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NASA Workshop on Impact Damage to Composites

Compiled by
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NASA Langley Research Center and held at
Langley Research Center
Hampton, Virginia
March 19–20, 1991

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National Aeronautics and
Space Administration

Langley Research Center
Hampton, Virginia 23665-5225

PREFACE

The *NASA Workshop on Impact Damage to Composites* was sponsored by the Mechanics of Materials Branch of NASA Langley Research Center. The workshop was held on March 19 and 20, 1991 at Langley Research Center, Hampton, Virginia. The objective was to review technology for evaluating damage tolerance of composite structures with impact damage and identify deficiencies. The number of registered participants was 40, representing industry, government, and universities; a list is included. The participants were specialists in the field of impact damage and composites. The workshop was divided into the following five sessions:

I - Impact Mechanics and Scaling
II - Damage and Strength Predictions
III - Standard Tests
IV - Design Criteria and Certification
V - Identification of Technology Deficiencies

The review, which was conducted in *Sessions I-IV*, consisted of invited talks that covered research and development, design, and criteria. Technology deficiencies were identified and discussed in Session V. Mr. C. C. Poe, Jr. moderated the sessions. This conference publication is a compilation of the slides used by the speakers in *Sessions I-IV* and a *List of Actions to Address Technology Deficiencies* that were recommended by the participants in *Session V*.

CONTENTS

LIST OF ACTIONS TO ADDRESS TECHNOLOGY DEFICIENCIES

General

- Standardize philosophy of distinguishing between damage resistance and damage tolerance.
- Standardize coupon type impact test.
- Develop methodology to relate results from standard impact tests of coupons to structures.
- Develop analyses to reduce the amount of testing currently used in the building block approach to verify a design.
- Develop progressive damage analyses.
- Address unique requirements of fuselage type structure.

Damage Resistance

- Develop a measure of extent and degree of impact damage, particularly one that relates to a threshold for detection.
- Develop understanding of strain-rate effects.
- Develop failure criteria to predict damage.
- Determine preload effects.
- Develop method to predict delamination growth in fatigue.
- Determine residual stress effects.
- Develop relationship between ply cracking, interlaminar toughness, and damage resistance.
- Develop local contact stiffness relationship that accounts for damage and other nonlinearities.

Damage Tolerance

- Develop failure criteria.
- Determine effects of biaxial loading and shear loading.

Criteria

- Survey airlines to determine frequency of impacts that can damage composite structures.
- Determine effects of applying limit load cycles to fatigue test article.
- Perform probability analysis to relate factor of safety to level of impact energy.
- Determine if criteria should result in strengths that are very sensitive to changes in level of energy or damage.
- Determine effects of oblique impacts.
- Determine how to account for impacter shape.

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Composition and Analysis of the Impact Force Curve

presented by

Wade C. Jackson

Mechanics of Materials Branch
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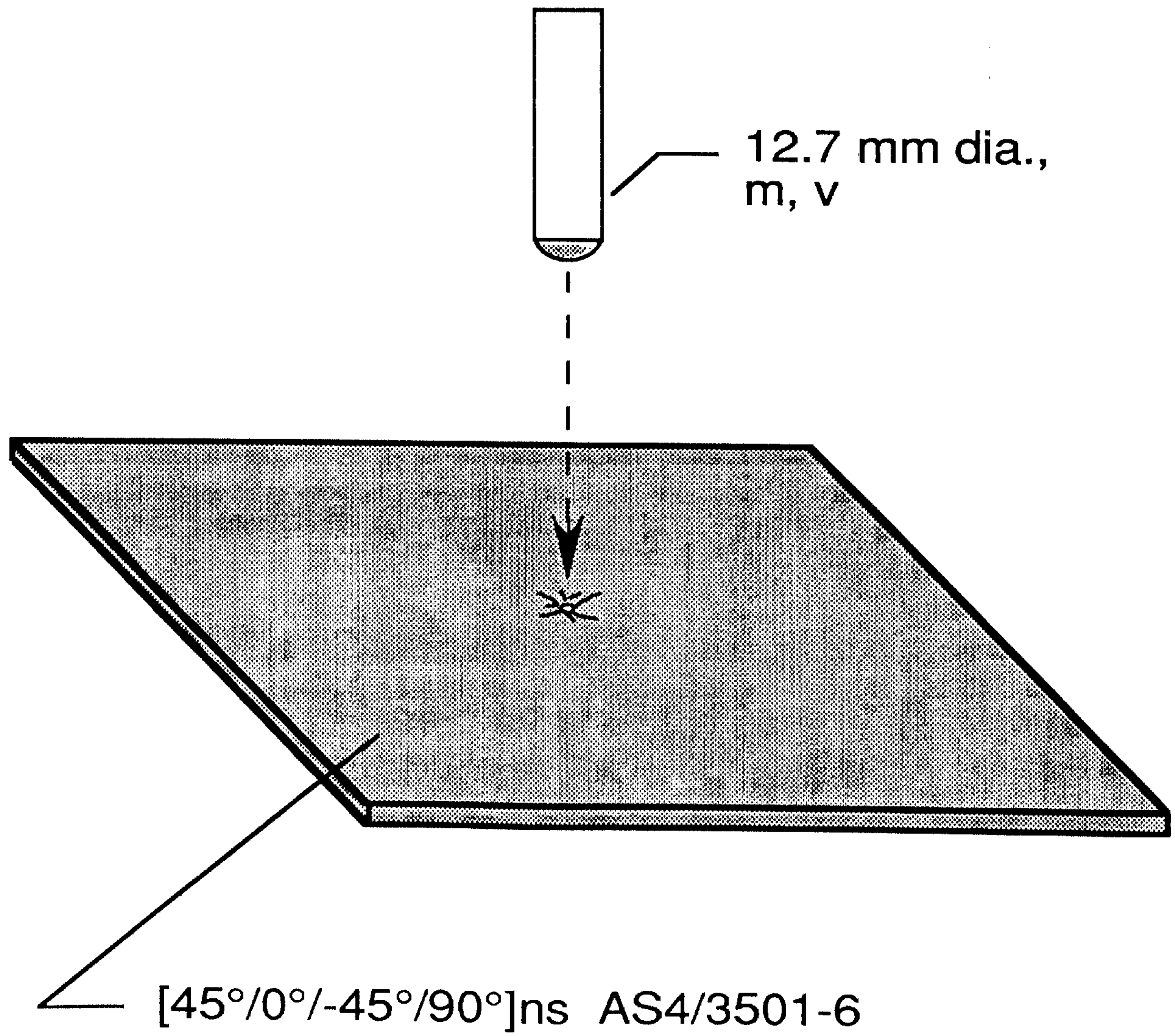
Impact Analysis Model



Analysis Methods

Computer Programs
- B.V. Sankar (University of Florida)
- C.T. Sun (Purdue University)

Energy-Balance Method (Greszczuk)

Zener/Olsson Solution

Mittal's Solution

Computer Programs

Sankar
- Dynamic Green's Function
- Series solution
- Limited to rectangular plates with simply-supported boundary conditions

Sun
- Finite element method
- Limited to rectangular plates but can accommodate any boundary conditions

Energy-Balance Method

$$K.E._{Impacter} = E_{Plate} + E_{Indentation}$$

$$\frac{1}{2}mv^2 = \int_0^{\delta_{max}} F\,d\delta + \int_0^{\alpha_{max}} F\,d\alpha$$

force-displacement relationship for plate

$$F = k\delta$$

Hertz's Law

$$F = n\alpha^{3/2}$$

Final Equation

$$\frac{1}{2}mv^2 = \frac{1}{2}\frac{F_{max}^2}{k} + \frac{2}{5}\frac{F_{max}^{5/3}}{n^{2/3}}$$

Zener/Olsson Model

One-parameter dimensionless analytical model.

Contact duration was less than the time required for bending waves to reflect from the boundaries.

Olsson extended Zener's solution to include anisotropic materials.

$$\frac{d^2\bar{\alpha}}{d\bar{t}^2} + \frac{3}{2}\bar{\lambda}\,\bar{\alpha}^{1/2}\frac{d\bar{\alpha}}{d\bar{t}} + \bar{\alpha}^{3/2} = 0$$

$$\bar{\alpha}(0) = 0 \qquad \frac{d\bar{\alpha}(0)}{d\bar{t}} = 1$$

$\bar{\alpha}$ = indentation

$\bar{\lambda}$ = dimensionless impact parameter

Mittal's Equation

Solved same problem as Zener but included the effects of shear deformation.

Final non-linear integral equation

$$\bar{F}^{2/3}(\bar{t}) - \bar{t} + \int_0^{\bar{t}} \bar{F}(\bar{t}')\left[(\bar{t} - \bar{t}') + \frac{4\lambda}{\pi}\left\{\frac{1}{2}\tan^{-1}\left(\frac{\bar{t} - \bar{t}'}{\beta'}\right) + \frac{\beta'}{\bar{t} - \bar{t}'}\right\}\right] d\bar{t}' = 0$$

Singularity

$\bar{F}$ = dimensionless force

λ = dimensionless impact parameter

β' = dimensionless shear parameter

Evaluated using a recursive relationship with a time step of 0.01

Results

Sample Force Histories

Composition of Impact Force Curve

Impact Animations

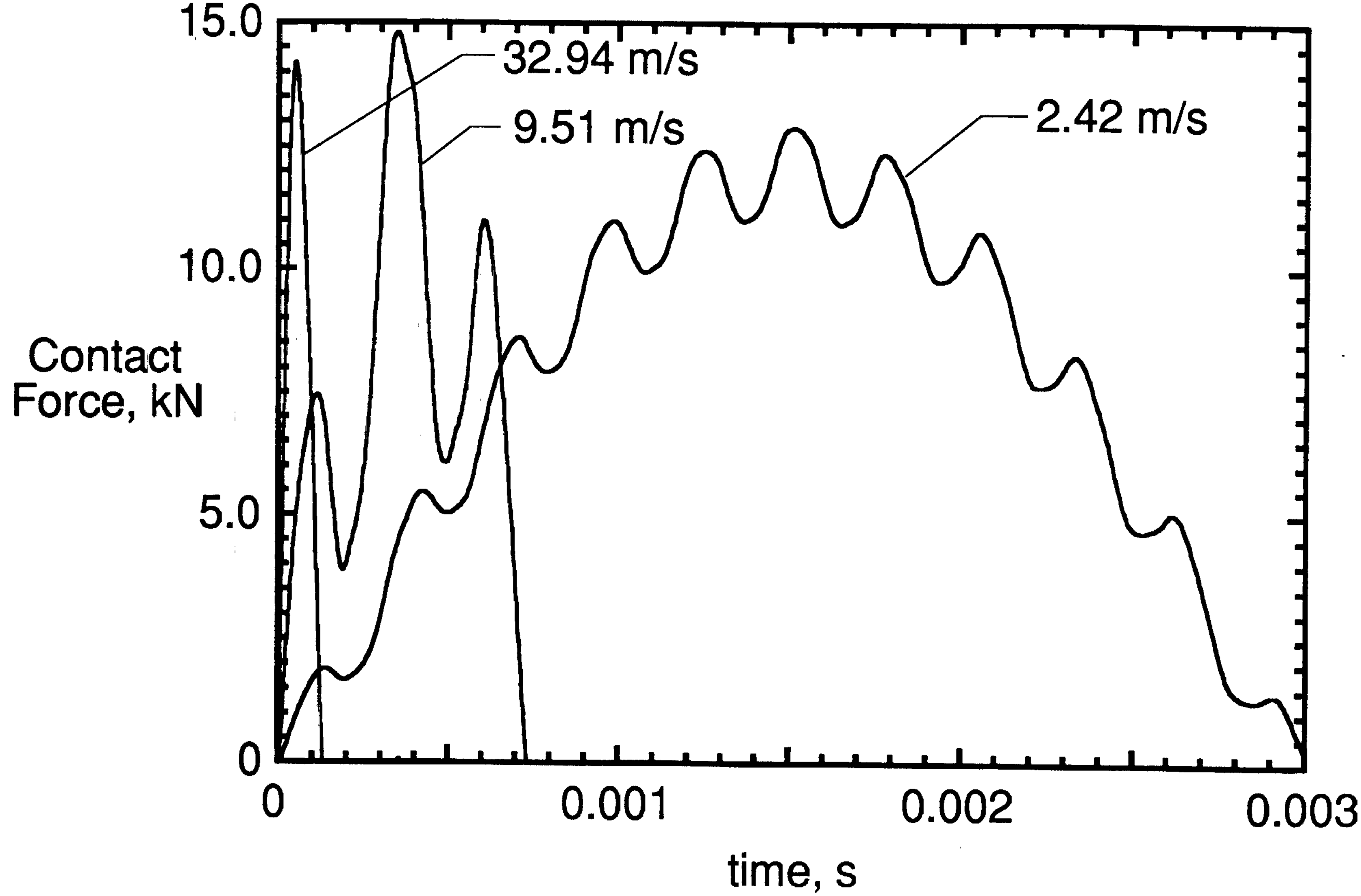
Correlation between Analyses

Displacements

Effect of Kinetic Energy, Plate Thickness, Shear Deformation, Plate Size, and Boundary Conditions

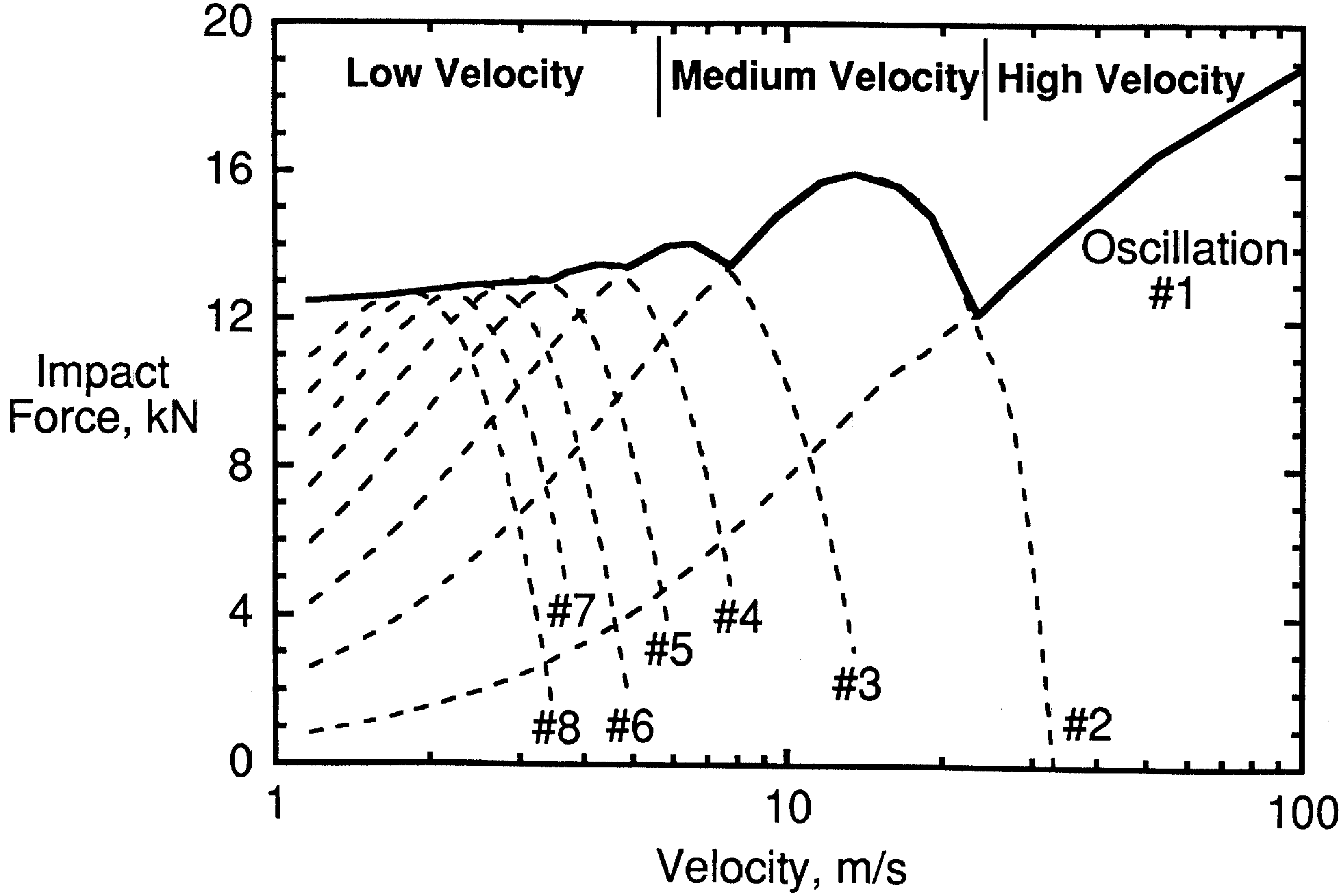
Sample Force Histories

48 ply, 12.7 x 12.7 cm ss-ss, K.E. = 13.6 J



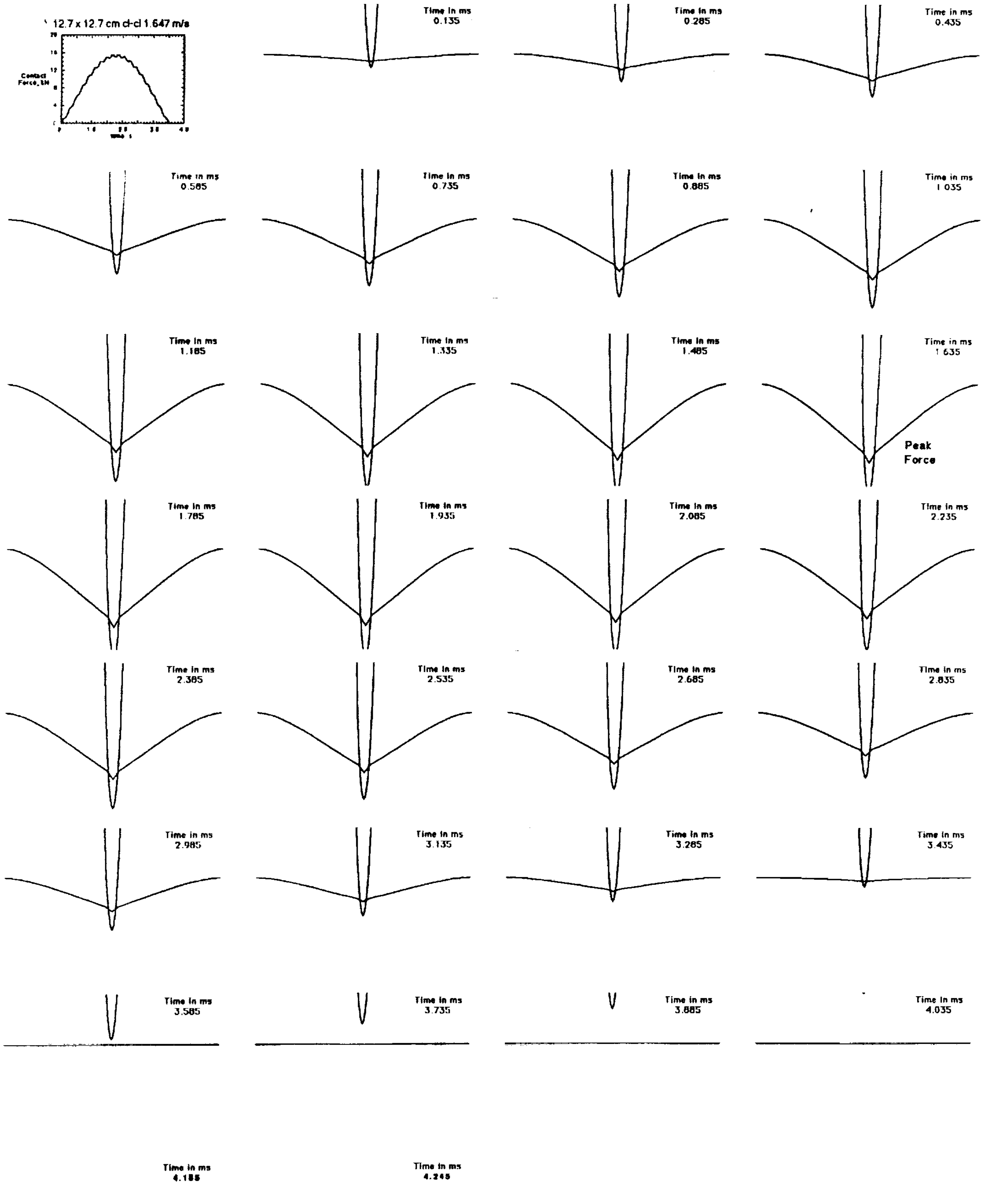
Composition of Impact Force Curve

48 ply, 12.7 x 12.7 cm ss-ss, K.E. = 13.6 J

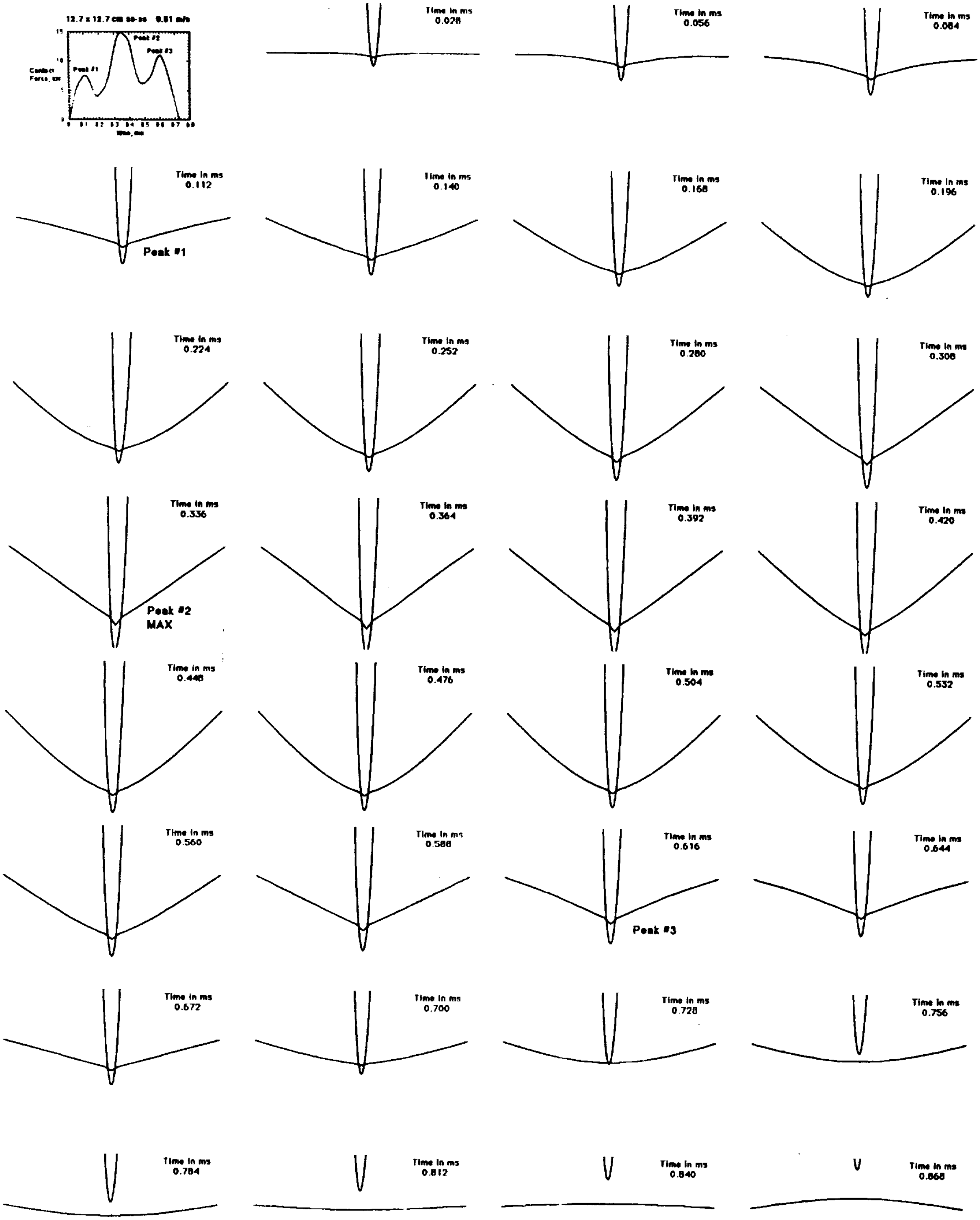


Impact Animations

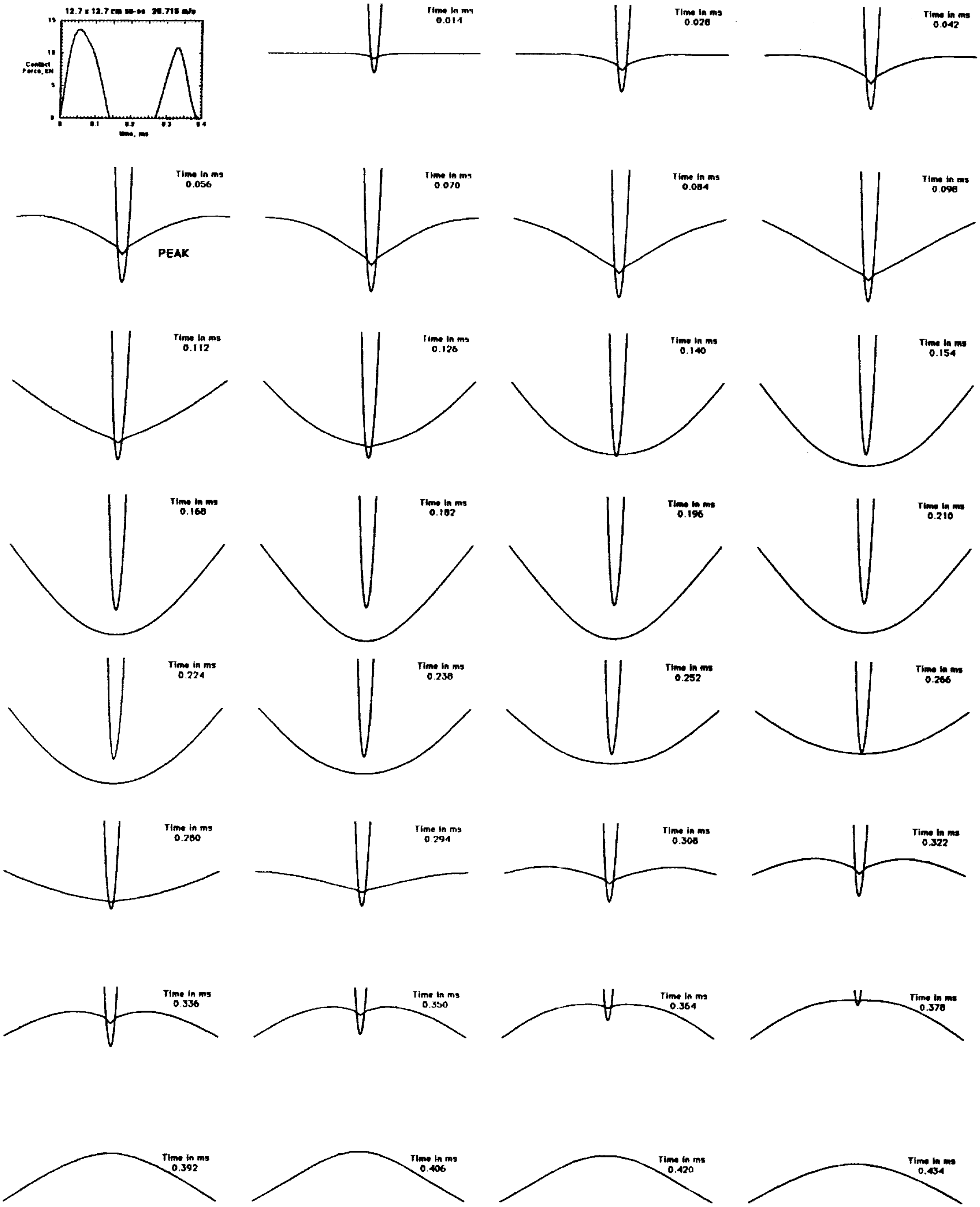
12.7 x 12.7 cm Cl-Cl, 13.56 J, v=1.647 m/s, 48-ply Quasi



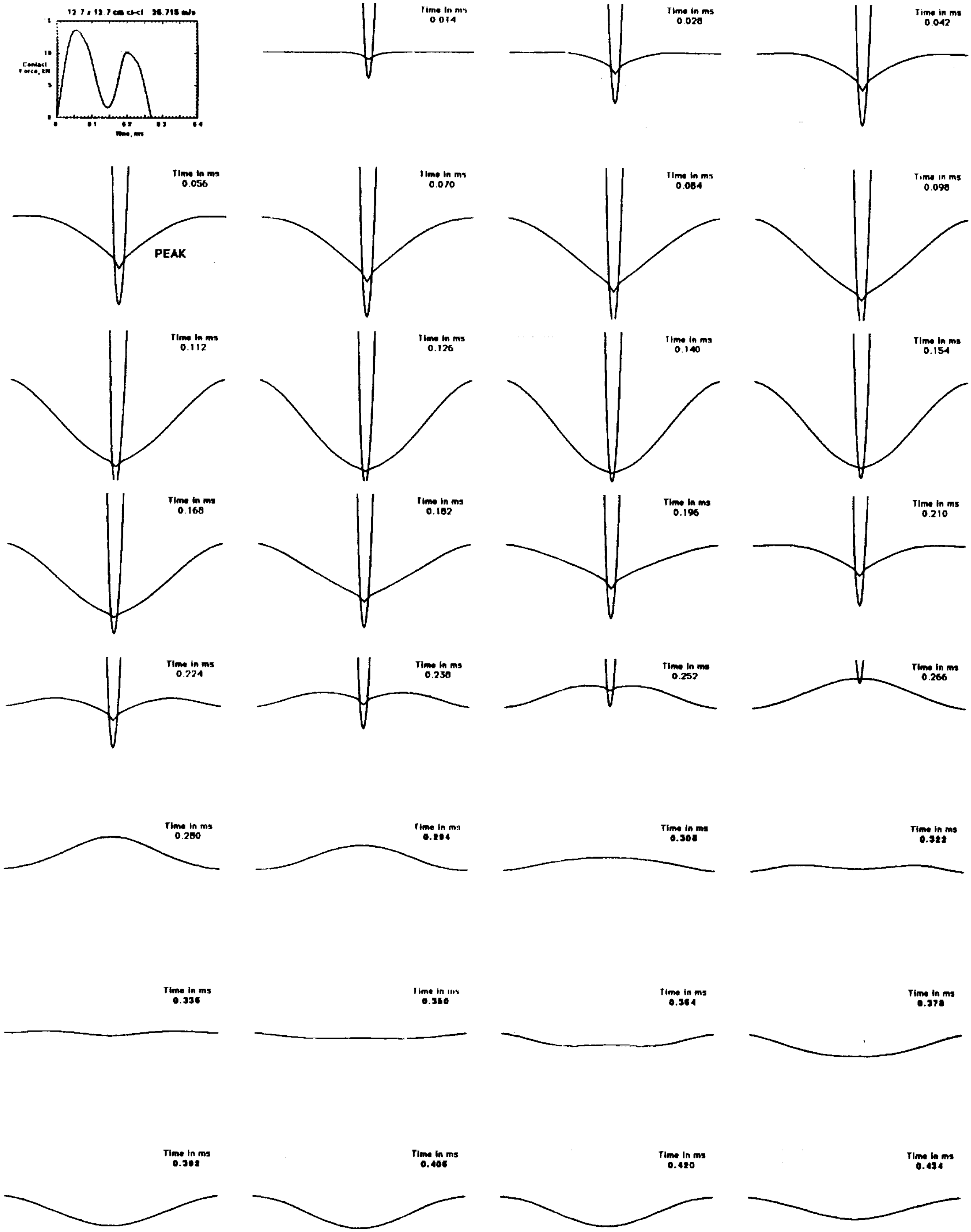
12.7 x12.7 cm SS-SS, 13.56 J, v=9.51 m/s, 48-ply Quasi



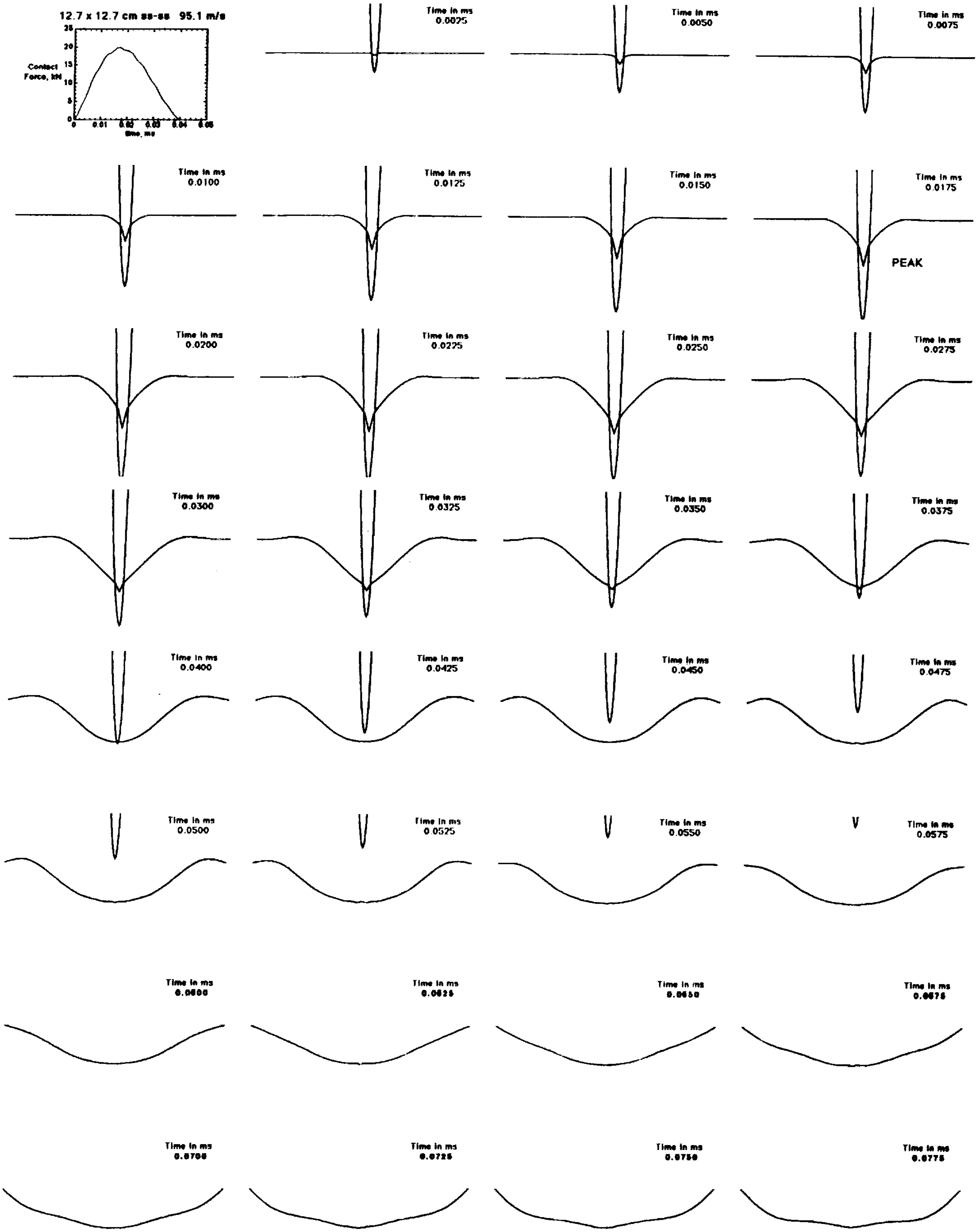
12.7 x 12.7 cm SS-SS, 13.56 J, v=26.715 m/s, 48-ply Quasi



12.7 x12.7 cm Cl-Cl, 13.56 J, v= 26.715, 48-ply Quasi

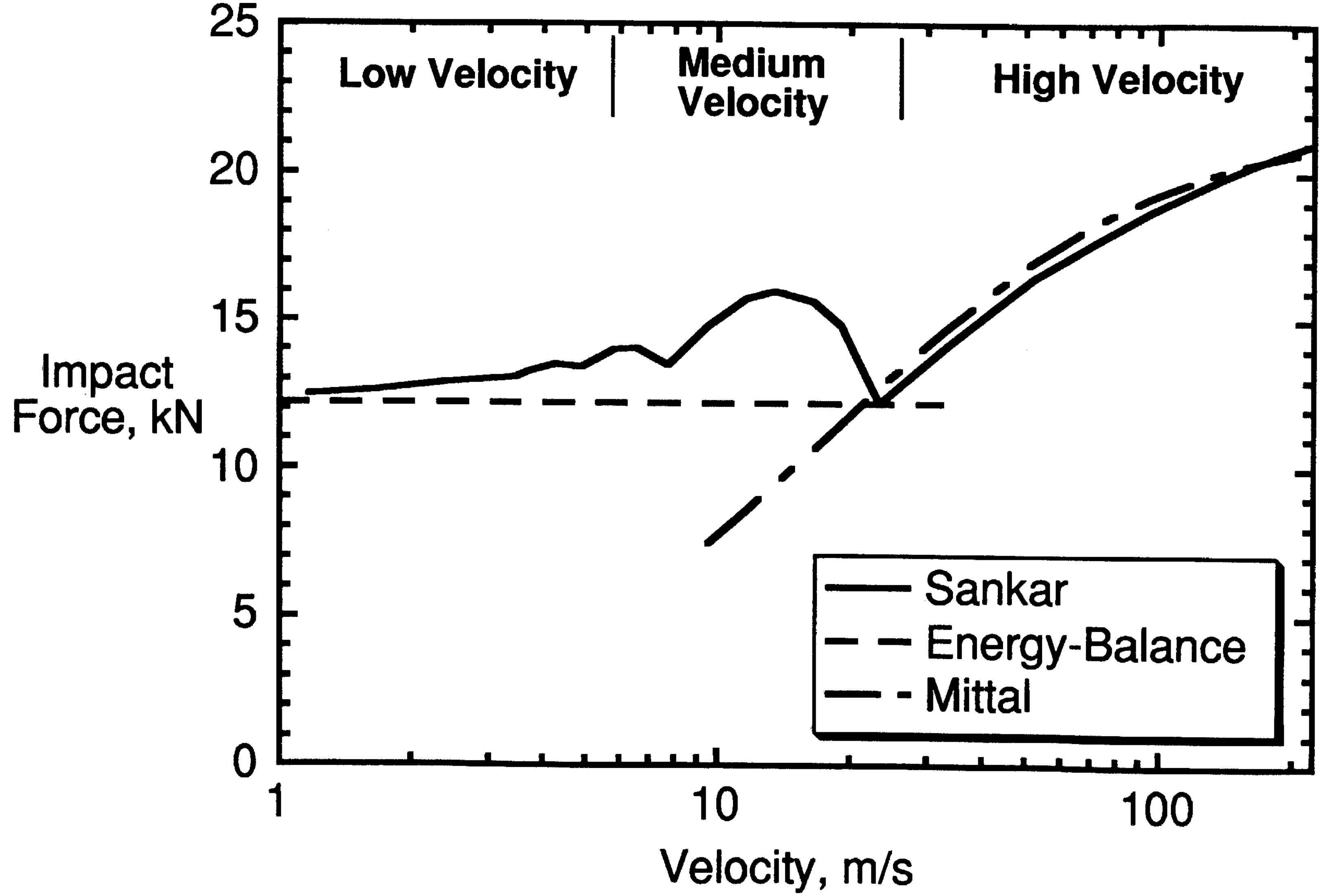


12.7 x 12.7 cm SS-SS, 13.56 J, v=95.1 m/s, 48-ply Quasi



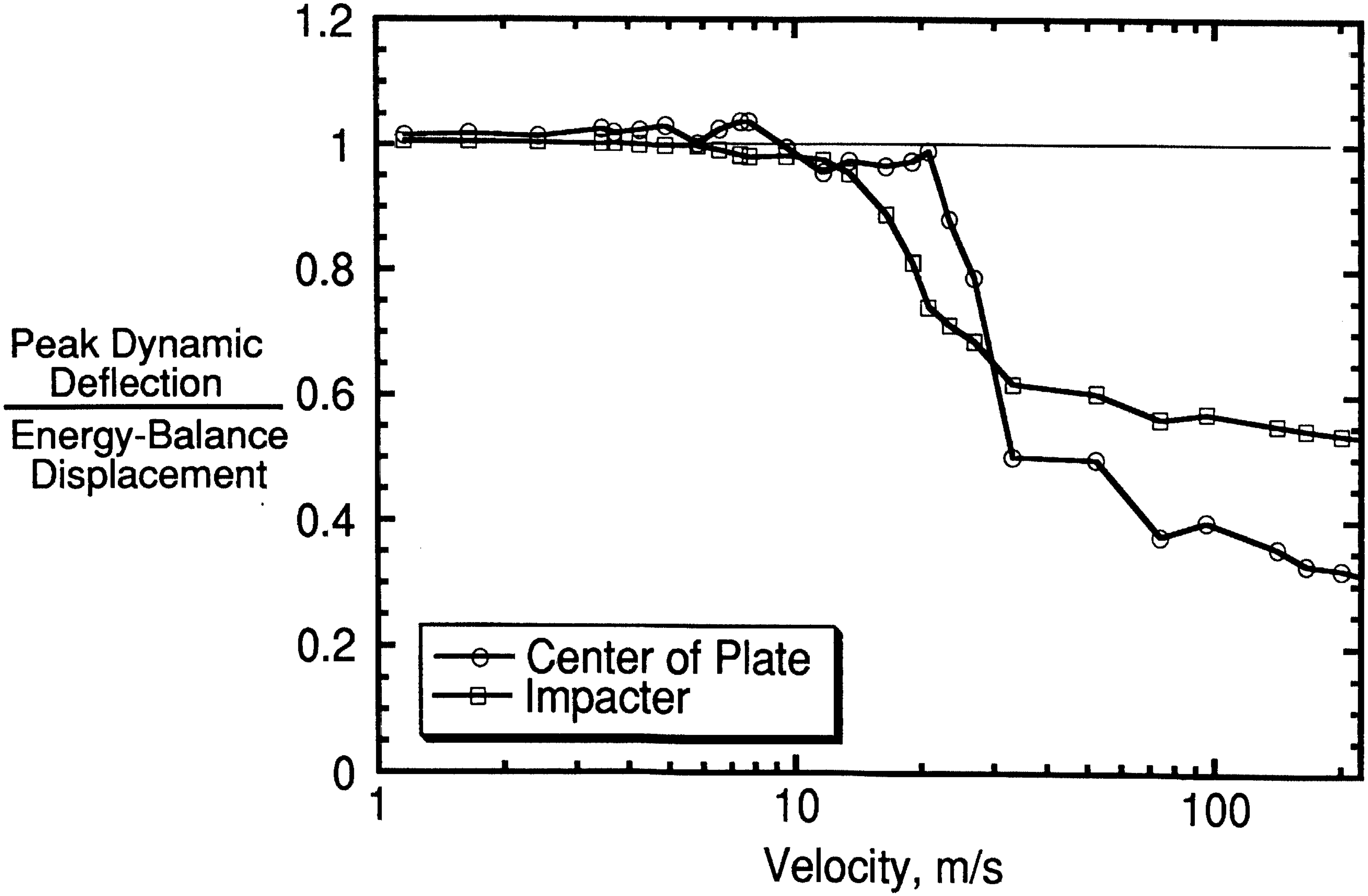
Correlation between Analyses

48 ply, 12.7 x 12.7 cm ss-ss, K.E. = 13.6 J



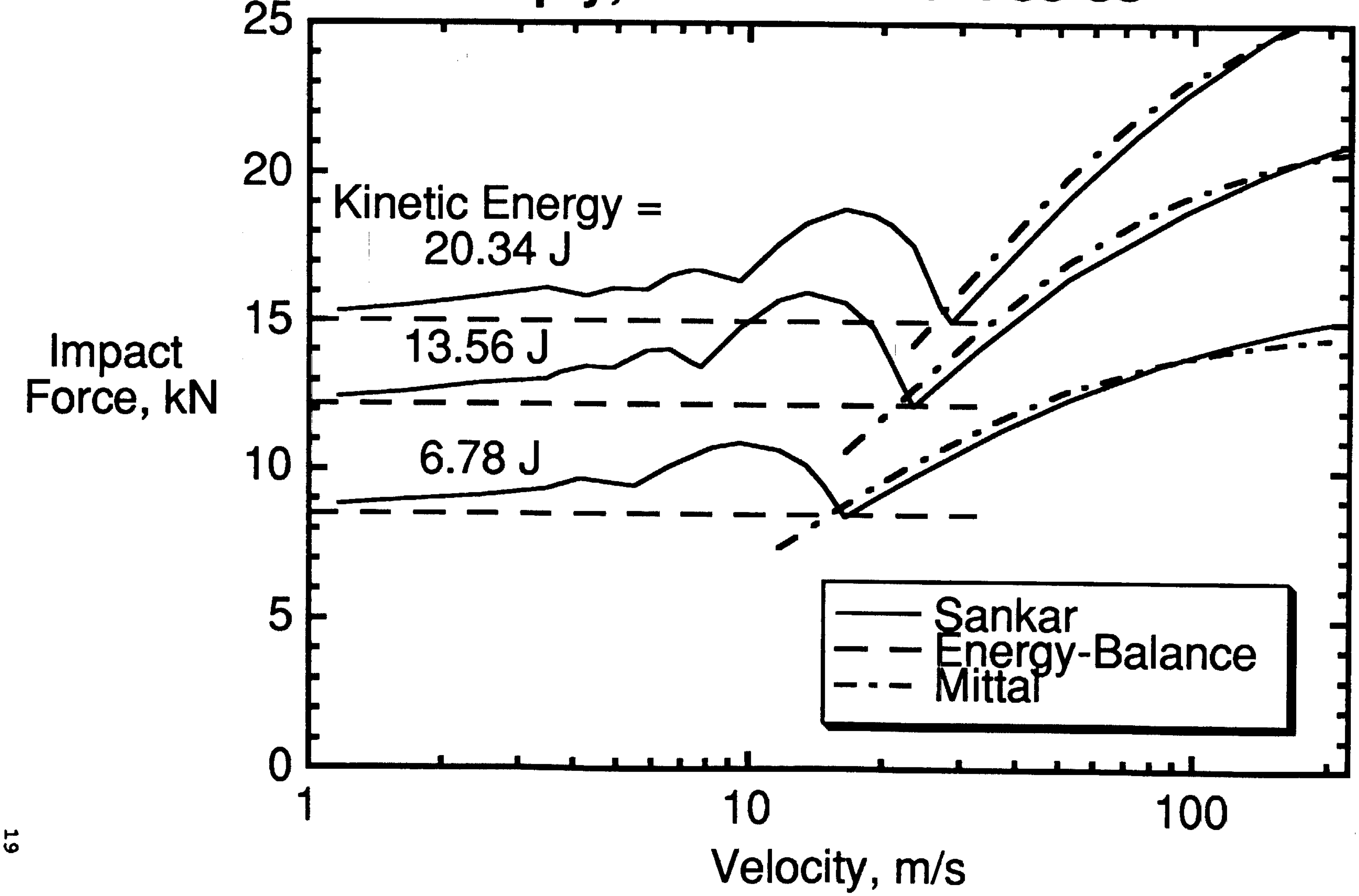
Normalized Dynamic Deflections

48 ply, 12.7 x 12.7 cm ss-ss, K.E. = 13.6 J



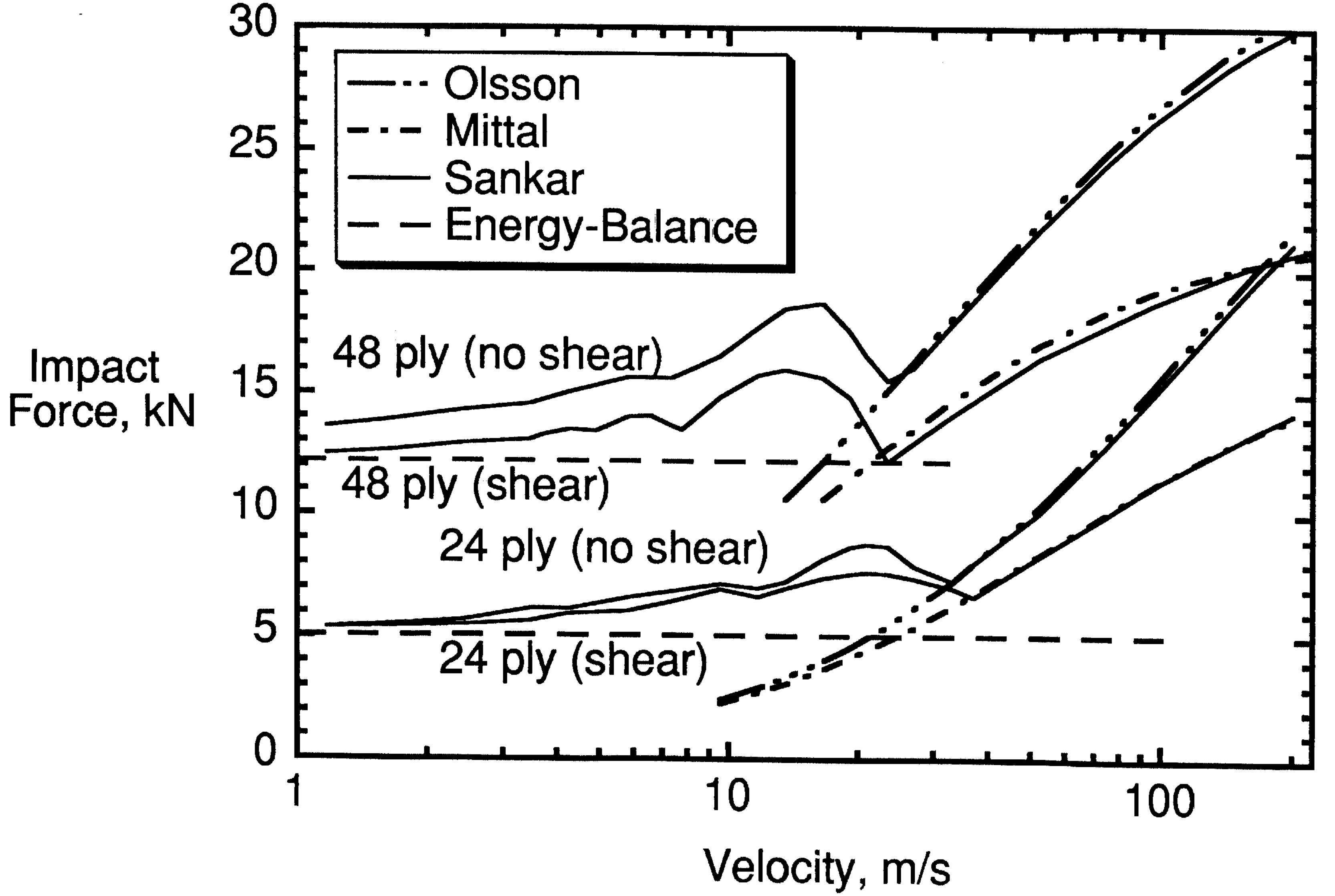
Effect of Kinetic Energy on Impact Force

48 ply, 12.7 x 12.7 cm ss-ss



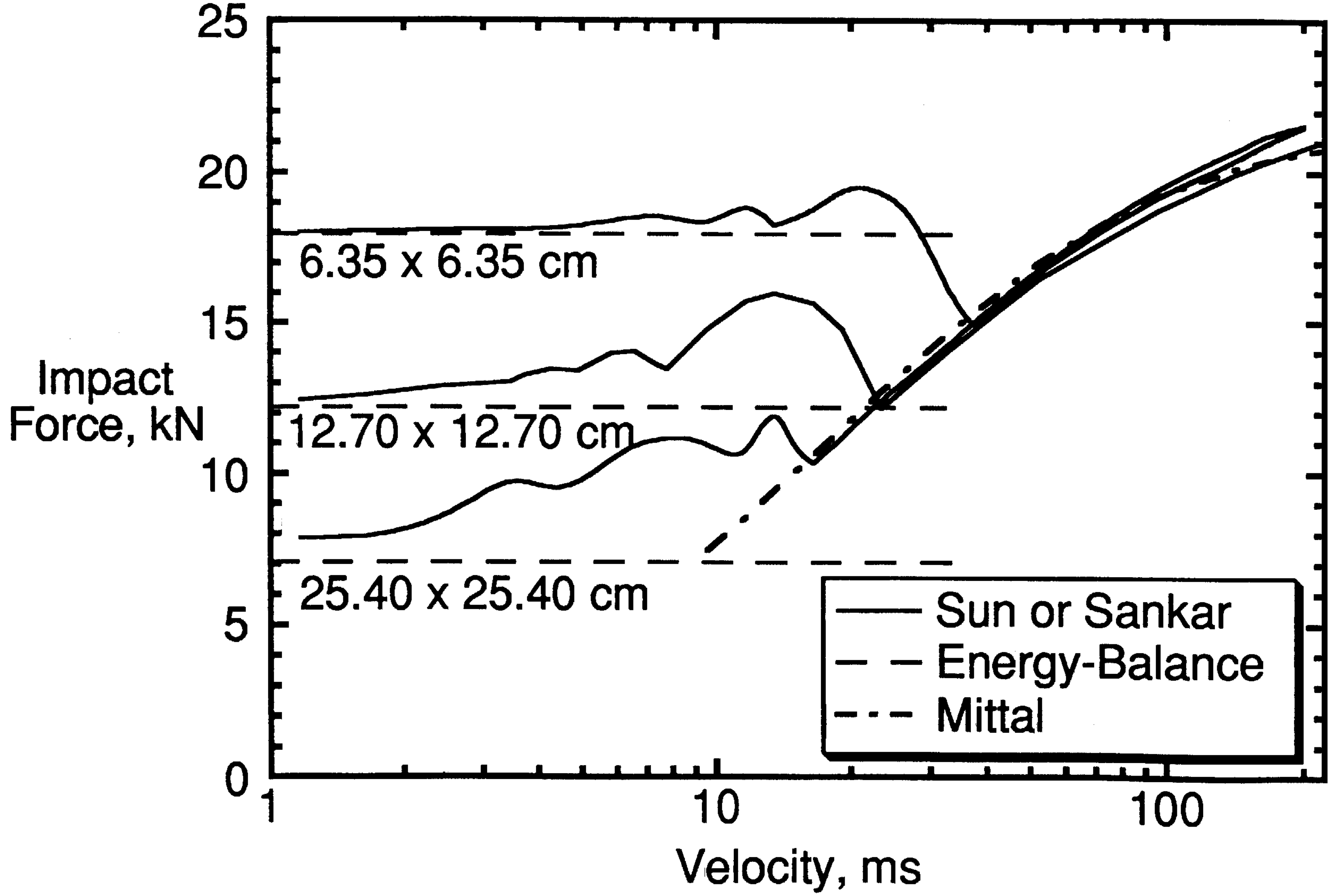
Effect of Shear on Impact Force

12.7 x 12.7 cm ss-ss, K.E. = 13.6 J



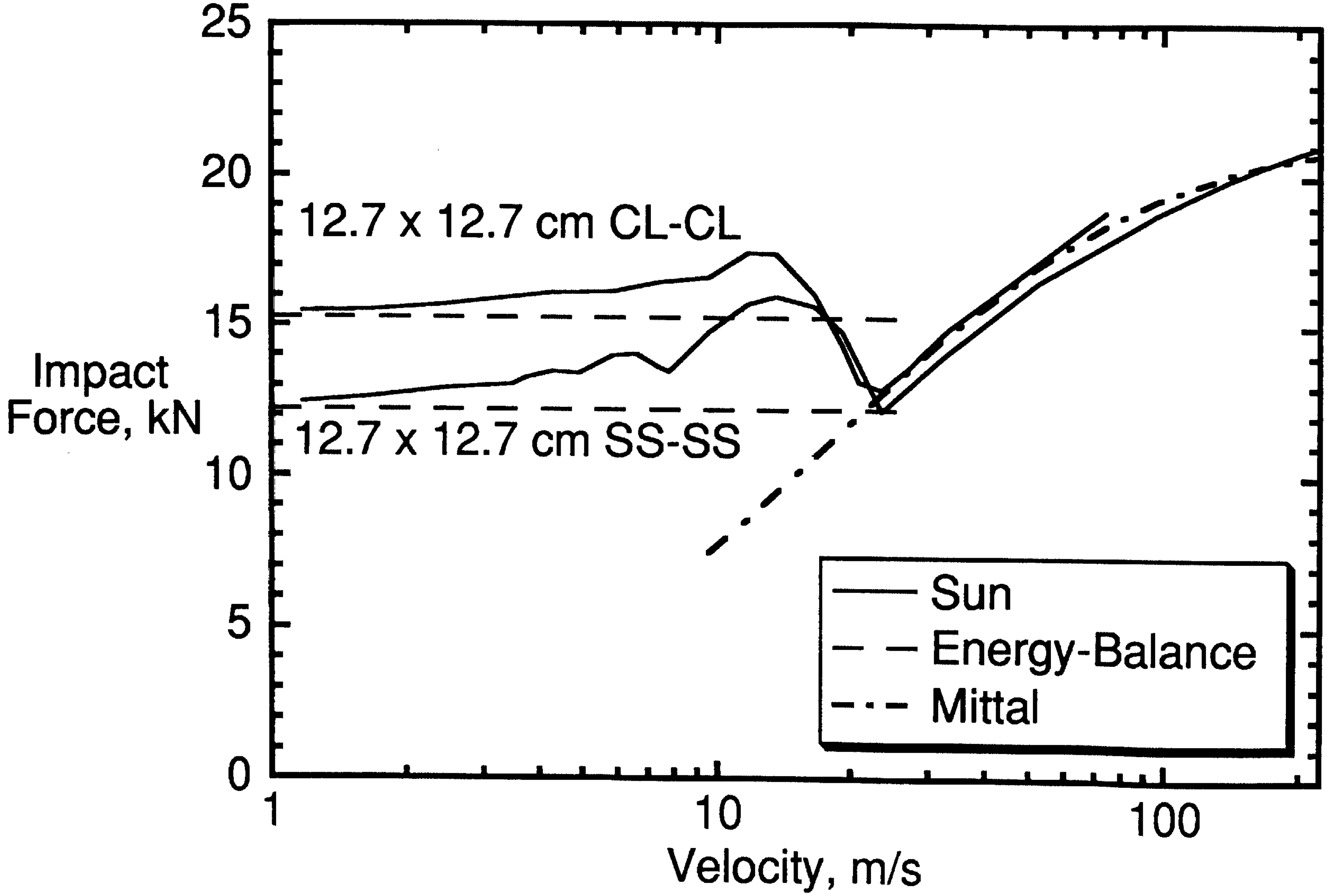
Effect of Plate Size on Impact Force

48 ply, ss-ss boundaries, K.E. = 13.6 J



Effect of Boundaries on Impact Force

48 ply, K.E. = 13.6 J



Concluding Remarks - Low Velocity

ENERGY-BALANCE:

Very simple to use and understand.

Accurately predicts impact force and displacements in the "low velocity" region.

Accuracy improves as contact duration increases and/or the plate vibration frequencies increase.

A static stress analysis should be adequate to model the impact problem.

Concluding Remarks - High Velocity

ZENER / OLSSON EQUATION:

Predicts force history very accurately and quickly in the "high velocity" region.

Limited to very thin laminates where shear deformation is not important.

MITTAL'S EQUATION:

Will accurately predict the force history for "high velocity" impacts for a given time step.

Solution is not convergent.

Limited to isotropic or quasi-isotropic plates.

Concluding Remarks - General

Impacts can be divided into three regions: low, medium, and high velocity.

Impact force curve can be constructed using an energy-balance and Mittal's equation.

In the "medium velocity" region, a program such as Sun's or Sankar's must be used to describe the impact.

IMPACTING LARGE COMPOSITE STRUCTURES AND SCALING IMPACT RESPONSE AND DAMAGE

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M. E. MORGAN

R. J. NUISMER

HERCULES AEROSPACE/COMPOSITE PRODUCTS

NASA WORKSHOP ON IMPACT DAMAGE TO COMPOSITES

19-20 MARCH 1991

IMPACTING LARGE/THICK STRUCTURES

THE USE OF FILAMENT WOUND CASES FOR THE SPACE SHUTTLE SOLID ROCKET BOOSTER REQUIRED AN UNDERSTANDING OF THE EFFECTS OF DAMAGE ON CASE PERFORMANCE

- EXTENSIVE HANDLING PRECAUTIONS COULD NOT TOTALLY ELIMINATE ACCIDENTAL IMPACT THREAT
- ACCEPT/REJECT CRITERIA BASED ON QUARTER-SCALES AND JOINT COUPON MATERIAL
- COUPON AND SUBSCALE RESULTS ARE DIFFICULT TO TRANSLATE TO FULL-SCALE SINCE METHODOLOGY IS NOT WELL UNDERSTOOD
- LACK OF REPRESENTATIVE FULL-SCALE IMPACT STRENGTH LOSS DATA WAS IMPETUS BEHIND SECOND PHASE OF DAMAGE CHARACTERIZATION WORK

HERCULES

- THE OBJECTIVE OF THE FWC IMPACT DAMAGE STUDY WAS TO VERIFY THE BURST INTEGRITY OF THE CASE AT VISUAL DAMAGE THRESHOLD (VDT)

- THE SECONDARY OBJECTIVE WAS TO AID IN ESTABLISHING IMPACT DAMAGE DISPOSITIONING TECHNIQUES

HERCULES

19-34/3

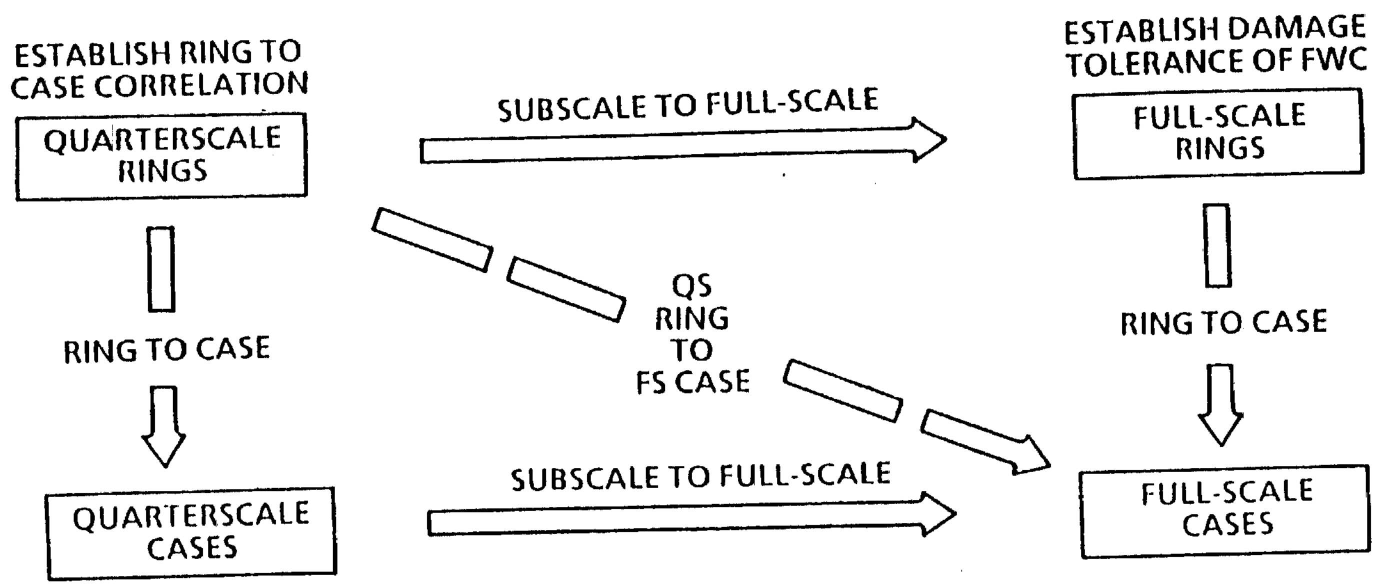
SECOND PHASE OF IMPACT CHARACTERIZATION ADDRESSES LACK OF FULL-SCALE DATA AND PREVIOUS CONCERNS ABOUT STRENGTH LOSS BELOW VISUAL THRESHOLD

- ESTABLISH VISUAL DAMAGE THRESHOLD (VDT) ON FULL-SCALE CASES (BOTH PAINTED AND NONPAINTED SURFACES)
- VERIFY WHETHER DAMAGE BELOW THE VDT DEGRADES CASE STRENGTH
- CHARACTERIZE EXTENT AND TYPE OF DAMAGE (FROM A VARIETY OF IMPACT CONDITIONS)
- USING EXISTING ANALYTICAL TECHNIQUES AND DEPLY DATA
  - INTRODUCE KNOWN AMOUNTS OF DAMAGE INTO FULL-SCALE TEST RINGS
  - COMPARE RING PERFORMANCE TO PREDICTIONS
- ASSESS DAMAGE TOLERANCE OF FWC RINGS/CASES FROM RING TEST DATA

HERCULES

19-43-3

FWC IMPACT DAMAGE INVESTIGATION LOGIC



HERCULES

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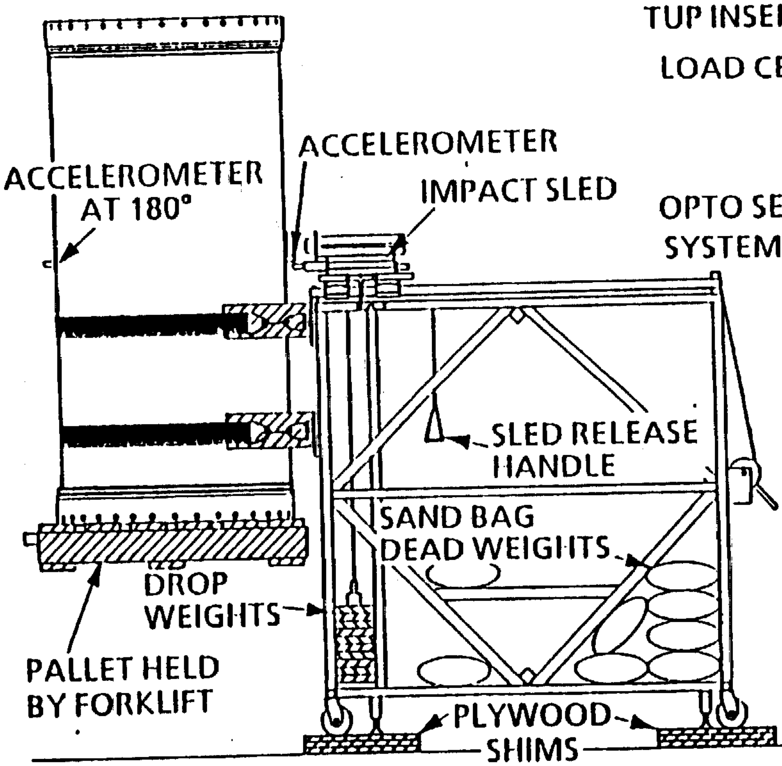
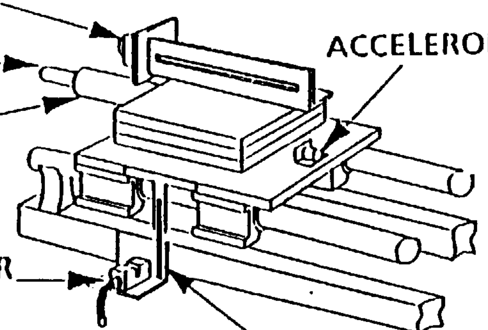
TEST CONFIGURATION USED FOR FWC PROVIDED REPRODUCIBLE INSTRUMENTED IMPACTS





IMPACT SLED CLOSE-UP

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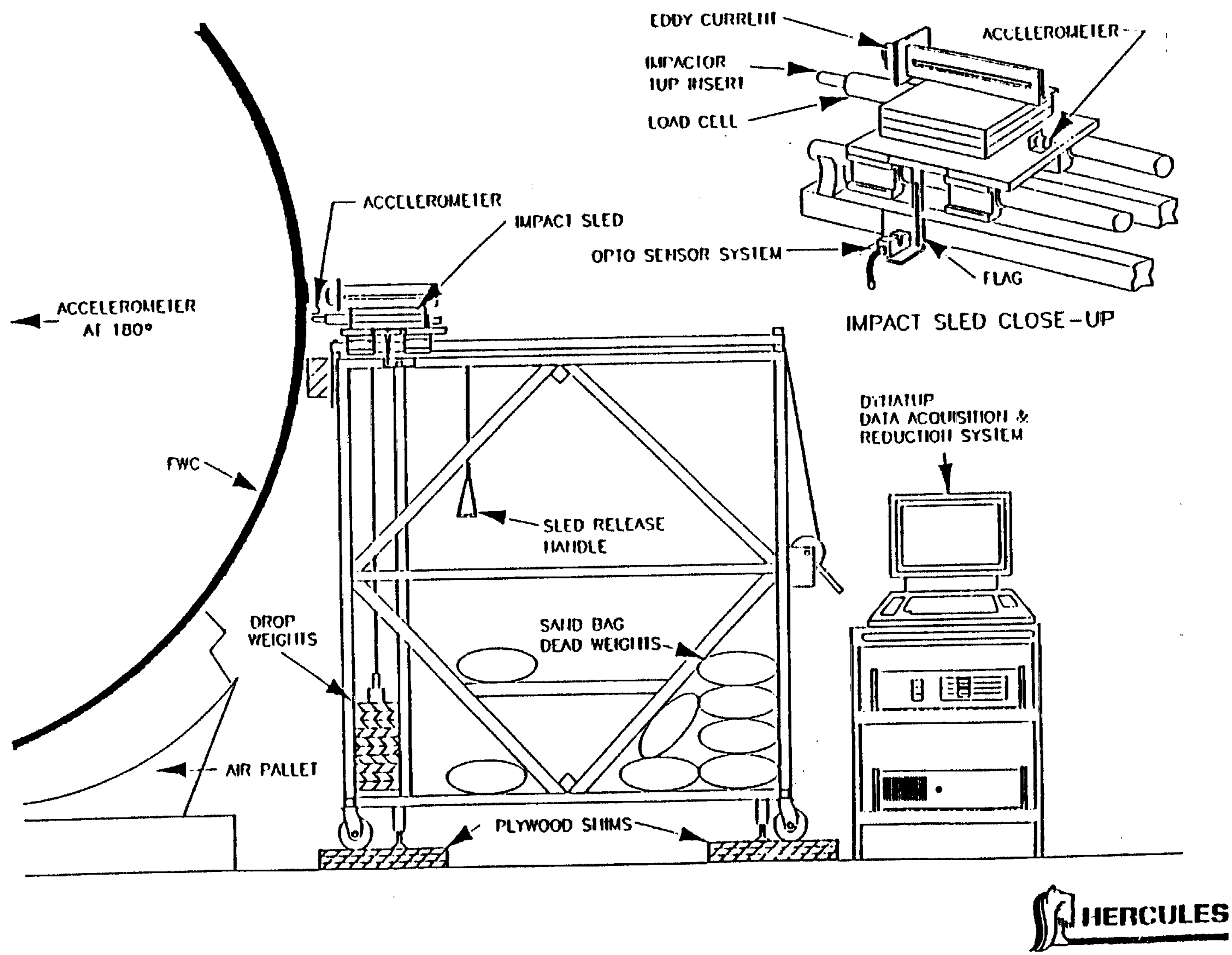
FWC IMPACT FIXTURE AND INSTRUMENTATION



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A MAJOR CONCERN IS UNDETECTED (BELOW VDT) DAMAGE THAT COULD REDUCE PERFORMANCE

- WHAT CONSTITUTED "VISIBLE" BECAME A MAJOR DILEMMA
- LOW LEVEL IMPACTS WERE IDENTIFIABLE IF IMPACT EVENT WAS OBSERVED AND LOCATION KNOWN
- ARMALON OVERWRAP TEXTURED IMPRINT PROVED TO BE HIGHLY SENSITIVE TO IMPACTS
- HUMAN FACTORS SUCH AS EYESIGHT, ABILITY, INSPECTION TECHNIQUES, AND LIGHTING COULD INFLUENCE VISUAL THRESHOLD

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19-43/9

MEASURABLE QUANTIFIED DEFINITION OF VISIBLE DAMAGE THRESHOLD DEVELOPED WITH HELP OF QC INSPECTION GROUPS

- FIELD INSPECTORS WOULD NOT KNOW WHERE TO LOOK
- THREE SEPARATE QC TEAMS FOUND 5-MIL DEPRESSIONS 100% OF THE TIME
    - HERCULES
    - MORTON THIOKOL
- DISTURBANCES LESS THAN 5 MILS WOULD BE HARD TO DETECT
- 5-MIL DEPRESSION IS GREATER THAN ARMALON IMPRINT TEXTURE

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QUANTITATIVE VISUAL DAMAGE THRESHOLD DEFINED FOR FWC

- VDT IS A VISUALLY DETECTED ANOMALOUS CONDITION ON THE FILAMENT WOUND SURFACE AS COMPARED TO THE ADJACENT COMPOSITE SURFACE. THIS CONDITION INCLUDES IMPRESSIONS, INDENTATIONS, GOUGES, SURFACE FINISH DISTURBANCES, OR FIBER DAMAGE OF DEPTH GREATER THAN 0.005 IN.
- VDT WAS "QUANTIFIED" TO PROVIDE INSPECTORS WITH A MEANS FOR SCREENING WHAT IS FOUND ON A CASE
- VDT LEVELS FOR FWC
  - 1-IN. DIAMETER IMPACTOR - 90 FT-LB
  - 0.5-IN. DIAMETER IMPACTOR - 20 FT-LB
  - THREE-SIDED CORNER IMPACTOR - 10 FT-LB

HERCULES

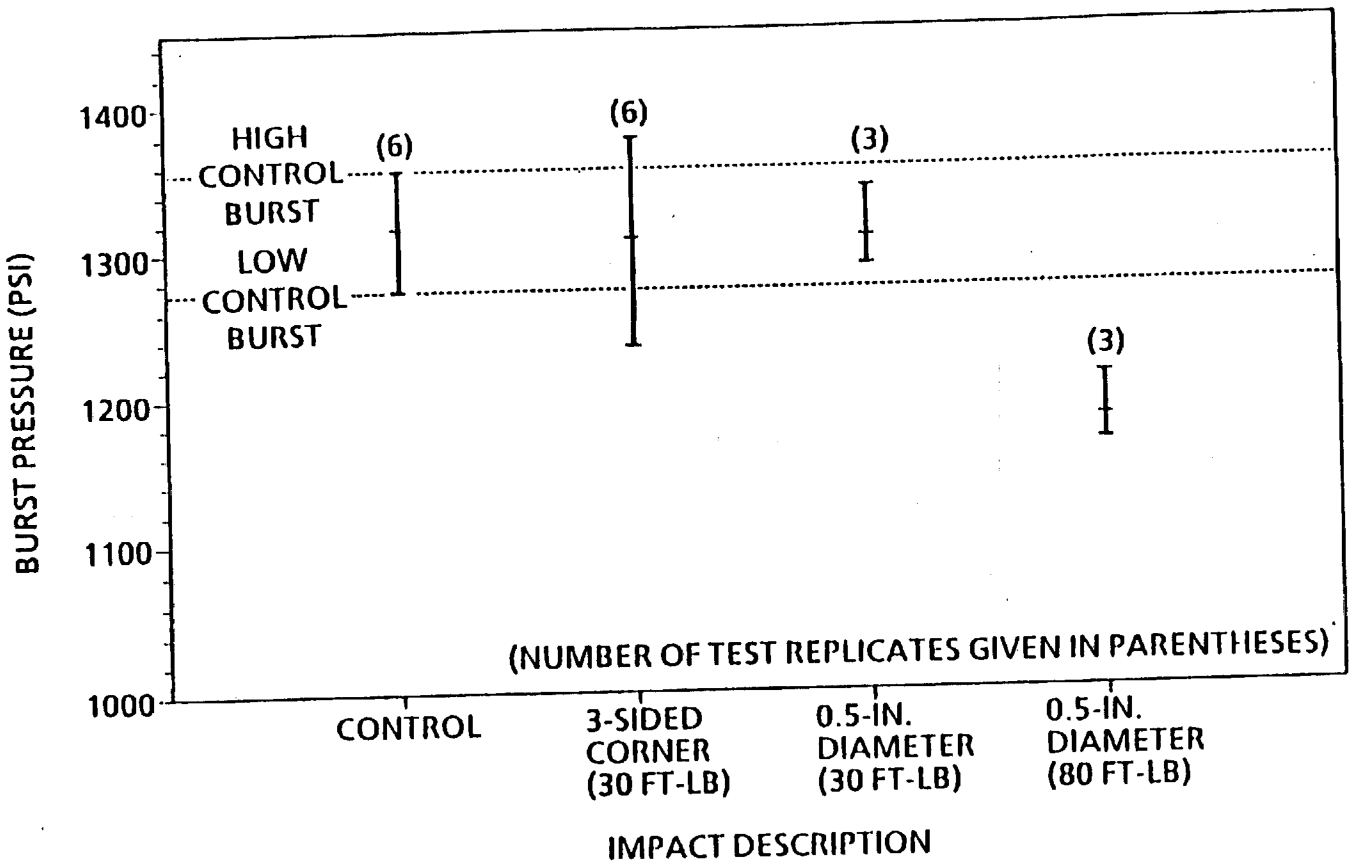
19-34/11

DAMAGE FROM IMPACT IN QS WAS DIFFERENT THAN IN FS

QS	FS
DELAMINATIONS DETECTED BY NDE AT ALL BUT LOWEST IMPACT LEVELS	NO DELAMINATIONS DETECTED--NDE TECHNIQUES COULD NOT LOCATE/ QUANTIFY DAMAGE (CROSS SECTION MICROSCOPY CONFIRMED NO DELAMINATIONS)
FIBER DAMAGE EXTENDED WELL BEYOND IMPACT SITE ON ALL SIDES	FIBER DAMAGE LOCALIZED UNDER IMPACT SITE
FIBER DAMAGE FOUND IN ALL 11 LAYERS AT SEVERE IMPACT CONDITIONS	FIBER DAMAGE FOUND 6 (OF 31) LAYERS DEEP AT SEVERE IMPACT CONDITIONS

HERCULES

ONLY HIGH LEVEL DAMAGE CAUSED STRENGTH LOSS (10%) ON QS RINGS



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HERCULES

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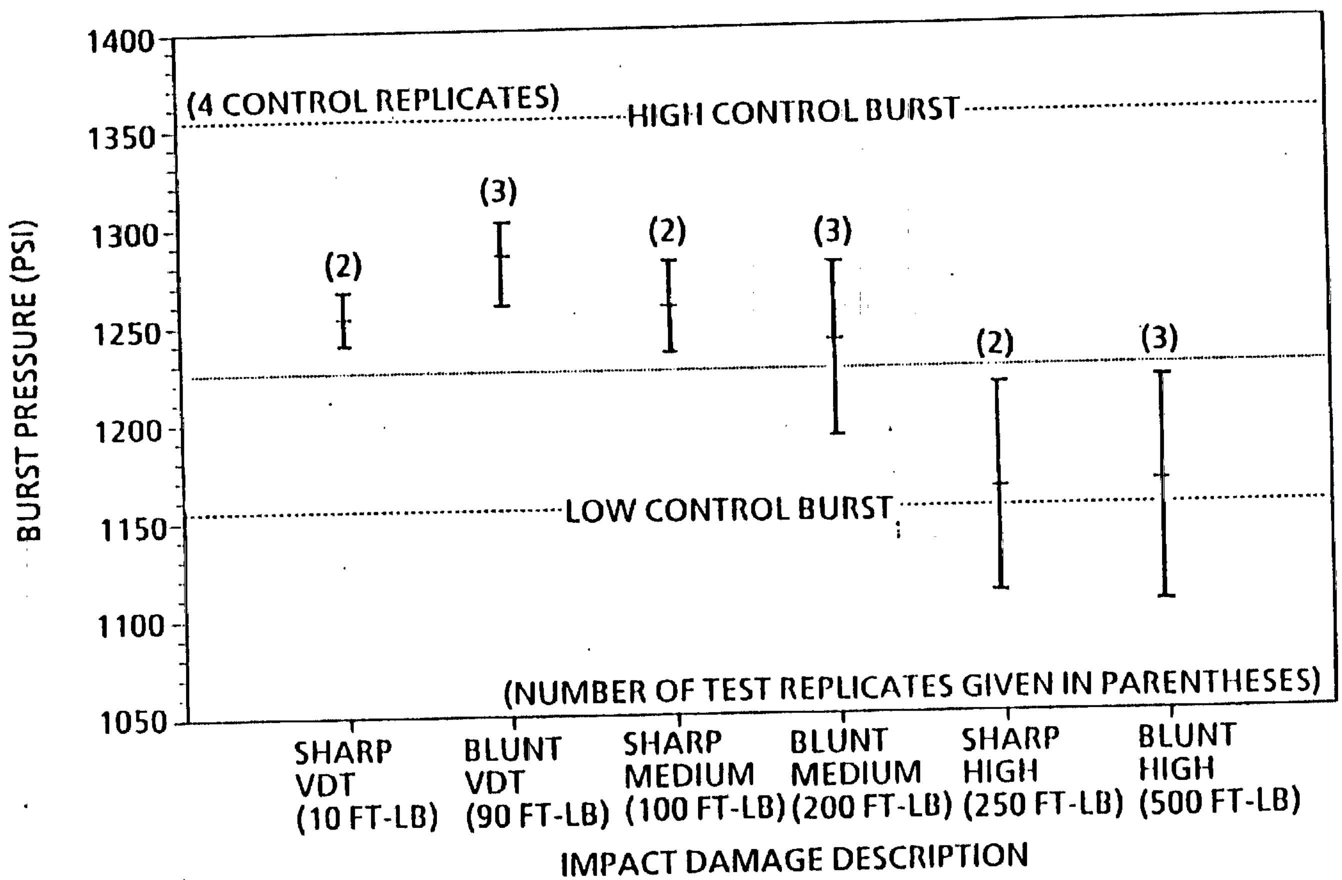
THE FWC IS TOLERANT TO HIGH IMPACT LEVELS BASED ON FS RING TESTING



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HERCULES

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SCALING IMPACT RESPONSE AND DAMAGE

DAMAGE ASSESSMENT FOR
COMPOSITE CASES
F04611-86-C-0040

HERCULES

RESULTS OF LITERATURE SEARCH/INDUSTRY SURVEY WERE PRESENTED AT WORKSHOP

- IN CONJUNCTION WITH JANNAF COMPOSITE MOTOR CASE SUBCOMMITTEE MEETING, NASA LANGLEY, 23-27 FEB 1987
- EXPANDED TO 2 DAYS AT AL REQUEST
- ATTENDANCE

INDUSTRY	GOVERNMENT	UNIVERSITY
• HERCULES	• AFML	• VPI
• THIOKOL	• AFWL	• DREXEL
• CSD	• AFFDL	• DELAWARE
• ASPC	• AFOSR	• MIT
• ARC	• JPL	• ALABAMA
• MDAC	• NRL	• DAYTON
• DUPONT	• NAVAL ORDNANCE	
• FIBERITE	• MICOM	
• UNION CARBIDE	• NASA	
• GENERAL RESEARCH	• ARMY MATERIALS LAB	
• MISSION RESEARCH		
• AVCO		

- PANEL DISCUSSION

HERCULES

1-261B/7

PROBLEM: IMPACT DAMAGE OF COMPOSITE CYLINDERS IS NOT WELL UNDERSTOOD

- Large Number of Parameters Makes Empirical Approach Difficult
- Empirical Approach too Costly for Large Cylinders

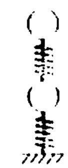
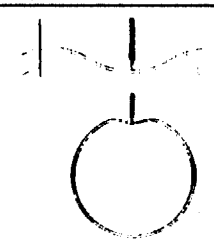
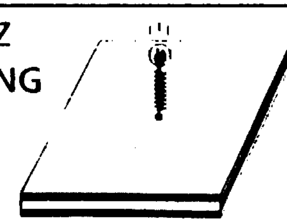
HERCULES

OBJECTIVE: GAIN SUFFICIENT UNDERSTANDING TO DESIGN COST EFFECTIVE CYLINDER IMPACT TESTS

- Understanding of Parameter Effects
- Understanding of Scaling Effects

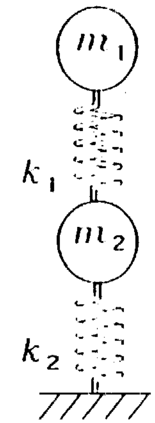
HERCULES

APPROACH: DO SUFFICIENT ANALYSIS UP-FRONT TO DESIGN INTELLIGENT TEST MATRIX

MODEL	FORMULATION	USE
2 DOF SPRING-MASS (R. J. NUISMER)	ASYMPTOTIC EXPANSION OF EXACT SOLUTION 	DEVELOP UNDERSTANDING OF QUASI-STATIC/ DYNAMIC TRANSITION
SCALING (J. MORTON)	DIMENSIONAL ANALYSIS	DEVELOP SCALING RULES
STATIC DEFLECTION OF PLATES AND CYLINDERS	CLOSED FORM AND FINITE ELEMENT 	INVESTIGATE DIFFERENCES IN PLATE AND CYLINDER RESPONSE TO IMPACT
PLATE RESPONSE TO IMPACT (D. S. CAIRNS/ S. R. SWANSON)	RAYLEIGH-RITZ TIME-MARCHING SHEAR DEFORMABLE 	QUANTITATIVE PREDICTION OF PLATE TO IMPACT

HERCULES

ANALYSIS RESULTS WERE INSTRUMENTAL IN DESIGN OF TEST MATRIX



IMPACTOR

$\omega_1^2 = k_1/m_1$

TARGET

$\omega_2^2 = k_2/m_2$

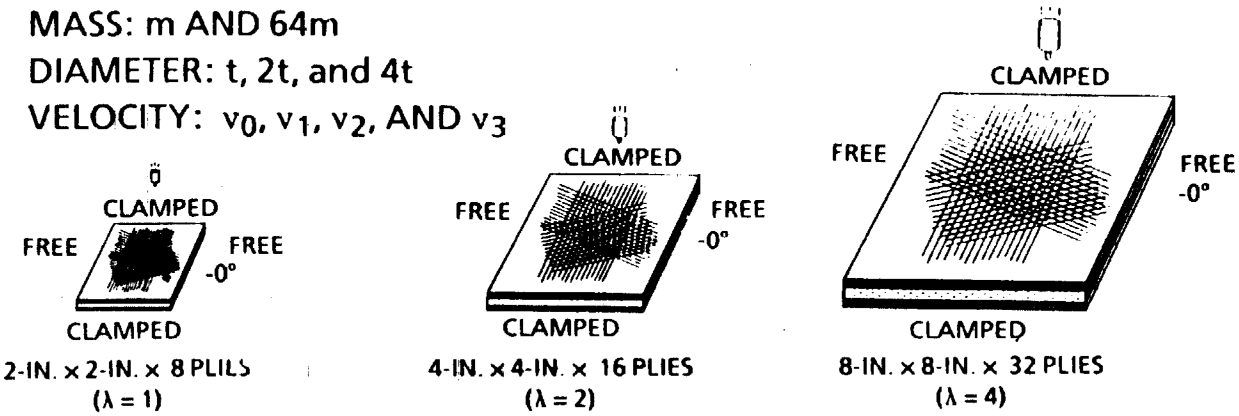
- RESULTS FROM 2 DOF SPRING-MASS IMPACT MODEL:
  - "TYPE" OF RESPONSE DEPENDS ON IMPACTOR/ TARGET FREQUENCY RATIO
  - "MAGNITUDE" OF RESPONSE ESSENTIALLY PROPORTIONAL TO IMPACTOR VELOCITY
- RESULTS FROM DIMENSIONAL ANALYSIS:
  - SIMULTANEOUS SCALEUP OF **BOTH** IMPACTOR AND TARGET GEOMETRY WHILE KEEPING MATERIALS AND IMPACTOR VELOCITY CONSTANT, LEADS TO IDENTICAL TARGET STRAINS AND CONTACT PRESSURES
- RESULTS FROM STATIC DEFLECTION MODELS:
  - RESPONSE TO IMPACT OF PLATES AND CYLINDERS CAN BE MADE SIMILAR

HERCULES

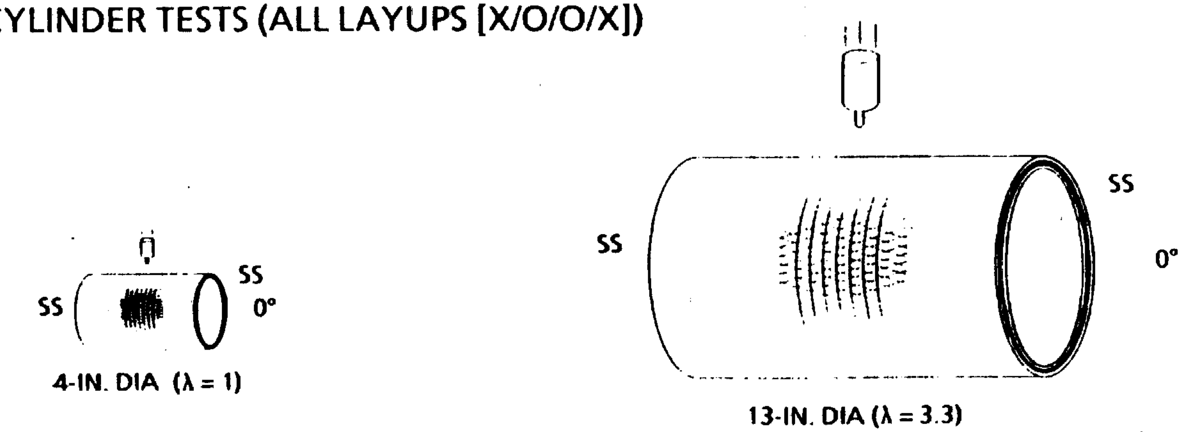
1-1980

EXPERIMENTAL TEST MATRIX BASED ON ANALYSIS RESULTS

- PLATE TESTS (ALL LAYUPS [X/O/O/X])
  - IMPACTOR:
    - MASS: m AND 64m
    - DIAMETER: t, 2t, and 4t
    - VELOCITY: v_0, v_1, v_2, AND v_3



- CYLINDER TESTS (ALL LAYUPS [X/O/O/X])



HERCULES

1 1977

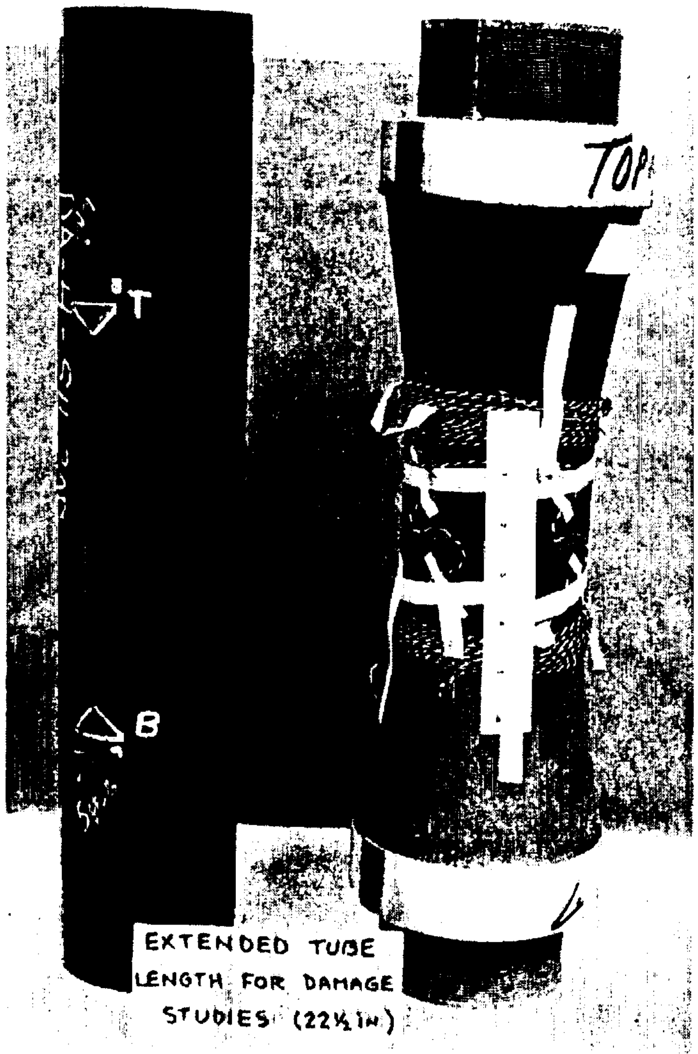
COMPREHENSIVE DATA WAS COLLECTED FROM IMPACT TESTS

- RESPONSE
  - IMPACT FORCE (QUASI-STATIC ONLY)
  - DEFLECTION (QUASI-STATIC ONLY)
  - STRAINS (SCALED GAGES)

- DAMAGE
  - VISUAL (FRONTSIDE/BACKSIDE DAMAGE)
  - NDE (DIGITAL ULTRASONICS)
  - DE (THERMAL DEPLY)

- RESIDUAL STRENGTH
  - TENSILE COUPONS FOR PLATES (SCALED)
  - BURST FOR CYLINDER (4" DIA. ONLY)

HERCULES



86C17258

1-1942

4-IN. TUBE PRESSURIZED SPECIMEN CONFIGURATION FOR RESIDUAL STRENGTH EVALUATION

HERCULES

ORIGINAL PAGE
BLACK AND WHITE PHOTOGRAPH

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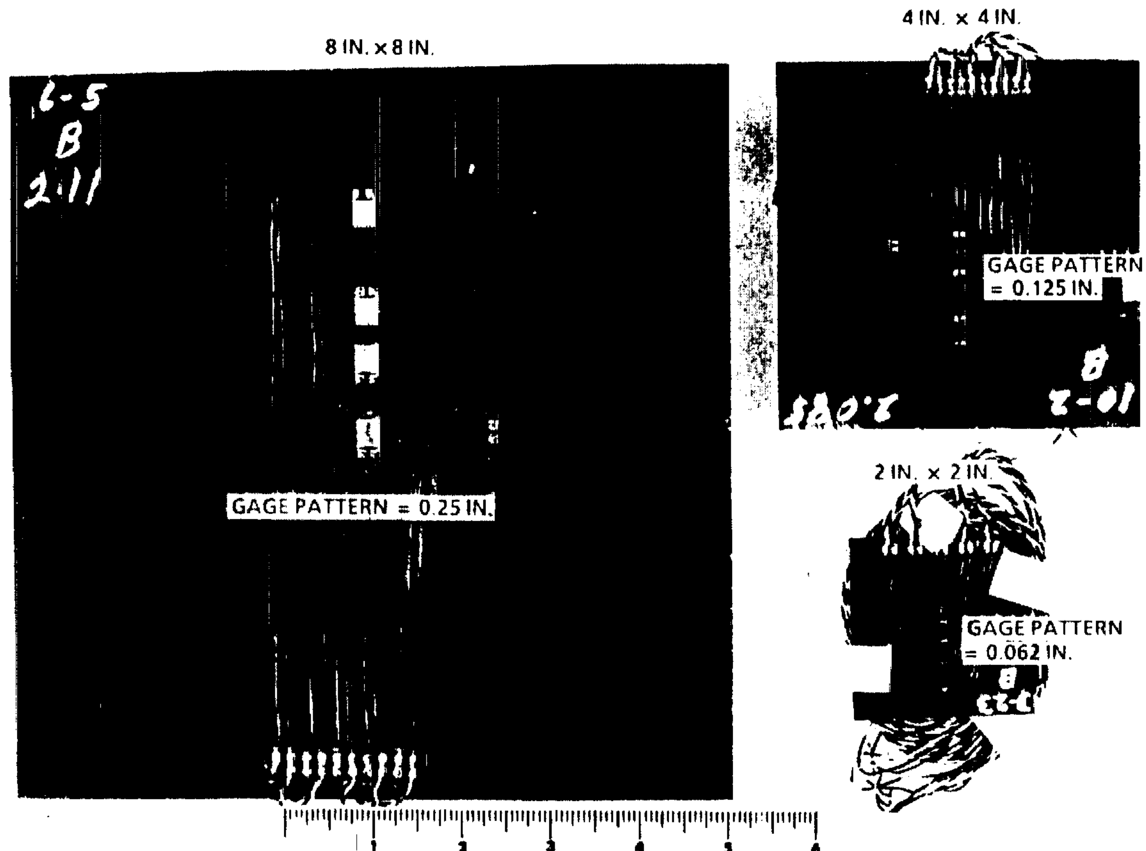
OF POOR QUALITY

SCALED PLATES AND STRAIN GAGES



89C13123

1-1949

HERCULES

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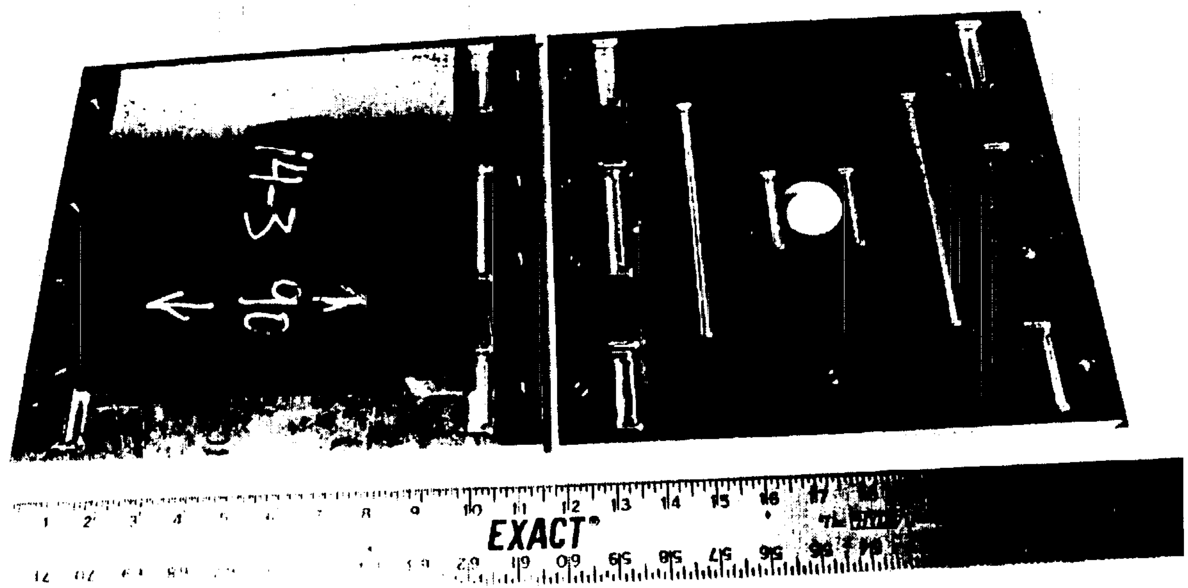
FLAT PLATE IMPACT HOLDING FIXTURE



88C7660

1-1944

HERCULES

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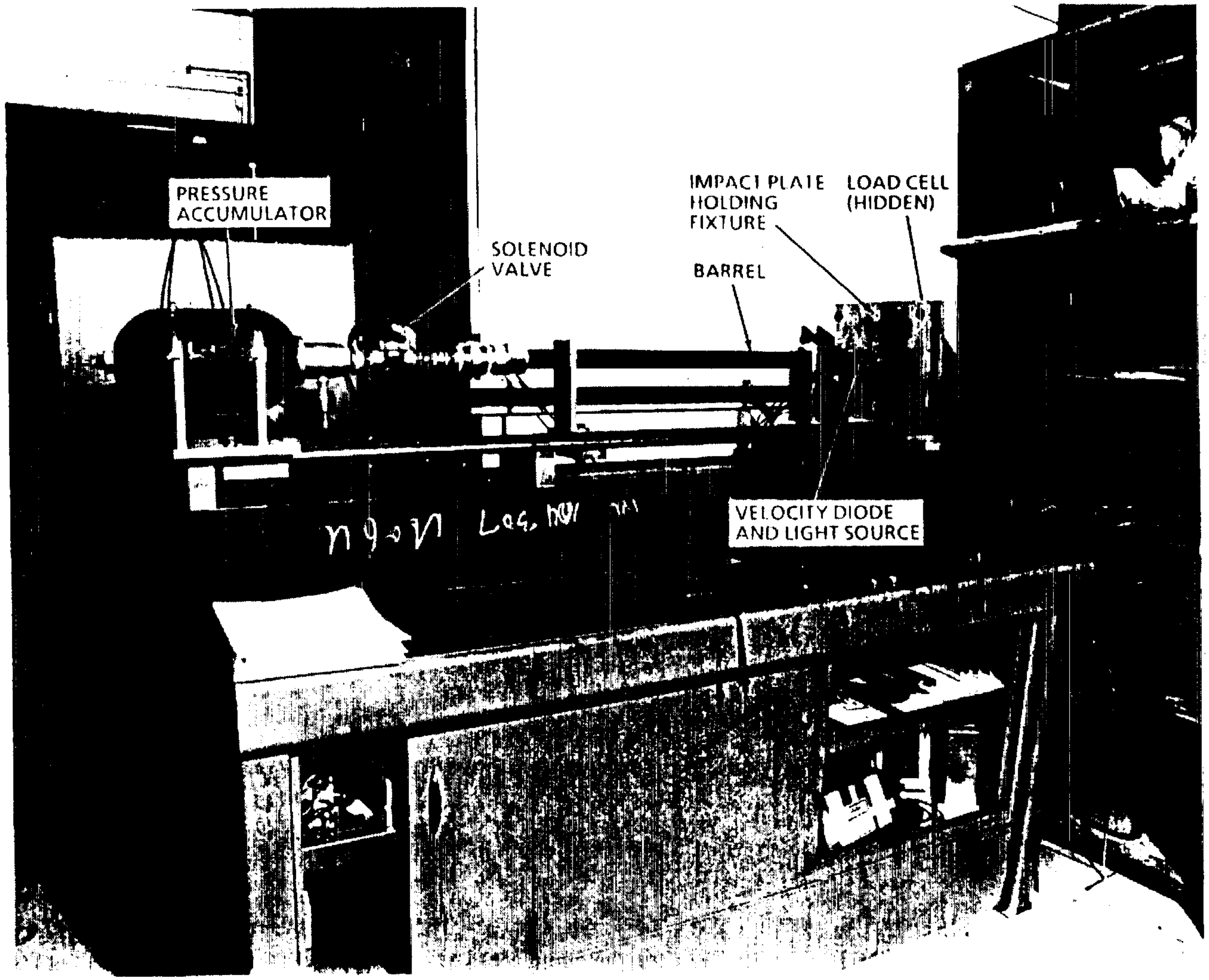
AIRGUN IMPACT TEST FIXTURE



88C7662

1-1947

HERCULES

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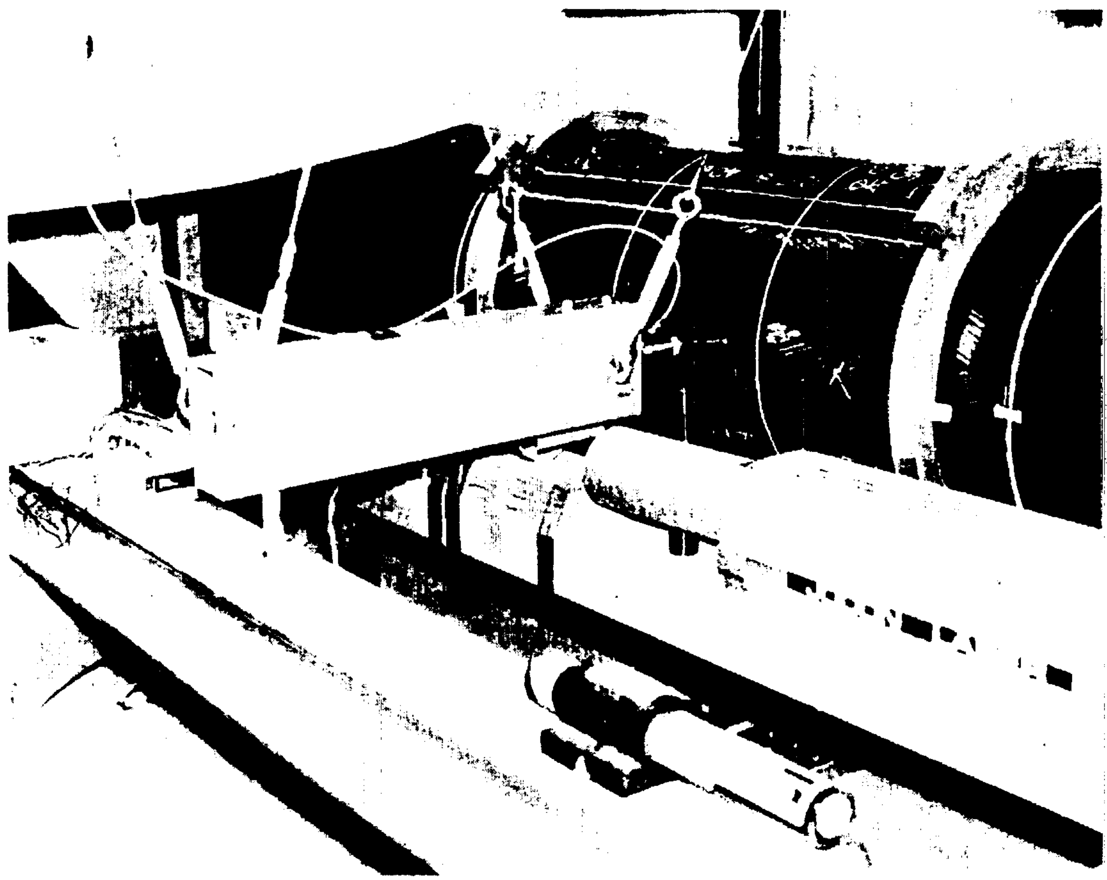
12-IN. CYLINDER IMPACT SITE CONFIGURATION--
TWELVE IMPACTS PER CYLINDER

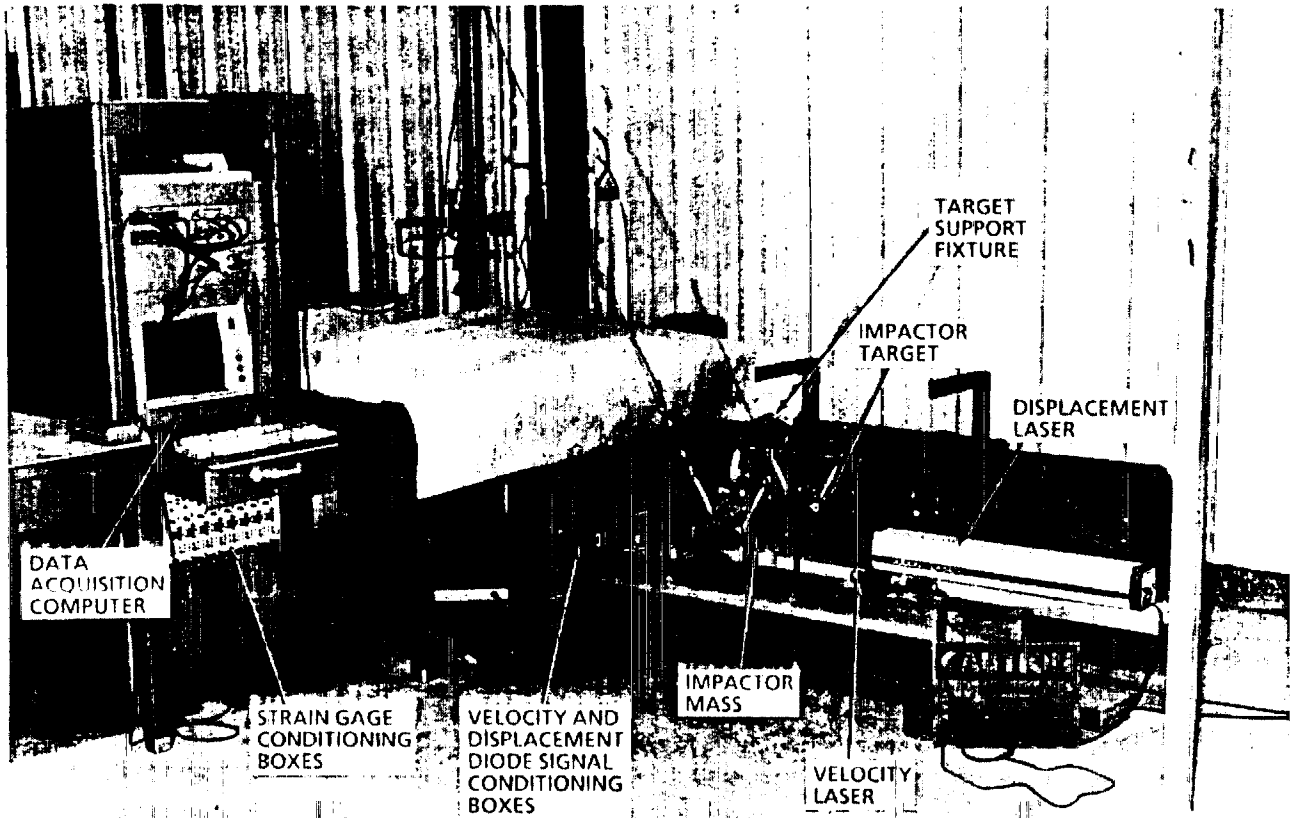


89C10483

1-1945

HERCULES

PENDULUM IMPACT TEST FIXTURE



89C10482

1-1948

HERCULES

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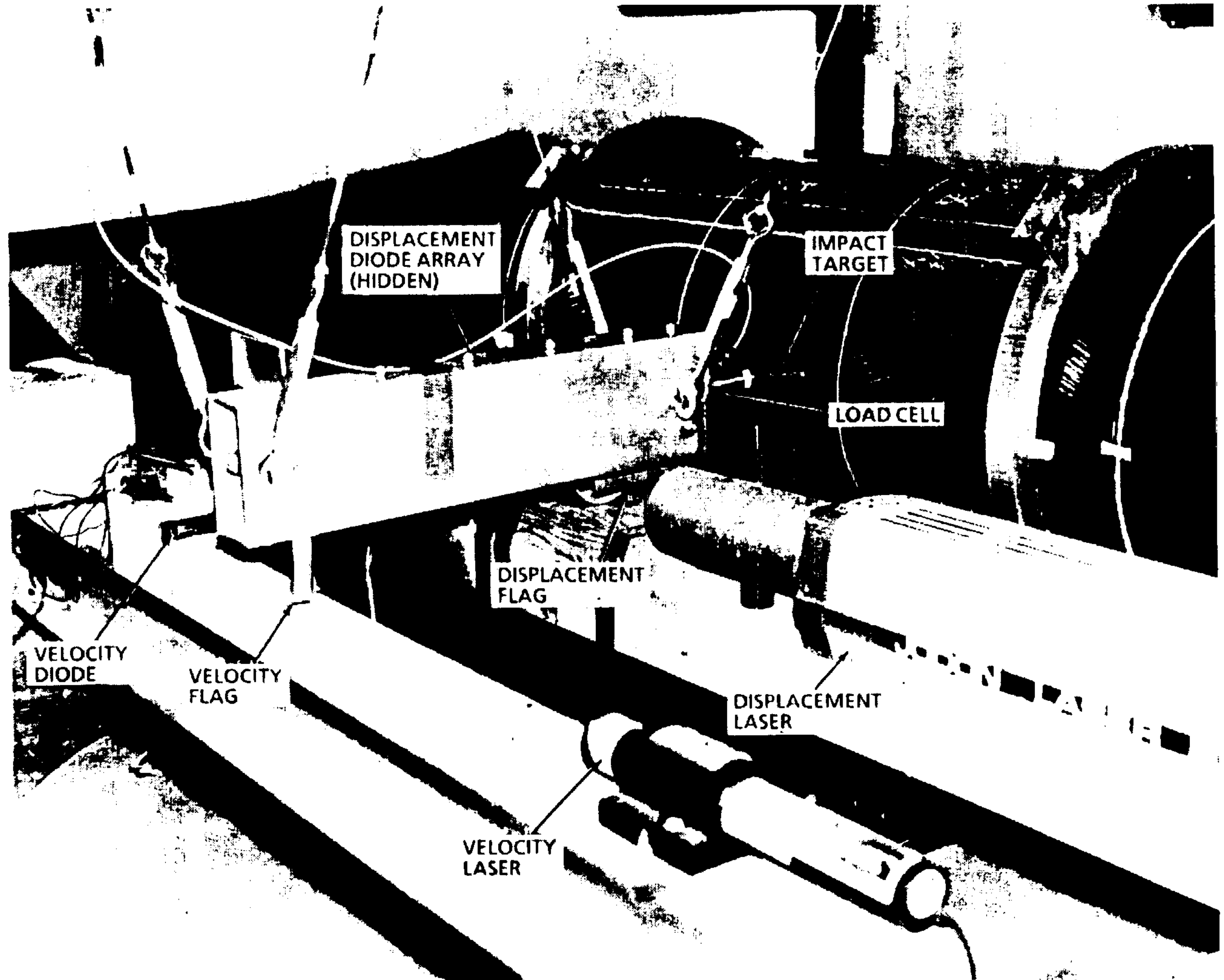
PENDULUM IMPACT EVENT INSTRUMENTATION



89C10483

HERCULES

1-1946

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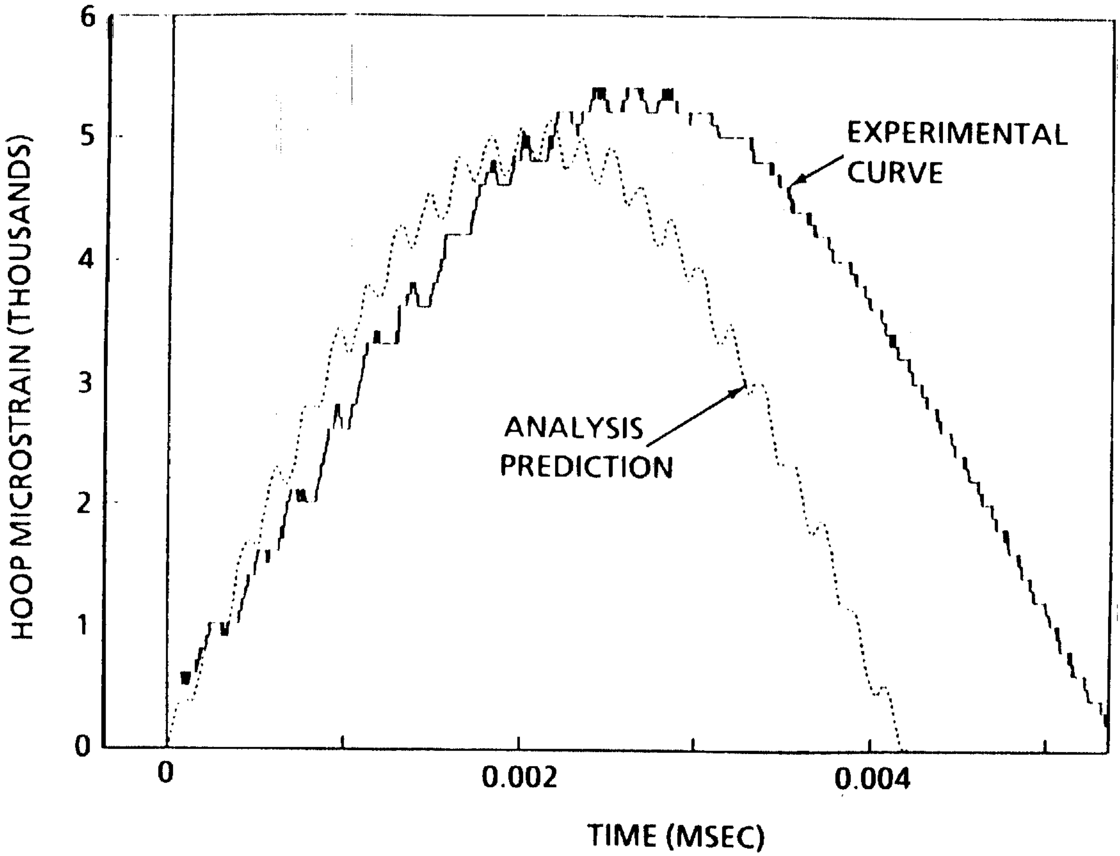
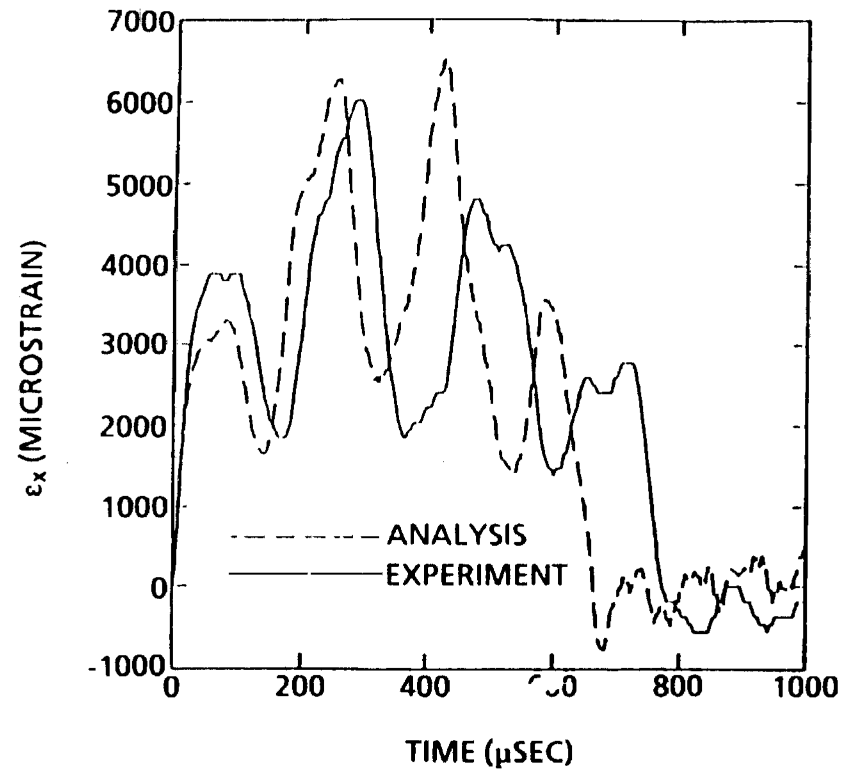
QUASI-STATIC RESPONSE OF PLATES PREDICTED WELL BY RAYLEIGH-RITZ, SHEAR DEFORMABLE, IMPACT MODEL



HERCULES

1 19/8

DYNAMIC RESPONSE OF PLATES PREDICTED WELL BY RAYLEIGH-RITZ, SHEAR DEFORMABLE, IMPACT MODEL



COMPARISON OF ANALYSIS WITH EXPERIMENT FOR STRAIN ε_x BEHIND IMPACT POINT IN 4-IN. x 4-IN. PLATE, IMPACT VELOCITY v_0 = 4.57 M/SEC (15 FT/SEC)

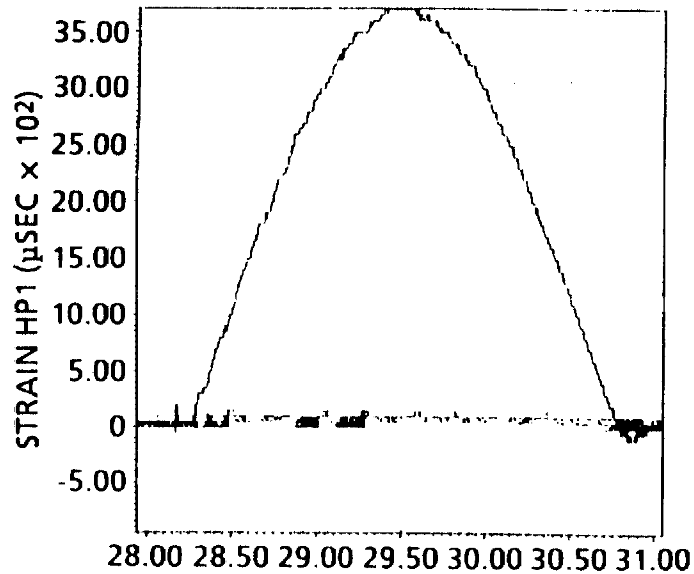
HERCULES

1 1981

PLATE RESULTS OBTAINED SUPPORT CONCLUSIONS FROM 2 DOF MODEL

- "TYPE" OF RESPONSE DEPENDS ON IMPACTOR/TARGET FREQUENCY RATIO (NOT VELOCITY OF IMPACTOR)

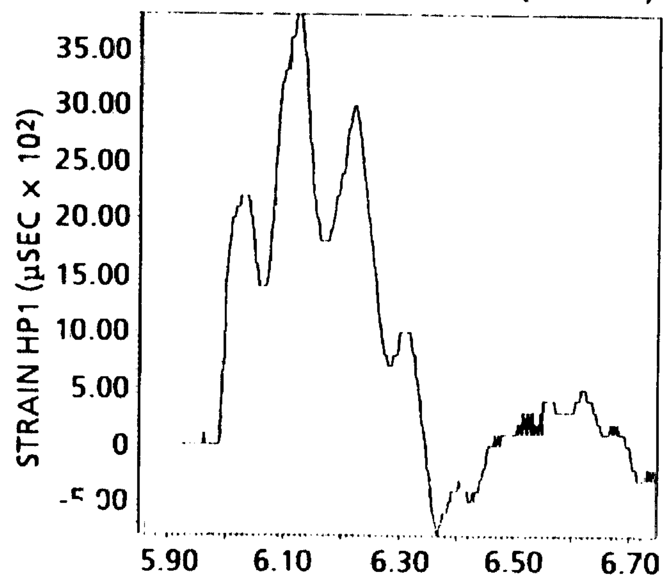
2-IN. × 2-IN. FLAT PLATE IMPACT--6-23
HOOP VERSUS TIME (PENDULUM)



QUASI-STATIC RESPONSE (PENDULUM)

ω_1 = 1600 Hz
ω_2 = 2260 Hz

IMPACT ON 2-IN. × 2-IN. WITH v_0 --3-15
HOOP NO. 1 VERSUS TIME (AIRGUN)



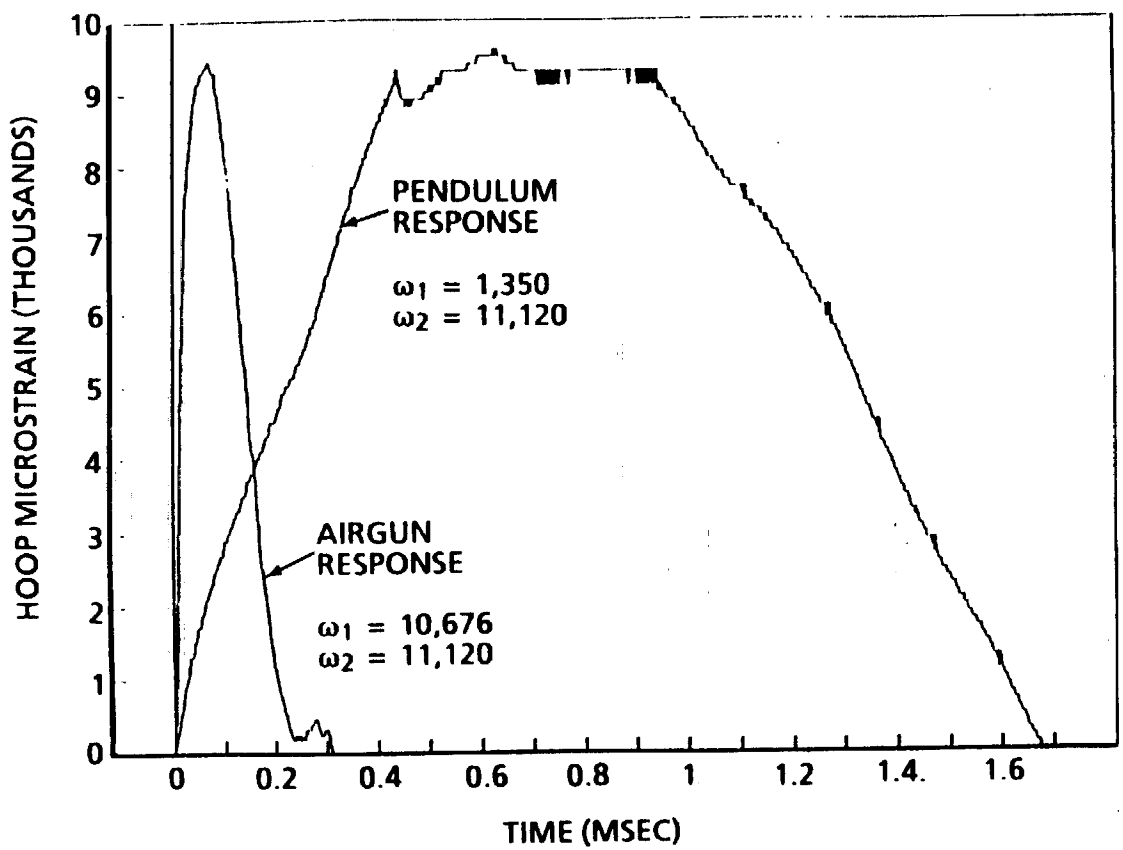
DYNAMIC RESPONSE (AIRGUN)

ω_1 = 12,500 Hz
ω_2 = 2,260 Hz

1-1981

HERCULES

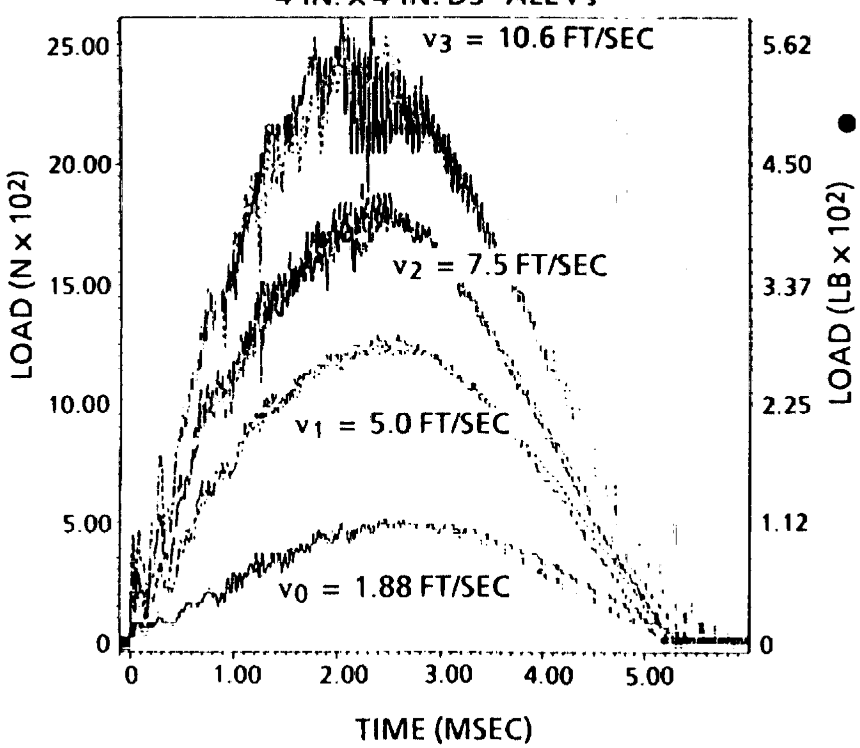
CYLINDER RESPONSE ALSO FOLLOWS THE FREQUENCY RATIO RULE



HERCULES

RESULTS OBTAINED SUPPORT CONCLUSIONS FROM 2 DOF MODEL

4-IN. × 4-IN. FLAT PLATE IMPACT --7-2; FILE: IM72V3
4-IN. × 4-IN. D3--ALL v's



- MAGNITUDE OF RESPONSE ESSENTIALLY PROPORTIONAL TO IMPACTOR VELOCITY (UP TO DAMAGE INITIATION)
- 2 DOF MODEL ACCURATELY PREDICTS CONTACT FORCE IN QUASI-STATIC RANGE

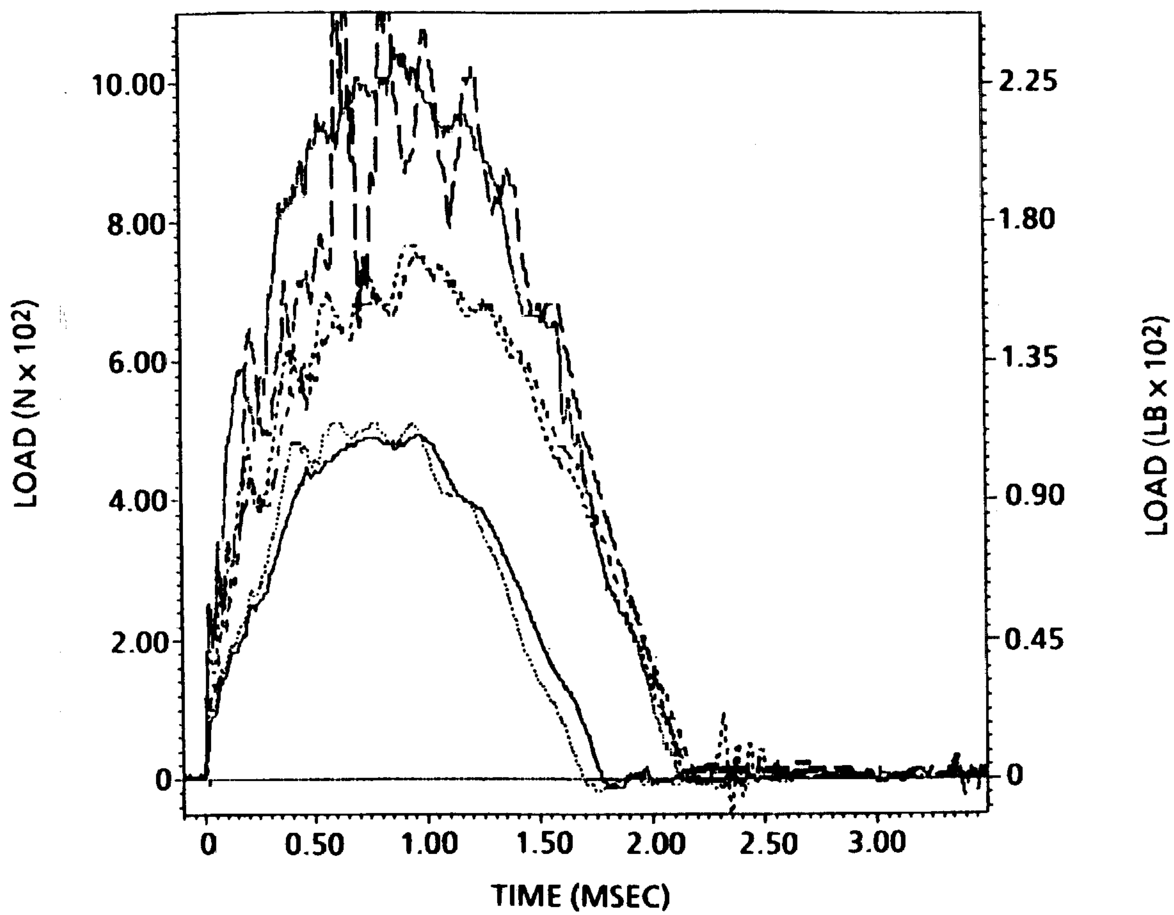
$$F(t) = \sqrt{2k_{eff}E}\sin\sqrt{\frac{k_{eff}}{m_1}}\,t = v\sqrt{m_1 k_{eff}}\sin\sqrt{\frac{k_{eff}}{m_1}}\,t$$

HERCULES

1-1987

THE TRENDS OBSERVED ON THE FLAT PLATES ARE CONSISTENT WITH THOSE OF THE CYLINDERS

4-IN. TUBE IMPACT TESTS--09D; FILE: 4IN9D
4-IN. D3 ALL v's

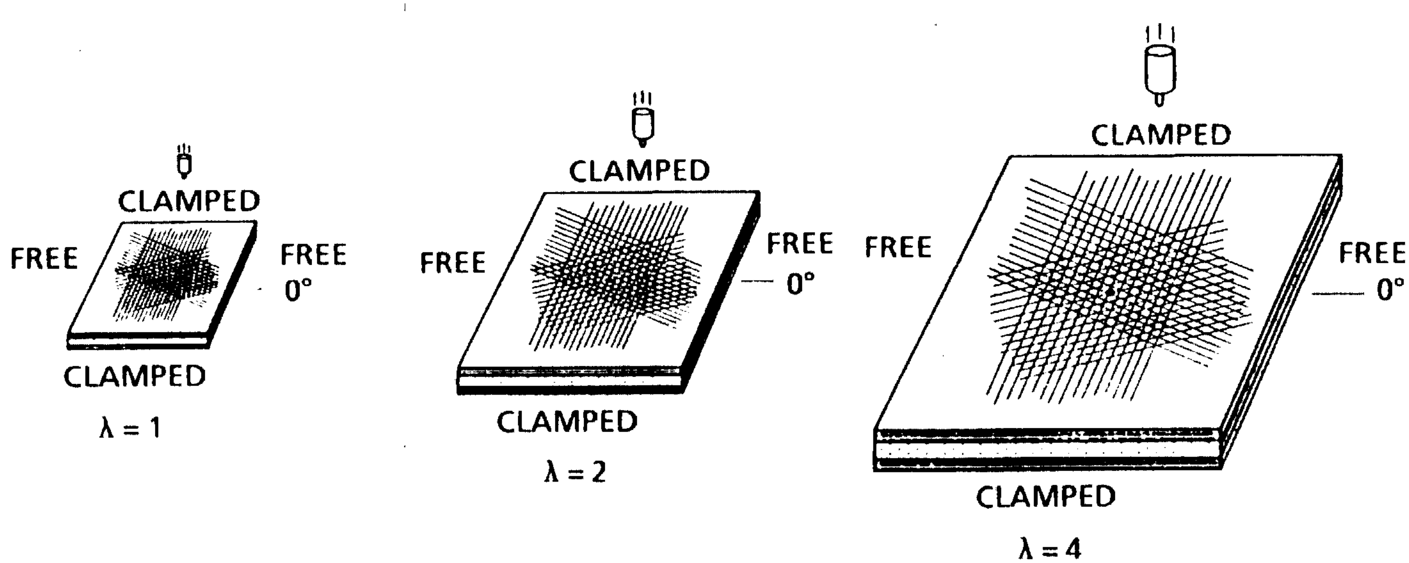


HERCULES

1 1961

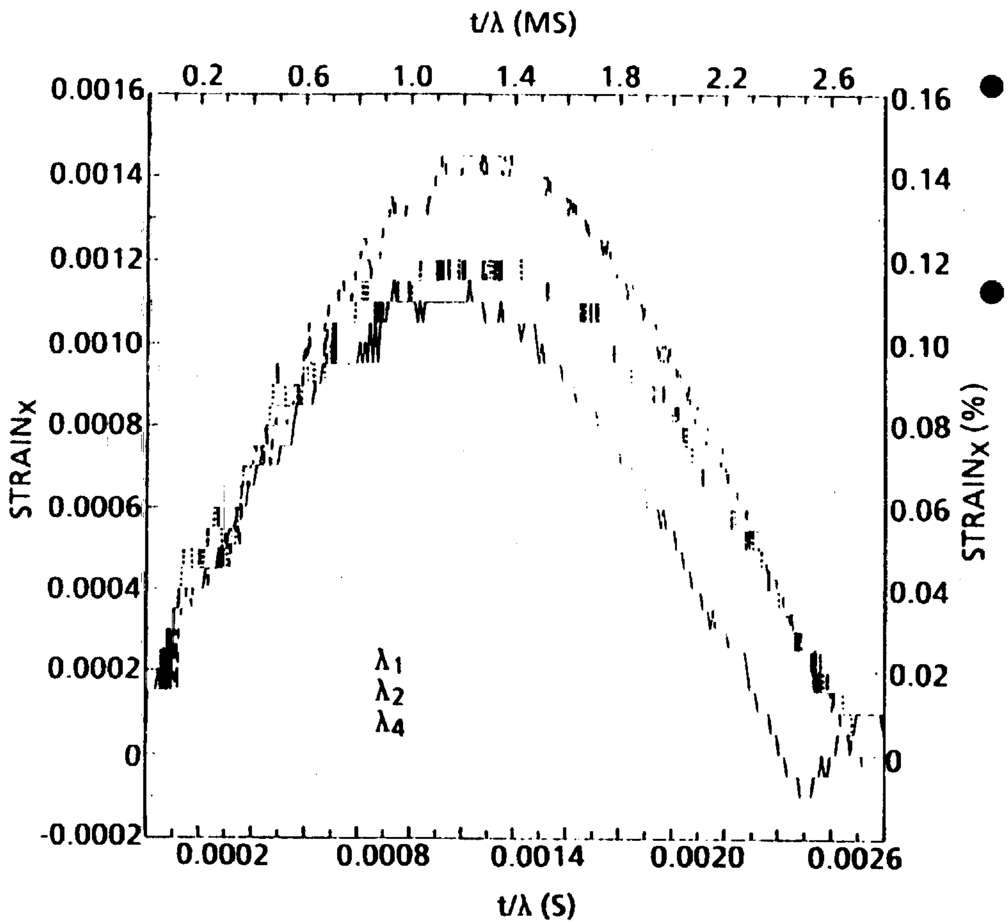
RESPONSE SCALES AS PREDICTED BY DIMENSIONAL ANALYSIS

- KEEPING MATERIALS AND IMPACT VELOCITY CONSTANT WHILE DIMENSIONS OF IMPACTOR AND TARGET SCALEUP BY $\lambda = 1,2,4$ RESULTS IN
  - CONSTANT STRAIN
  - CONSTANT CONTACT PRESSURE (FORCE SCALES AS λ^2)
  - IMPACT DURATION INCREASES AS λ



HERCULES

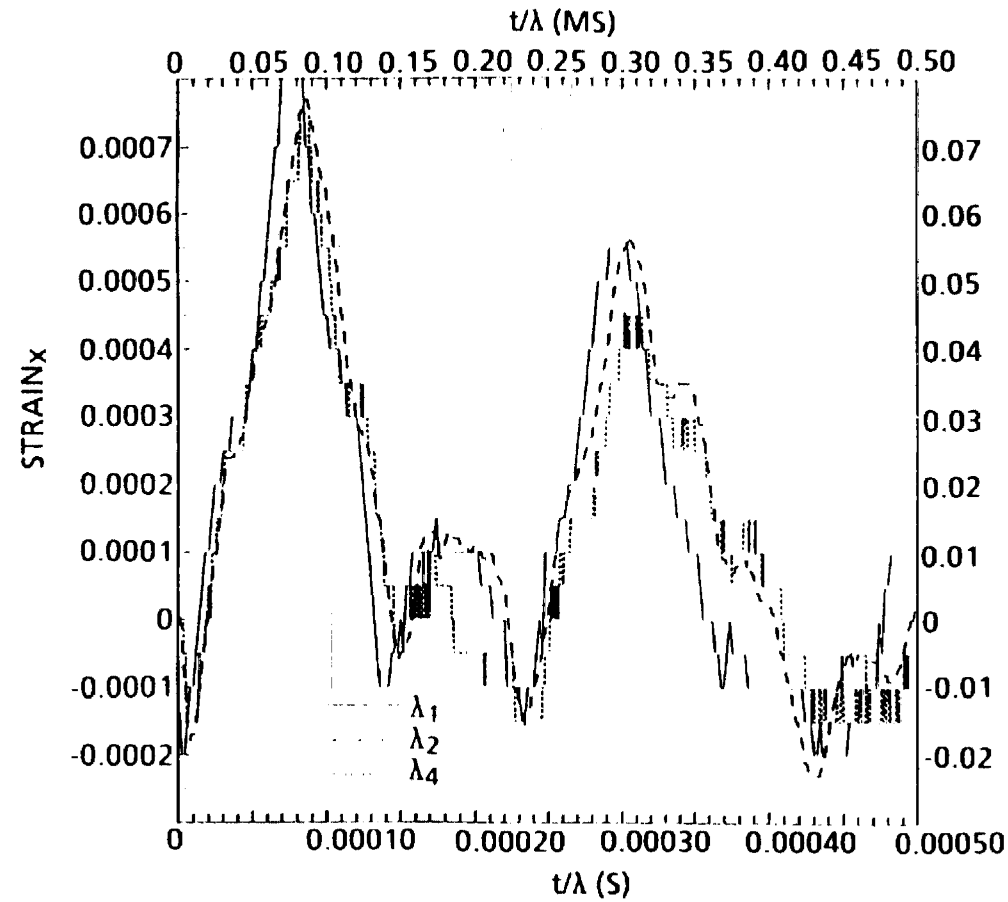
1-1968



- QUASI-STATIC PLATE STRAIN RESPONSE SCALES FOR SCALED IMPACT
- EVENT DURATION IS SCALED BY λ AS PREDICTED BY THE SCALING RULES

HERCULES

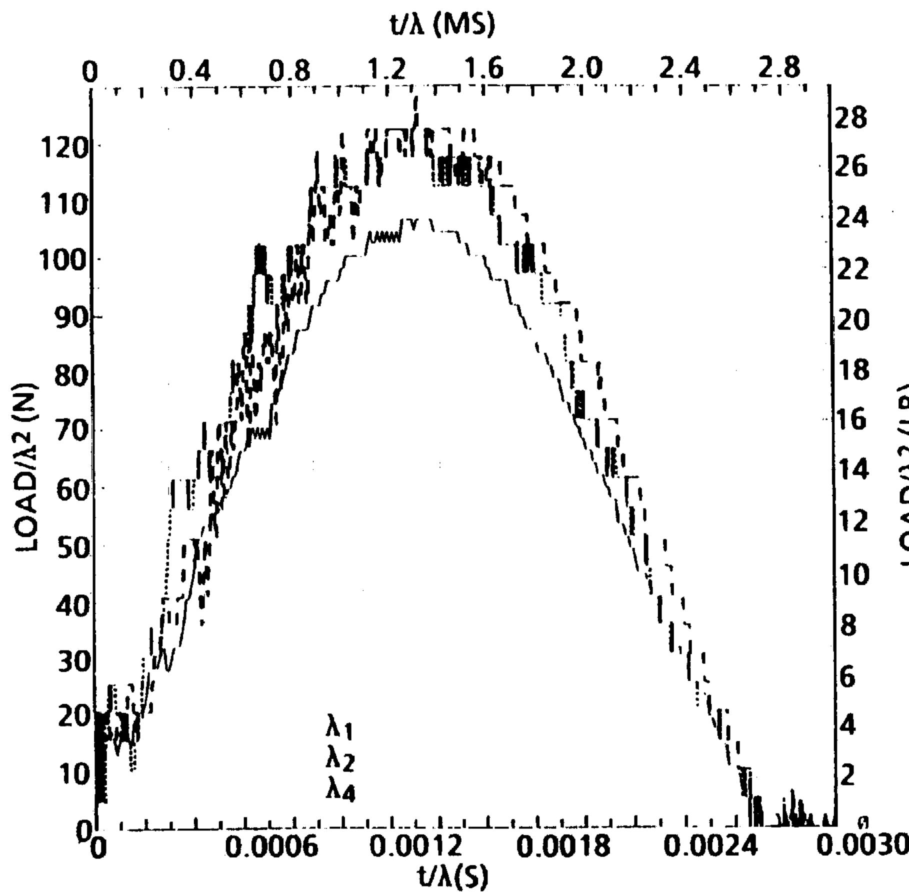
1-1971



- DYNAMIC PLATE STRAIN RESPONSE SCALES FOR SCALED IMPACT
- EVENT DURATION IS SCALED BY λ AS PREDICTED BY THE SCALING RULES

HERCULES

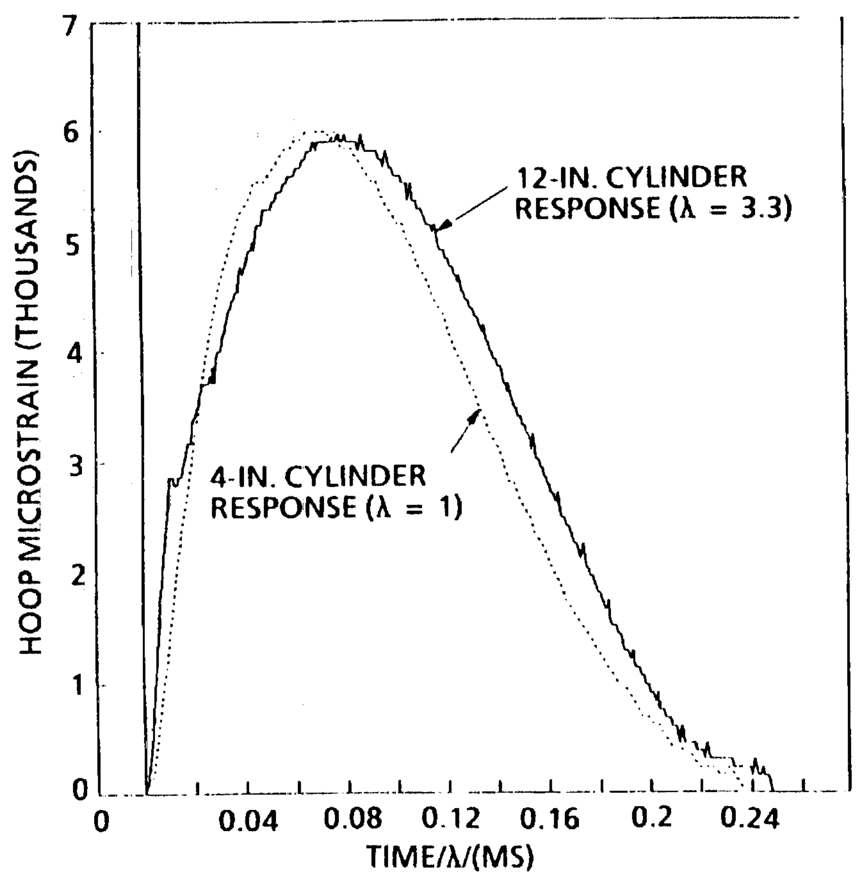
1-19/1



- CONTACT FORCE IS SCALED BY λ^2 AS PREDICTED BY THE SCALING RULES

HERCULES

1 1972



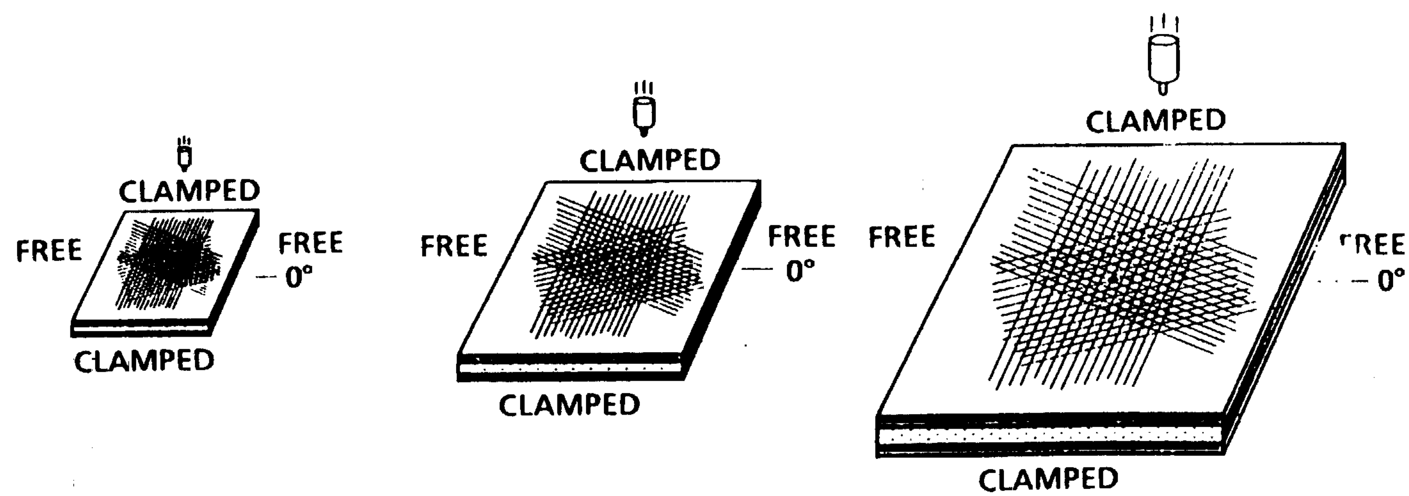
- CYLINDER STRAIN RESPONSE SCALES FOR SCALED IMPACT
- EVENT DURATION IS SCALED BY λ AS PREDICTED BY THE SCALING RULES

HERCULES

1-1974

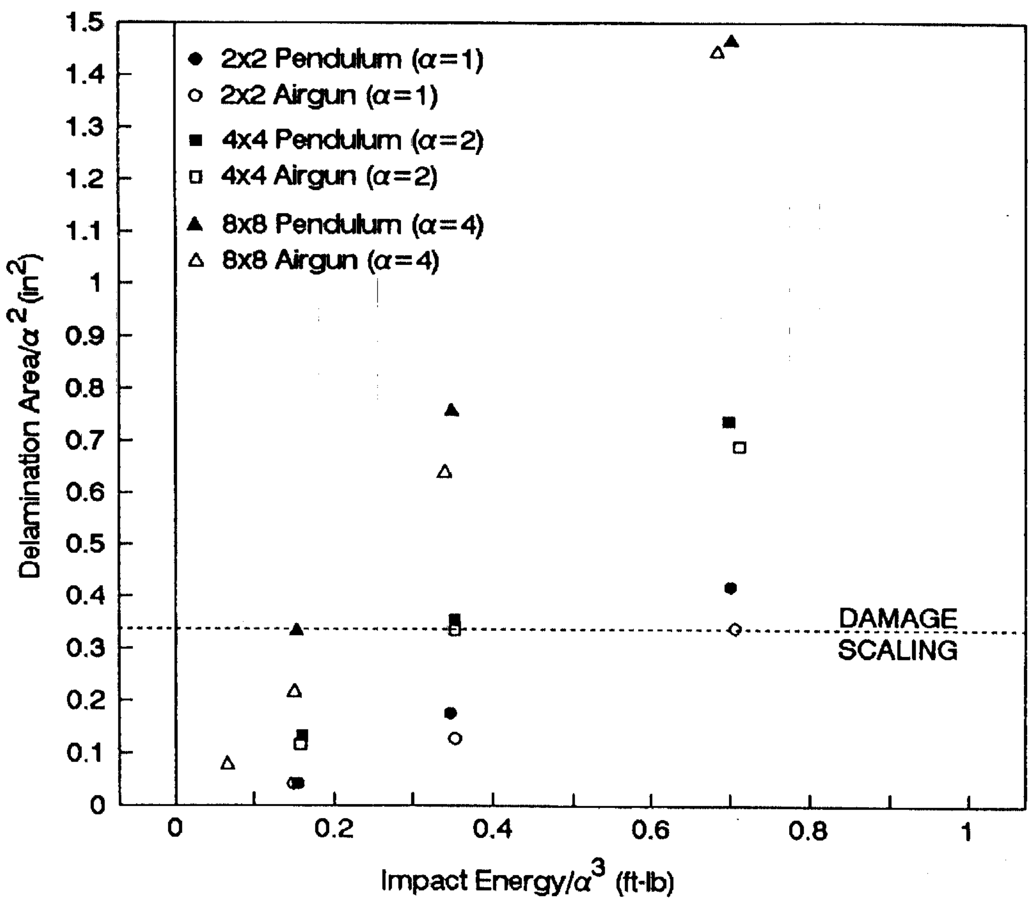
SIZE EFFECTS APPEAR IN DAMAGE AND STRENGTH SCALING

- IN SPITE OF IDENTICAL STRAINS AND CONTACT PRESSURES, MORE DAMAGE OCCURS AND STRENGTHS GO DOWN AS TARGET GETS LARGER
- DIFFERENT FAILURE MECHANISMS MAY BE AT WORK FOR DIFFERENT TYPES OF DAMAGE
  - DELAMINATION AREA APPEARS TO BE RELATED TO IMPACT ENERGY
  - FIBER BREAKAGE APPEARS TO BE RELATED TO IMPACT STRESS



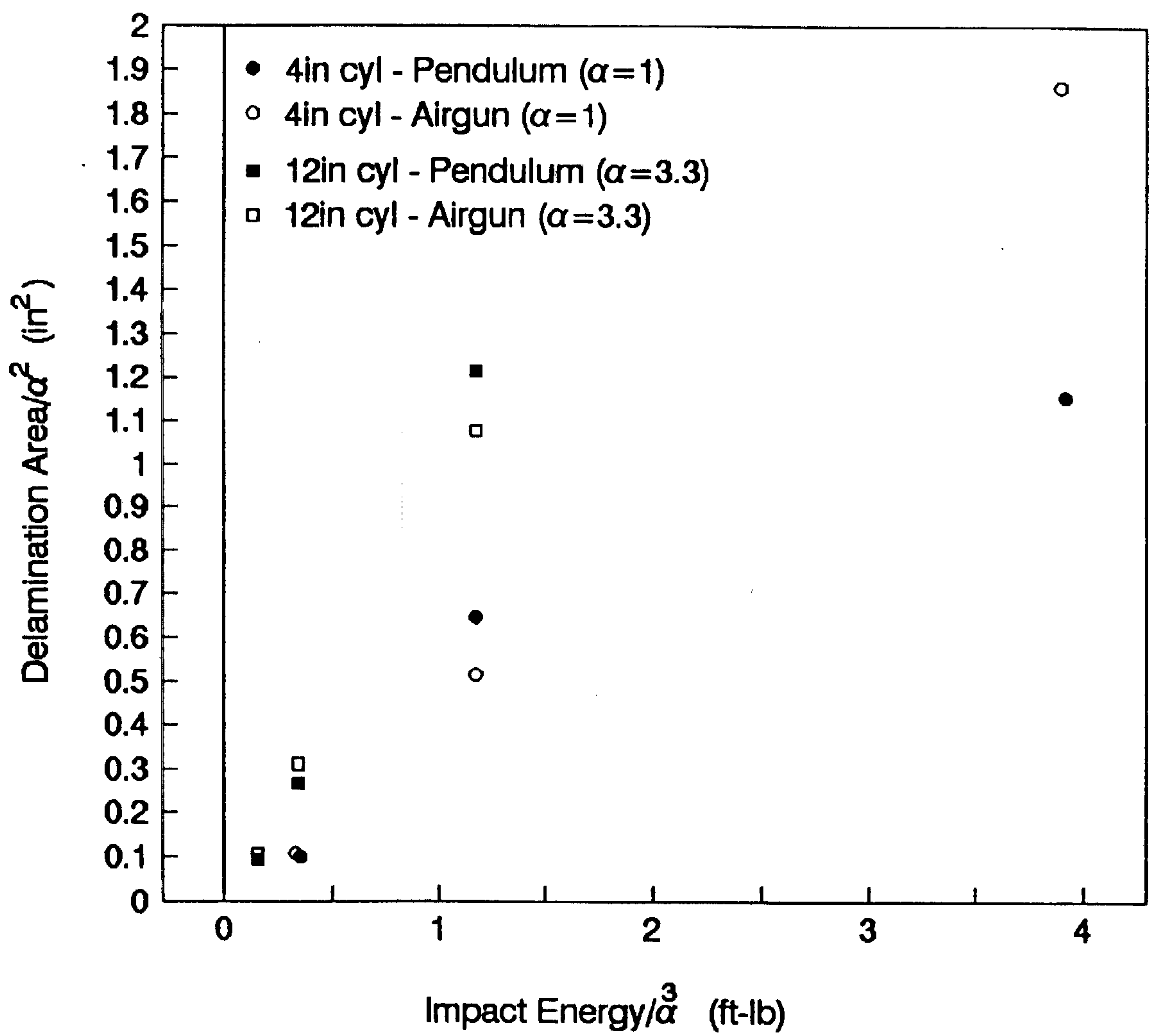
HERCULES

1-1965



- DELAMINATION AREAS INDICATE THE FRACTURE MECHANICS SCALING APPROACH IS VALID
- IMPACTOR SIZE HAD NO DISCERNABLE EFFECT
- QUASI-STATIC AND DYNAMIC DOMAINS PRODUCED SIMILAR DELAMINATION DAMAGE

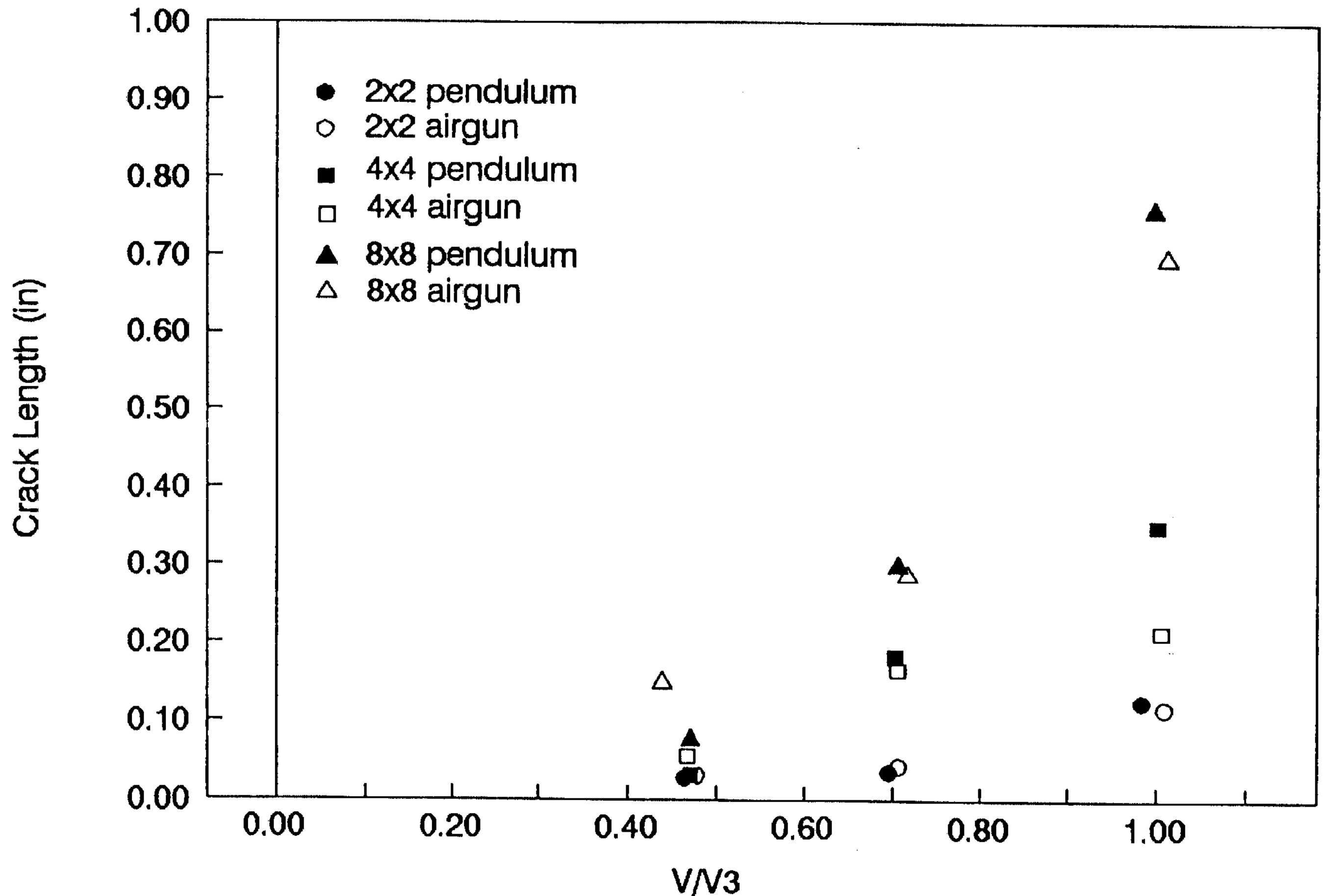
HERCULES



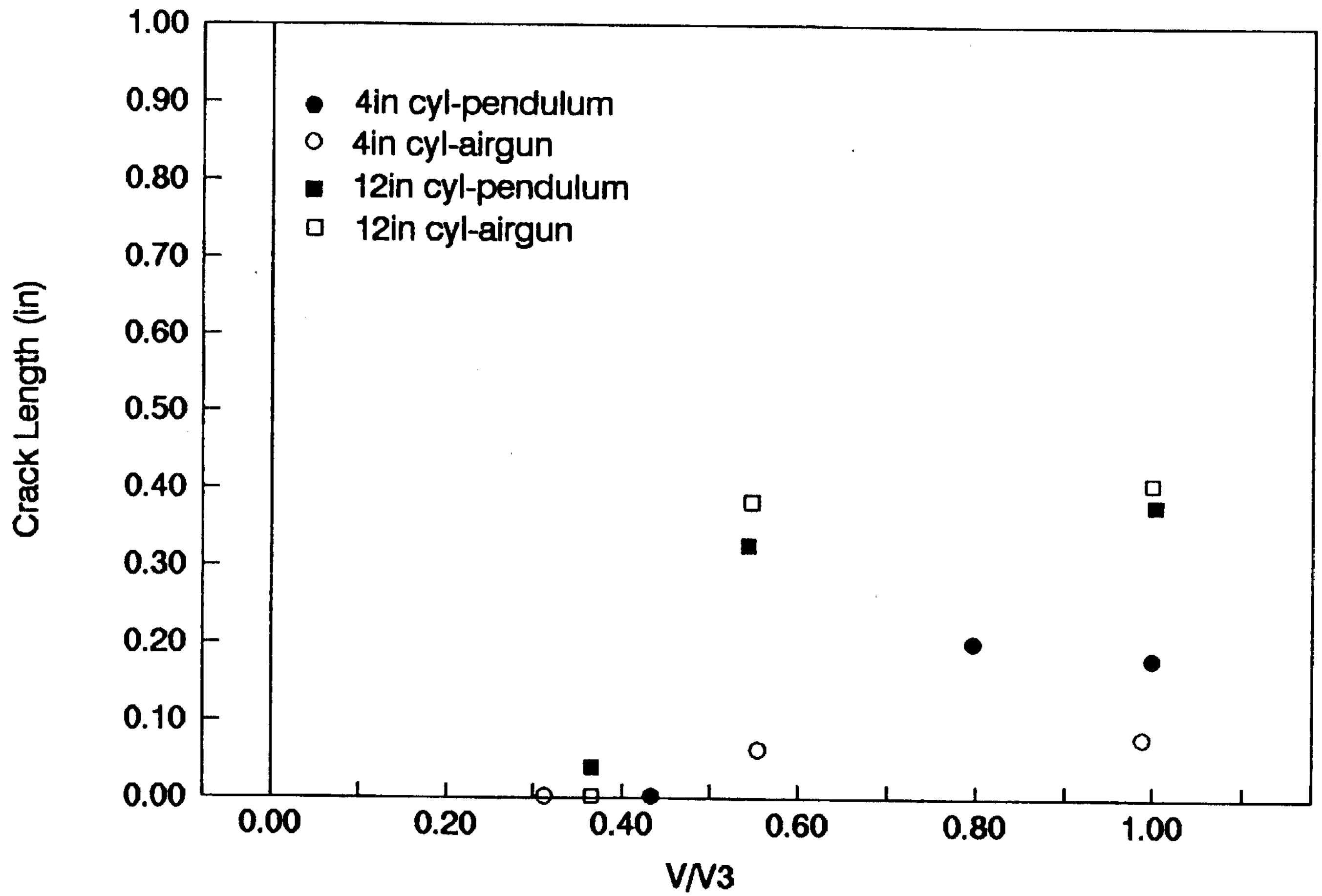
- CYLINDERS FOLLOWED TRENDS SIMILAR TO THOSE OF THE PLATES
- MORE SCATTER WAS FOUND IN THE CYLINDER DELAMINATION DATA THAN IN THE PLATE DATA

HERCULES

FIBER BREAKAGE APPEARS TO BE RELATED TO IMPACT STRESS FOR PLATES

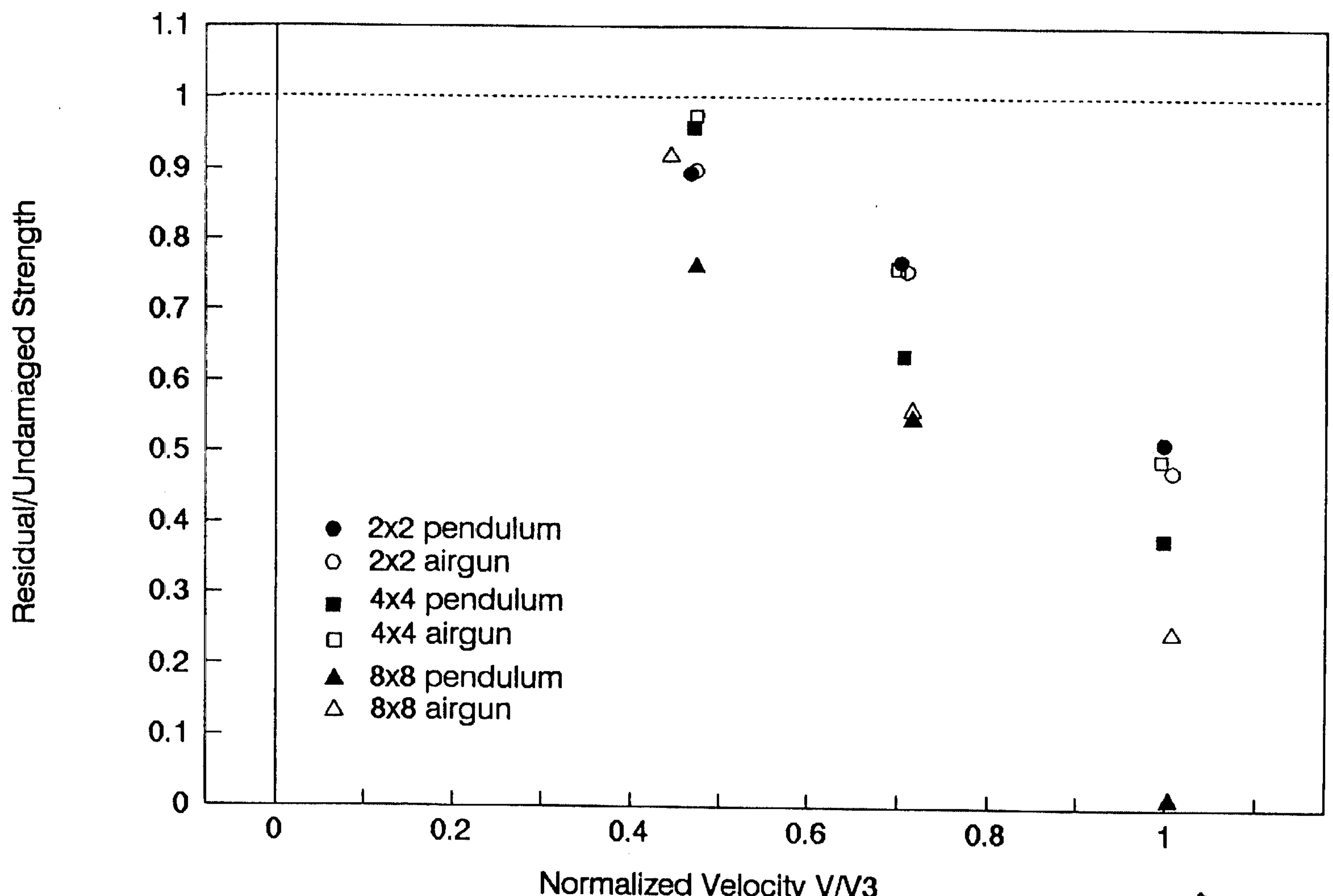


FIBER BREAKAGE DATA FOR CYLINDERS IS LESS CONCLUSIVE BECAUSE OF INCREASED DATA SCATTER



HERCULES

RESIDUAL STRENGTH FOLLOWS DAMAGE SCALING TRENDS FOR PLATES



HERCULES

4-IN. × 4-IN. PLATE VISUAL IMPACT DAMAGE

V1D1

V1D2

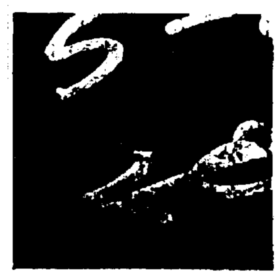
V1D3

V2D1

V2D2

V2D3

V3D1



V3D2

V3D3

88C11362, 88C11364, 88C11366
88C11372, 88C11409, 88C11432
88C11435, 88C11440, 88C11447

1-1941

HERCULES

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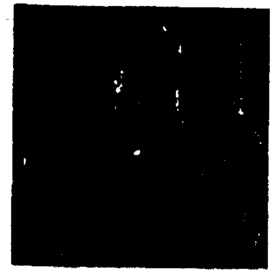
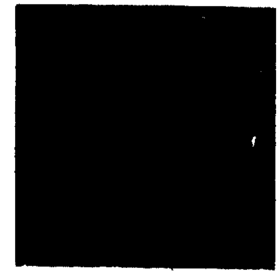
4-IN. TUBE VISUAL IMPACT DAMAGE

V1D1



V1D2



V1D3

V2D1

V2D2

V2D3

V3D1

V3D2

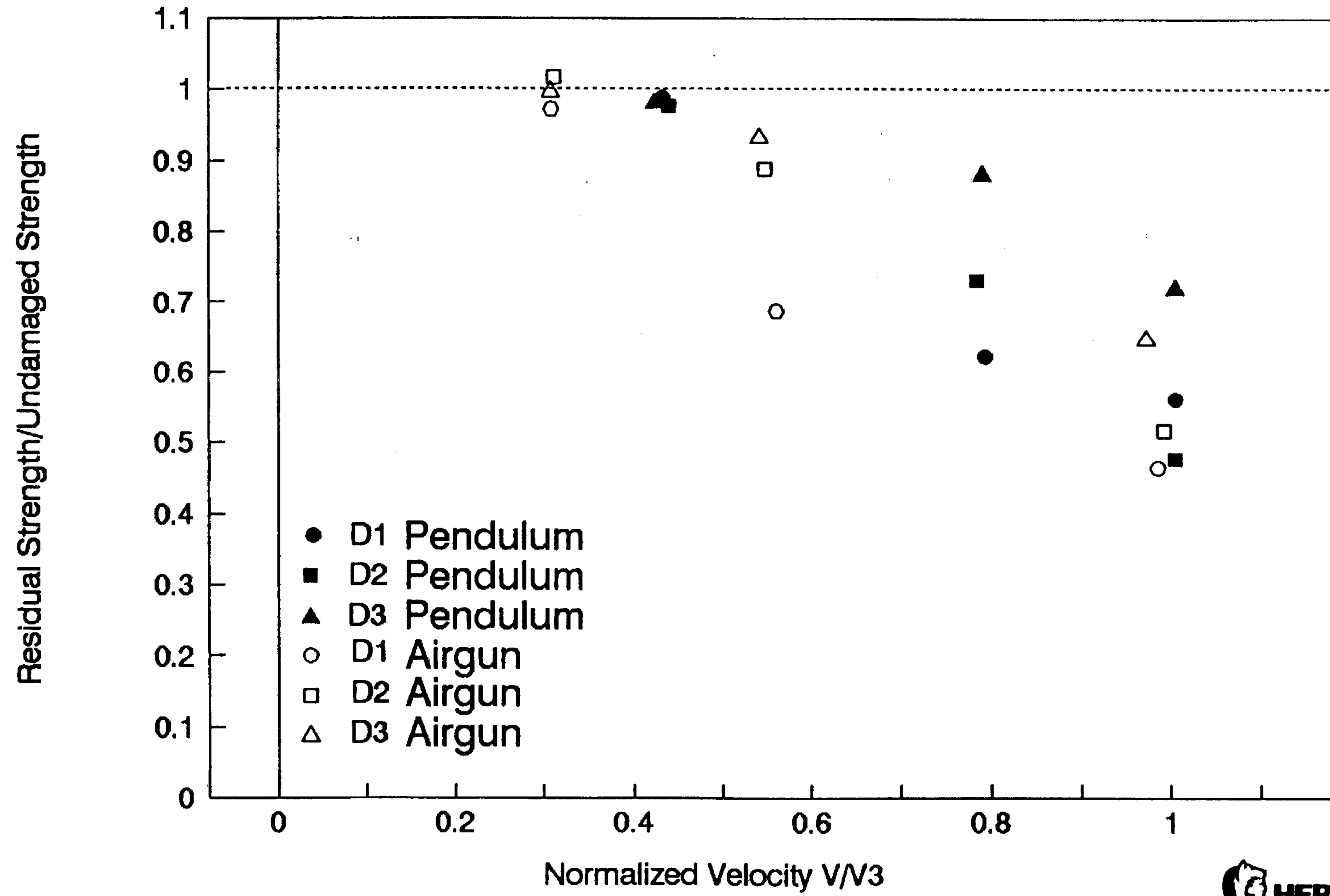
V3D3

89C6560, 89C6561, 89C6562
89C6564, 89C6566, 89C6585
89C6586, 89C6589, 89C6593

1-1940

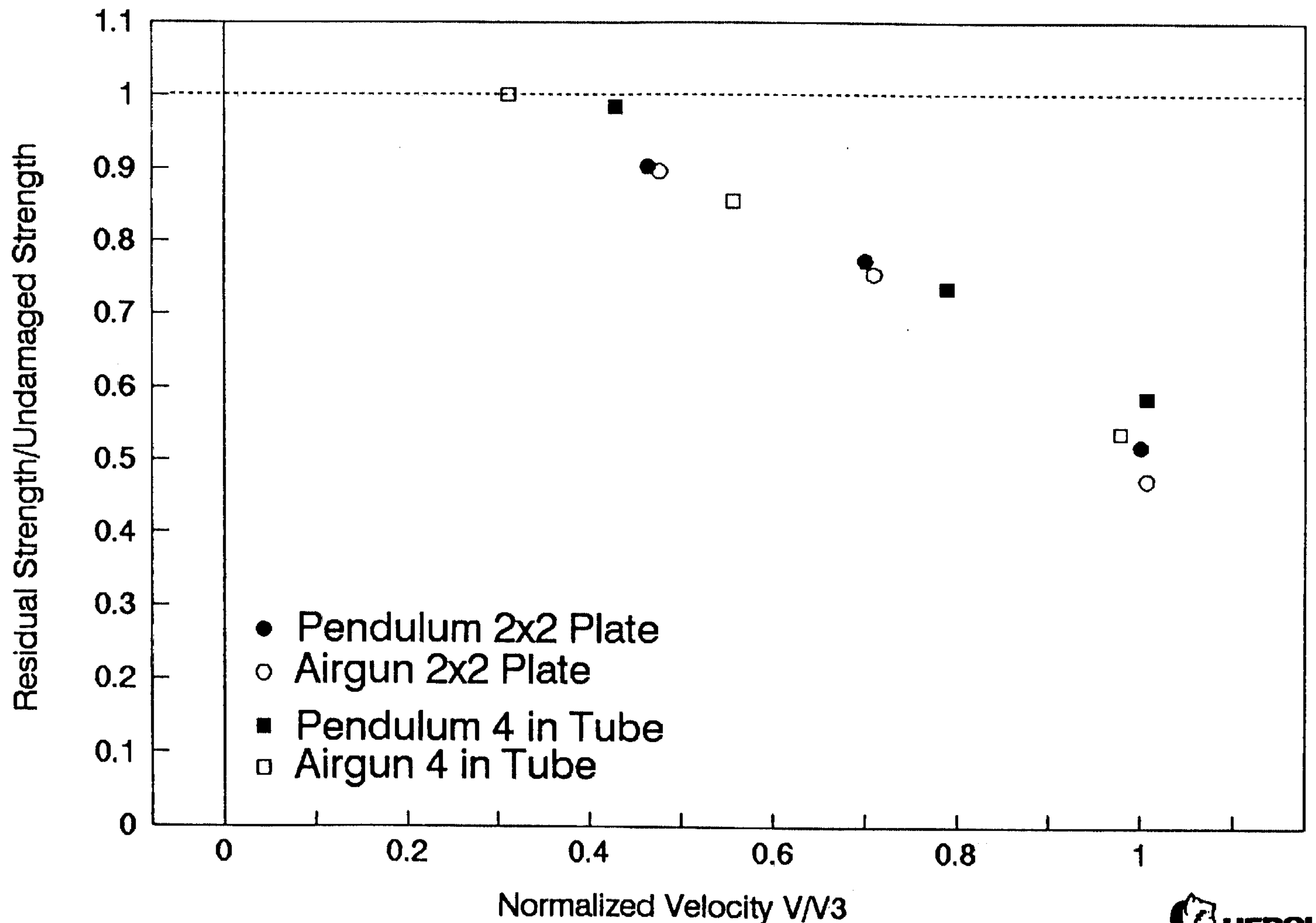
HERCULES

TUBE RESIDUAL STRENGTH DATA SUPPORTS DAMAGE TRENDS, WITH IMPACTOR GOEMETRY EFFECTS



HERCULES

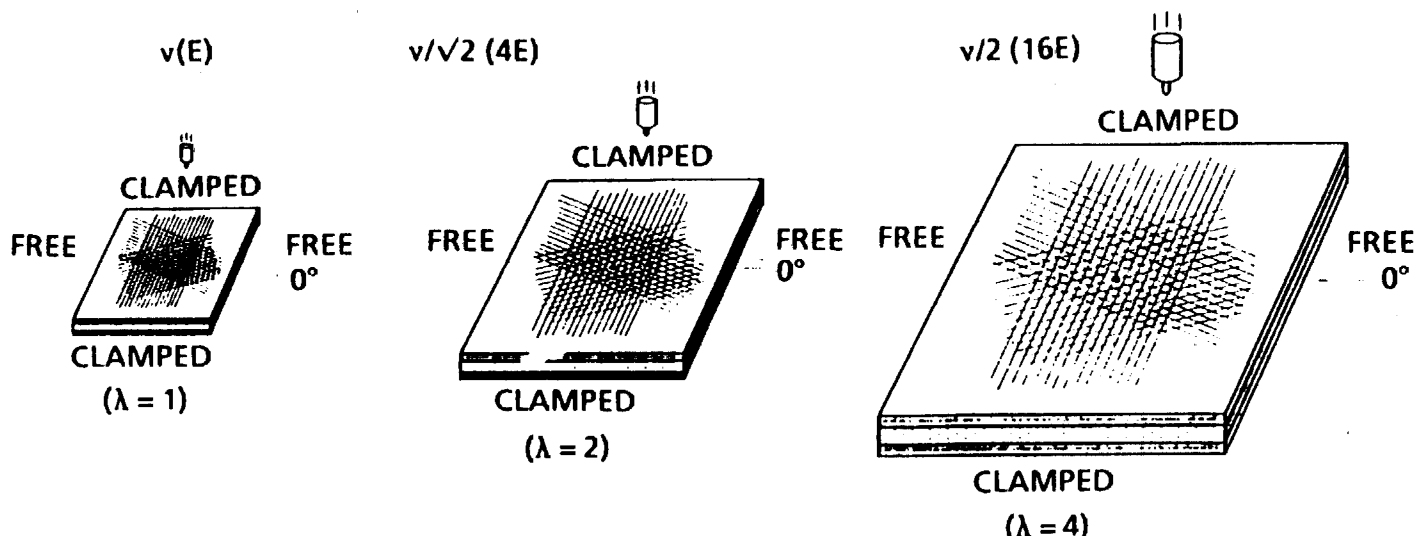
SUPERIMPOSED RESIDUAL STRENGTH DATA FOR 2X2 FLAT PLATES AND 4-IN. TUBES (SAME LAYUP AND THICKNESS) SUPPORTS FEASIBILITY OF SUBELEMENT APPROACH



HERCULES

DAMAGE AND STRENGTH SCALING MAY BE ABLE TO BE ACHIEVED THROUGH RESPONSE SCALING WITH REDUCED IMPACT VELOCITIES

- SCALING IMPACTOR AND TARGET GEOMETRIES RESULTS IN SAME "TYPE" OF RESPONSE
- REDUCING IMPACT VELOCITY AS SCALE INCREASES LOWERS APPLIED STRESSES/ STRAINS AND MAY RESULT IN SIMILAR DAMAGE AND STRENGTH LOSS WITH SCALEUP
- PRELIMINARY RESULTS SUGGEST VELOCITY SHOULD SCALE DOWN AS $\lambda^{-1/2}$



HERCULES

1-1964

CONCLUSIONS

- IMPACTOR/TARGET FREQUENCY RATIO DETERMINES "TYPE" OF IMPACT EVENT (<u>NOT</u> IMPACT VELOCITY OR IMPACTOR/TARGET MASS RATIO)
- SIMPLE PLATE MODEL OF IMPACT ACCURATELY PREDICTS RESPONSE (STRAINS, CONTACT FORCE AND DEFLECTION)
- DIMENSIONAL ANALYSIS ACCURATELY PREDICTS RESPONSE SCALING
- DAMAGE SCALING IS MORE COMPLICATED BUT MAY BE POSSIBLE USING RESPONSE SCALING LAWS WITH REDUCED IMPACT VELOCITIES
- PRELIMINARY RESULTS INDICATE SIMILAR PLATE AND CYLINDER RESPONSE TO IMPACT

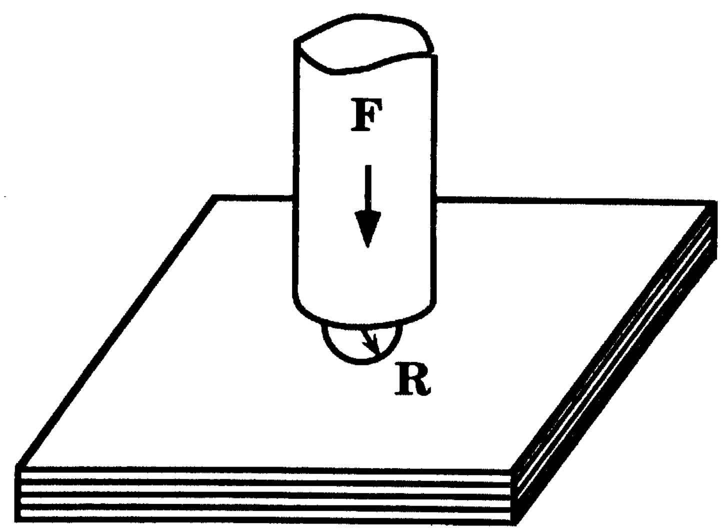
HERCULES

DELAMINATIONS IN COMPOSITE PLATES UNDER TRANSVERSE STATIC OR IMPACT LOADS

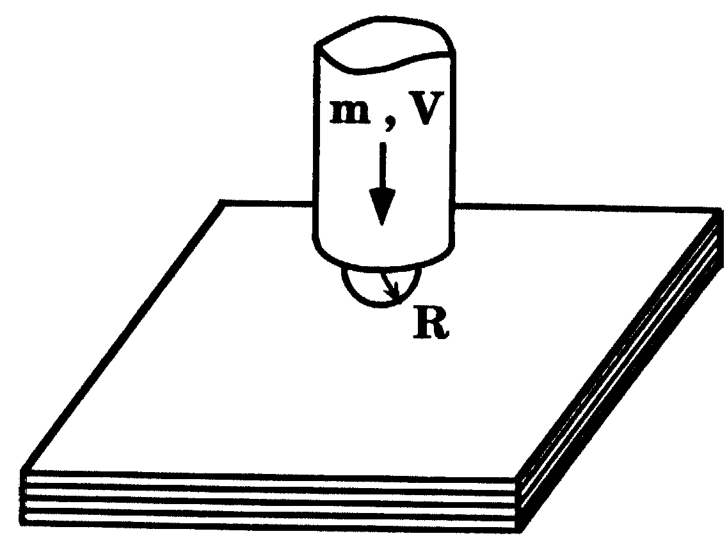
Scott R. Finn and George S. Springer

Department of Aeronautics and Astronautics
Stanford University
Stanford, California 94305

STATIC



DYNAMIC (IMPACT)



non-penetrating

edges clamped, simply supported, or free

OBJECTIVE

A MODEL TO PREDICT:

- **Damage Initiation Load**
- **Delamination**

  Locations
  Sizes
  Shapes

NEED:

- **Stress Analysis**
- **Damage Model**

  understanding of phenomena

  data

DAMAGE INITIATION

- **OBSERVATION: Delaminations Accompanied by Matrix Cracking**
- **POSTULATE: Matrix Cracking is a Precursor to Delamination**

Initiation Load when Matrix Cracking Occurs

a) Gosse, et al

$$\frac{1}{2}\left(\sigma_{yy} + \sigma_{zz}\right) + \sqrt{\frac{1}{4}\left(\sigma_{yy} - \sigma_{zz}\right)^2 + \sigma_{yz}^2} \geq Y$$

b) 3-D Tsai-Wu

c) 3-D Hashin

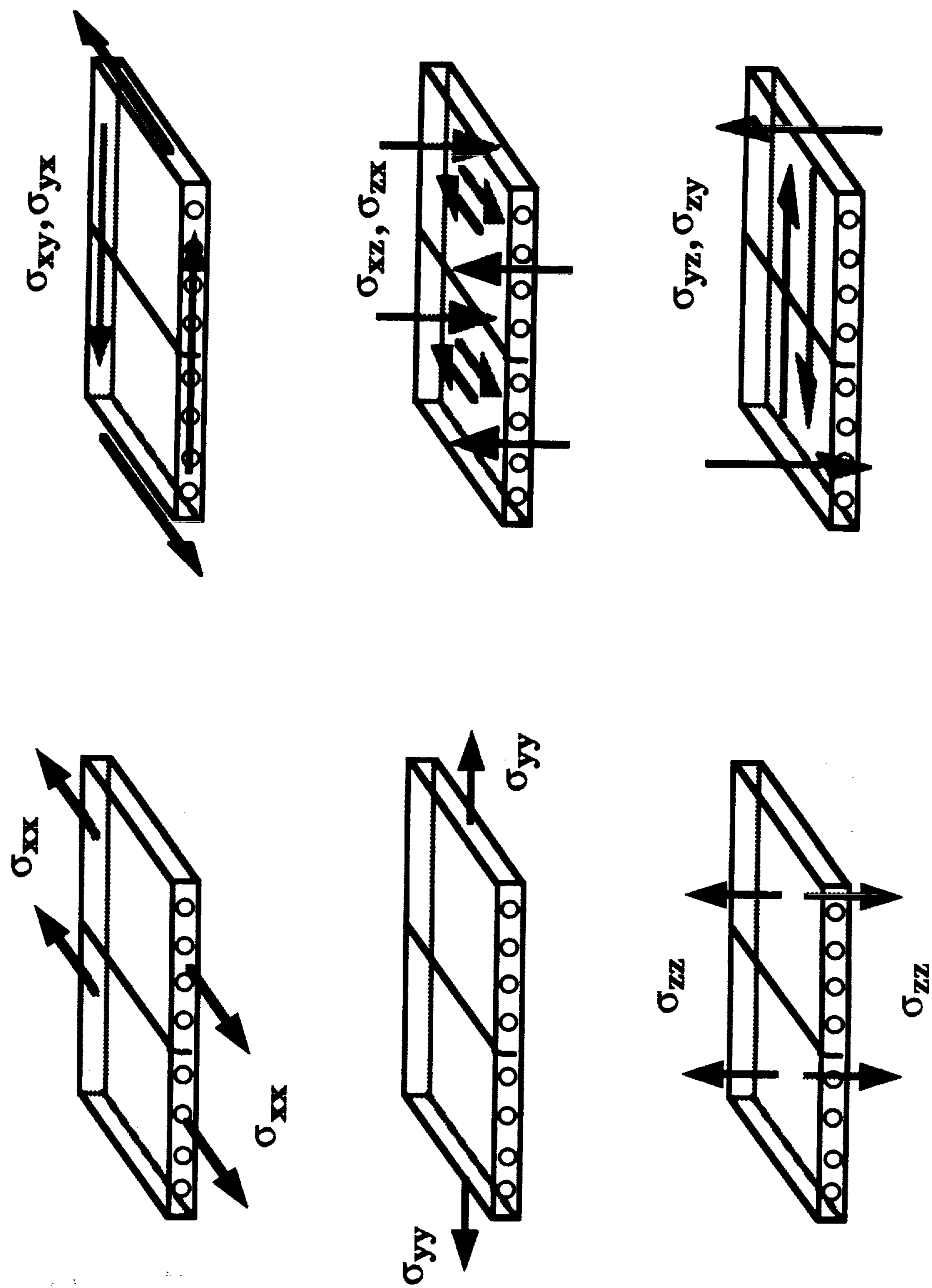
DELAMINATION

POSTULATES

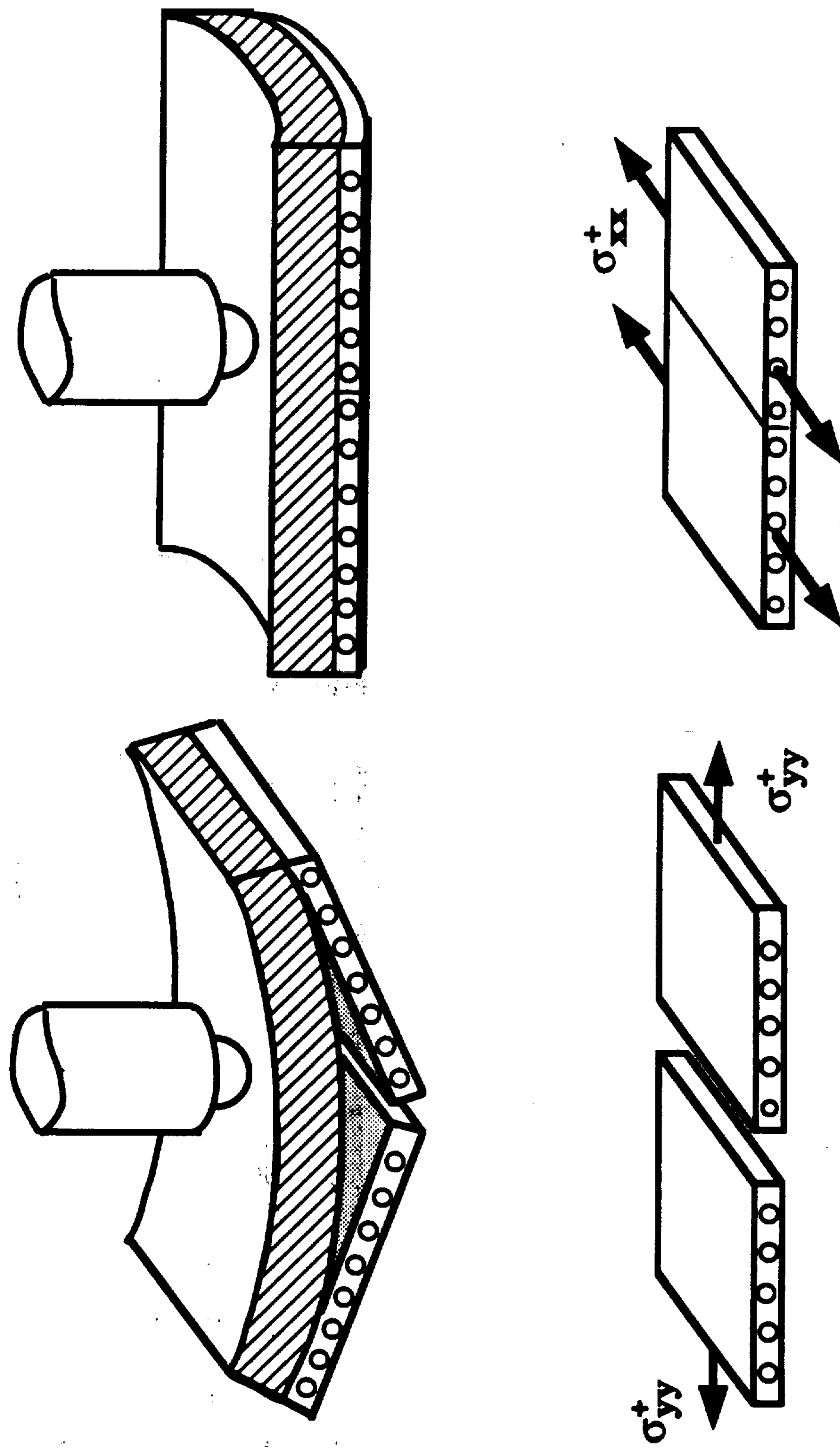
- **Matrix Cracking Must Be Present**
- **Crack Must Open**
- **Sufficient Strain Energy Available to Cause Delamination**

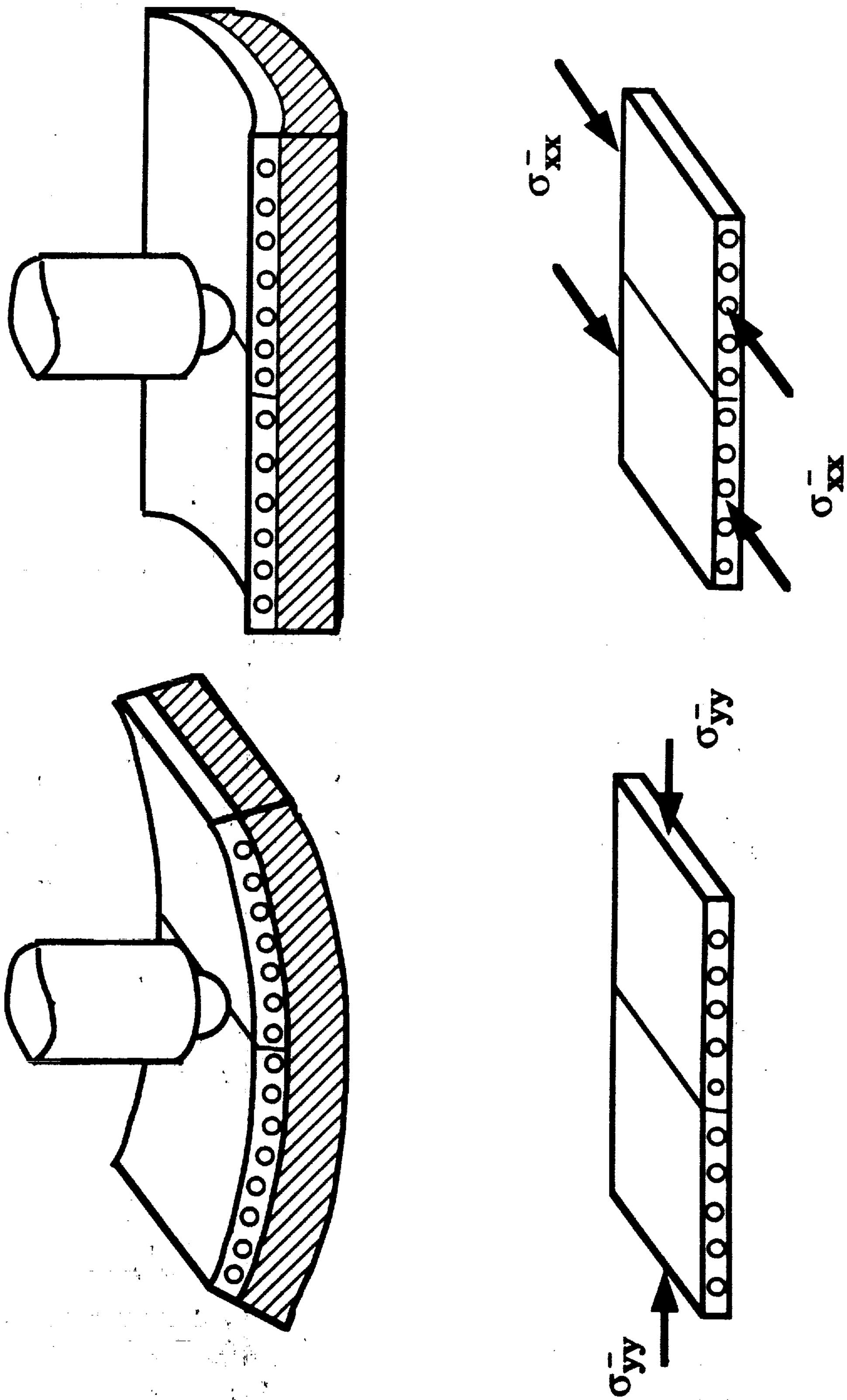
$$\hat{S} \geq \Gamma dA$$

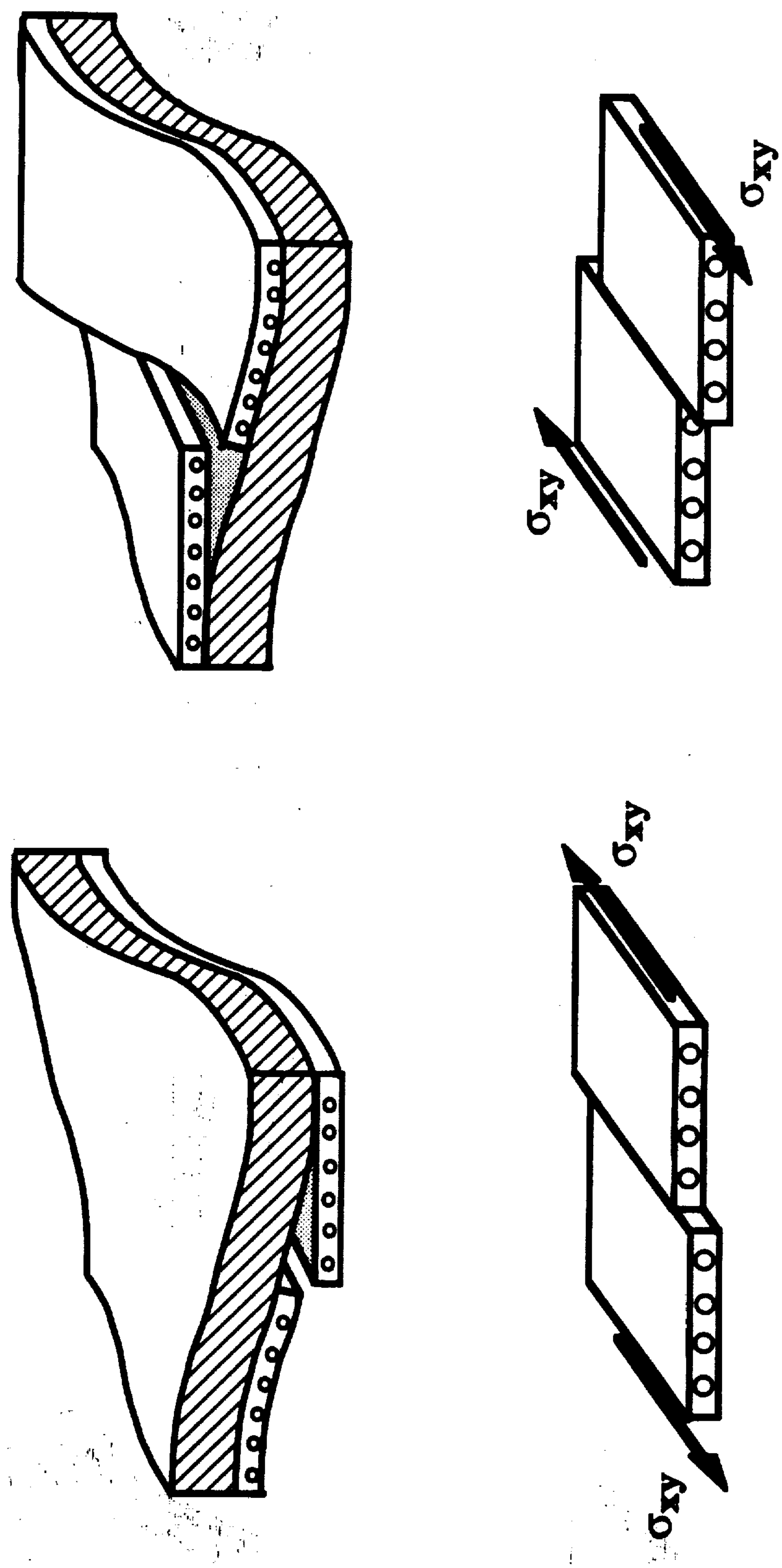
function of stresses



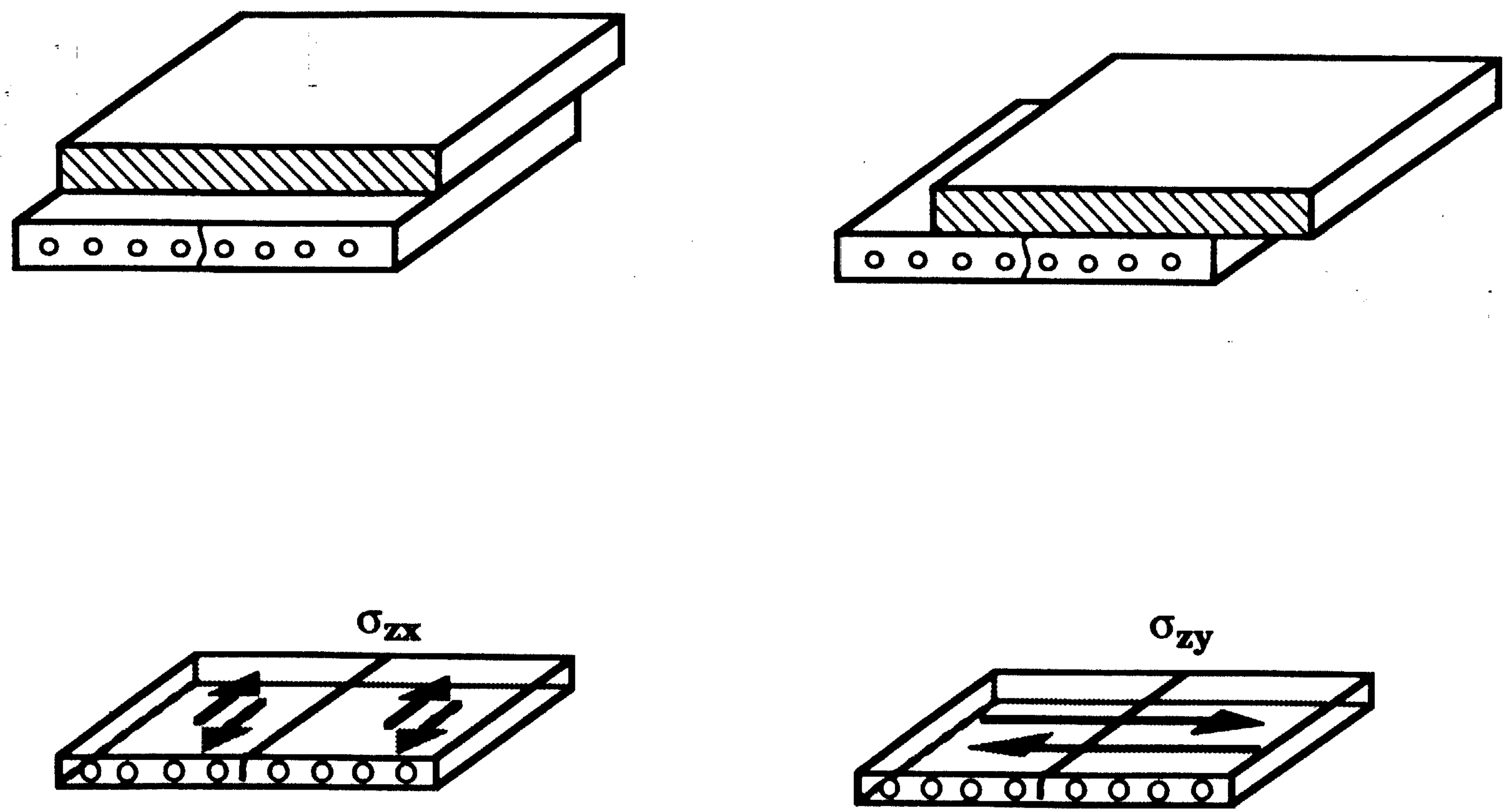
σ_{xy}, σ_{yx}
σ_{xz}, σ_{zx}
σ_{yz}, σ_{zy}
σ_{xx}
σ_{xx}
σ_{yy}
σ_{yy}
σ_{zz}
σ_{zz}


σ^+_{xx}
σ^+_{xx}
σ^+_{yy}
σ^+_{yy}

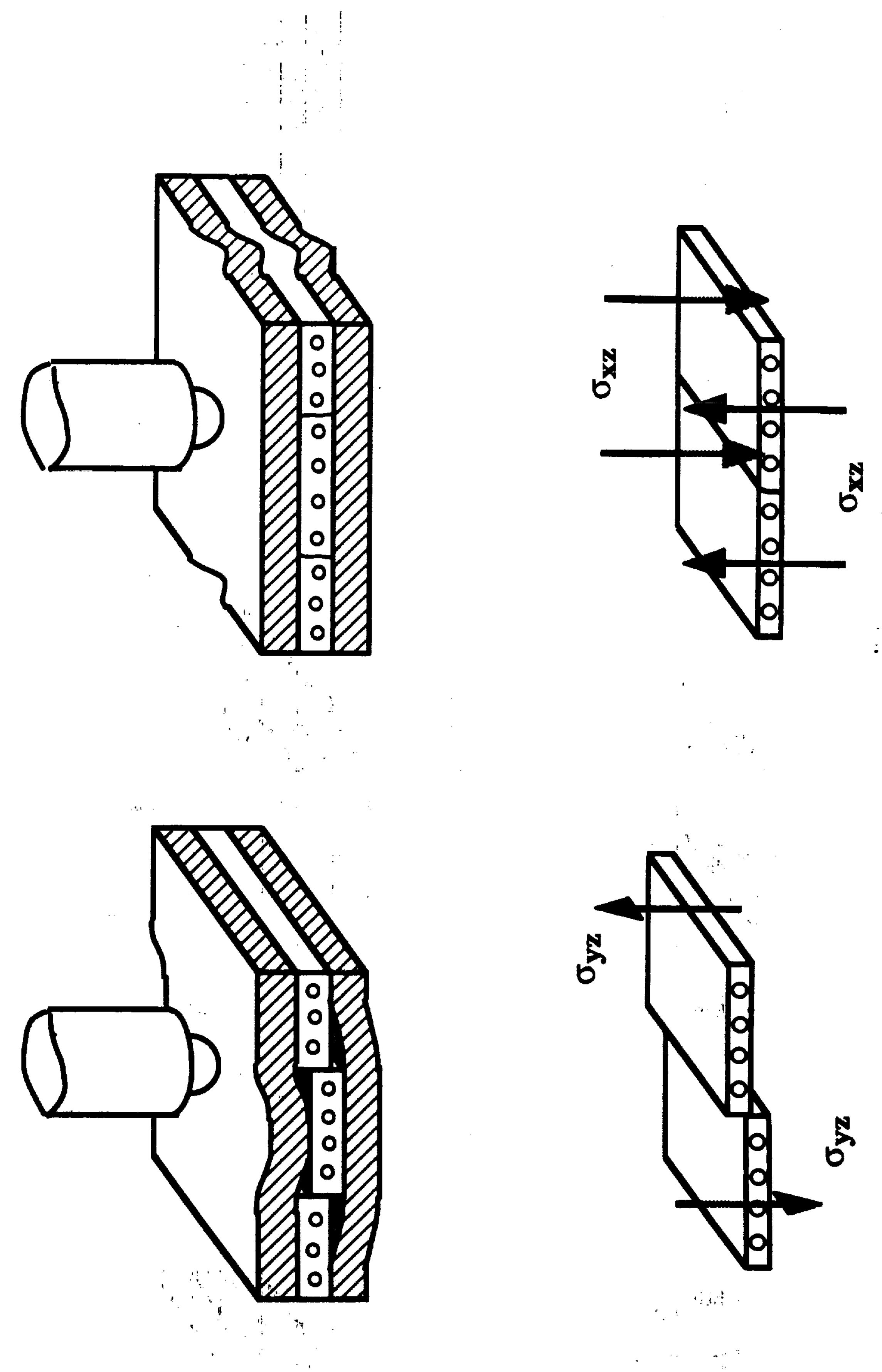

σ^-_{xx}
σ^-_{xx}
σ^-_{yy}
σ^-_{yy}



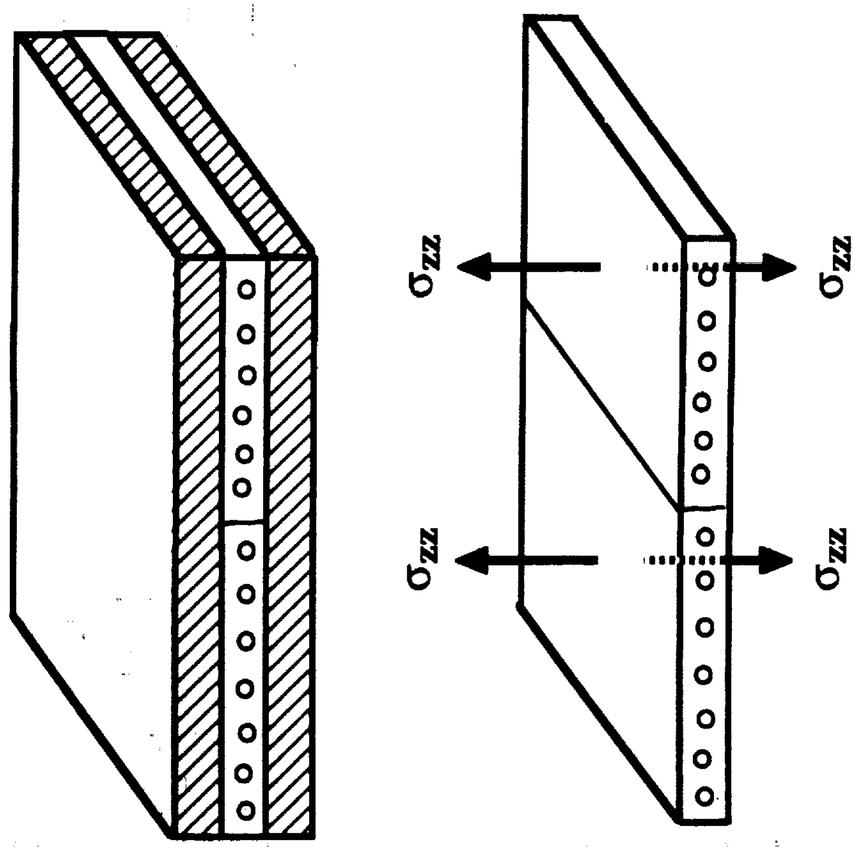
σ_{xy}
σ_{xy}
σ_{xy}
σ_{xy}

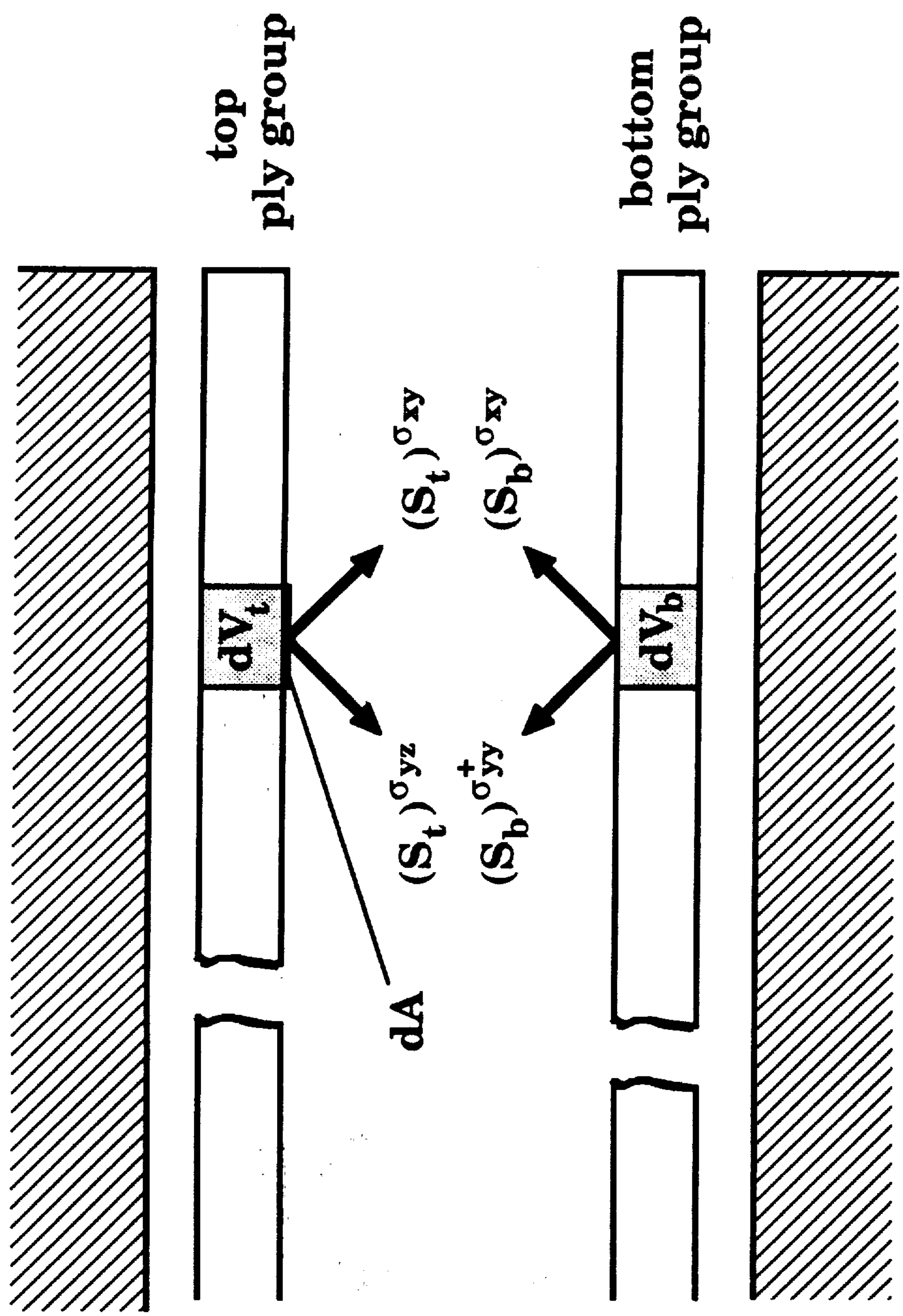


σ_{zx}
σ_{zy}



C-2


σ_{zz}
σ_{zz}
σ_{zz}
σ_{zz}



$$S_t\,dV_t + S_b\,dV_b \geq \Gamma\,dA$$

ANALYSIS - SOLUTION

- **STRESS ANALYSIS**

  Modified Wu-Springer Finite Element Method

  Each Edge: Free, Simply Supported, or Clamped

  Static or Dynamic Load

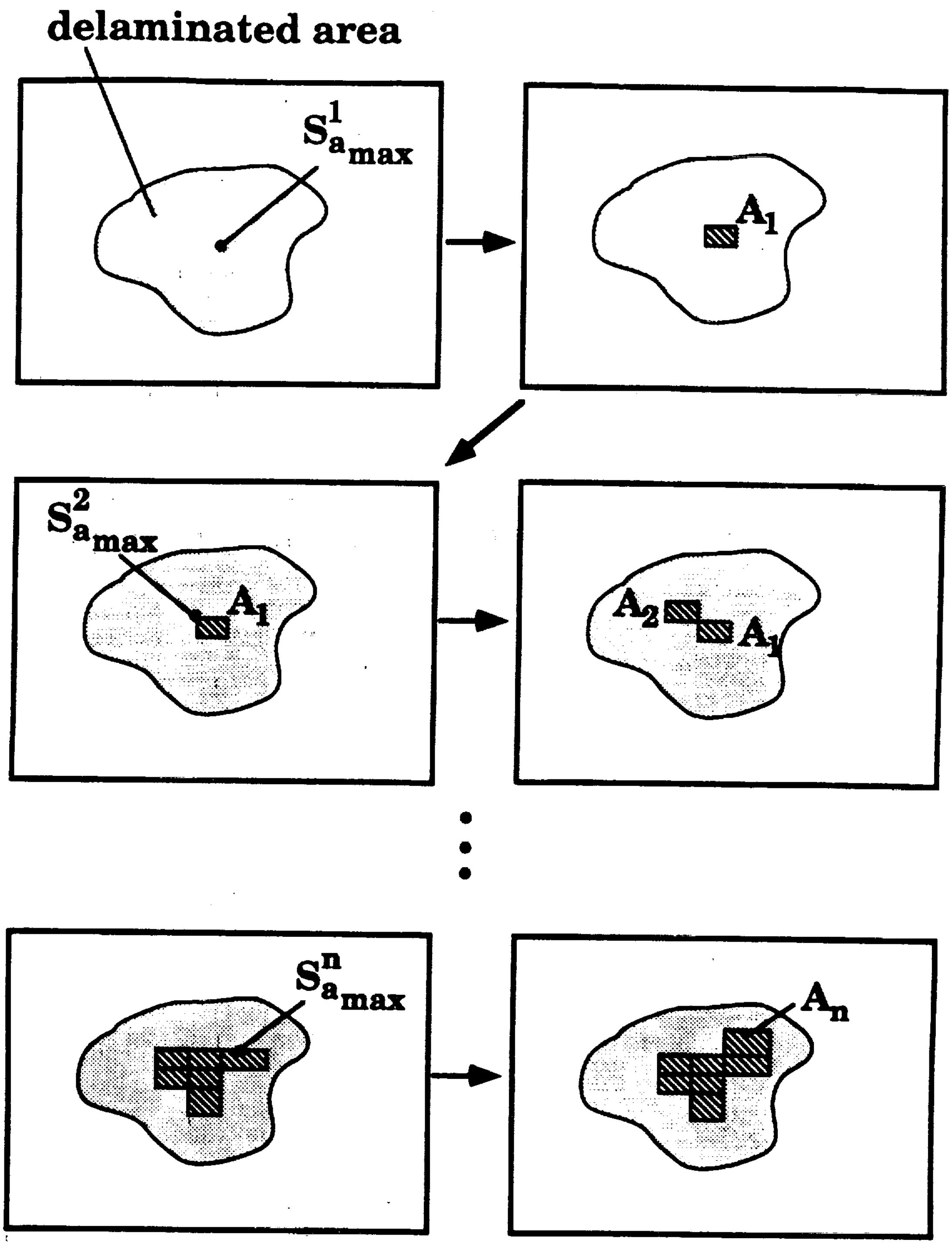
  Load Applied Over Finite Area

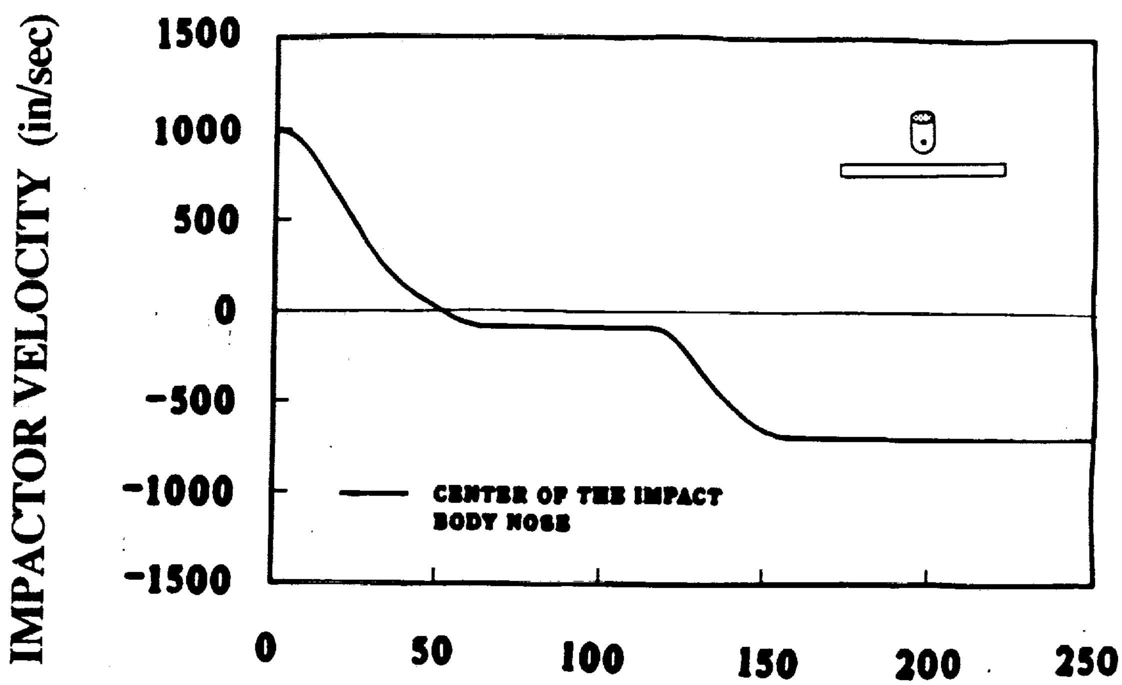
  Use Plate Symmetry for Computational Speed

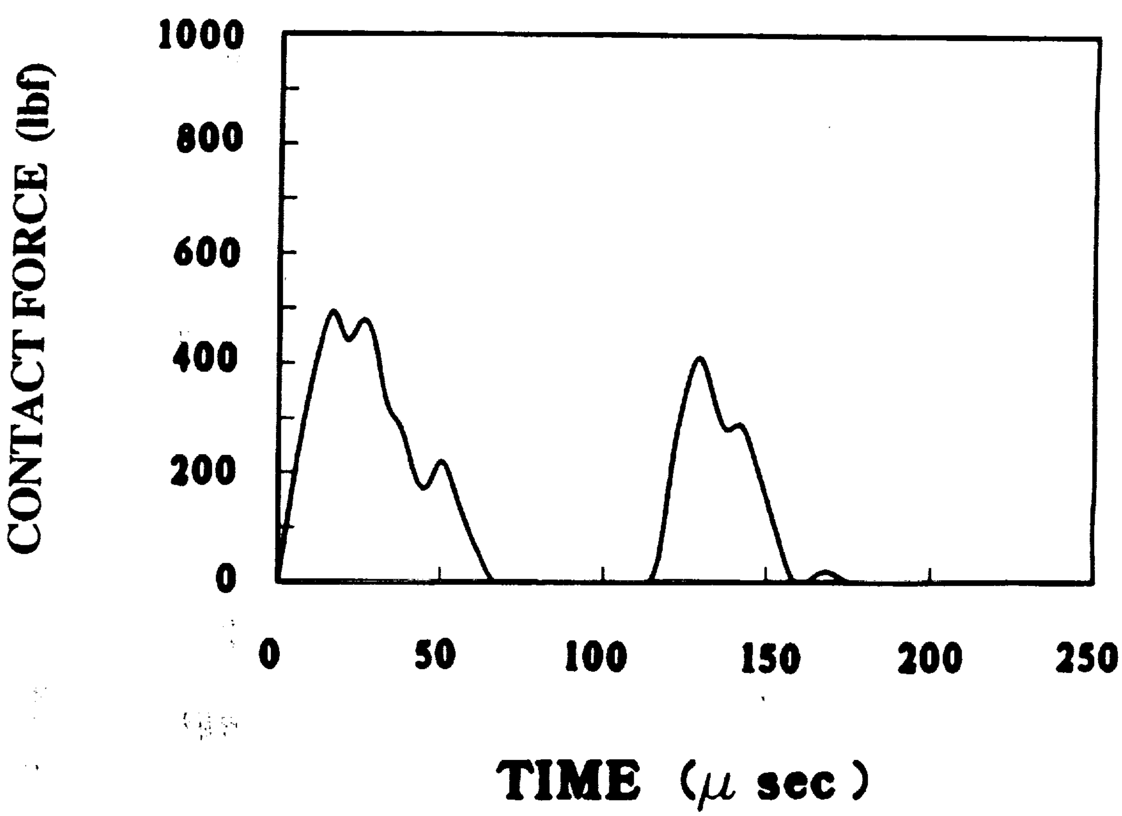
- **DELAMINATION**

  Step by Step

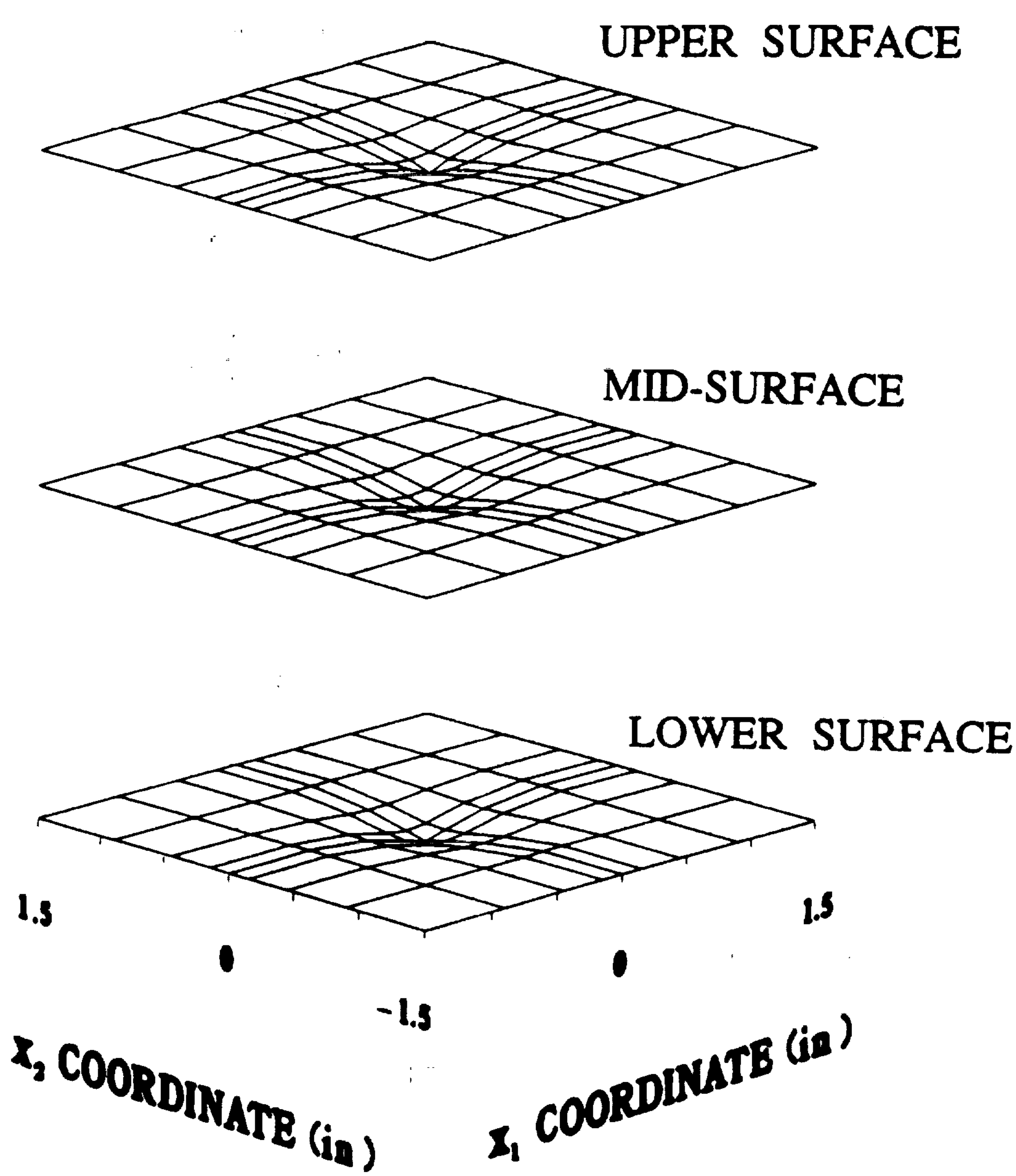
- **USER FRIENDLY COMPUTER CODES**


delaminated area
$S^1_{a_{max}}$
A_1
$S^2_{a_{max}}$
A_1
A_2
A_1
$S^n_{a_{max}}$
A_n

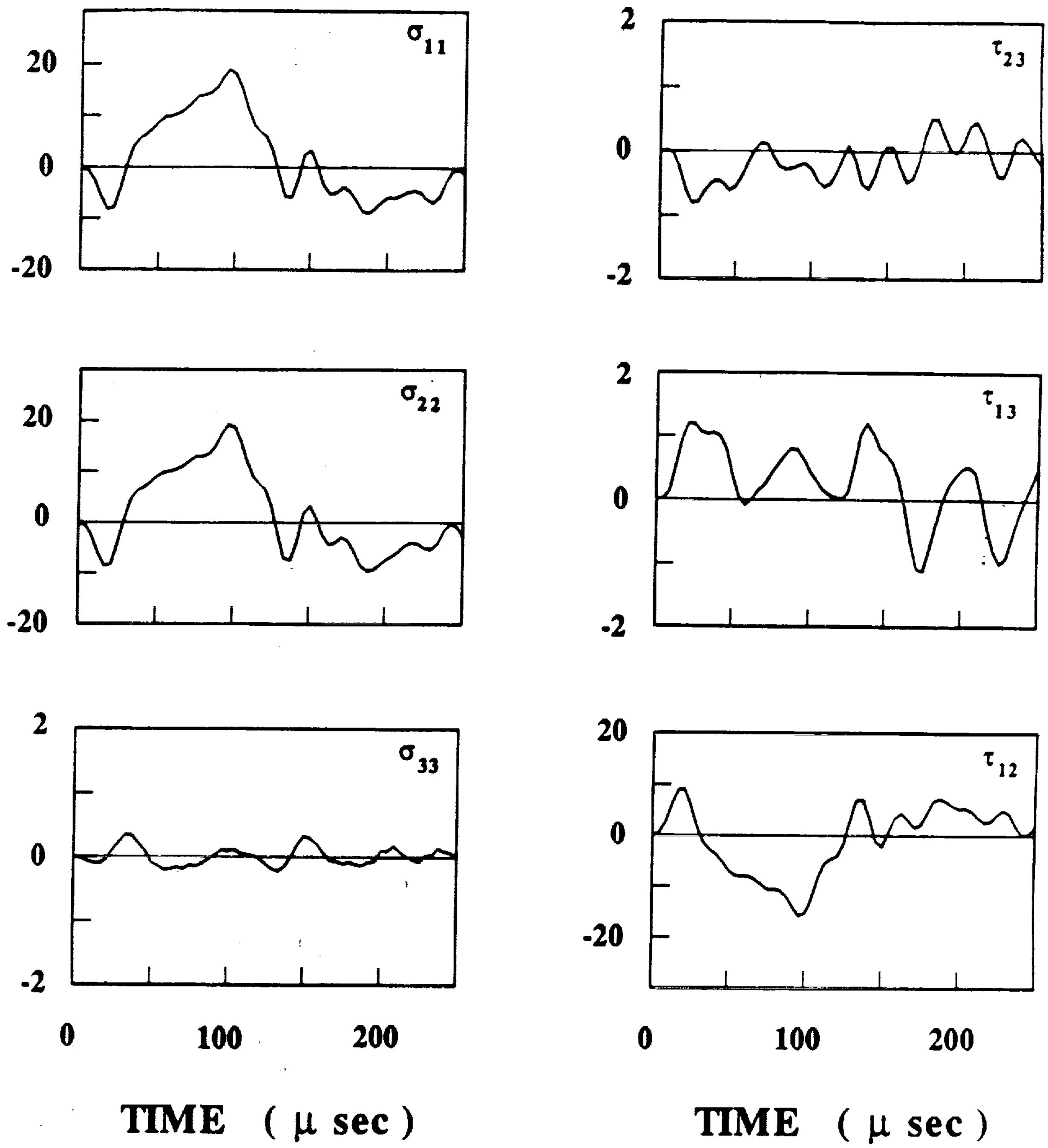

IMPACTOR VELOCITY (in/sec)
1500
1000
500
0
-500
-1000
-1500
CENTER OF THE IMPACT
BODY NOSE
0
50
100
150
200
250

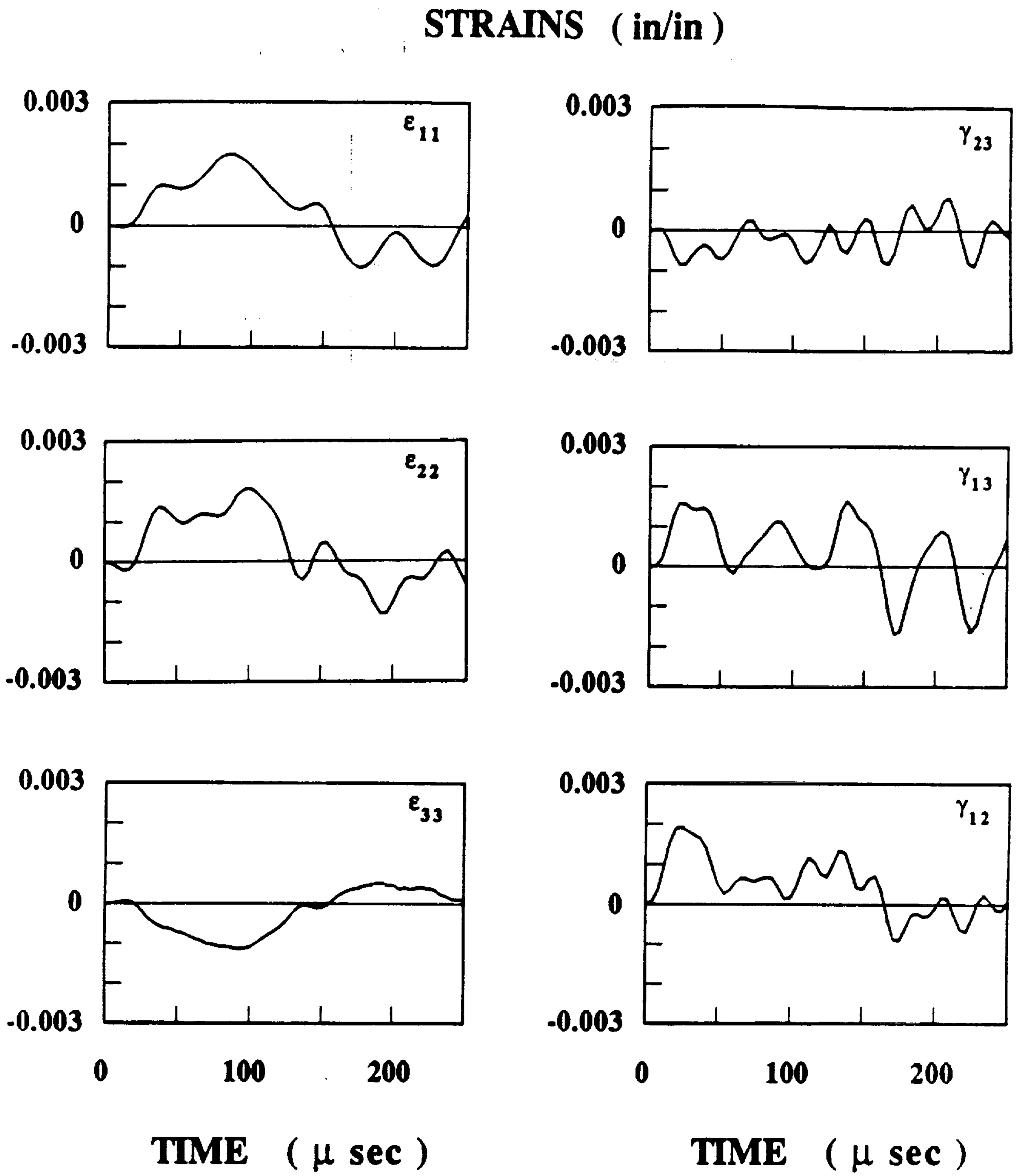

CONTACT FORCE (lbf)
1000
800
600
400
200
0
0
50
100
150
200
250
TIME (μ sec)

DISPLACEMENTS at TIME = 25 μ sec



STRESSES (ksi)




STRAINS (in/in)
0.003
0
-0.003
ε_{11}
γ_{23}
ε_{22}
γ_{13}
ε_{33}
γ_{12}
0
100
200
TIME (μ sec)
TIME (μ sec)

COMPUTER CODES

- STRESS ANALYSIS (IMPACT)
- DELAMINATION ANALYSIS (DELAM-TRL)
- COMBINED STRESS AND DELAMINATION ANALYSIS (IMPACT-ST)

USER FRIENDLY

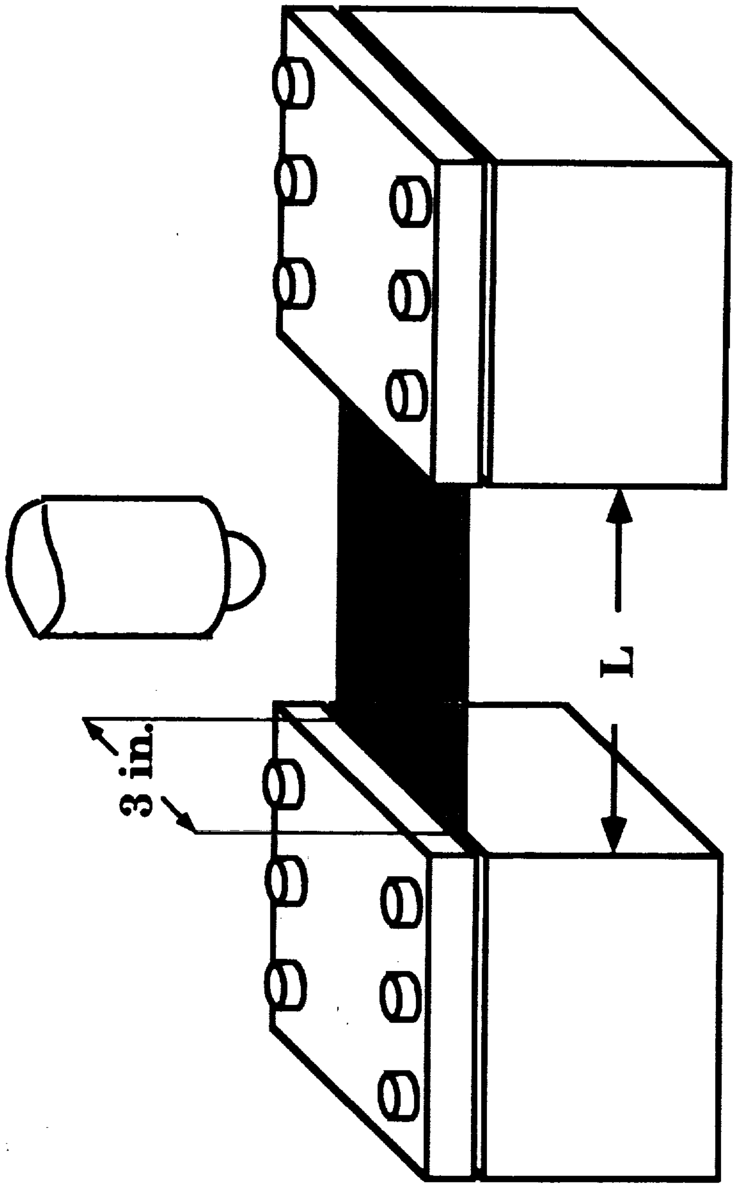
TESTS

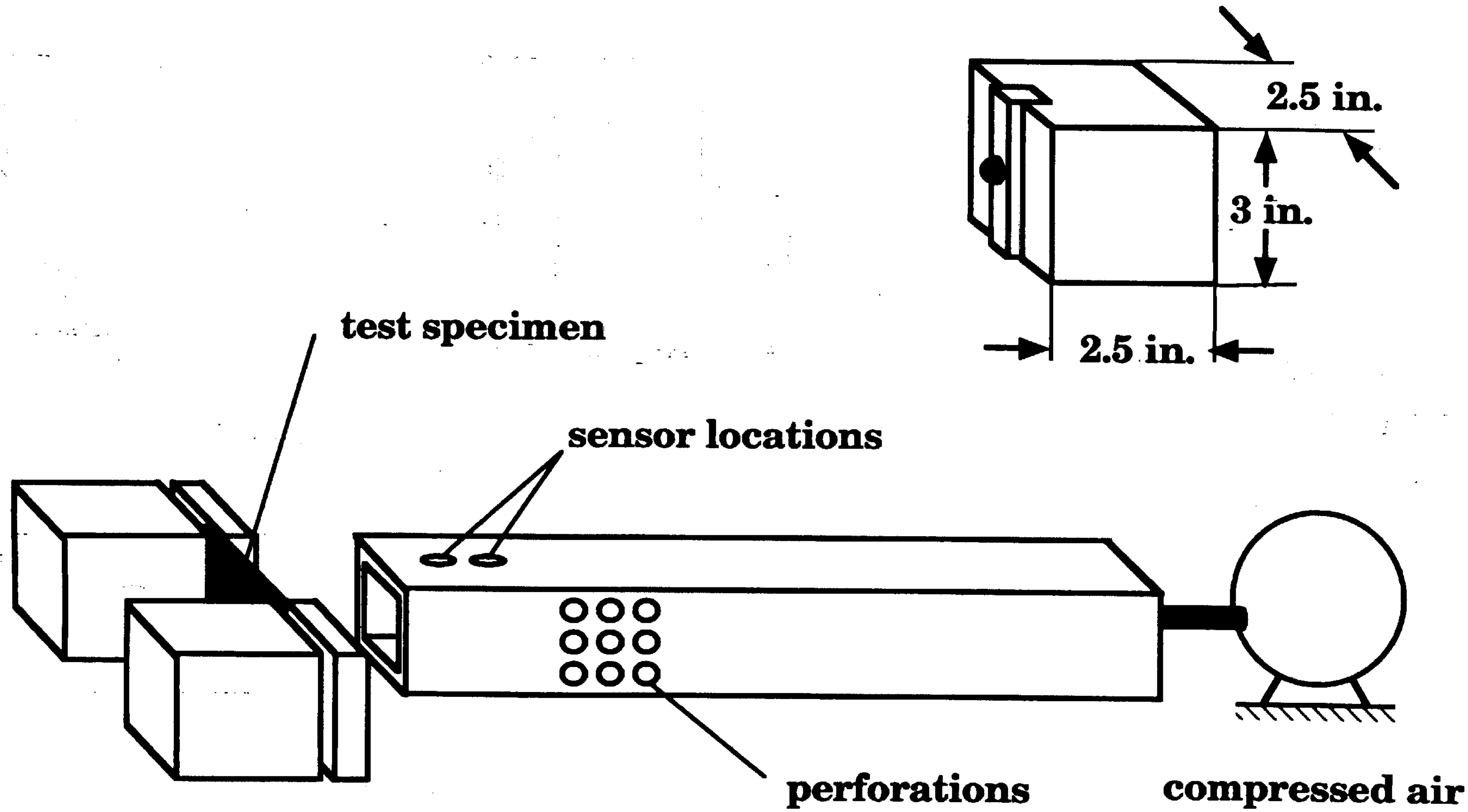
OBJECTIVE:

- **To Obtain Comprehensive, Systematic Data Set**
  - **To Gain Understanding of the Phenomena**
  - **Verify Present Model**
  - **Verify Future Models**

STATIC

DYNAMIC


3 in.
L


2.5 in.
3 in.
2.5 in.
test specimen
sensor locations
perforations
compressed air

TESTS

Materials (all ICI/Fiberite) :

T300/976	**(Thermoset)**
IM7/977-2	**(Toughened)**
APC-2	**(Thermoplastic)**

IMPACT	**STATIC**
V = 50-225 in/sec	**E = 0 - 65 lbf-in**
m = 0.355 - 0.963 lbm	
R = 0.125, 0.25 in.	**R = 0.25, 0.5, 1.0 in.**
L = 4 in.	**L = 3, 4, 5 in.**

DAMAGE INSPECTION

C-Scan

X-Ray

Microscope (sections)

MEASUREMENTS

- Damage Initiation Load
- Occurrence of Matrix Cracking
- Delamination

  Shapes
  Sizes
  Locations

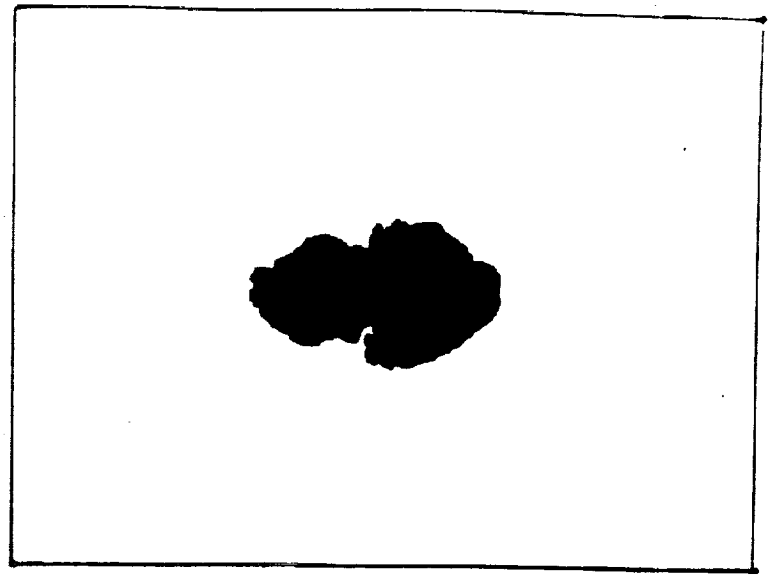


Illustration of a typical C-scan result

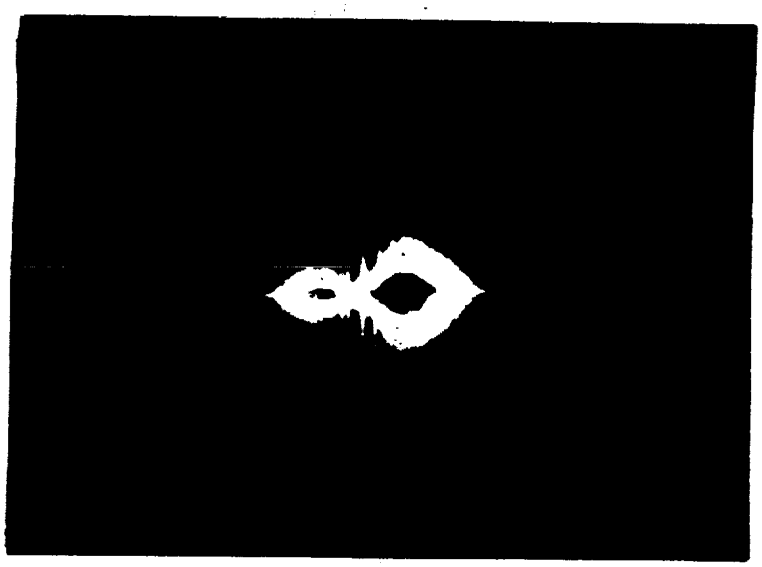
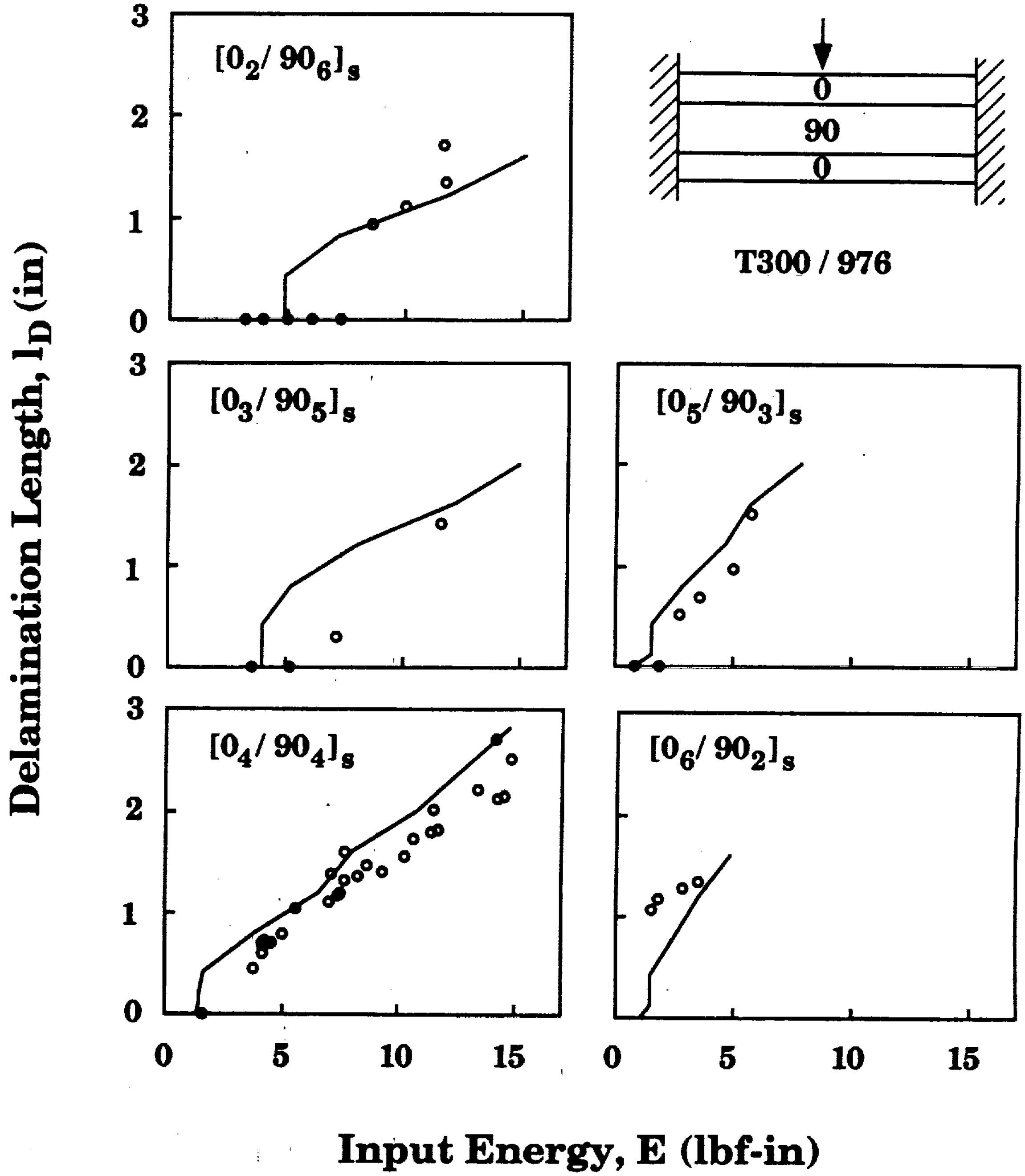


Illustration of a typical x-ray result

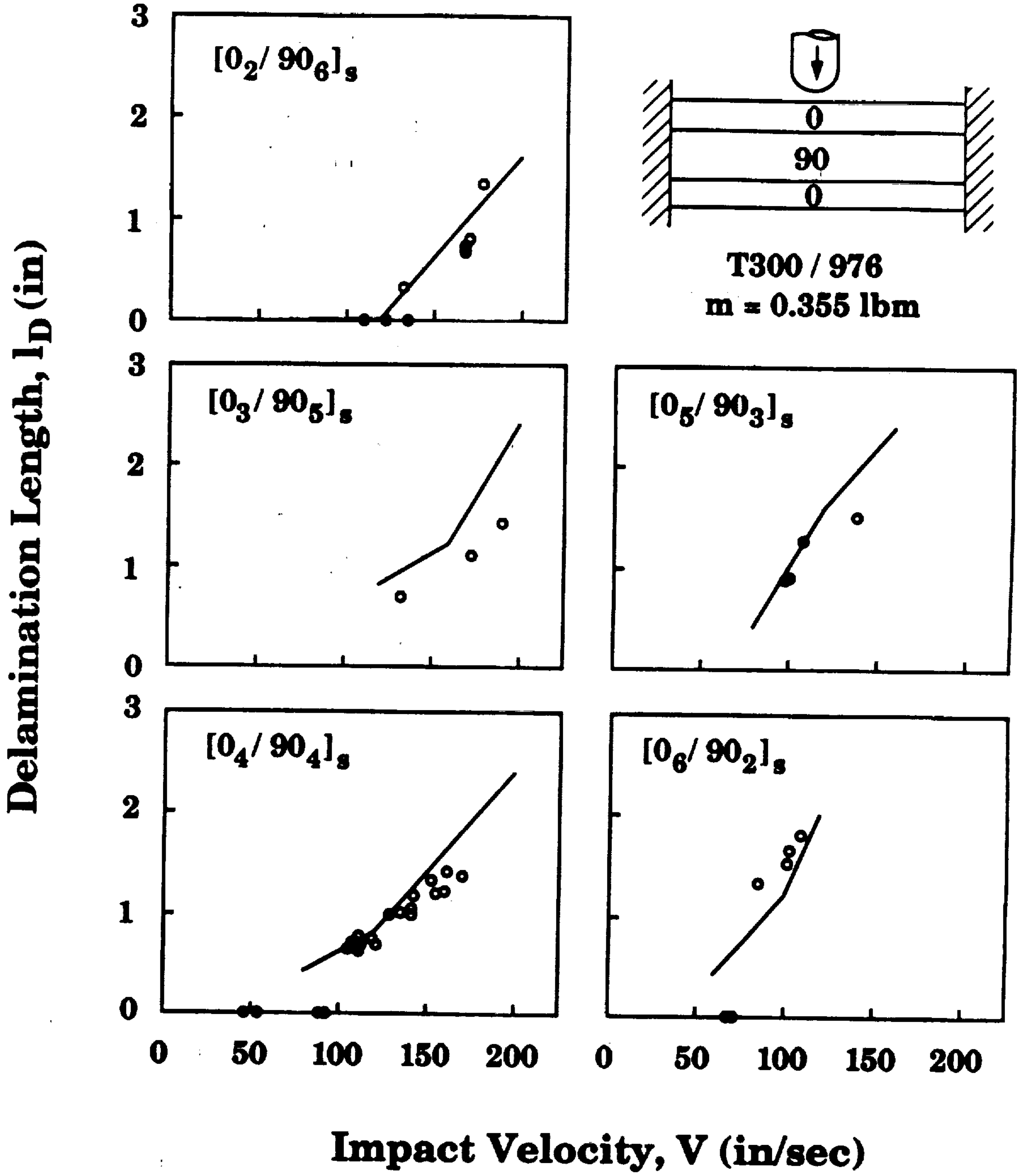
EFFECTS INVESTIGATED

- Thickness of Bottom Ply Group
- Mismatch Angle
- Plate Thickness
- Nose Radius
- Plate Length

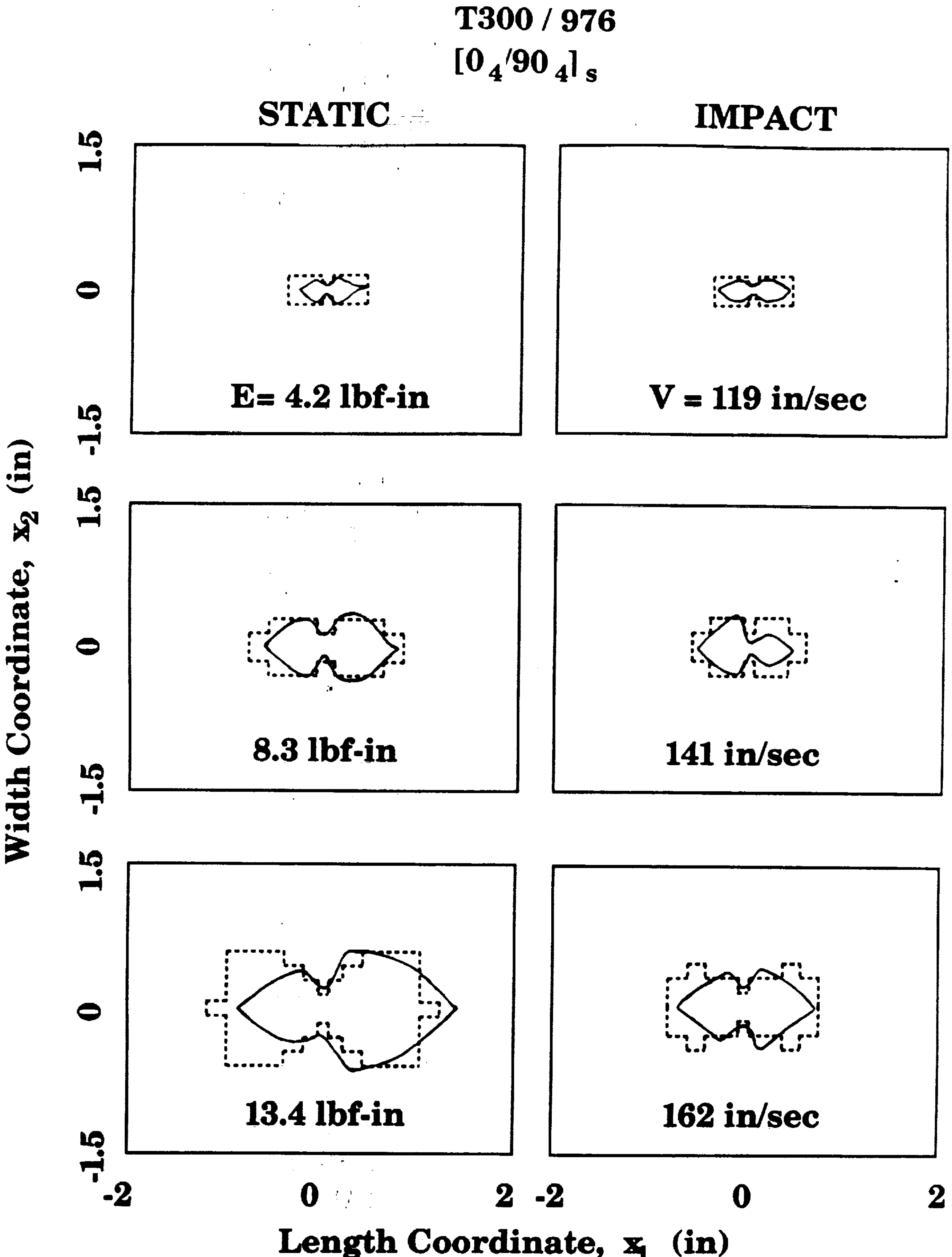
STATIC


[0$_2$/ 90$_6$]$_s$
[0$_3$/ 90$_5$]$_s$
[0$_5$/ 90$_3$]$_s$
[0$_4$/ 90$_4$]$_s$
[0$_6$/ 90$_2$]$_s$
0
90
0
T300 / 976
Delamination Length, l_D (in)
Input Energy, E (lbf-in)
0
1
2
3
0
5
10
15

IMPACT



[0$_2$/ 90$_6$]$_s$
[0$_3$/ 90$_5$]$_s$
[0$_5$/ 90$_3$]$_s$
[0$_4$/ 90$_4$]$_s$
[0$_6$/ 90$_2$]$_s$
0
90
0
T300 / 976
m = 0.355 lbm
Delamination Length, l_D (in)
Impact Velocity, V (in/sec)
0
1
2
3
0
50
100
150
200


T300 / 976
[0 4/90 4] s
STATIC
IMPACT
E= 4.2 lbf-in
V = 119 in/sec
8.3 lbf-in
141 in/sec
13.4 lbf-in
162 in/sec
Width Coordinate, x2 (in)
1.5
0
-1.5
-2
2
Length Coordinate, x1 (in)

SUMMARY

- Tests

  Insight

  Large Data Set

- Damage Initiation Model
- Delamination Model
- User Friendly Codes

  Engineering Design Tool

- Verification

IMPACT DAMAGED COMPOSITES, PART I: DAMAGE SIMULATION AND STRENGTH PREDICTIONS

Larry B. Ilcewicz

and

Ernest F. Dost

Boeing Commercial Airplane Group

Impact Damaged Composites, Part 1: Damage Simulation and Strength Predictions

- Transport Fuselage Structures → *L. Ilcewicz*
- A Need to Understand Fundamentals → *L. Ilcewicz*
- Types of Impact Damage States → *E. Dost*
- Damage Simulation & CAI Strength Prediction → *E. Dost*
- Part 1 Summary → *E. Dost*

Impact Damaged Composites, Part 2: Standard Tests for Fuselage Structural Issues

- Fuselage Impact Damage Characterization → *E. Dost*
- Standard Specimen Tests for Fuselage Applications → *W. Avery*
- Part 2 Summary → *W. Avery*

Impact Damaged Composites, Part 1: Damage Simulation and Strength Predictions

L. Ilcewicz, 3/19/91

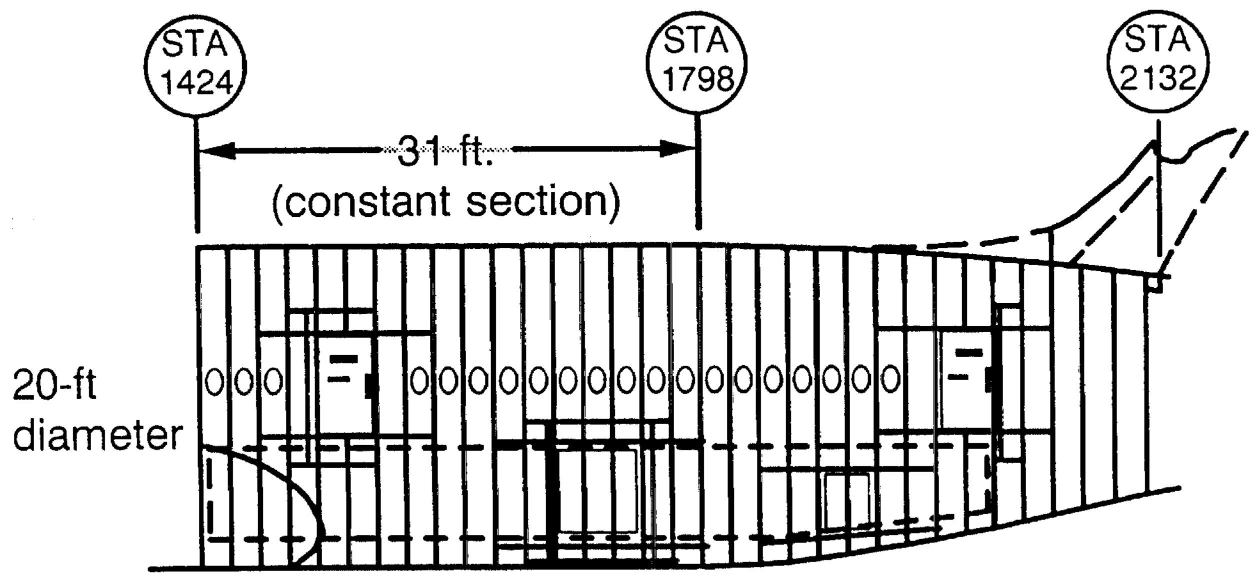
Transport Fuselage Structures

- Loads and Technical Issues
- Problem Definition

A Need to Understand Fundamentals

- Damage Resistance versus Damage Tolerance
- Silver Bullet Syndrome

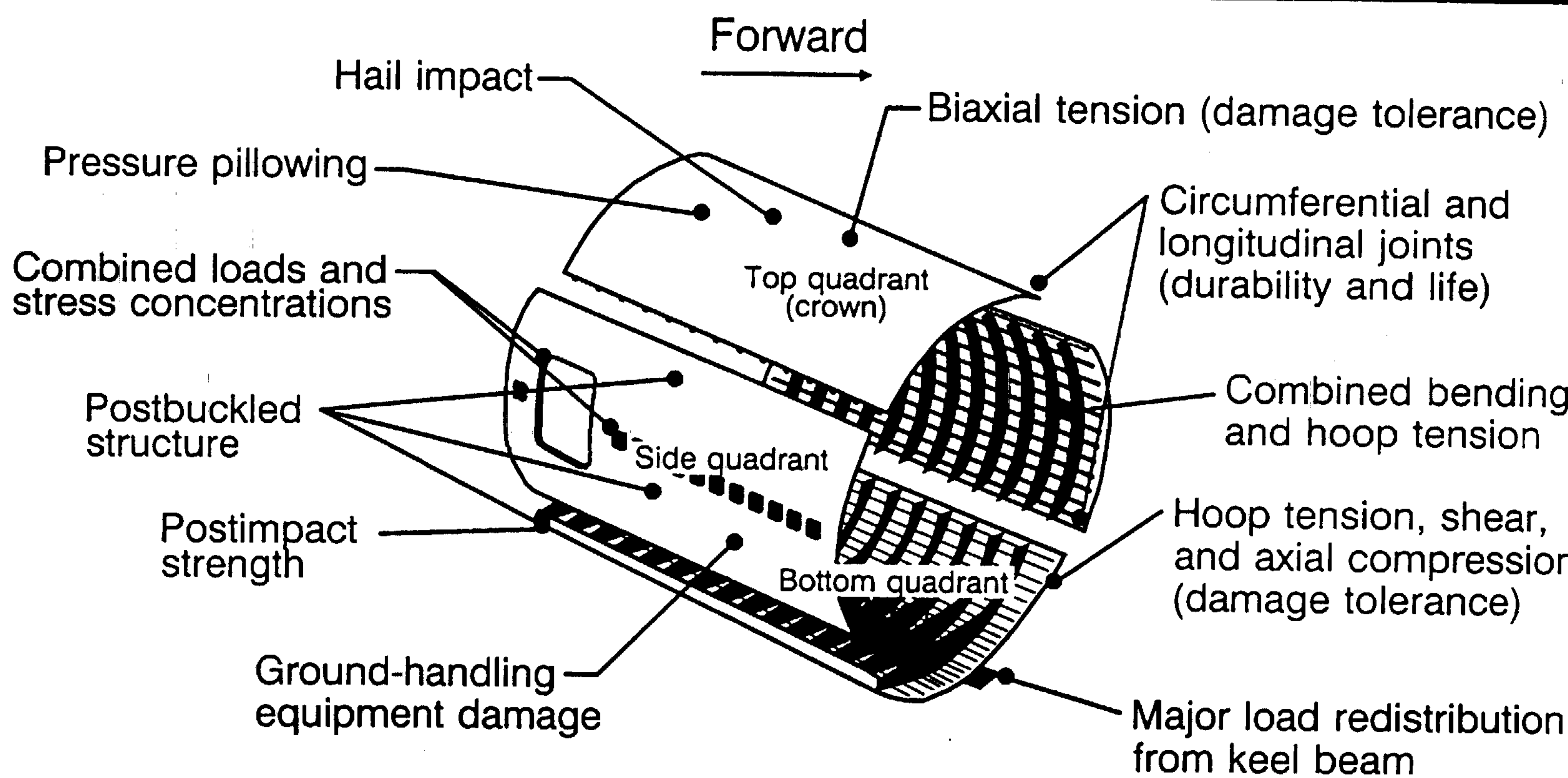
Fuselage Section Used for ATCAS Studies



767-X - Sec.46

- Use actual 767-X loads

Design Drivers for Composite Fuselage



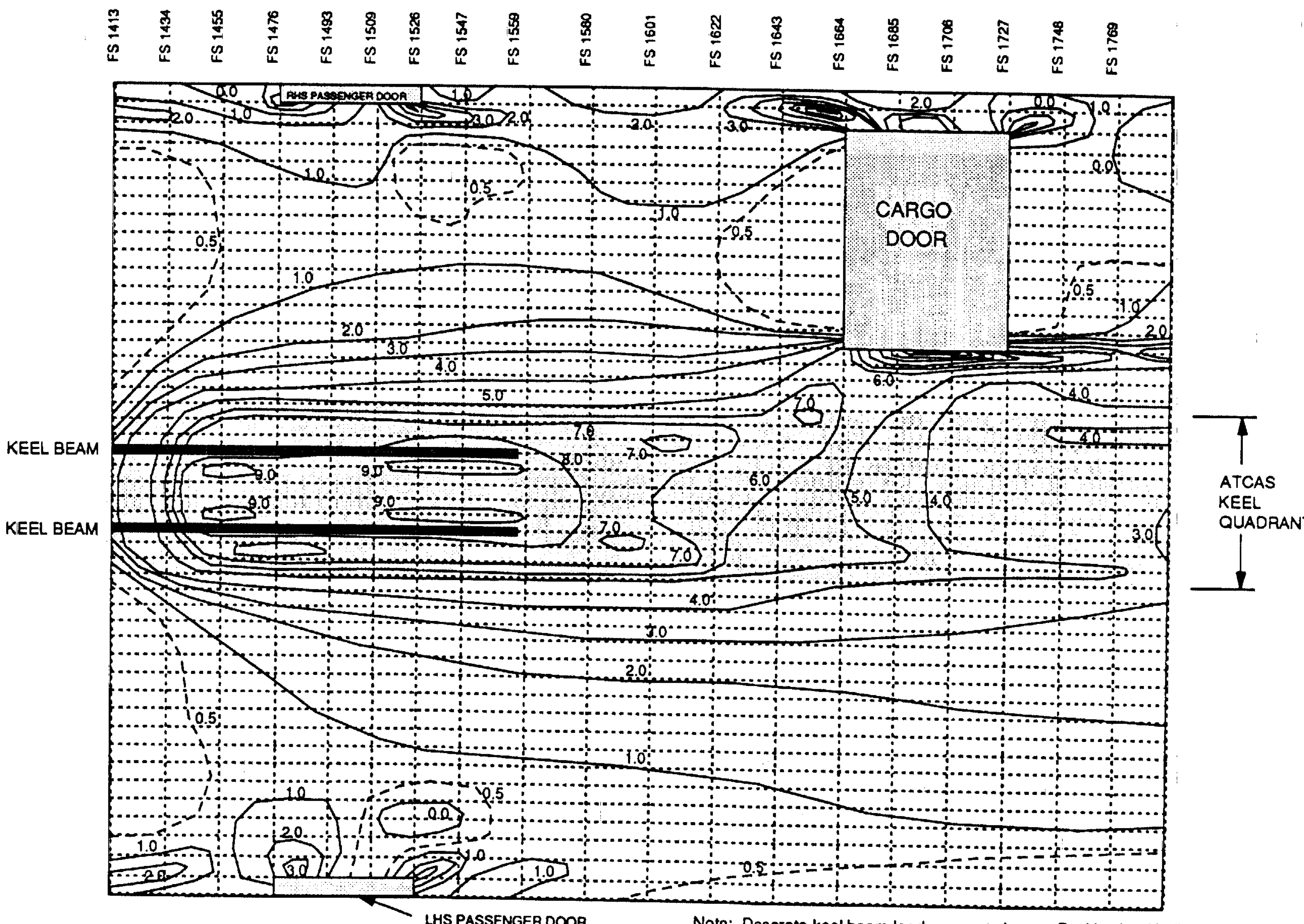
Program addresses critical structural issues.

NASA/BOEING ATCAS

AL1823.04 DR1D

Critical Compression Load Condition in Keel and Lower Side Quadrants Aft of Wheel Well

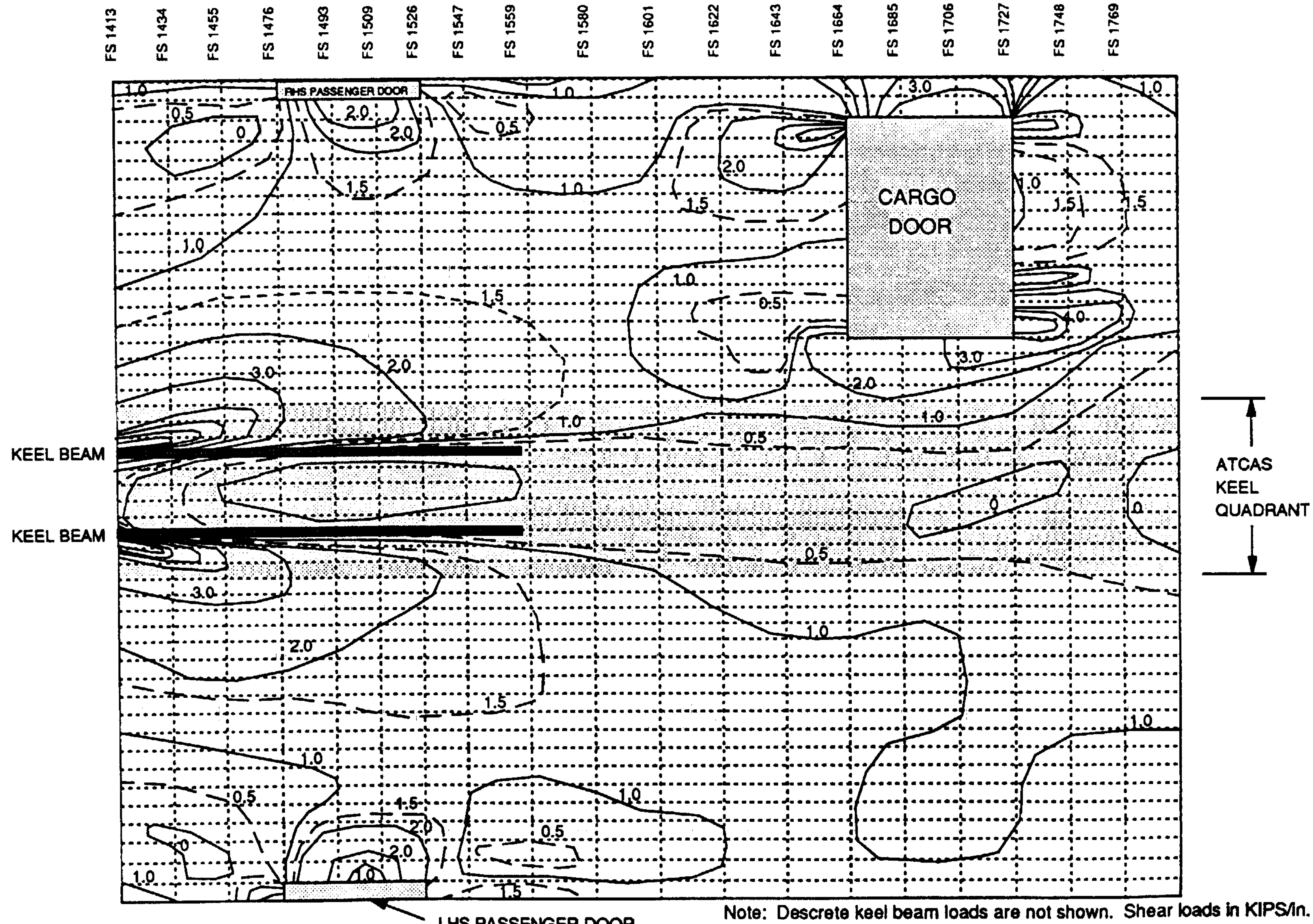
(Note Load Redistribution Near Keel Beams and Major Cutouts)



Note: Descrete keel beam loads are not shown. Positive load indicates COMPRESSION (KIPS/IN). All loads are for 2.5 g ULTIMATE load condition.

Critical Shear Load Condition in Keel and Lower Side Quadrants Aft of Wheel Well

(Note Load Redistribution Near Keel Beams and Major Cutouts)

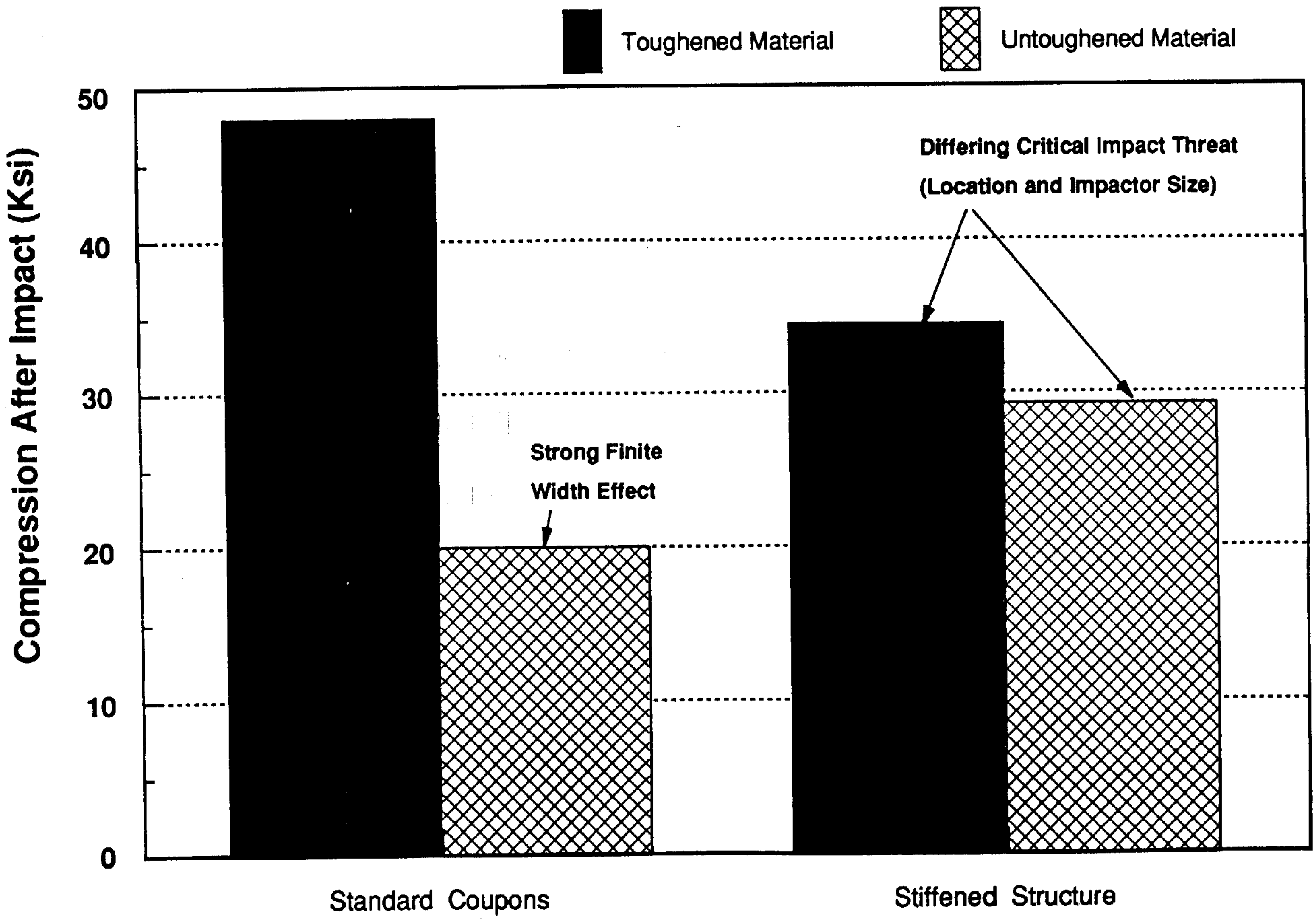


Note: Descrete keel beam loads are not shown. Shear loads in KIPS/in.
All loads shown are for 2.5 g ULTIMATE load condition.

Problem Definition

- *Fundamental Understanding of Material Failure Mechanisms*
  - Damage = f(Impact Variables)
  - Combined Loads
- *"Engineering" Analysis Methods Verified By Tests*
  - Damage Simulation
  - Failure Criteria
- *Identify Critical Impact Threats for Fuselage Structure*
  - Critical Damage = f(Fuselage Location)
  - Design Criteria (ULTIMATE, LIMIT, SAFE FLIGHT)
- *Methods to Relate Coupon Screening Tests to Structural Performance*
  - Material Selection

Material Models and Damage Characterization Tests are Needed to Understand the Critical Impact Threat for New Material Forms



Issues that Should be Addressed to Make Material Selections that are Both Economically & Structurally Sound

Additional Tests and Analysis Needed to Evaluate the Relationship Between Coupon and Structural Performance

Effect of Criteria Must Be Understood, e.g.,

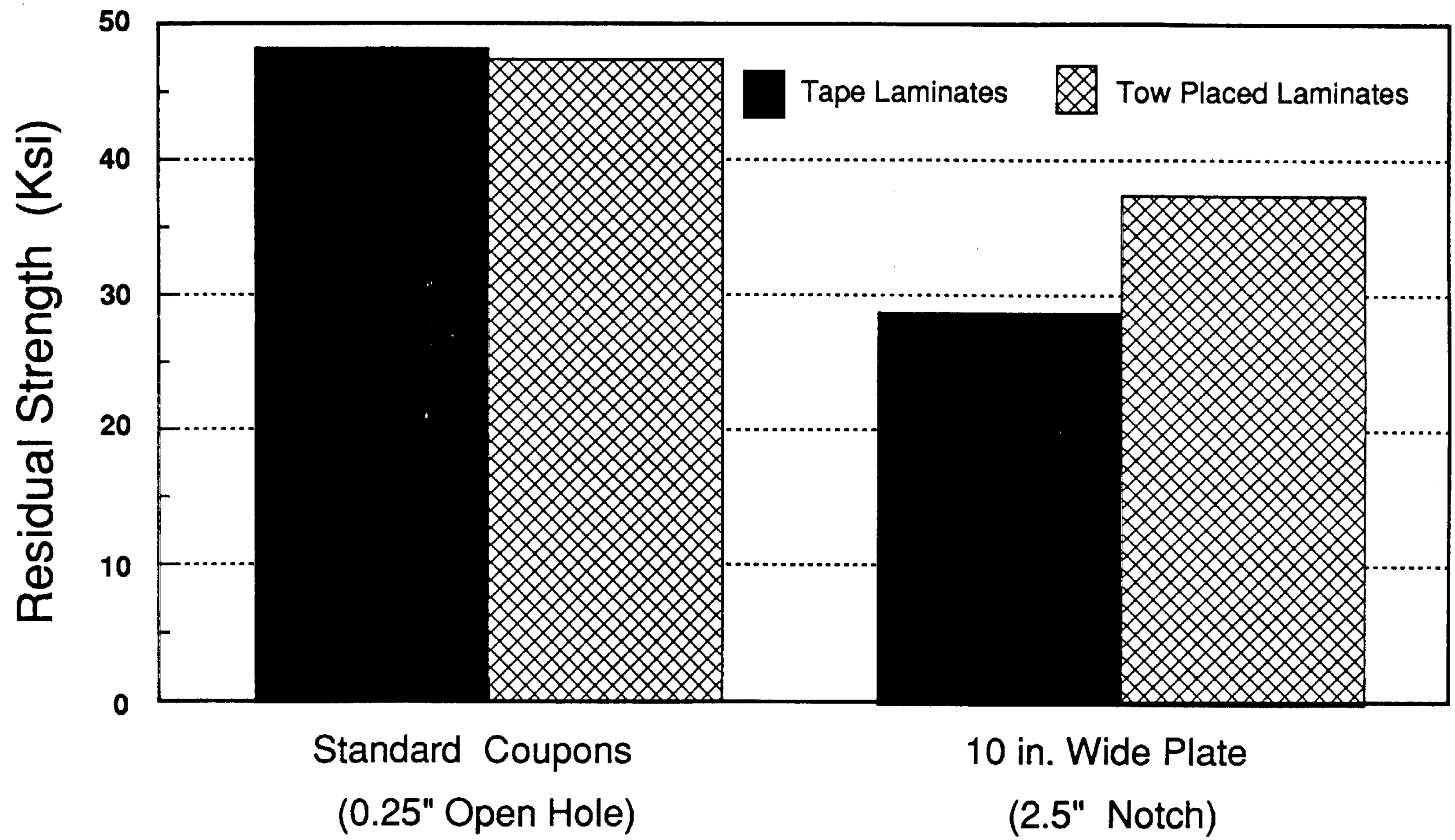
- **Impact Level**
- **Impact Location**
- **Damage Type and Size**

Effect of Scale Must Be Understood, e.g.,

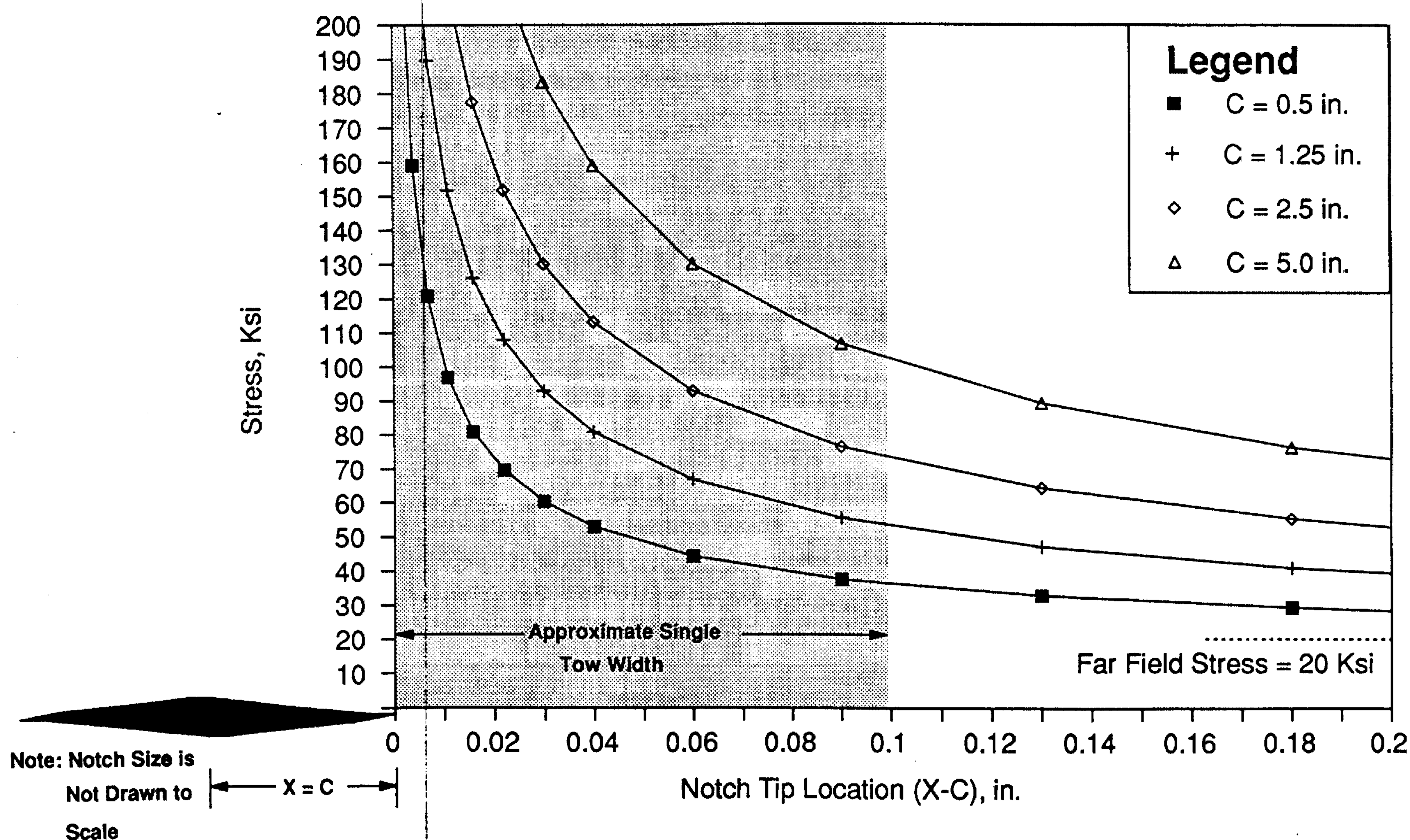
- **Impact Boundary Condition**
- **Finite Width Effects**
- **Changing Failure Mechanisms**
- **Interactions With Material Dimensional Scales**

Relationship Between Material Screening Test Results and Structural Performance

Notched Tensile Strength

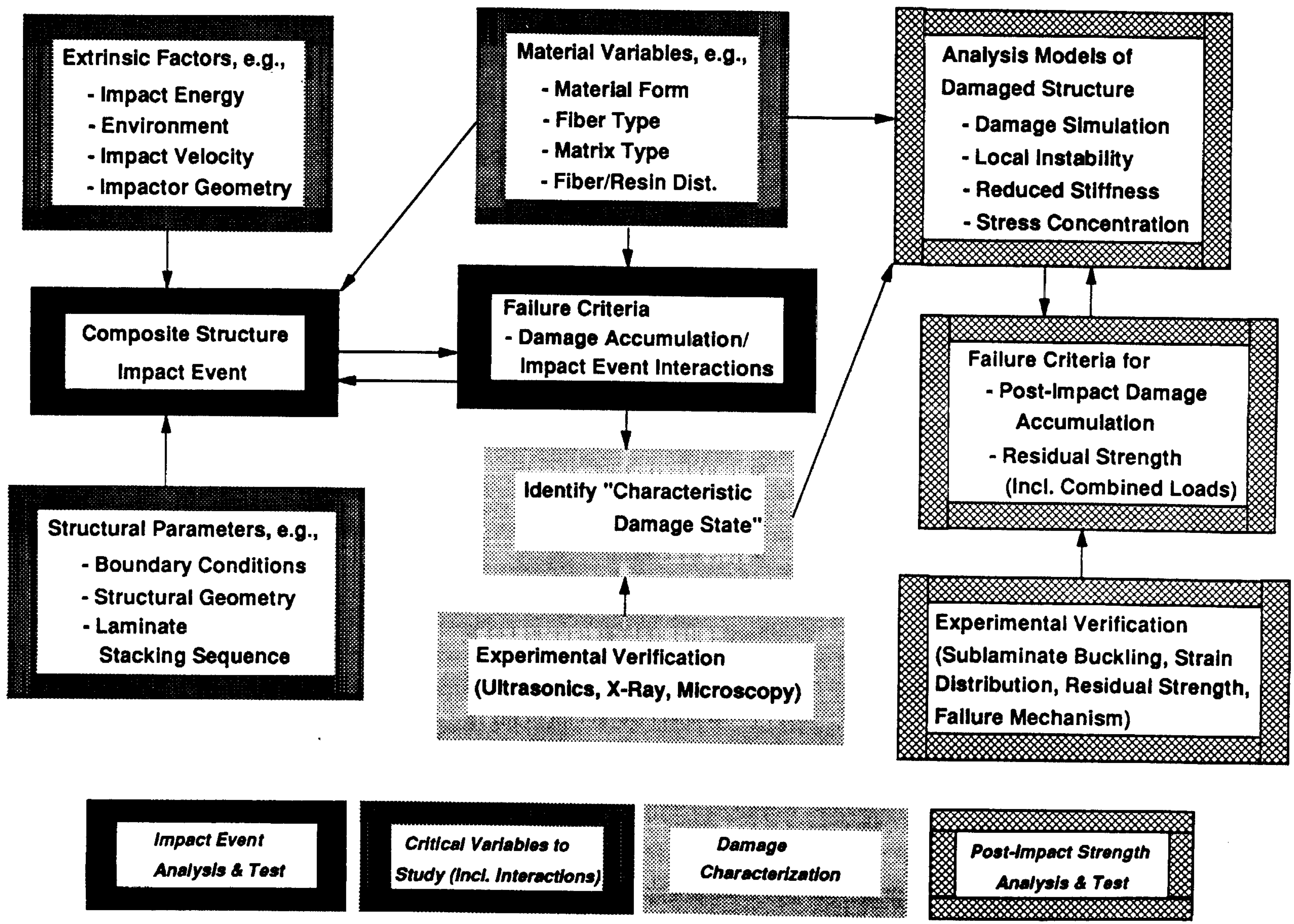


Notched Plate Stress Distributions and Tow Material Dimensional Scale

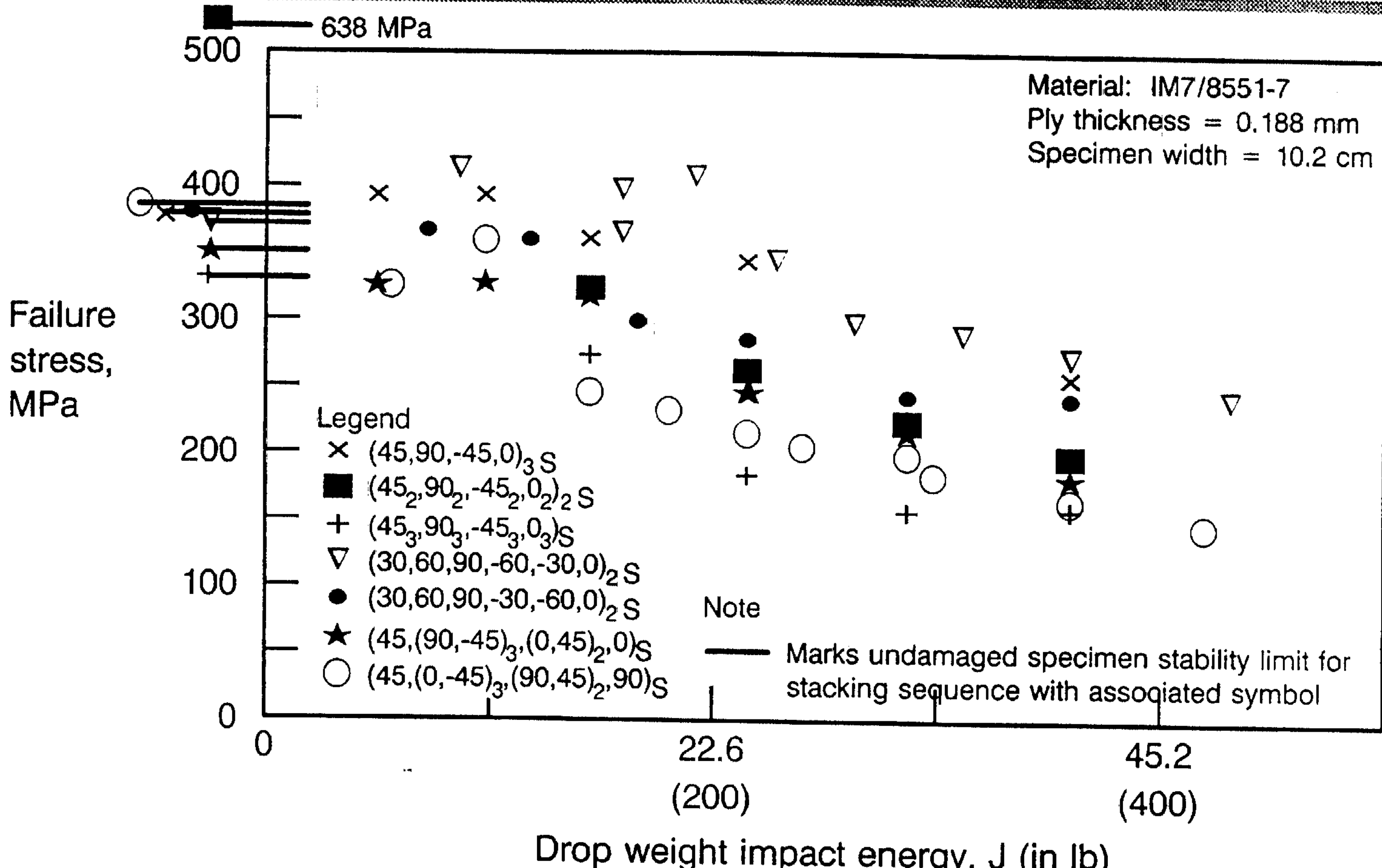


A Need to Understand Fundamentals in Four Areas

Damage Resistance Versus Damage Tolerance



Experimental Data Showing Postimpact Compression Performance as a Function of Laminate Stacking Sequence

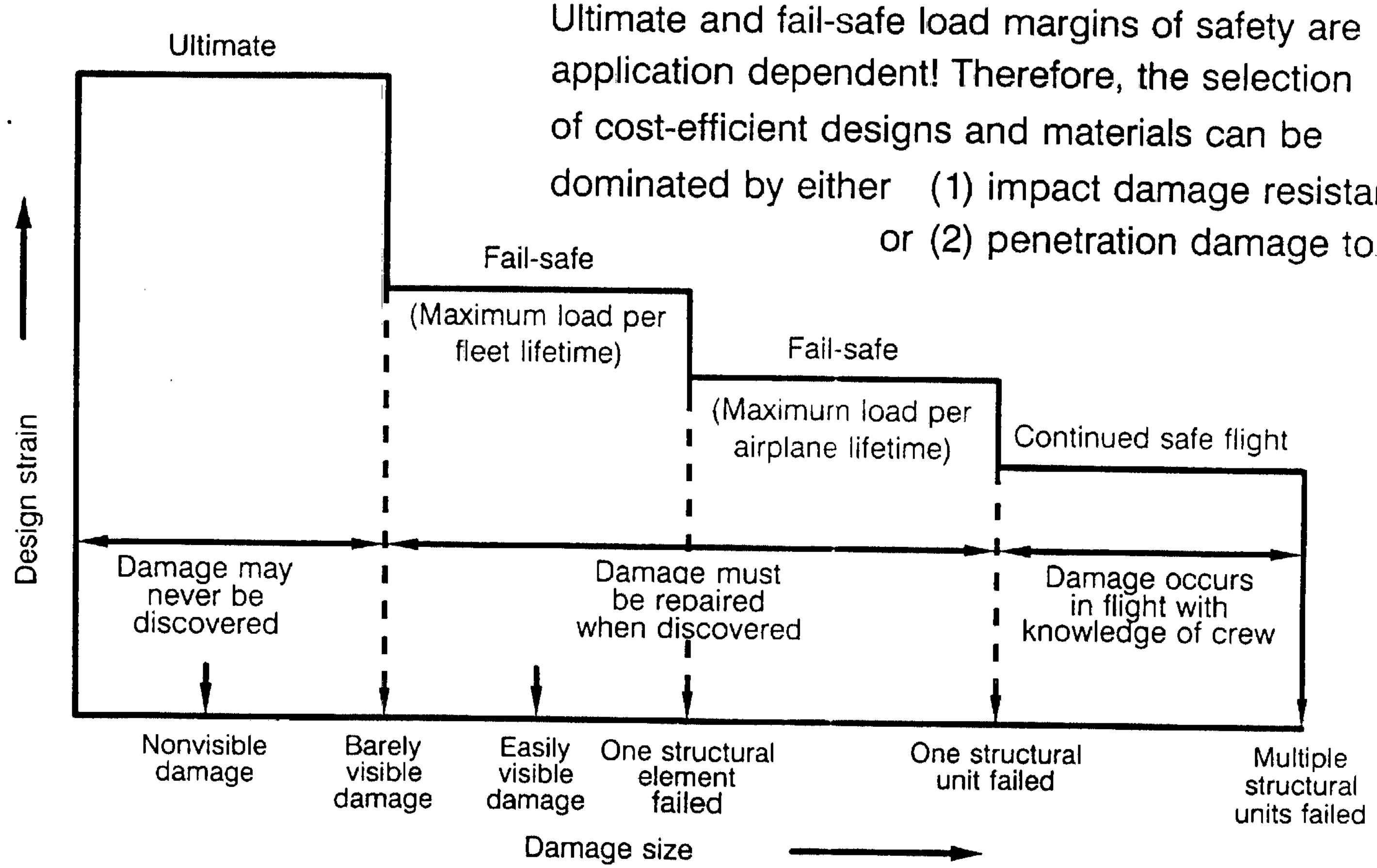


Boeing ATCAS Goals in Studying Impact Damaged Composites

- *Identify Critical Impact Threats for Fuselage Structure*
- *Characterization of Important Features of Damage State*
- *Damage Simulation That Works for Coupons and Structures*
- *Predict Residual Strength for Different Load Combinations*

Damage Tolerance: Impact Damage and Through Penetration

Ultimate and fail-safe load margins of safety are application dependent! Therefore, the selection of cost-efficient designs and materials can be dominated by either (1) impact damage resistance or (2) penetration damage tolerance.



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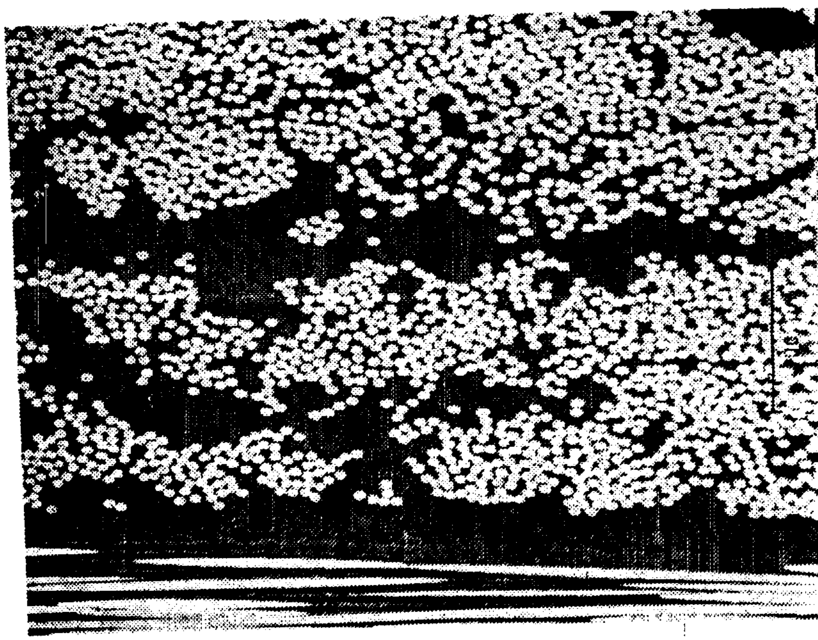
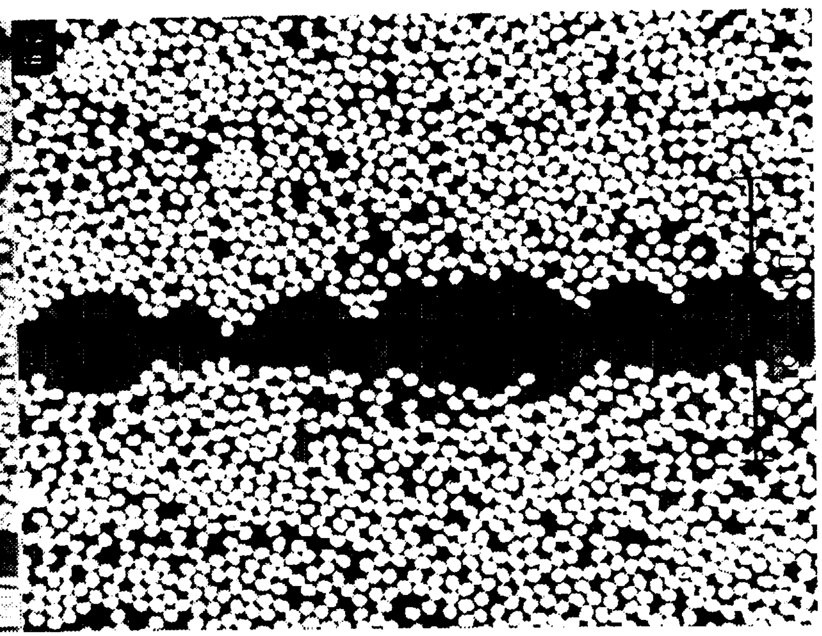
ATCAS

AL1823.36 DR1G

Resin Rich Interlaminar Layer Architecture



IM7/8551-7

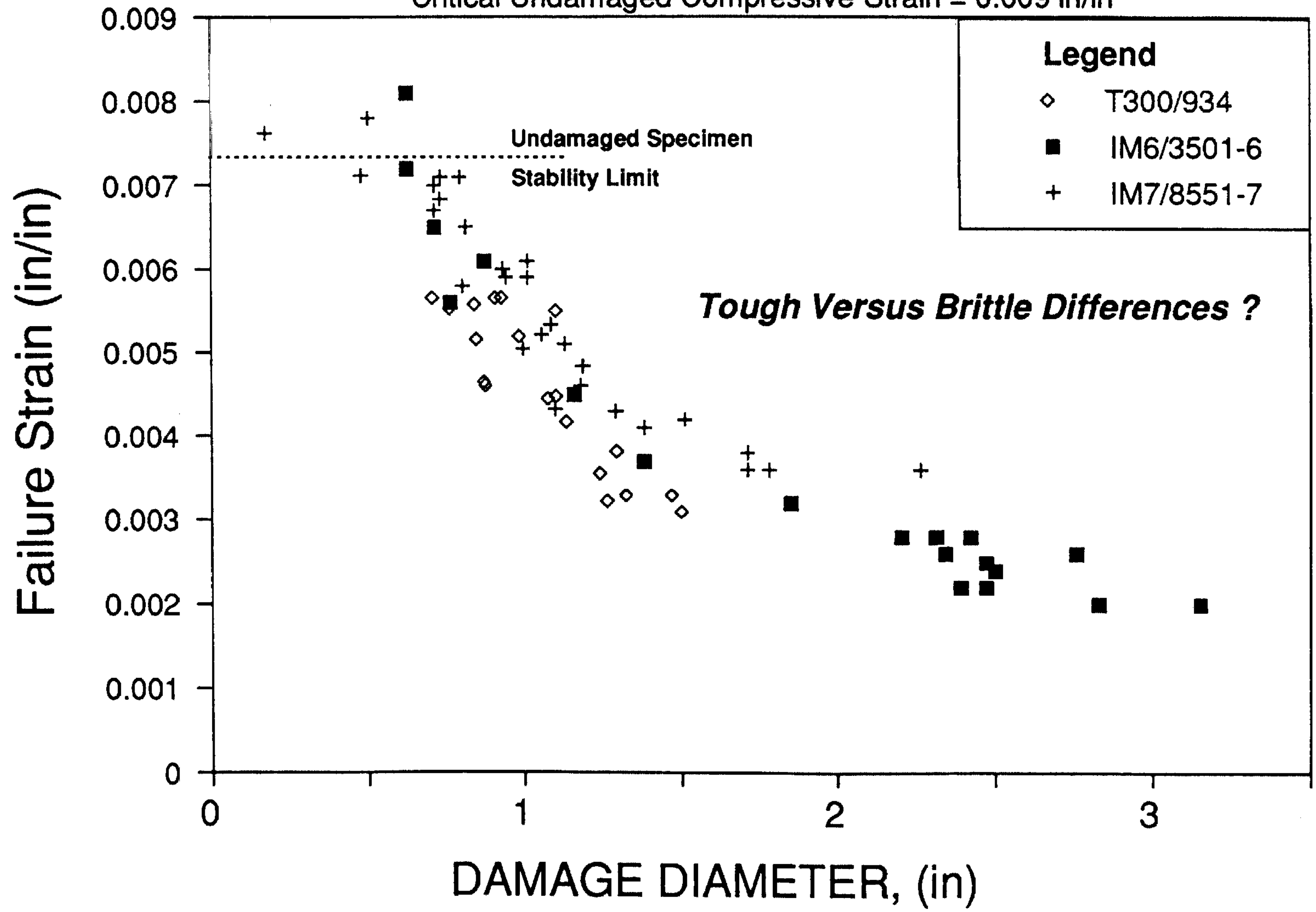


T800H/3900-2

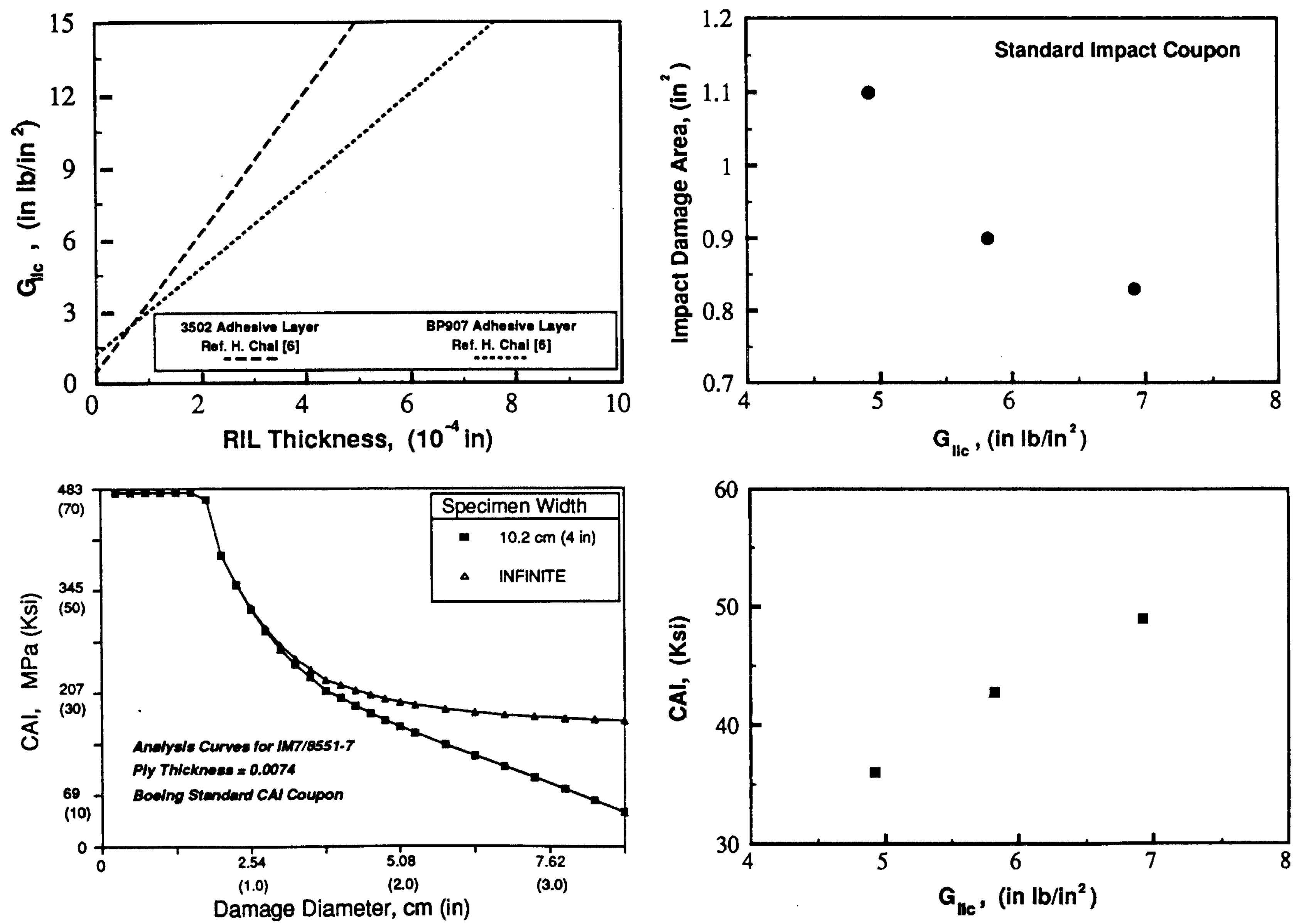
Compressive Damage Tolerance of Three Materials

Standard Quasi-Isotropic Layup, Ply Thickness = 0.0074 in.

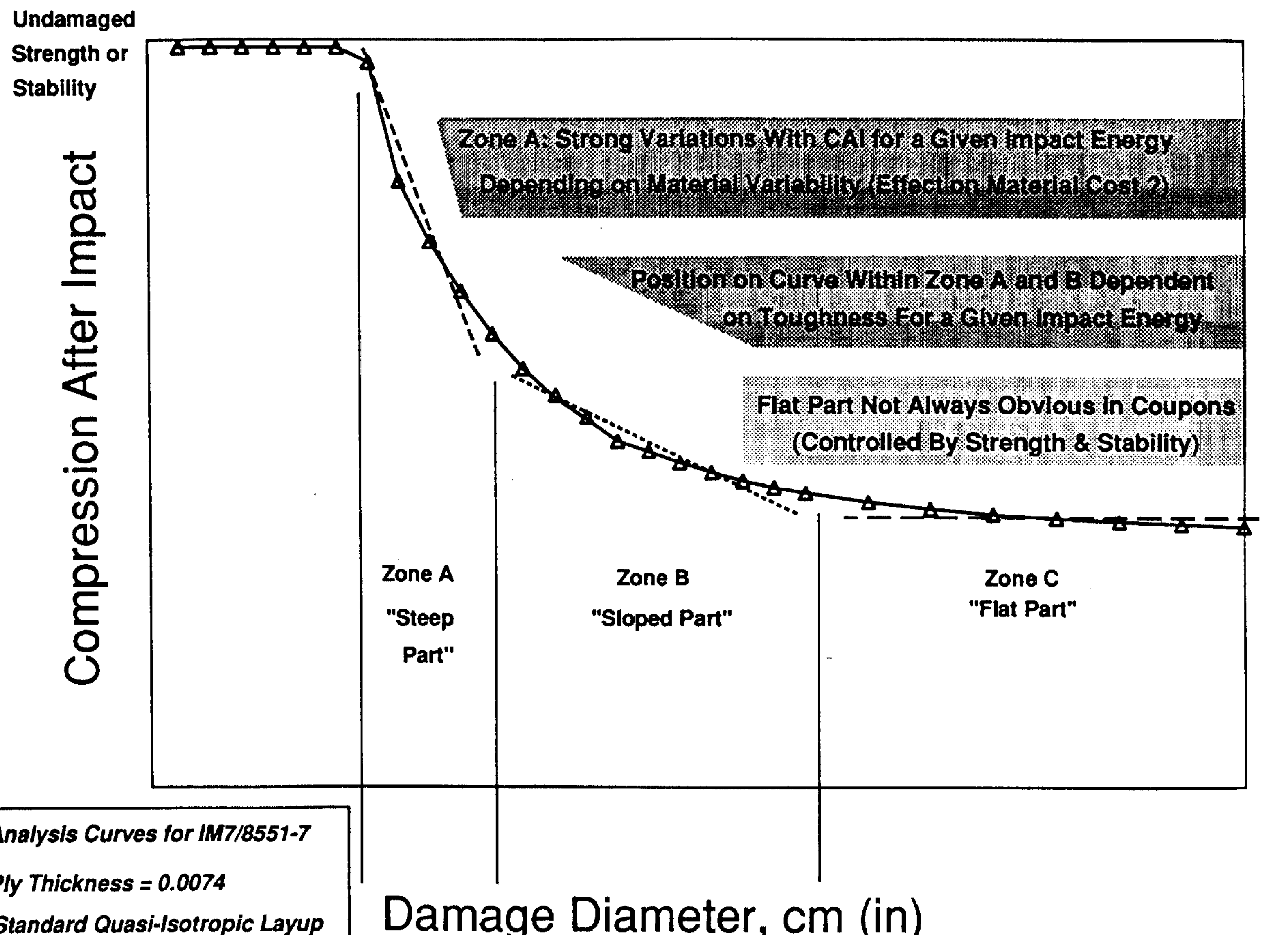
Critical Undamaged Compressive Strain = 0.009 in/in



Interlaminar Toughness is Critical to Impact Damage Resistance



Three Parts to a Damage Tolerance Curve Dominated by Sublaminate Stability



Analysis Curves for IM7/8551-7

Ply Thickness = 0.0074

Standard Quasi-Isotropic Layup

Corrected to Infinite Width

Beware of Alternate Failure Modes for New Material Forms

Toughened, Resin-Rich Interlaminar Layer for Shear Delamination Resistance Without Compromising Hot/Wet Requirements

- Evidence of Strong Increase in Mode II Toughness With Increasing Load Rate
- Localized Fiber Failure May Have Differing Relationships With Impact Variables

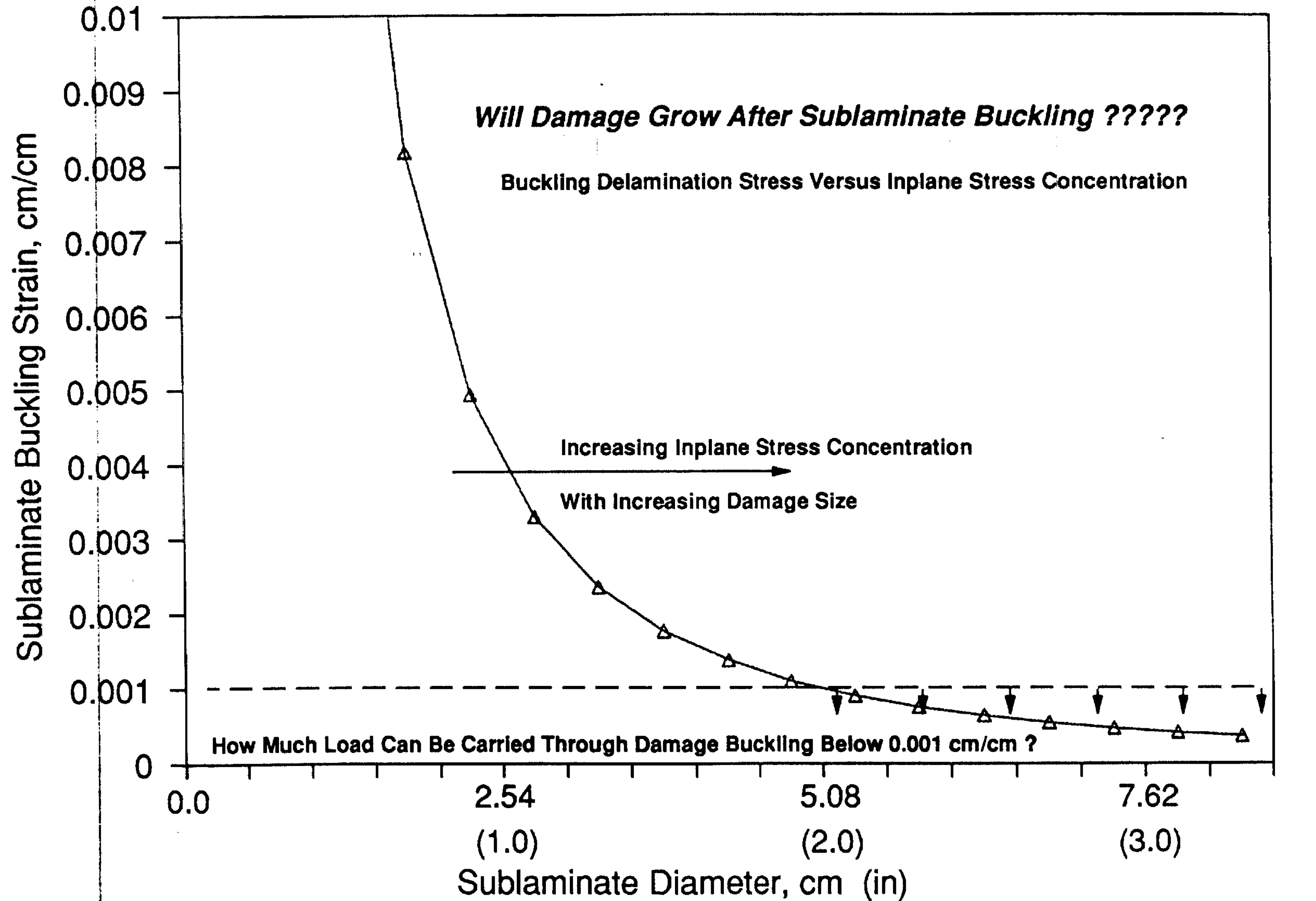
Laminate Stitching to Enhance Sublaminate Stability With Traditional "Brittle" Matrix Systems

- Should Consider Angle Impact and Larger Diameter Impactor for Fiber Failure
- What About Flat Part of Damage Tolerance Curve (Large Through Penetrations)

There's No Such Thing as a Free Lunch ???

With Proper Fundamental Understanding of Materials, Mechanics, and Design Criteria, Impact Damage Issues for New Material Forms Can Be Solved, Leading to Safe and Efficient Use of Composites

Shape of Sublaminate Stability Curve for Damage in Standard Coupon



Impact Damaged Composites, Part 1: Damage Simulation and Strength Predictions

Transport Fuselage Structures → *L. Ilcewicz*

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Potential Impact Damage States

Impact event

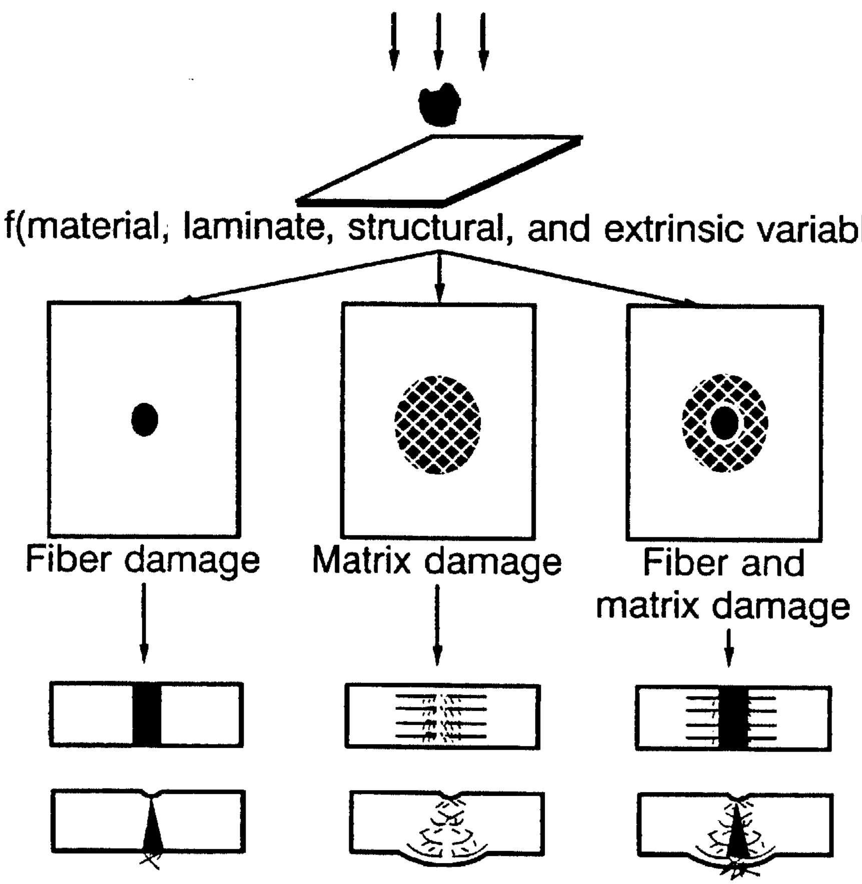
Impact damage = f(material, laminate, structural, and extrinsic variables)

Planar view

Cross-sectional view

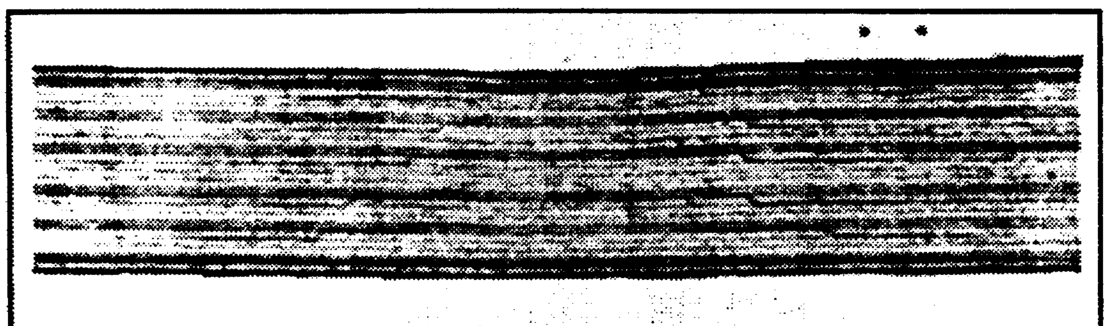
Symmetric damage

Unsymmetric damage



Cross-Sections of Impacted Specimens Demonstrating Failure Modes

Localized
fiber failure



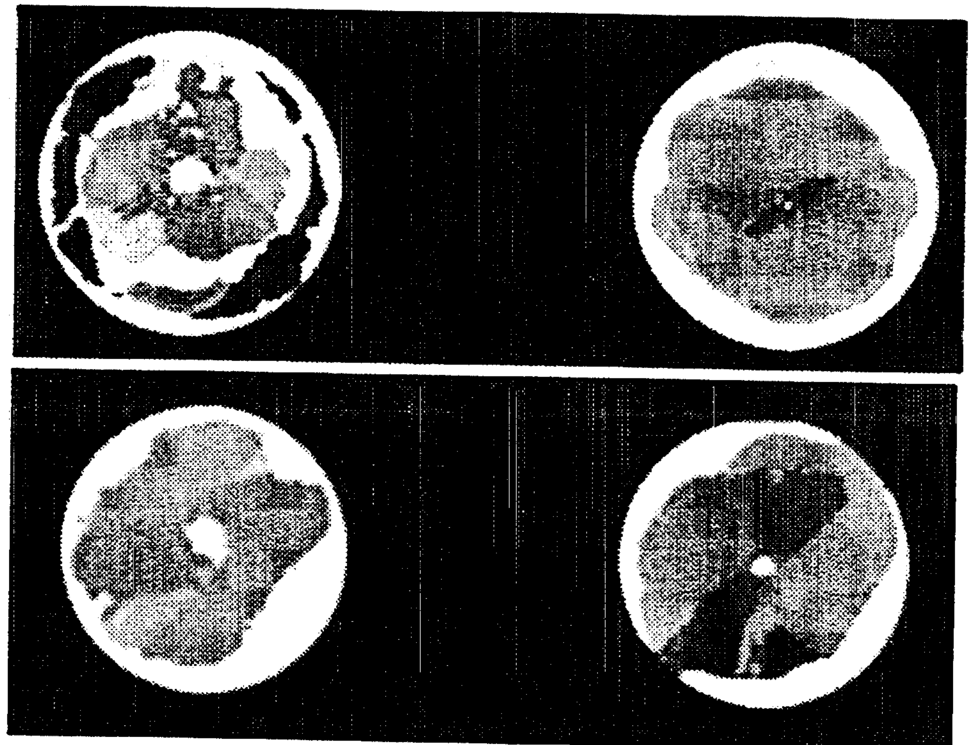
Delamination and
transverse cracks

ATCAS

HL1568B.15

ORIGINAL PAGE IS
OF POOR QUALITY

A Comparison of Front and Back Surface Pulse-Echo C-Scans for Two Material Types



Front

Back

IM7/8551-7
$(45,0,-45,90)_{3S}$

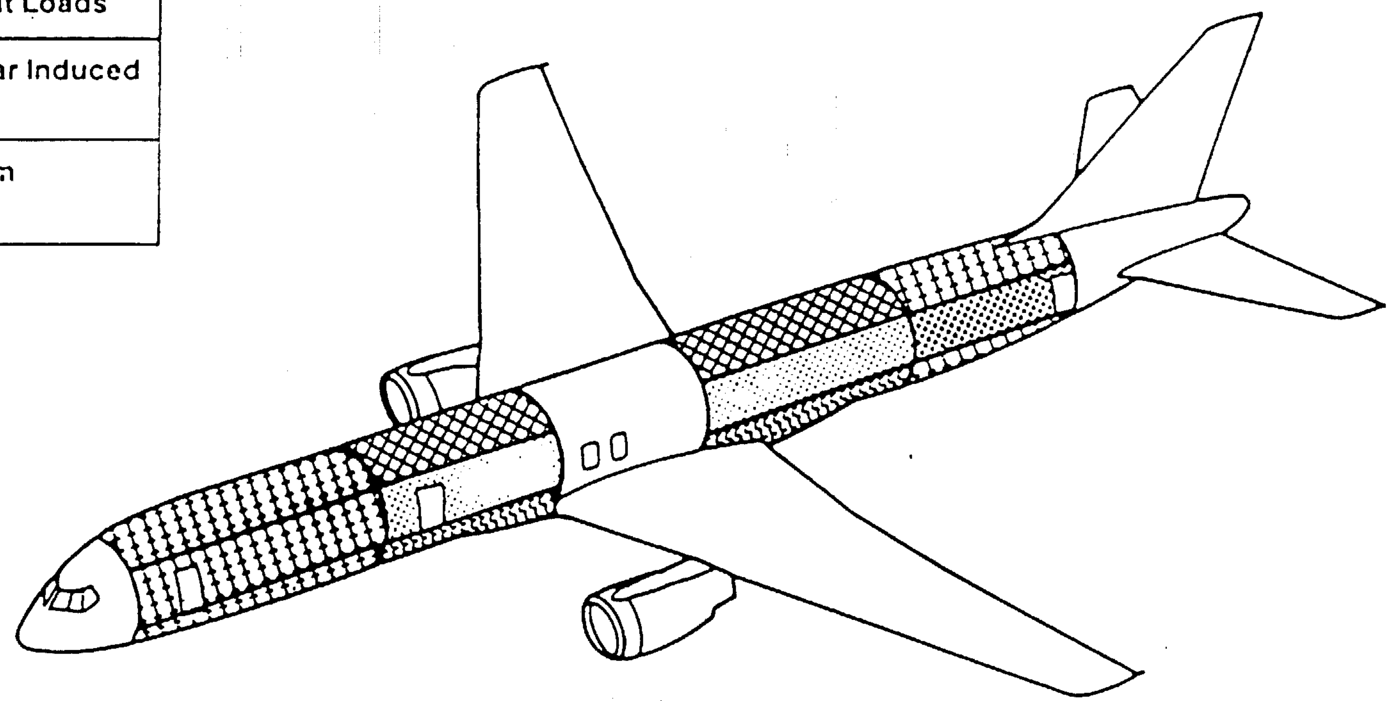
T300/934
$(45,90,-45,0)_{3S}$

NASA/BOEING
ATCAS

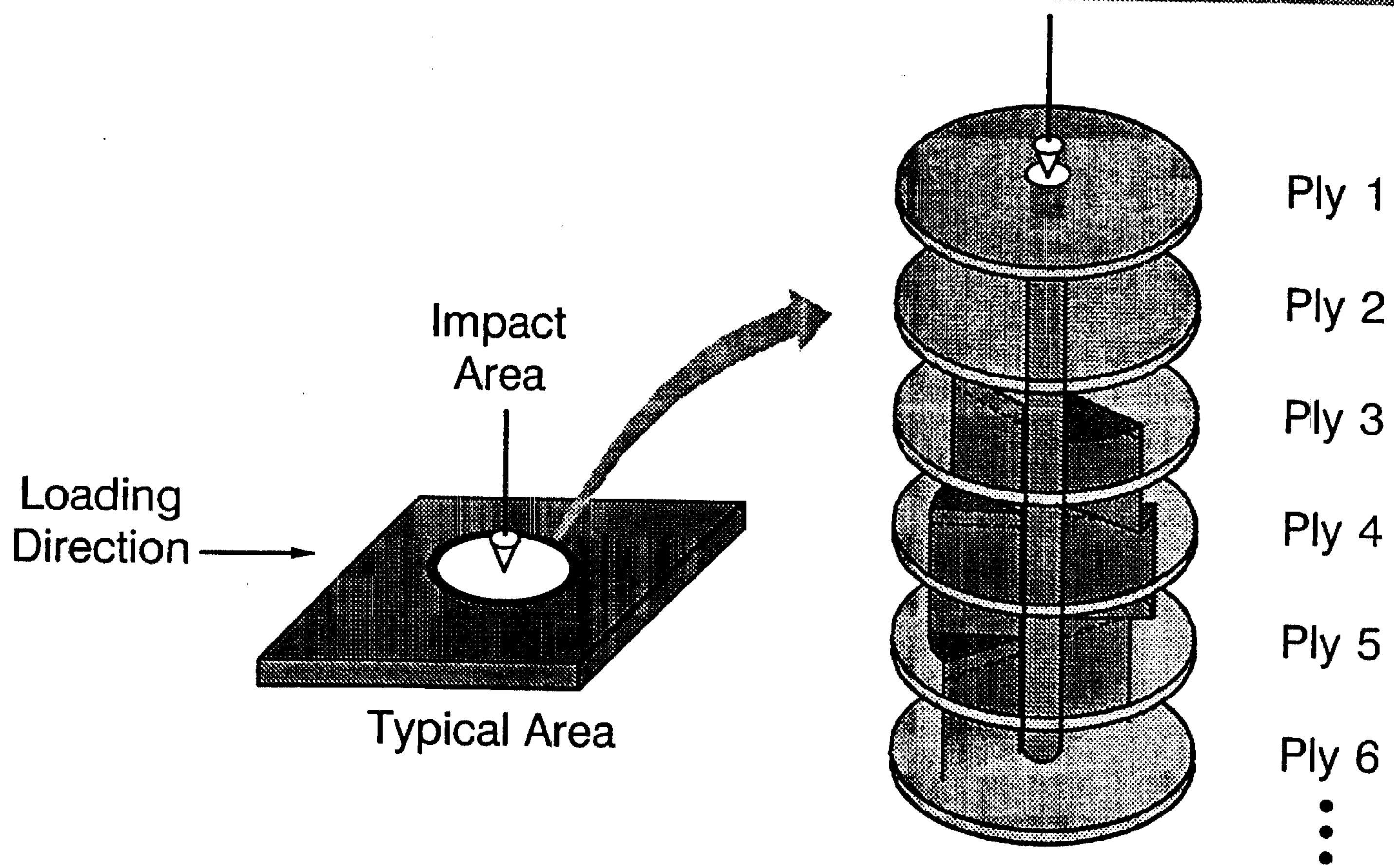
HL15889 15 R2db

Aircraft Fuselage Critical Load Conditions

	Critical Load Condition
	Pressure Only
	Pressure With Tension Induced From Flight Loads
	Pressure With Compression Induced From Flight Loads
	Pressure With Shear Induced From Flight Loads
	Shear Induced From Flight Loads



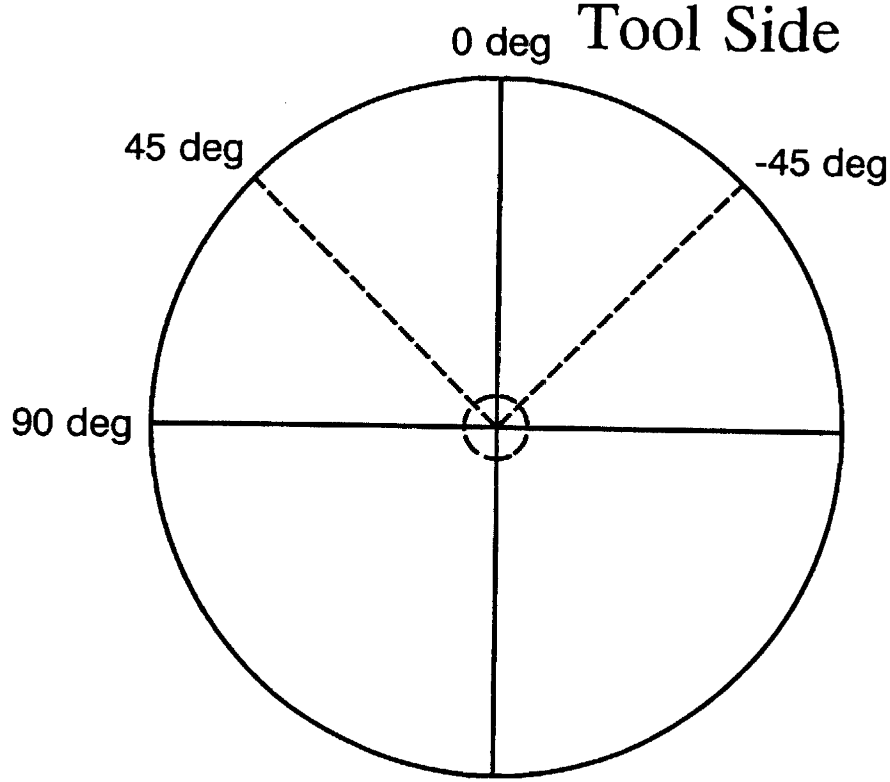
Characteristic Damage State Caused by Impact



HL1568.28 R1h

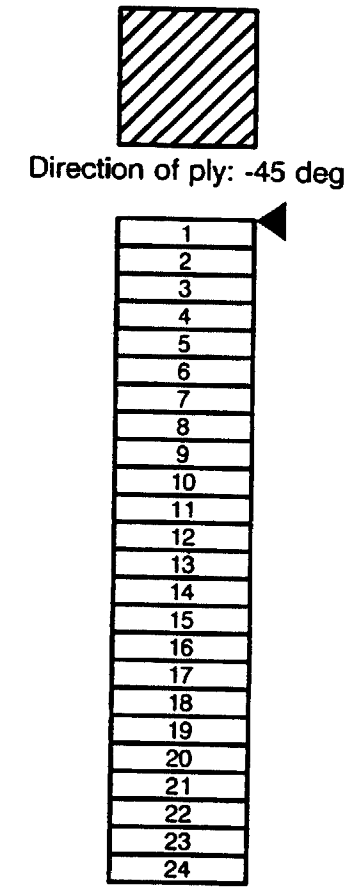
ACDP10
Ply 1

Tool Side



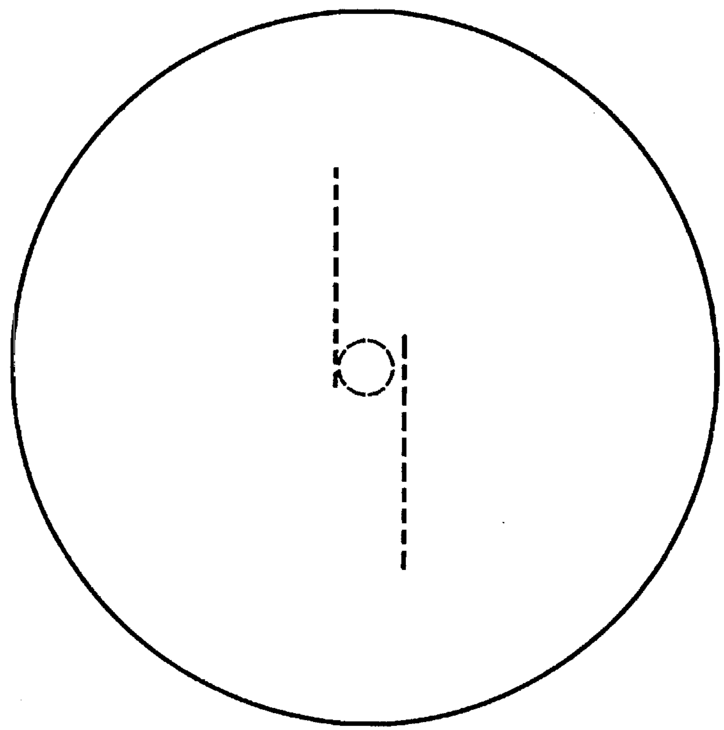
Scale factor 1 in = 0.67 in

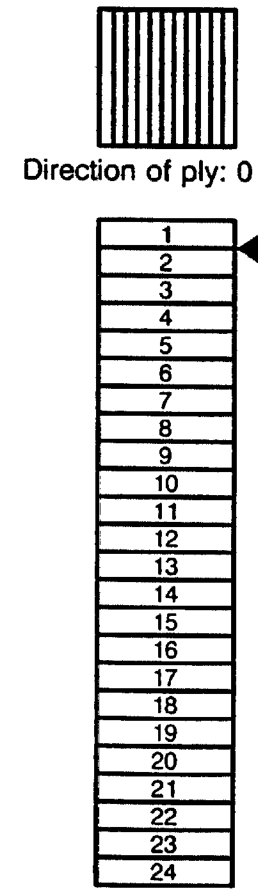
Stacking sequence: (-45/0/45/90)



HL1568.20 R1h

Ply 2





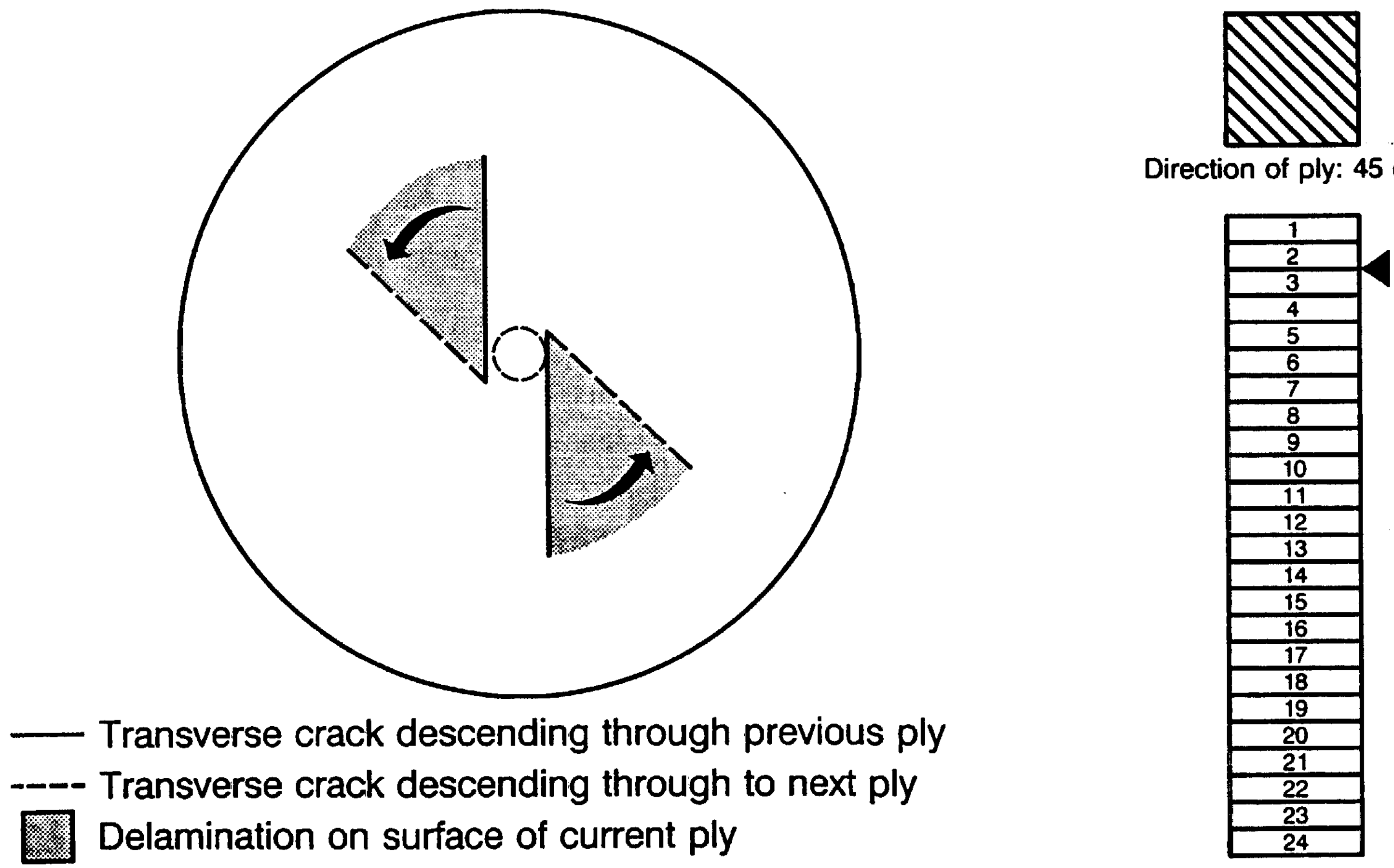
— Transverse crack descending through previous ply

---- Transverse crack descending through to next ply

Delamination on surface of current ply

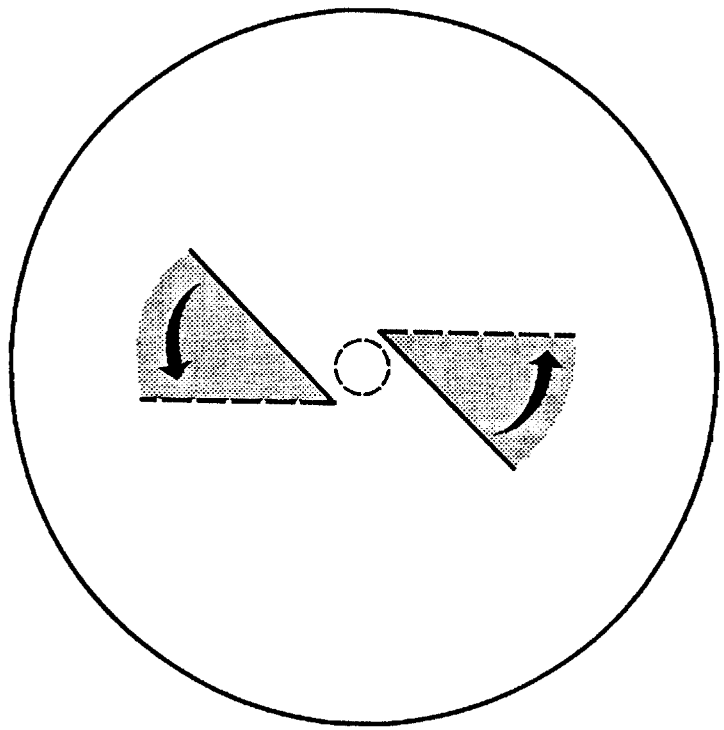
HL1568.21

Ply 3



HL1568.22 R1h

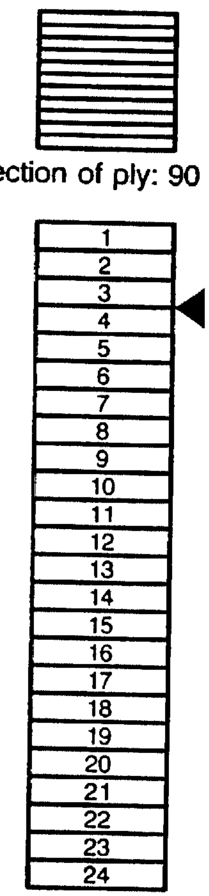
Ply 4



— Transverse crack descending through previous ply

---- Transverse crack descending through to next ply

Delamination on surface of current ply



Direction of ply: 90 deg

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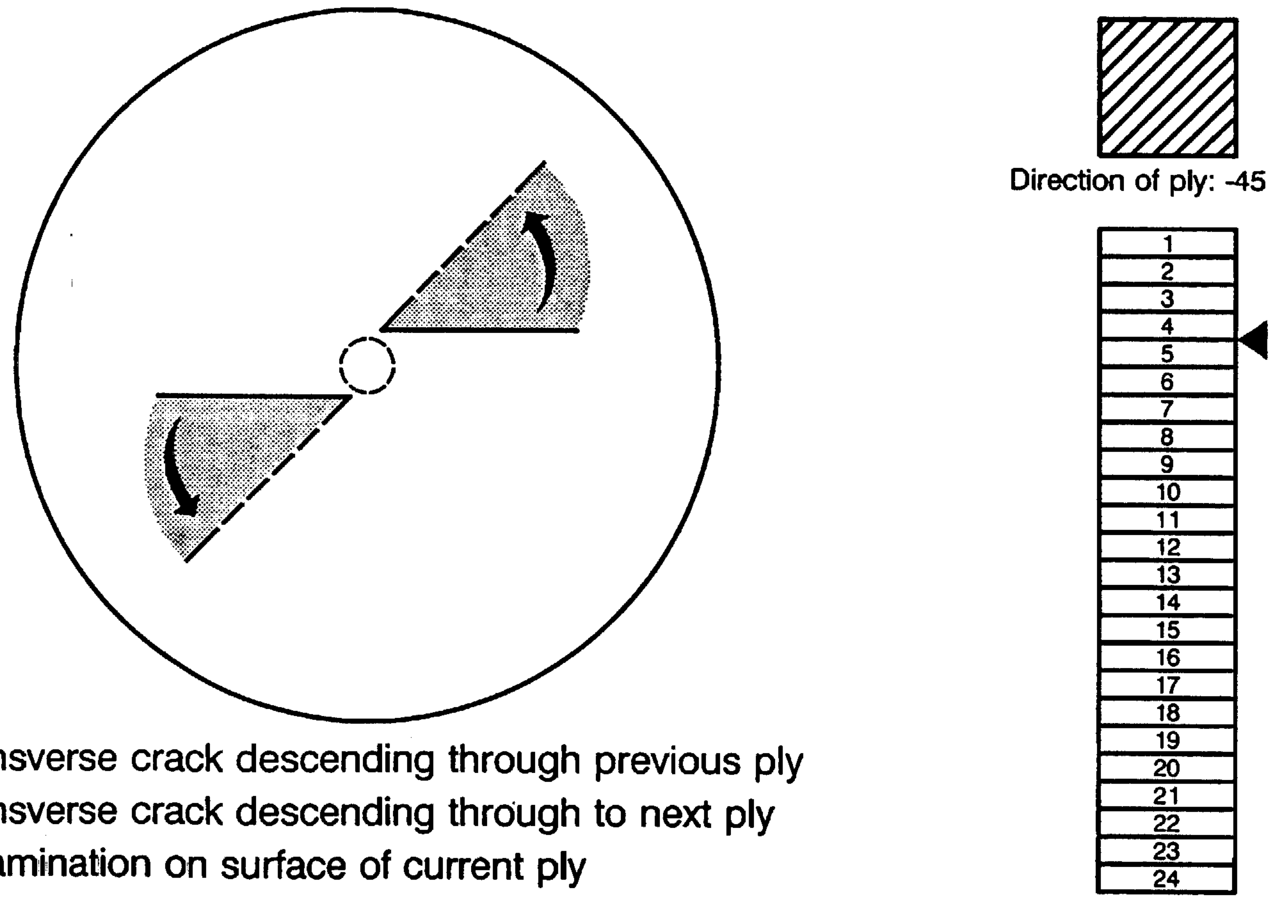
HL1568.23 R1h

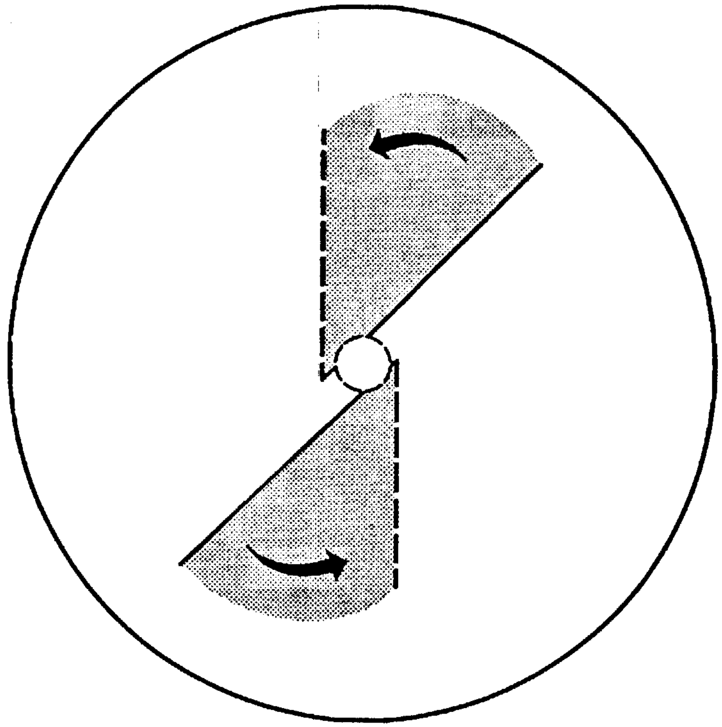
Ply 5



—— Transverse crack descending through previous ply
---- Transverse crack descending through to next ply
Delamination on surface of current ply

HL1568.24 R1h

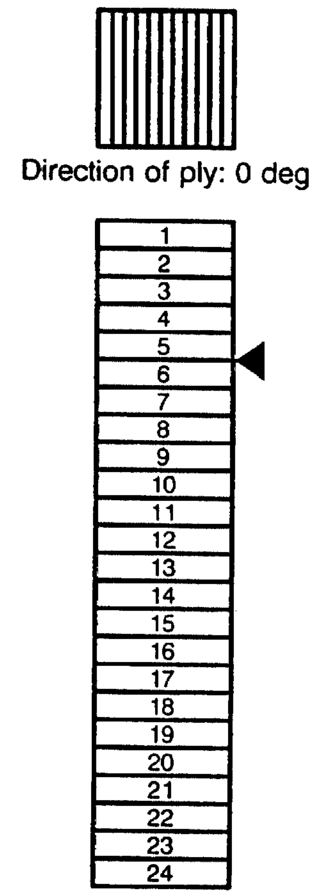
Ply 6



— Transverse crack descending through previous ply

---- Transverse crack descending through to next ply

▒ Delamination on surface of current ply



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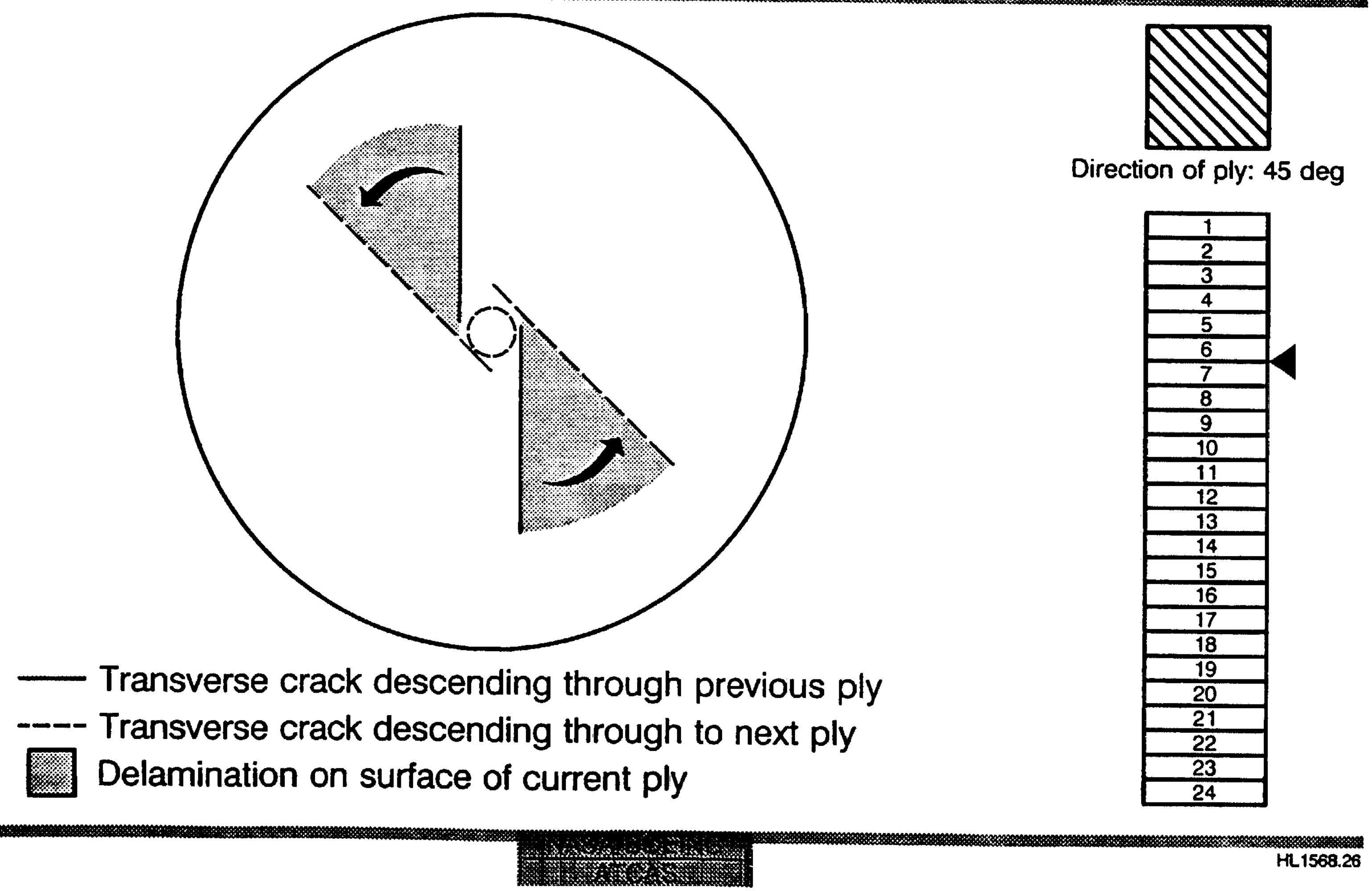
HL1568.25 r1h

Ply 7


Direction of ply: 45 deg
1
2
3
4
5
6
7
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9
10
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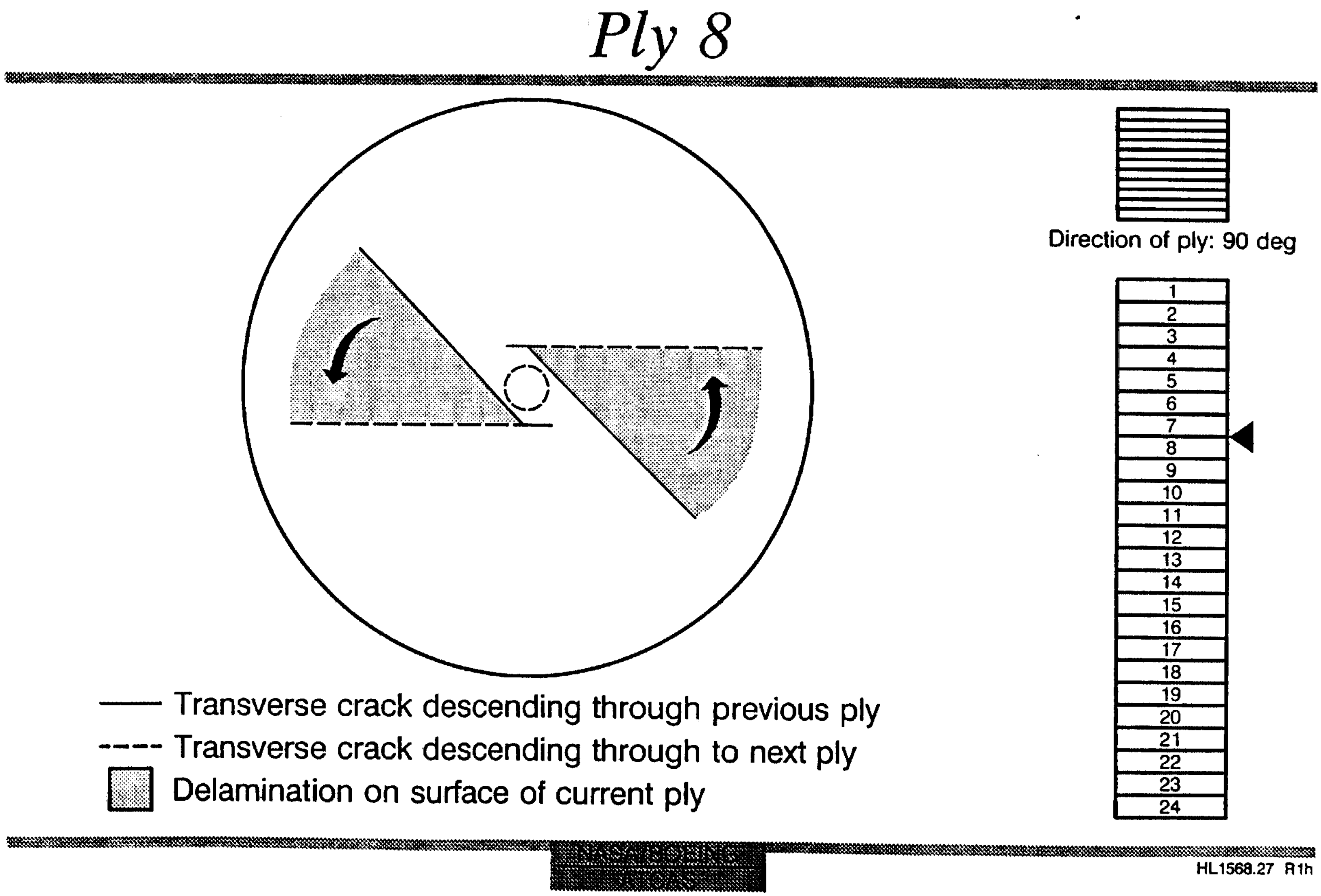
12
13
14
15
16
17
18
19
20
21
22
23
24
Transverse crack descending through previous ply
Transverse crack descending through to next ply
Delamination on surface of current ply
NASA/BOEING
ATCAS
HL1568.26


Ply 8
Direction of ply: 90 deg
1
2
3
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6
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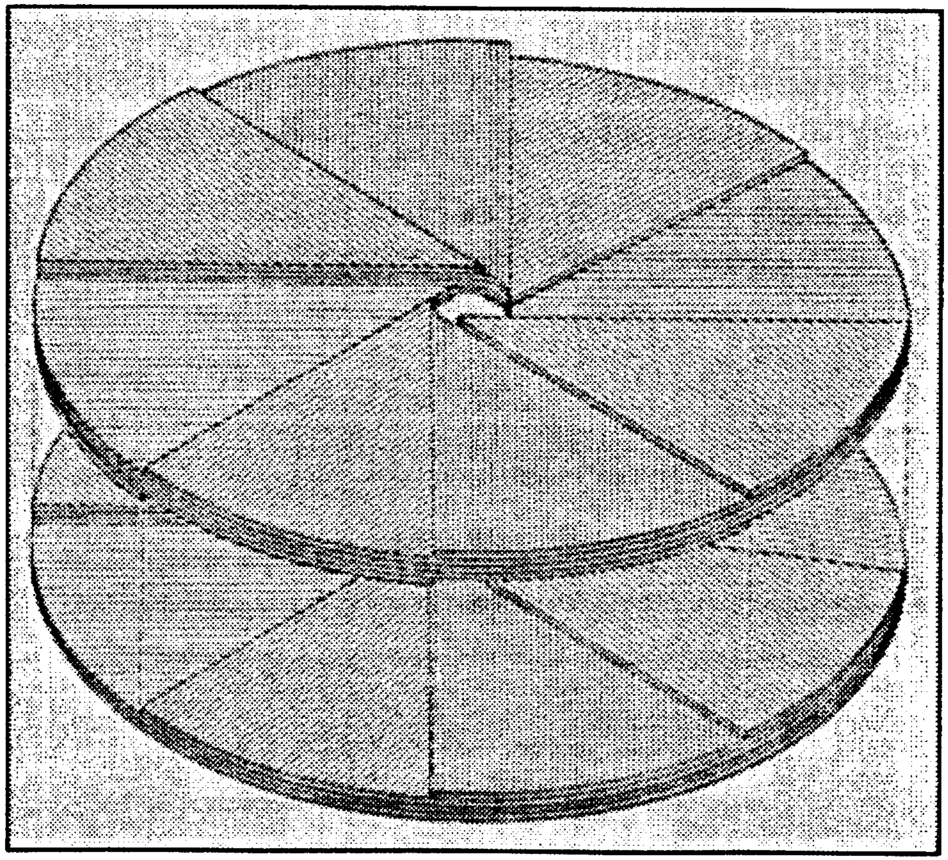
11
12
13
14
15
16
17
18
19
20
21
22
23
24
Transverse crack descending through previous ply
Transverse crack descending through to next ply
Delamination on surface of current ply
HL1568.27 R1h

Sublaminates for (-45,0,45,90)ns Laminates



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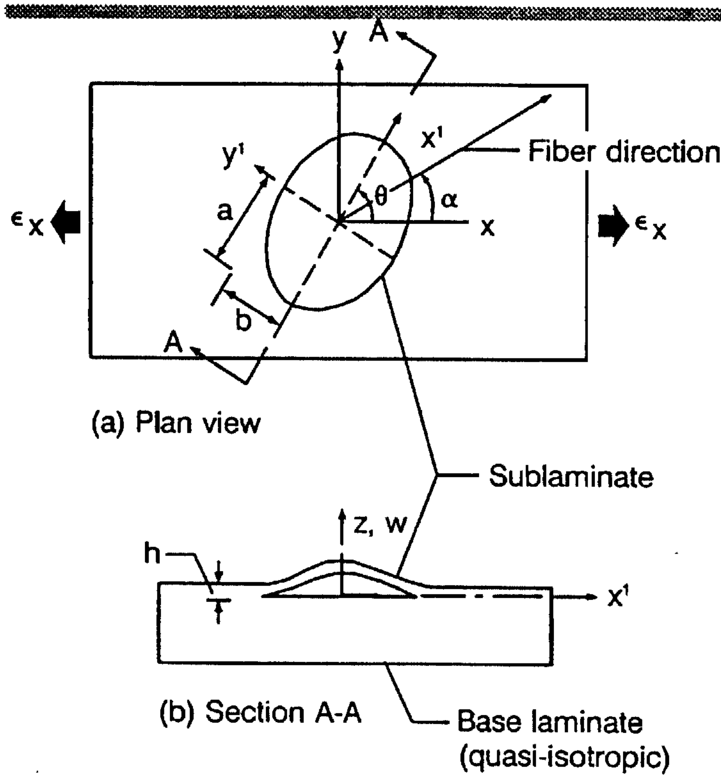
ATCAS

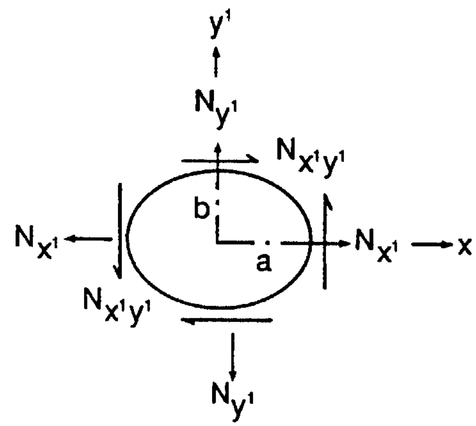
C1568B.07

Sublaminate Buckling Model



(a) Plan view

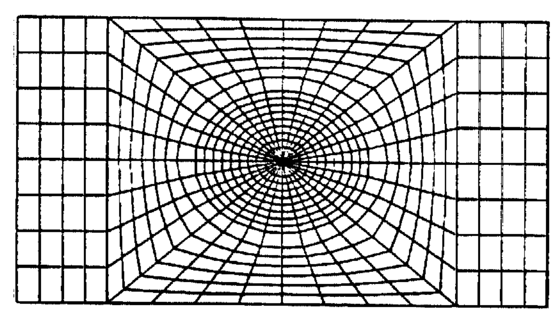
(b) Section A-A



(c) Free body diagram of the sublaminate

Buckling of a sublaminate in a quasi-isotropic composite laminate, K. N. Shivakumar and J. D. Whitcomb, NASA TM 85755, February 1984

STAGSC-1



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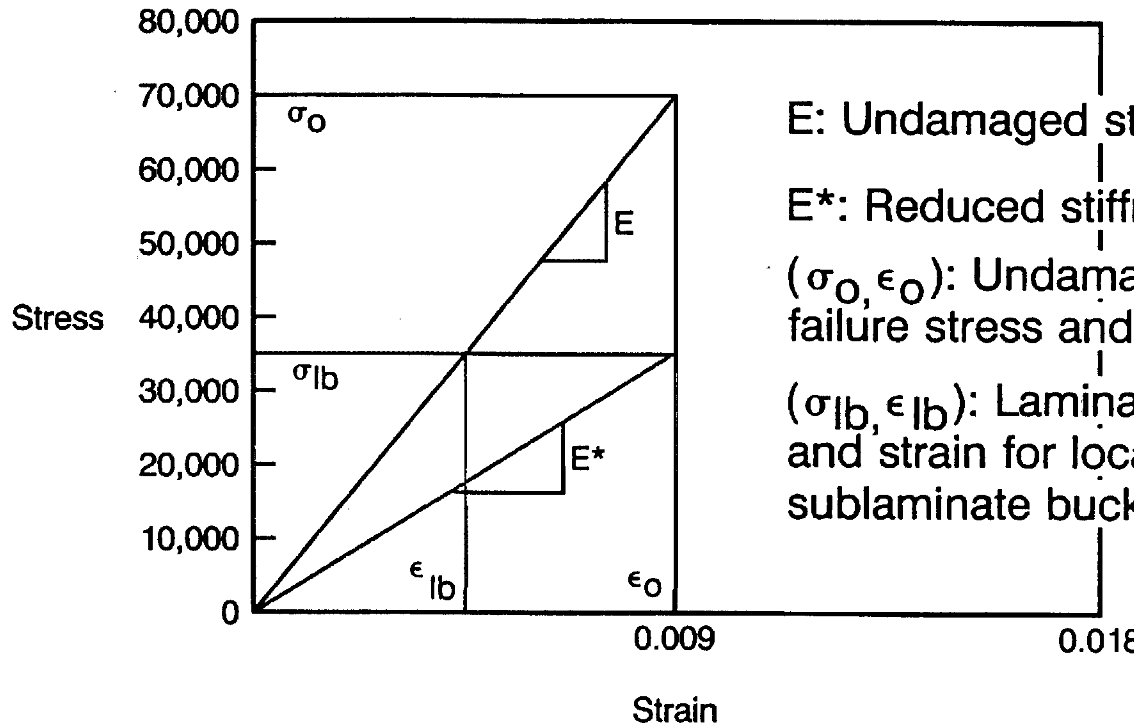
CL1568.03 R1rk

Reduced Stiffness Calculation



E: Undamaged stiffness

E*: Reduced stiffness

(σ_o, ϵ_o): Undamaged laminate failure stress and strain

$(\sigma_{lb}, \epsilon_{lb})$: Laminate stress and strain for local sublaminate buckling

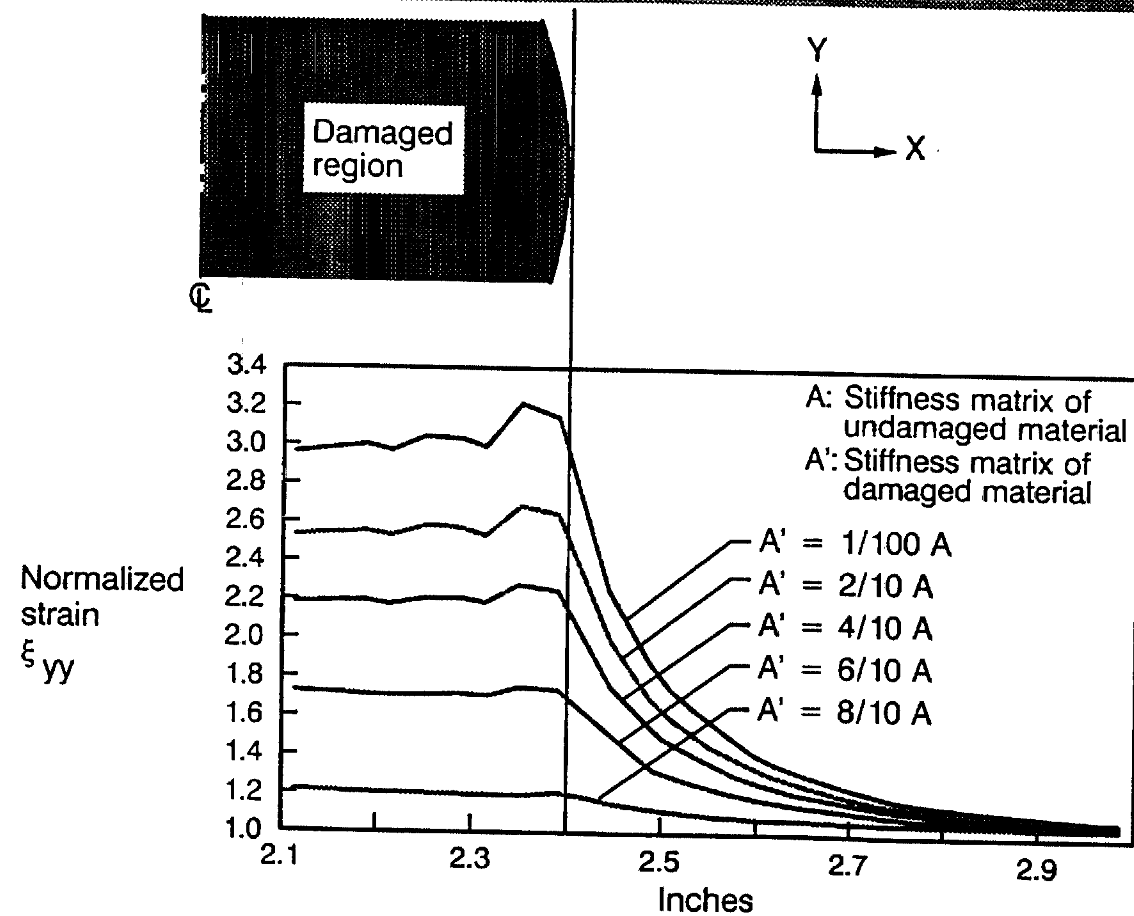
$$E^* = \frac{E \ \epsilon_{lb}}{\epsilon_0}$$

Assumptions:

1. Load carried by damaged area becomes constant following local buckling
2. Strain compatibility between damaged and undamaged material at failure

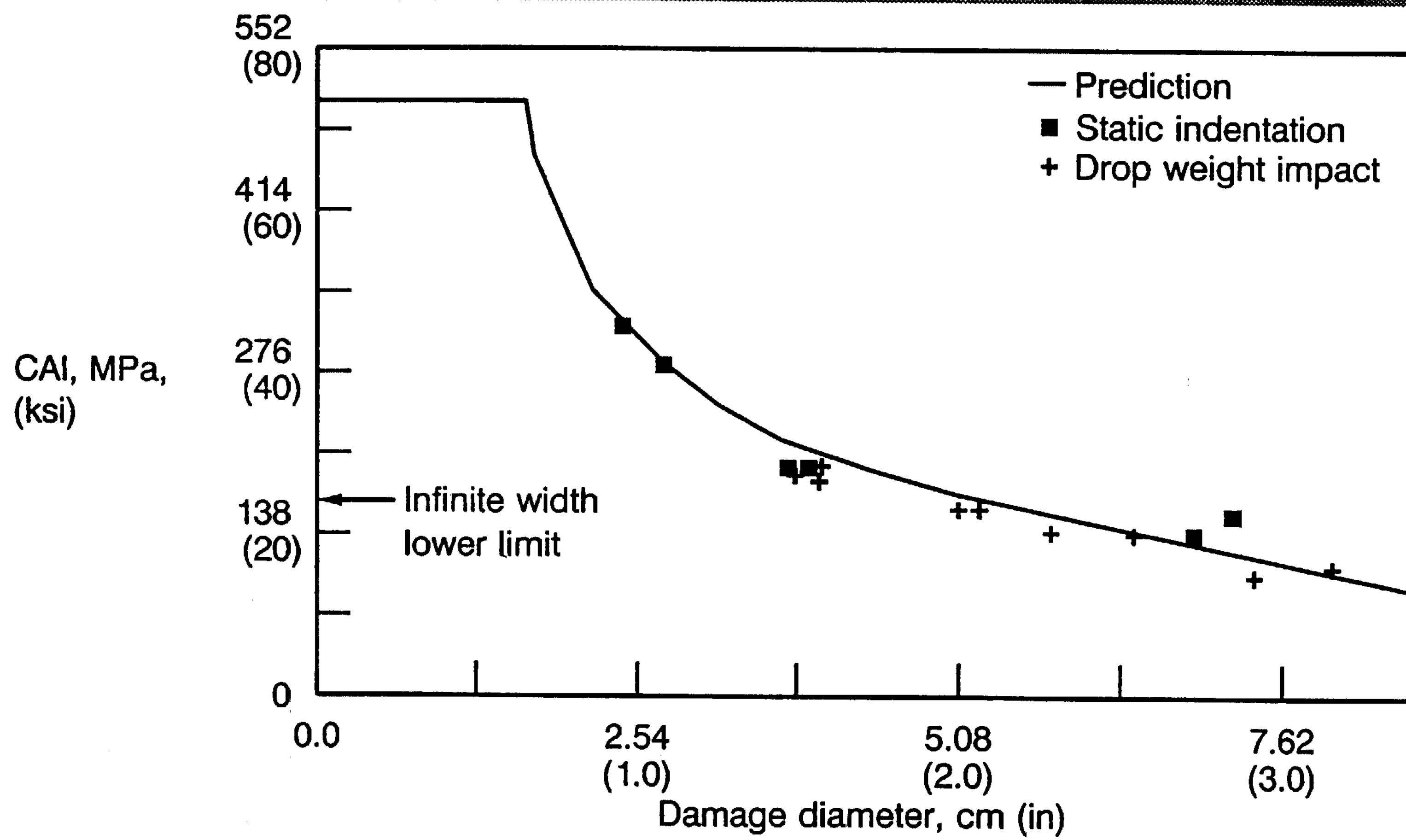
CL1568.04 R1h

Strain Distribution Near Damaged Region



CL1568.05 R2h

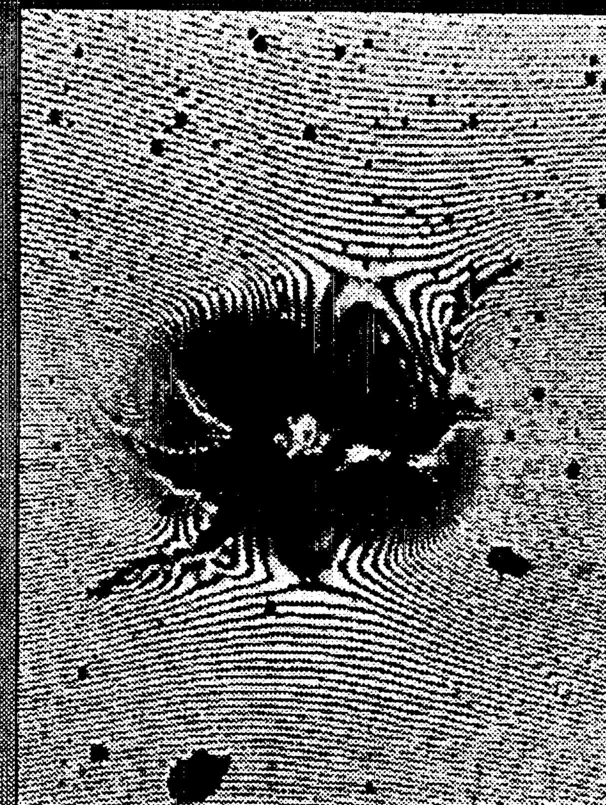
Predictions and Experimental Results for 12.7-cm Wide Specimens AS6/3501-6, $(45/0/-45/90)_{5S}$, and Ply Thickness = 0.0188 cm



CL1568.17 R2h

Experimental Determination of Sublaminate Buckling and Strain Distribution of Impacted Laminates

Out-of-plane displacement contours

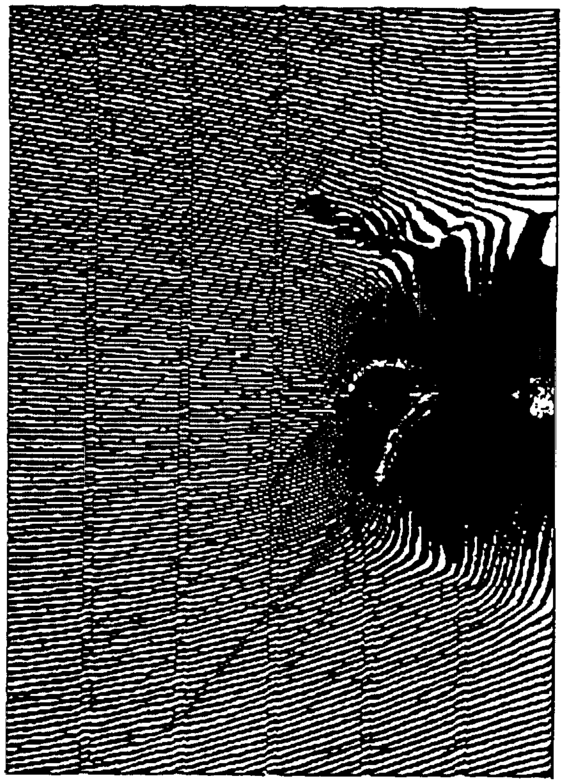


In-plane displacement contours

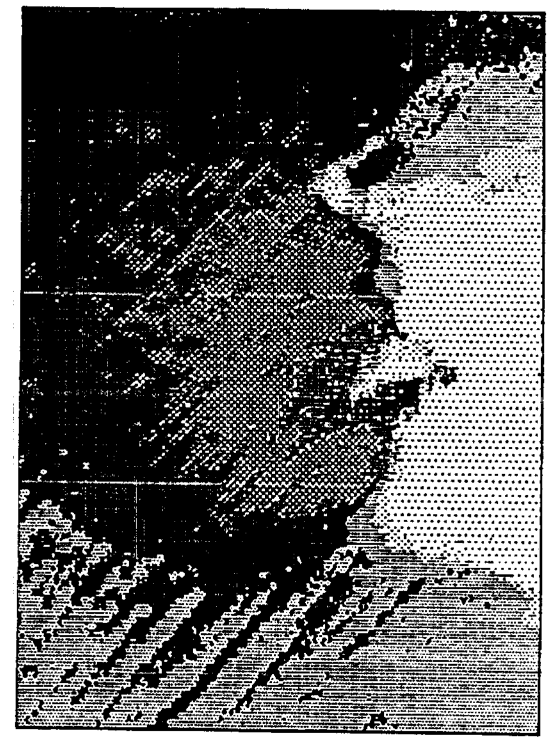
NASA/BOEING
ATCAS

Experimental Displacement and Strain Contours

IM7/8551-7, $(45,90,-45,0)_{3S}$,Damage Diameter = 1.28 in.



Axial Displacement field, Specimen 88/77-1-1 23000 lb



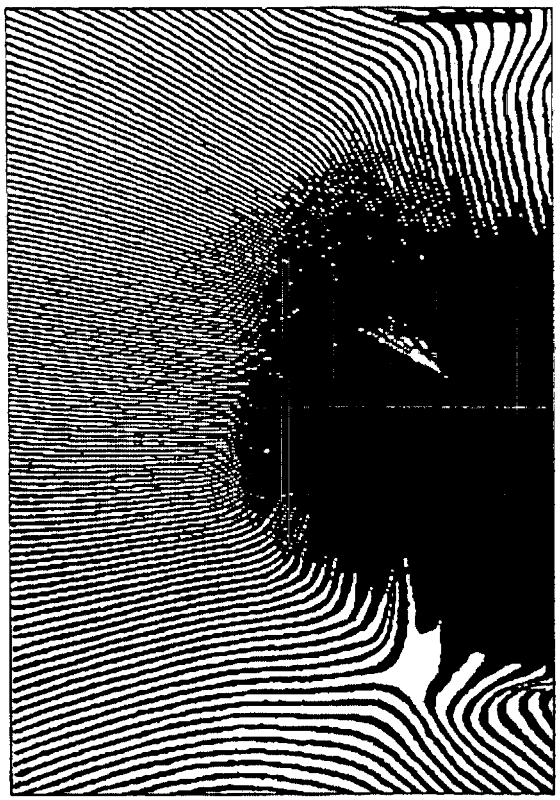
Axial Strain Field, Specimen 88/77-1-1. 23000 lb

Measured Strain Concentration at 80% of failure is between 1.19 and 1.72

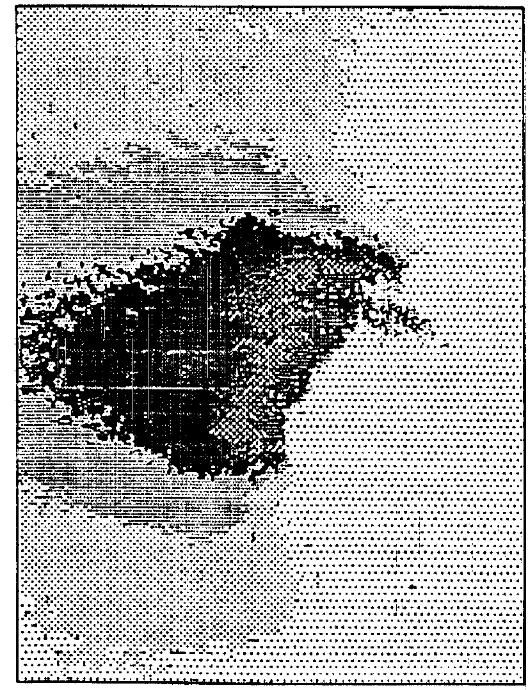
Calculated Strain Concentration at failure is 1.92

Experimental Displacement and Strain Contours

IM7/3501-6, $(45,90,-45,0)_{3S}$, Damage Diameter = 2.6 in.



Axial displacement Field, Specimen 90/177-2. 13000 lb



Axial Strain Field, Specimen 90/177-2. 13000 lb

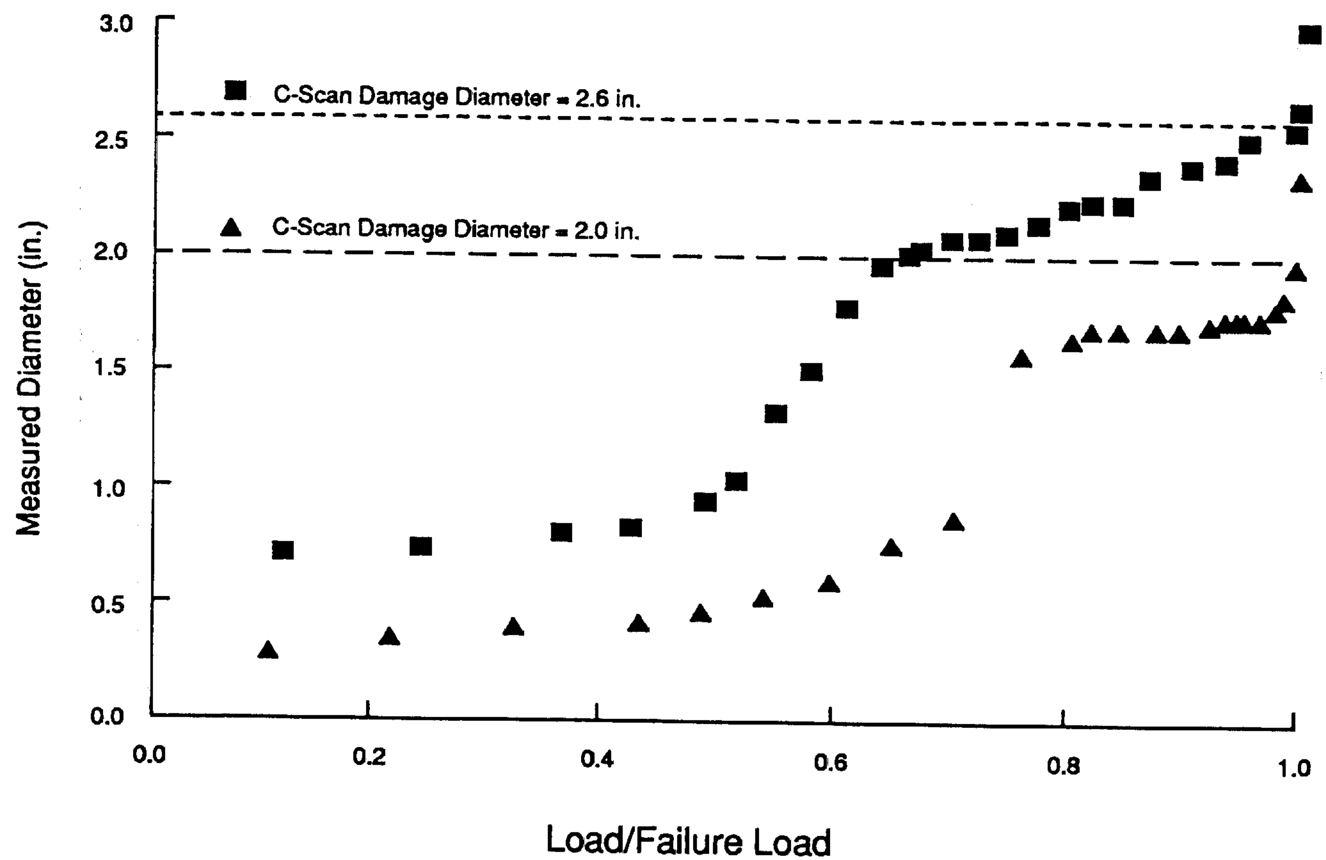
Measured Strain Concentration at 80% of failure is between 2.9 and 6.6

Calculated Strain Concentration at failure is 4.4

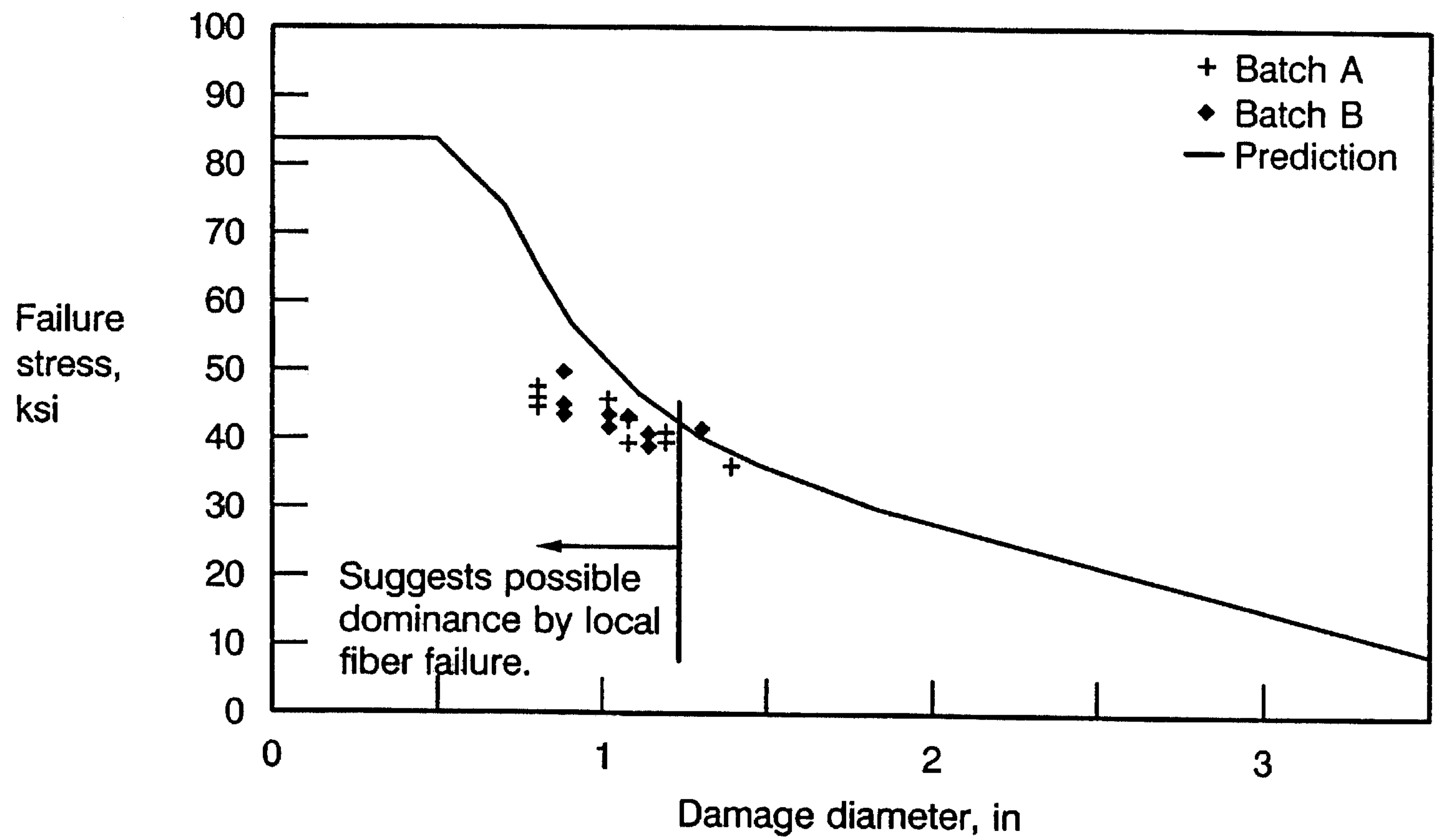
ORIGINAL PAGE IS
OF POOR QUALITY

Diameter of Buckled Region vs. Normalized Load

IM7/3501-6, $(45,90,-45,0)_{3S}$

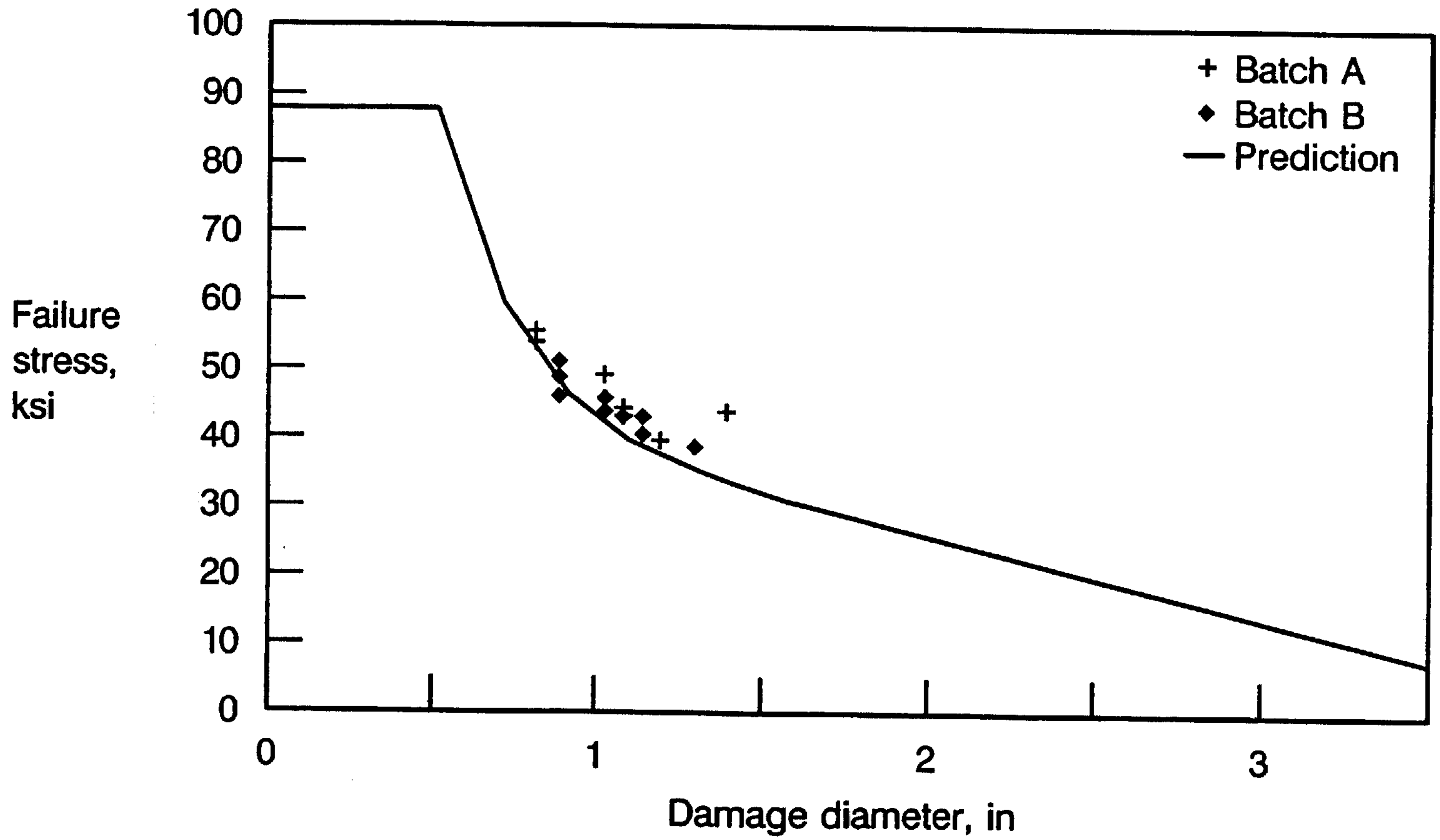


Predictions and Experimental Results for 10.2-cm Wide Specimens T800H/3900-2, $(45/0/-45/90)_{3S}$ and Ply Thickness = 0.0194 cm



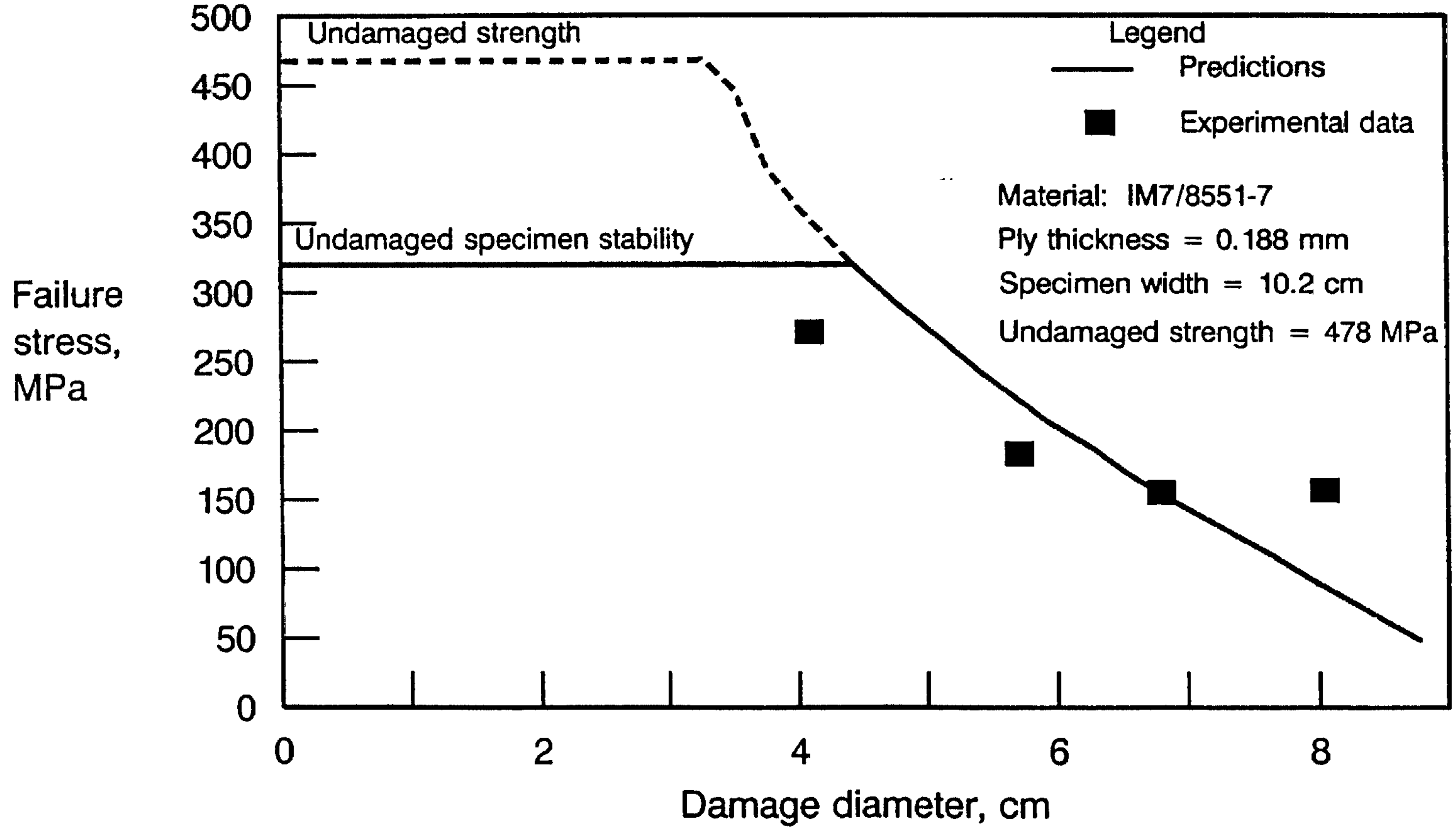
CL1568.24

Predictions and Experimental Results for 10.2-cm Wide Specimens T800H/3900-2, $(45/0/-45/90)_{4S}$ and Ply Thickness = 0.0151 cm



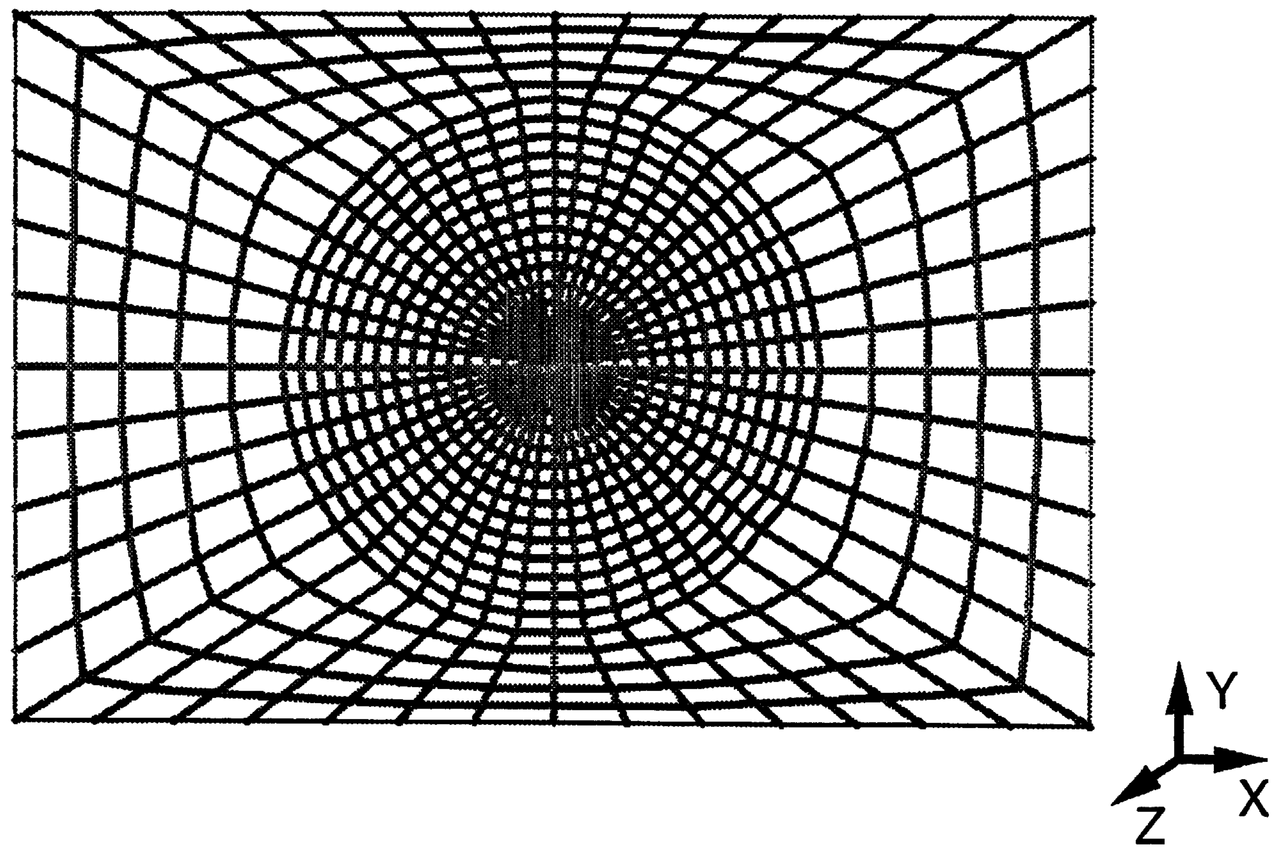
CL1568.23

Sublaminate Stability/Reduced Stiffness Predictions of CAI for Laminate Layup: $(45_3,90_3,-45_3,0_3)_S$



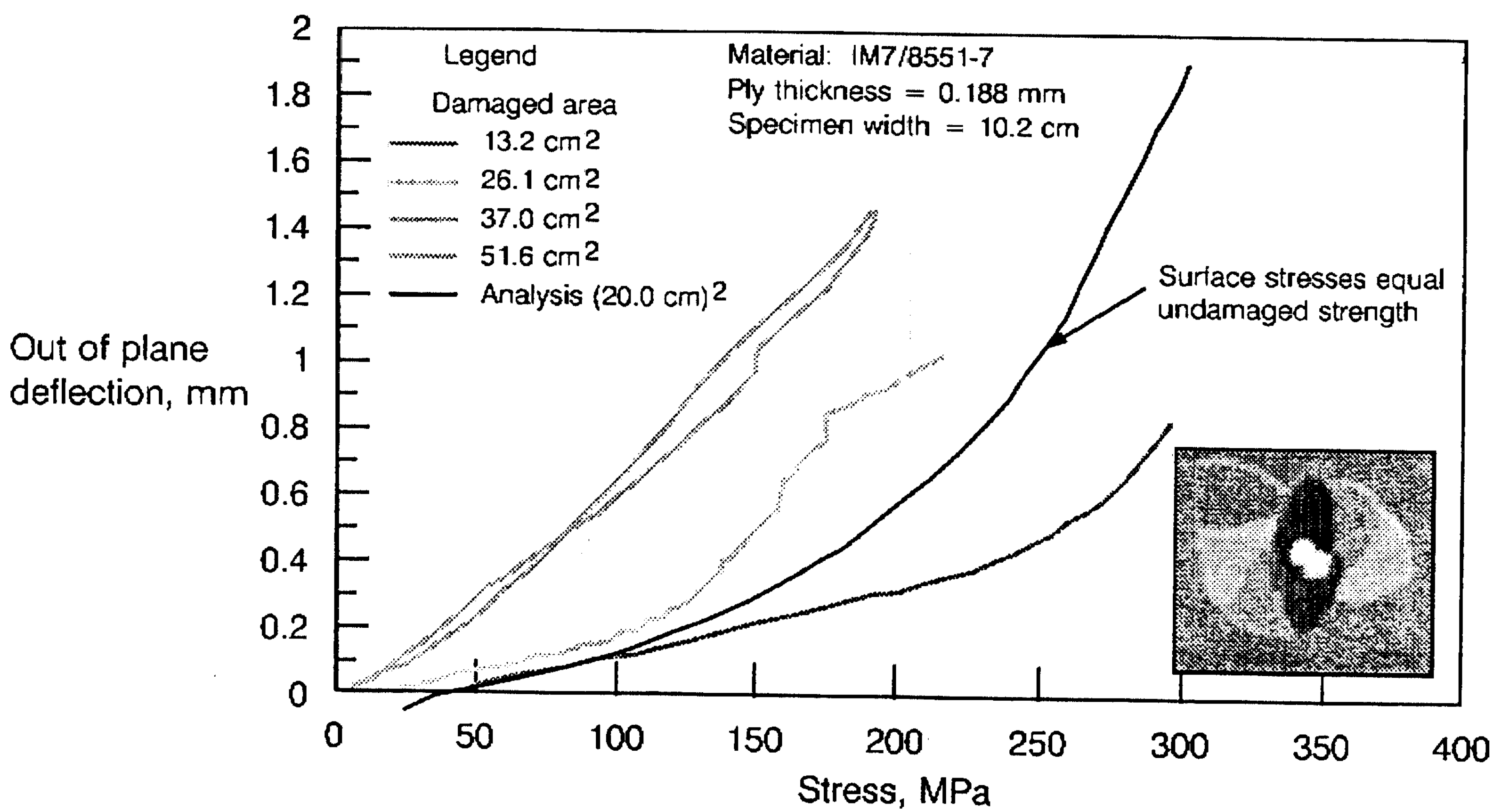
HL1568.10 R3mk

STAGSC-1 Finite Element Grid



HL1568.11

LVDT Measured Specimen Stability
Laminate Layup: $(45_3/90_3/-45_3/0_3)_S$



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D18238 08 DR1s

Summary

- Characteristic damage states (CDS) were classified for impacted laminates
- Local fiber failures and matrix damage were found to compete for dominance of the CDS in a toughened material with small damage
- Test data on compression after impact (CAI) performance versus drop weight impact energy indicated a strong effect of stacking sequence
- Delaminations in the CDS were determined using ultrasonics
- CAI for laminates with symmetric size distributions of matrix damage through the thickness was predicted with a sublaminate stability/stress redistribution analysis method
- Post-impact performance for laminates with unsymmetric size distributions of matrix damage through the thickness was best termed a change in specimen stability, requiring geometrically nonlinear analysis

HL1568.14 r1h

TOWARDS A METHODOLOGY FOR THE ASSESSMENT OF IMPACT OF COMPOSITE STRUCTURES

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GOAL

Develop a consistent methodology (philosophy)

to assess composite structures subjected to impact

TWO MAJOR ISSUES

Damage Resistance: Measure of the damage incurred by a material/structure due to a particular event.

Damage Tolerance: Measure of the ability of a material/structure to "perform" (given particular requirements) with damage present.

MODULARIZED APPROACH OVERVIEW

Damage Resistance

1. Global Analysis of Impact Event

↓

Loads on Structure

↓

2. Local Deformation and Strain Response Analysis

↓

Local Strain Field

↓

3. Failure Criteria

↓

Damage Predictions

↓

Damage Tolerance

4. Degraded Property Model

↓

Damaged Stress/Strain Field

↓

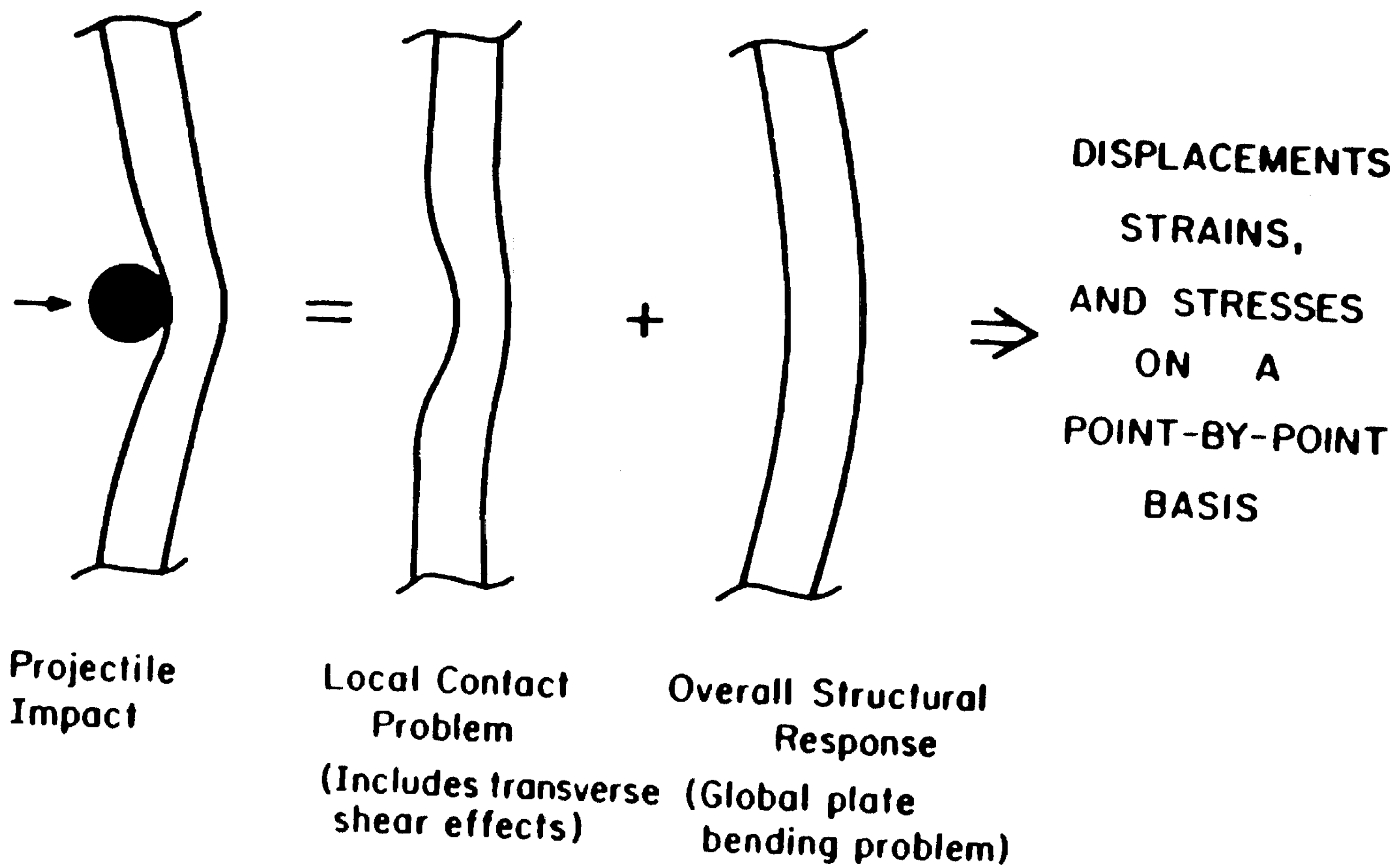
5. Failure Criteria for Component

↓

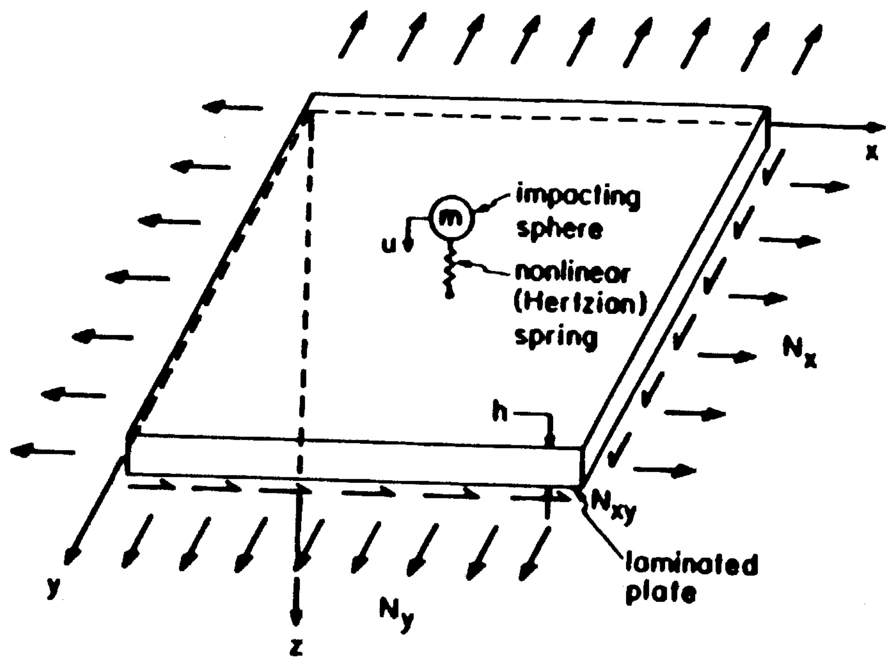
Performance Prediction

DAMAGE RESISTANCE

RESPONSE OF LAMINATE TO IMPACT DIVIDED INTO LOCAL AND GLOBAL RESPONSES



IMPACT EVENT GLOBAL MODEL

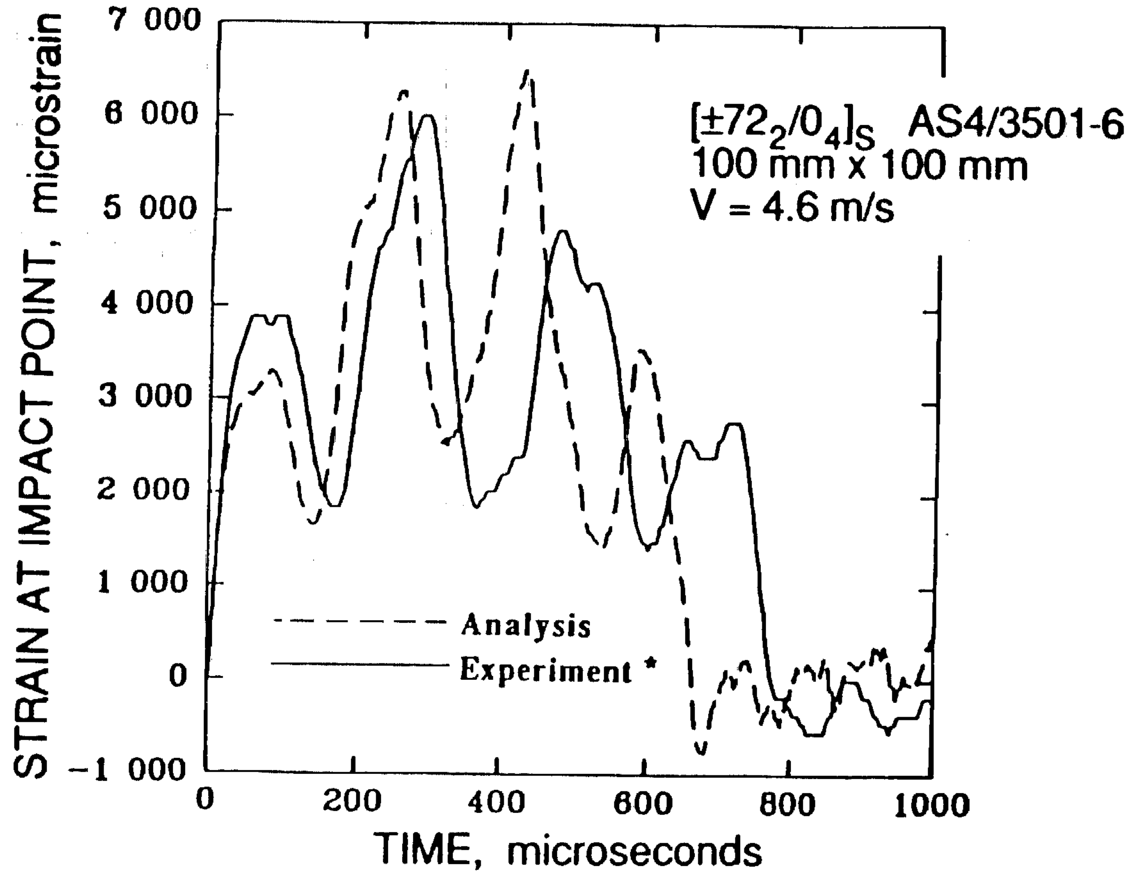


Assumed Modes Rayleigh - Ritz Analysis

- Reissner - Mindlin Plate Theory (shear deformation)
- Nonlinear Forcing Function (Hertzian Spring)
- Effect of In - Plane Loading (N_x, N_y, N_{xy})
- Bending - Twisting Coupling (D_{16}, D_{26} terms)

Newmark Beta implicit time marching scheme

GLOBAL DYNAMIC RESPONSE IS WELL-PREDICTED



* Y. Qian and S.R. Swanson, "Experimental Measurement of Impact Response in Carbon/Epoxy Plates", Proceedings of the 30th AIAA/ASME/ASCE/AHS Structures, Structural Dynamics and Materials Conference, 1989.

MODULARIZED APPROACH OVERVIEW

Damage Resistance

1. Global Analysis of Impact Event

↓

Loads on Structure

↓

2. **Local Deformation and Strain Response Analysis**

↓

Local Strain Field

↓

3. Failure Criteria

↓

Damage Predictions

↓

Damage Tolerance

4. Degraded Property Model

↓

Damaged Stress/Strain Field

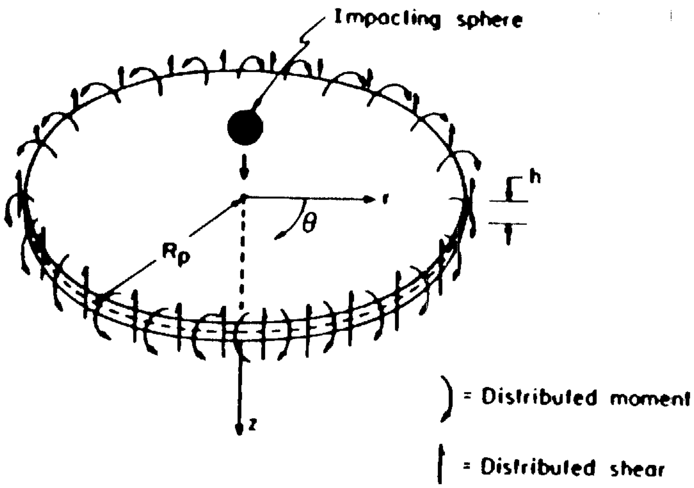
↓

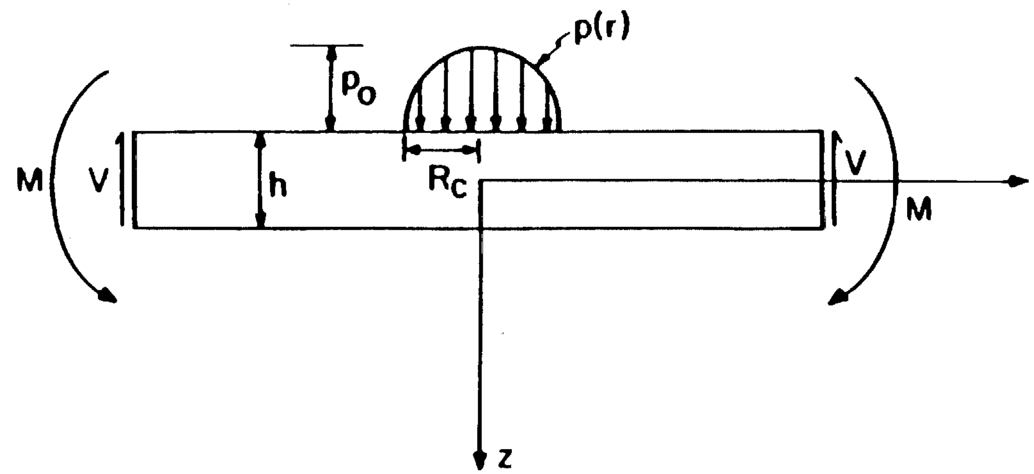
5. Failure Criteria for Component

↓

Performance Prediction

LOCAL CONTACT PROBLEM SOLUTION APPROACH

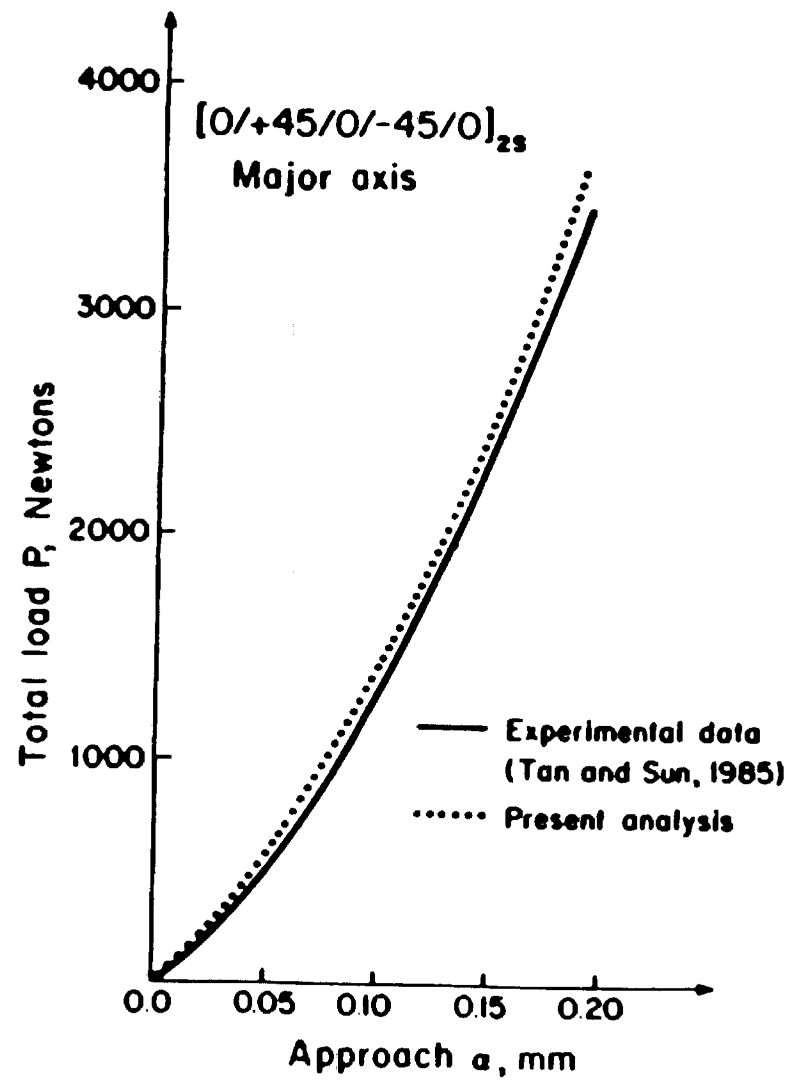




- Constitutive properties smeared through-the-thickness
- Axisymmetry assumed
- D'Alembert inertial terms included in z-direction
- Stress function utilized
- Contact and body forces expanded as Fourier-Bessel series

Resulting strains rotated into ply principal axes

LOCAL STATIC INDENTATION RESPONSE IS WELL-PREDICTED



MODULARIZED APPROACH OVERVIEW

Damage Resistance

1. Global Analysis of Impact Event

↓

Loads on Structure

↓

2. Local Deformation and Strain Response Analysis

↓

Local Strain Field

↓

3. Failure Criteria

↓

Damage Predictions

↓

Damage Tolerance

4. Degraded Property Model

↓

Damaged Stress/Strain Field

↓

5. Failure Criteria for Component

↓

Performance Prediction

DAMAGE PREDICTIONS

- Applied on a ply-by-ply-basis
- Maximum Strain Criterion:

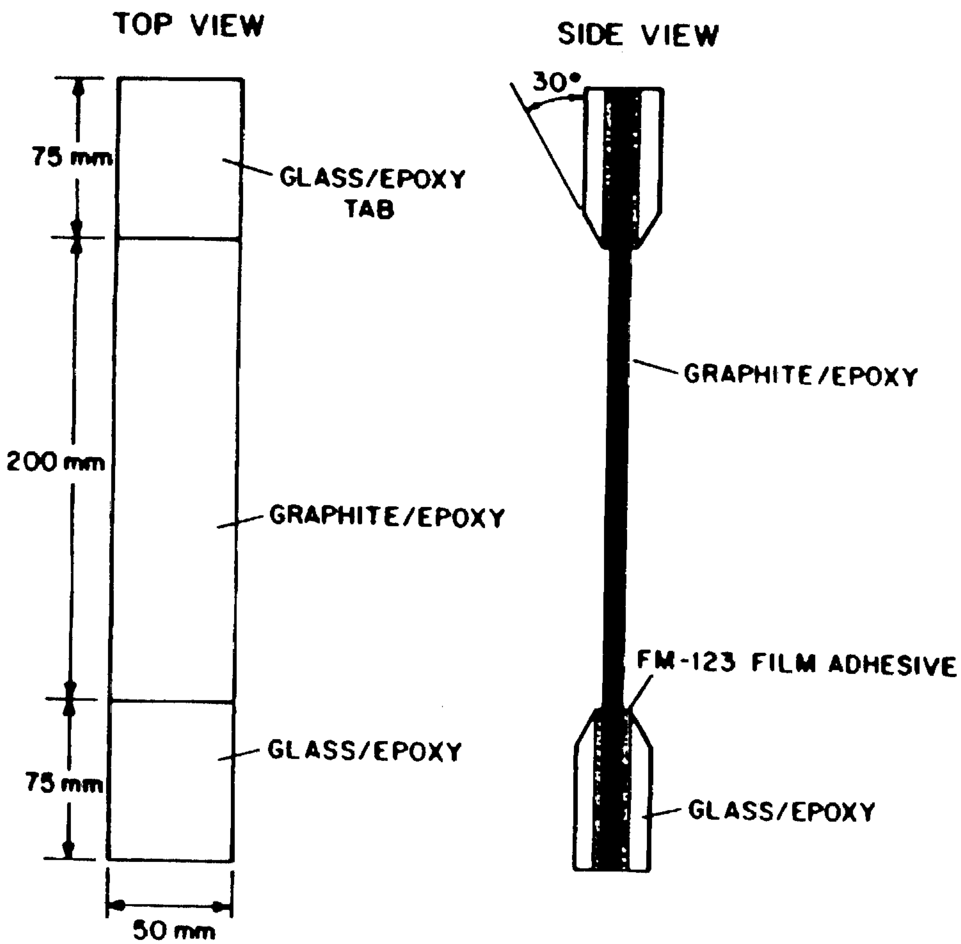
$$\frac{\varepsilon_{11}}{\varepsilon_{11ult}} > 1 \qquad \frac{\gamma_{12}}{\gamma_{12ult}} > 1$$

$$\frac{\varepsilon_{22}}{\varepsilon_{22ult}} > 1 \qquad \frac{\gamma_{13}}{\gamma_{13ult}} > 1$$

$$\frac{\varepsilon_{33}}{\varepsilon_{33ult}} > 1 \qquad \frac{\gamma_{23}}{\gamma_{23ult}} > 1$$

- Mode of failure indicated

SPECIFIC EXAMPLE



Layup

- AS4/3501-6 Graphite / Epoxy
- $[\pm 45/0]_{2S}$

Geometry

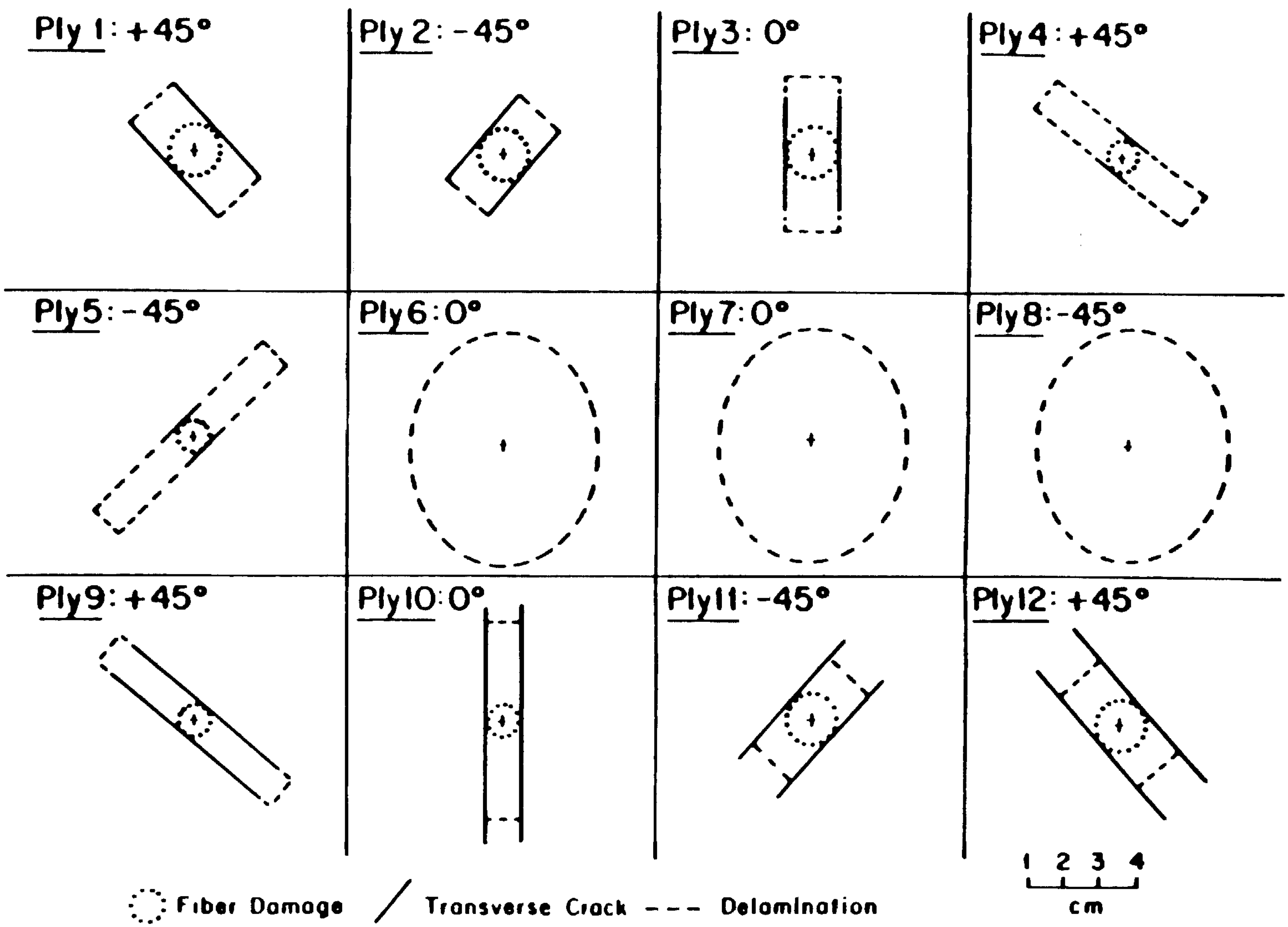
- Length = 190 mm
- Width = 70 mm
- Clamped (x) - Free (y)

Impact

- 12.7 mm diameter steel sphere
- Impact Energy = 8 Joules
- Impact Speed = 43 m/s

PLY-BY-PLY DAMAGE PREDICTIONS

$[\pm 45/0]_{2S}$



NDE RESULTS

$[\pm 45/0]_{2S}$ AS4/3501-6 Graphite/Epoxy
190 mm x 70 mm Clamped (x) - Free (y)
12.7 mm diameter steel sphere
at 42.8 m/s impact speed (7.95 Joules)

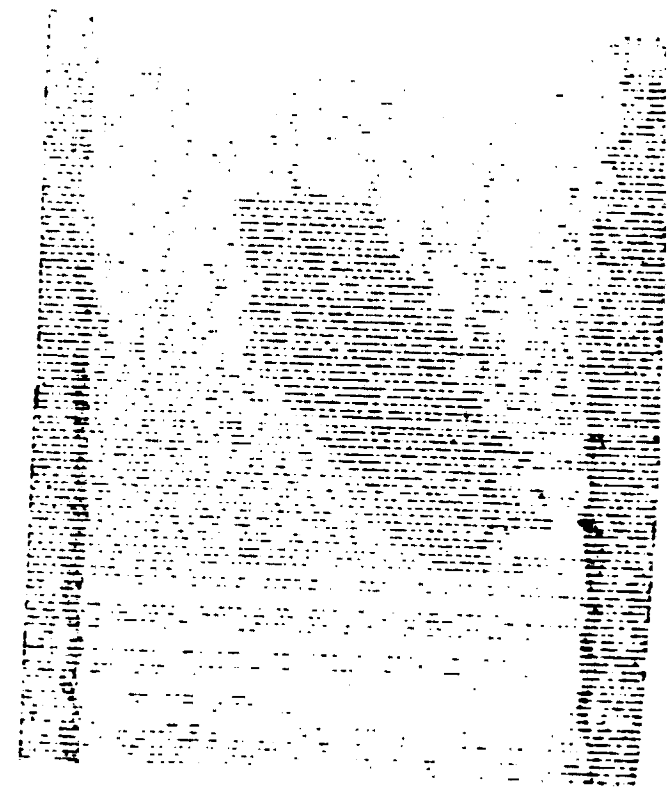
X-Ray Photograph
(DIB enhanced)

Ultrasonic C-Scan
(pulse-echo)

NDE RESULTS

$[\pm 45/0]_{2S}$ AS4/3501-6 Graphite/Epoxy
190 mm x 70 mm Clamped (x) - Free (y)
12.7 mm diameter steel sphere
at 42.8 m/s impact speed (7.95 Joules)

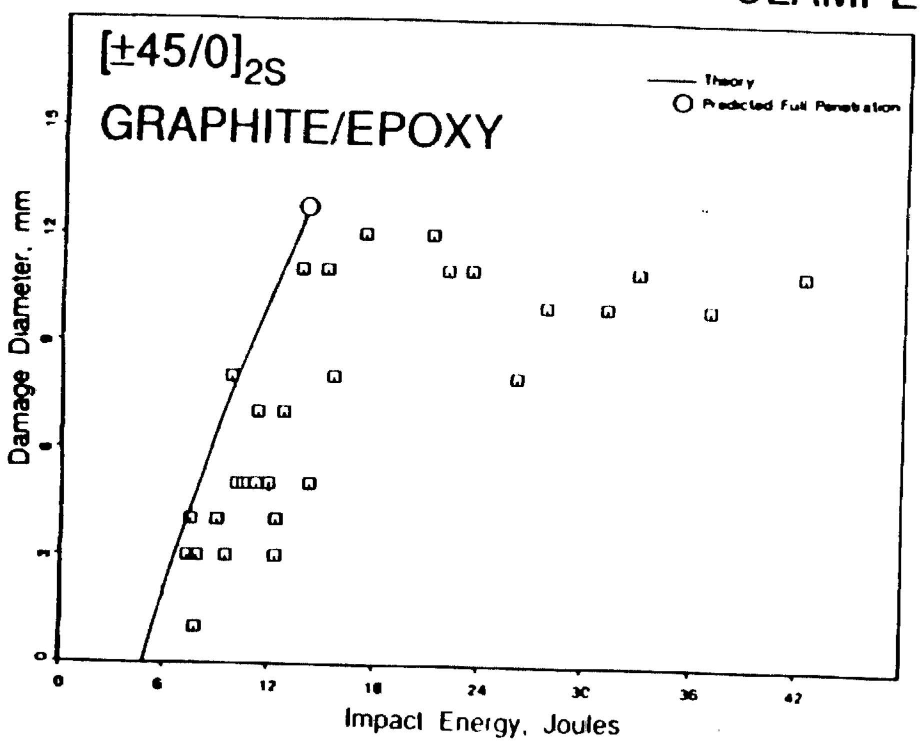
X-Ray Photograph
(DIB enhanced)

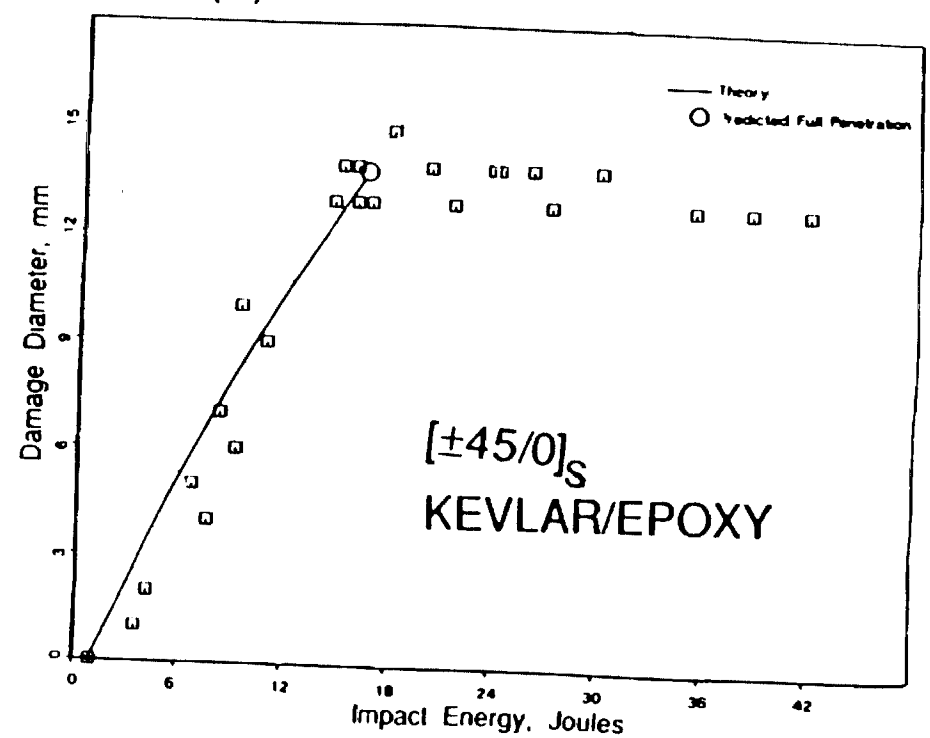


Ultrasonic C-Scan
(pulse-echo)

INTEGRATED DAMAGE IS WELL-PREDICTED

CLAMPED(X) - FREE(Y)





IMPORTANT FACTORS IN DAMAGE RESISTANCE

- It is necessary to treat impact at both the local and global levels *(they can be separated)*
- Parameters have influence in different regimes
  - lower velocity: boundary conditions important
  - higher velocity: constituent mass plays a dominant role

  response length versus span is key
- Impactor energy is not the only controlling factor

  mass and velocity must be specified
- Presence of damage can affect local response

  ability of surface to conform to indentor is important
- Preload does affect damage resistance

MATERIAL INFLUENCE ON DAMAGE RESISTANCE

- Mass and layup affect structural properties, inertia, and thus the impact event

- Basic material strengths control resulting damage near impactor

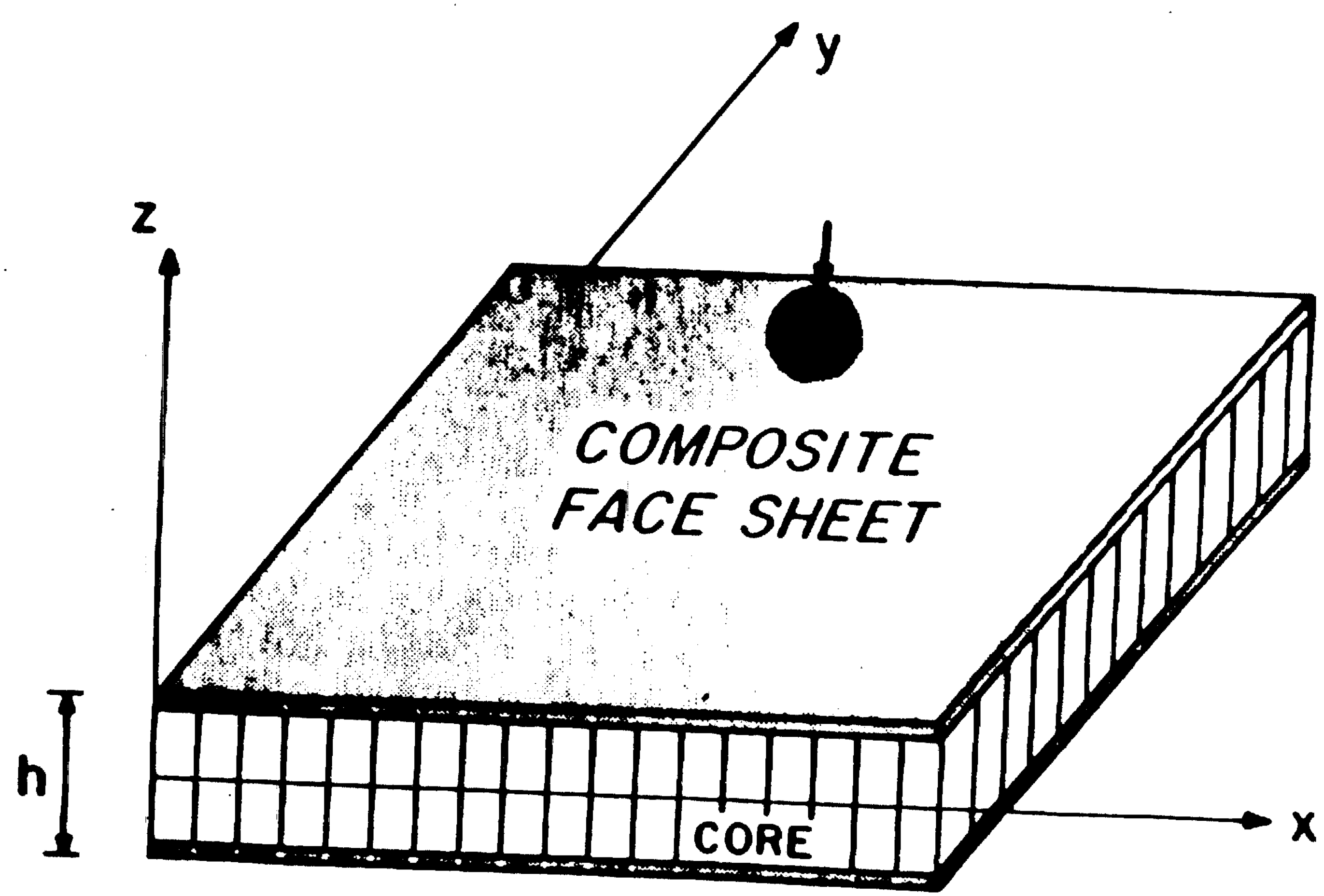
Damage resistance is a combined material/structural property

NEEDS

DAMAGE RESISTANCE

- Include more details in local contact stiffness (material nonlinearities and damage)
- Include high strain rate material properties (especially for high strain-to-failure systems)
- Incorporate influence of preload
- Acquire better failure criteria
- Determine need to do progressive damage analysis
- Extend general methodology to more complicated structure (once modes of structures are known, structure can be analyzed in this manner)
- Use analysis to do proper scaling

METHODOLOGY CAN BE APPLIED TO SANDWICH CONSTRUCTION



ALL DAMAGE POSSIBILITIES MUST BE CONSIDERED

- Facesheet damage
  - front
  - back
- Core damage
- Debond damage
- Damage combinations

DAMAGE

TOLERANCE

MODULARIZED APPROACH OVERVIEW

Damage Resistance

1. Global Analysis of Impact Event

↓

Loads on Structure

↓

2. Local Deformation and Strain Response Analysis

↓

Local Strain Field

↓

3. Failure Criteria

↓

Damage Predictions

↓

Damage Tolerance

4. Degraded Property Model

↓

Damaged Stress/Strain Field

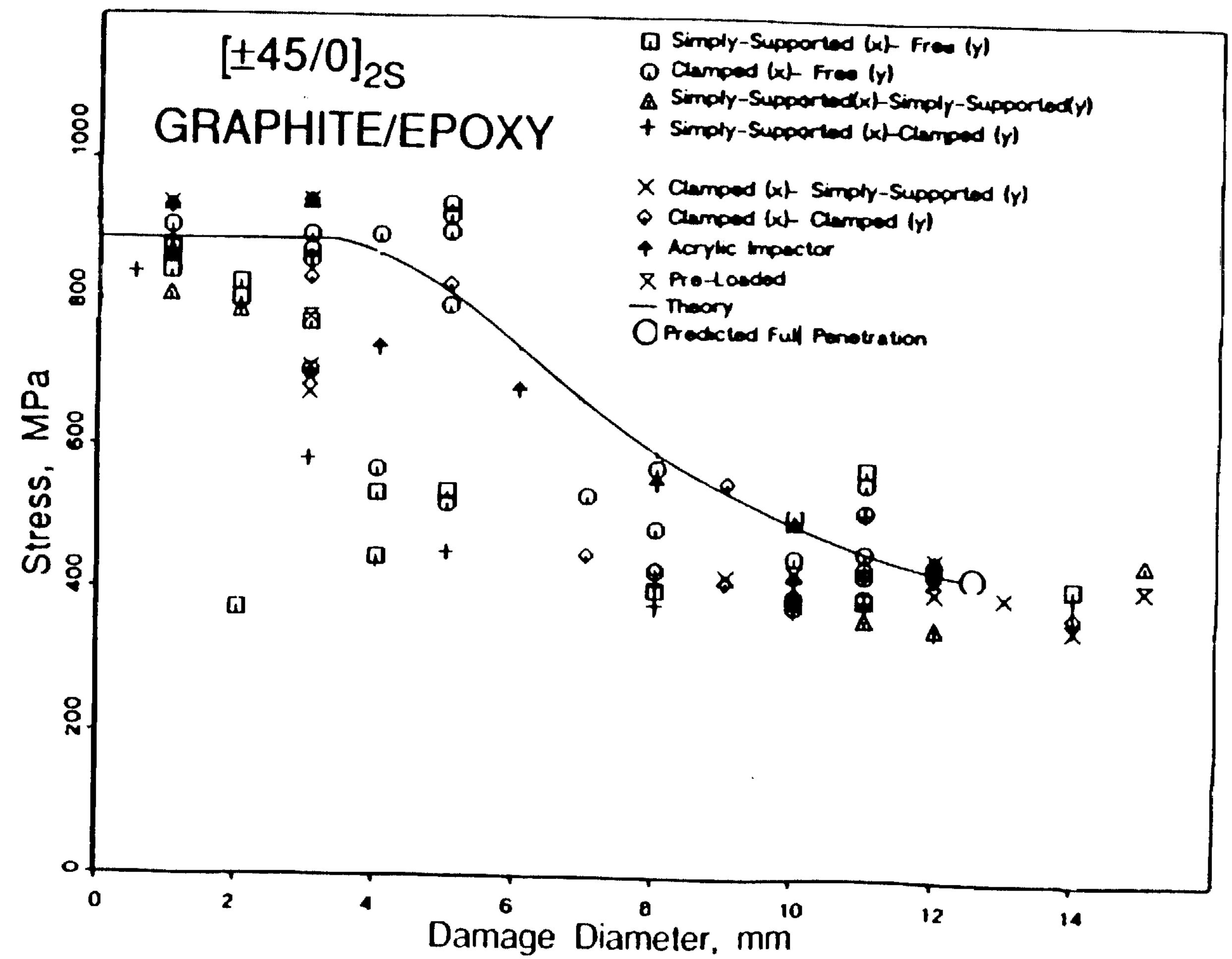
↓

5. Failure Criteria for Component

↓

Performance Prediction

POST-IMPACT RESIDUAL STRENGTH IS A FUNCTION OF DAMAGE PRESENT



FRACTURE OCCURS AT TWO LEVELS

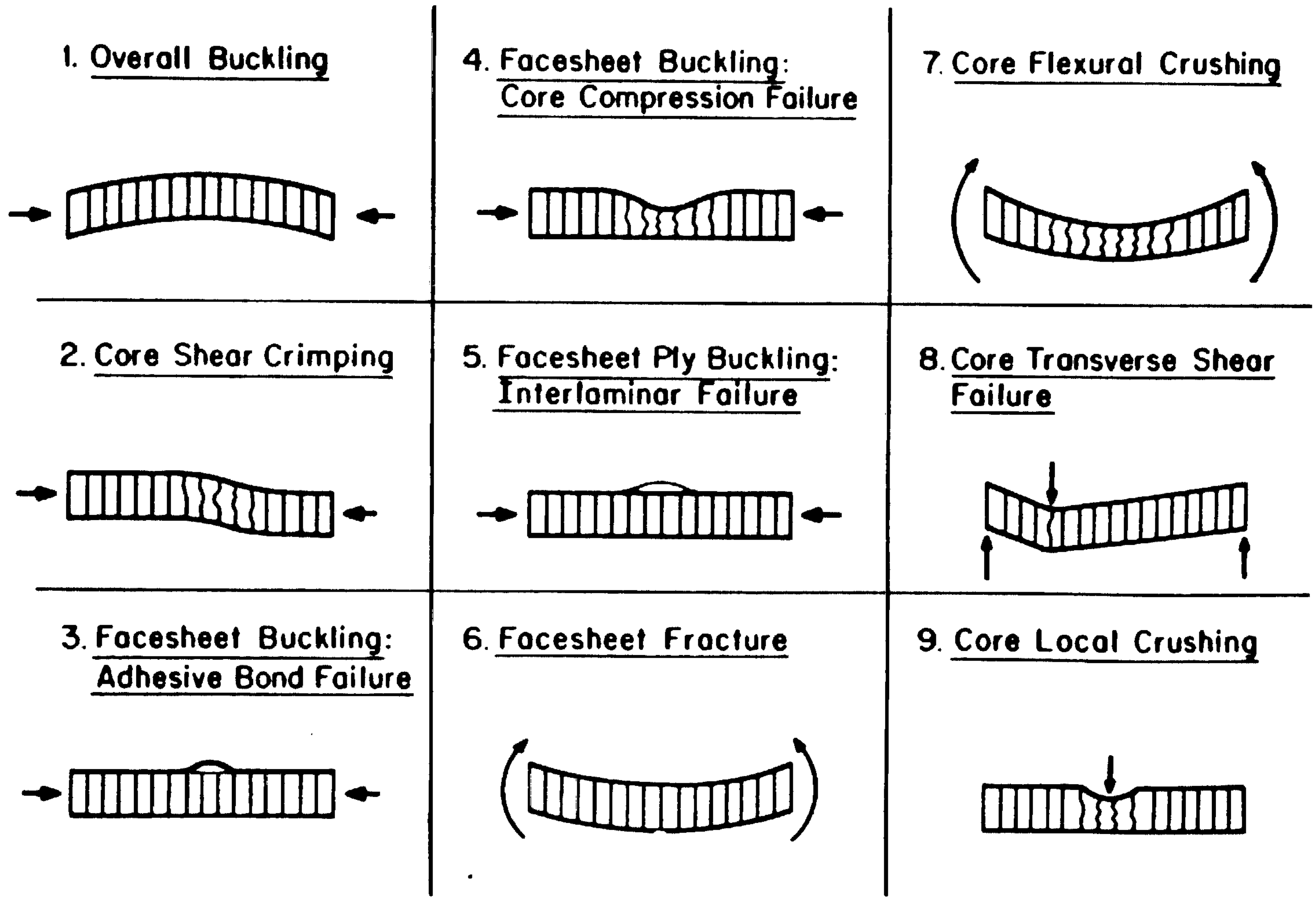
- **Global level** *(structural dominated)*

  Failure as a result of structural mechanism (especially under compression)

- **Local level** *(material property)*

  Failure initiates from local fracture

SANDWICH CONSTRUCTION INTRODUCES MANY POTENTIAL FAILURE MODES



DIFFERENT ANALYSES NEEDED FOR VARIOUS LOADING TYPES/POTENTIAL FAILURE MODES

- Many different types of damage occur
  - transverse cracks
  - delaminations
  - fiber fracture
  - others (just beginning to be defined)
- Many different modes of failure/fracture possible
  - depends on geometrical configuration
  - depends on loading
- Each damage type and mode requires consideration
- Requires medium mechanics (ply level) or micromechanics approach

TENSILE

RESIDUAL STRENGTH

MODULARIZED APPROACH OVERVIEW

Damage Resistance

1. Global Analysis of Impact Event

↓

Loads on Structure

↓

2. Local Deformation and Strain Response Analysis

↓

Local Strain Field

↓

3. Failure Criteria

↓

Damage Predictions

↓

4. Degraded Property Model

↓

Damaged Stress/Strain Field

↓

5. Failure Criteria for Component

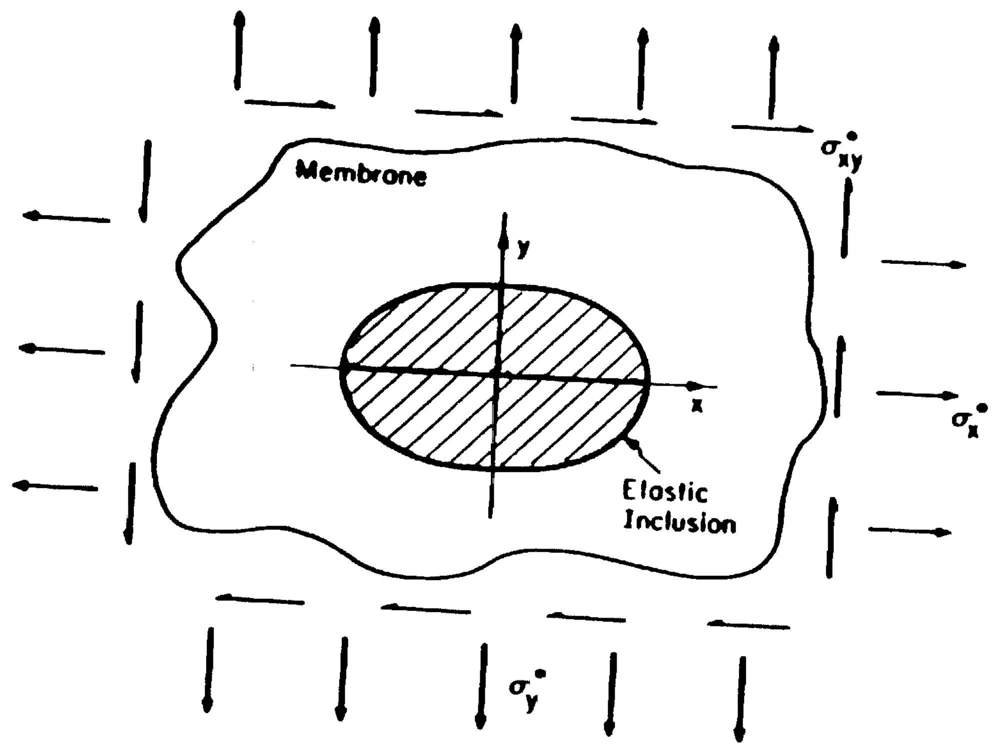
↓

Performance Prediction

Damage Tolerance

DAMAGED LAMINATE STRAIN ANALYSIS

EQUIVALENT MEMBRANE MODEL
(anisotropic inclusion)



Model Based on Lekhnitskii's Complex Potentials

DEGRADATION ASSUMPTIONS

- Delaminations and isolated angle ply splits do not cause significant degradation under tensile loading

- Only fiber breakage creates a significant reduction in constitutive properties

- All elastic constants set to zero in region of fiber breakage

MODULARIZED APPROACH OVERVIEW

Damage Resistance

1. Global Analysis of Impact Event

↓

Loads on Structure

↓

2. Local Deformation and Strain Response Analysis

↓

Local Strain Field

↓

3. Failure Criteria

↓

Damage Predictions

↓

Damage Tolerance

4. Degraded Property Model

↓

Damaged Stress/Strain Field

↓

5. Failure Criteria for Component

↓

Performance Prediction

RESIDUAL STRENGTH PREDICTION FOR IN-PLANE FRACTURE

- Utilize strain from Equivalent Membrane Model
- Average strain concept applied on laminate basis
- Maximum 0° fiber strain assumed to control failure

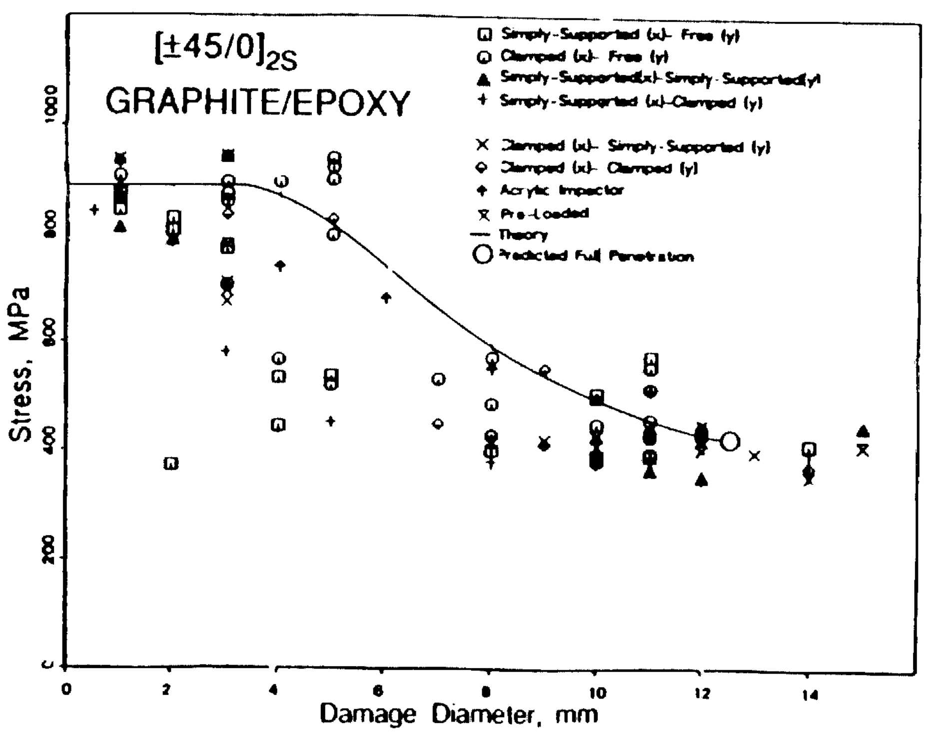
$$S.R. = \frac{\bar{\varepsilon}_x^{o}}{\frac{1}{a_o}\int_0^{a_o} \bar{\varepsilon}_x(x,y)\, dr}$$

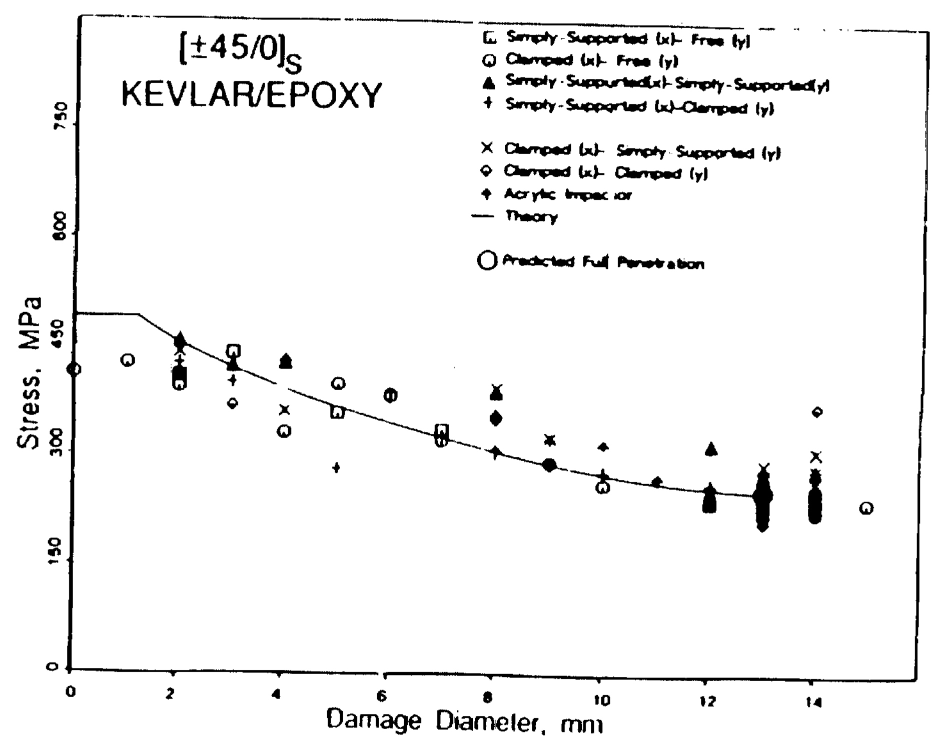
where $\bar{\varepsilon}_x^{o}$ = far field laminate strain

$\bar{\varepsilon}_x(x,y)$ = strain distribution

a_o = material parameter (3.8 mm here)

IN-PLANE FRACTURE IS REASONABLY WELL-PREDICTED





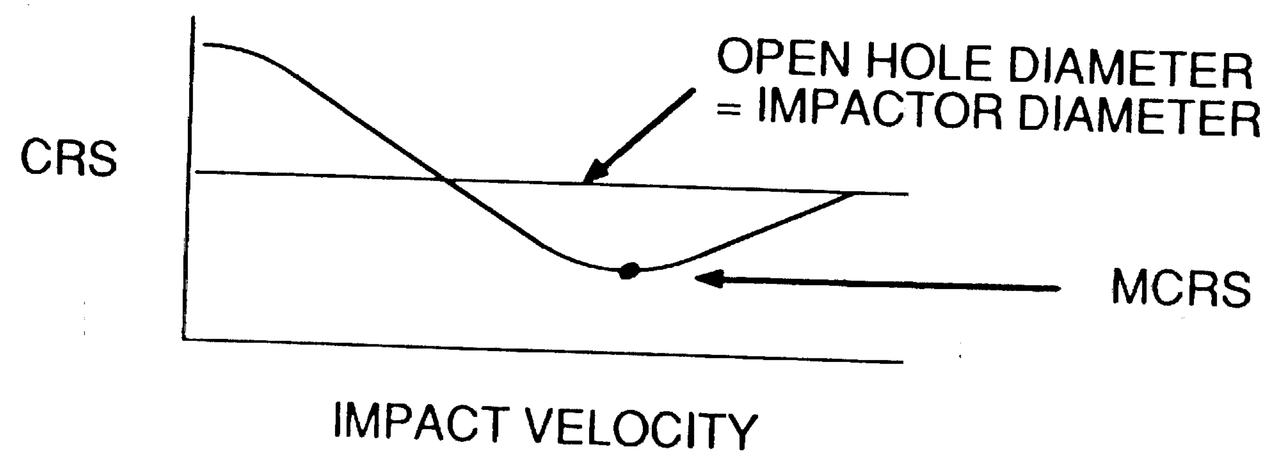
COMPRESSIVE

RESIDUAL STRENGTH

PREDICTION METHODOLOGIES FOR COMPRESSIVE RESIDUAL STRENGTH

- A significant amount of work has been published on the compressive residual strength of notches, holes, and imbedded delaminations.
- Very little has been published on multiple imbedded delaminations or combined damage (i.e. impact).
- The effect of impact on compressive residual strength cannot be estimated by an "equivalent-sized" notch, hole, imbedded delamination, or some combination thereof.
- Two-dimensional planform damage information is not sufficient to predict compressive residual strength due to impact.

A MINIMUM COMPRESSIVE RESIDUAL STRENGTH (due to impact) EXISTS



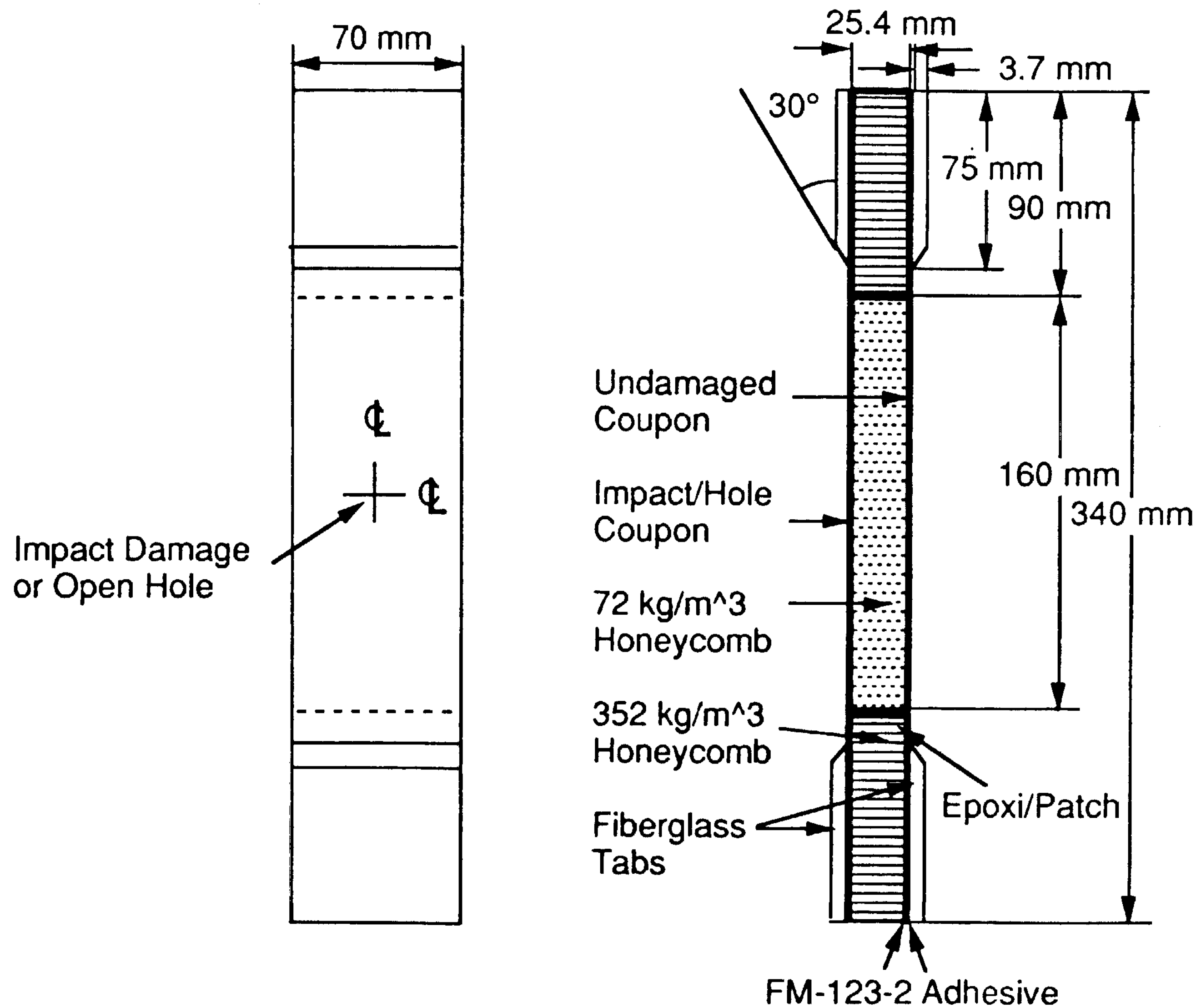
What damage state governs this minimum point?

Is this minimum point a laminate characteristic or is it dependent upon impact method?

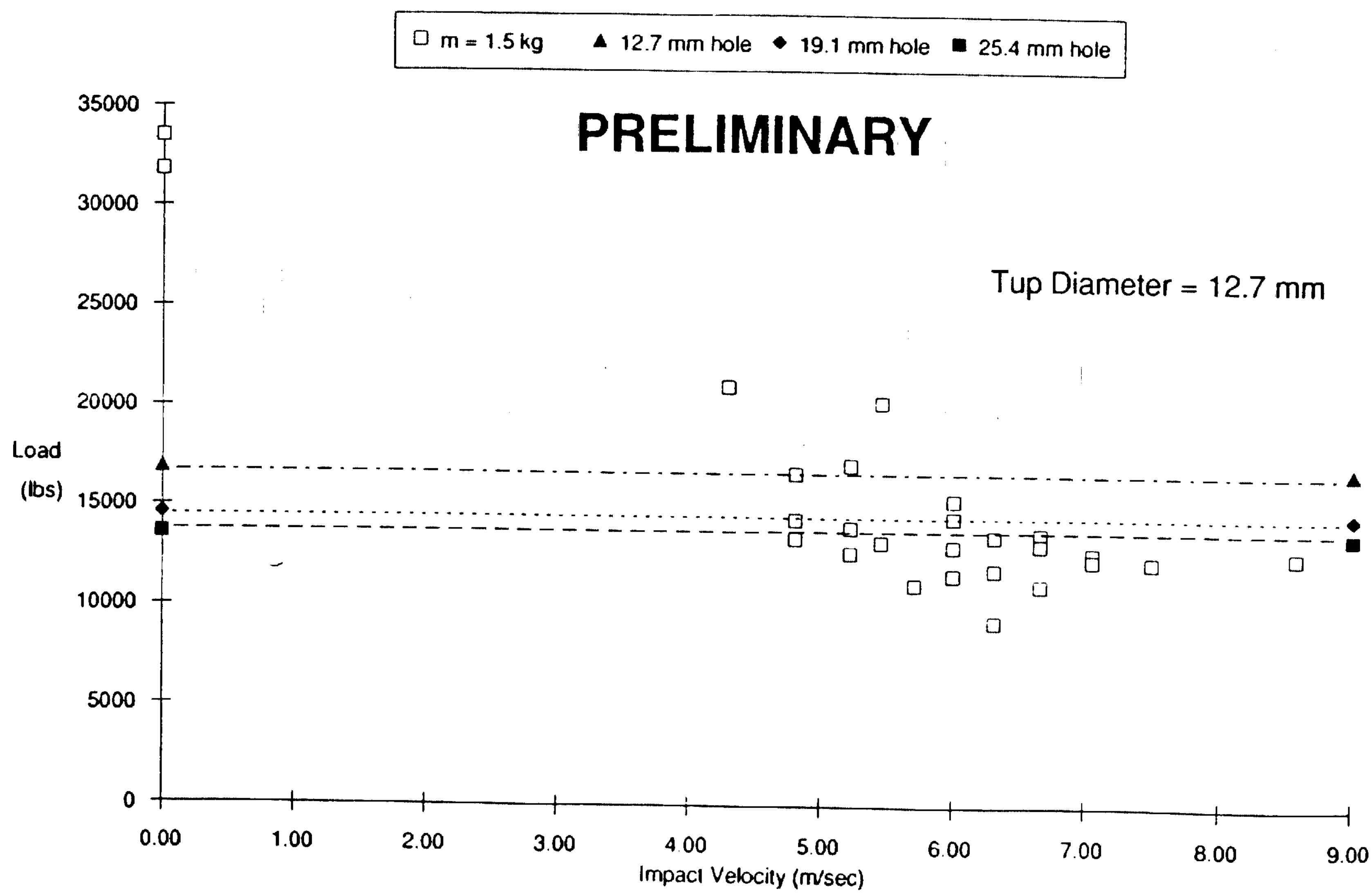
GENERAL TEST PROCEDURES

- $[\pm45/0]_{2S}$ laminate of AS4/3501-6 graphite/epoxy
- 12.7 mm diameter spherical impactor
- Impact coupon and inspect damage
- Bond damaged coupon and an undamaged coupon to aluminum honeycomb base
- Test configuration quasi-statically in compression

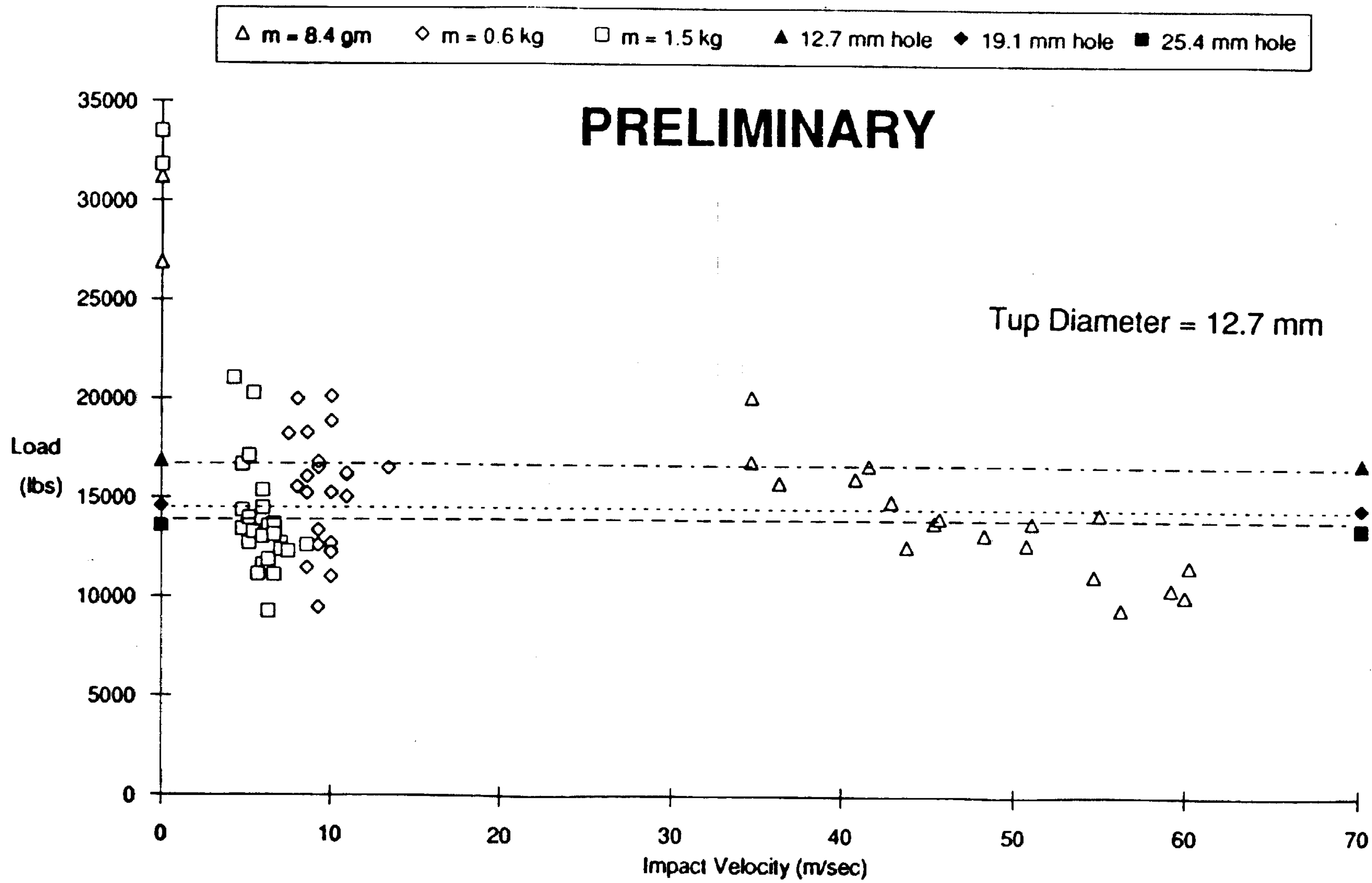
COMPRESSION TEST SPECIMEN



TEST DATA SHOWS MINIMUM EXISTS



MINIMUM APPEARS TO BE INDEPENDENT OF IMPACTOR MASS

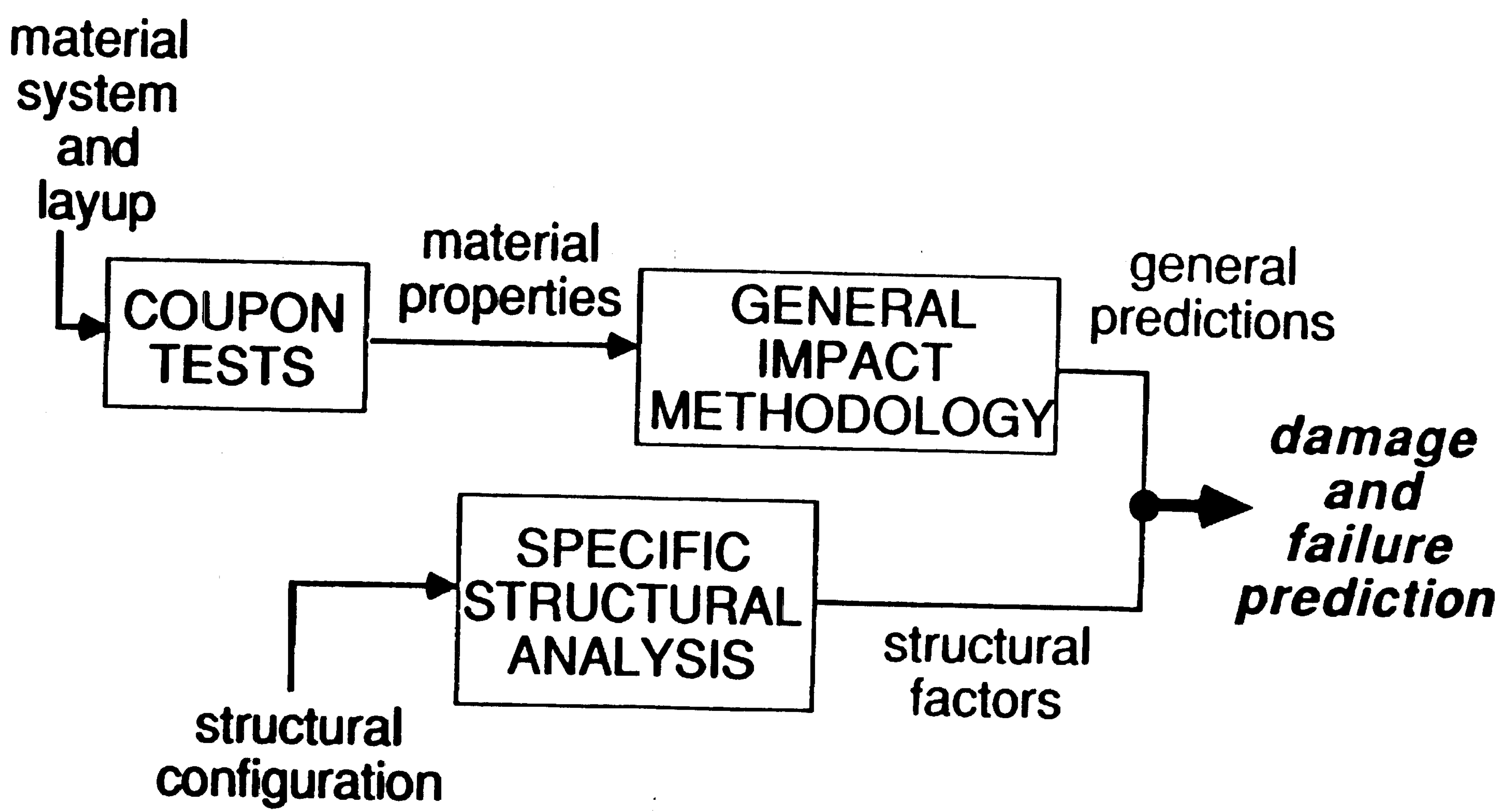


NEEDS

DAMAGE TOLERANCE

- Develop progressive damage model for in-plane failure
- Model "actual" damage
- Develop a repertoire of damage tolerance tools for each failure mode (especially structural failure)

EXTEND METHODOLOGY TO STRUCTURAL CONFIGURATIONS



SUMMARY

- A modularized framework and consistent approach for the treatment of composite structures subjected to impact have been developed
- Some refinement of impact analysis is needed
- Integrated through-the-thickness representations (planform) of the damage are generally inappropriate for residual strength predictions
- Modelling the actual (3-D) damage distribution is essential
- A minimum compressive residual strength exists which may be independent of impactor mass

SUMMARY (cont.)

- Development of damage "metrics" is needed
- Need to develop a repetoire of tools for residual strength assessment
- Development of better failure criteria is key in damage resistance and damage tolerance assessments

Langley-10b

STANDARD IMPACT TESTS USED AT LOCKHEED

C. F. GRIFFIN
T. GILLETTE

NASA WORKSHOP ON IMPACT DAMAGE TO COMPOSITES
MARCH 19-20, 1991
LOCKHEED AERONAUTICAL SYSTEMS COMPANY

AGENDA

- LAMINATE IMPACT TESTS
- SUBCOMPONENT IMPACT TESTS
- COMPONENT IMPACT TESTS

LAMINATE IMPACT TESTS

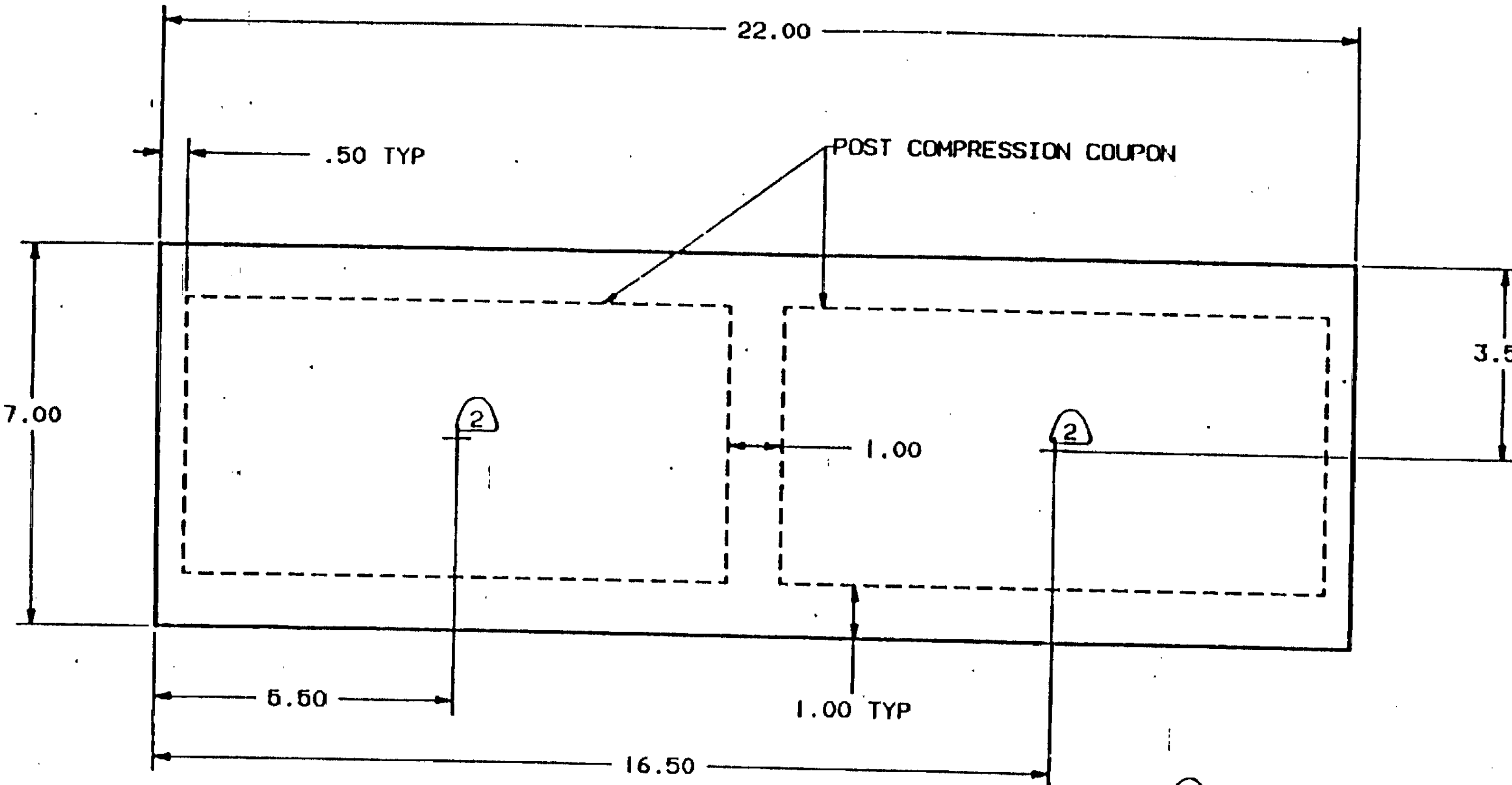
PURPOSE:

- QUANTITATIVE EVALUATION OF THE DAMAGE TOLERANCE OF COMPOSITE MATERIALS
- DATA OFTEN USED TO ESTABLISH PRELIMINARY DESIGN STRENGTH

DATA OBTAINED:

- IMPACT DAMAGE CHARACTERISTICS VERSUS IMPACT ENERGY
- COMPRESSION STRENGTH AFTER IMPACT

IMPACT COMPRESSION SUBPANEL



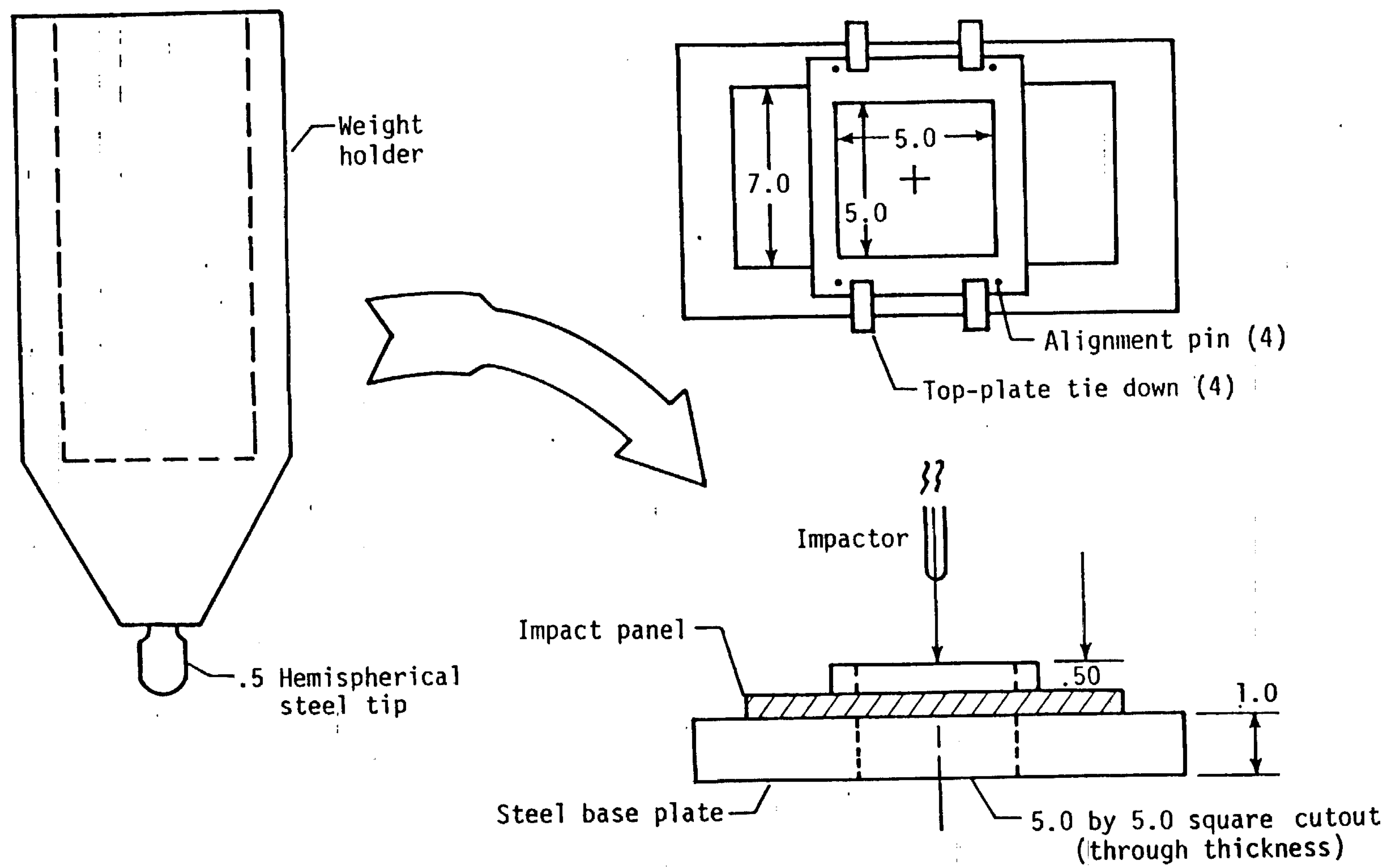
2 IMPACT LOCATIONS.

1 ALL DIMENSIOMS ARE IN INCHES.

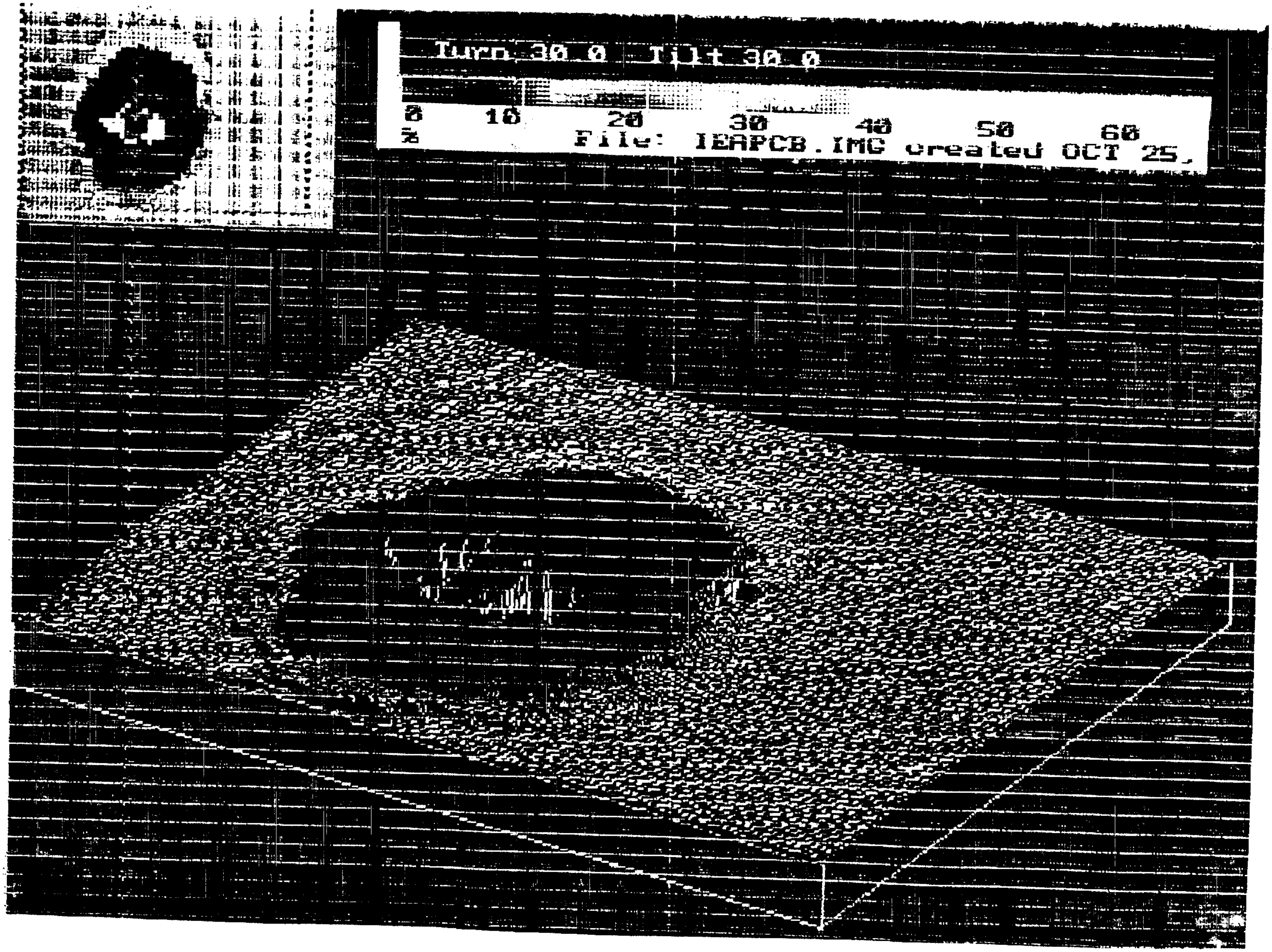
NOTES:

C-7
IMPACT COMPRESSION SUBPANEL

COUPON LEVEL IMPACT TEST APPARATUS



PLY LEVEL IMPACT DAMAGE C-SCAN ANALYSIS



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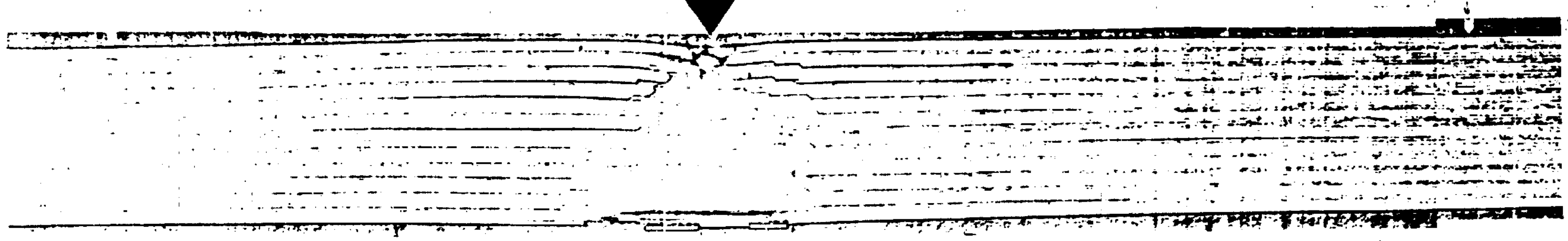
INTERNAL DAMAGE DUE TO IMPACT

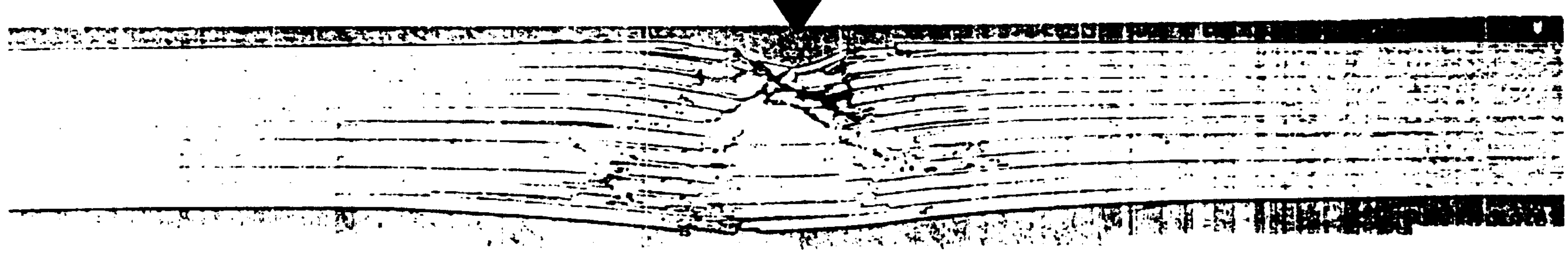
HIGH STRAIN CELION/974 48 PLY QUASI-ISOTROPIC LAMINATE
1/2 IN. DIA. IMPACTOR

40 FT LB IMPACT



10X

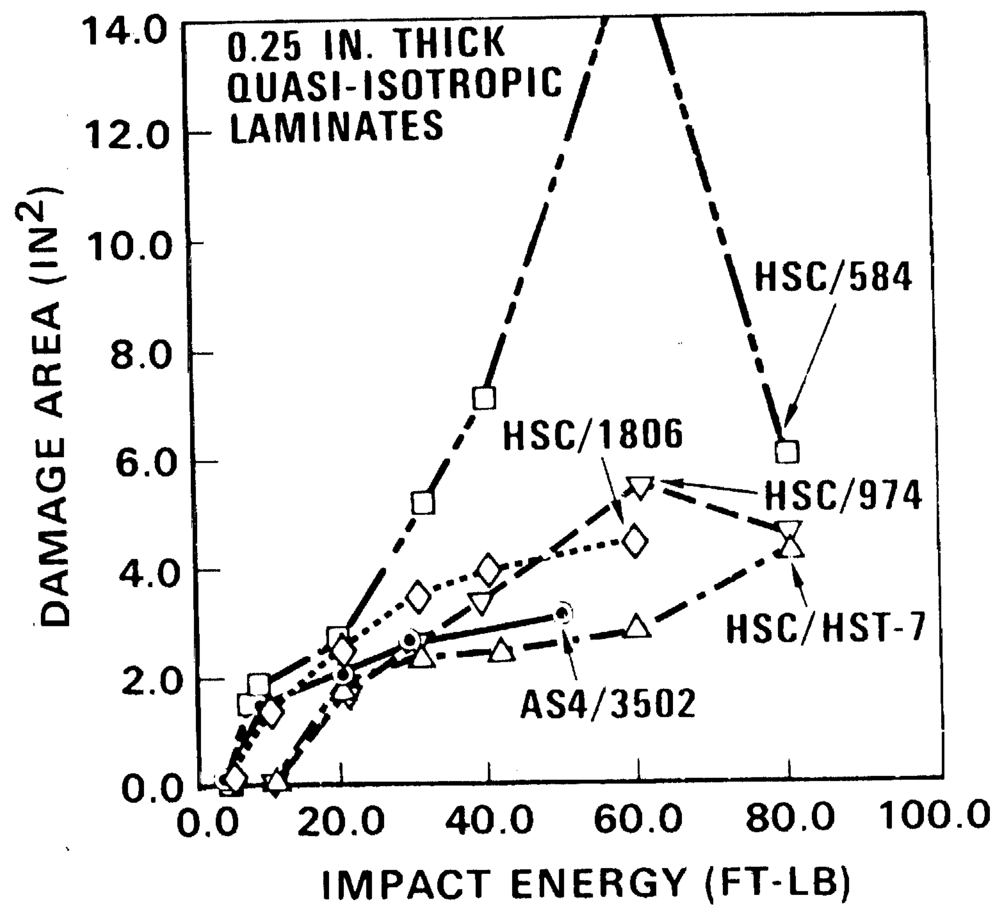
60 FT LB IMPACT



10X

CFG-14

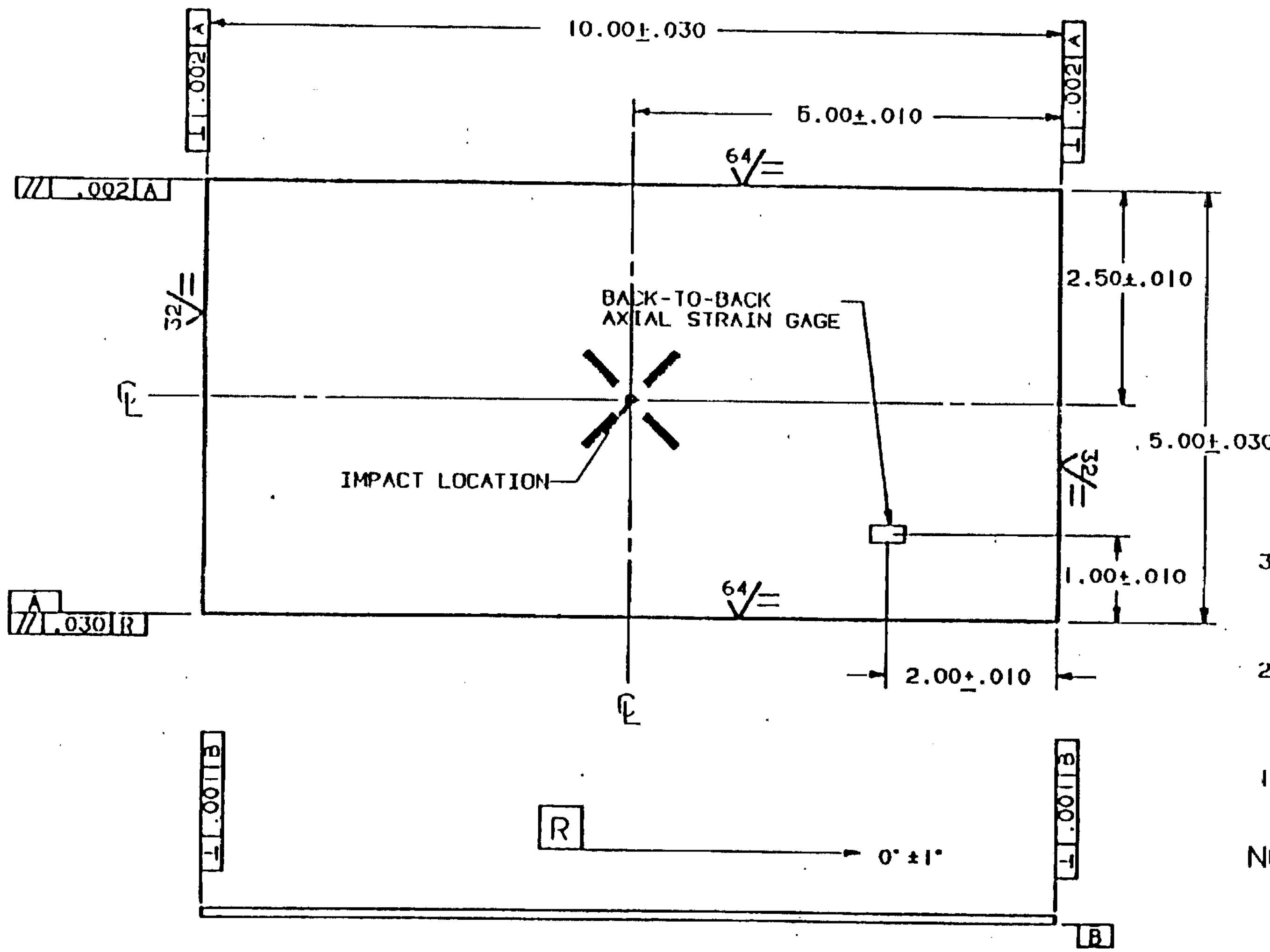
IMPACT RESPONSE — RESIN EFFECTS



CFG-13

7

LOCKHEED STANDARD COMPRESSION AFTER IMPACT COUPON 5 IN. X 10 IN.



NOTES:

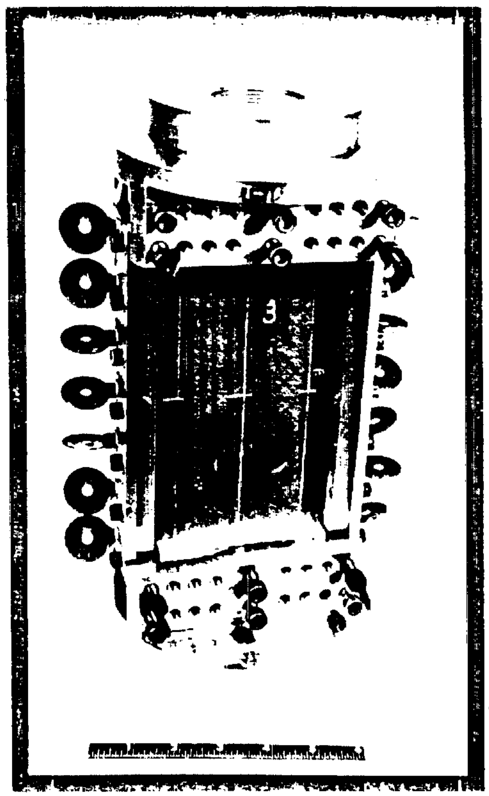
1 ALL DIMENSIONS ARE IN INCH

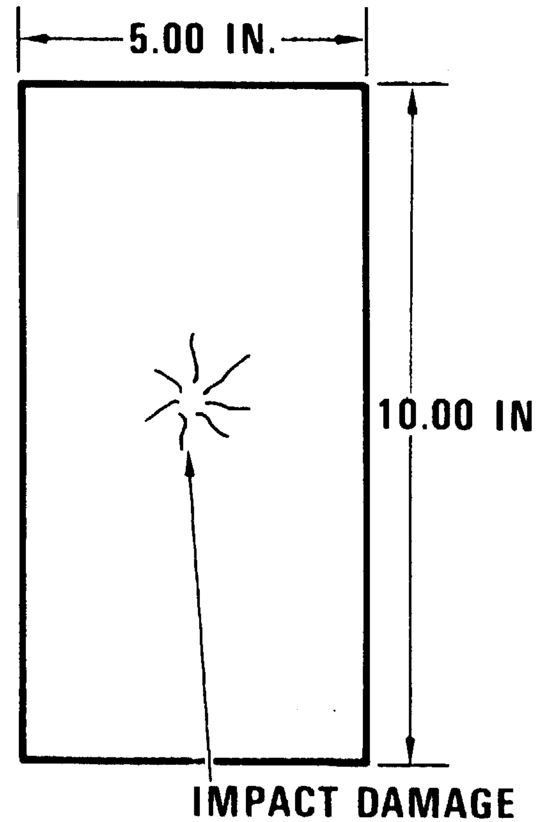
2 NOMINAL THICKNESS IS .250 IN. (REF).

3 EDGES TO BE SQUARE WITHOUT BURRS. DO NOT BREAK EDGES.

C-8
IMPACTED COMPRESSION COUPON

COMPRESSION TEST FIXTURE AND SPECIMEN





CFG-16

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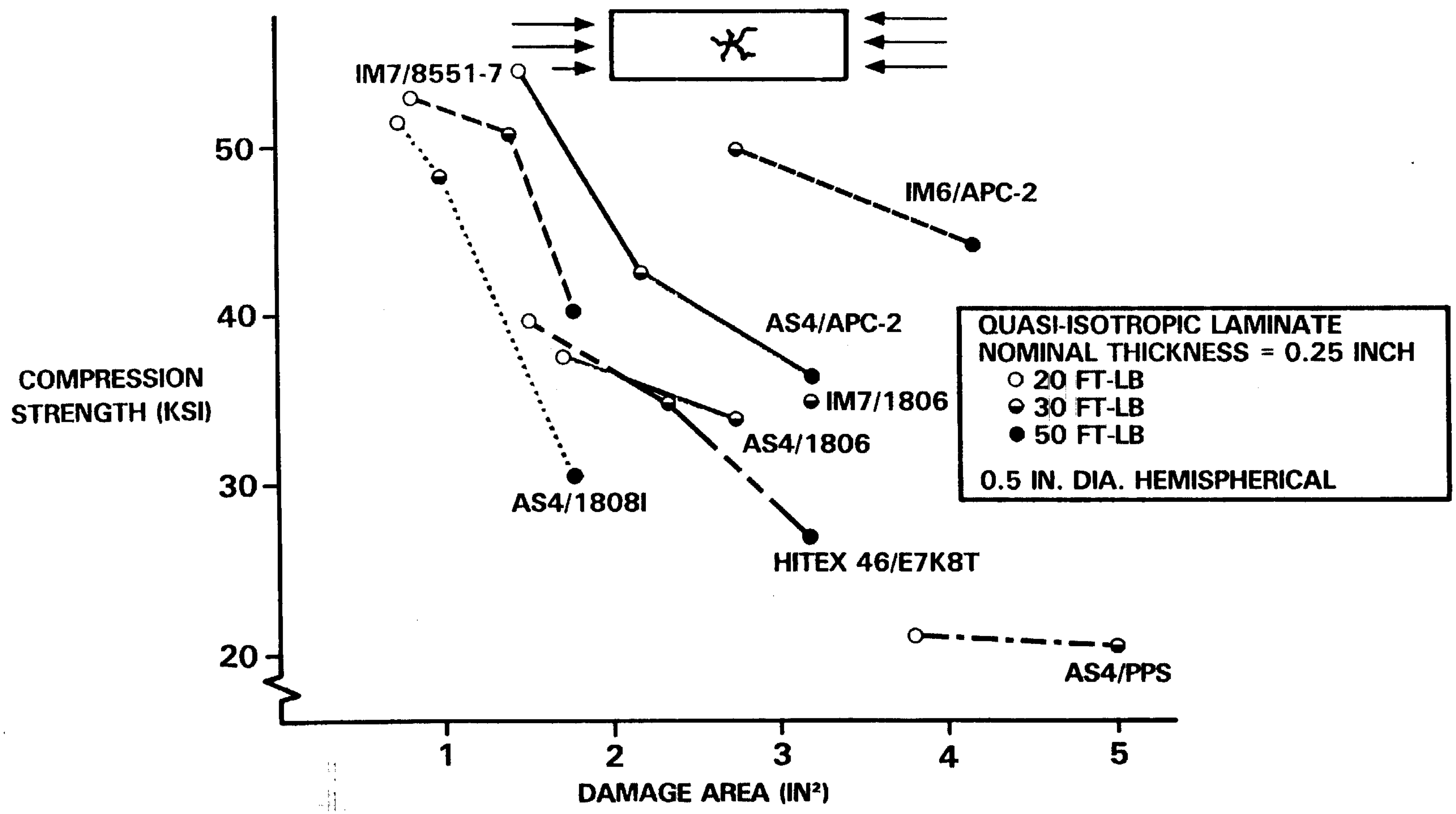
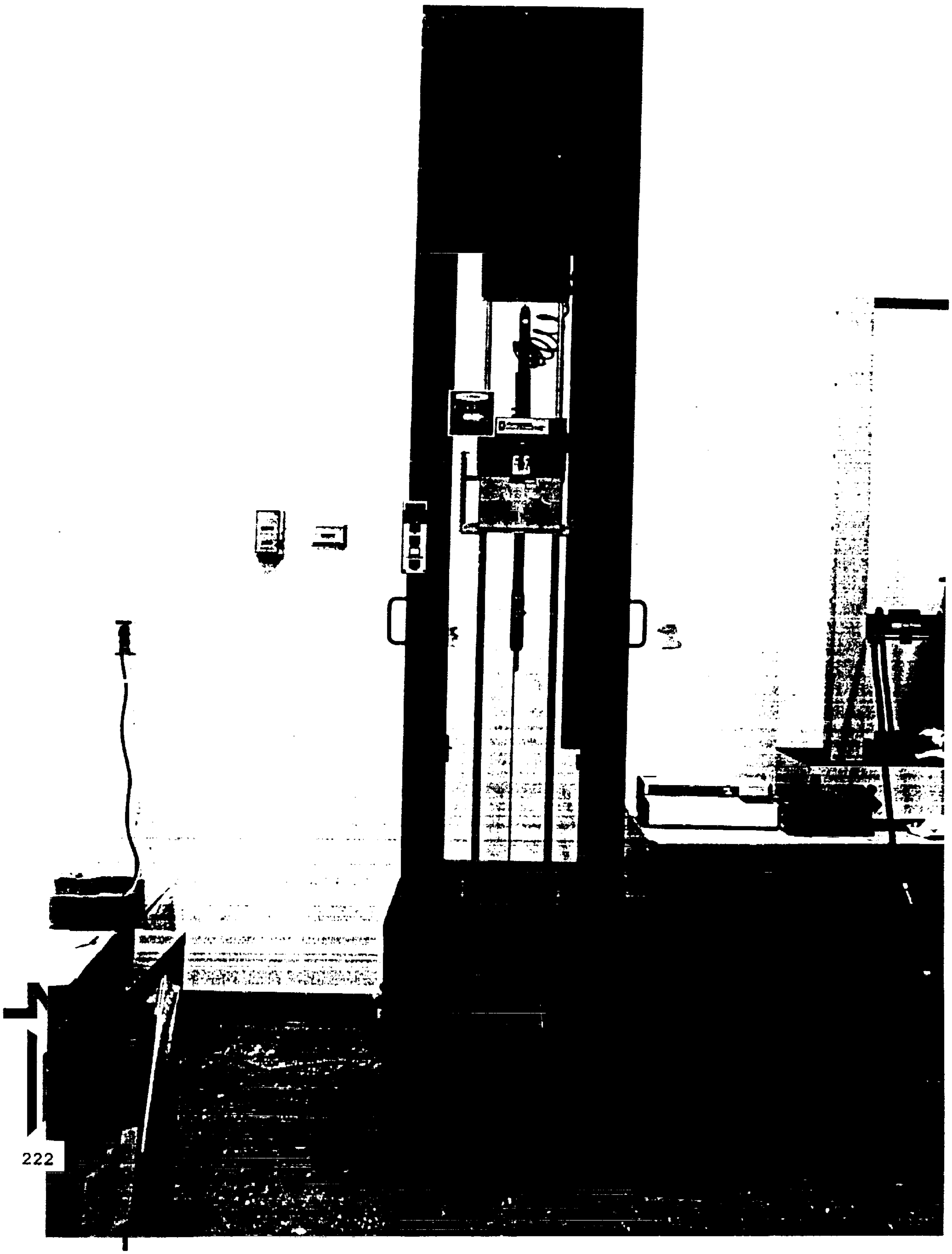
COMPRESSION AFTER IMPACT

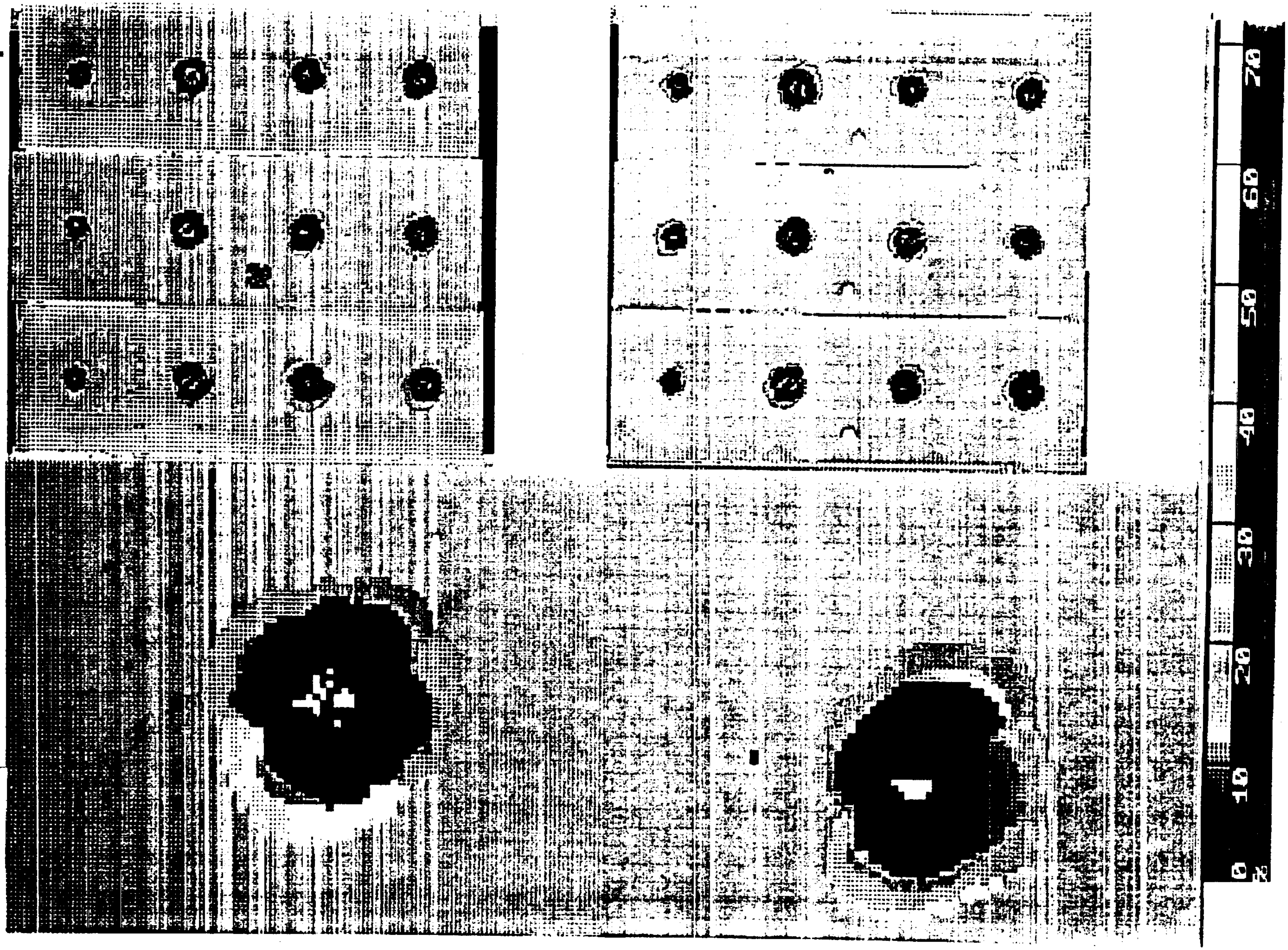


7624-047
7-11-88

DYNATUP IMPACT SYSTEM



COMPARATIVE C-SCANS OF DYNATUP AND LOCKHEED IMPACT SYSTEMS



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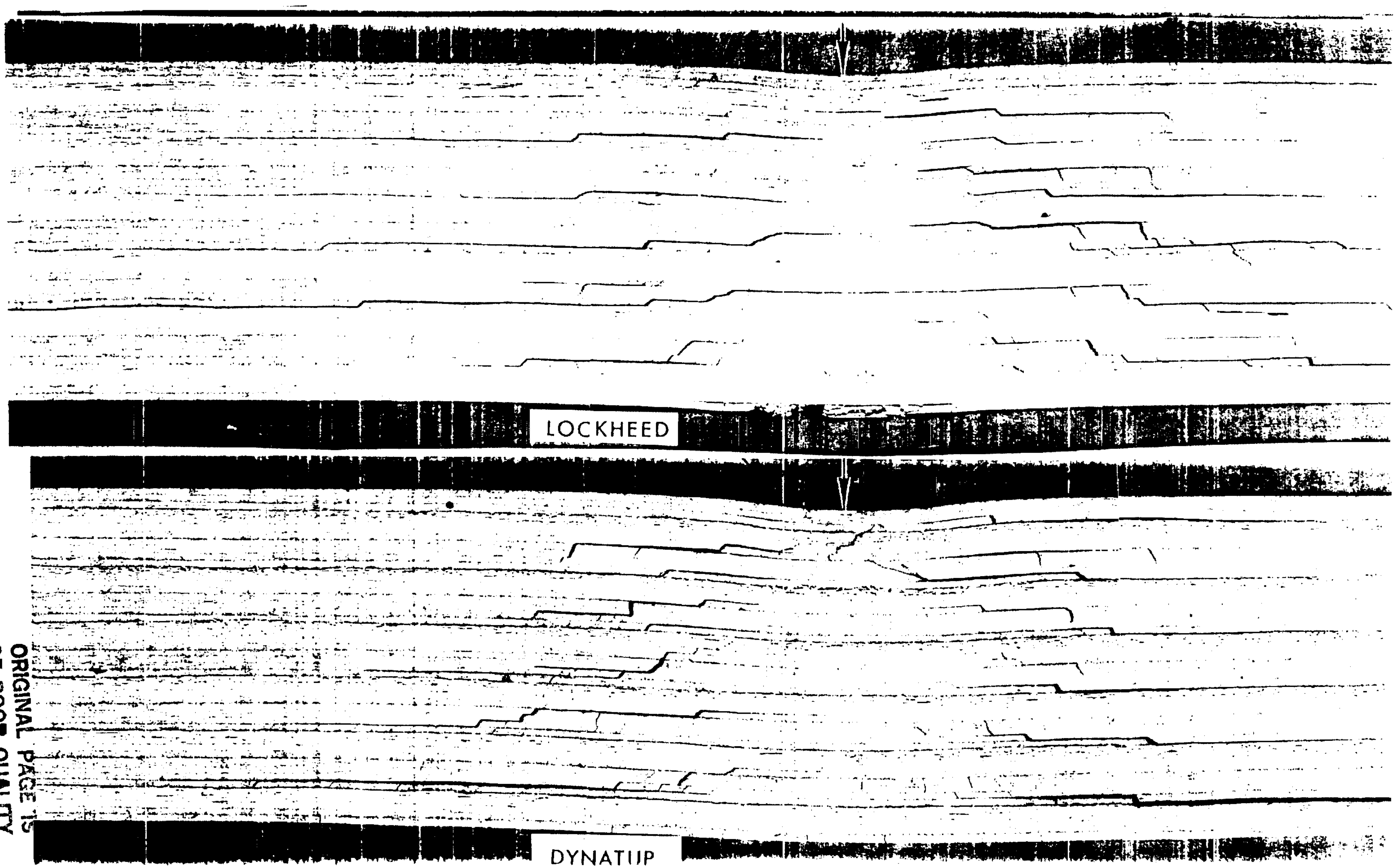
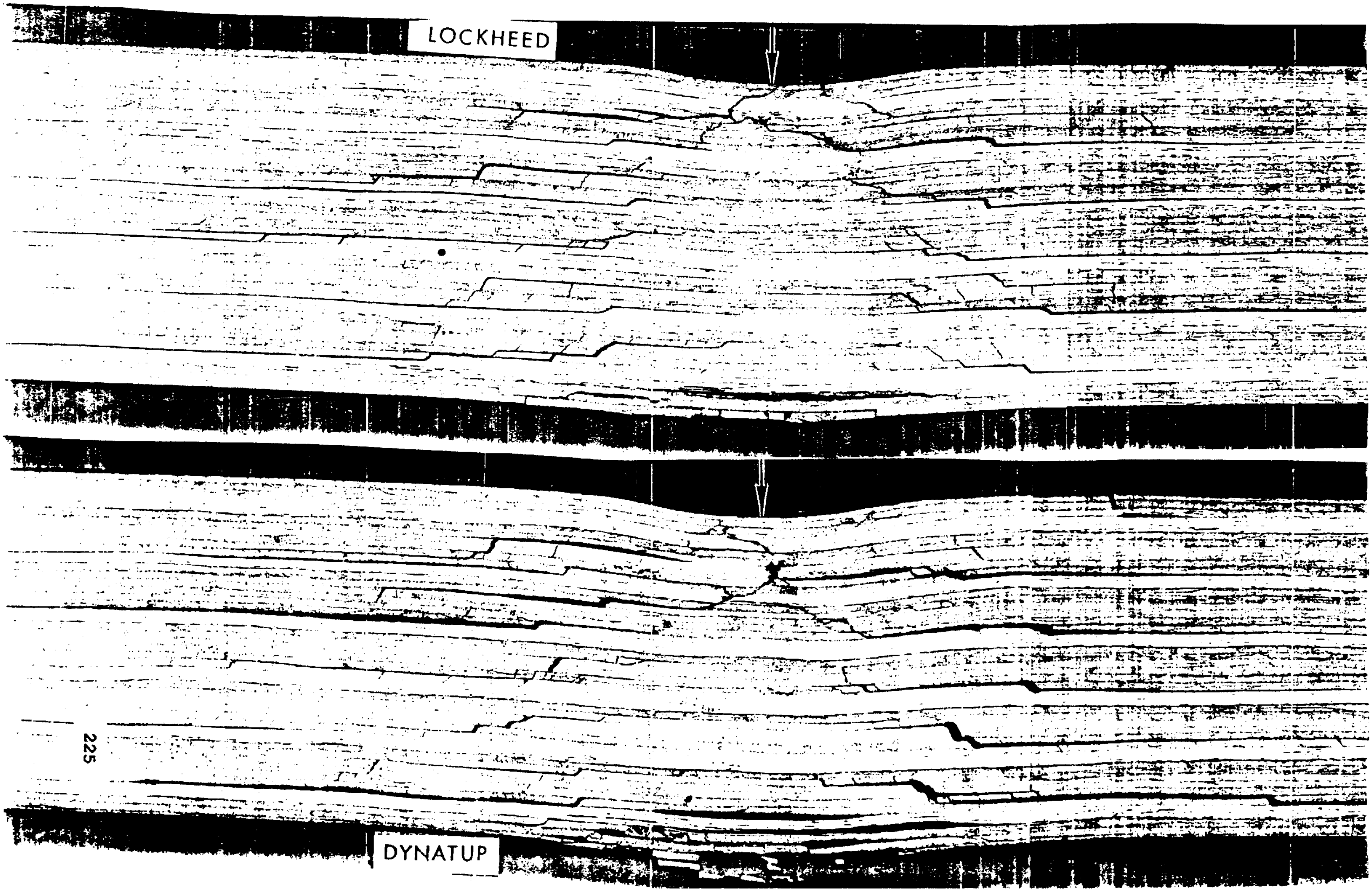
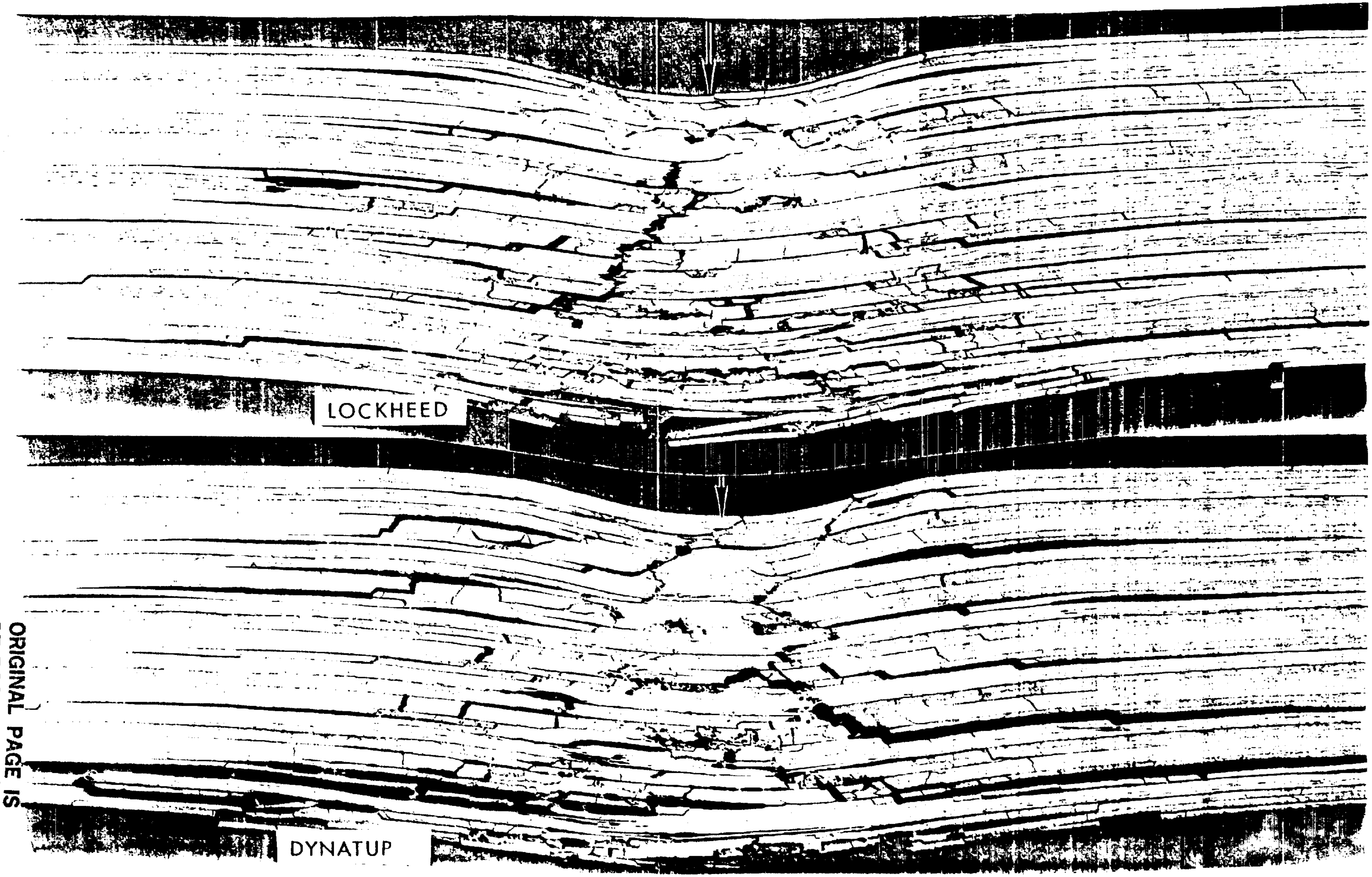
MICROGRAPHIC COMPARISON OF DYNATUP AND LOCKHEED SYSTEMS
960 IN. LBS./IN.



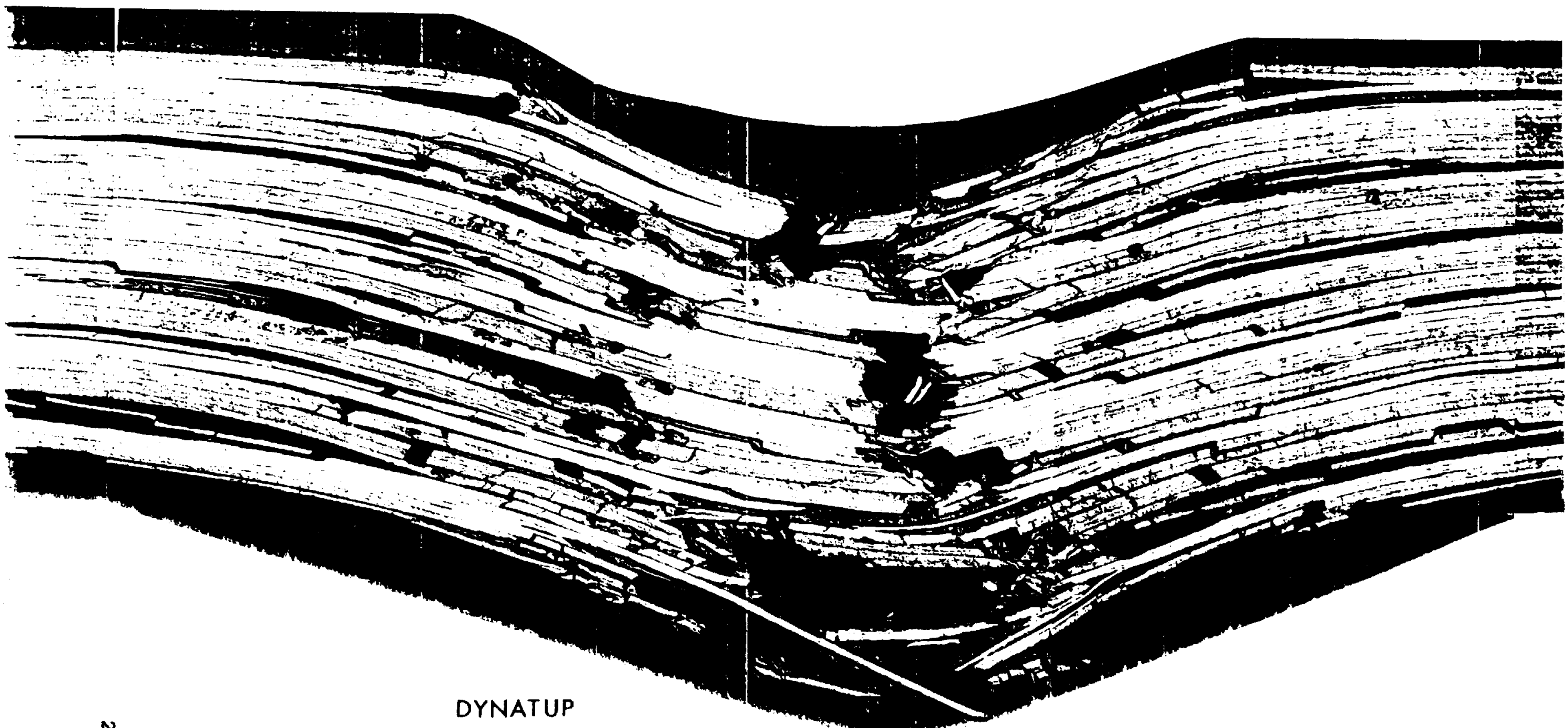
MICROGRAPHIC COMPARISON OF DYNATUP AND LOCKHEED SYSTEMS 1440 IN. LBS./IN.



MICROGRAPHIC COMPARISON OF DYNATUP AND LOCKHEED SYSTEMS 1920 IN. LBS./IN.

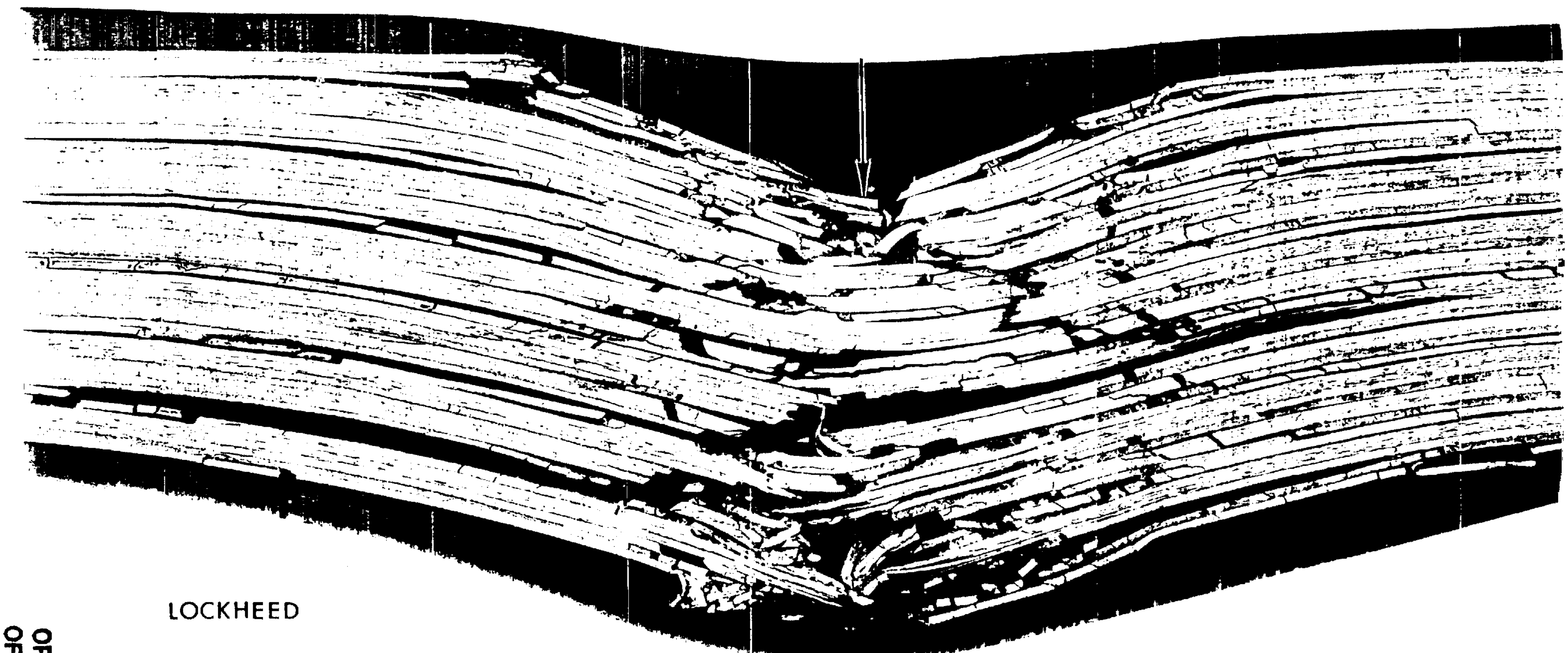


MICROGRAPHIC COMPARISON OF DYNATUP AND LOCKHEED SYSTEMS 2880 IN. LBS./IN.



DYNATUP

MICROGRAPHIC COMPARISON OF DYNATUP AND LOCKHEED SYSTEMS
2880 IN. LBS./IN.



LOCKHEED

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SUBCOMPONENT IMPACT TESTS

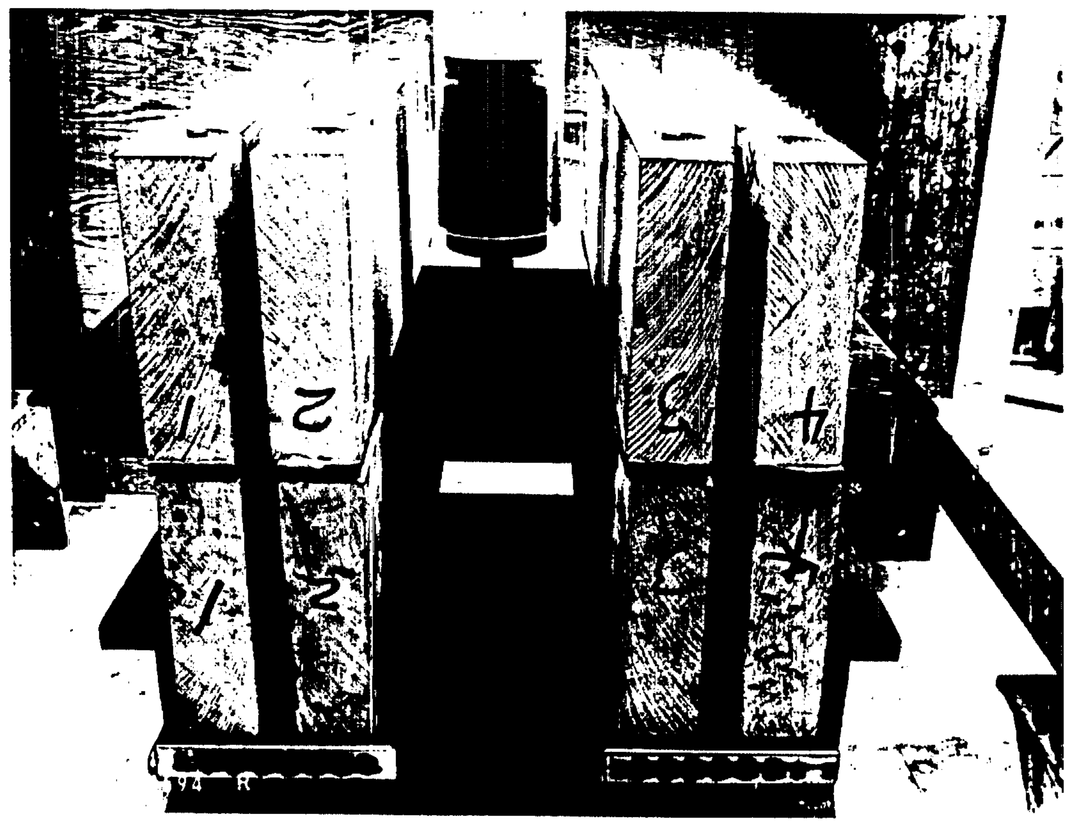
PURPOSE:

- EVALUATION OF GEOMETRICAL DETAILS ON IMPACT DAMAGE TOLERANCE
- DETERMINATION OF ALLOWABLE DESIGN STRENGTH

DATA OBTAINED:

- IMPACT DAMAGE CHARACTERISTICS VERSUS PANEL CONFIGURATION AND IMPACT LOCATION
- STRENGTH AFTER IMPACT

THREE STRINGER IMPACT TEST SETUP



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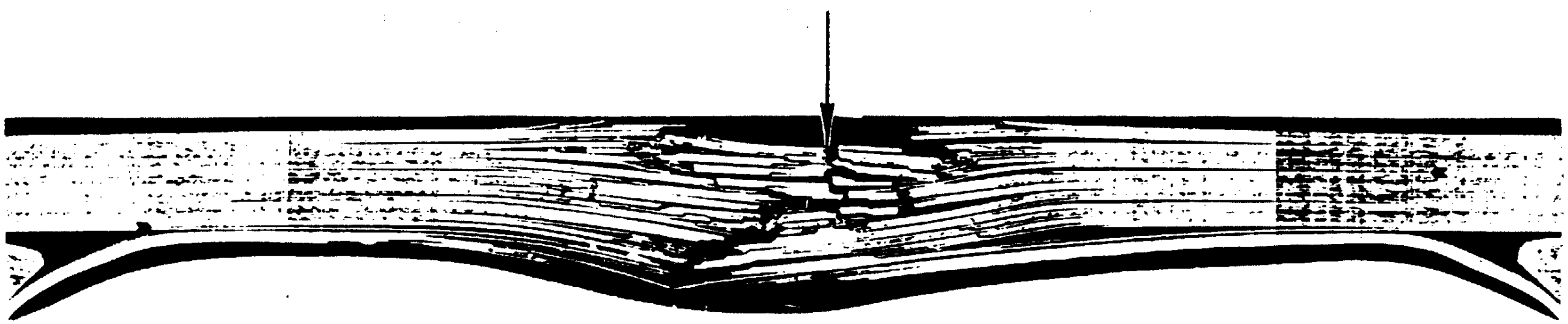
MICROSECTION OF DAMAGE DUE TO IMPACT BETWEEN STRINGERS



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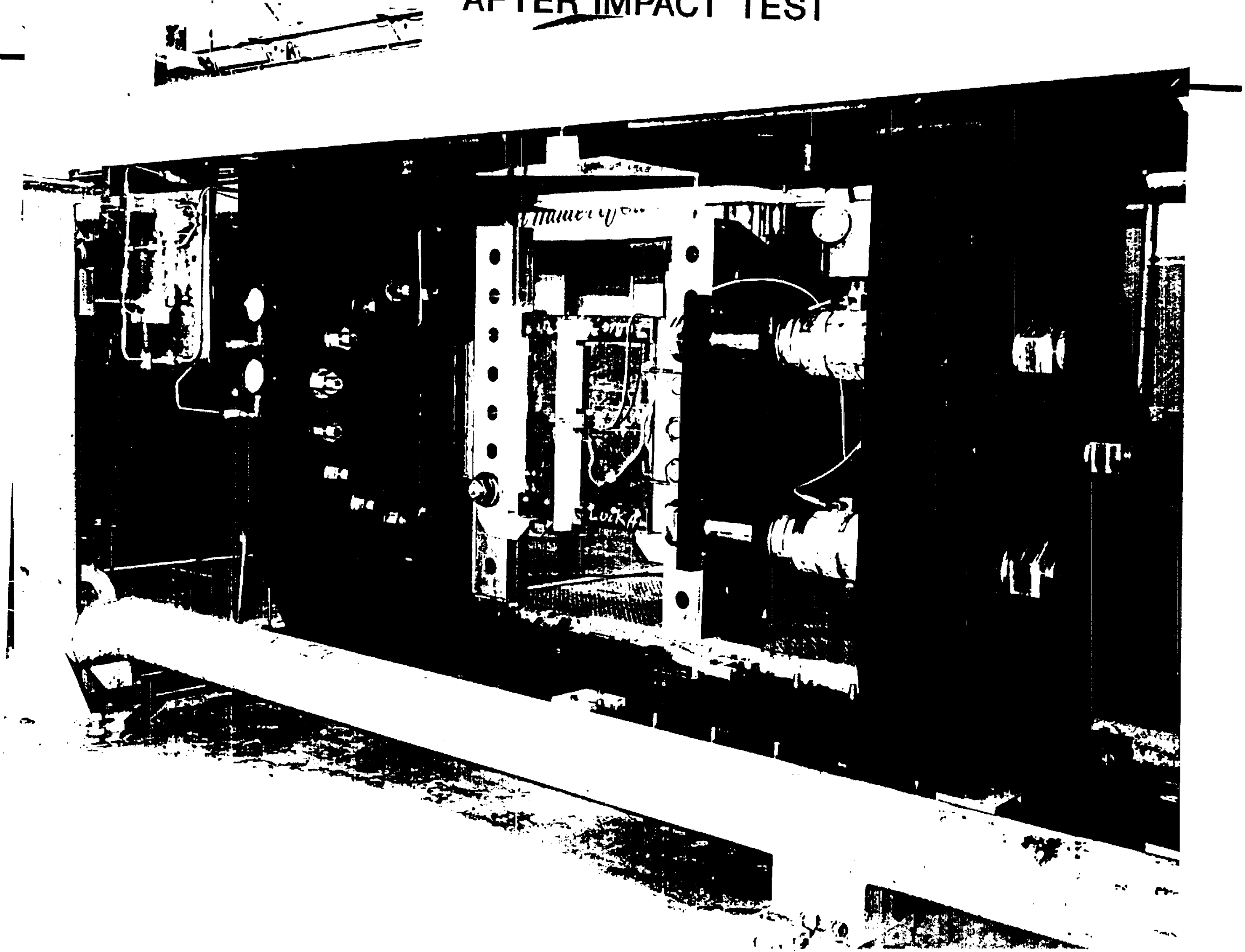
THREE STRINGER PANEL STRENGTH AFTER IMPACT TEST



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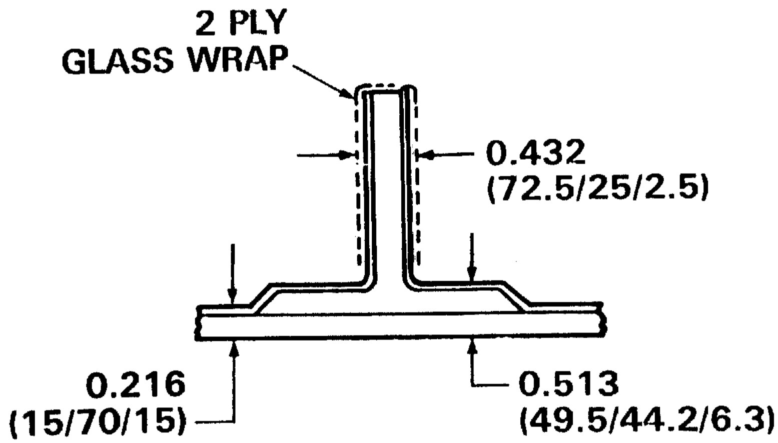
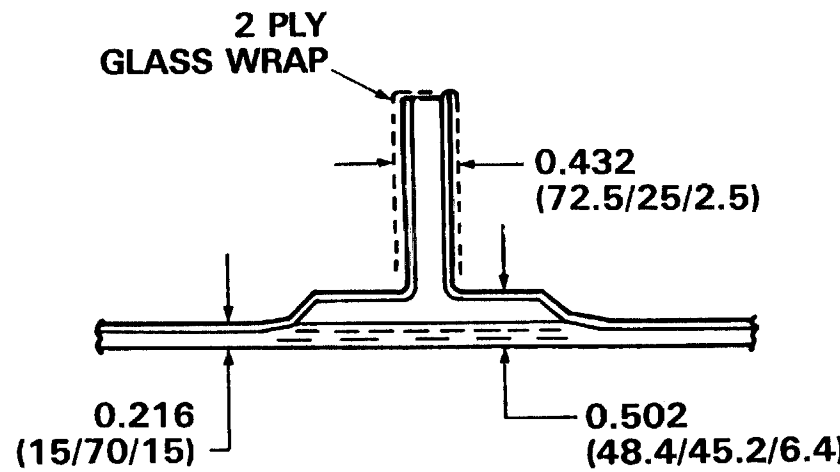
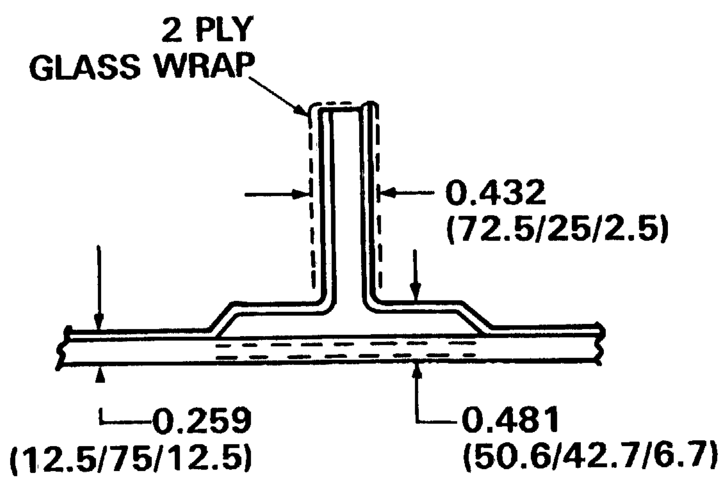
AS4/1806 'T' STIFFENED PANELS



BASELINE



DISCRETE PAD-UP



INTEGRAL PAD-UP

7624-056
7-11-88

TEE STIFFENED PANEL COMPRESSION TESTS

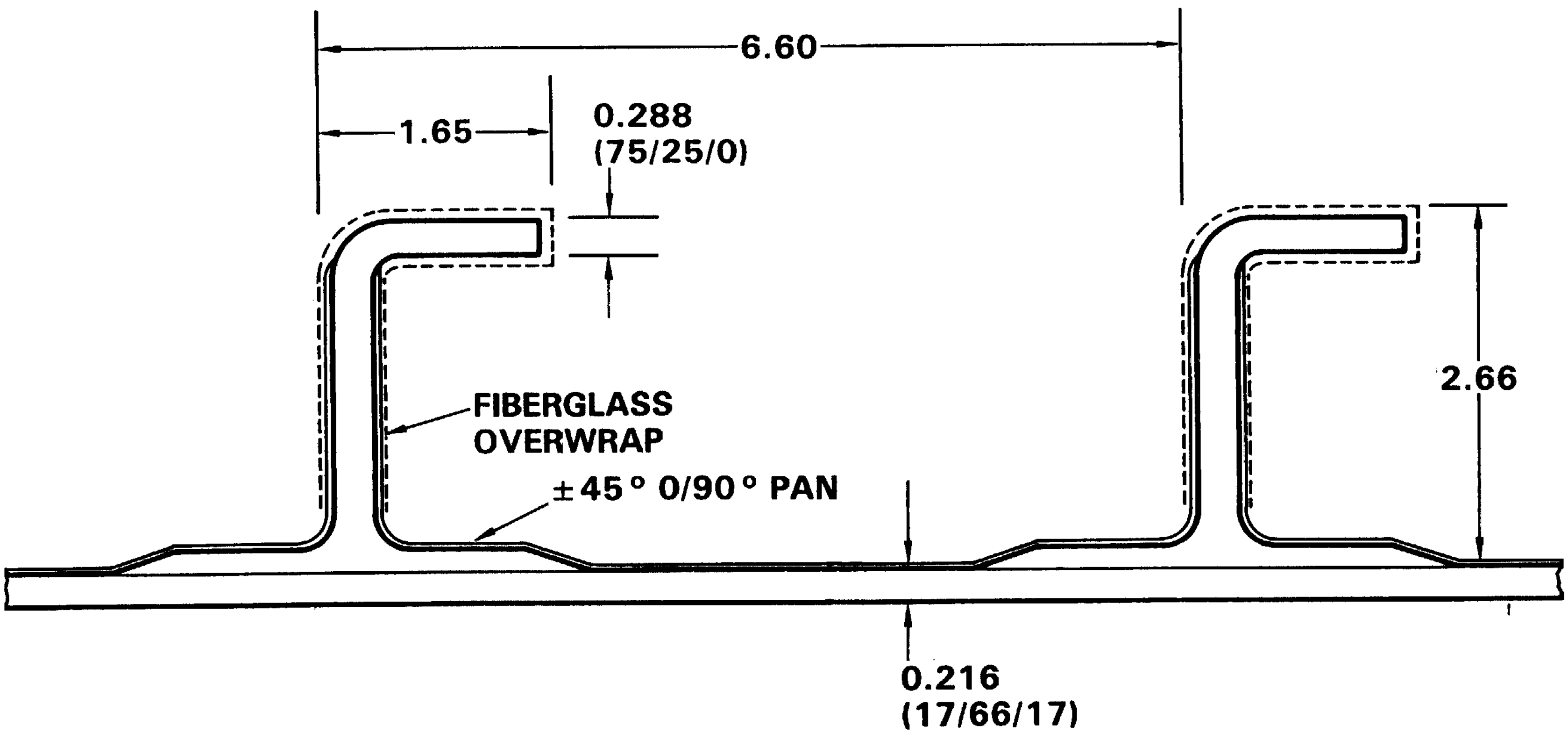
MATL: AS4/1806 TAPE   PANEL WEIGHT: 0.034 LB/IN^2

	TEST CONFIGURATION	FAILURE LOAD (KIPS)	Nx (LB/IN) Nx_{ULT} = 22,000 LB/IN	FAILURE STRAIN (IN/IN)
BASELINE	CRIPPLING, 0.25 DIA. FASTENERS	499.8	35,200	0.0064
	POST-IMPACT 80 FT-LB	488.5	34,400	0.0071
	POST-IMPACT 80 FT-LB	495.0 (1) / 413.9 (2)	23,800 (1) / 19,900 (2)	0.0045 (1) / 0.0047 (2)
	POST-IMPACT 40 FT-LB	499.2 (1) / 528.3 (2)	24,000 (1) / 25,400 (2)	0.0045 (1) / 00.061 (2)
INTEGRAL PAD-UP	POST-IMPACT 80 FT-LB	453.0	31,900	0.0056
	POST-IMPACT 100 FT-LB	563.7	39,700	0.0068
DISCRETE PAD-UP	POST-IMPACT 80 FT-LB	409.0	28,800	0.0049
	POST-IMPACT 100 FT-LB	420.3	29,600	0.0050

(1) CENTER STIFFENER FAILURE   (2) RESIDUAL STRENGTH

7624-057
7-11-88

AS4/1806 FABRIC J-STIFFENED DESIGN

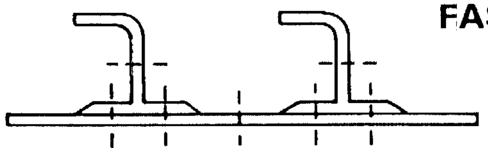
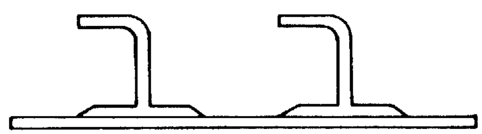
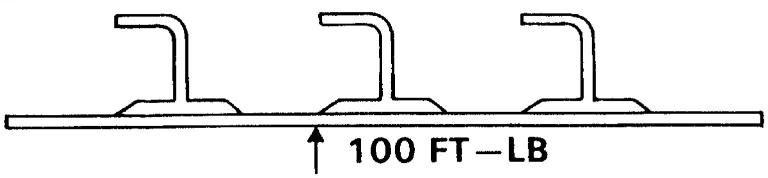
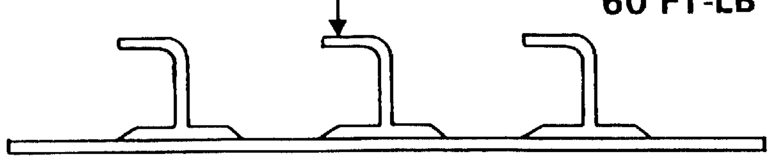


Nx_{ULT} = 22,000 LB/IN

7624-028
7-11-88

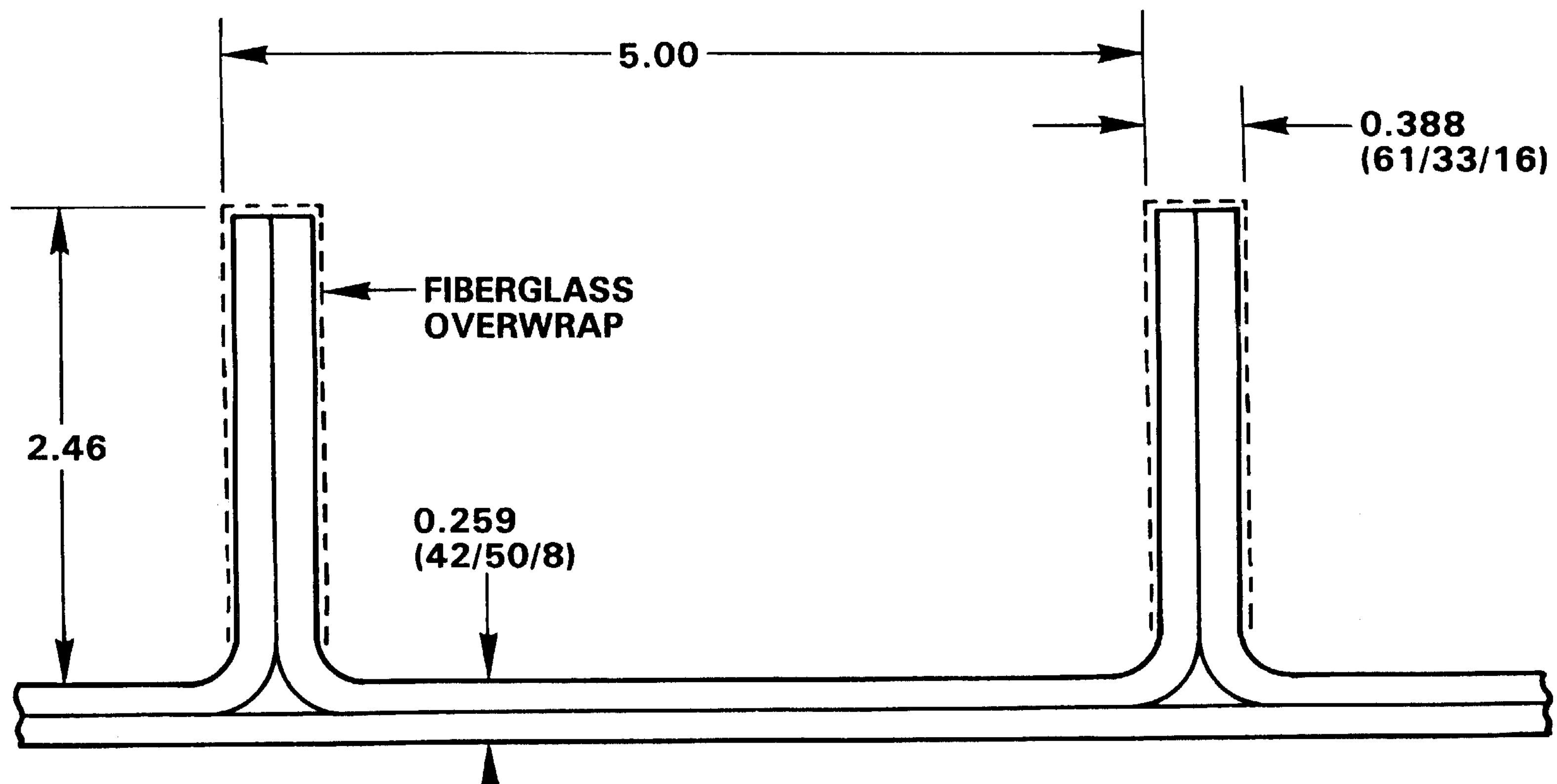
J-STIFFENED PANEL COMPRESSION TESTS

MATL: AS4/1806 FABRIC

TEST CONFIGURATION	FAILURE LOAD (KIPS)	Nx (LB/IN) Nx_{ULT} = 22000 LB/IN	FAILURE STRAIN (IN/IN)
0.25 DIA FASTENERS  CRIPPLING	484.2	34100	0.0064
 COLUMN	329.7	27570	0.0045
 100 FT—LB POST-IMPACT	524.9	25240	0.0048
POST-IMPACT 60 FT-LB 	541.3	26020	0.0046

7624-030
7-11-88

IM7/8551-7 TAPE BLADE STIFFENED DESIGN

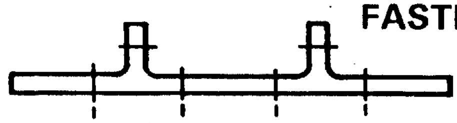
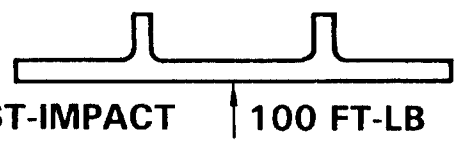
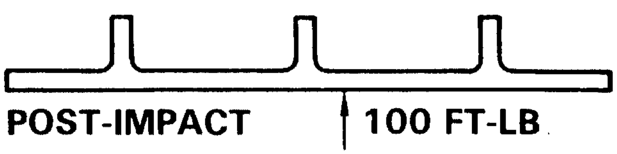
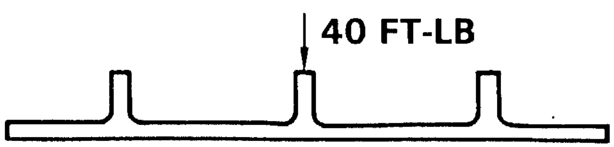


N_xULT = 22,000 LB/IN

7624-035
7-11-88

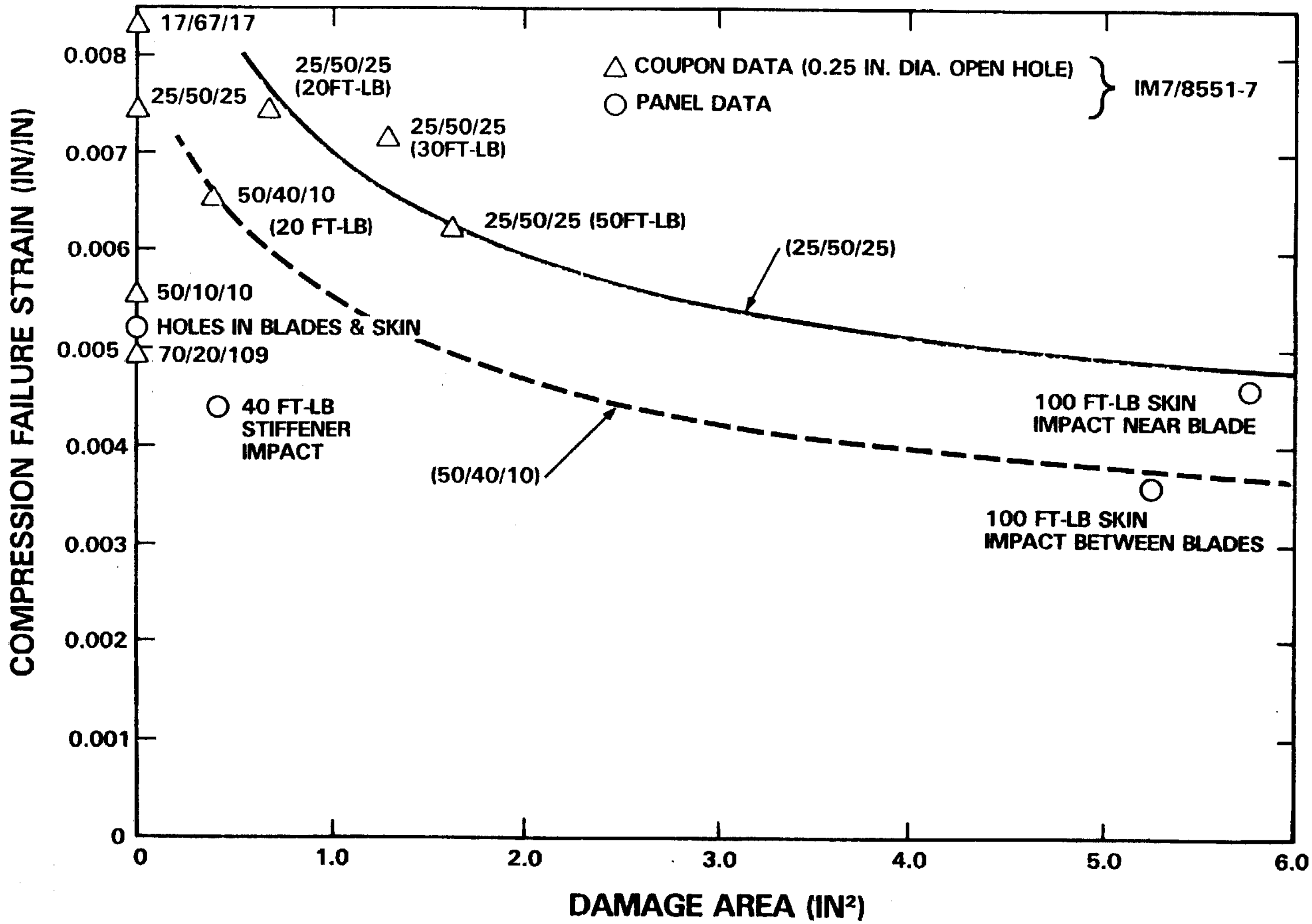
BLADE STIFFENED PANEL COMPRESSION TESTS

MATL: IM7/8551-7 TAPE   PANEL WEIGHT: 0.027 LB/IN^2

TEST CONFIGURATION	FAILURE LOAD (KIPS)	Nx (LB/IN) Nx_{ULT} = 22,000 LB/IN	FAILURE STRAIN (IN/IN)
 0.25 DIA FASTENERS CRIPPLING	282.1	25,880	0.0052
 POST-IMPACT 100 FT-LB	211.8	19,430	0.0036
 POST-IMPACT 100 FT-LB	373.2	23,550	0.0046
40 FT-LB  POST-IMPACT	361.2	22,790	0.0044

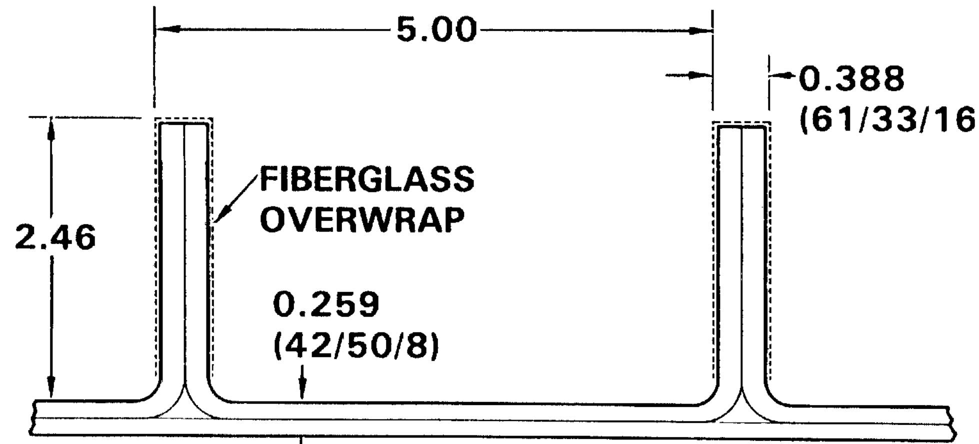
7624-037
7-11-88

COMPARISON OF COUPON DATA TO BLADE STIFFENED PANEL DATE



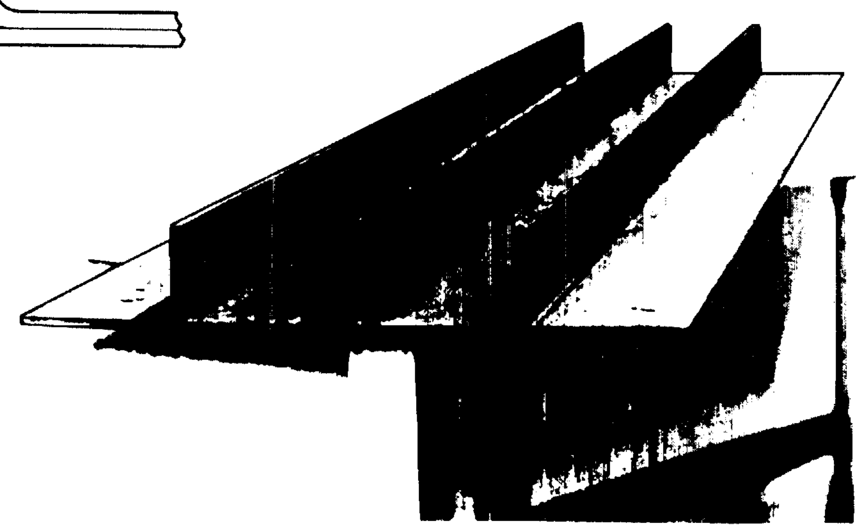
7624-038
7-11-88

BASELINE BLADE STIFFENED DESIGN



Nx_{ULT} = 22,000 LB/IN.

AS4/2220



7624-060
7-11-88

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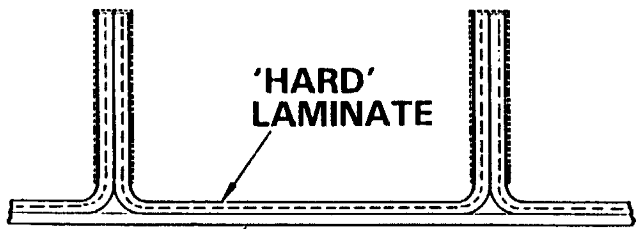
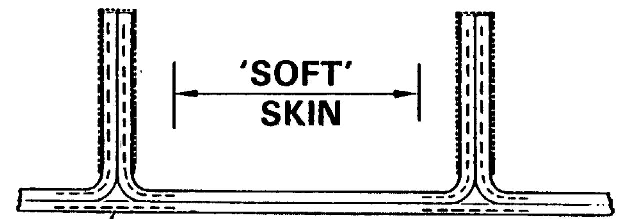
DAMAGE TOLERANCE IMPROVEMENT CONCEPTS

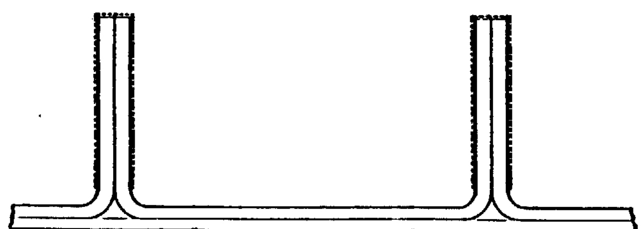


'SOFT' LAMINATE WITH
ADHESIVE INTERLEAVE
ϵ = 0.00450 IN./IN.

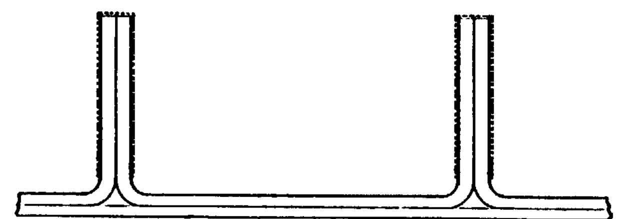


0° INSERT

WITH INSERT ϵ = 0.00397 IN./IN.
WITHOUT INSERT ϵ = 0.00528 IN./IN.



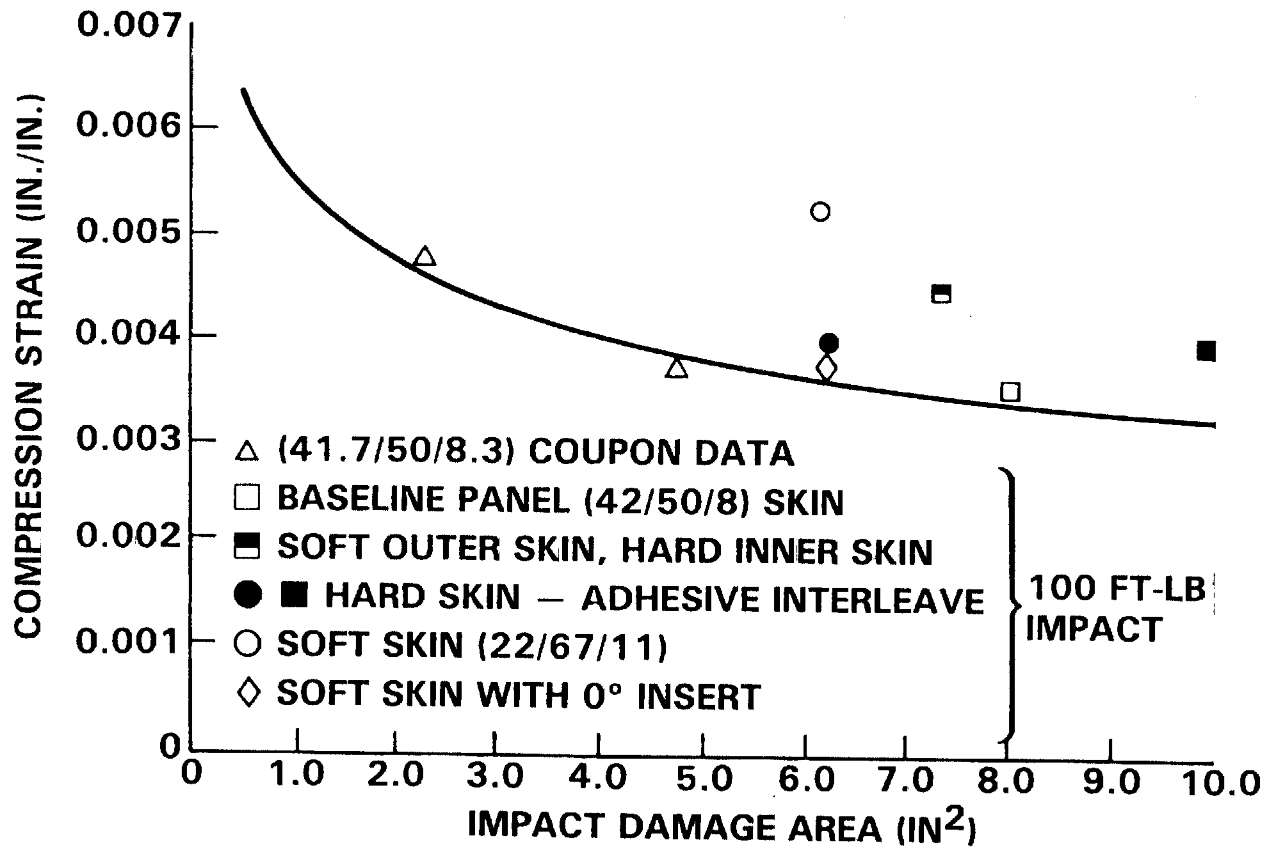
ADHESIVE INTERLEAVE BETWEEN
0° & 45° PLIES
ϵ = 0.0040 IN./IN.



BASELINE
'HARD' SKIN
ϵ = 0.00354 IN./IN.

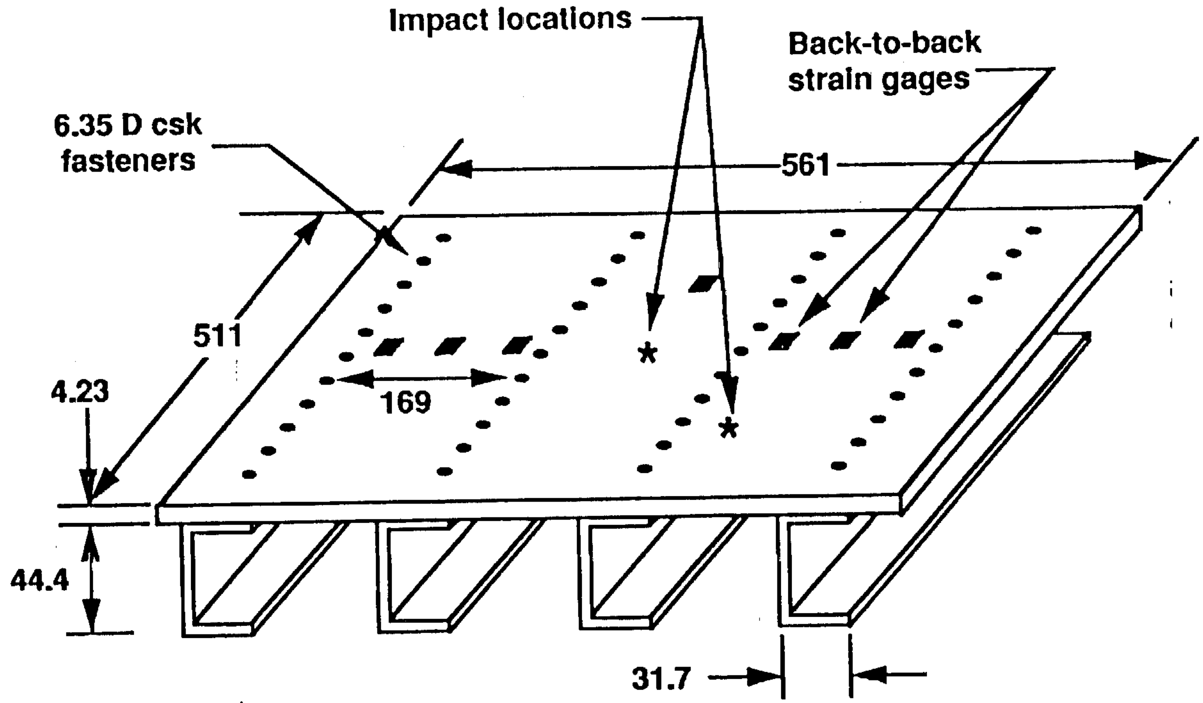
7624-061
7-11-88

AS4/2220 COUPON AND BLADE STIFFENED PANEL DATA



7624-062
7-11-88

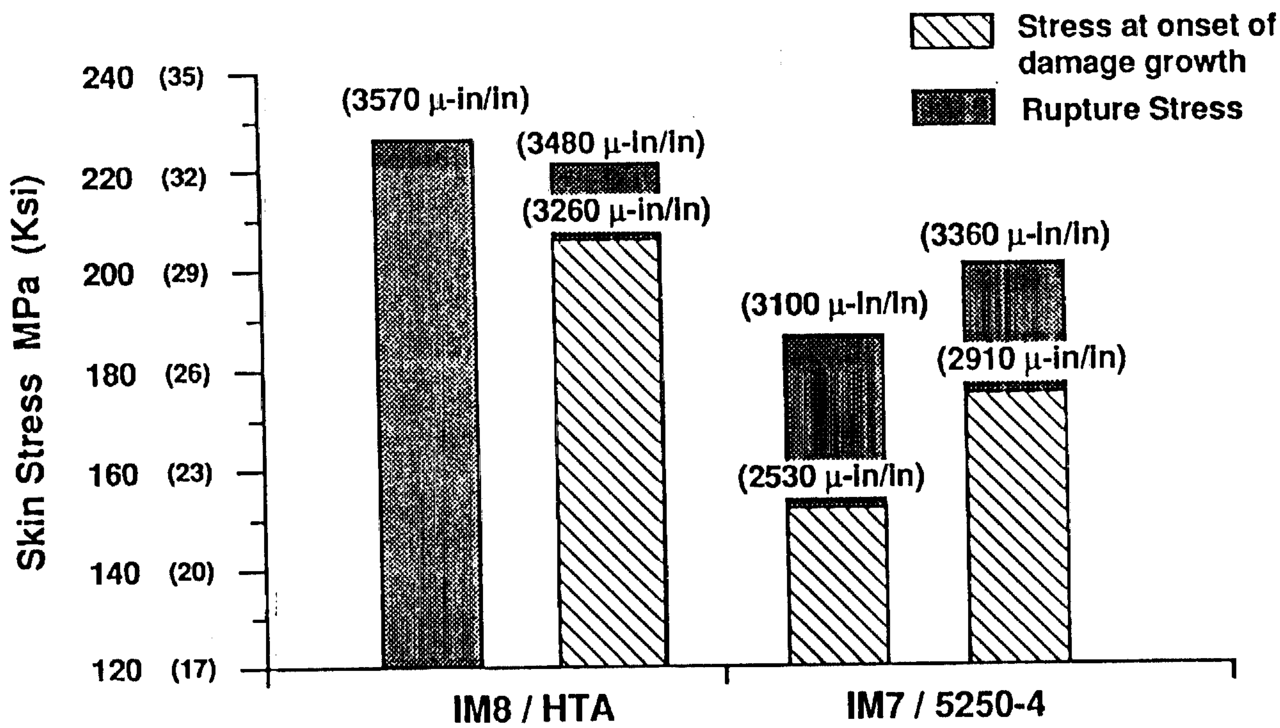
SCHEMATIC OF 4-STRINGER PANEL TEST ARTICLE



All dimensions are in millimeters.

11-29-90
GJB

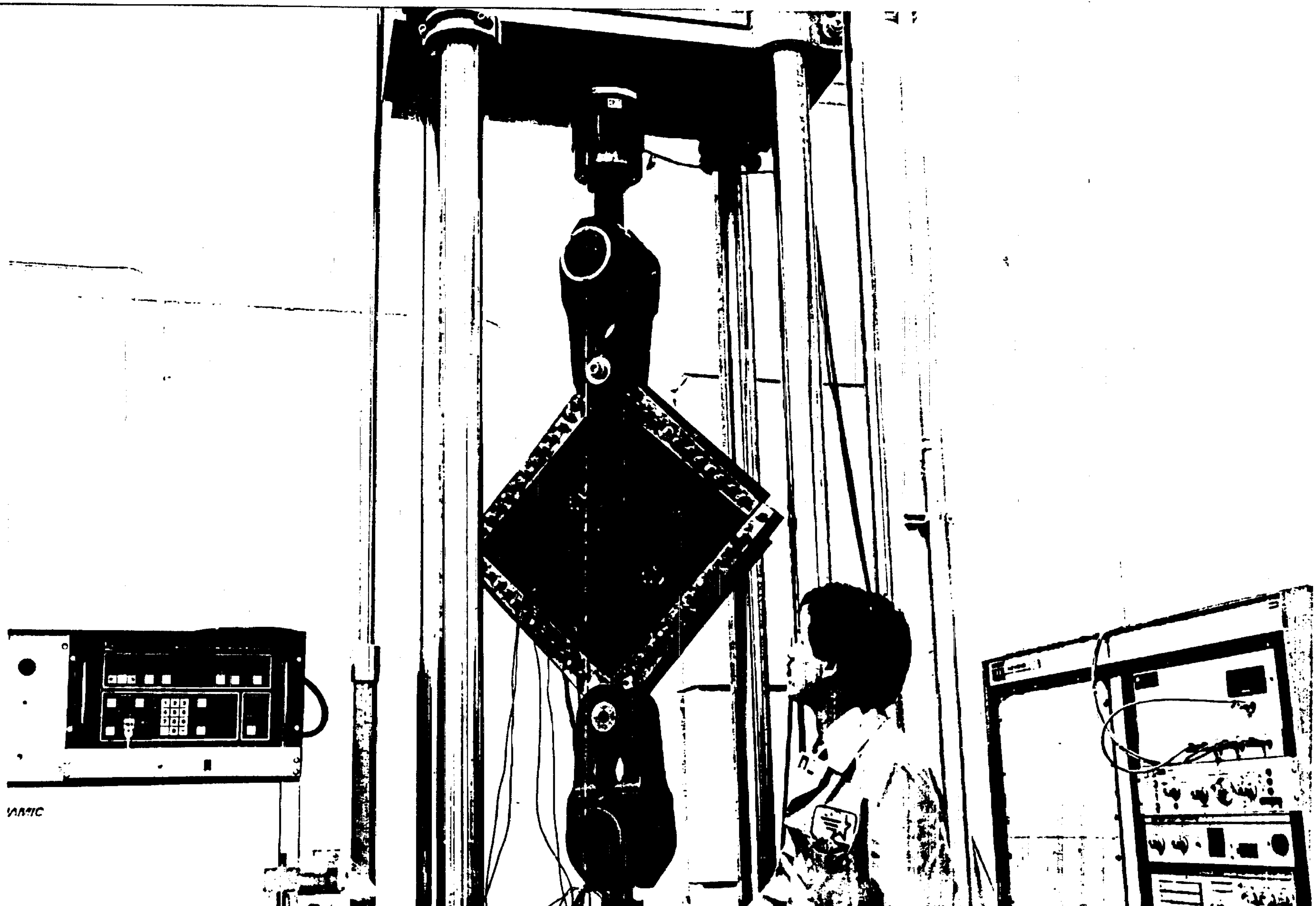
STATIC STRENGTH COMPARISON OF 4-STRINGER PANELS



Damage Area (cm^2)	17.7	10.1	74.0	19.0
Dent Depth (mm)	.864	1.70	4.09	thru

11-29-90
GJB

SHEAR PANEL IMPACT DAMAGE STRENGTH TEST
STATIC AND FATIGUE



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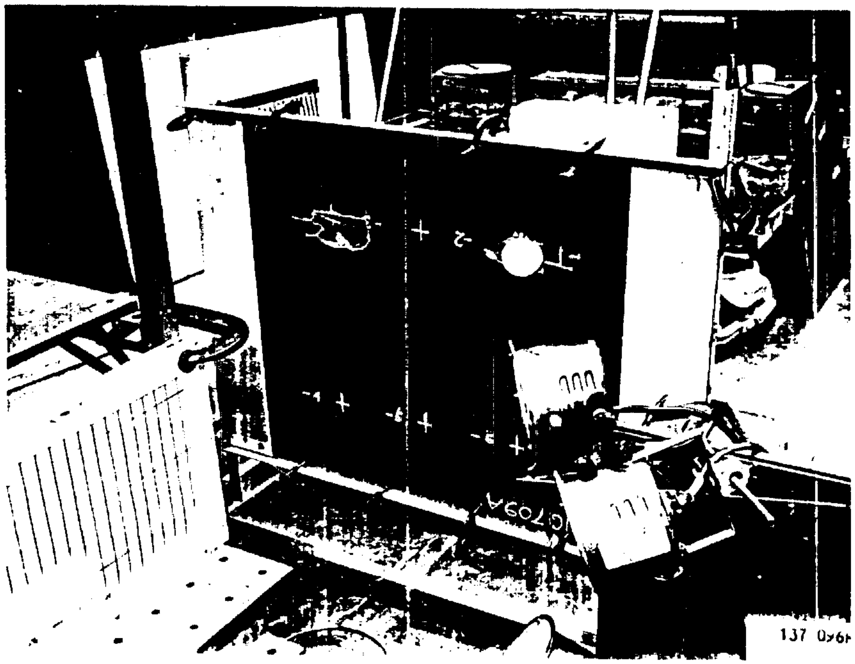
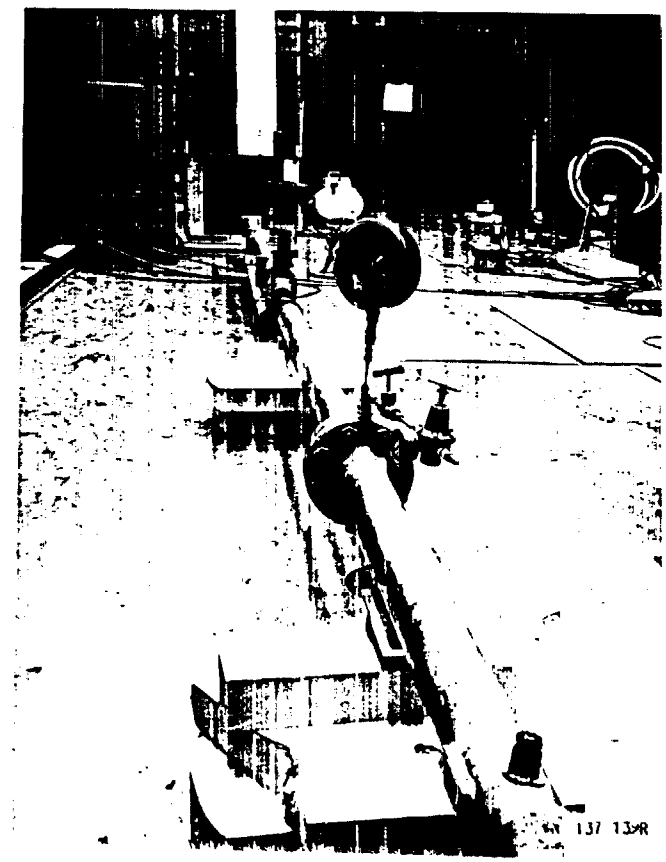
HIGH VELOCITY IMPACT TESTS





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COMPONENT IMPACT TESTS

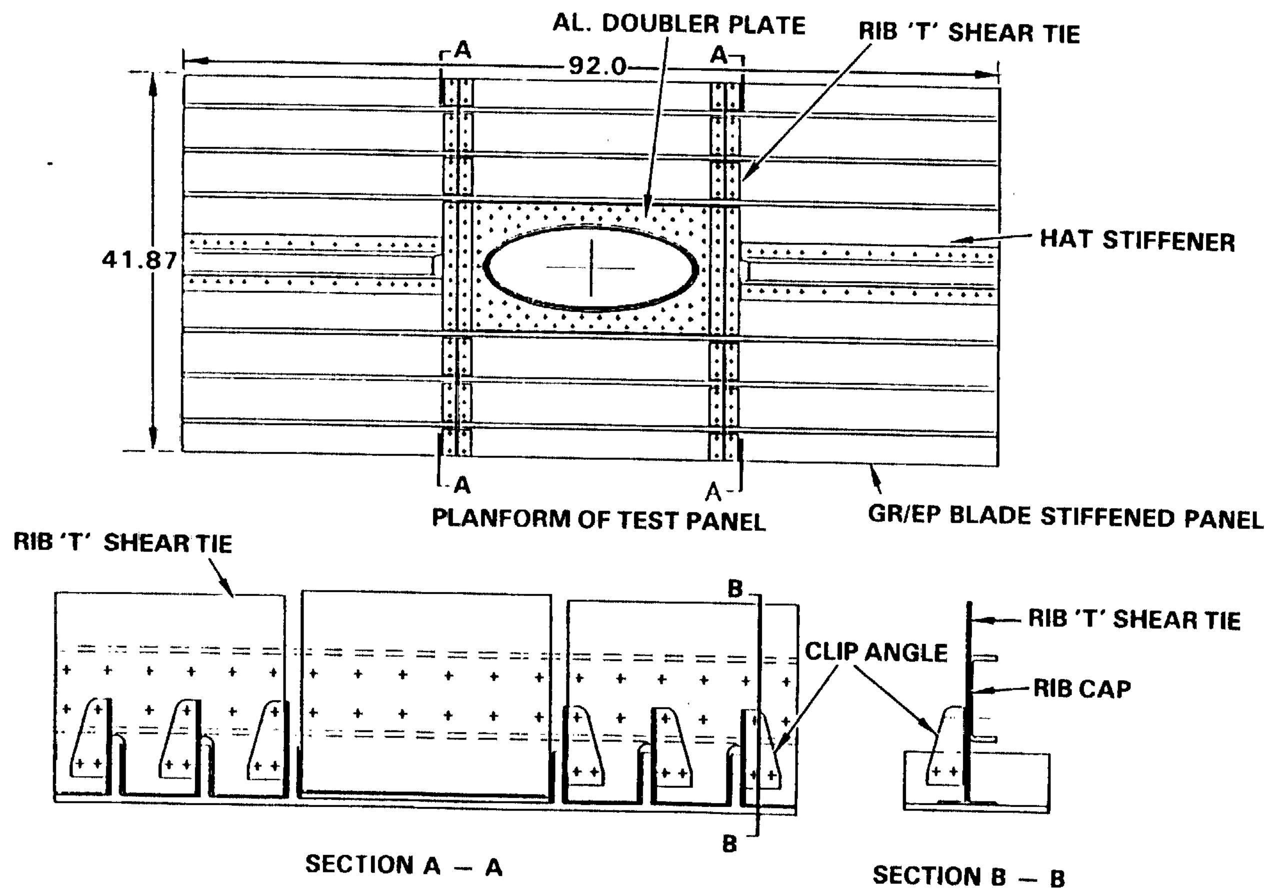
PURPOSE:

- VERIFICATION OF IMPACT DAMAGED STRENGTH

DATA OBTAINED:

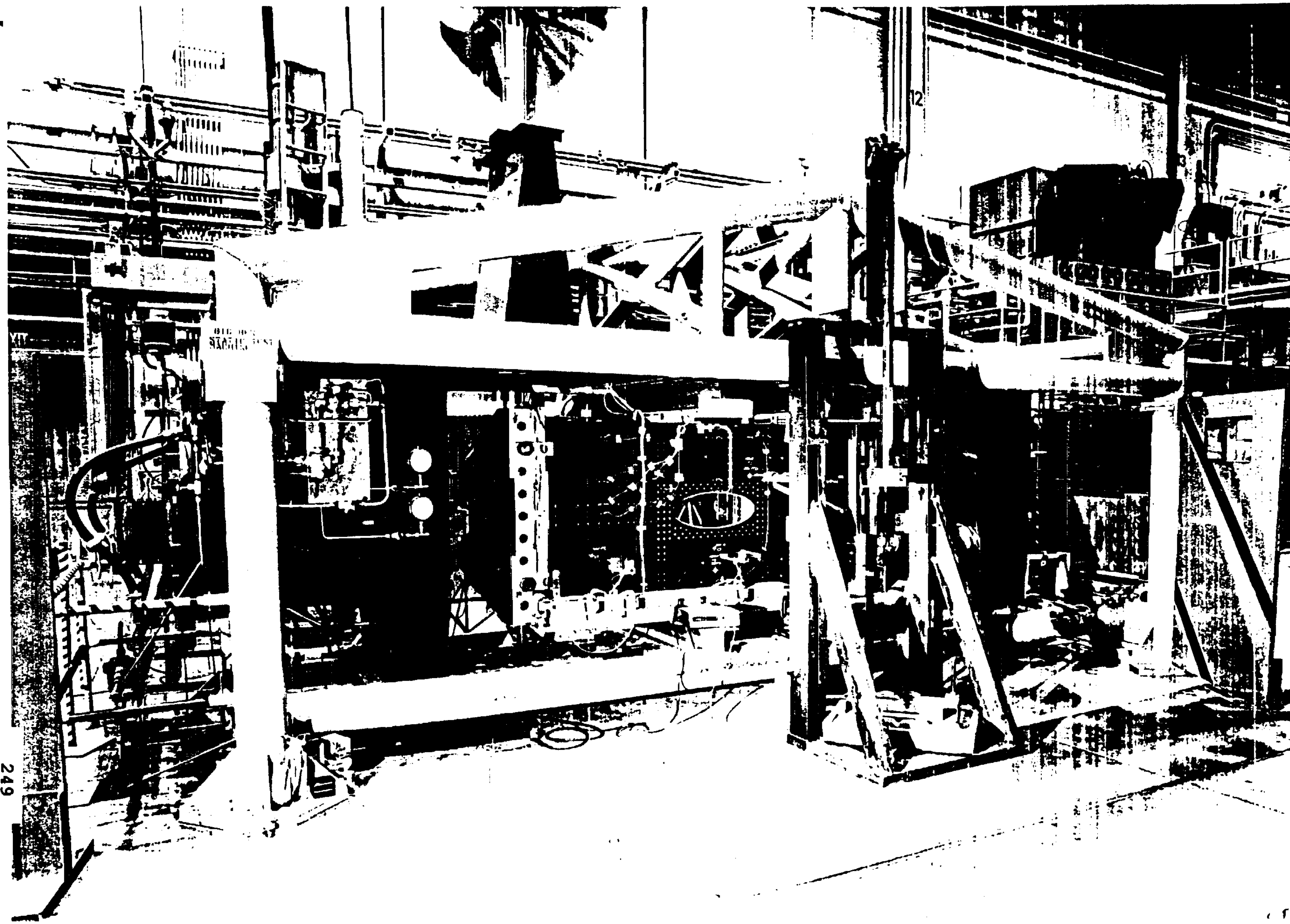
- IMPACT DAMAGE CHARACTERISTICS
- STRENGTH AFTER IMPACT

STIFFENED PANEL WITH CUT-OUT



7624-065
7-11-88

STIFFENED PANEL WITH CUT-OUT IMPACT DAMAGE STRENGTH TEST

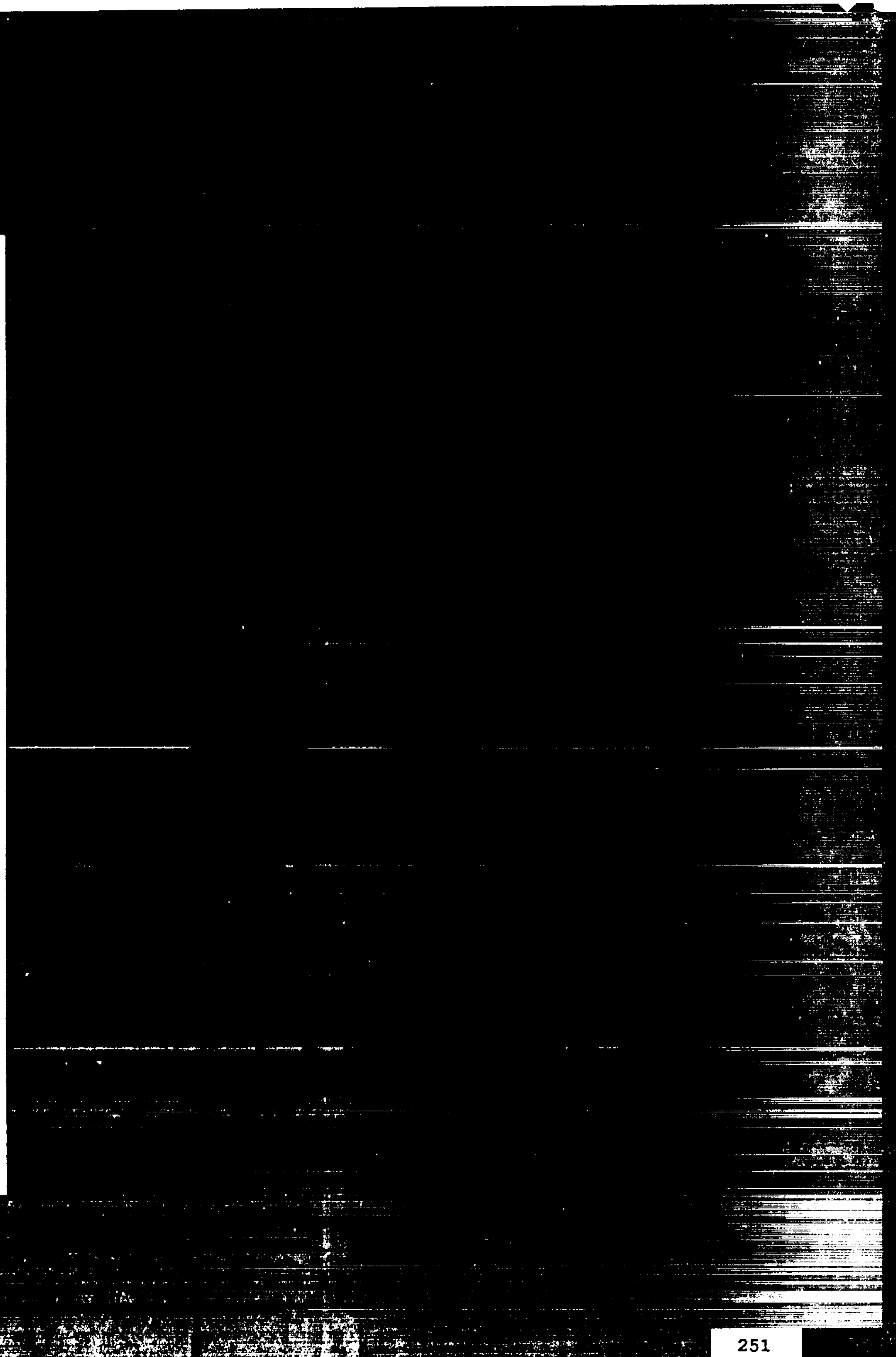


PANEL TEST RESULTS

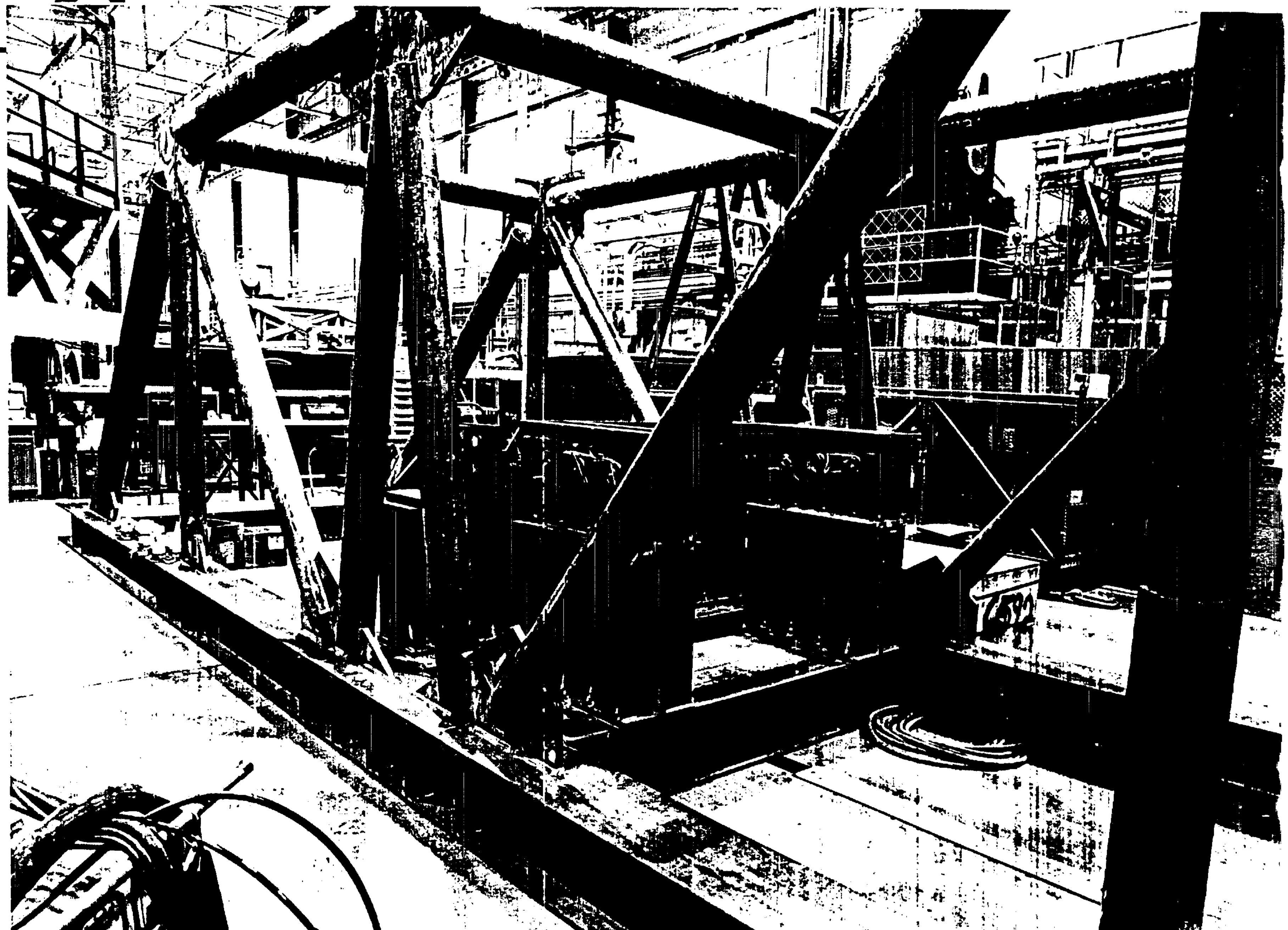
TEST CONDITION	COMPRESSION LOAD (LB/IN)	MAX. STRAIN (IN/IN)	FAR FIELD STRAIN (IN/IN)
UNDAMAGED	16100	0.0041	0.0020
DAMAGED — 2 SKIN IMPACTS AT 100 FT-LB — 1 BLADE IMPACT AT 30 FT-LB	23900	0.0066	0.0033

7624-066A
7-11-88

BOX BEAM COMPONENT



BOX BEAM COMPONENT TEST SETUP



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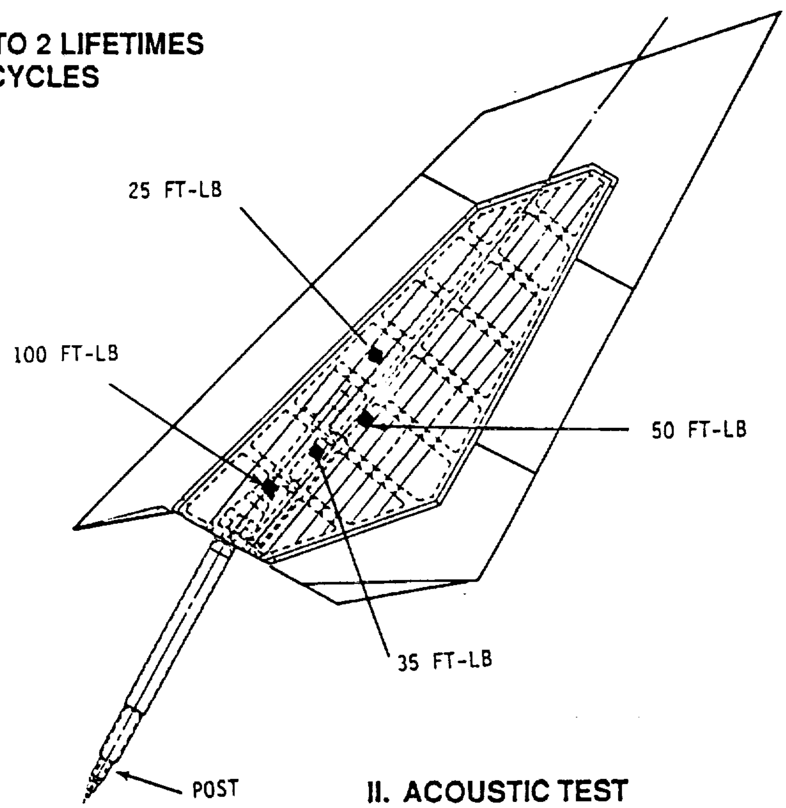
FULL SCALE RUDDER TEST PROGRAM

I. STATIC / FATIGUE TEST

- TESTED TO DESIGN ULTIMATE LOADS NO VISIBLE DAMAGE

- SUBJECTED TO 2 LIFETIMES OF FATIGUE CYCLES



II. ACOUSTIC TEST

- SUBJECTED TO AN ACOUSTIC / THERMAL EQUIVALENT OF 2 LIFETIMES
  - NO VISIBLE DAMAGE
  - SOME DELIMITATIONS ALONG RISERS DISCOVERED AFTER ONE LIFETIME (POSSIBLY THERE SINGLE INITIAL MANUFACTURING). NO GROWTH OF DELAMS, HOWEVER, FROM 1 LIFETIME TO 2.

III. STATIC TEST 70 FAILURE

- RUDDER SURFACE IMPACTED 4 LOCATIONS

- FAILURE OCCURRED @ 125% OF DESIGN ULTIMATE BENDING MOMENT.

SUMMARY

- LAMINATE IMPACT TESTS ARE ADEQUATE TO DETERMINE RELATIVE IMPACT DAMAGE BEHAVIOR OF COMPOSITES.
- INDUSTRY STANDARD FOR COUPON IMPACT TESTS SHOULD BE ESTABLISHED.
- SUBCOMPONENT IMPACT TESTS ARE REQUIRED TO ESTABLISH GEOMETRICAL EFFECTS AND DESIGN SPECIFIC STRENGTH ALLOWABLES.
- COMPONENT IMPACT TESTS ARE REQUIRED TO VERIFY STRUCTURAL INTEGRITY.

IMPACT DAMAGED COMPOSITES, PART II: STANDARD TESTS FOR FUSELAGE STRUCTURAL ISSUES

Ernest F. Dost

and

William B. Avery

Boeing Commercial Airplane Group

Impact Damaged Composites, Part 1: Damage Simulation and Strength Predictions

Transport Fuselage Structures → *L. Ilcewicz*

A Need to Understand Fundamentals → *L. Ilcewicz*

Types of Impact Damage States → *E. Dost*

Damage Simulation & CAI Strength Prediction → *E. Dost*

Part 1 Summary → *E. Dost*

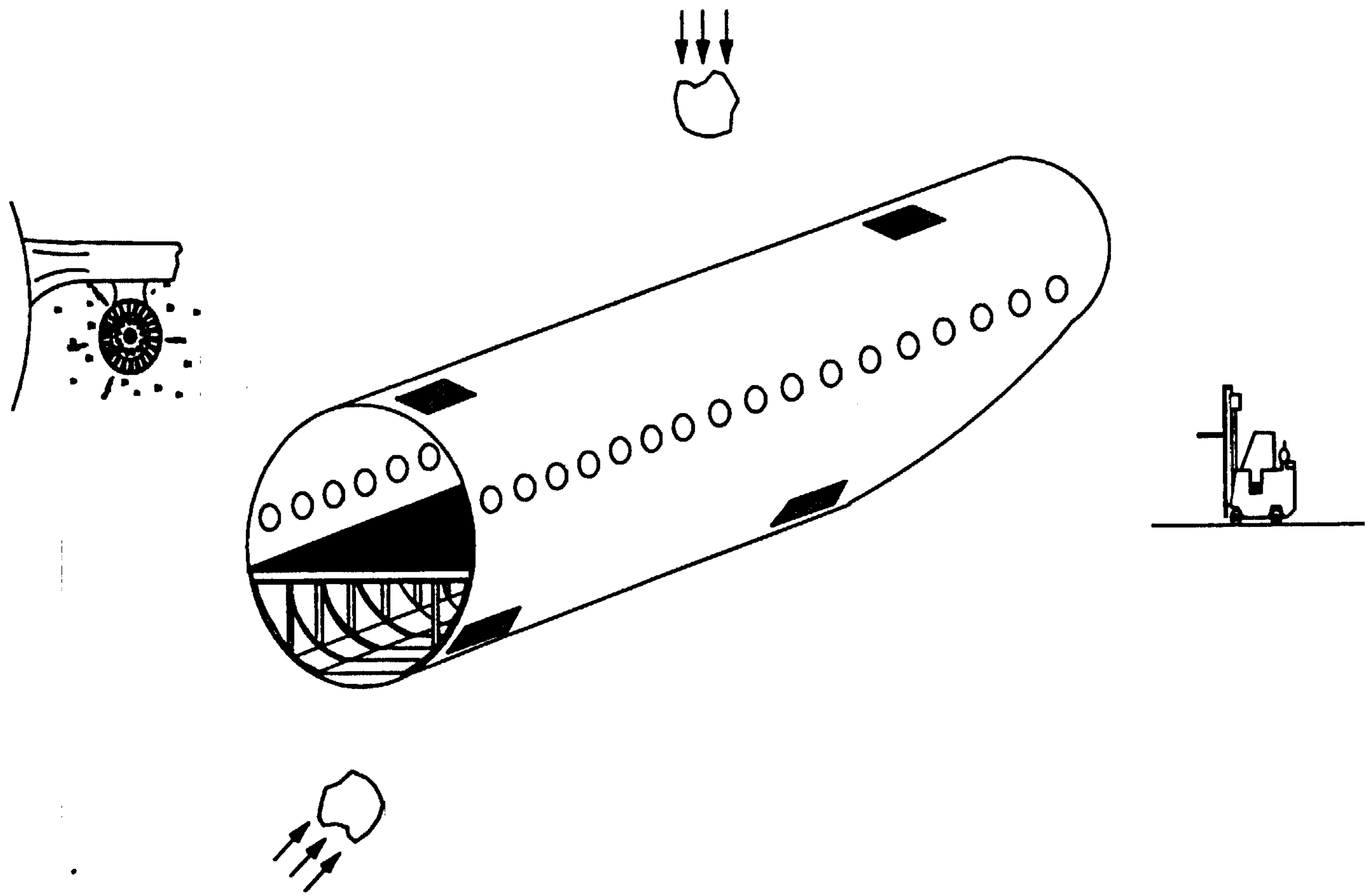
Impact Damaged Composites, Part 2: Standard Tests for Fuselage Structural Issues

Fuselage Impact Damage Characterization → *E. Dost*

Standard Specimen Tests for Fuselage Applications → *W. Avery*

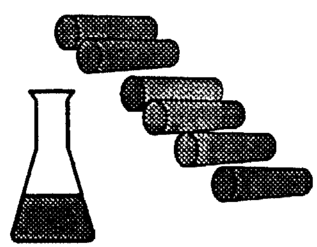
Part 2 Summary → *W. Avery*

Objective: Identify Critical Impact Threats for Fuselage



Impact Designed Experiment Involving Multiple Variables

Material variables



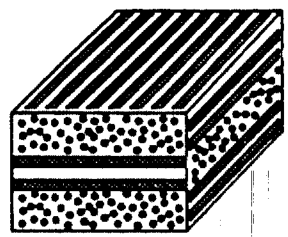
Fiber
- AS4
- IM7

Resin
- 938 (3501-6)
- 977-2

Fiber volume
- 0.480
- 0.565

Material form
- Tape
- Tow

Laminate variables

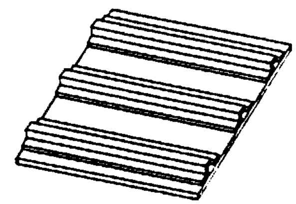


Stiffener layup
- Hard
- Soft

Skin layup
- Hard
- Soft

Thickness
- Thick (approximately 0.2 in)
- Thin (approximately 0.1 in)

Structural variables

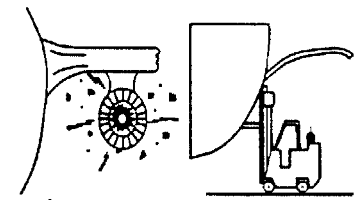


Stiffener type
- Blade
- Hat

Stiffener spacing
- 7 in
- 12 in

Stiffener adhesive layer
- With
- Without

Extrinsic variables



Impact mass
- 0.5 lbm
- 12.0 lbm

Impact energy (skin/stiffener)
- 80 in-lb/200 in-lb
- 1,200 in-lb/2,000 in-lb

Impact temperature
- 70°F
- 180°F

Impact diameter
- 0.25 in
- 1.0 in

Impactor tup shape
- Flat
- Spherical

Impactor stiffeness
- 0.5 Msi
- 30 Msi

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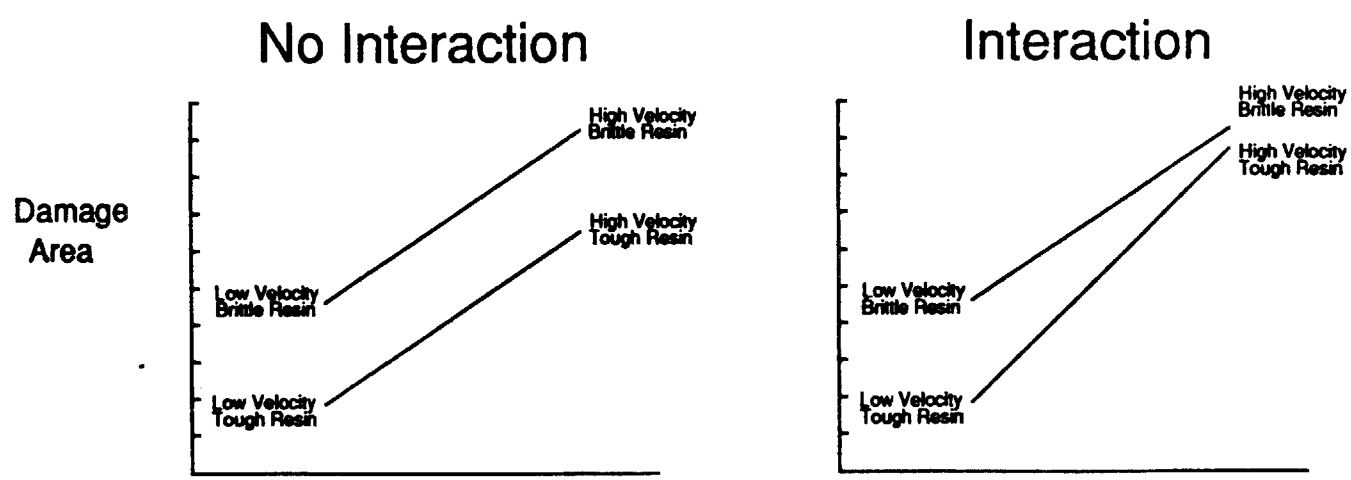
ATCAS

DL1623.11 DR1s

THE DESIGN of EXPERIMENTS (DOE) TECHNIQUE

Example: 4 Variables		
Change One Variable (at a time) 5 Tests Main Effects No Interactions	Fully Crossed Experiment 16 Tests Main Effects All Interactions	Fractional Factorial Designed Experiment 8 Tests Main Effects Interactions



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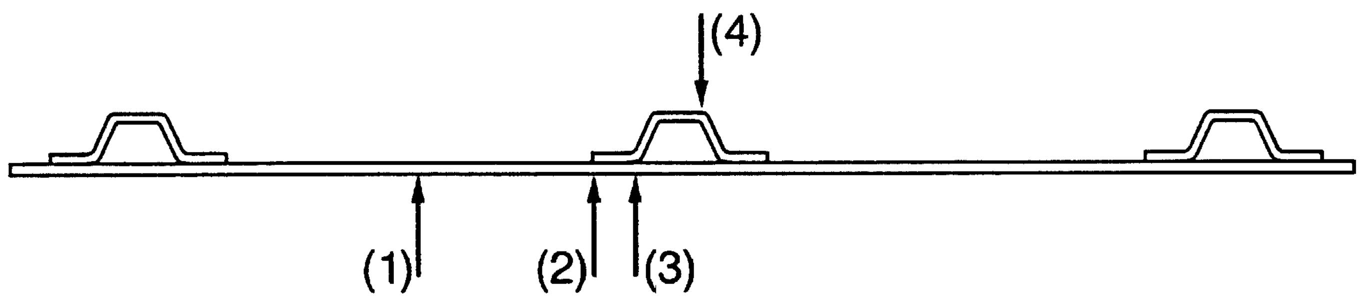
THE DESIGN of EXPERIMENTS (DOE) TECHNIQUE

Summary for 16 Variables

One Variable at a Time:	17 Tests
Fully Crossed Experiment:	65,536 Tests
Fractional Factorial:	**32 Tests**

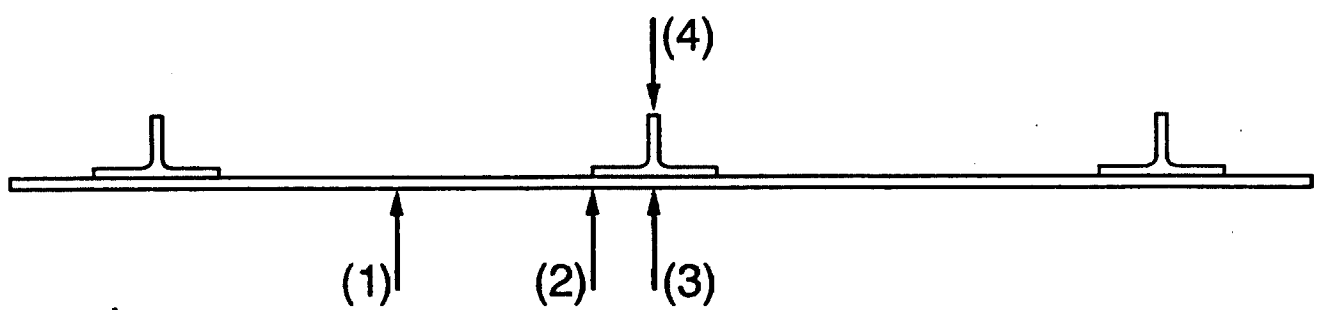
dec1189e

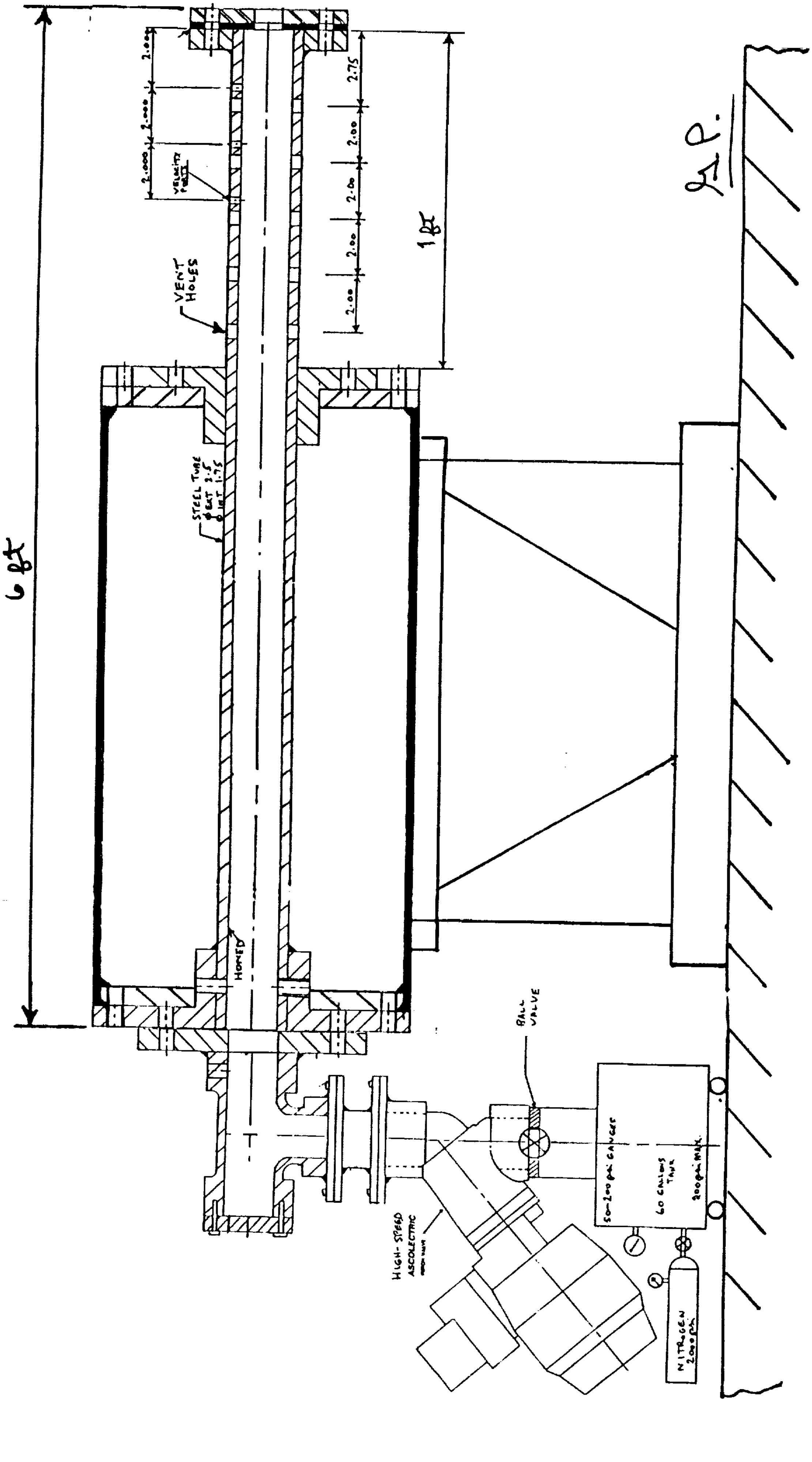
Critical Impact Locations



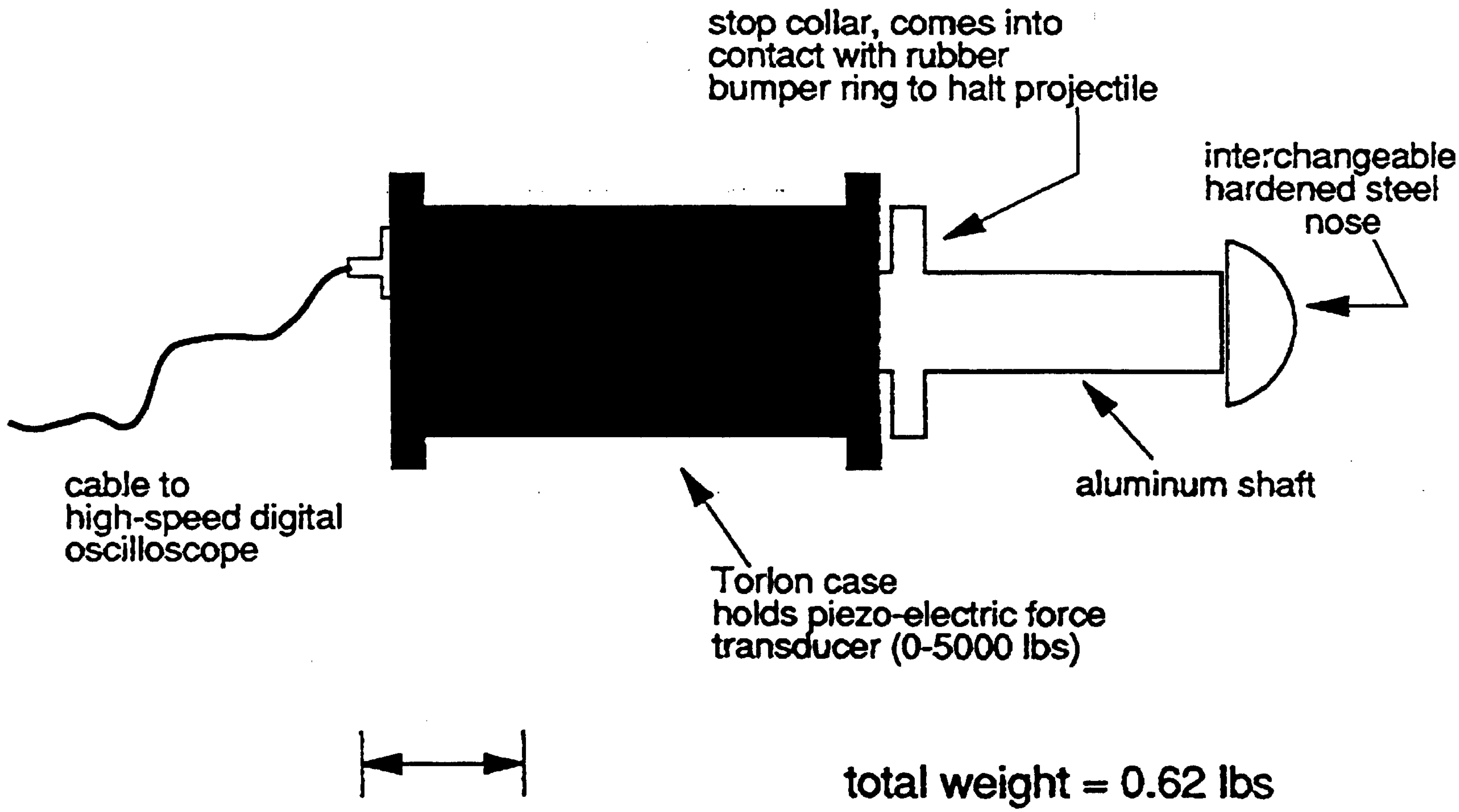
(1) Skin midbay
(2) Edge of stiffener attachment flange

(3) Base of stiffener web
(4) Top of stiffener




6 ft
1 ft
VENT HOLES
VELOCITY PORTS
STEEL TUBE
HONED
BALL VALVE
HIGH-SPEED ASCOLECTRIC
60 GALLONS TANK
NITROGEN

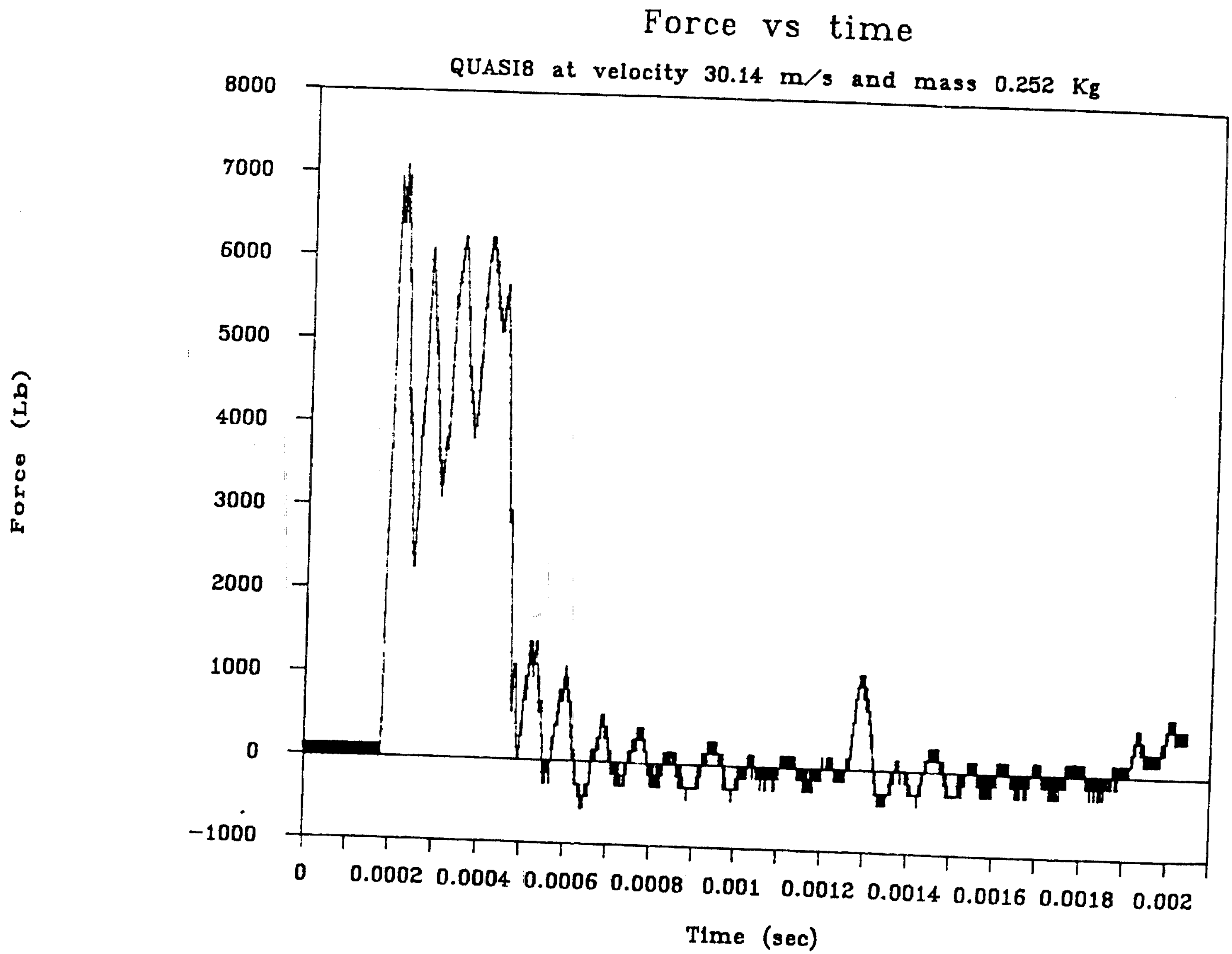
High Velocity, Low Mass, Instrumented Tup



total weight = 0.62 lbs

dynamic calibration factor = 1.65

UBC
Mar 14/91


Force vs time
QUASI8 at velocity 30.14 m/s and mass 0.252 Kg
Force (Lb)
8000
7000
6000
5000
4000
3000
2000
1000
0
−1000
0
0.0002
0.0004
0.0006
0.0008
0.001
0.0012
0.0014
0.0016
0.0018
0.002
Time (sec)

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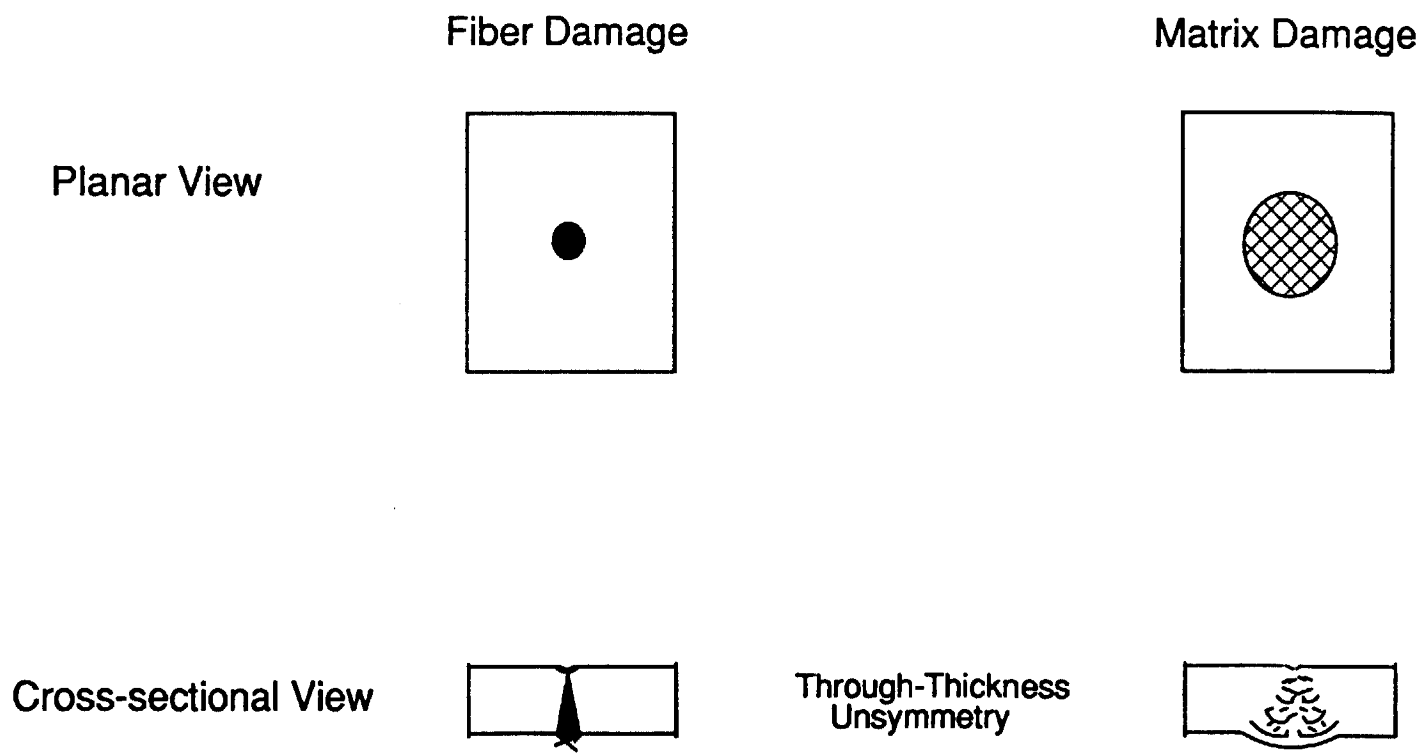
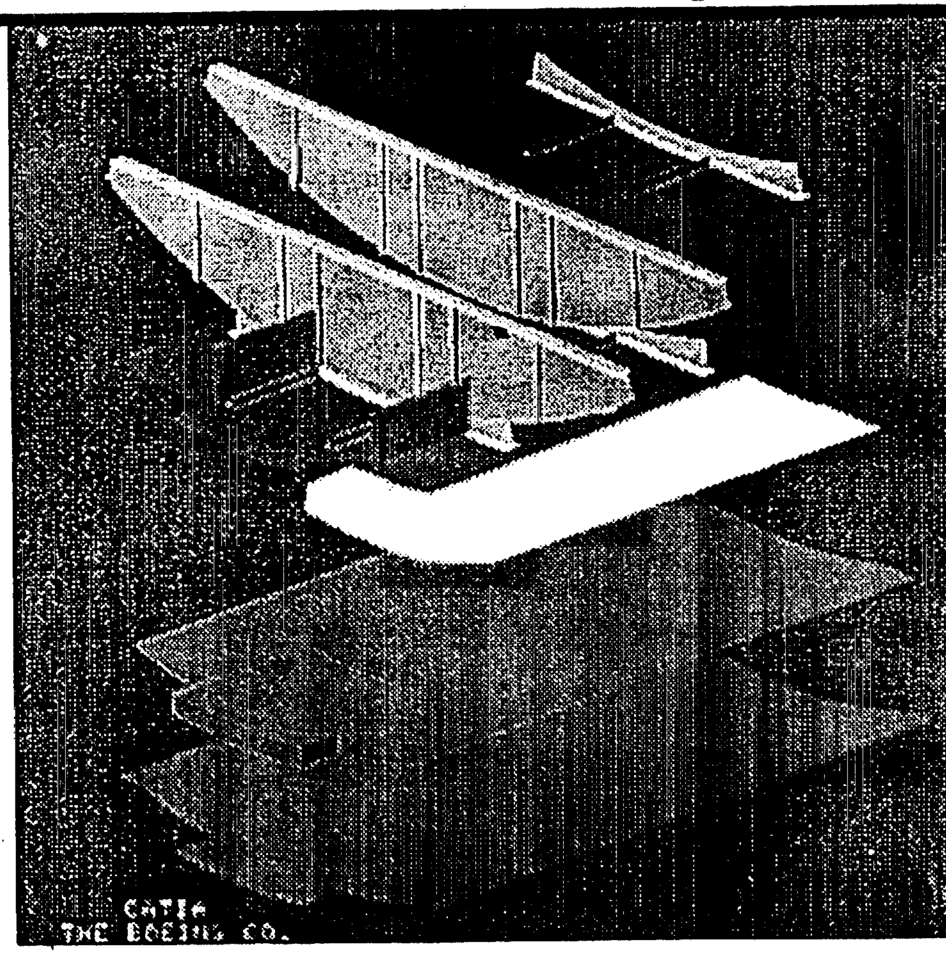
IMPACT DAMAGE ZONE MEASURED RESPONSE VARIABLES

Overall Damage Geometry and Distribution



First Keel Design



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ATCAS

A1623B.54 D

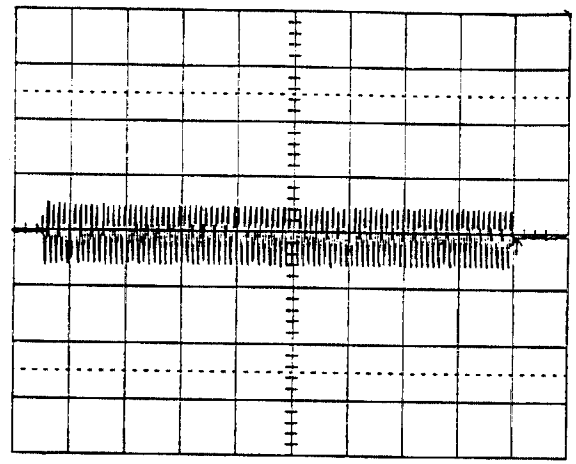
1st Sandwich Impact Damage Resistance Designed Experiment

Variables Studied:
- Facesheet Thickness
- Core Density
- Fabrication Sequence
- Facesheet Material

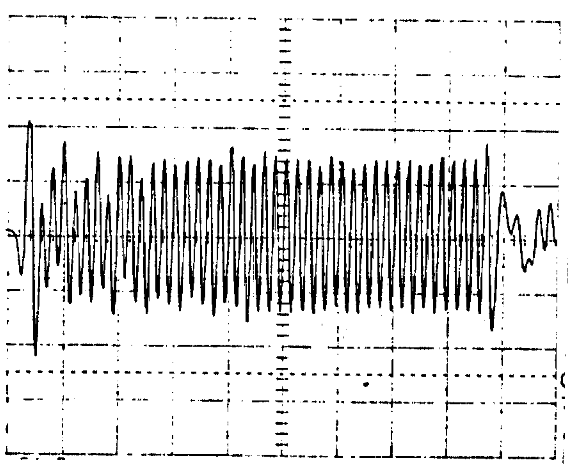
Run Number	Laminate Thickness	Core Density	Fabrication Sequence	Facesheet Material
1	0.045	110 WF	Precured Skin	IM7/938
2	0.045	110 WF	Cocured Skin	IM7/8551-7
3	0.045	200 WF	Cocured Skin	IM7/938
4	0.045	200 WF	Precured Skin	IM7/8551-7
5	0.134	200 WF	Precured Skin	IM7/938
6	0.134	200 WF	Cocured Skin	IM7/8551-7
7	0.134	110 WF	Cocured Skin	IM7/938
8	0.134	110 WF	Precured Skin	IM7/8551-7

Thru-Transmission Ultrasound - Signal Response
Rohacell Foam Sandwich Panels

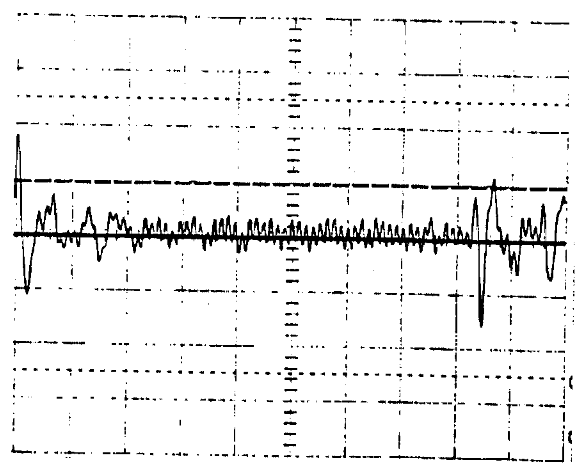
Sample Input Signal



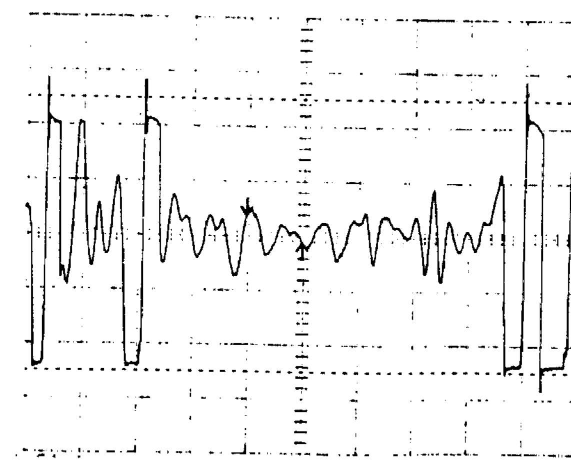
Recieved Signals



500 KHZ



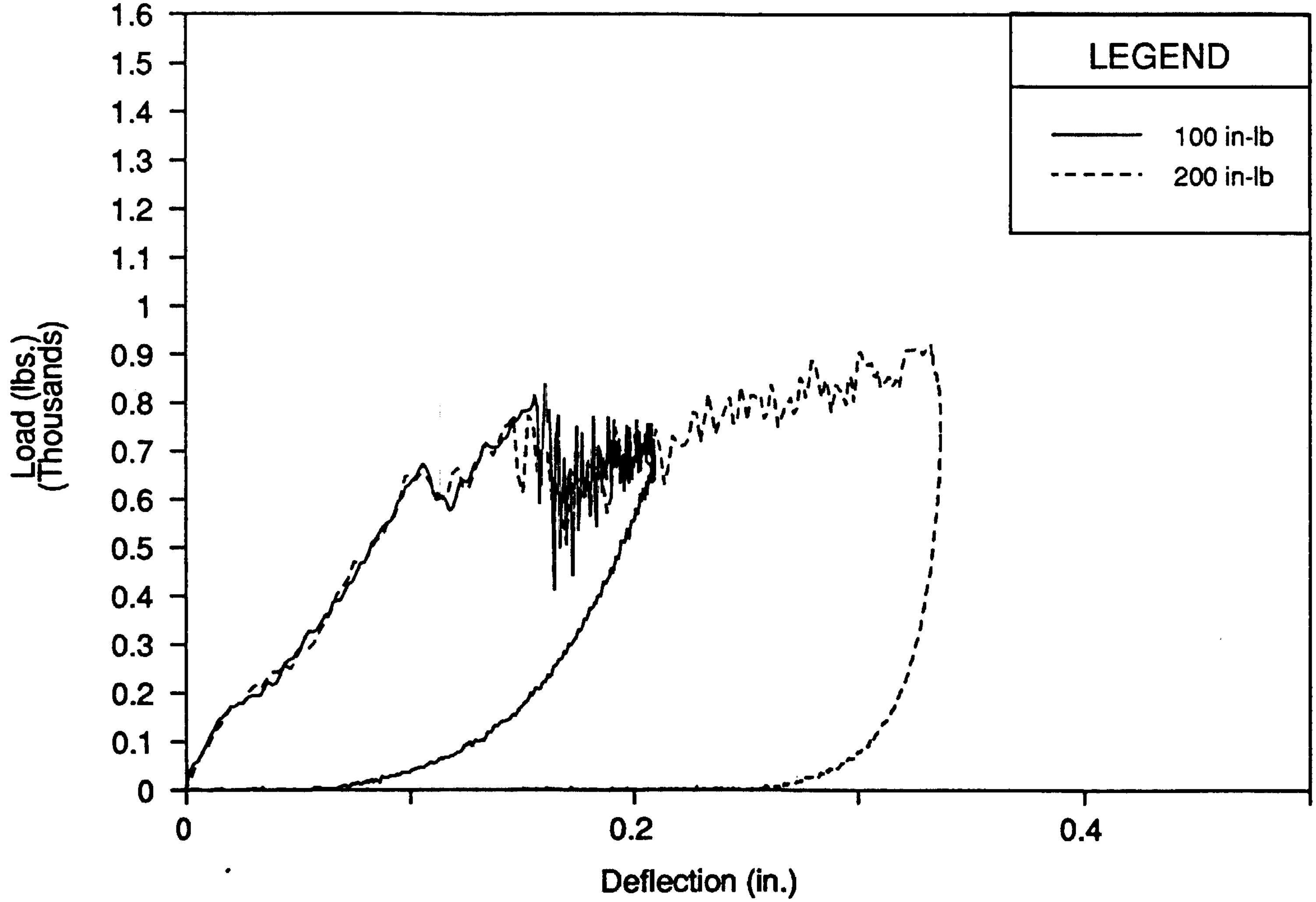
750 KHZ

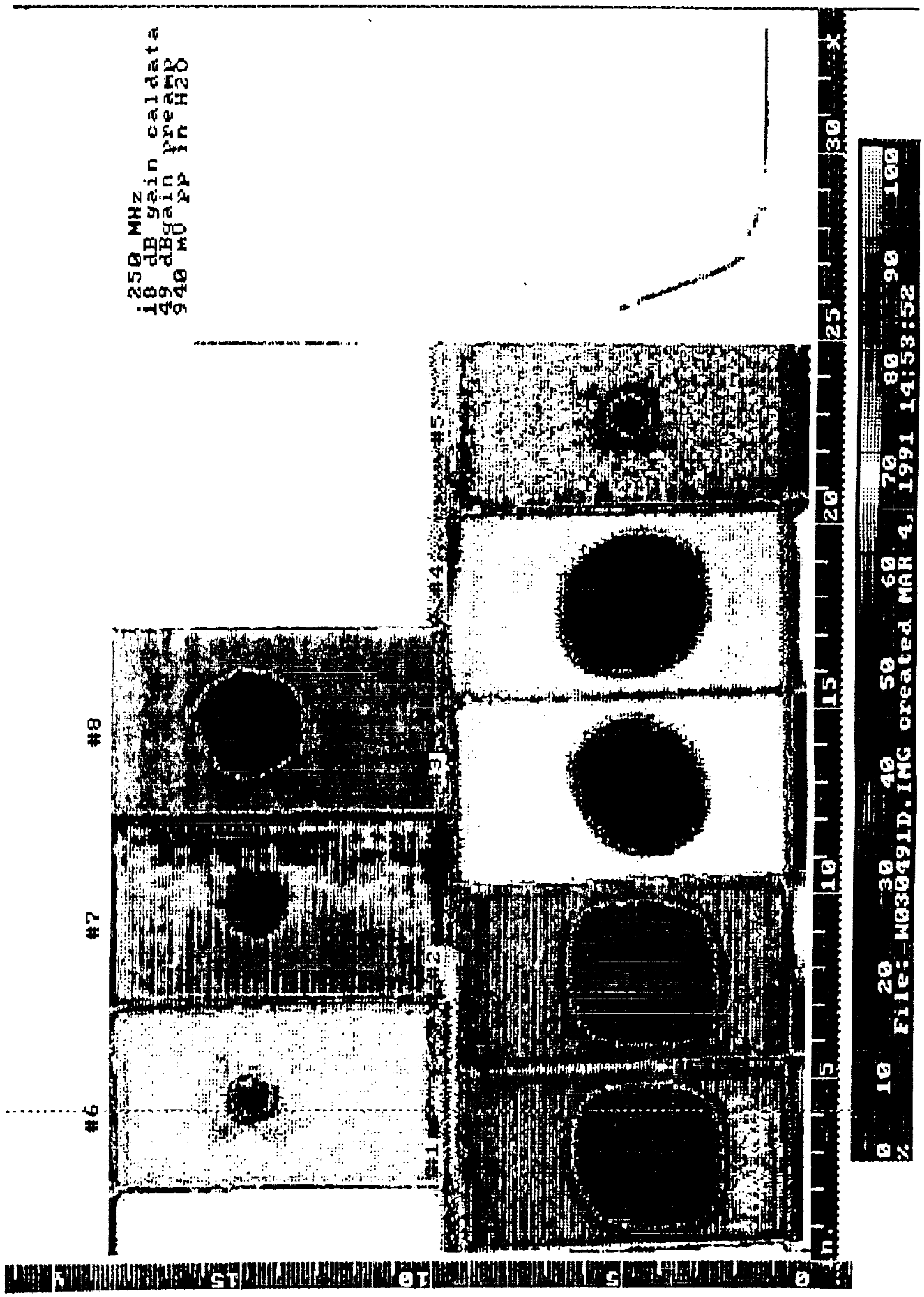


1000 KHZ

IMPACT LOAD vs. DEFLECTION

Facesheet: IM7/8551-7, Thickness = .088 in.   Core Density: 110 WF





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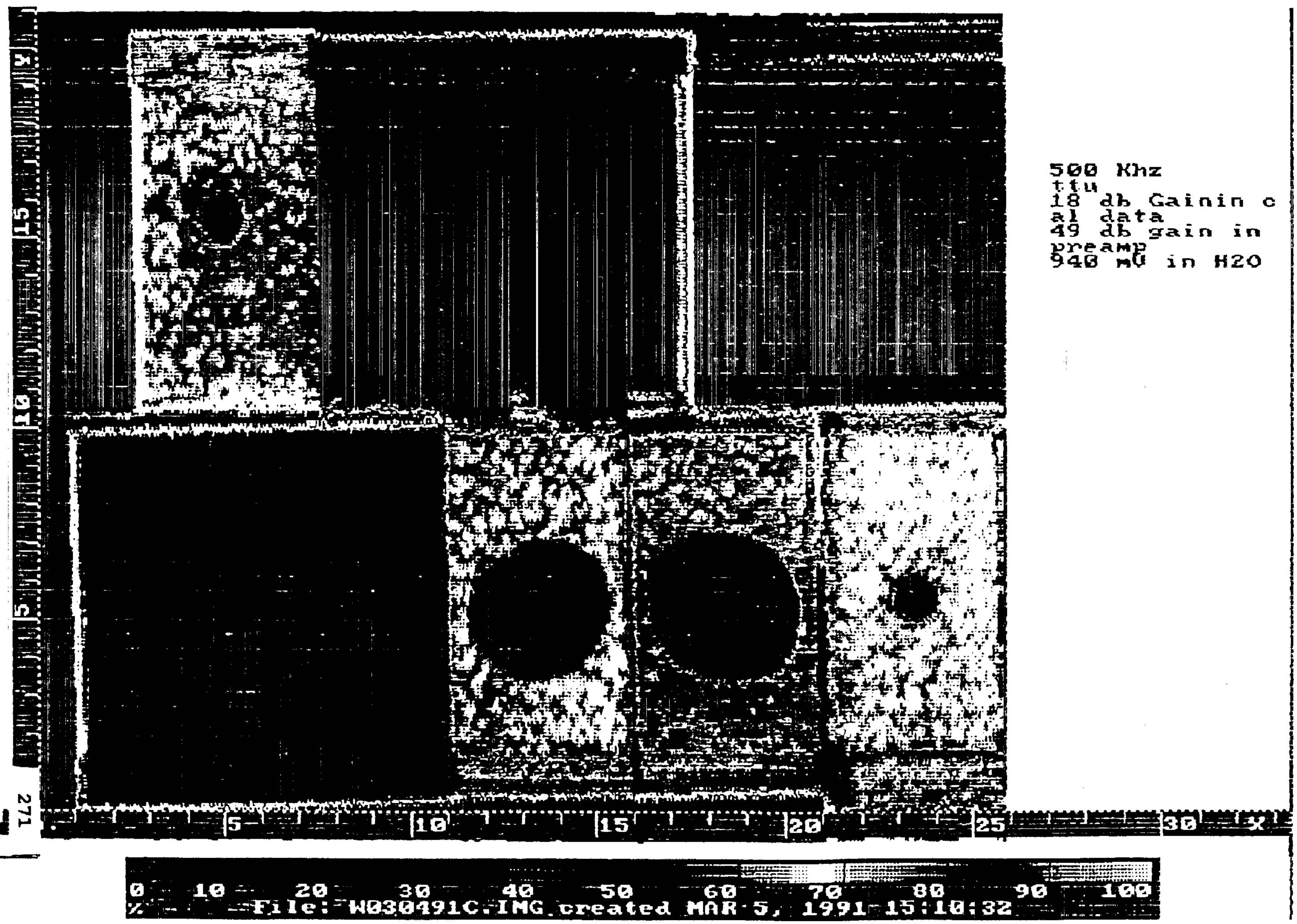

500 Khz
ttu
18 db Gainin c
al data
49 db gain in
preamp
940 mV in H2O
Y
5
10
15
5
10
15
20
25
30
X
0 10 20 30 40 50 60 70 80 90 100
%
File: W030491C.IMG created MAR 5, 1991 15:10:32

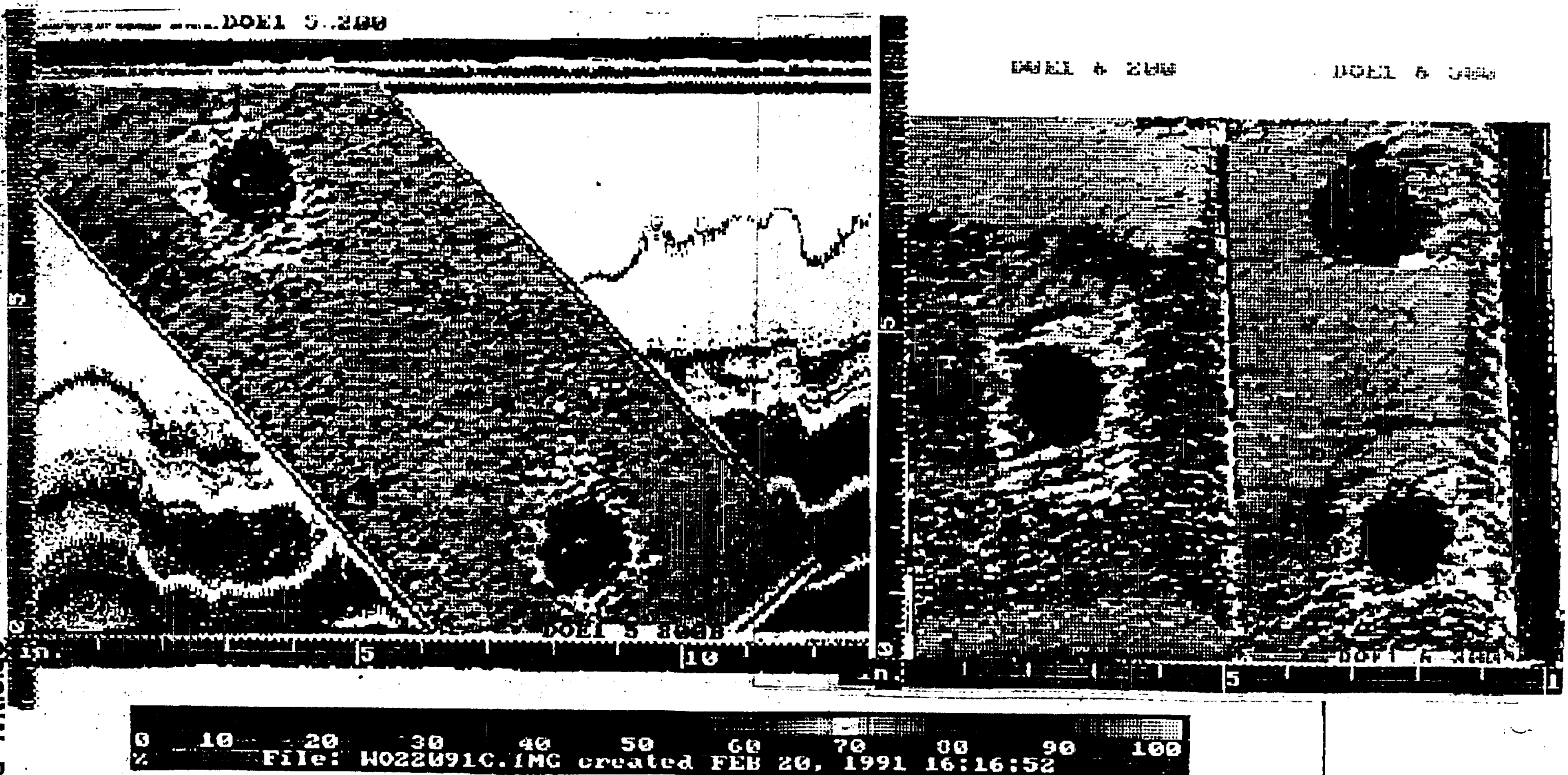
5 MHZ Pulse-Echo Time of Flight C-Scans
Rohacell Foam Sandwich Panels

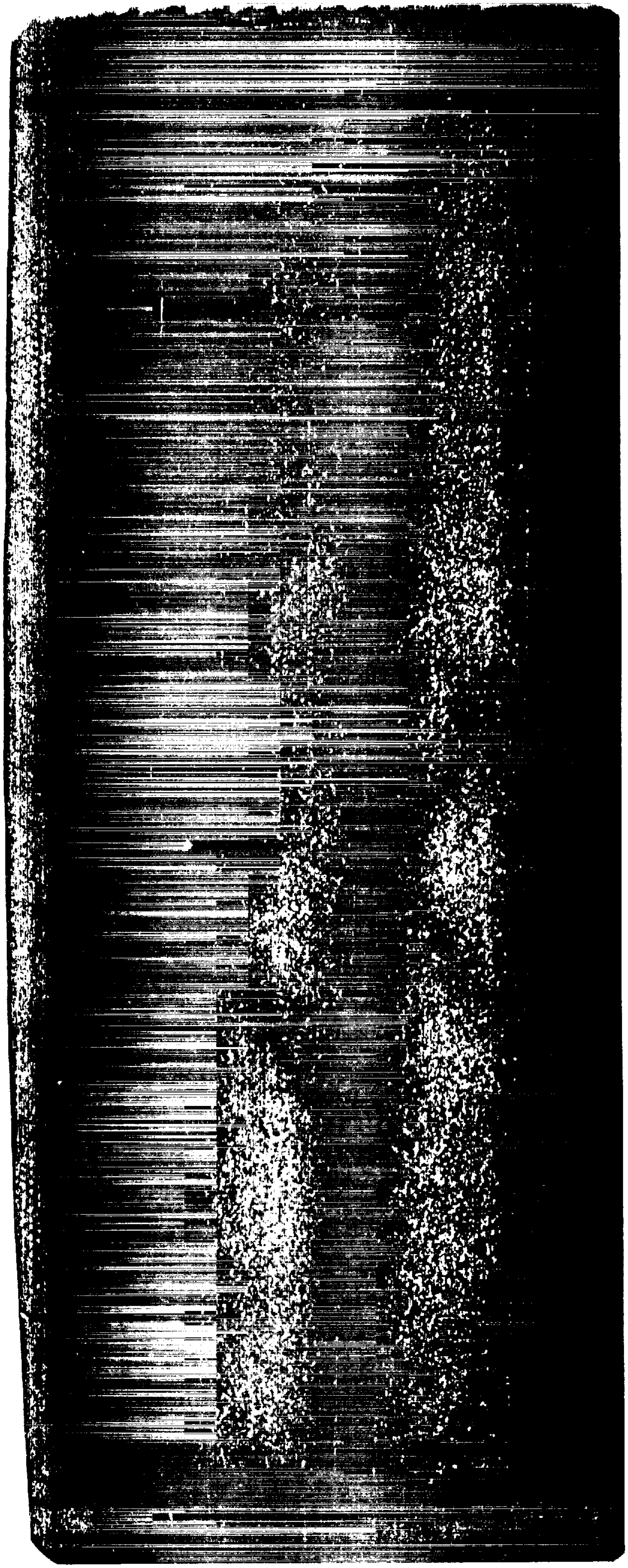
Facesheet: IM7/938

Core Density: 13 lb/ft

Facesheet: IM7/8551-7

Core Density: 13 lb/ft





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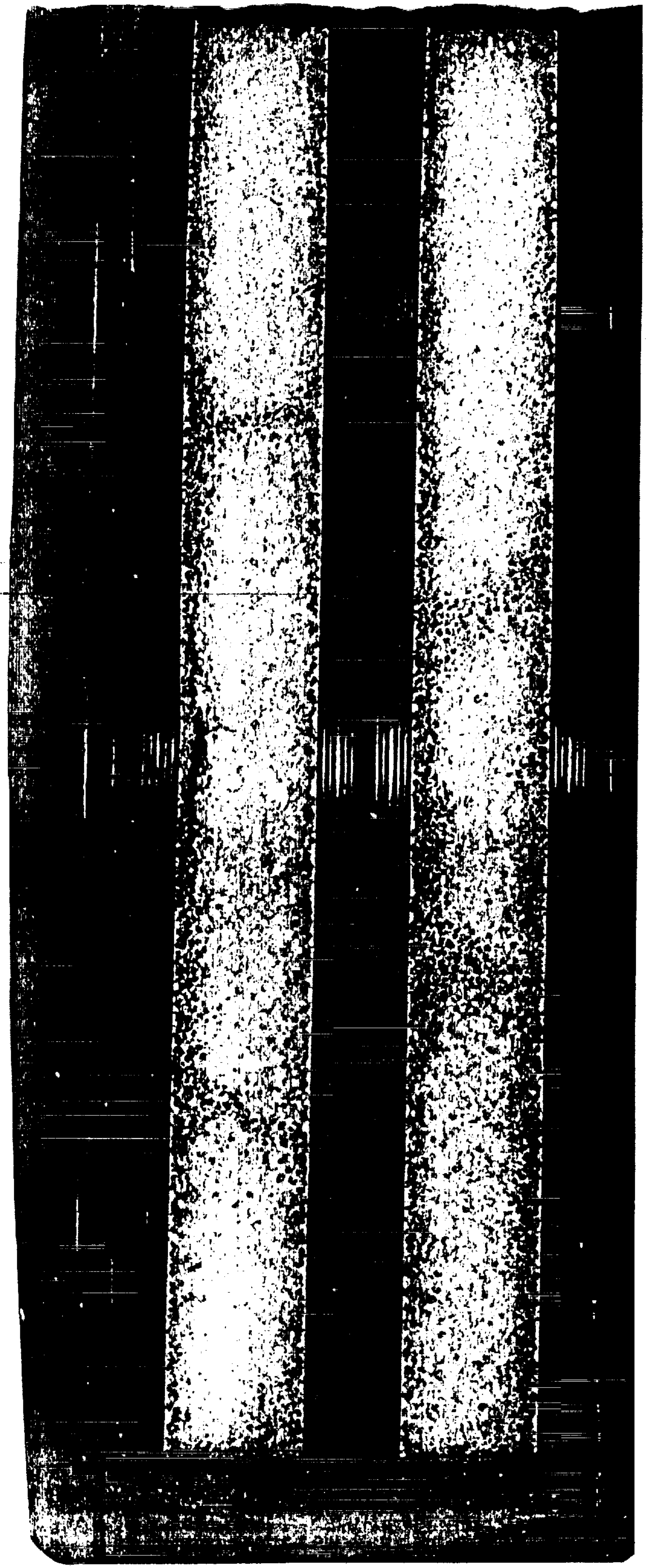
1-2-2115-B

1-8.200 B-G



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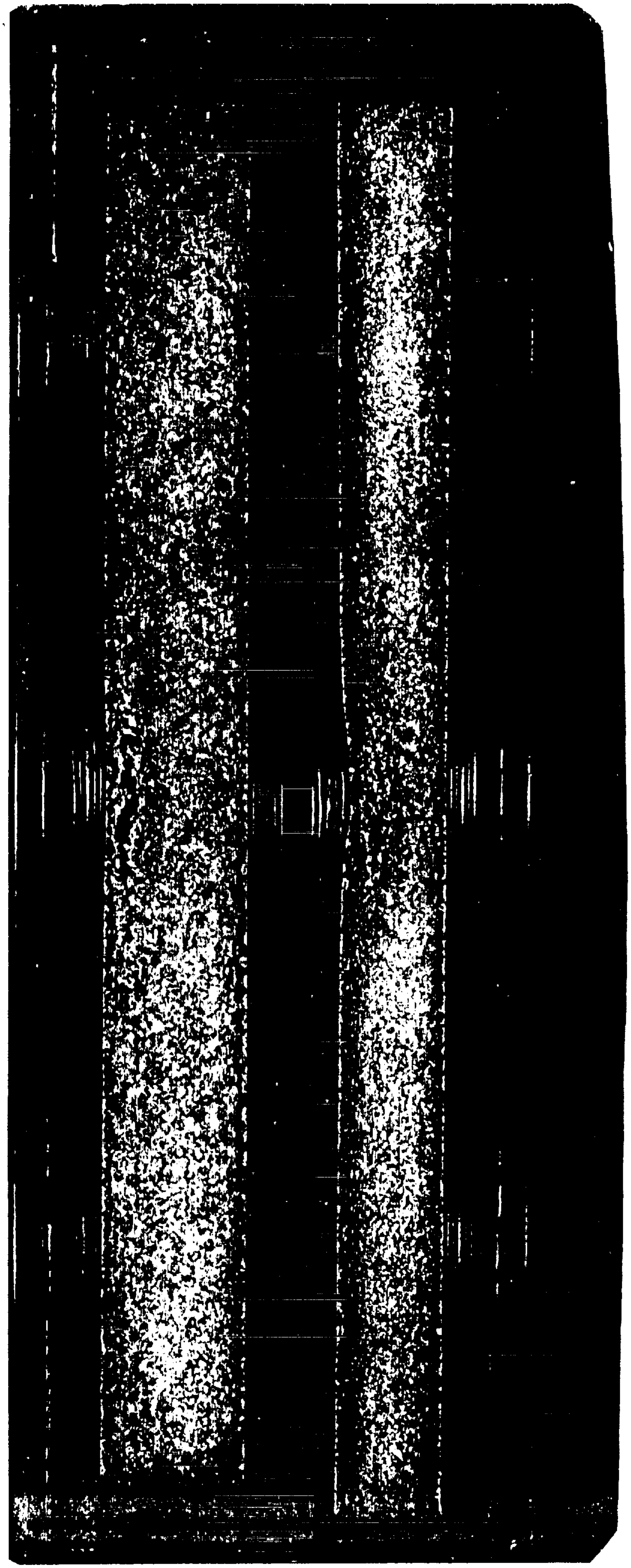

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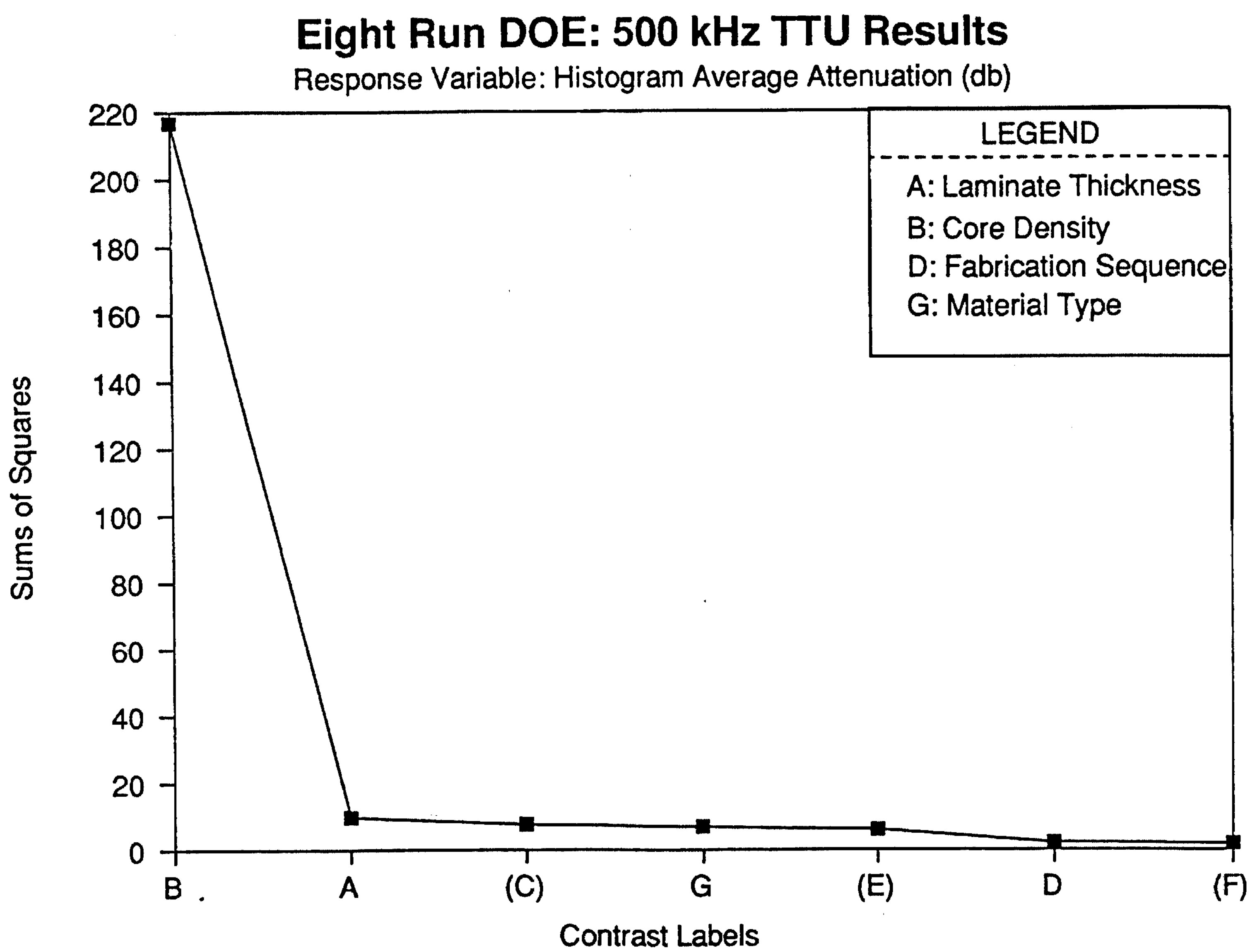
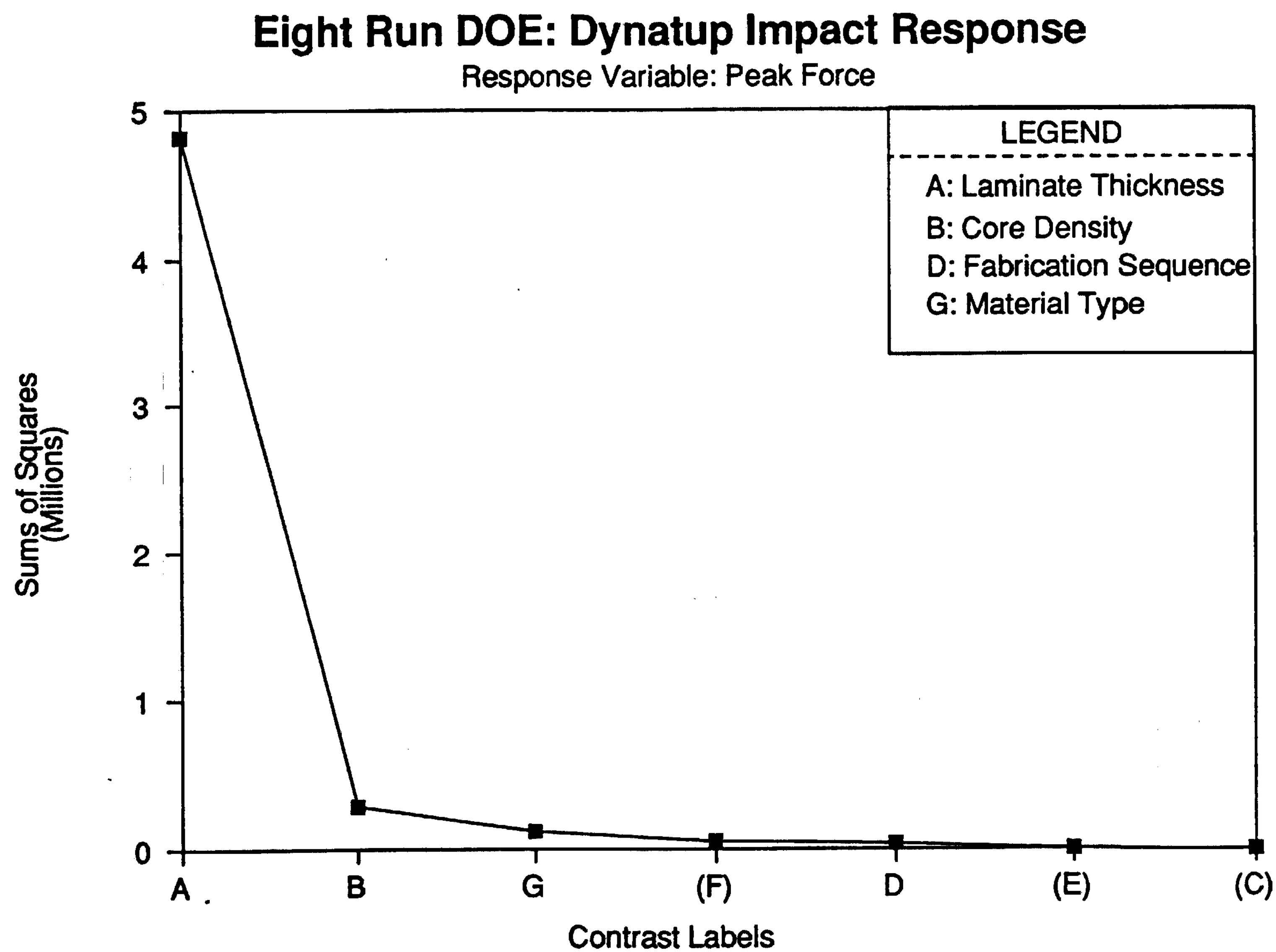
1-6-20xb-4

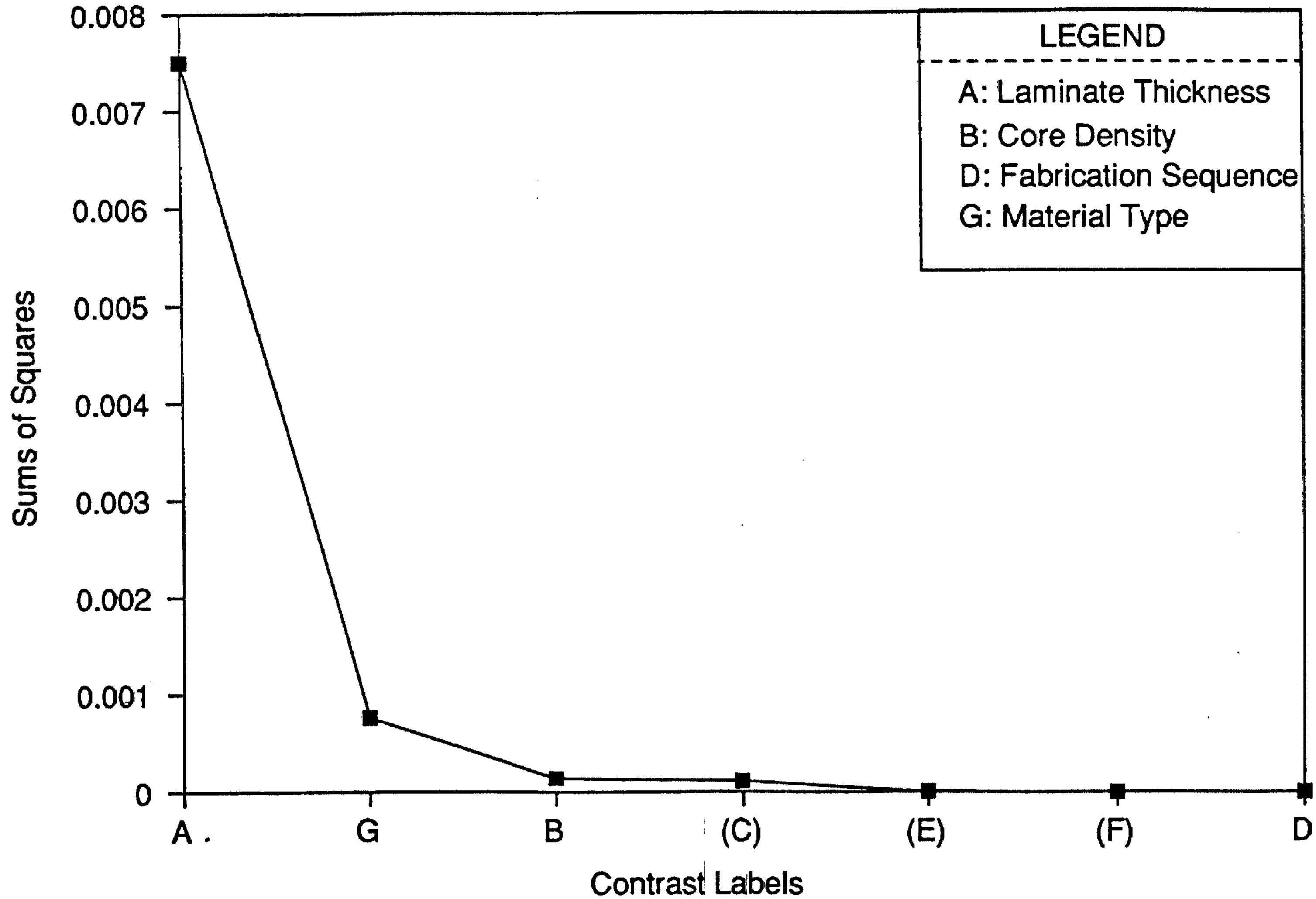

Eight Run DOE: 500 kHz TTU Results
Response Variable: Histogram Average Attenuation (db)
Sums of Squares
220
200
180
160
140
120
100
80
60
40
20
0
B
A
(C)
G
(E)
D
(F)
Contrast Labels
LEGEND
A: Laminate Thickness
B: Core Density
D: Fabrication Sequence
G: Material Type

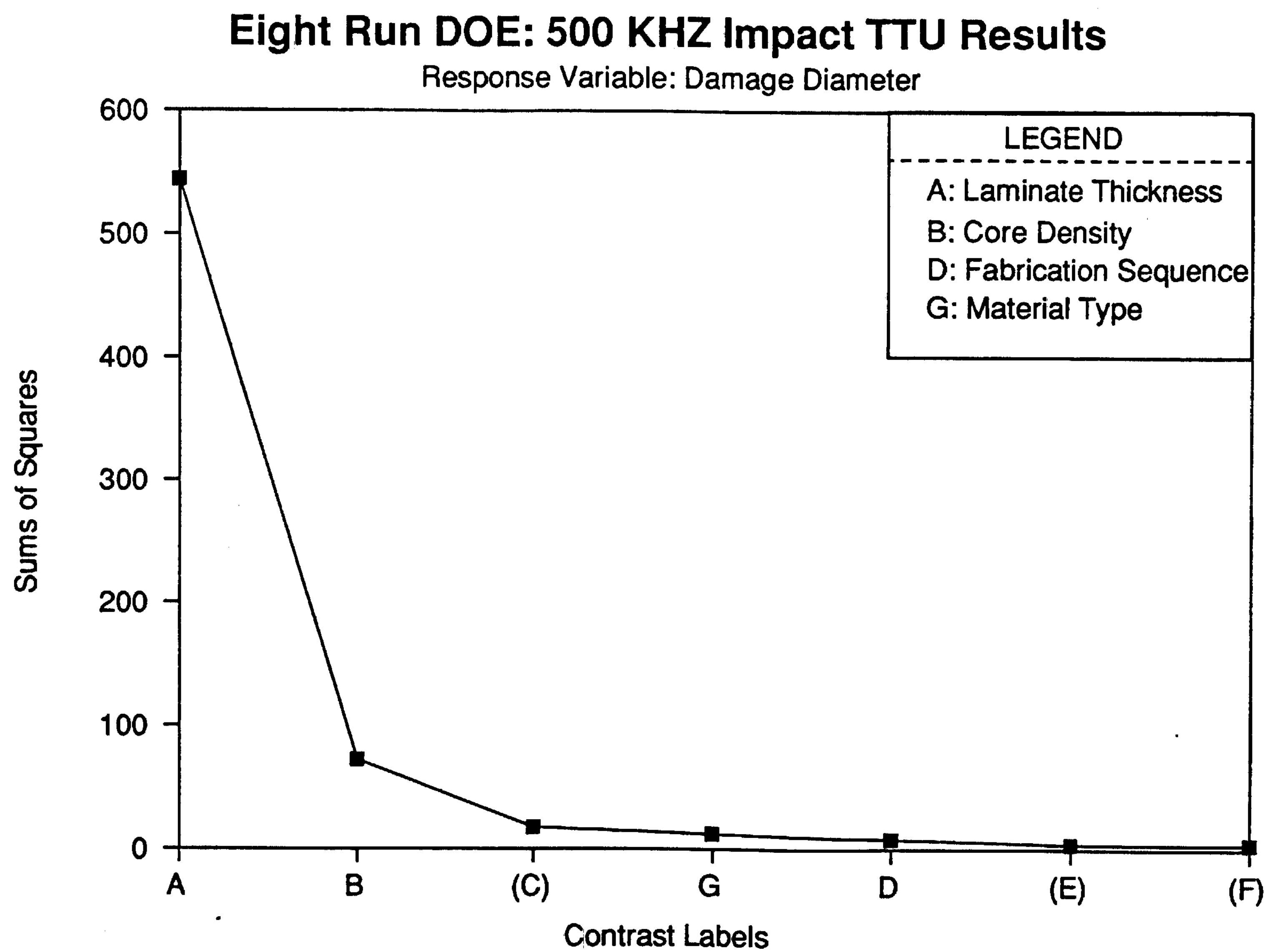


Eight Run DOE: Dynatup Impact Response
Response Variable: Peak Force
LEGEND
A: Laminate Thickness
B: Core Density
D: Fabrication Sequence
G: Material Type
Sums of Squares (Millions)
0
1
2
3
4
5
A
B
G
(F)
D
(E)
(C)
Contrast Labels

Eight Run DOE: Dynatup Impact Data

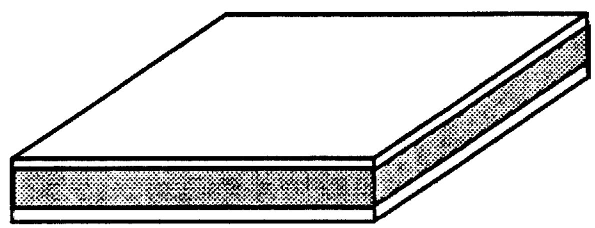
Response Variable: Dent Depth



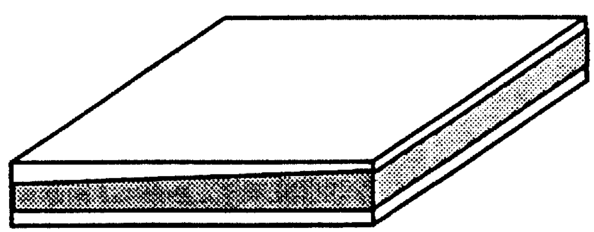

Eight Run DOE: 500 KHZ Impact TTU Results
Response Variable: Damage Diameter
LEGEND
A: Laminate Thickness
B: Core Density
D: Fabrication Sequence
G: Material Type
Sums of Squares
600
500
400
300
200
100
0
A
B
(C)
G
D
(E)
(F)
Contrast Labels

Impact Damage Resistance and Residual Strength of Sandwich Structure for Aircraft Fuselage

Constant gauge



Tapered



Variables

- Core type: Foam versus honeycomb
- Core density: 6 to 18 lb/ft^3
- Core thickness: 0.25 to 0.75 in
- Face sheet thickness: 0.1 to 0.3 in
- Fiber type: AS4 versus IM7
- Matrix type: 3501-6 versus 8551-7
- Impactor shape: Flat versus spherical
- Impactor diameter: 0.25 to 1.0 in

- Laminate layup: Hard versus soft
- Fiber type: AS4 versus IM7
- Matrix type: 938 versus 977-2
- Fiber volume: 0.480 versus 0.565
- Material form: Tow versus tape

Impact Damaged Composites, Part 1: Damage Simulation and Strength Predictions

- Transport Fuselage Structures → *L. Ilcewicz*
- A Need to Understand Fundamentals → *L. Ilcewicz*

- Types of Impact Damage States → *E. Dost*
- Damage Simulation & CAI Strength Prediction → *E. Dost*
- Part 1 Summary → *E. Dost*

Impact Damaged Composites, Part 2: Standard Tests for Fuselage Structural Issues

- Fuselage Impact Damage Characterization → *E. Dost*
- Standard Specimen Tests for Fuselage Applications → *W. Avery*
- Part 2 Summary → *W. Avery*

OUTLINE

- COMPRESSION AFTER IMPACT
- TENSION AFTER THROUGH PENETRATION
- COMPRESSION AFTER THROUGH PENETRATION
- OTHER MATERIAL SCREENING NEEDS
- SUMMARY

COMPRESSION AFTER IMPACT

ARE EXISTING SPECIMEN GEOMETRIES AND STANDARD TEST PROCEDURES SUFFICIENT?

YES, IF YOU UNDERSTAND:

- THE IMPACT INDUCED DAMAGE STATE
- SPECIMEN BUCKLING LIMITS
- FINITE WIDTH CORRECTION FACTORS
- IMPACTOR SIZE, SHAPE, WEIGHT, AND STIFFNESS EFFECTS
- CAI FAILURE MODES
- PLY THICKNESS AND STACKING SEQUENCE EFFECTS
- SUBLAMINATE STABILITY
- SPECIMEN CURVATURE EFFECTS
- ROLL OF UNDAMAGED COMPRESSION STRENGTH
- THE EFFECT OF STIFFENERS
- AND THE INTER-RELATIONSHIPS BETWEEN ALL OF THE ABOVE

NO / MAYBE, IF YOU INCLUDE:

- COMPRESSION AFTER THROUGH PENETRATION
- SHEAR AFTER IMPACT
- COMBINED LOADING
- TAPERED THICKNESS

IMPACT VARIABLES

- STATIC INDENTATION VS. DROP WEIGHT VS. CONTROLLED VELOCITY
  - RATE SENSITIVE MATERIALS?
  - DAMAGE STATE CHANGE WITH IMPACT RATE?

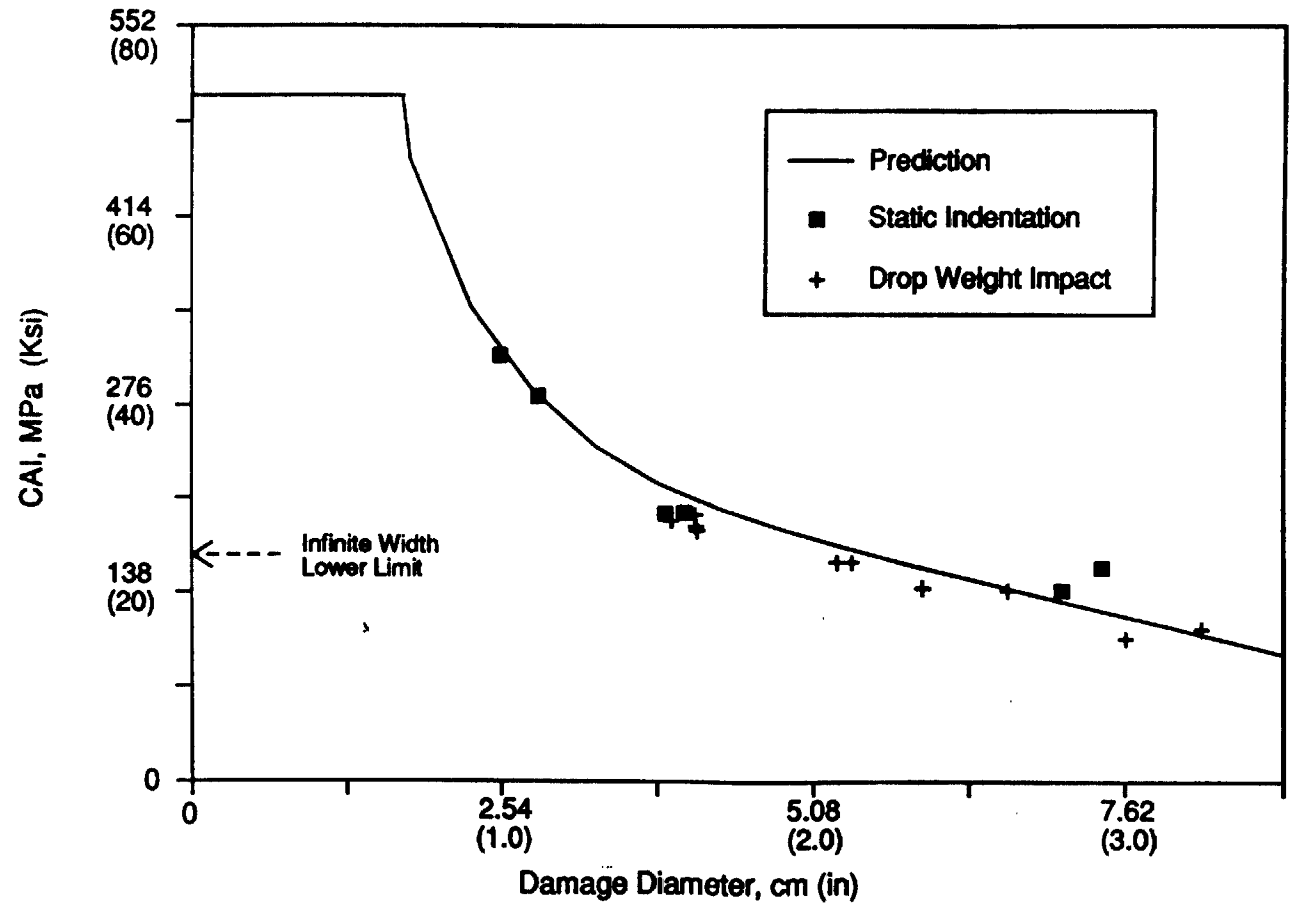
- IMPACTOR VARIABLES
  - GEOMETRY (DIAMETER, SHAPE)
    DAMAGE STATE CHANGES WITH IMPACTOR DIAMETER
  - MASS
  - STIFFNESS

  LEAD
  STEEL
  NYLON

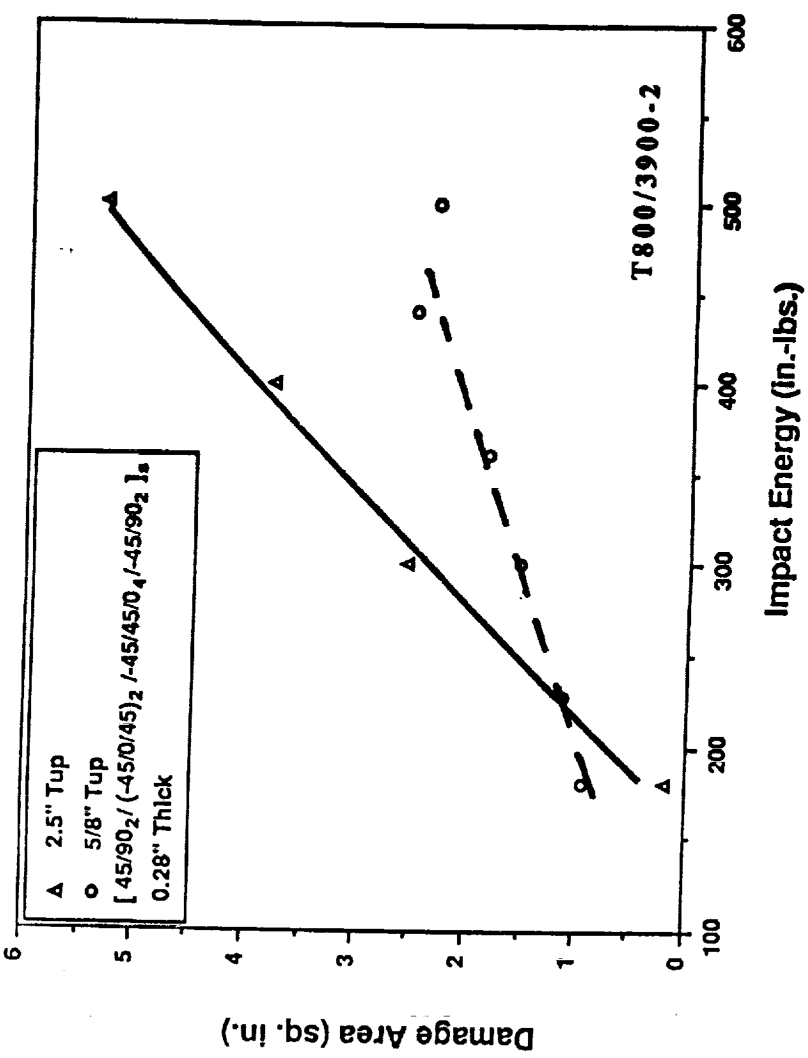
- RANGE OF IMPACT ENERGIES
  - NEED TO UNDERSTAND VISIBILITY AS A FUNCTION OF ENERGY
  - LOW ENERGY (SMALL DAMAGE) VS. HIGH ENERGY (LARGE DAMAGE)
    - DAMAGE STATE CHANGES
    - FAILURE MECHANISMS CHANGE

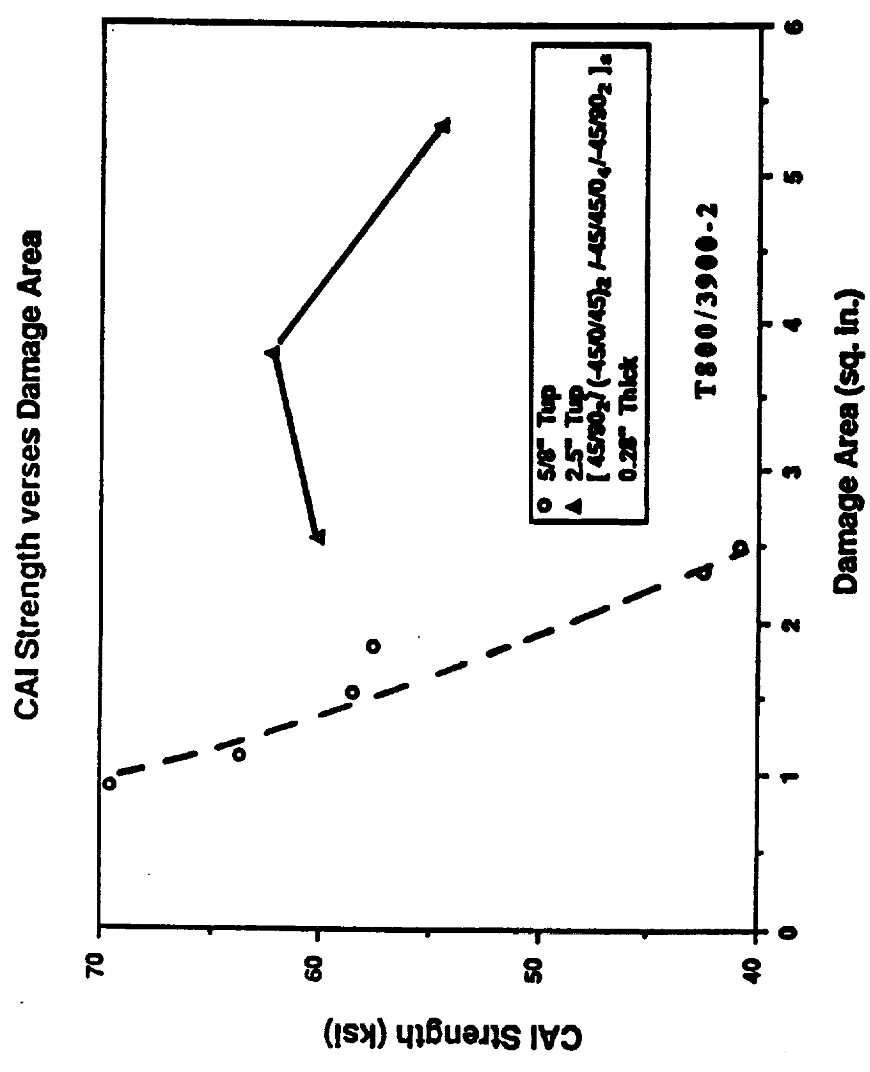
C-4

Predictions and experimental results for 12.7 cm wide specimens, AS6/3501-6,$(45/0/-45/90)_{5S}$,and ply thickness = 0.0188 cm.



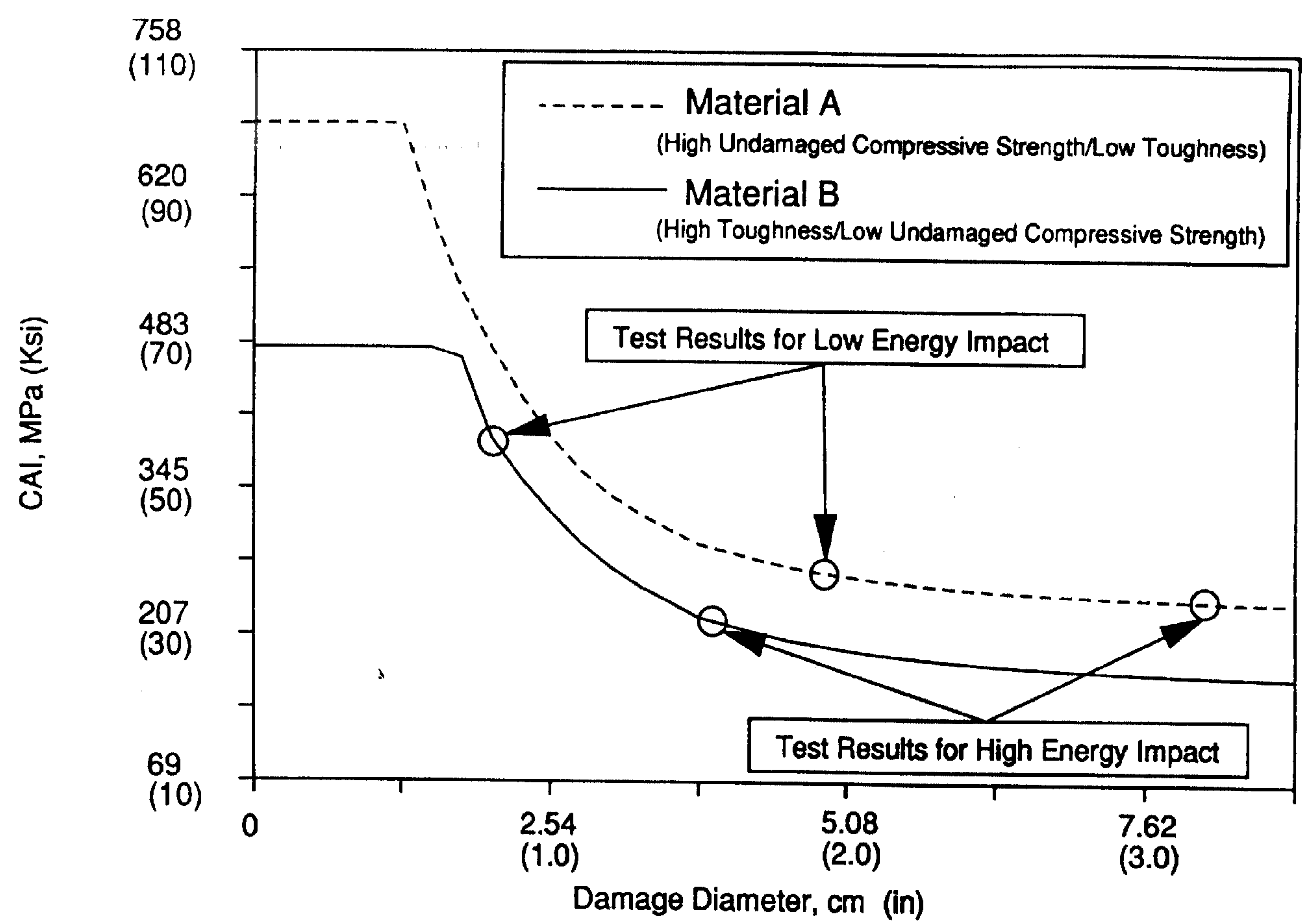
Impact Energy as a Function of
Damage Area





CAI Strength verses Damage Area
5/8" Tup
2.5" Tup
[45/90$_2$/(-45/0/45)$_2$ /-45/45/0$_4$/-45/90$_2$]s
0.28" Thick
T800/3900-2
Damage Area (sq. in.)
CAI Strength (ksi)
0
1
2
3
4
5
6
40
50
60
70

Theoretical damage tolerance curves for two material types.



NEED FOR ENHANCED DATA EVALUATION

- INSTRUMENTED IMPACT DATA
  - FORCE-TIME CURVES
  - FORCE-DISPLACEMENT CURVES
- DAMAGE CHARACTERIZATION
  - FIBER FAILURE VS. DELAMINATION
  - NDE TECHNIQUES: PULSE-ECHO TIME-OF-FLIGHT C-SCAN
- SUPPORTING ANALYSIS
  - SPECIMEN BUCKLING LIMITS
  - FINITE WIDTH EFFECTS VS. DAMAGE SIZE
- FAILURE MODE IDENTIFICATION

Example of Instrumented Impact Results and Application To Impact Modeling

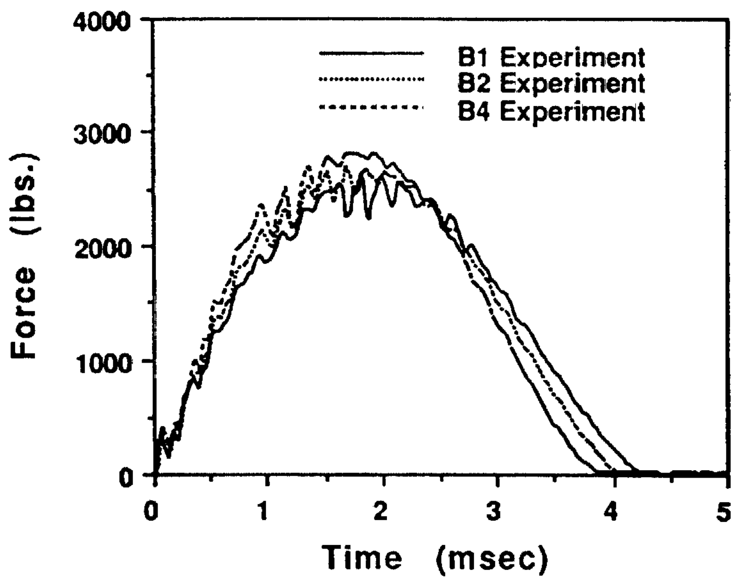


Figure 10. Instrumented Impact Results for the Resin Content Study

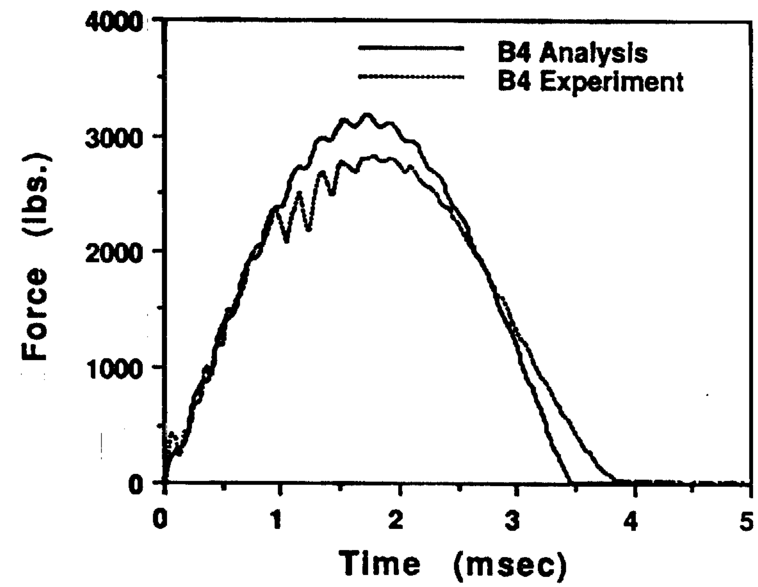


Figure 11. Predicted and Experimental Results for a Specimen from Batch B4, Impacted at an Energy of 180 in-lb.

T800/3900-2

B1: Low Resin Content

B2: Intermediate Resin Content

B4: High Resin Content

Ref.: Dodd Grande, et al., "Effects of Intra- and Inter-Laminar Resin Content on the Mechanical Properties of Toughened Composite Materials." NASA Conference Publication 3104, pp. 455-476.

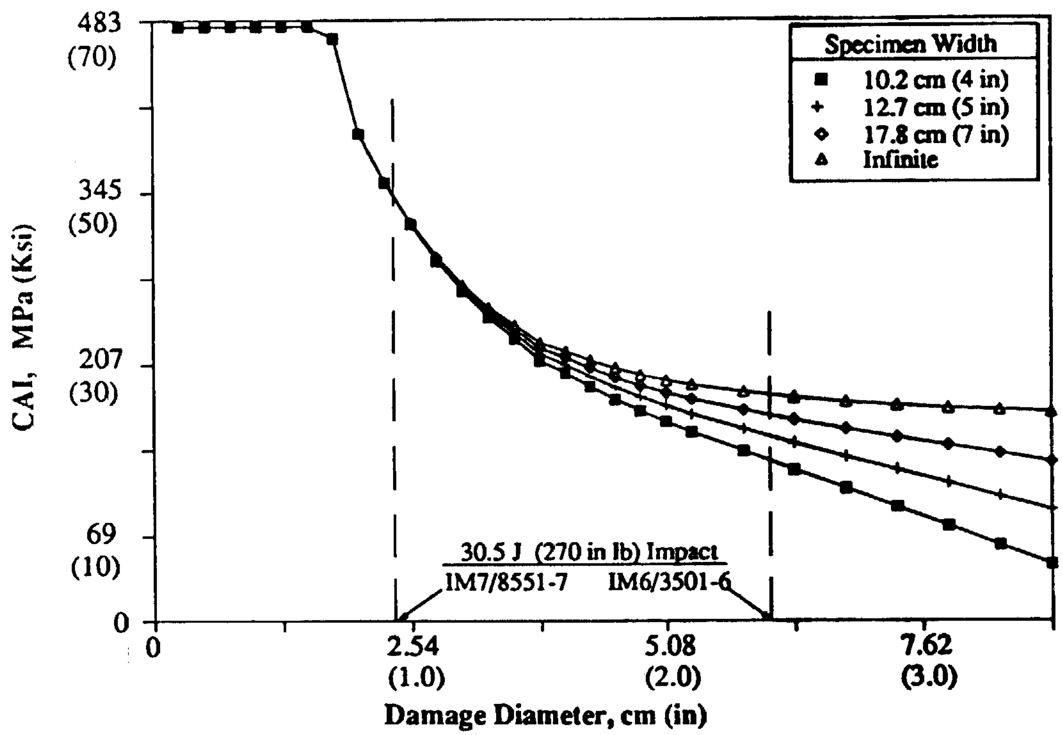
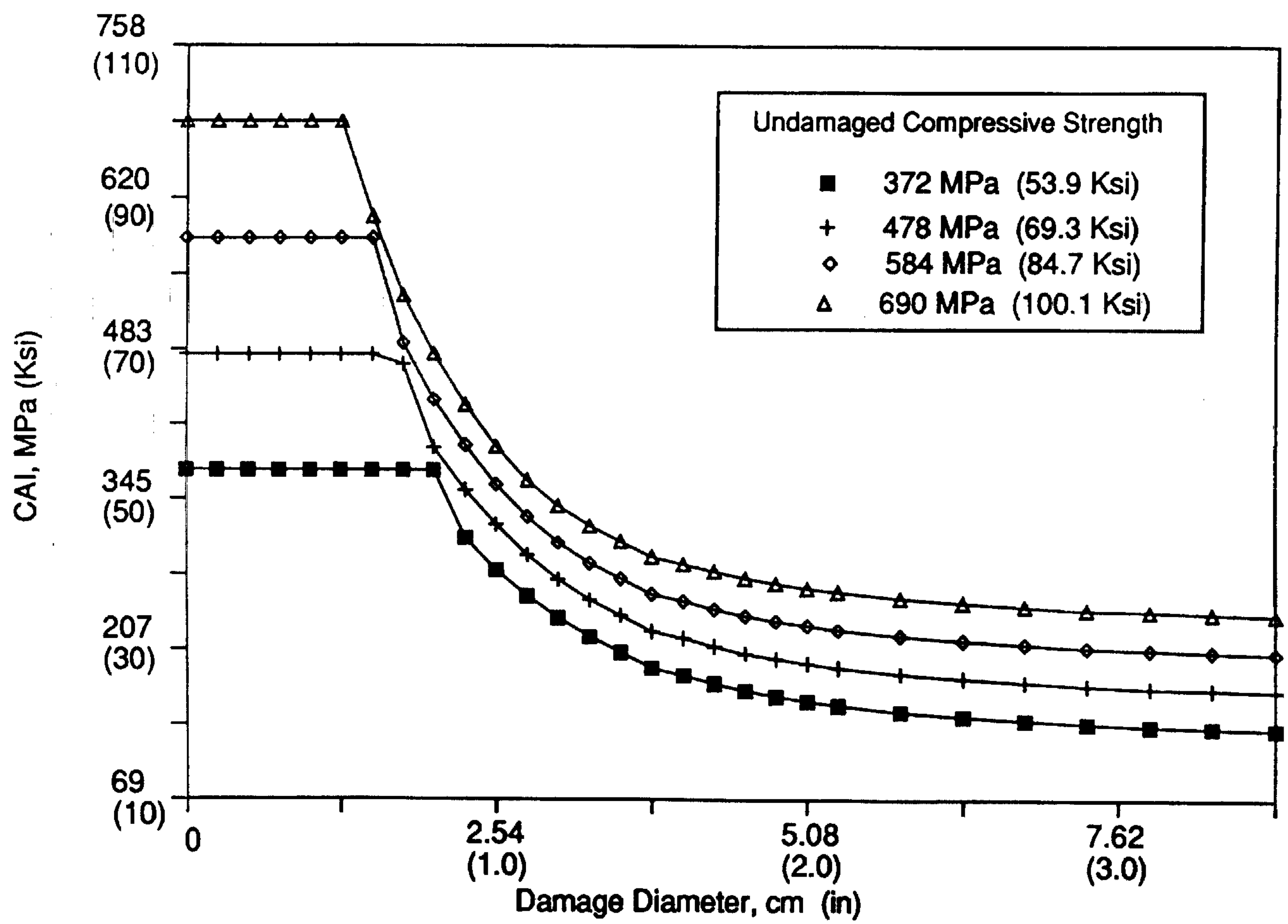


Figure 5. Finite specimen width effects for IM6/3501-6 moduli, $(45/0/-45/90)_{nS}$ ($n \geq 3$), undamaged compressive strength = 478 MPa, and ply thickness = 0.0188 cm.

MATERIAL, LAMINATE, AND STRUCTURAL VARIABLES

- TOUGHNESS VS. BASIC COMPRESSION STRENGTH
- PLY THICKNESS AND LAMINATE STACKING SEQUENCE
  - EFFECT ON SPECIMEN STABILITY
  - EFFECT ON SUBLAMINATE STABILITY
  - EFFECT ON DAMAGE STATE

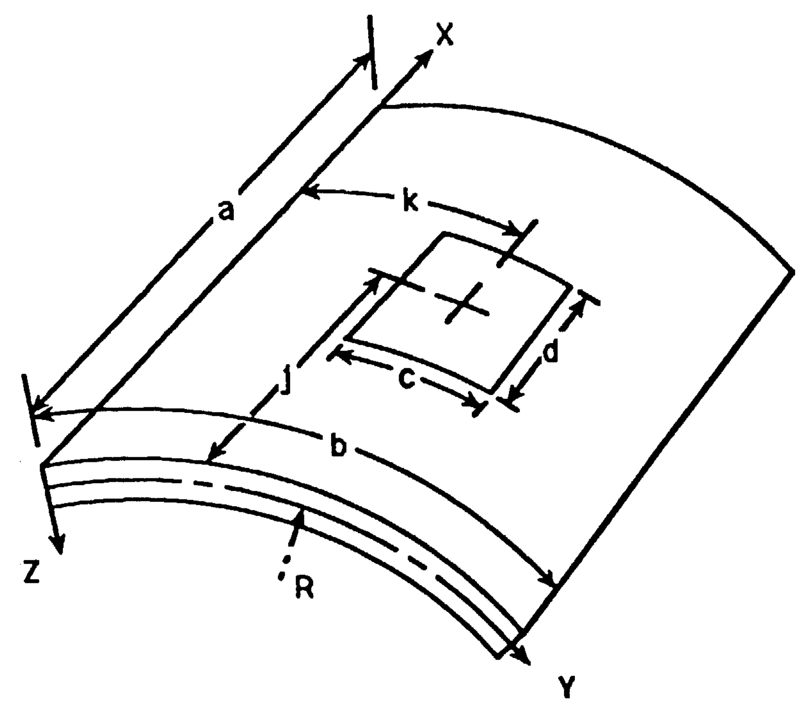
Predictions of the effects of undamaged compressive strength for $(45/0/-45/90)_{nS}$ (n≥3), IM6/3501-6 moduli, and ply thickness = 0.0188 cm.

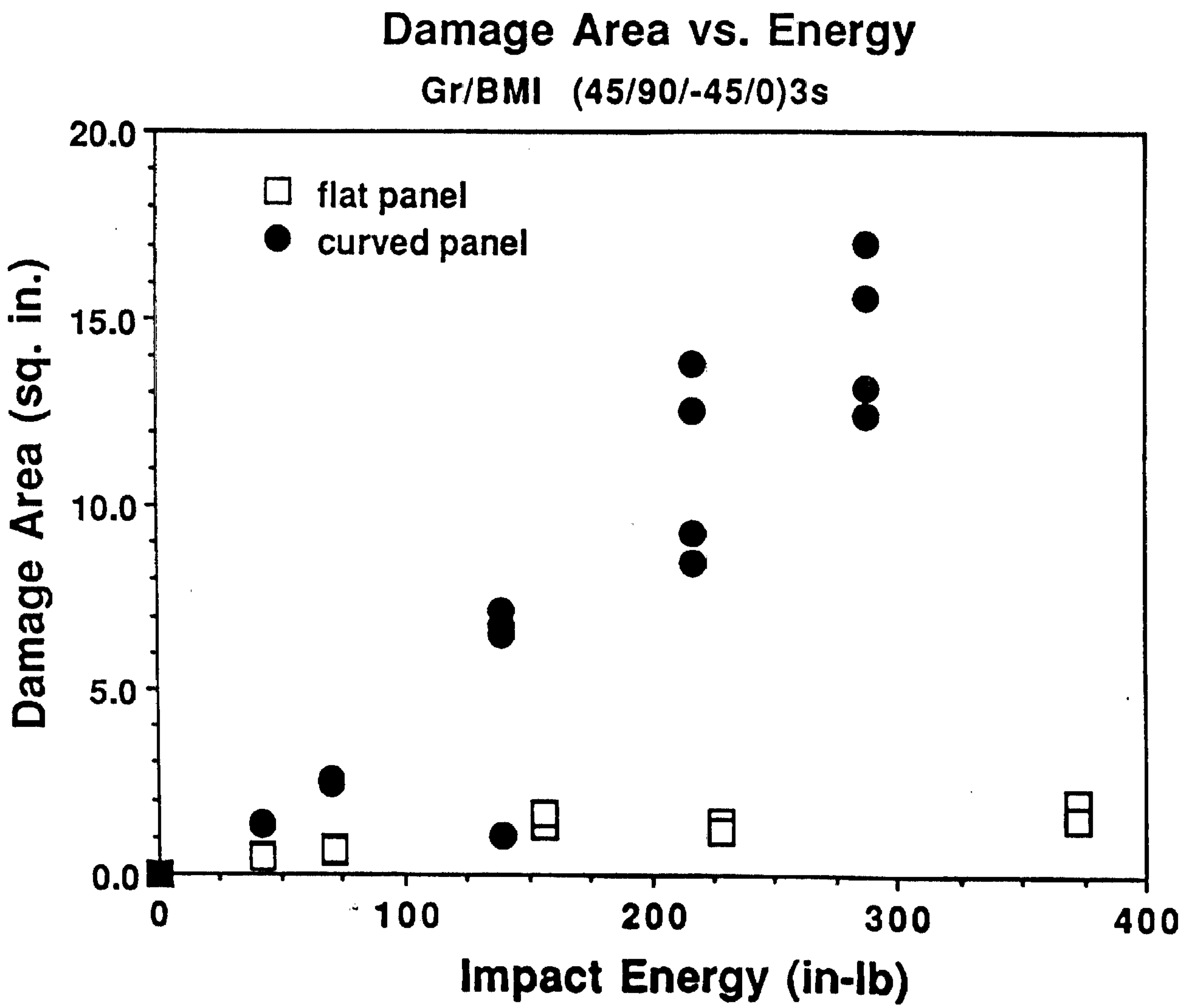


SPECIMEN CURVATURE / SPECIMEN THICKNESS

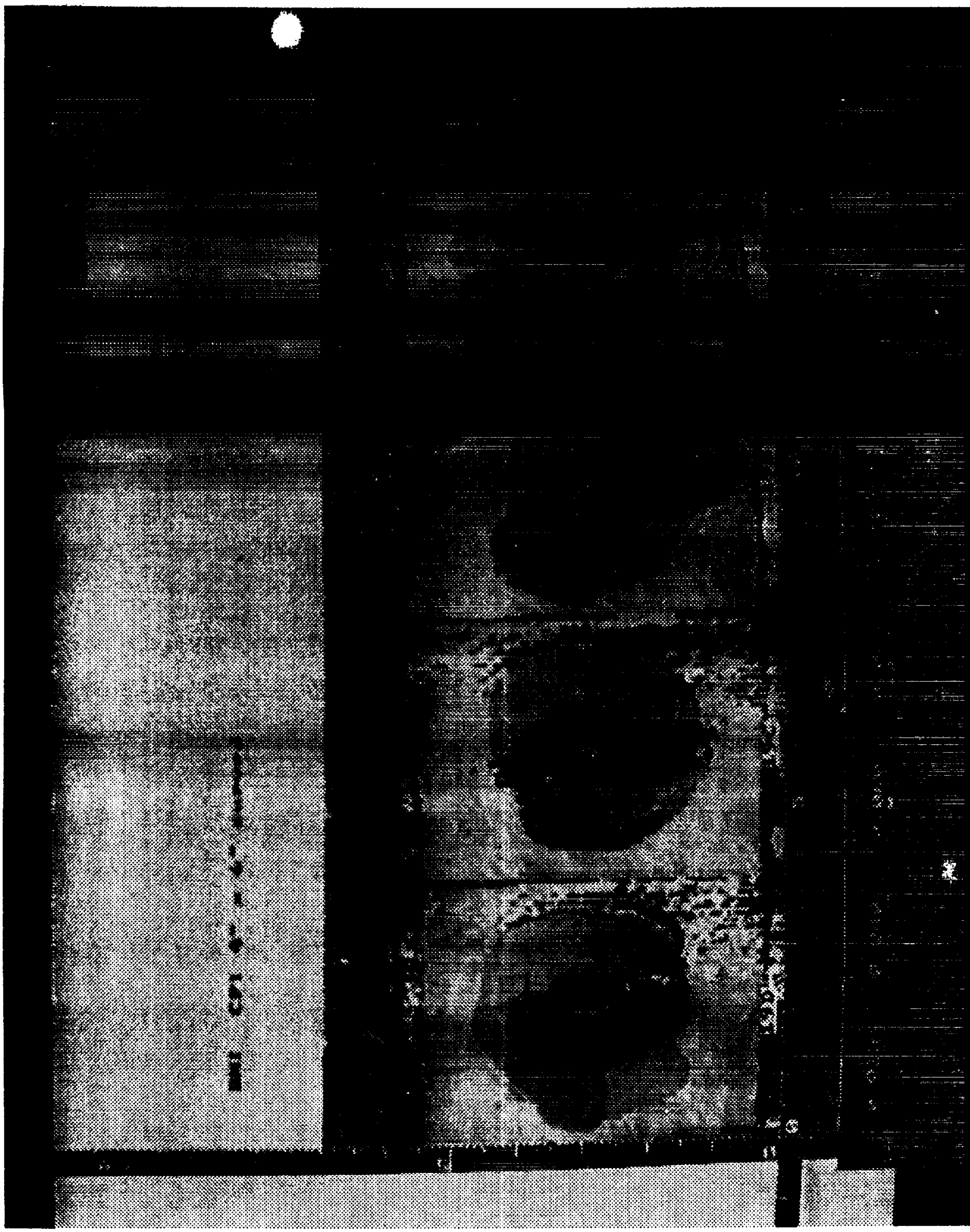
- INCREASED INTERLAMINAR SHEAR COMPONENT RESULTS IN LARGER DELAMINATIONS
- CURVATURE INCREASES SUBLAMINATE STABILITY
- RESULTS FOR HITEX-46/V398-G BMI LAMINATES

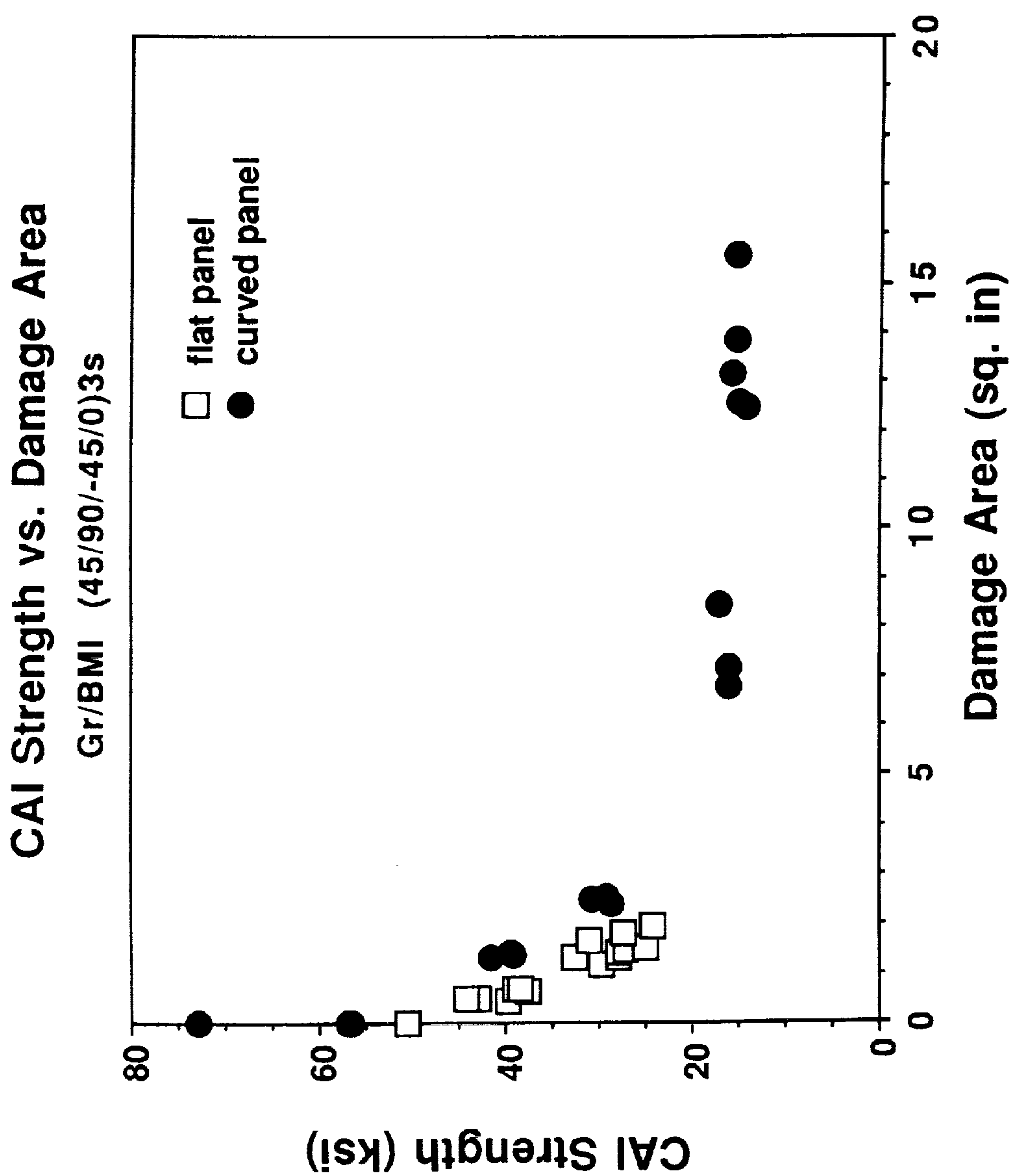
CURVED CAI SPECIMEN



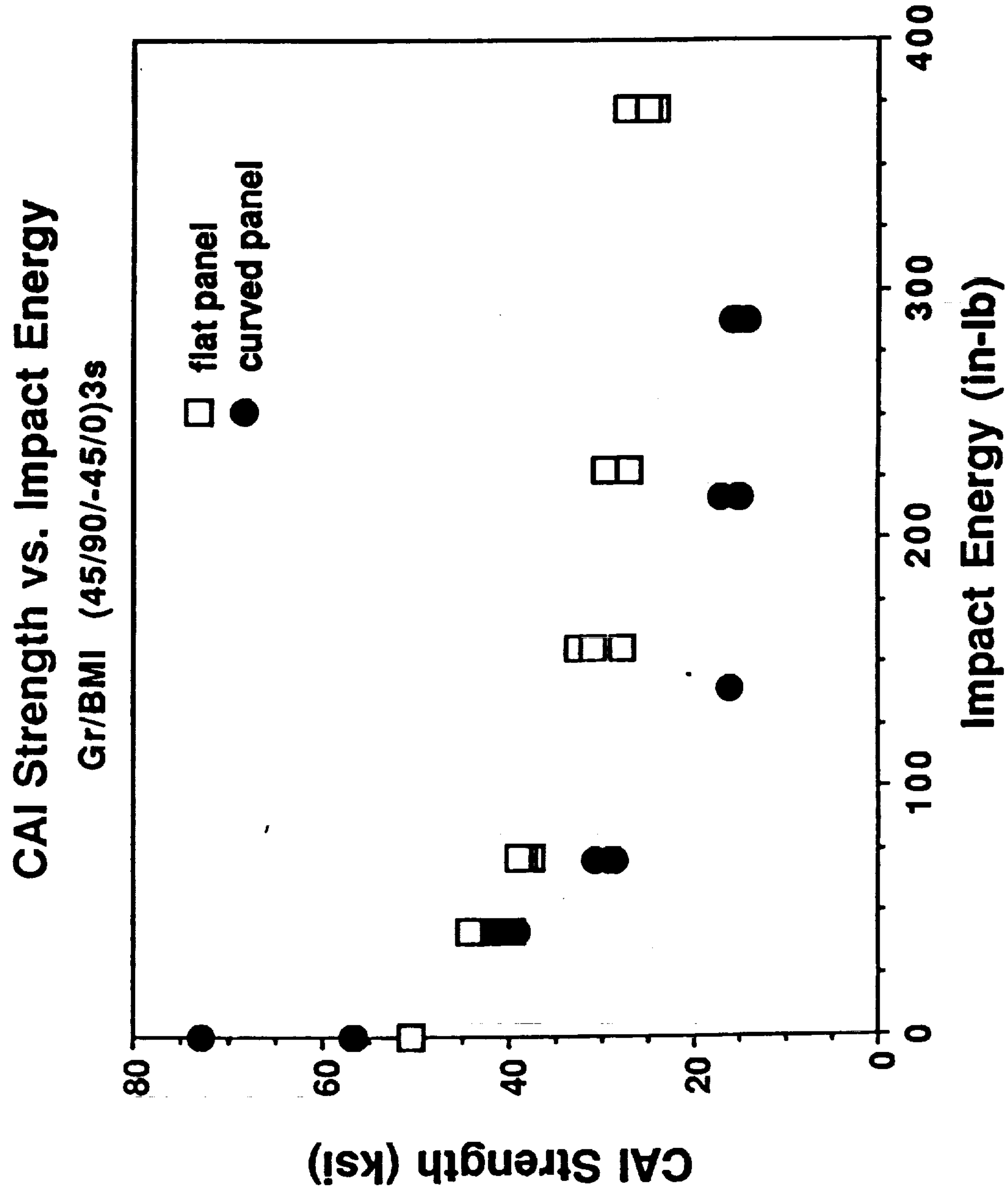

Damage Area vs. Energy
Gr/BMI (45/90/-45/0)3s
flat panel
curved panel
Damage Area (sq. in.)
20.0
15.0
10.0
5.0
0.0
0
100
200
300
400
Impact Energy (in-lb)

ASG-2369.bw





CAI Strength vs. Damage Area
Gr/BMI (45/90/-45/0)3s
flat panel
curved panel
CAI Strength (ksi)
80
60
40
20
0
0
5
10
15
20
Damage Area (sq. in)


CAI Strength vs. Impact Energy
Gr/BMI (45/90/-45/0)3s
flat panel
curved panel
CAI Strength (ksi)
Impact Energy (in-lb)
0
20
40
60
80
0
100
200
300
400

RECOMMENDED CAI STRENGTH TEST EVALUATION SCHEMES

- STUDY A RANGE OF IMPACT LEVELS.
- CONSIDER POSSIBLE EFFECTS OF IMPACTOR GEOMETRY, MASS AND VELOCITY.
- USE INSTRUMENTED IMPACT TEST METHODS TO QUANTIFY DAMAGE RESISTANCE.
- NONDESTRUCTIVELY MEASURE IMPACT DAMAGE SIZE PRIOR TO THE CAI TEST.
- ANALYTICALLY SCALE CAI RESULTS TO ELIMINATE FINITE SPECIMEN WIDTH EFFECTS.
- VALIDATE ANALYSIS WITH TEST RESULTS TO ADD CONFIDENCE IN USING THE MODELS TO EXTRAPOLATE THE DATA BASE.
- EXAMINE BROKEN SPECIMENS TO JUDGE FAILURE MODE FOLLOWING CAI TESTS.

ELEMENT GEOMETRIES

1. CRIPPLING SPECIMENS
   - IMPACT DAMAGED IN STRUCTURAL CONFIGURATIONS
2. SKIN/STRINGER SEPARATION
   - IMPACT DAMAGED IN STRUCTURAL CONFIGURATIONS

TENSION AFTER THROUGH PENETRATION

<u>BOEING CROWN STUDIES:</u>

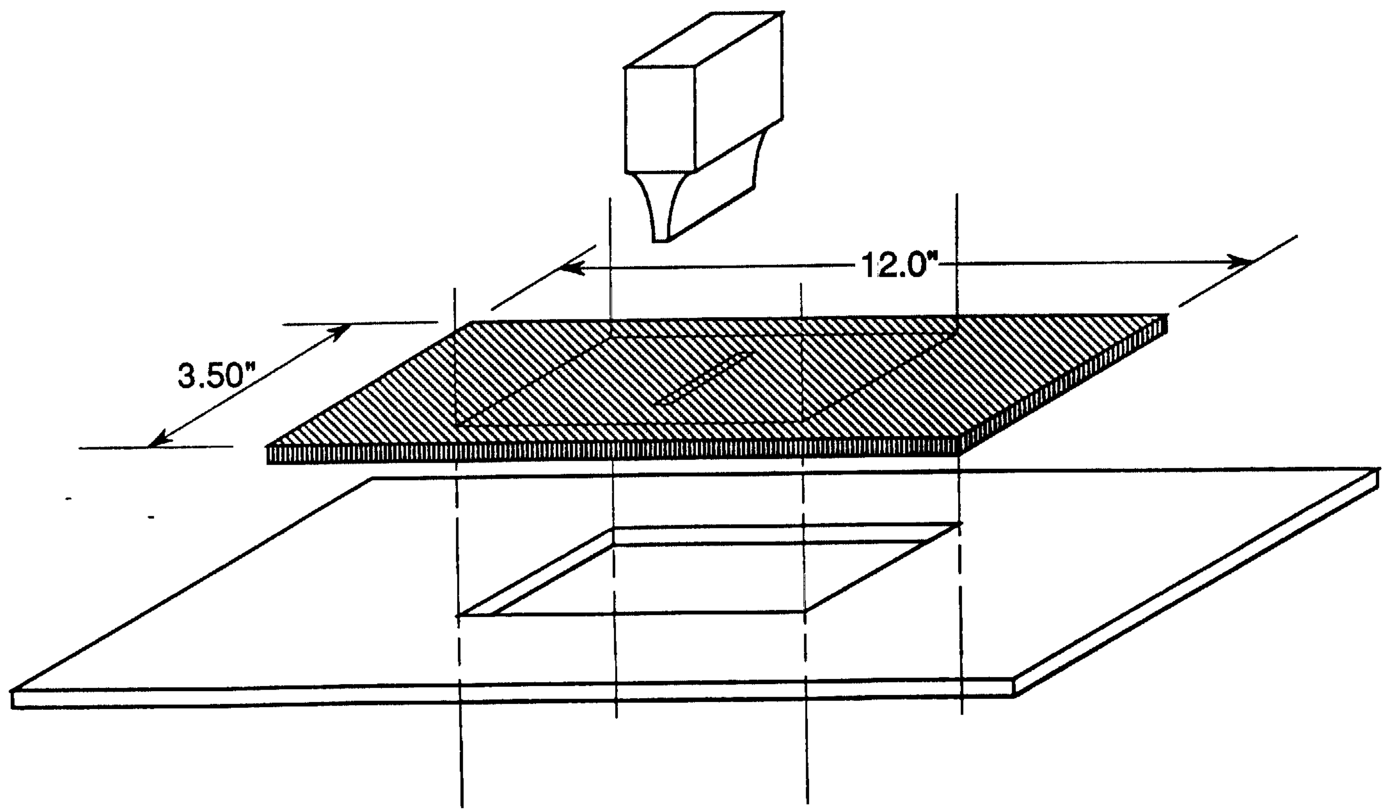
TEST METHOD

- 3.5" X 12" SPECIMEN
- 2.5" X 5" WINDOW IN SUPPORT FIXTURE
- 0.875" X 0.060" BLADE

IMPACT RESULTS

- THINNER SPECIMENS EXHIBIT FIBER DAMAGE
- THICKER SPECIMENS EXHIBIT EXTENSIVE DELAMINATION

Expanded View of Blade Impacter



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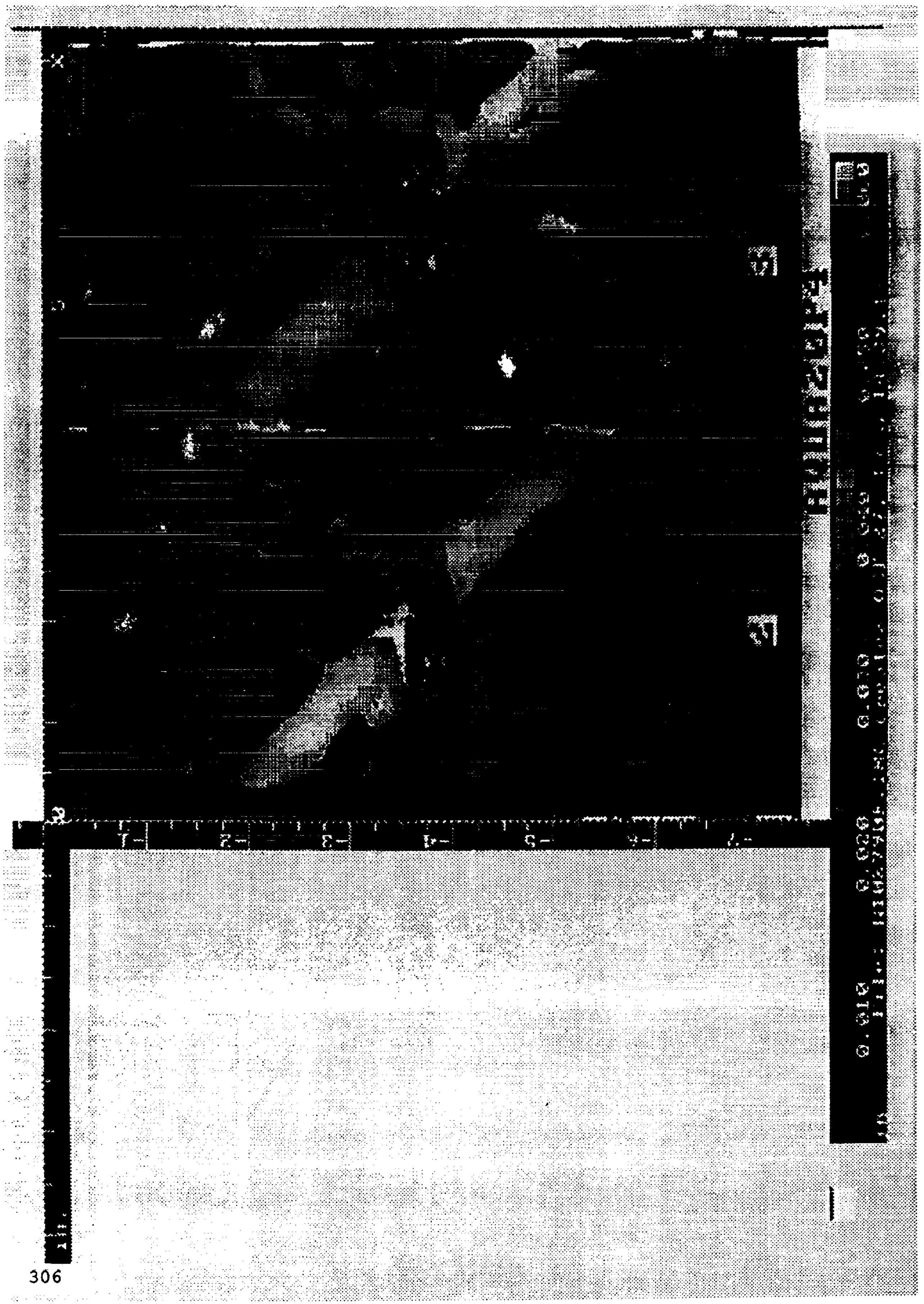

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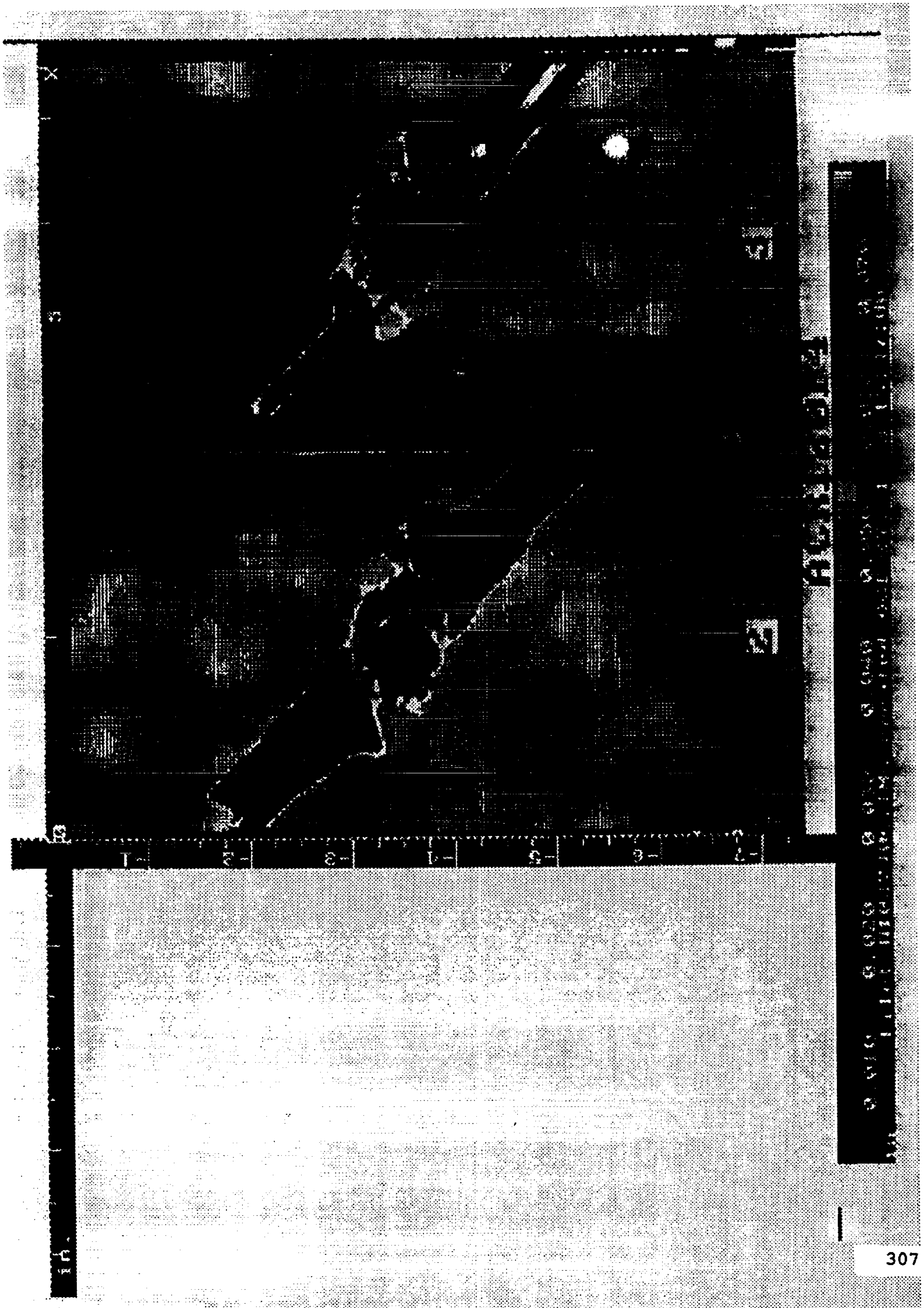

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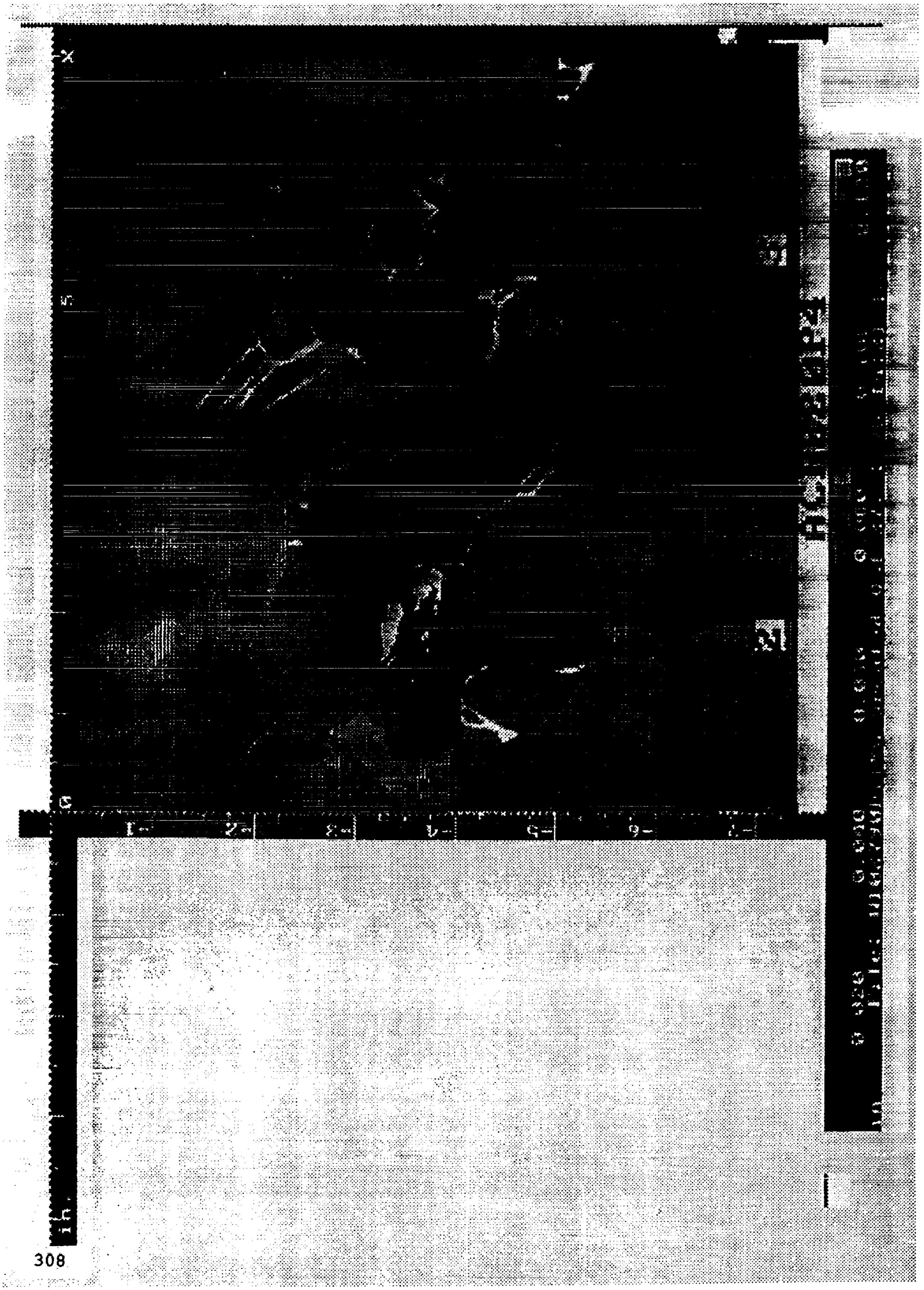

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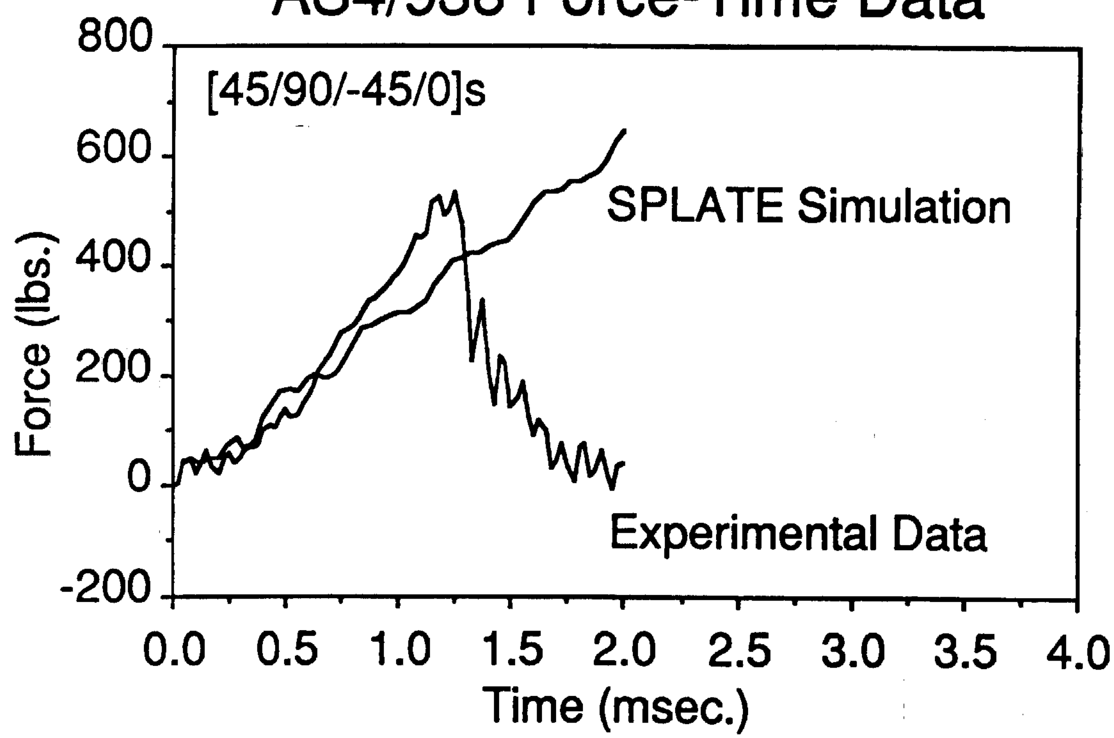
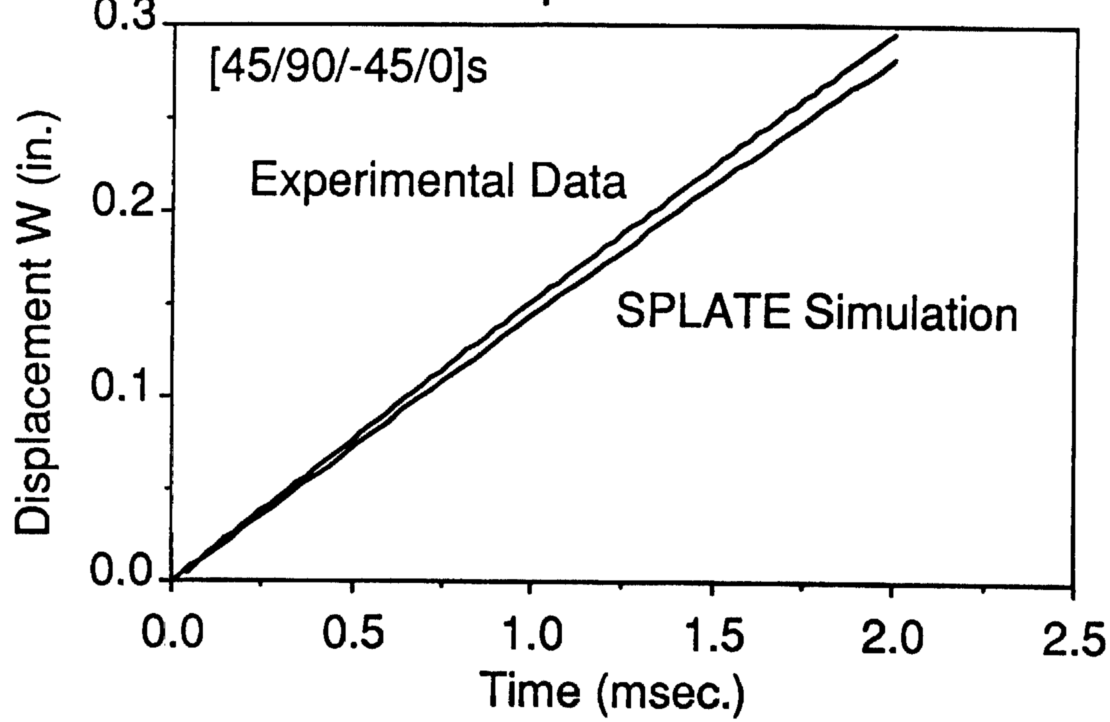
SPLATE Simulation of Blade Impact
AS4/938 Force-Time Data



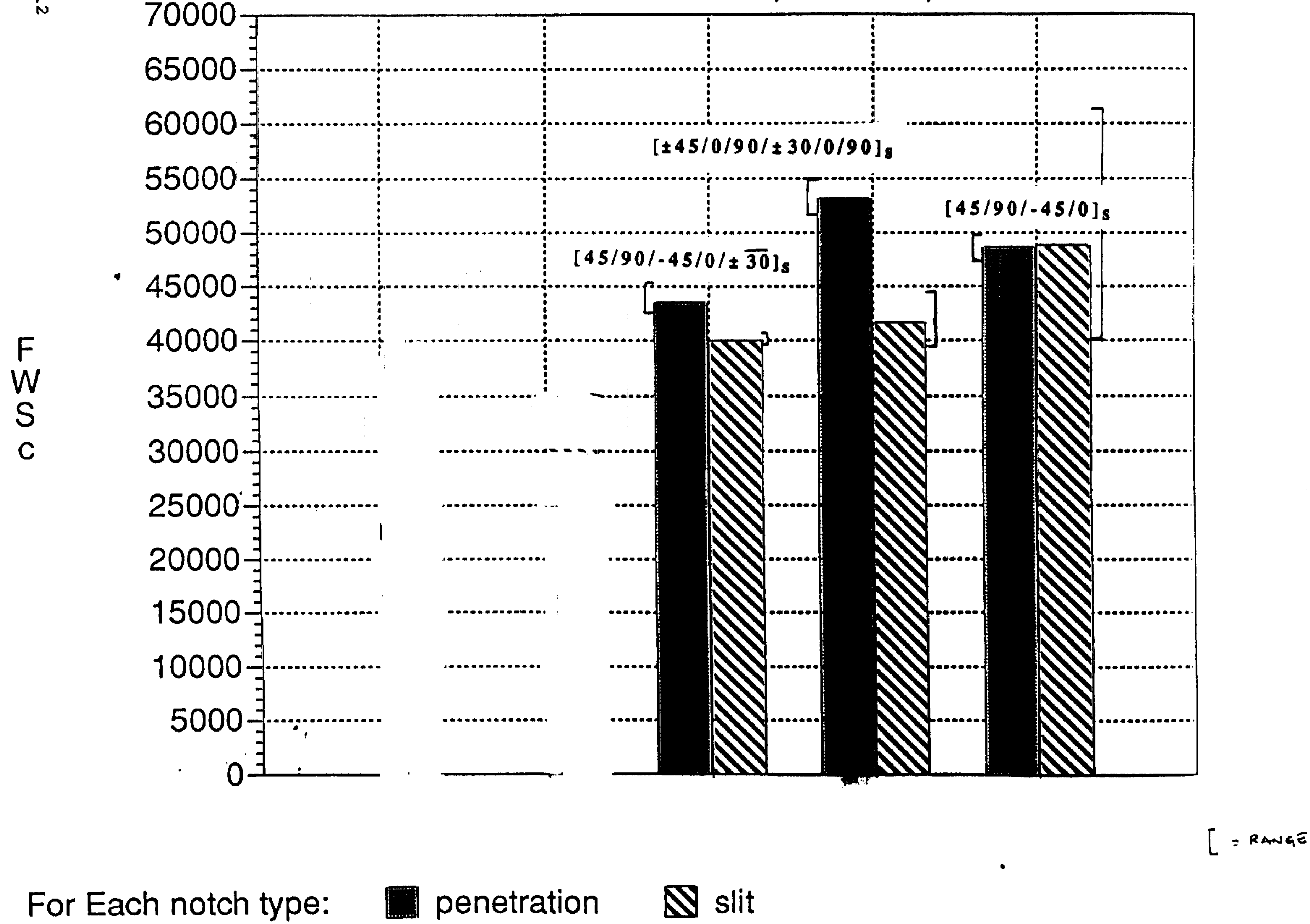
SPLATE Simulation of Blade Impact
AS4/938 Displacement Curves



TENSION RESIDUAL STRENGTH

- FOR THICKER SPECIMENS RESIDUAL STRENGTH OF PENETRATION SPECIMENS IS GREATER THAN SLIT SPECIMENS
- FOR THINNER SPECIMENS RESIDUAL STRENGTHS APPROXIMATELY EQUAL OR SLIGHTLY LESS

Comparison of Slit and Penetration Strengths
Material = IM6/3501-6, w/d = 4, w = 3.5



HYPOTHESIS

- IN THICK LAMINATES THE PENETRATION EVENT CAUSES PRIMARILY DELAMINATIONS BEYOND THE PENETRATION, WHICH SOFTENS THE NOTCH TIP REGION AND THEREFORE INCREASES STRENGTH

- IN THINNER LAMINATES THE PENETRATION CAUSES FIBER BREAKAGE BEYOND THE PENETRATION, WHICH INCREASES THE EFFECTIVE NOTCH LENGTH AND REDUCES STRENGTH

RECOMMENDED STANDARD COUPON

SPECIMEN:	3.5" X 12"
CUT-OUT:	2.5" X 5"
IMPACTOR:	0.875" X 0.060"

ENGINEERING EVALUATION NEEDED

- LARGE DEFLECTIONS
- NON-LINEAR EFFECTS
- SPECIMEN THICKNESS EFFECTS
- CUT-OUT SIZE EFFECTS

ALL RELATED

OTHER MATERIAL SCREENING NEEDS

1. COMPRESSION AFTER THROUGH PENETRATION
2. SHEAR STRENGTH AFTER IMPACT
3. COMBINED LOAD REDISTRIBUTION
   - COMPRESSION / SHEAR
   - COMPRESSION / HOOP TENSION
   - IMPACT AND THROUGH PENETRATION
   - TAPERED THICKNESS (INTERLAMINAR SHEAR LOAD TRANSFER)
4. SANDWICH SPECIMEN GEOMETRIES
   - 5"X10" WITH 4"X4" CUT-OUT IN SUPPORT FIXTURE

Impact Damaged Composites: Summary for Parts 1 and 2

Critical Impact Threats for Fuselage Structure Must Be Identified

Accurate Predictions of Post-Impact Strength Require Characterization of Important Damage Features

Complex Damage States Must be Simulated (e.g., Sublaminate Buckling Approach) to Allow Engineering Analysis of Structural Configurations

The Post-Impact Strength/Damage Relationships for Fuselage Structure Will Depend on Location, Combined Load States, and Load Redistribution

Fundamentals Must be Understood to Relate Coupon Screening Tests to Fuselage Structural Performance

References Giving More Information on Topics Covered

1.) Gosse, J.H., Mori, P.B.Y., "Impact Damage Characterization of Graphite/Epoxy Laminates," In Proc. of 3rd Tech Conf. of American Society for Composites, Technomic Publ. Co., 1988, pp. 344-353.

2.) Dost, E.F., Ilcewicz, L.B., Gosse, J.H., "Sublaminate Stability Based Modeling of Impact Damaged Composite Laminates," In Proc. of 3rd Tech. Conf. of Society for Composites, Technomic Publ. Co., 1988, pp. 354-363.

3.) Ilcewicz, L.B., Dost, E.F., Coggeshall, R.L., "A Model for Compression After Impact Strength Evaluation," In Proc. of 21st Inter. SAMPE Tech. Conf., Soc. for Adv. of Material and Process Engineering, 1989, pp. 130-140.

4.) Avery, W.B., "A Semi-Discrete Approach to Modeling Post-Impact Compression Strength of Composite Laminates," In Proc. of 21st Inter. SAMPE Tech. Conf., Soc. for Adv. of Material and Process Engineering, 1989, pp. 141-151.

5.) Dost, E.F., Ilcewicz, L.B., Avery, W.B., Coxon, B.R., "The Effects of Stacking Sequence On Impact Damage Resistance and Residual Strength for Quasi-Isotropic Laminates," In Composite Materials: Fatigue and Fracture, ASTM STP 1110, 1991.

6.) Avery, W.B., Grande, D.H., "Influence of Materials and Layup Parameters On Impact Damage Mechanisms," In Proc. of 22nd Inter. SAMPE Tech. Conf., Soc. for Adv. of Material and Process Engineering, 1990, pp. 470-483.

7.) Grande, D.H., Ilcewicz, L.B., Avery, W.B., Bascom, W.D., "Effects of Intra- and Inter-Laminar Resin Content On Mechanical Properties of Toughened Composite Materials," In Proc. of First NASA Advanced Composite Technology Conf. (Part 2), 1990, pp. 455-475.

8.) Dost, E.F., Avery, W.B., Swanson, G.D., Lin, K.Y., "Developments In Impact Damage Modeling for Laminated Composite Structures," In Proc. of First NASA Advanced Composite Technology Conf. (Part 2), 1990, pp. 721-736.

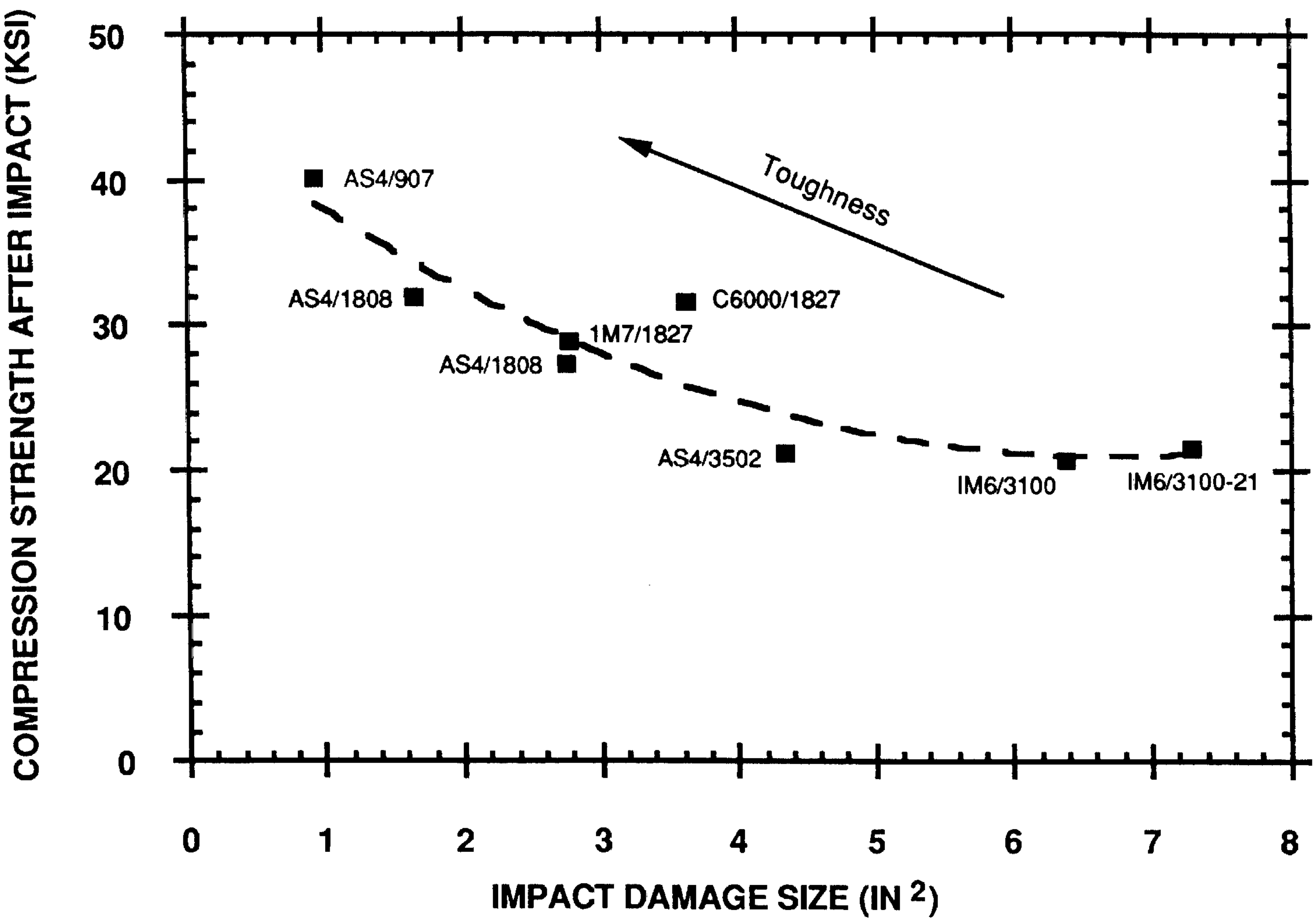
IMPACT DAMAGE RESISTANCE

and

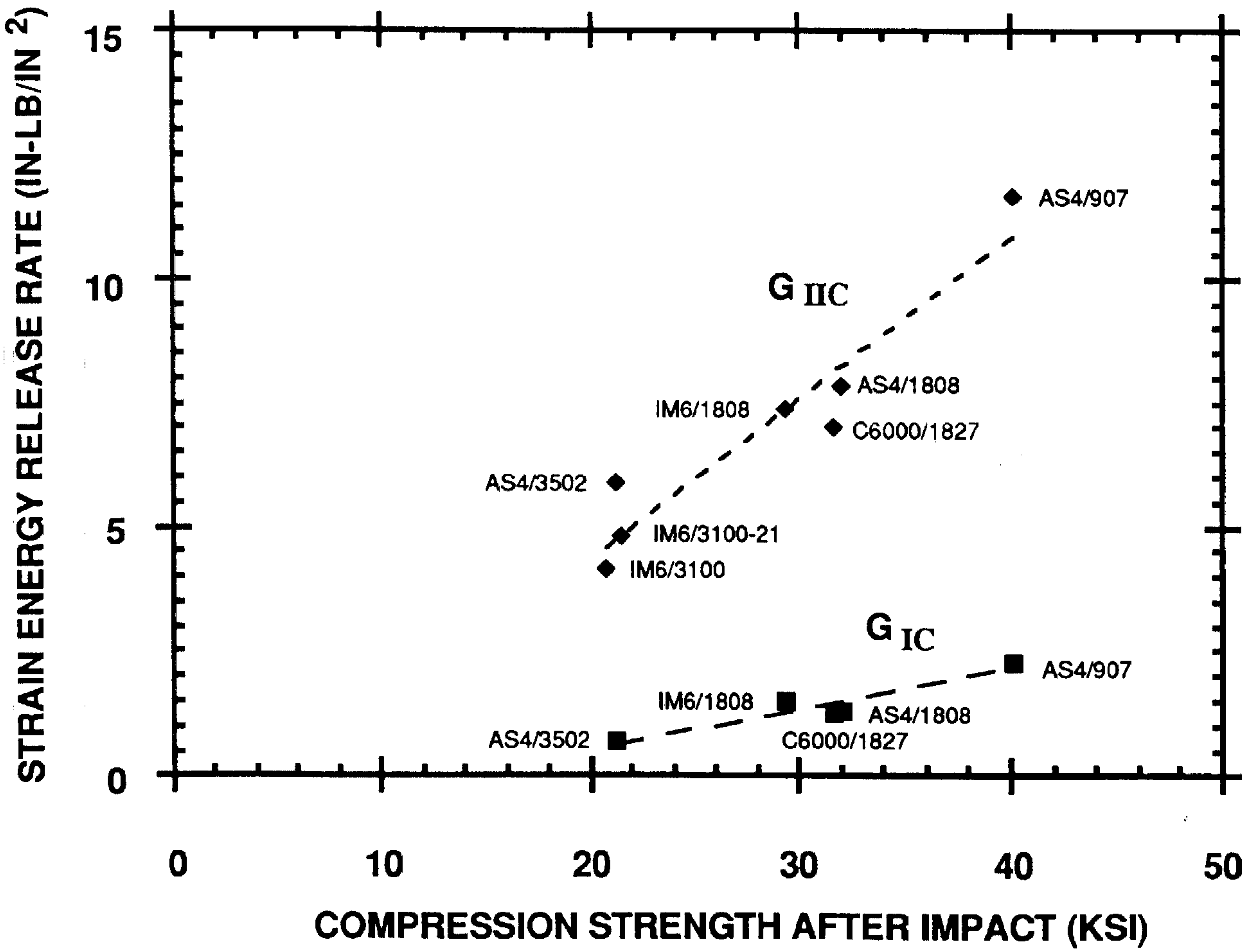
MATERIAL TOUGHNESS

JOHN E. MASTERS
LOCKHEED ENGINEERING AND SCIENCE

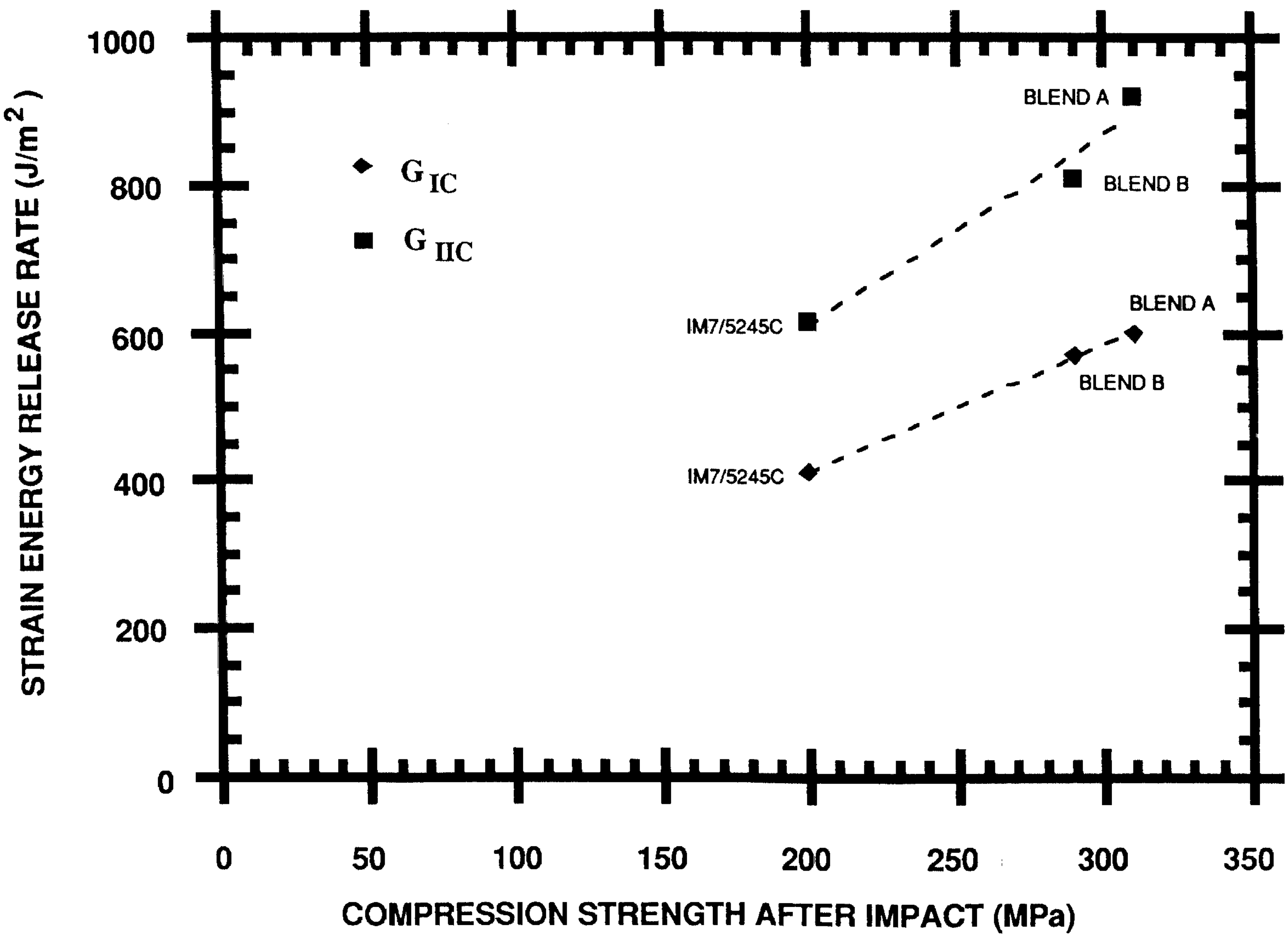
DAMAGE SIZE vs. RESIDUAL COMPRESSION STRENGTH
(Single Resin Systems)



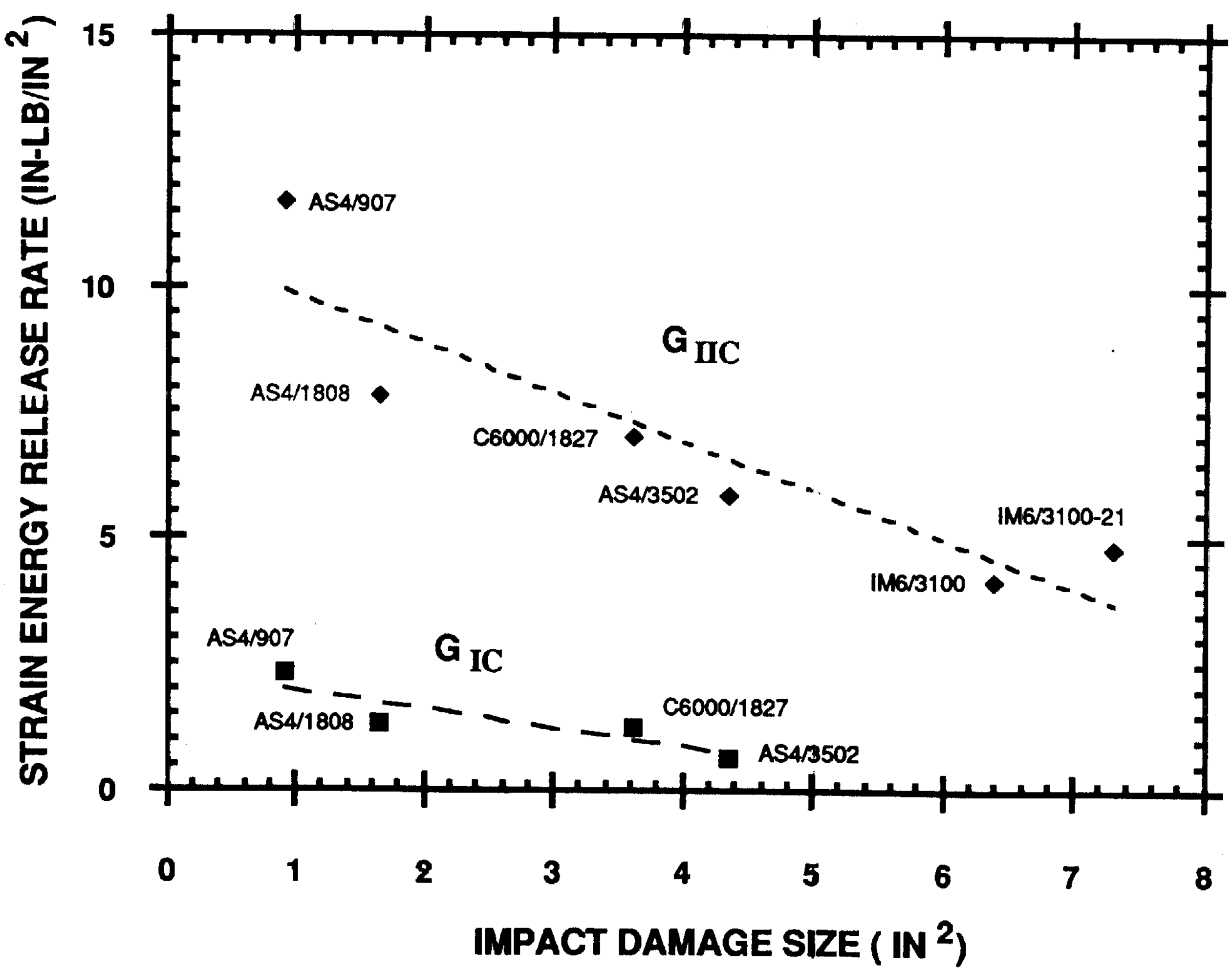
COMPRESSION STRENGTH AFTER IMPACT
vs.
STRAIN ENERGY RELEASE RATE
(Single Resin Systems)



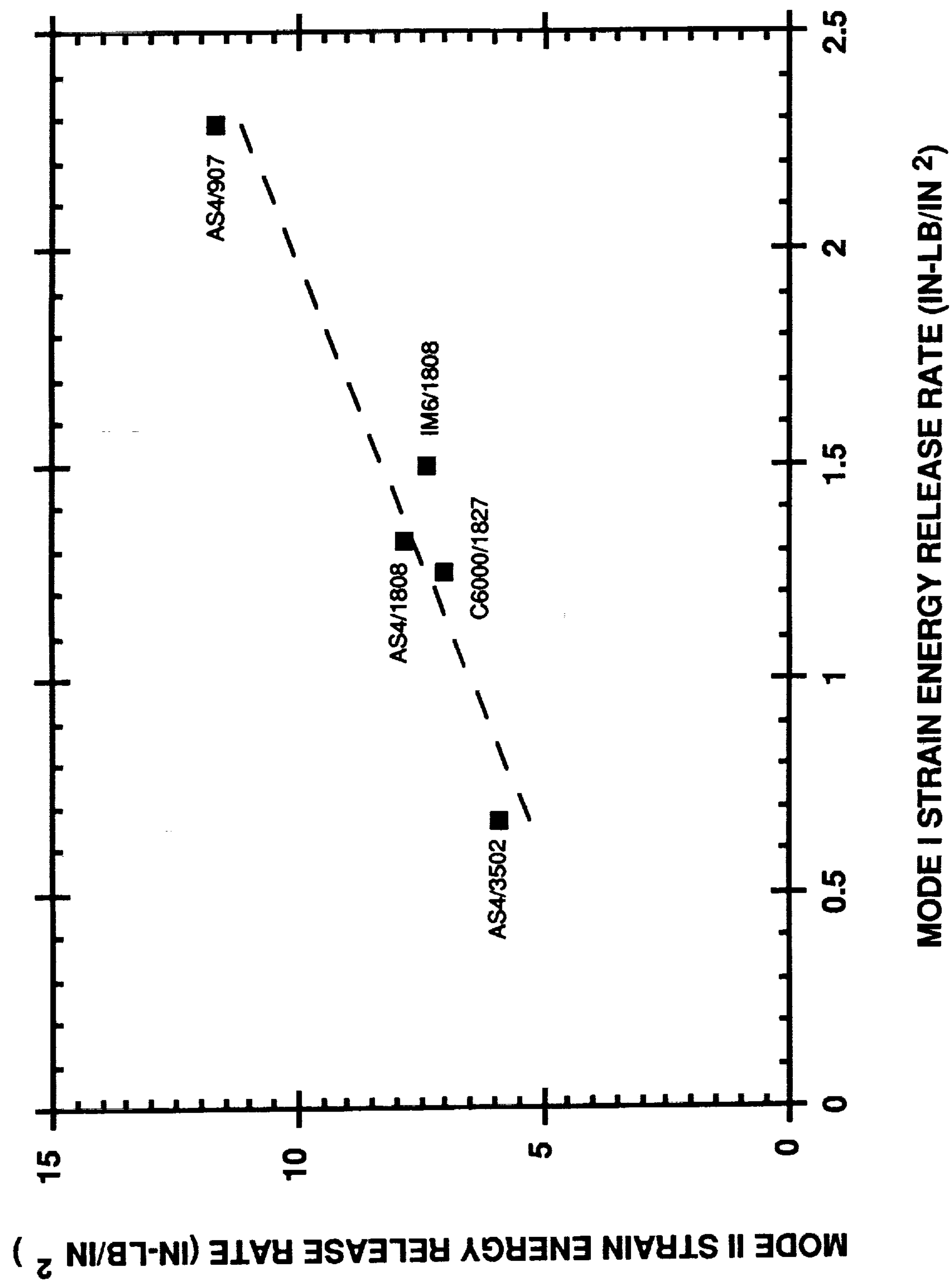
COMPRESSION STRENGTH AFTER IMPACT vs. STRAIN ENERGY RELEASE RATE (Thermoset-Thermoplastic Blends BASF)



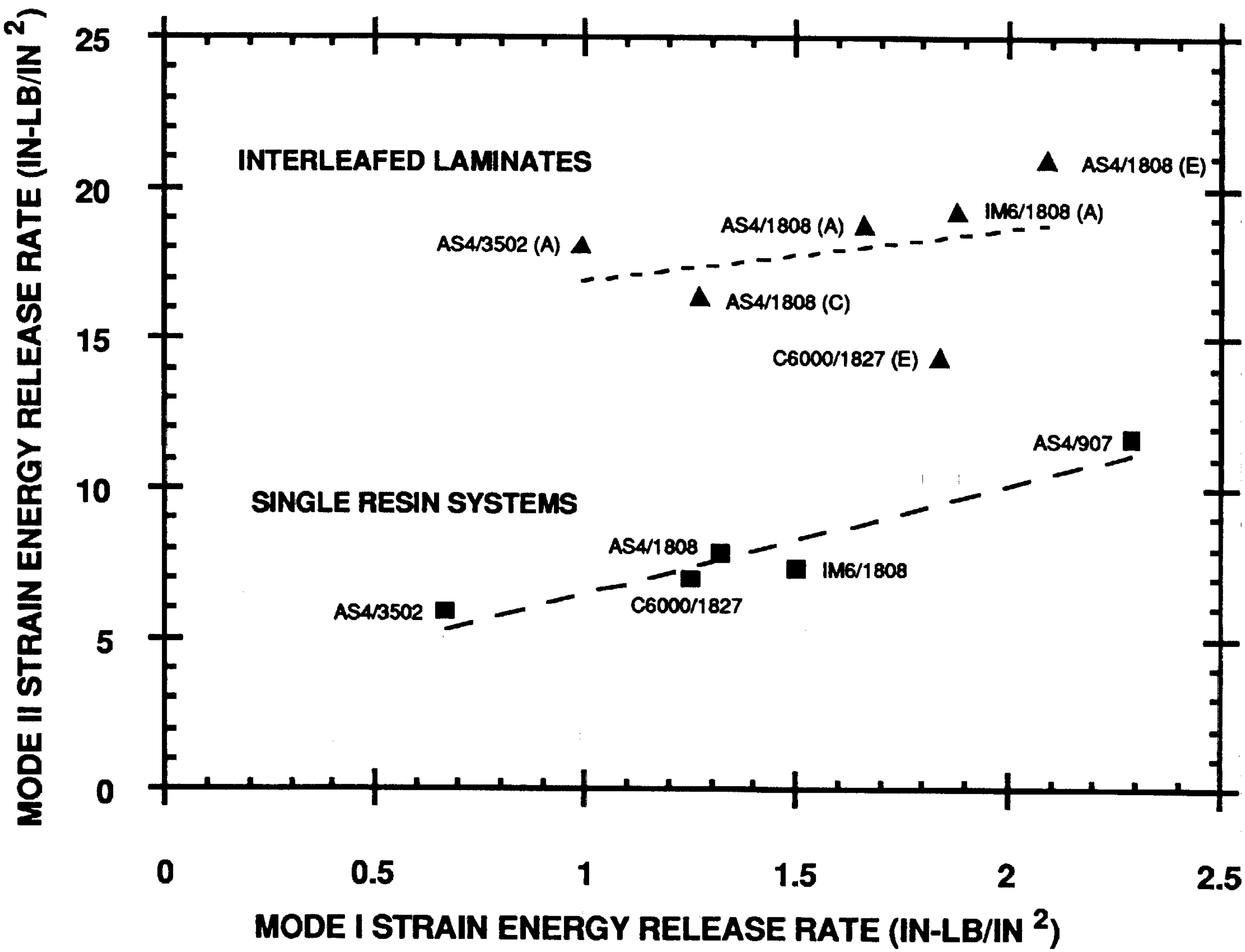
IMPACT DAMAGE SIZE
vs.
STRAIN ENERGY RELEASE RATE
(Single Resin Systems)



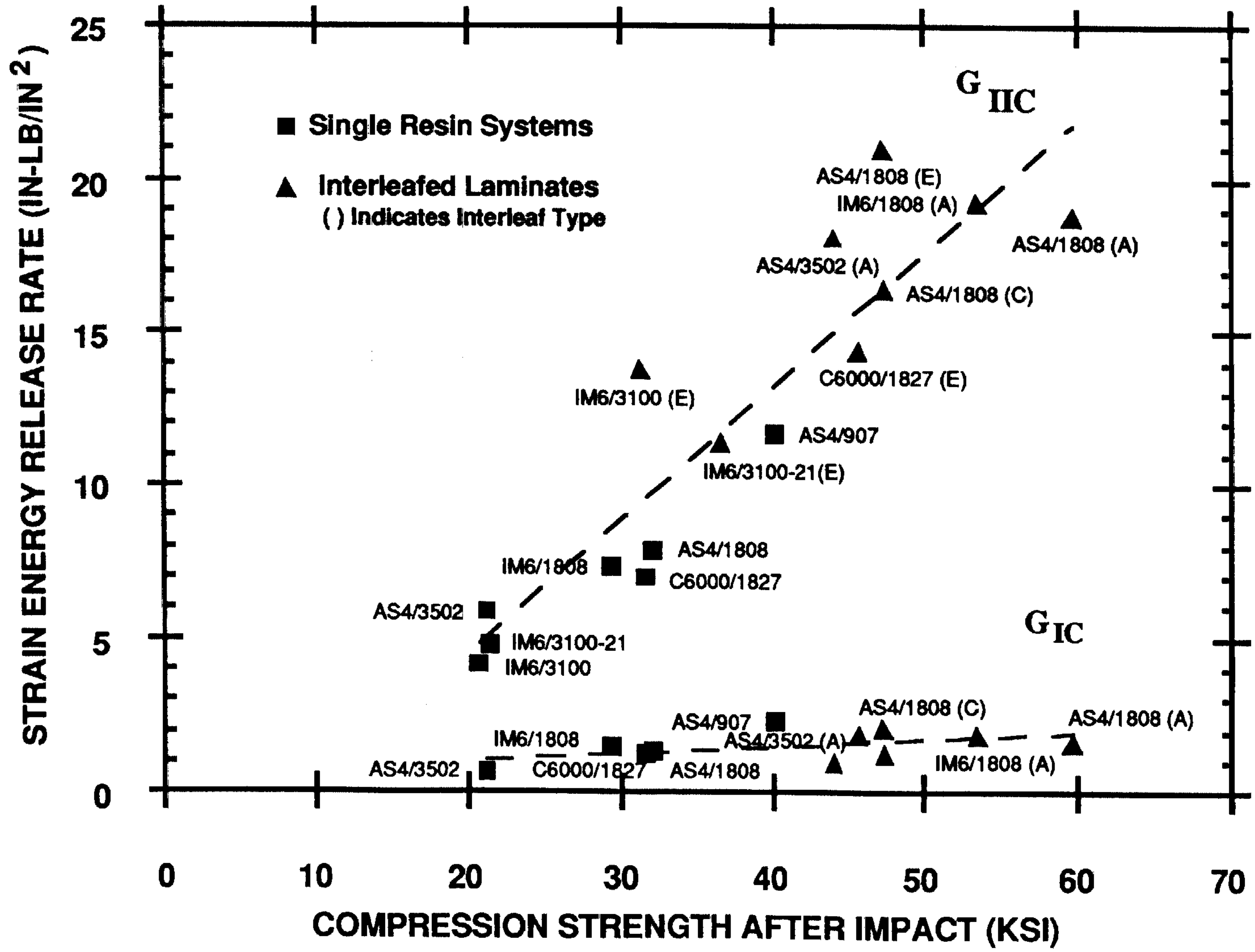
MODE I vs. MODE II STRAIN ENERGY RELEASE RATES
(Single Resin Systems)



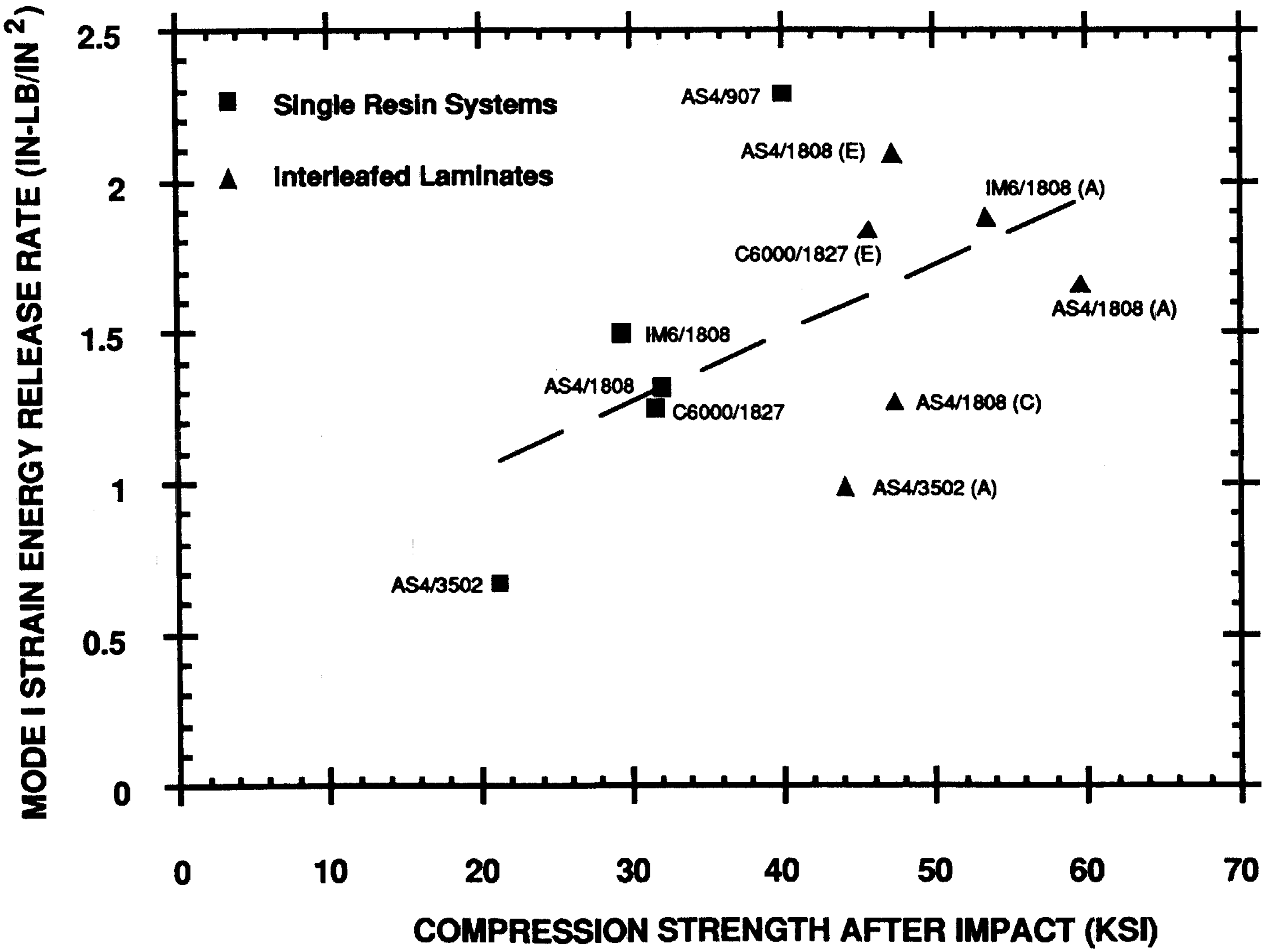
MODE I vs. MODE II STRAIN ENERGY RELEASE RATES
(Single Resin and Interleafed Systems)



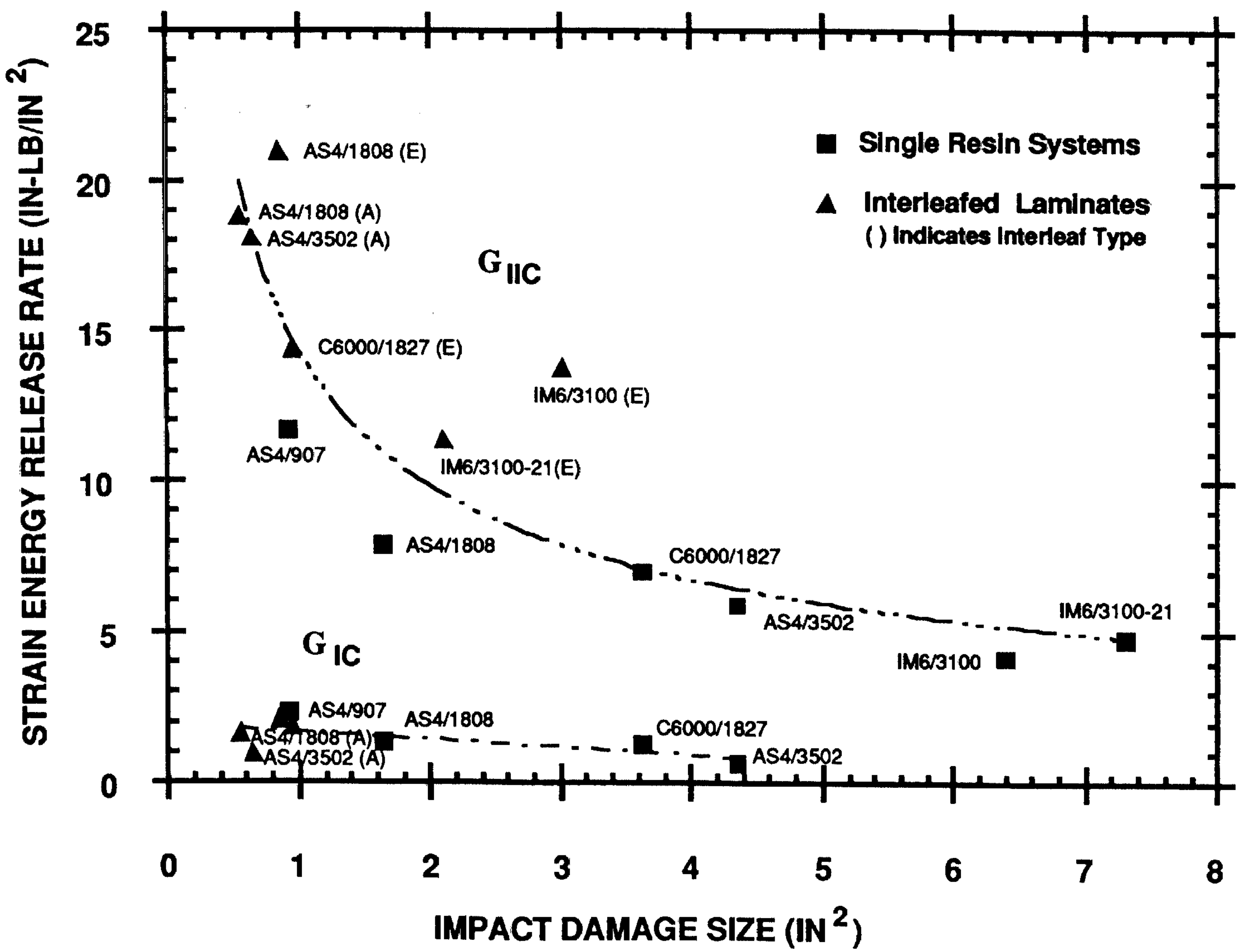
COMPRESSION STRENGTH AFTER IMPACT
vs.
STRAIN ENERGY RELEASE RATE
(Single Resins and Interleafed Laminates)



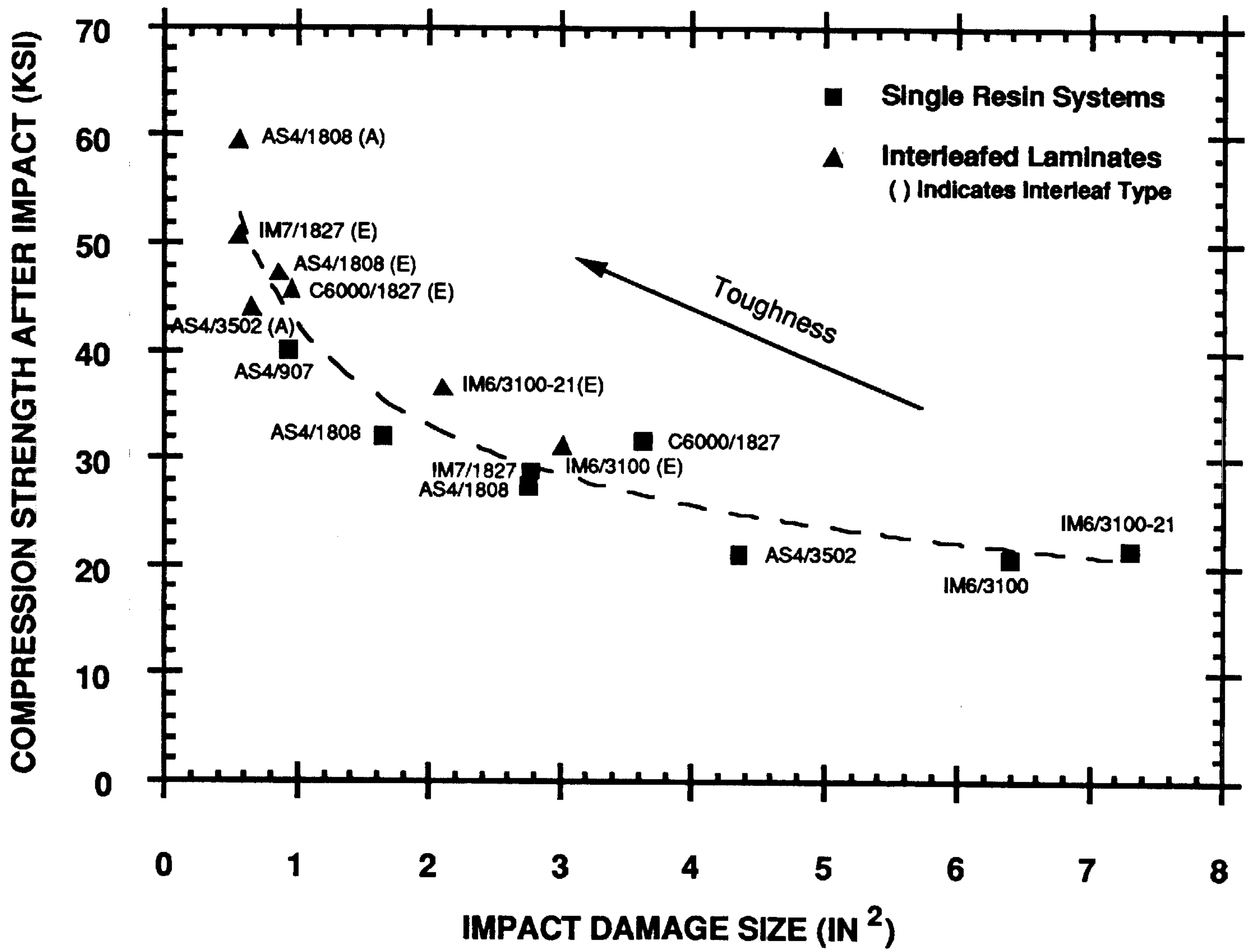
COMPRESSION STRENGTH AFTER IMPACT
vs.
MODE I STRAIN ENERGY RELEASE RATE



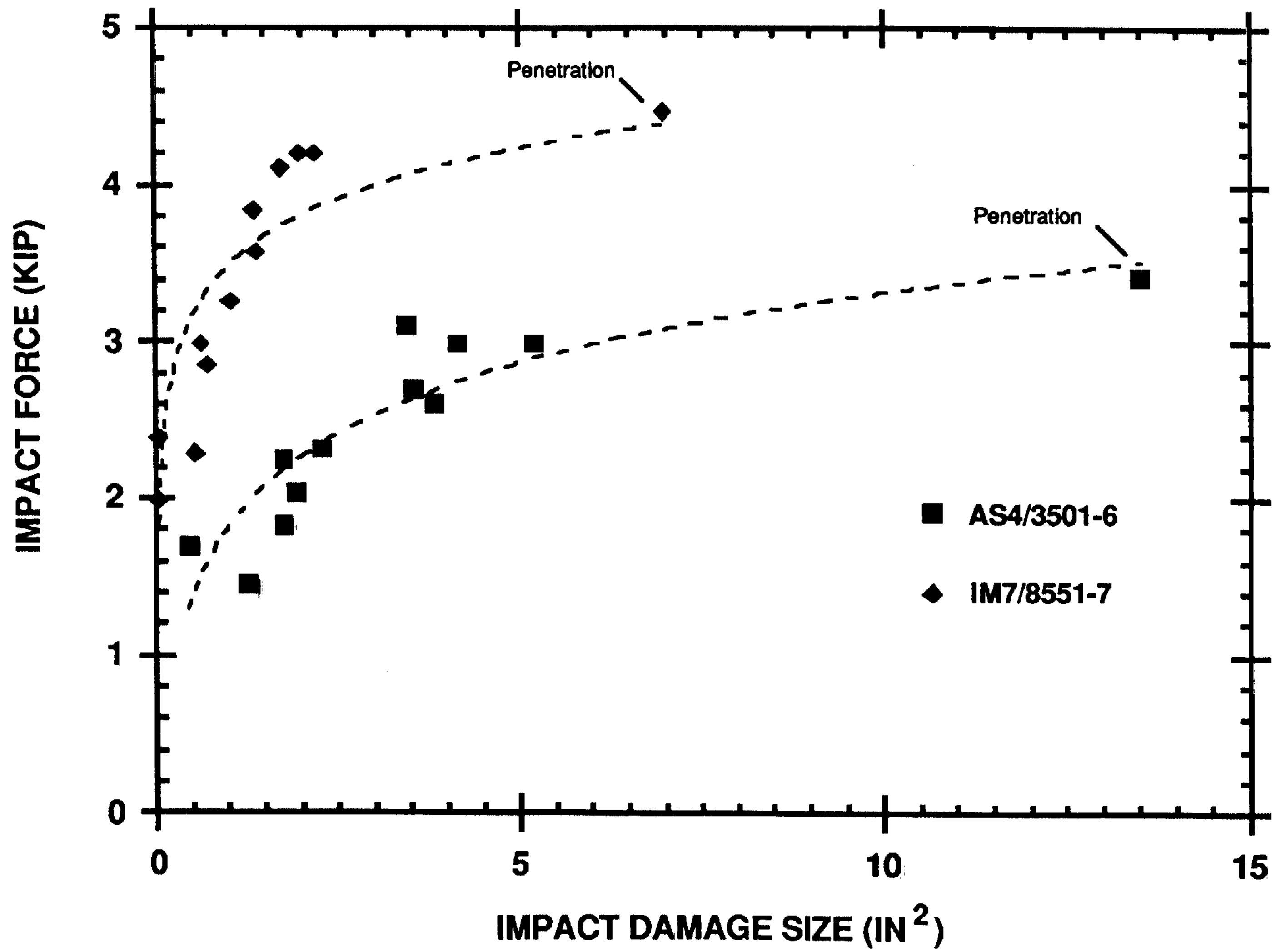
DAMAGE SIZE vs. STRAIN ENERGY RELEASE RATE
(Single Resins and Interleafed Laminates)



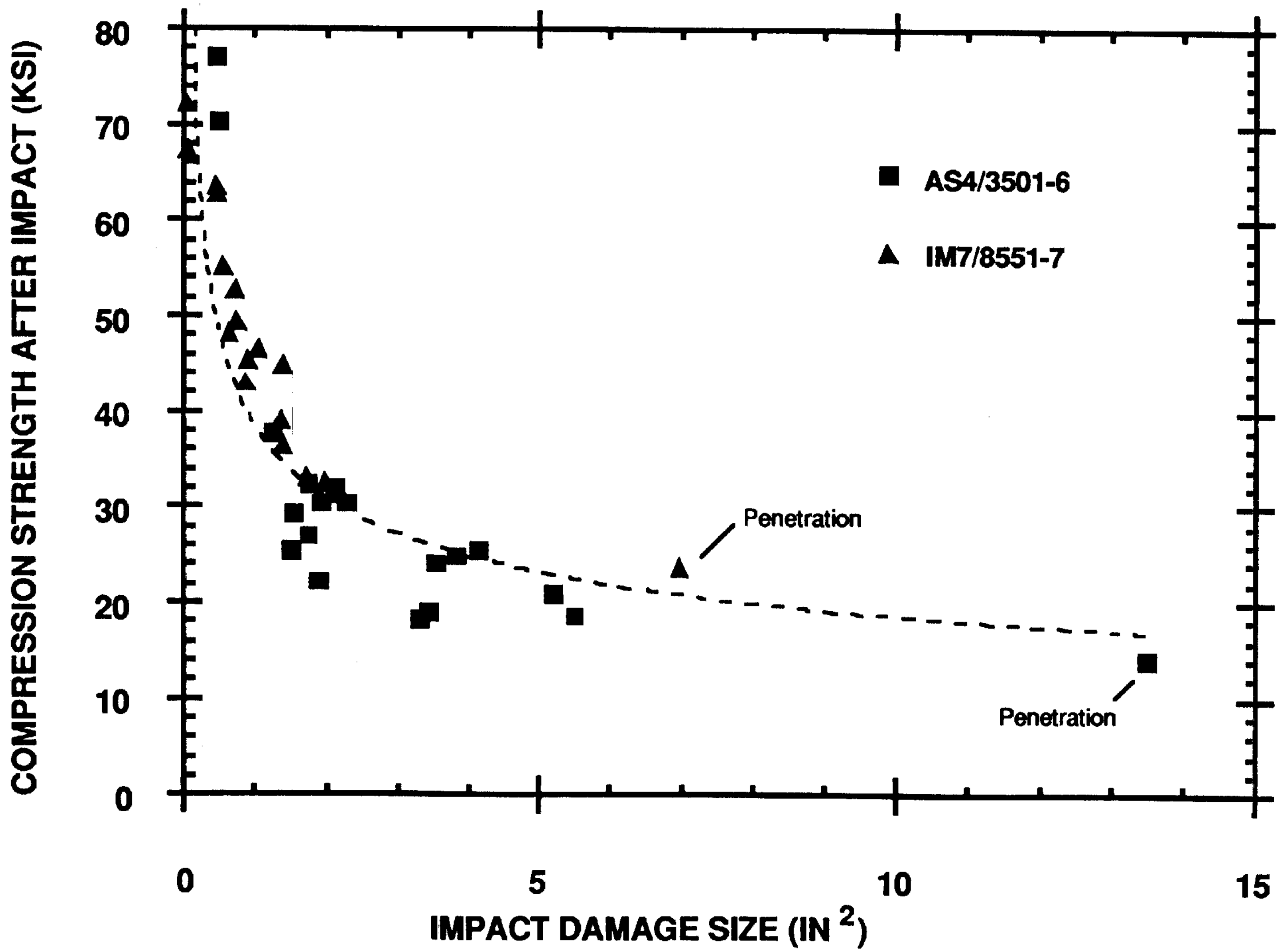
DAMAGE SIZE vs. RESIDUAL COMPRESSION STRENGTH
(Single Resin and Interleafed Laminates)



DAMAGE SIZE vs. IMPACT FORCE
(AS4/3501-6 and IM7/8551-7)



DAMAGE SIZE vs. RESIDUAL COMPRESSION STRENGTH
(AS4/3501-6 and IM7/8551-7)



SUMMARY:

Residual Compression Strength after Impact is a combined measure of

- Material Resistance to Impact-Induced Delamination
- Material Tolerance to Damage Propagation under Load

Impact Resistance is Directly Proportional to Material Toughness

Strong Correlation between Mode II Strain Energy Release Rate and Impact Resistance (Measured by Impact Delamination Size)

Material Tolerance to Impact Damage Insensitive to Toughness for a given Flaw Size

STANDARD IMPACT TESTS

PRESENTED AT

NASA WORKSHOP ON IMPACT DAMAGE TO COMPOSITES

NASA LARC
MARCH 20, 1991

BY
DR. VICTOR L. CHEN
PRINCIPAL ENGINEER
COMPOSITES R&D
DOUGLAS AIRCRAFT COMPANY

STANDARD IMPACT TESTS

COMPRESSION AFTER IMPACT

NASA ST-1

ST-1 SET-UP

SRM 2-88 (SACMA)

NASA ST-1

$[+45/0/-45/90]_{NS}$, 12" X 7" X 1/4"

10 LB. 1/2" HEMISPHERICAL STEEL TIP

NOMINAL 20 FT-LB IMPACT

REPLICA OF THREE

SACMA RECOMMENDED METHOD 2-88

$[+45/0/-45/90]_{NS}$, 6" X 4" X T

11 LB 5/8" HEMISPHERICAL STEEL TIP

1500 IN-LB/IN IMPACT

COMPRESSION AFTER IMPACT TEST

ID	SPECIMEN SIZE	NO. OF PLIES	IMPACT ENERGY FT-LBS.	IMPACT IN-LB/IN THICK	DAMAGE SIZE AREA IN/SQ IN	COMPRESSION STRENGTH KSI	FAILURE STRAIN %	REMARKS
31	2.5x10	32	CONTROL	- -	-- --	76.6	1.33	0.178 INCH
41	2.5x10	48	CONTROL	- -	-- --	91.8	1.71	0.266 INCH
34	NASA ST-1	32	14.2	957	1.43 1.71	31.3	0.47	
44	ST-1	48	21.3	958	1.46 1.83	32.8	0.50	TUP DIA=0.5IN
37	ST-1	32	22.0	1486	1.25 1.35	29.2	0.44	
47	ST-1	48	33.2	1501	2.10 3.5	26.9	0.40	
310	4x6	32	22.0	1488	1.28 2.4	27.7	0.42	
410	4x6	48	33.3	1499	2.53 4.8	25.6	0.39	

MAT: AS4/8552 LAMINATE: $(0, 45, 90, -45)_{NS}$

ISSUES ON COMPRESSION AFTER IMPACT TEST

- IMPACT ENERGY OR IMPACT ENERGY/THICKNESS
- IMPACTOR SIZE
- SPECIMEN SIZE, THICKNESS
- OTHER VARIATIONS - DYNAMIC IMPACT
PRELOADING
MULTIPLE IMPACT

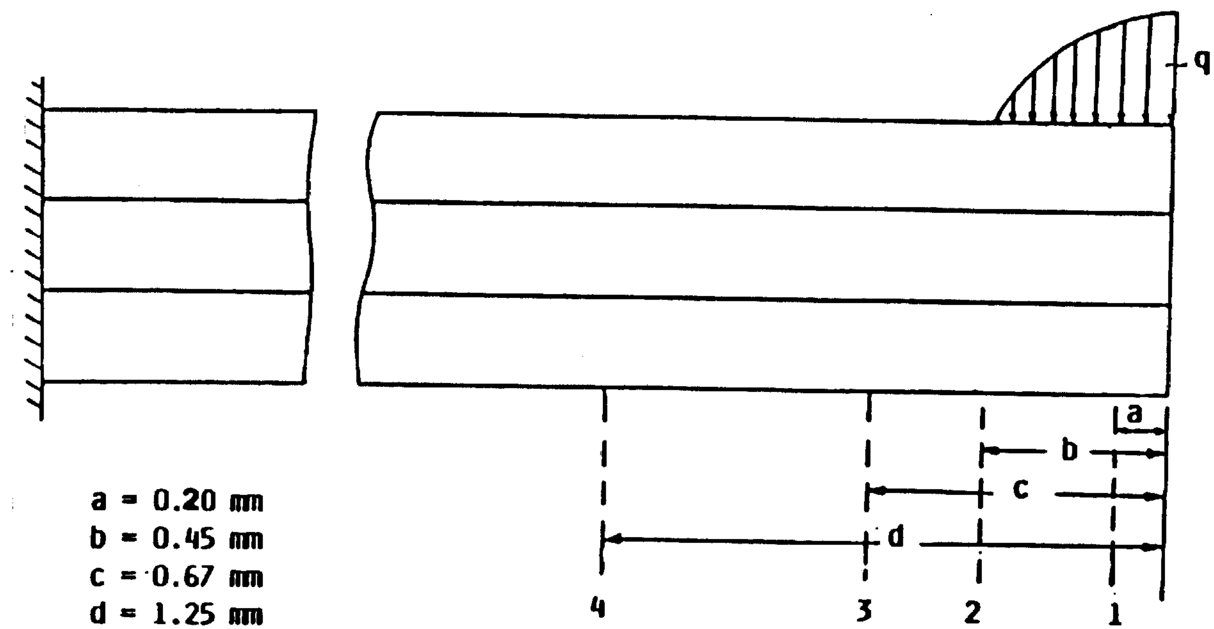


FIG. 12—*Stress output locations for Figs. 14 to 22.*

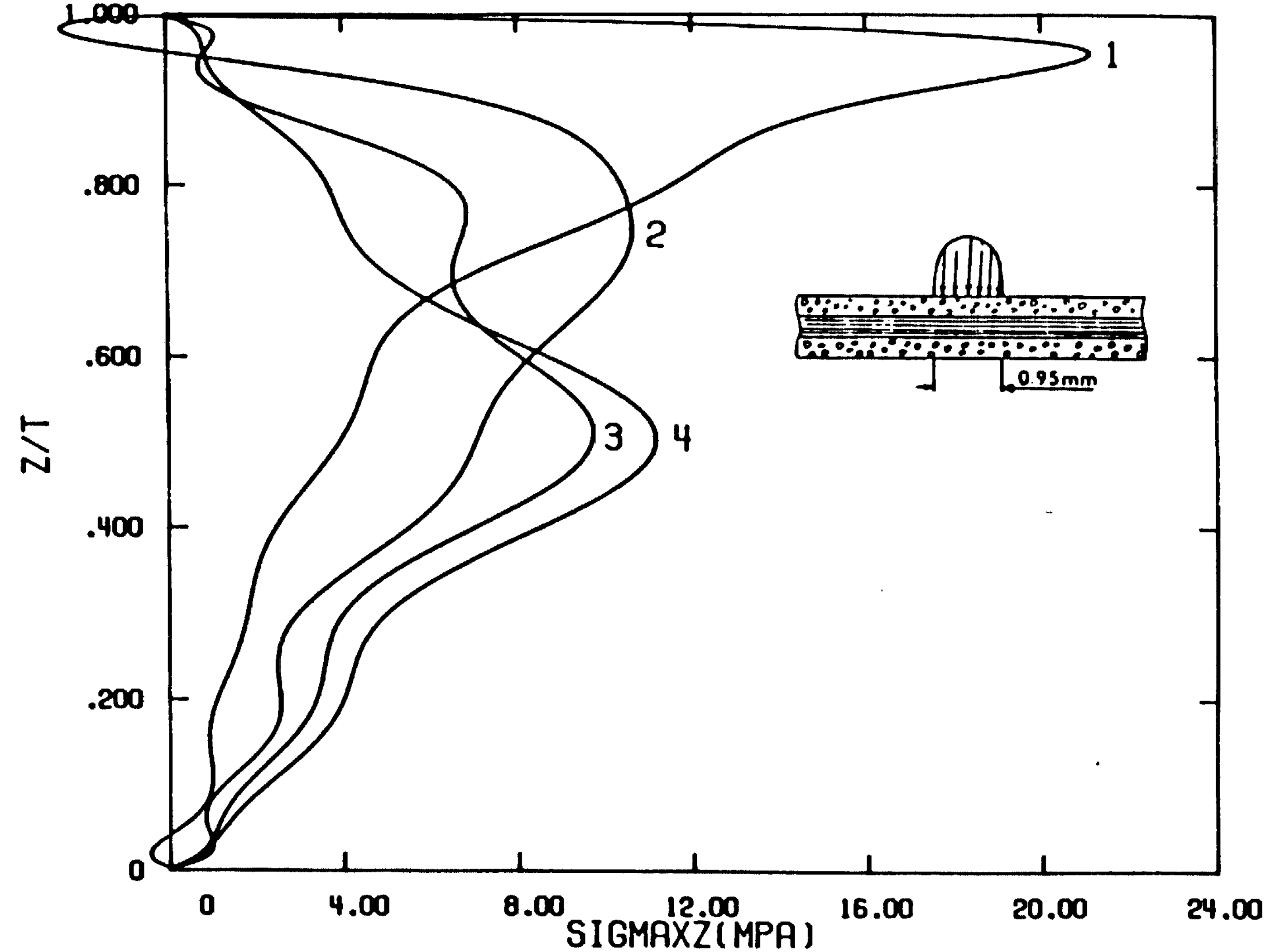


FIG. 14—*Transverse shear stress distribution in a [$90_s/0_s/90_s$] lay-up at 10 μs.*

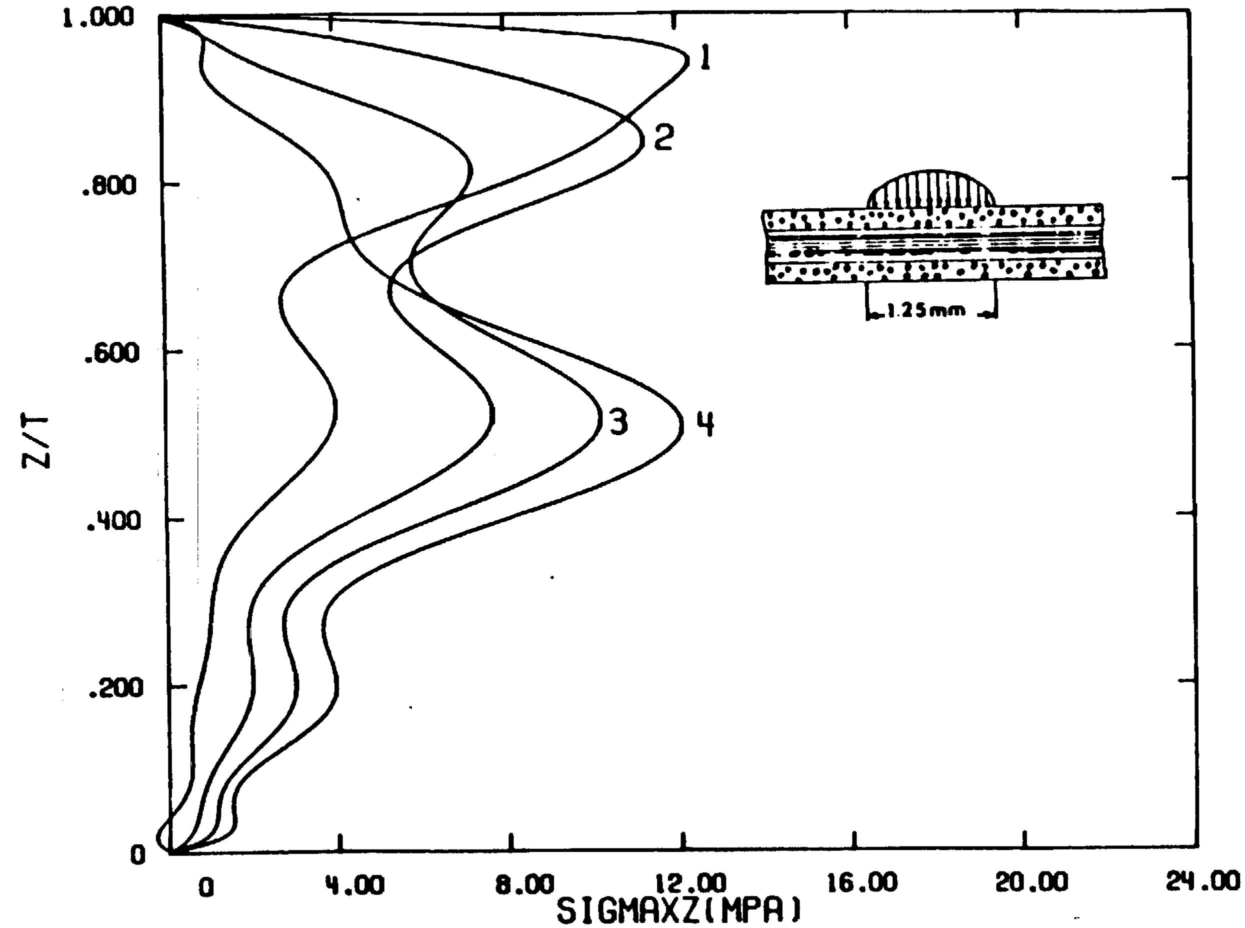


FIG. 18—*Effect of contact area on shear stress distribution of a* $[90_S/0_S/90_S]$ *laminate.*

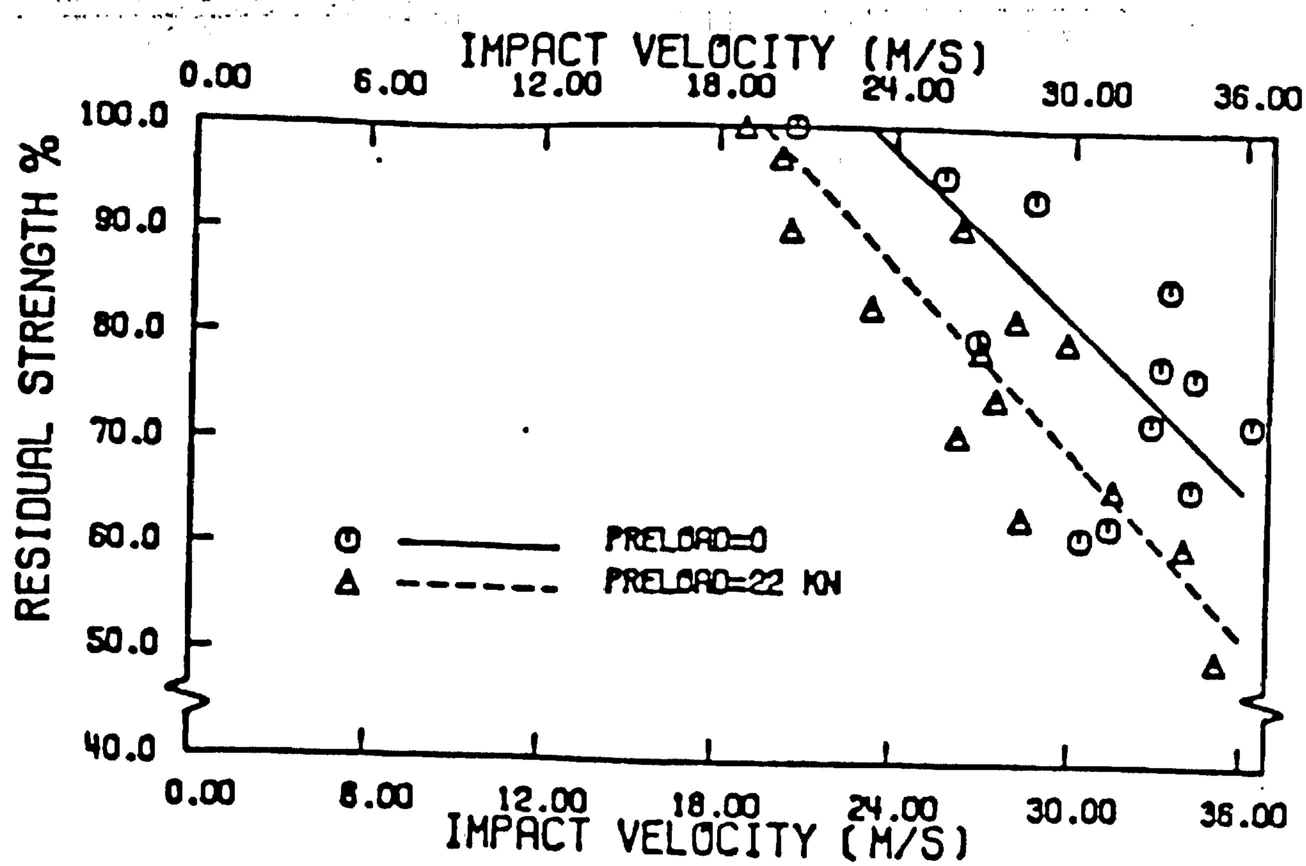
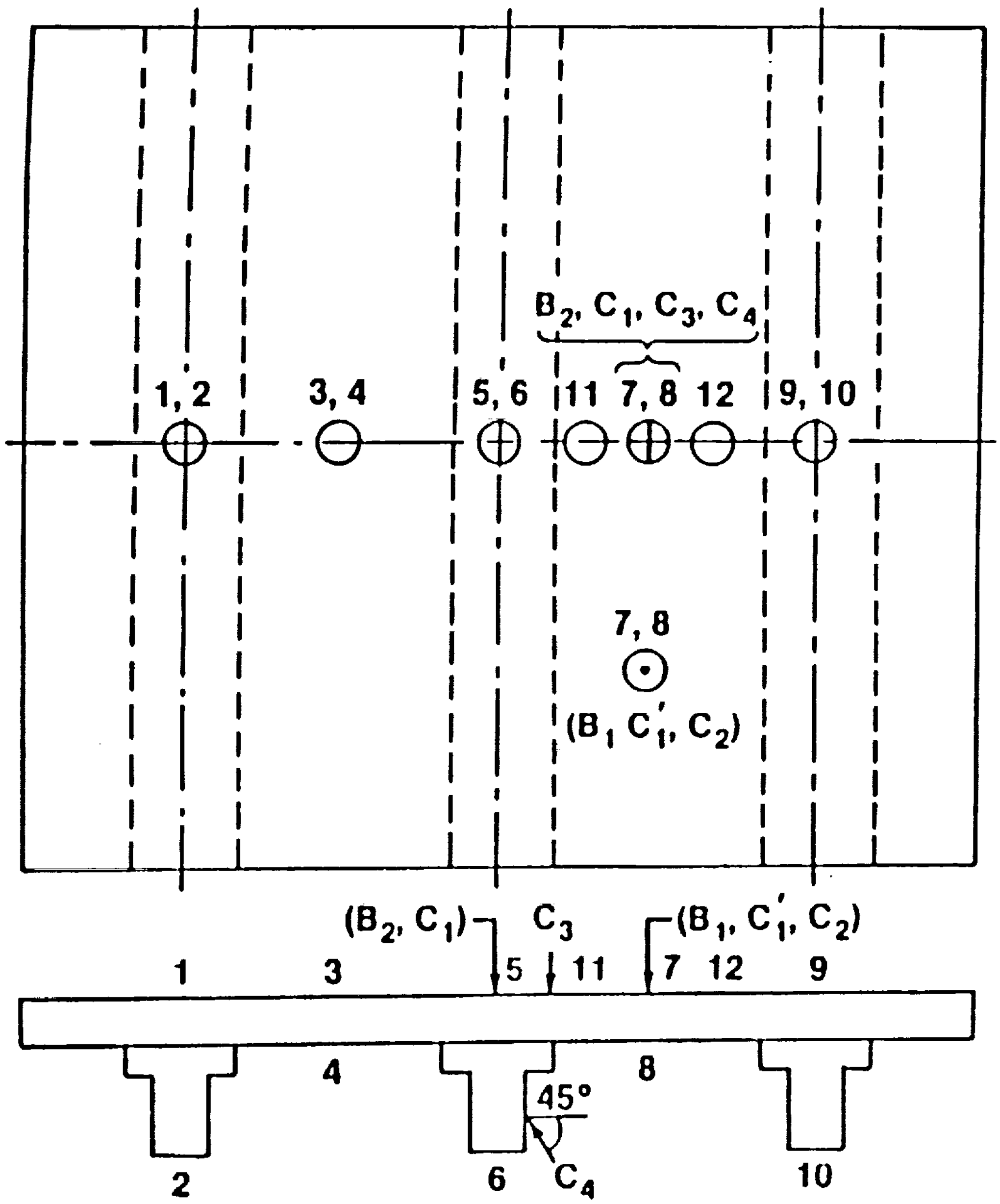


Fig. 2 Effect of initial stresses on the residual strength.

CAI OF STIFFENED PANELS

- IMPACT SITES
- SUPPORT DURING IMPACT
- AUXILIARY TESTS - PEEL MOMENT TEST
  PULL-OFF INTERACTION TEST
  STIFFENER CRIPPLING TEST

STRAIN GAGE AND IMPACT LOCATIONS ON PANELS



1,2 . . . 12 = STRAIN GAGE LOCATIONS
↓ = IMPACT SITES
$B_1\ C_1$, ETC. = PANEL NUMBERS

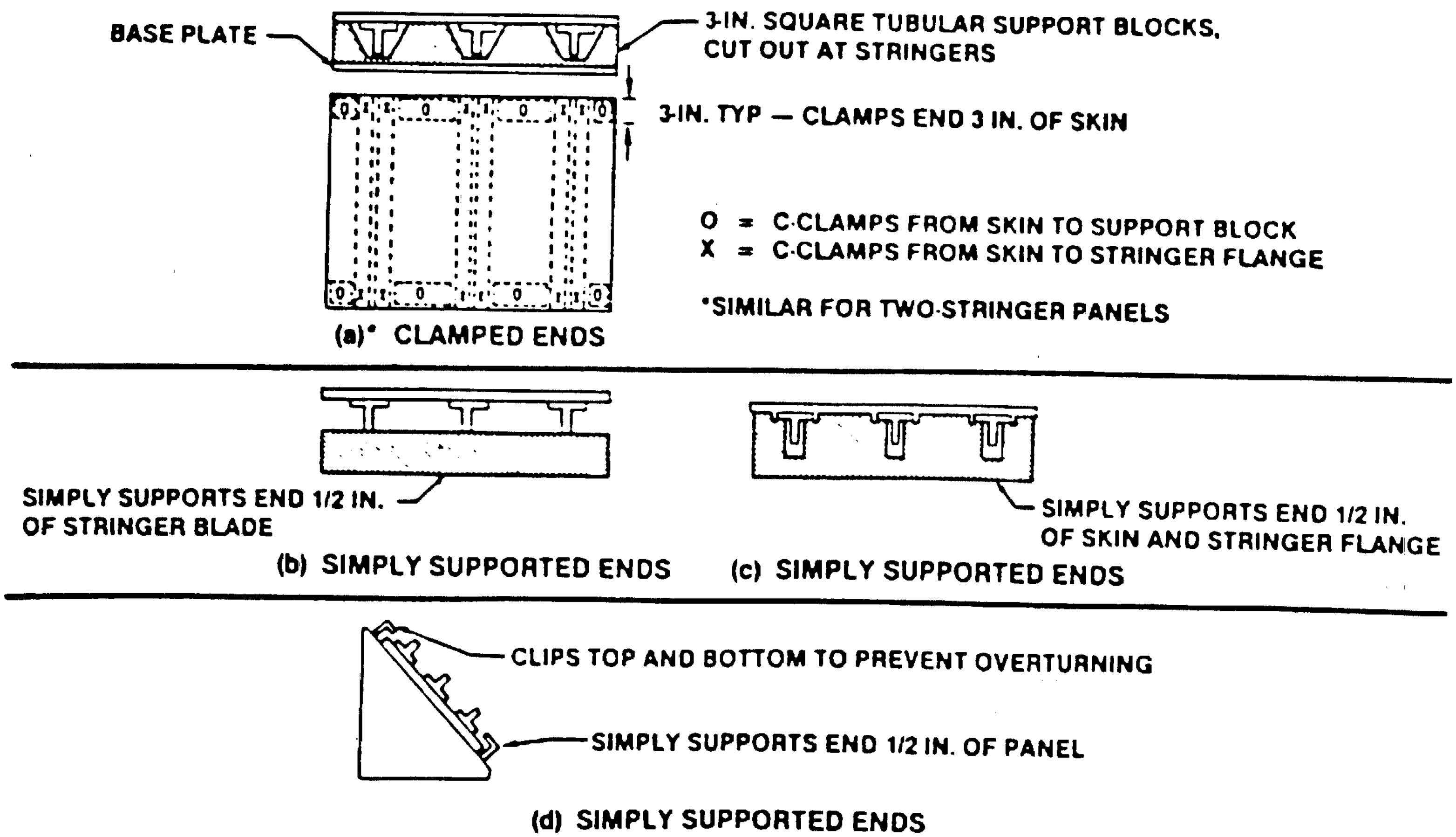
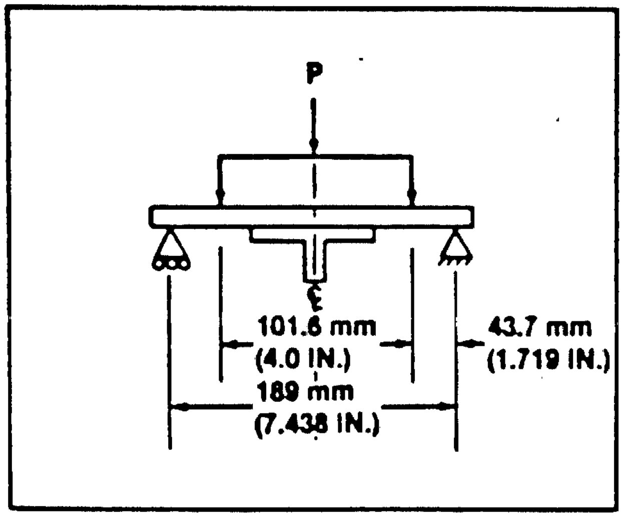
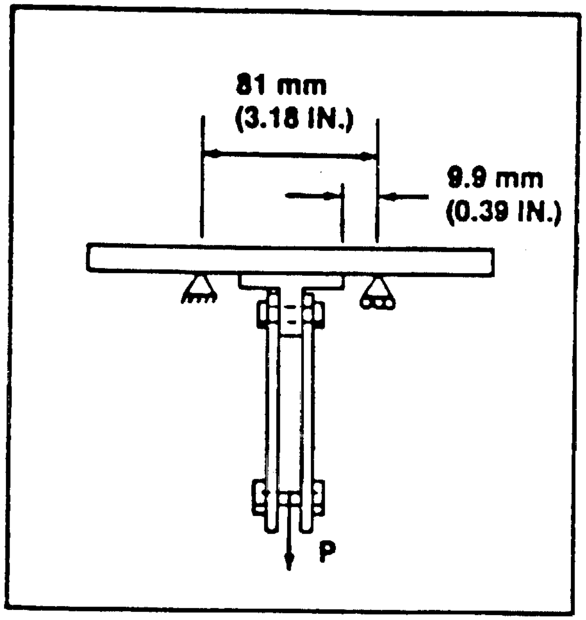


Figure 3. Support Conditions for Impact Tests



TEST RESULTS					
PANEL TYPE	TEST	P N	P (LB)	$M_p = 0.215$ P N-m/mm	(IN.-LB/IN.)
IV	1	7,784	(1,750)	1.67	(376)
	2	7,473	(1,680)	1.61	(361)
	3	7,673	(1,725)	1.65	(371)
AVERAGE				1.64	(369)
V	1	9,340	(2,100)	2.01	(451)
	2	9,674	(2,175)	2.08	(467)
	3	9,452	(2,125)	2.03	(457)
AVERAGE				2.04	(458)

Figure 4. Peel Moment Test



TEST RESULTS					
PANEL TYPE	TEST	RUNNING P N/mm	(LB/IN.)	$M_P = (P/2)(9.9\ mm)$ N·m/mm	(IN.-LB/IN.)
VI	1	144	(825)	0.72	(161)
	2	177	(1,008)	0.88	(197)
	3	137	(785)	0.68	(153)
AVERAGE				0.76	(170)
ADJUSTED DATA TO ACCOUNT FOR 3-DEG MISALIGNMENT					
VI	1	194	(1,105)	0.92	(207)
	2	239	(1,362)	1.09	(245)
	3	185	(1,058)	1.02	(229)
AVERAGE				1.01	(227)

Figure 5. Pull-Off Interaction Test

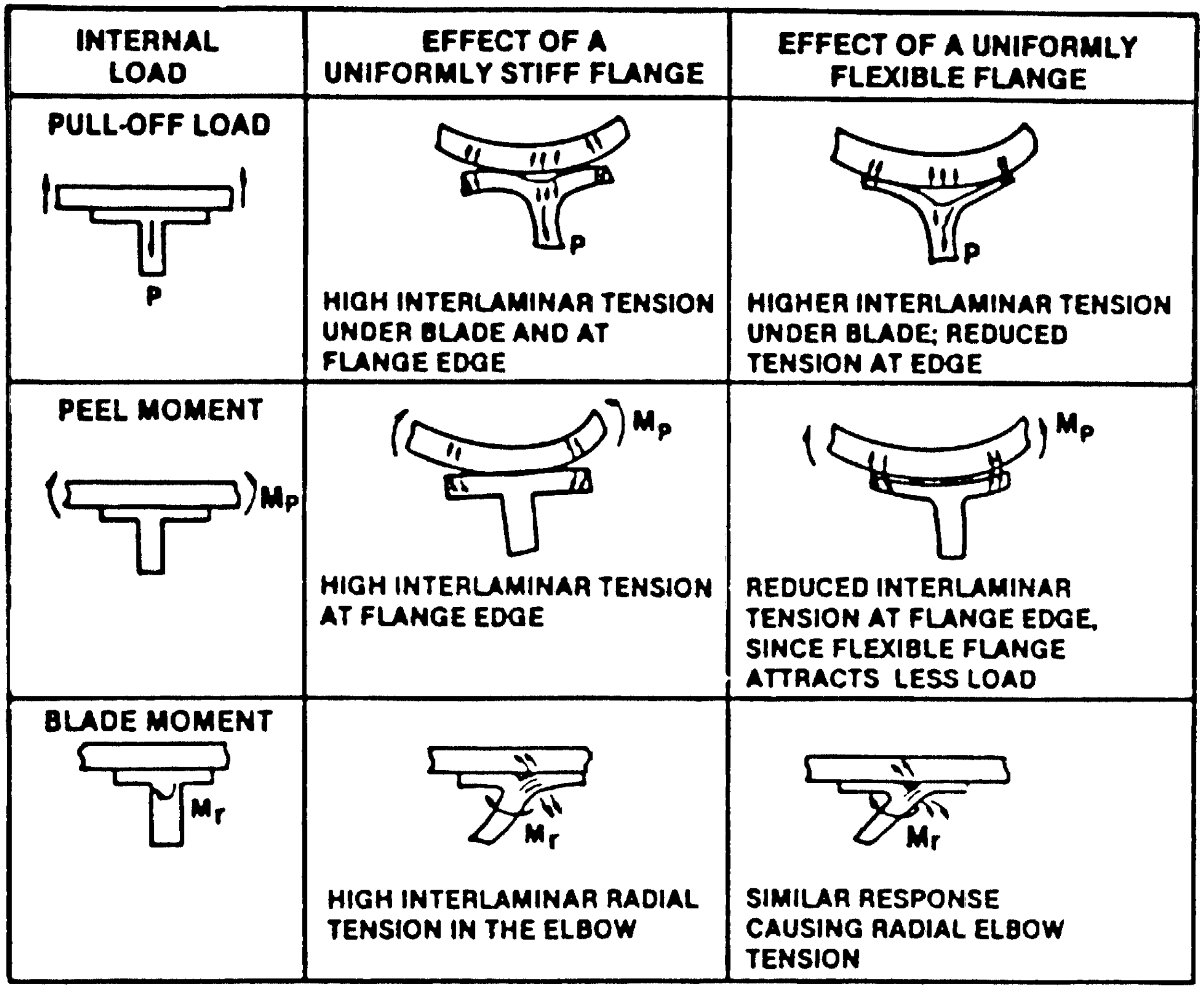


Figure 6. Effects of Internal Loads on Bonded Stringer Interlaminar Tension Stresses

TABLE 3
RESIDUAL COMPRESSION STRENGTH OF POSTIMPACT STIFFENED PANELS

PANEL ID	MATERIAL	LENGTH, WIDTH cm (IN.)	THICKNESS SKIN, FLANGE (NO. OF PLIES)	IMPACT LOCATION	IMPACT ENERGY J (FT-LB)	FAILURE STRESS MPa (KSI)	FAILURE STRAIN (PERCENT)	NO. OF STIFFENERS
1	T700/3620	40 (15.75) 39 (15.4)	18, 9	MID-STRINGER	4.3 (3.21)	—	0.385	3, HAT
2	T700/3620	40 (15.75) 39 (15.4)	18, 8	STRINGER EDGE	4 (3.05)	—	0.385	3, HAT
3	T700/3620	40 (15.75) 39 (15.4)	18, 9	MIDBAY	4.2 (3.15)	—	0.494	3, HAT
4	T700/3620	40 (15.75) 39 (15.4)	18, 9	STRINGER EDGE	12.26 (9.05)	—	0.298	3, HAT
5	CARBON/EPOXY TAPE	51 (20) 34.3 (13.5)	41, 41	MIDBAY	12.2 (9.0)	—	0.52	3, BLADE ADHESIVELY BONDED
6	CARBON/EPOXY	48.1 (18.9) 36.6 (14.4)	41, 41	MIDBAY,	14.2 (10.51)	—	0.37	3, BLADE ADHESIVELY BONDED
7	CARBON/EPOXY	48.8 (18.9) 57.2 (22.5)	48, 48	MID-STRINGER	13.4 (9.87)	—	0.47	3, HAT
8	AS4/3501-6	61 (24) 45.7 (18)	48, 24	MIDBAY, MID-STRINGER, STRINGER EDGE	135.5 (100) EACH	—	0.37	3, CHANNEL MECHANICALLY FASTENED
9	AS4/3501-6	61 (24) 45.7 (18)	48, 24	MIDBAY	135.5 (100)	—	0.38	3, CHANNEL MECHANICALLY FASTENED
10	AS4/3501-6	61 (24) 45.7 (18)	48, 24	MIDBAY, MID-STRINGER, STRINGER EDGE	112.5 (83) EACH	—	0.38	3, CHANNEL MECHANICALLY FASTENED
11	AS4/3501-6	61 (24) 45.7 (18)	48, 24	CENTERS OF TWO BAYS	81.3 (60) EACH	—	0.32	3, CHANNEL MECHANICALLY FASTENED
12	AS4/3501-6	61 (24) 45.7 (18)	48, 24	CENTERS OF TWO BAYS	54.2 (40)	—	0.35	3, CHANNEL MECHANICALLY FASTENED
13	AS6/5245-6	61 (24) 45.7 (18)	48, 24	MIDBAY, MID-STRINGER, STRINGER EDGE	135.5 (100) EACH	—	0.39	3, CHANNEL MECHANICALLY FASTENED
14	IM6/1808I	51 (20) 48 (19)	39, 14	MIDBAY	135.5 (100)	292 (42.3)	—	3, BLADE
15	IM6/1808I	51 (20) 48 (19)	54, 14	MIDBAY	135.5 (100)	361 (52.3)	—	3, BLADE A-BONDED
16	IM6/1808I	51 (20) 48 (19)	66, 14	MIDBAY	135.5 (100)	477 (69.1)	—	3, BLADE A-BONDED
17	IM6/1808I	51 (20) 48 (19)	40, 14	MIDBAY	135.5 (100)	271 (39.3)	—	3, BLADE A-BONDED
18	IM6/1808I	51 (20) 48 (19)	54, 36	MIDBAY	135.5 (100)	225 (32.6)	0.433	3, BLADE A-BONDED
19	IM6/1808I	51 (20) 48 (19)	54, 36	MID-STRINGER	135.5 (100)	267 (39)	0.400	3, BLADE A-BONDED
20	IM6/1808I	51 (20) 48 (19)	54, 18	MID-STRINGER	135.5 (100)	321 (46.5)	0.50	3, BLADE A-BONDED
21	IM6/1808I	51 (20) 48 (19)	54, 18	MIDBAY	135.5 (100)	267 (39)	—	3, BLADE A-BONDED
22	IM6/1808I	51 (20) 48 (19)	54, 18	MID-STRINGER	135.5 (100)	323 (47)	0.444	3, BLADE A-BONDED
23	IM6/1808I	51 (20) 48 (19)	54, 18	MIDBAY	271 (100)	184 (26.7)	0.333	3, BLADE A-BONDED
24	IM6/1808I	51 (20) 48 (19)	54, 18	SIDE OF BLADE	144 (106)	425 (617)	0.689	3, BLADE A-BONDED
25	IM7/8551-7	51 (20) 48 (19)	63, 18	MIDBAY	135.5 (100)	302 (43.8)	—	3, BLADE A-BONDED
26	IM7/8551-7	51 (20) 48 (19)	63, 18	MIDBAY	135.5 (100)	326 (47.3)	—	3, BLADE A-BONDED
27	IM6/1808I	142 (56) 84 (33)	54, 18	MIDBAY	135.5 (100)	—	0.410	3, BLADE A-BONDED
28	IM6/1808I	142 (56) 84 (33)	54, 18	MIDBAY	135.5 (100)	—	0.424	3, BLADE A-BONDED

WHY IMPACT TEST?

- o DAMAGE TOLERANCE CRITERIA
- o TYPE OF (IN-SERVICE) DAMAGE
- o DAMAGE RESISTANCE
- o COMPRESSION AFTER IMPACT TEST

PRELIMINARY DAMAGE TOLERANCE CRITERIA

Flaw/Damage Type	Flaw/Damage Size (1)
Scratches	Assume the presence of a surface scratch that is 4.0 in long and 0.02 in deep
Delamination	Assume the presence of an interply delamination that has an area equivalent to a 2.0-in diameter circle with dimensions most critical to its location (2)
Impact Damage	Assume the presence of damage caused by the impact of a 1.0-in diameter hemispherical impactor with 100 ft-lb of kinetic energy or with that kinetic energy required to cause a dent 0.10 in deep, whichever is least

(1) For limited access areas such as the interior of the wing, the contractor shall have the option of proposing an inspection procedure before closeout which will allow the assumed damage area size to be reduced

(2) This requirement also accounts for delamination that might occur and be nondetected as a result of in-service repair

IMPACT DAMAGE

MATRIX CRACKING

- TRANSVERSE SHEAR CRACK
- BENDING (TENSION) CRACK

DELAMINATION

FIBER BREAKAGE

- (THROUGH-THE-THICKNESS CRACK)

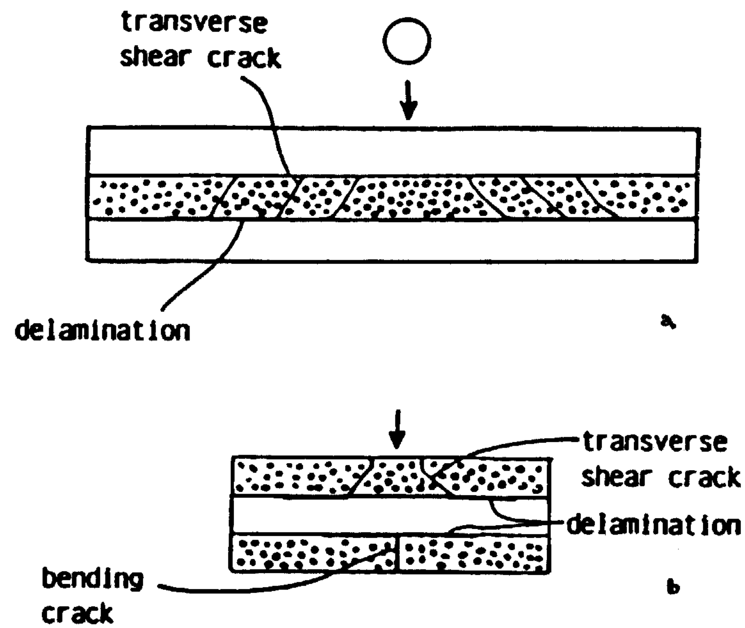
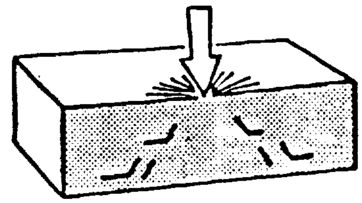
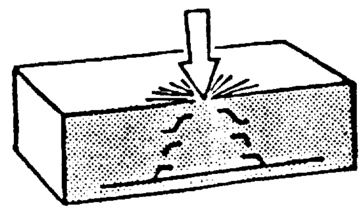


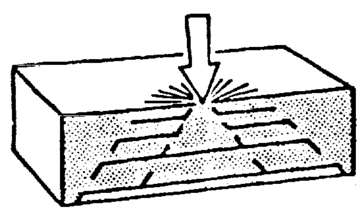
FIG. 1—*Nomenclature of damage modes.* (a) *Longitudinal section.* (b) *Transverse section.*



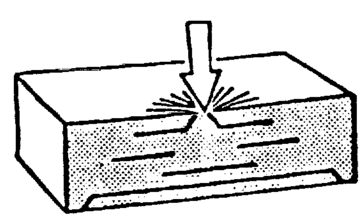
Mechanism 1: Tension, no delamination or shear



Mechanism 2: Tension and shear, tension dominated



Mechanism 3: Tension and shear, shear dominated



Mechanism 4: Shear, small amounts of tension

Figure 1. Fracture Mechanisms

DAMAGE RESISTANCE

IMPACT DAMAGE =

F (STACKING SEQUENCE, IMPACT ENERGY)

- o SPECIMEN SUPPORT
- o IMPACT FORCE
- o CONTACT STRESS DISTRIBUTION

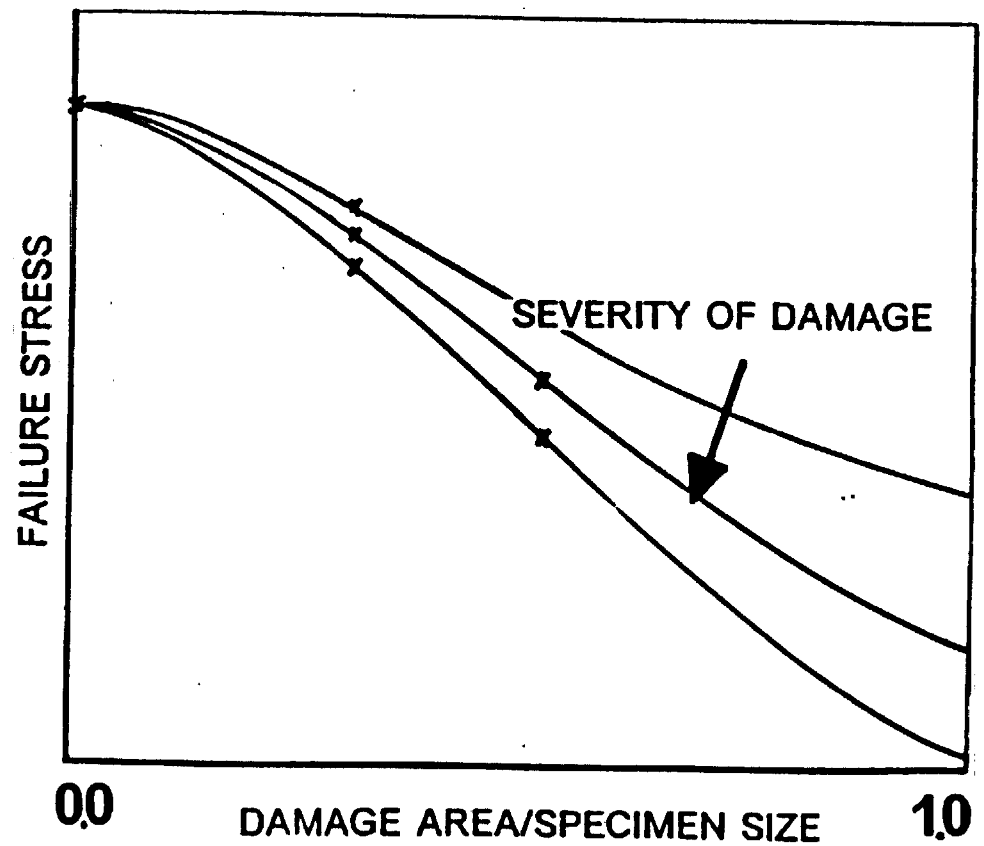
DAMAGE TOLERANCE

CAI STRENGTH

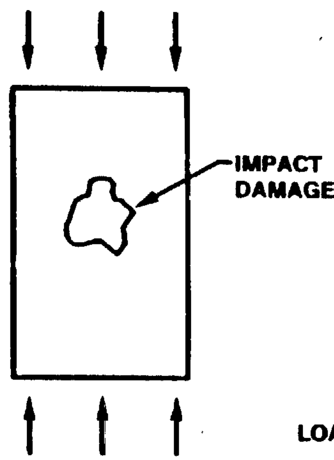
$$= F\left(\frac{\text{DAMAGE SIZE}}{\text{SPECIMEN SIZE}}\right)$$

- o WHAT KIND OF DAMAGE
- o SHOULD WE CONSIDER TAI
  SAI

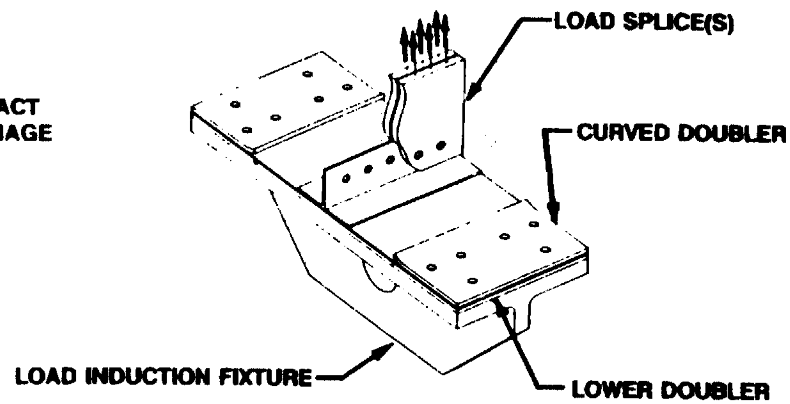
WHAT IS RESIDUAL STRENGTH



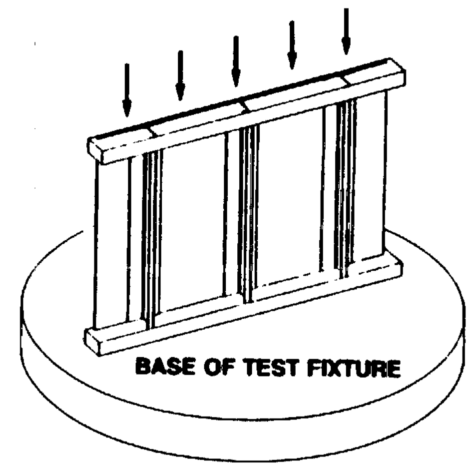
RESIDUAL STRENGTH PREDICTION DEVELOPMENT



CAI



STIFFENER PULL-OFF



DAMAGED STIFFENED
PANEL COMPRESSION TEST

ANALYSIS
- DAMAGE MODELING (REDUCED STIFFNESS AND STRENGTH)
- ULTIMATE STRENGTH PREDICTION ON DAMAGE PARTS (FINITE ELEMENT ANALYSIS WITH PARAMETERS OBTAINED FROM DAMAGE MODEL)

G154.88

Douglas Resin Selection

Resin Supplier Data → IRAD Screening → ICAPS Basic Properties → Selected Resin Design Properties

Resin Supplier Data:

- Specific Weight
- Modulus (RTD)
- Modulus (H/W)
- elongation (RTD)
- Glass Transition temperature
- Moisture Absorption

ICAPS Basic Properties:

- Tension
- Compression (RTD)
- Compression (H/W)
- CAI (RTD)
- CAI (H/W)
- Bearing By-pass

Selected Resin Design Properties:

- El
- Flt
- Flc
- Et, Ftt, Ftc
- vlt
- Puht
- Plht
- Fbru
- Puhc, Plhc
- F(0), F(90)

RELEVANCE OF IMPACTER SHAPE TO TENSION STRENGTH OF THICK CARBON/EPOXY WITH NONVISIBLE DAMAGE

C. C. Poe, Jr.
NASA Langley Research Center

NASA Workshop on Impact Damage to Composites
NASA Langley Research Center
Hampton, Virginia
March 19 & 20, 1991

OBJECTIVE

PRESENT METHOD TO PREDICT TENSION STRENGTH
WITH BARELY VISIBLE IMPACT DAMAGE --
IRRESPECTIVE OF IMPACTER SHAPE

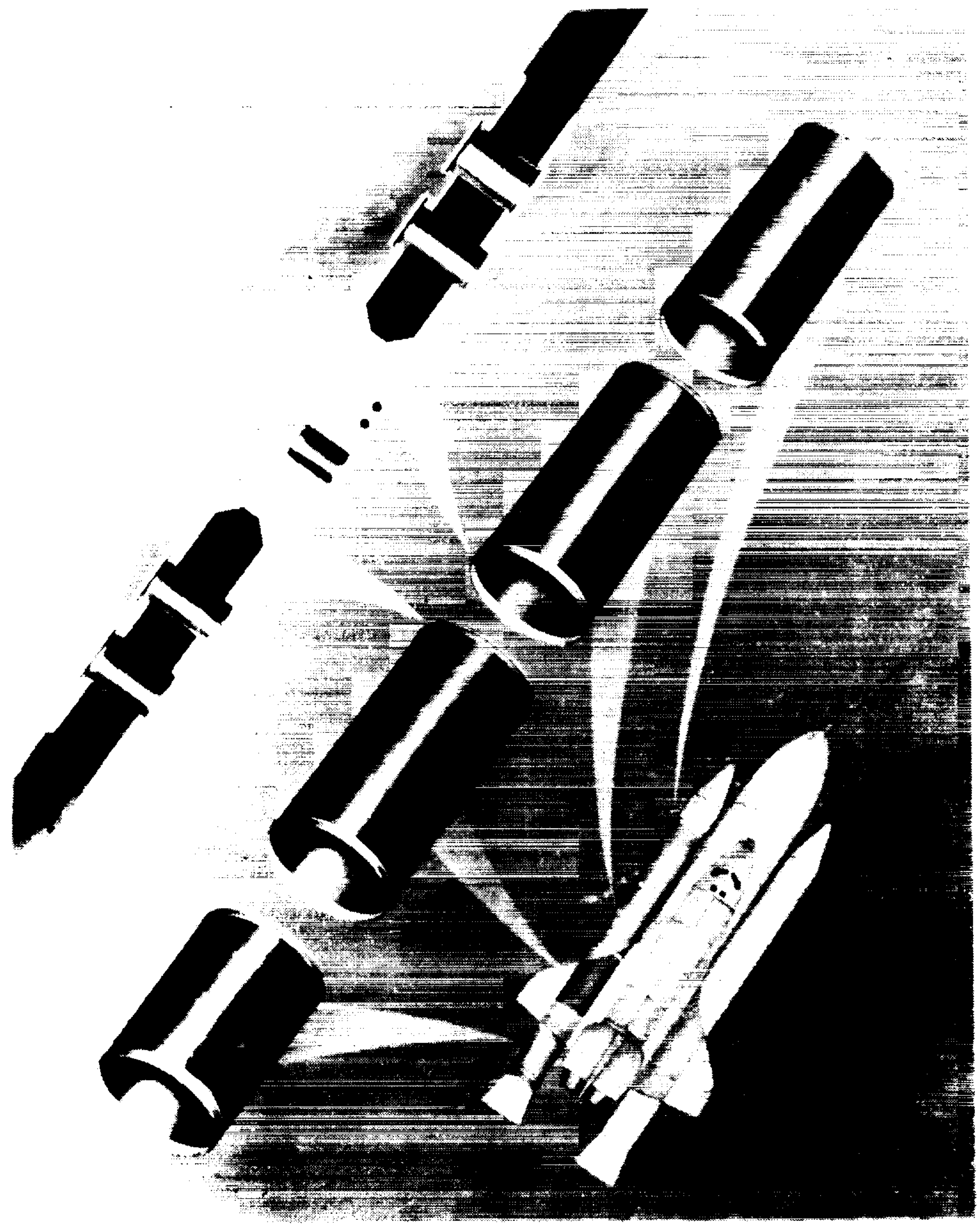


Figure 1.– Space shuttle solid rocket motor.

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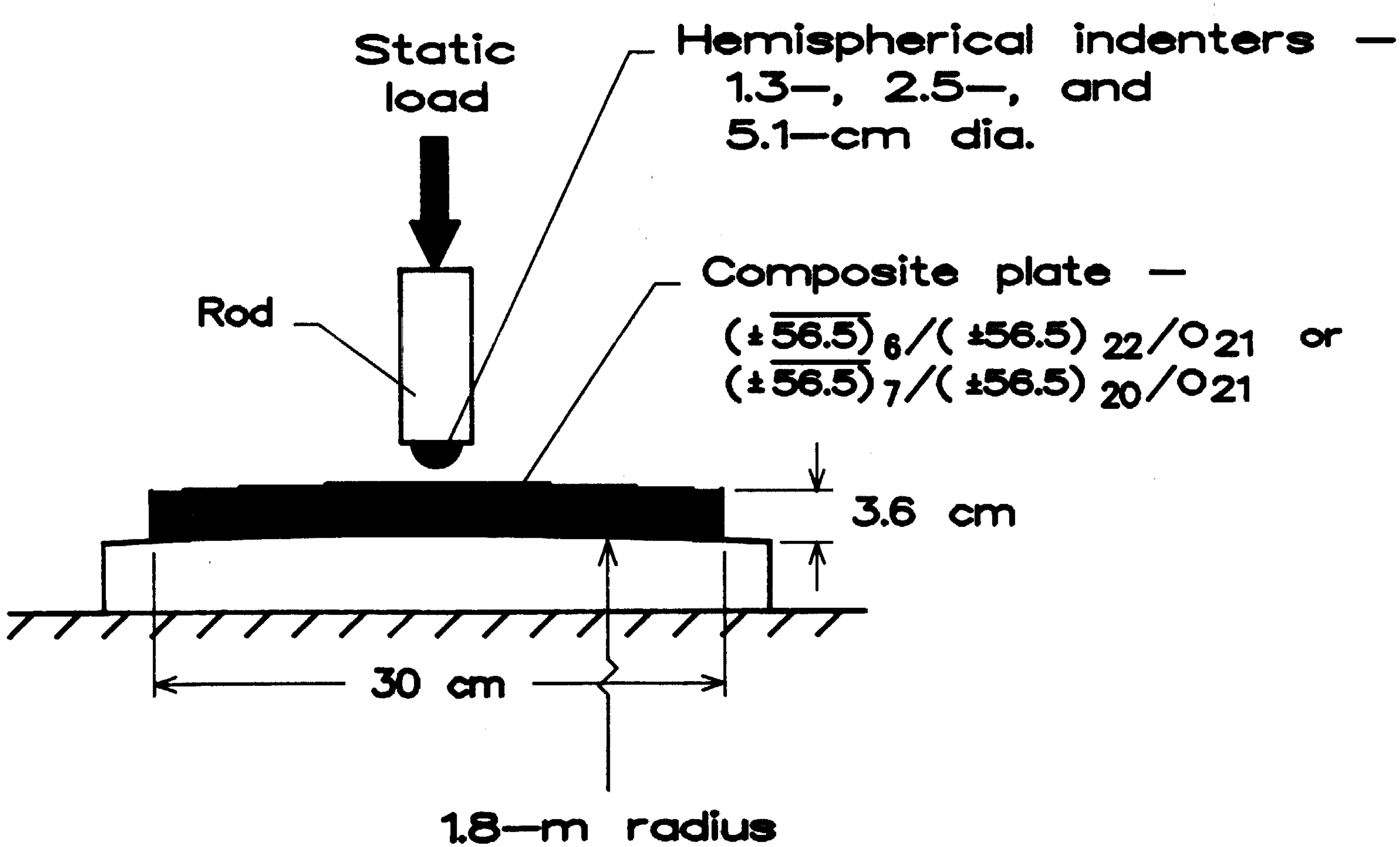
OUTLINE

PART I - DAMAGE FROM SIMULATED IMPACTS USING HEMISPHERICAL INDENTERS -- ANALYSIS & EXPERIMENTS

PART II - STRENGTH AFTER LOW-VELOCITY IMPACTS -- ANALYSIS & EXPERIMENTS

TEST SPECIMEN AND APPARATUS



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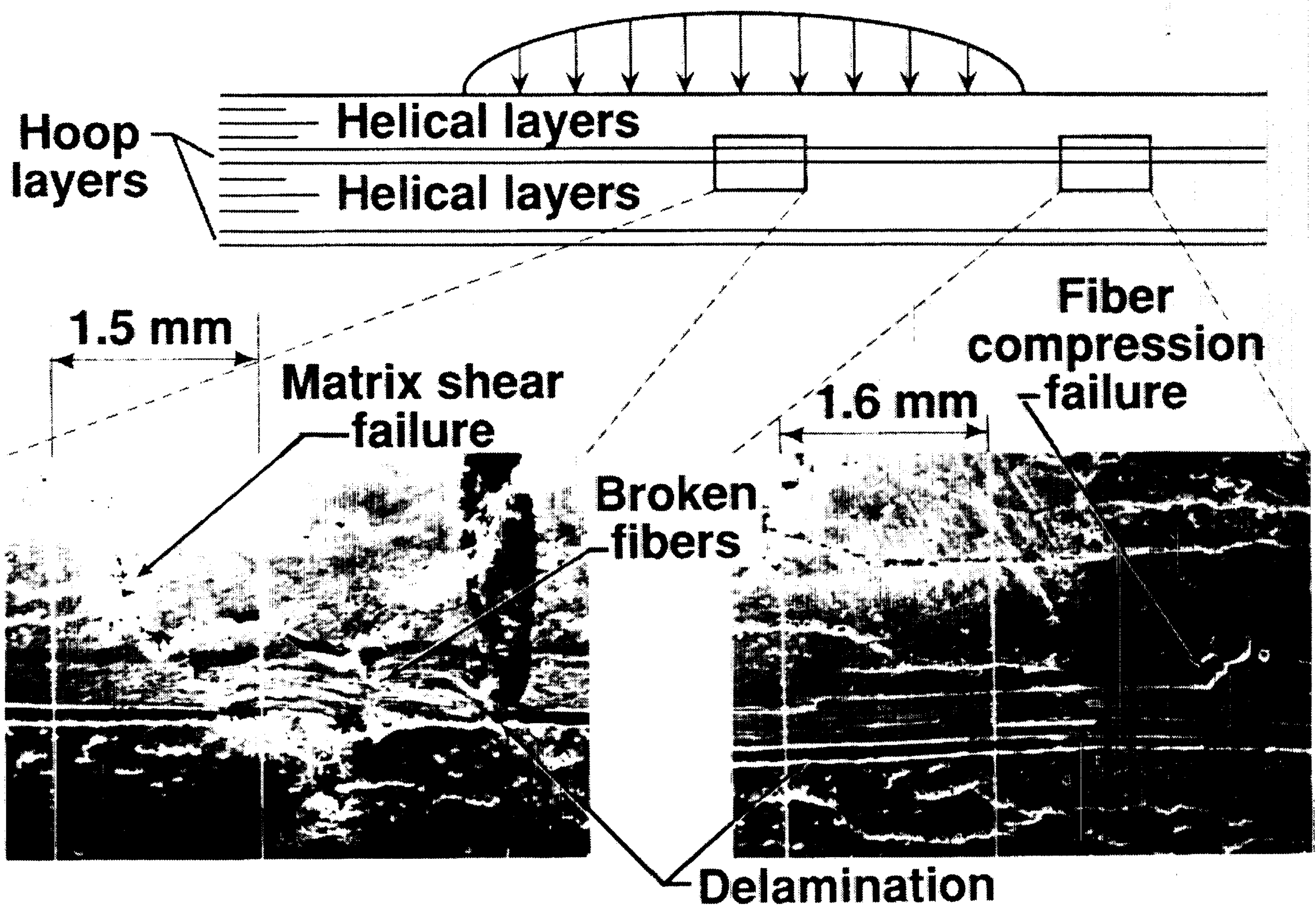
SEM PHOTOGRAPHS OF DAMAGE
5.08-cm-dia Indenter With 589 MPa Pressure



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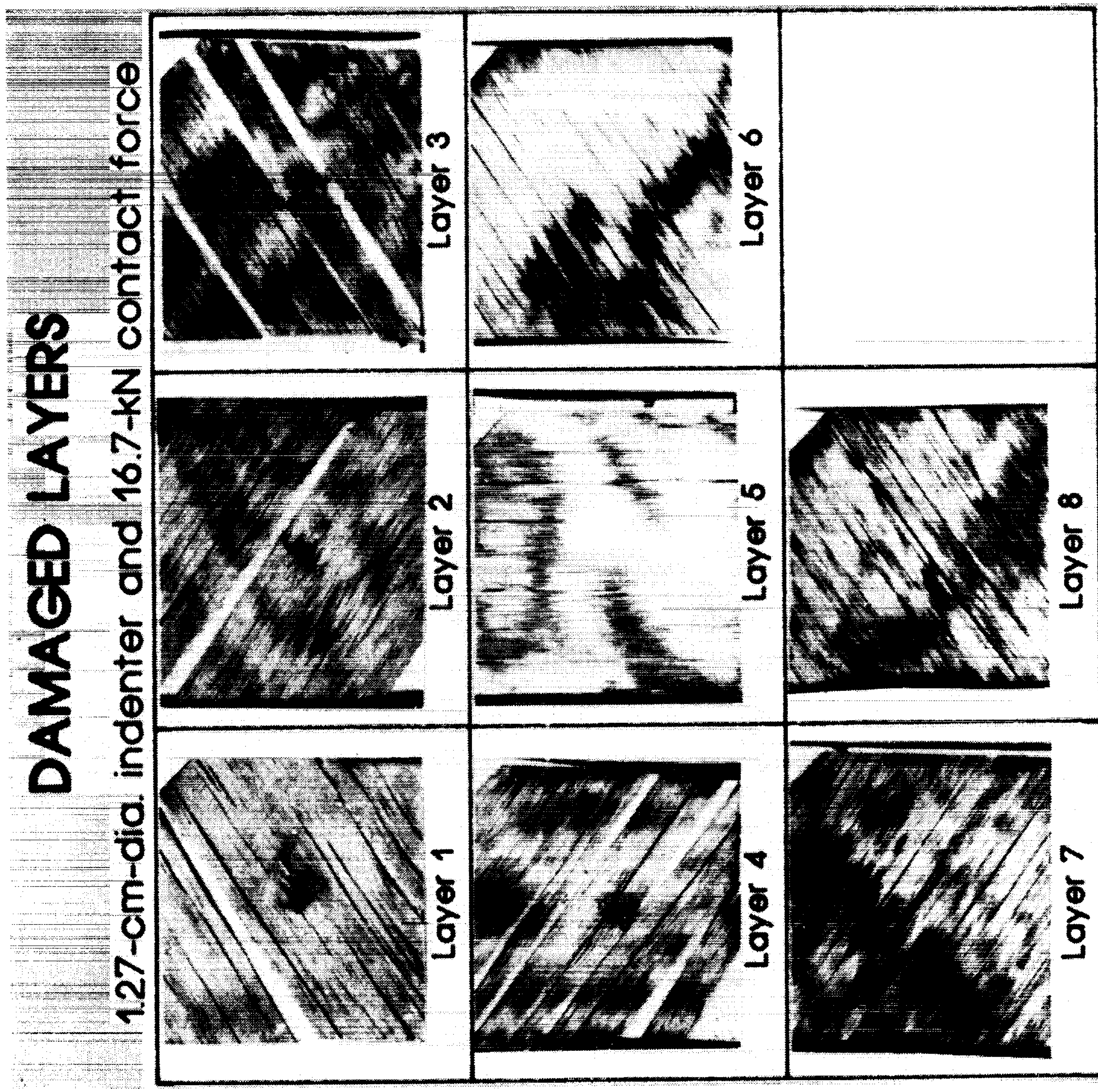
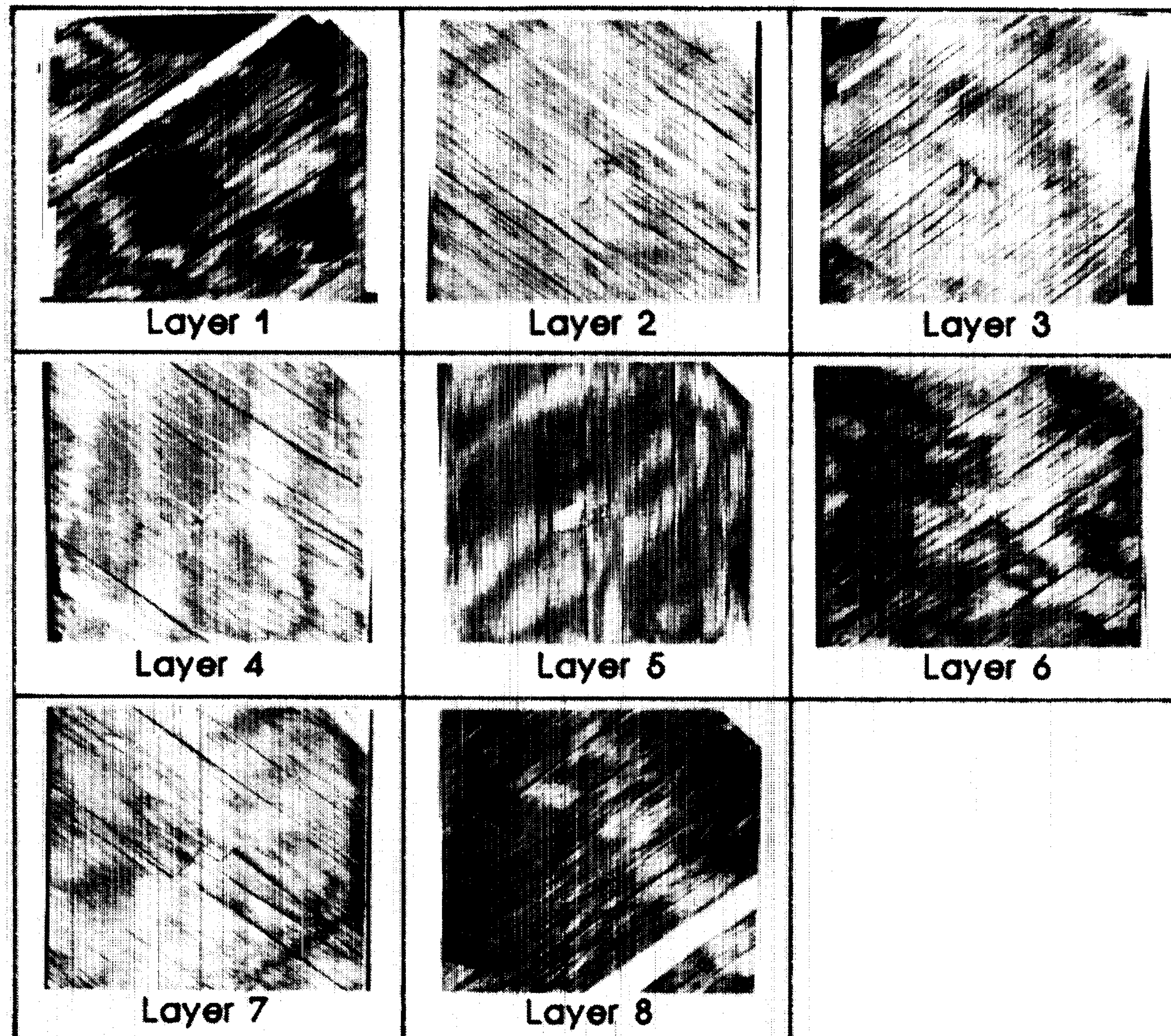

DAMAGED LAYERS
1.27-cm-dia. indenter and 16.7-kN contact force
Layer 1
Layer 2
Layer 3
Layer 4
Layer 5
Layer 6
Layer 7
Layer 8

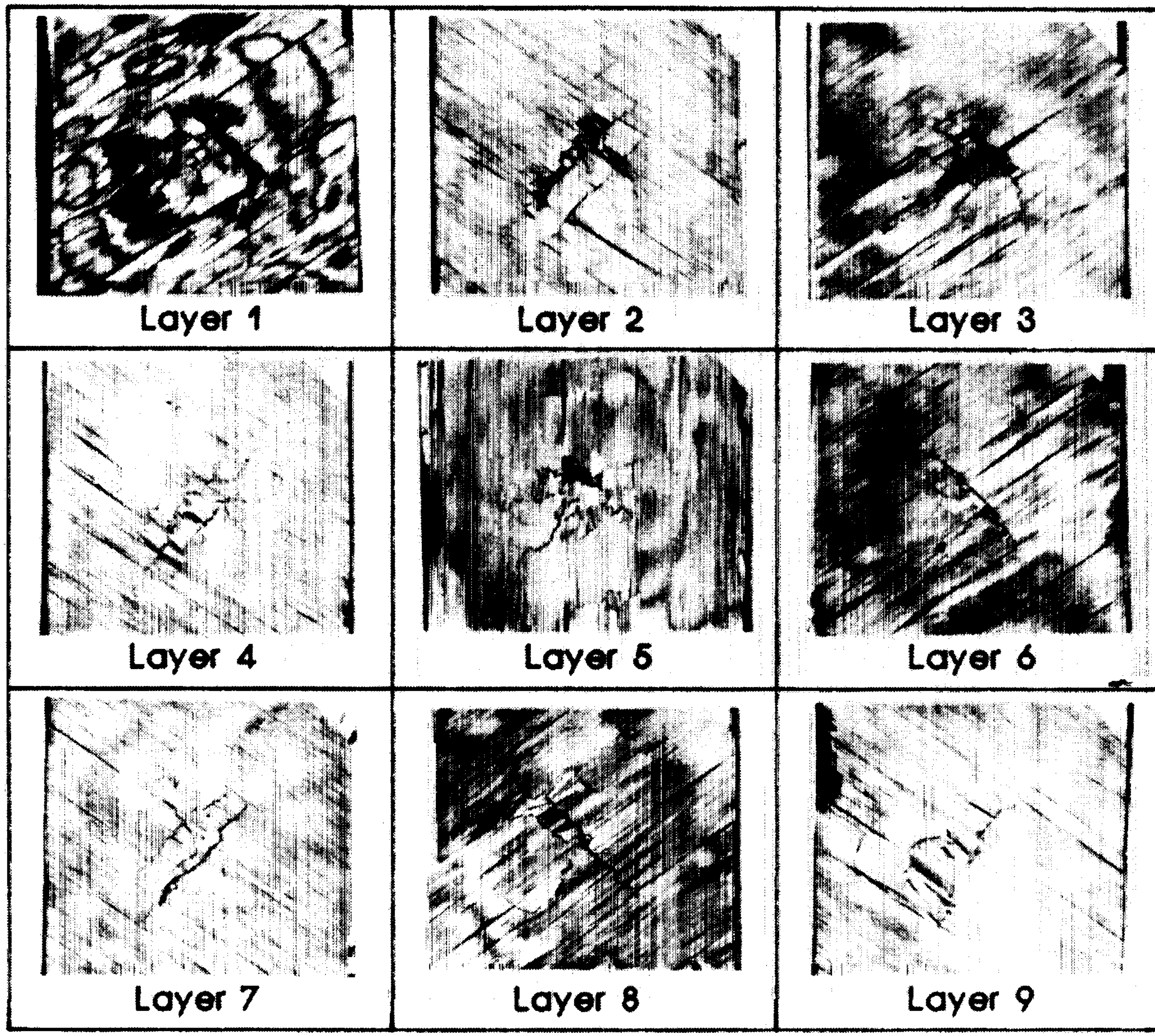
DAMAGED LAYERS

2.54-cm-dia. indenter and 66.7-kN contact force



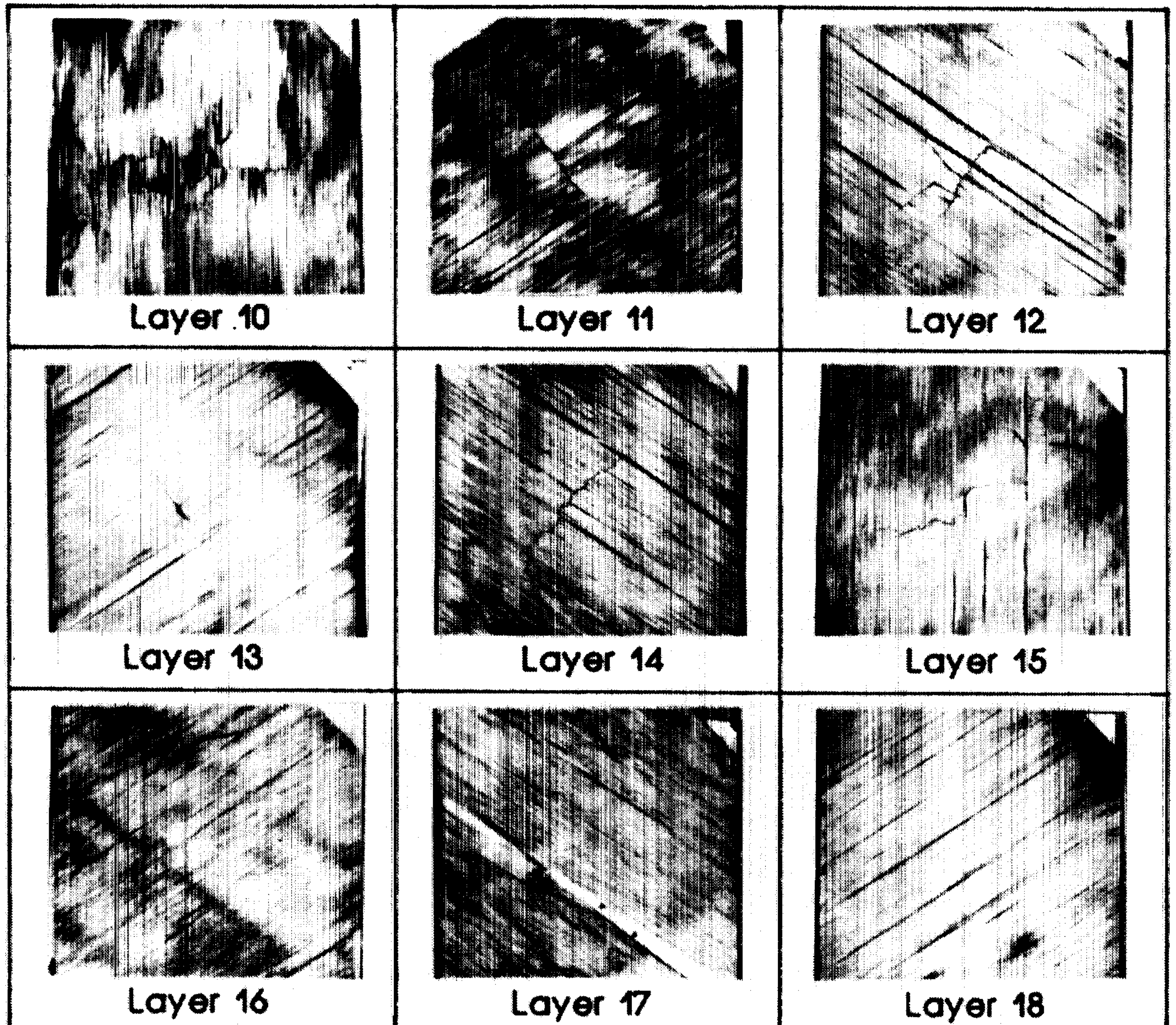
DAMAGED LAYERS 1-9

5.08-cm-dia. indenter and 267-kN contact force



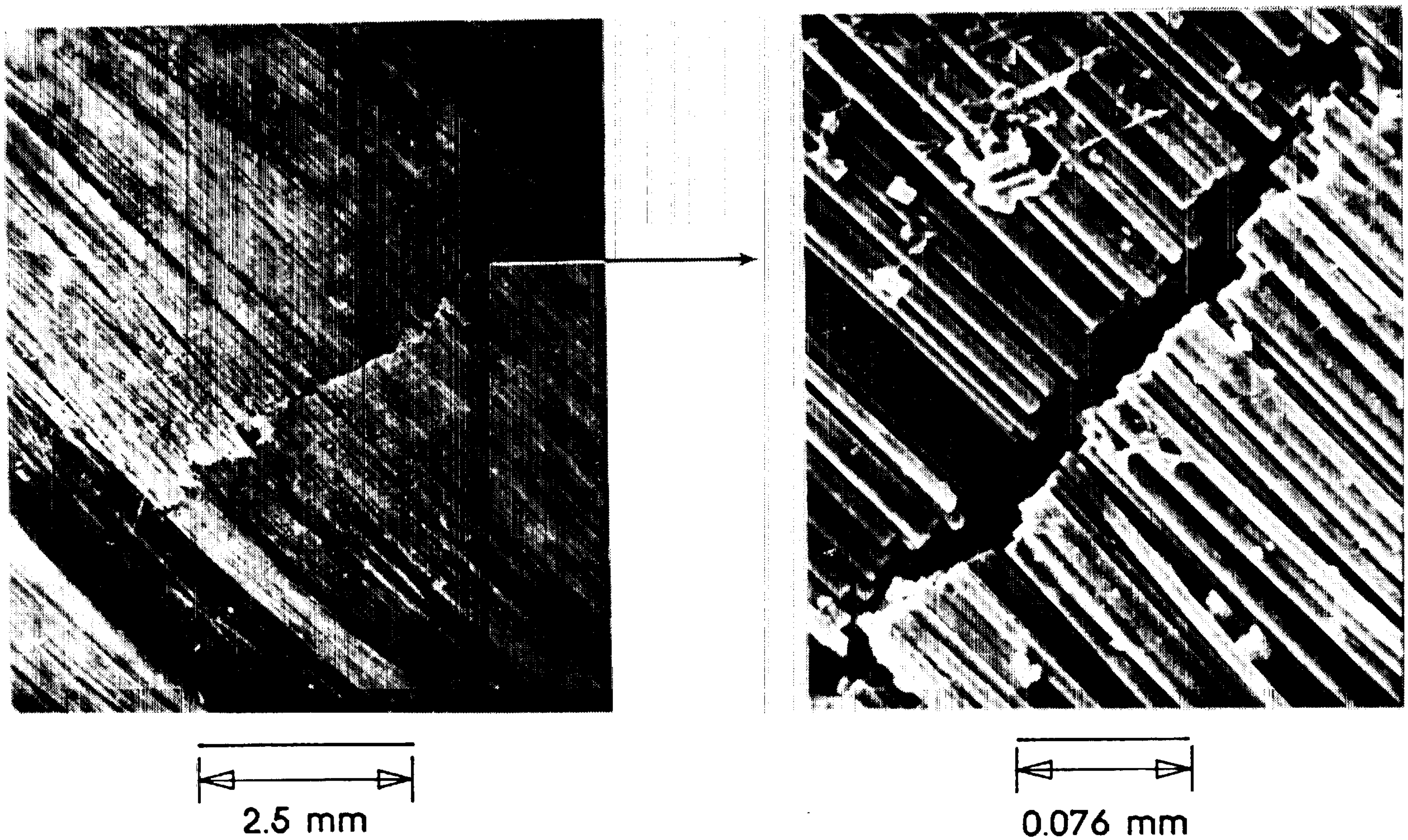
DAMAGED LAYERS 10-18

5.08-cm-dia. indenter and 267-kN contact force

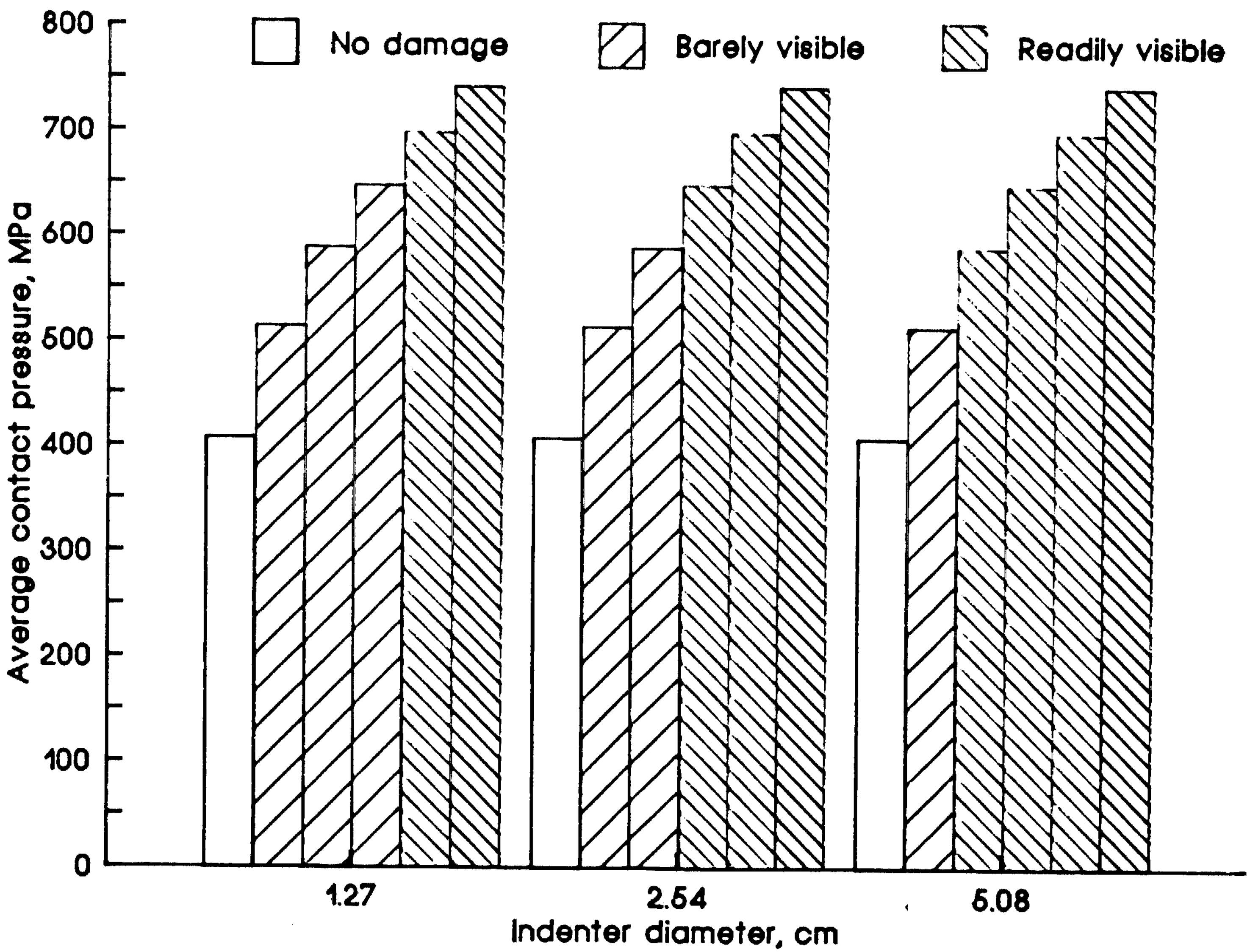


MAGNIFIED IMPACT DAMAGE IN 7TH LAYER

2.54-CM-INDENTER AND 54.3-KN IMPACT FORCE

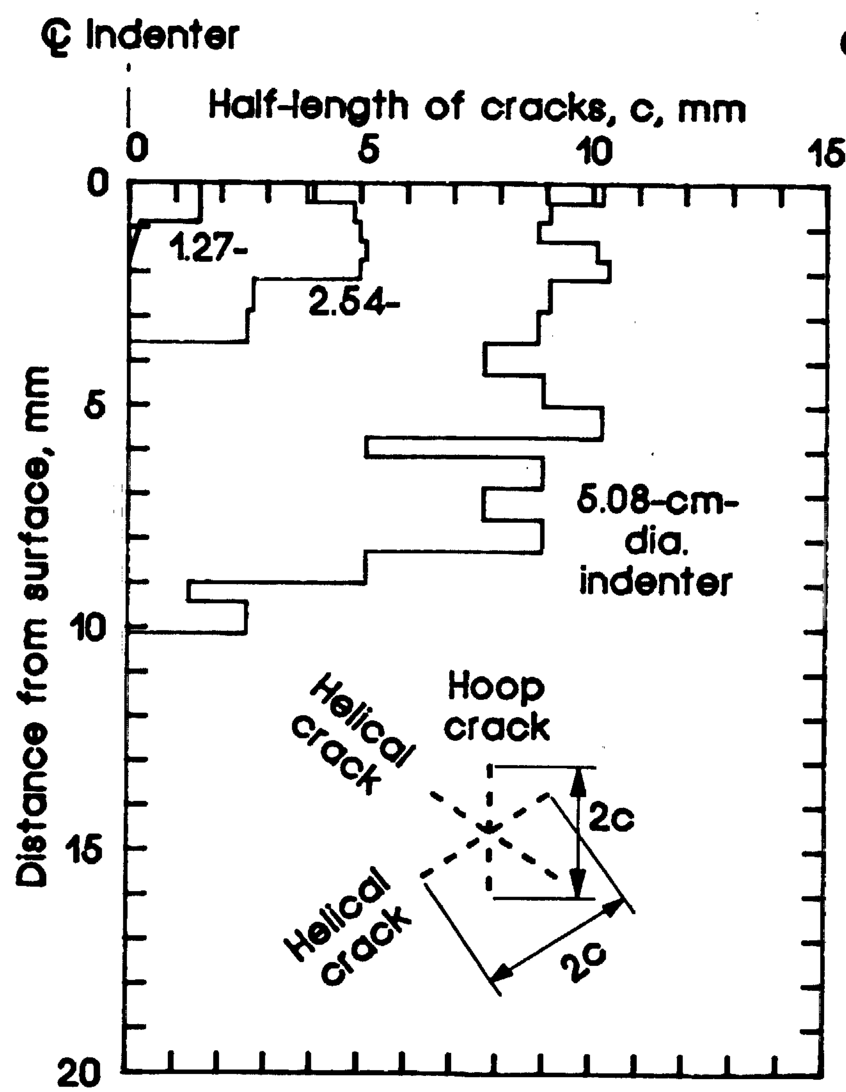


VISIBILITY OF SURFACE DAMAGE

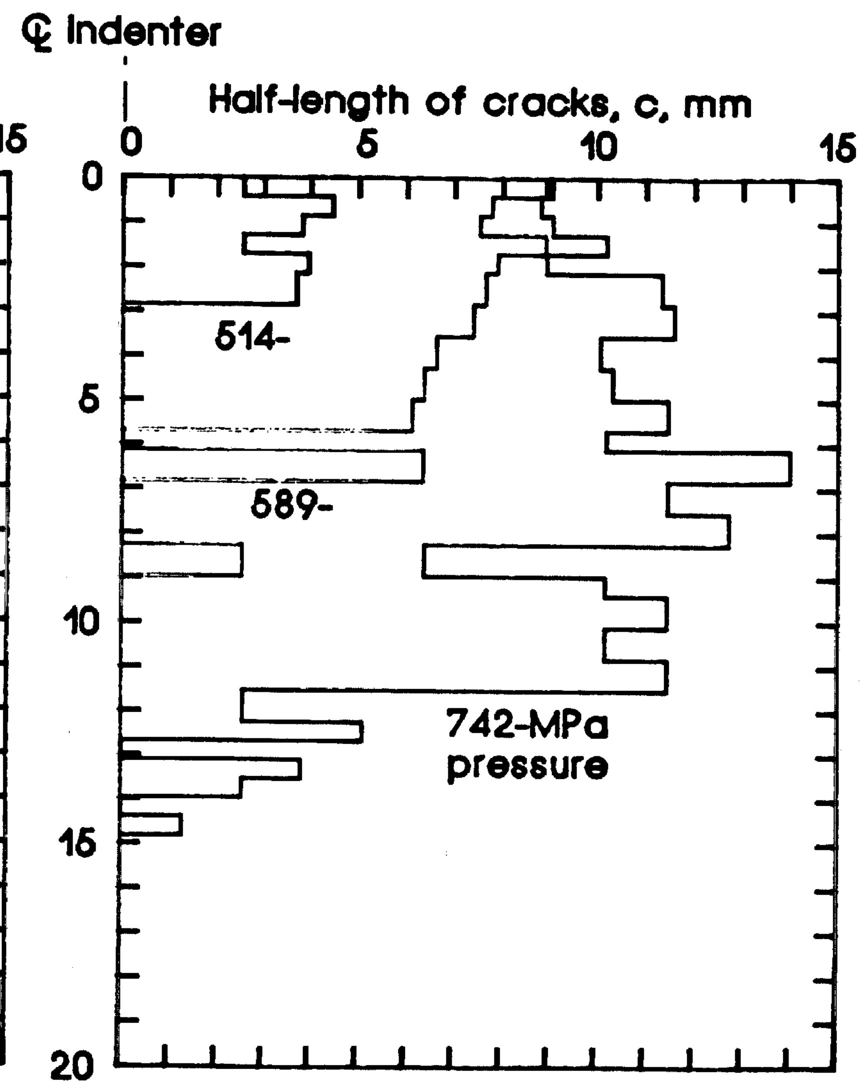


SLIDE16.ANN

FIBER BREAK PROFILES FROM INDENTATION TESTS



AVERAGE CONTACT PRESSURE
OF 648 MPa



INDENTER DIAMETER
OF 5.08 CM

\ASC88\SLIDE3C

ANALYTICAL APPROACH

- CALCULATE CONTACT STRESSES USING A. E. H. LOVE'S SOLUTION FOR HEMISPHERICAL PRESSURE APPLIED TO ISOTROPIC, SEMI-INFINITE BODY.

- CALCULATE DAMAGE SIZE USING MAXIMUM STRESS CRITERIA.

HERTZ LAW

FOR TRANSVERSELY ISOTROPIC BODIES -

Indentation.-

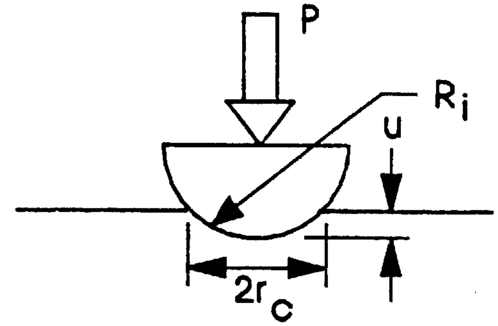
$$u = R_i^{-1/3}(P/n_o)^{2/3}$$

Pressure.-

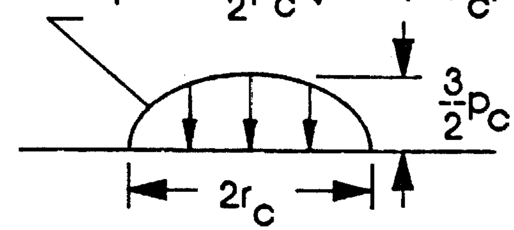
$$p_c = (n_o/R_i)^{2/3}P^{1/3}/\pi$$

Contact radius.-

$$r_c = (PR_i/n_o)^{1/3}$$
$$= \pi p_c R_i/n_o$$



$$p(r) = \frac{3}{2}p_c\sqrt{1-(r/r_c)^2}$$



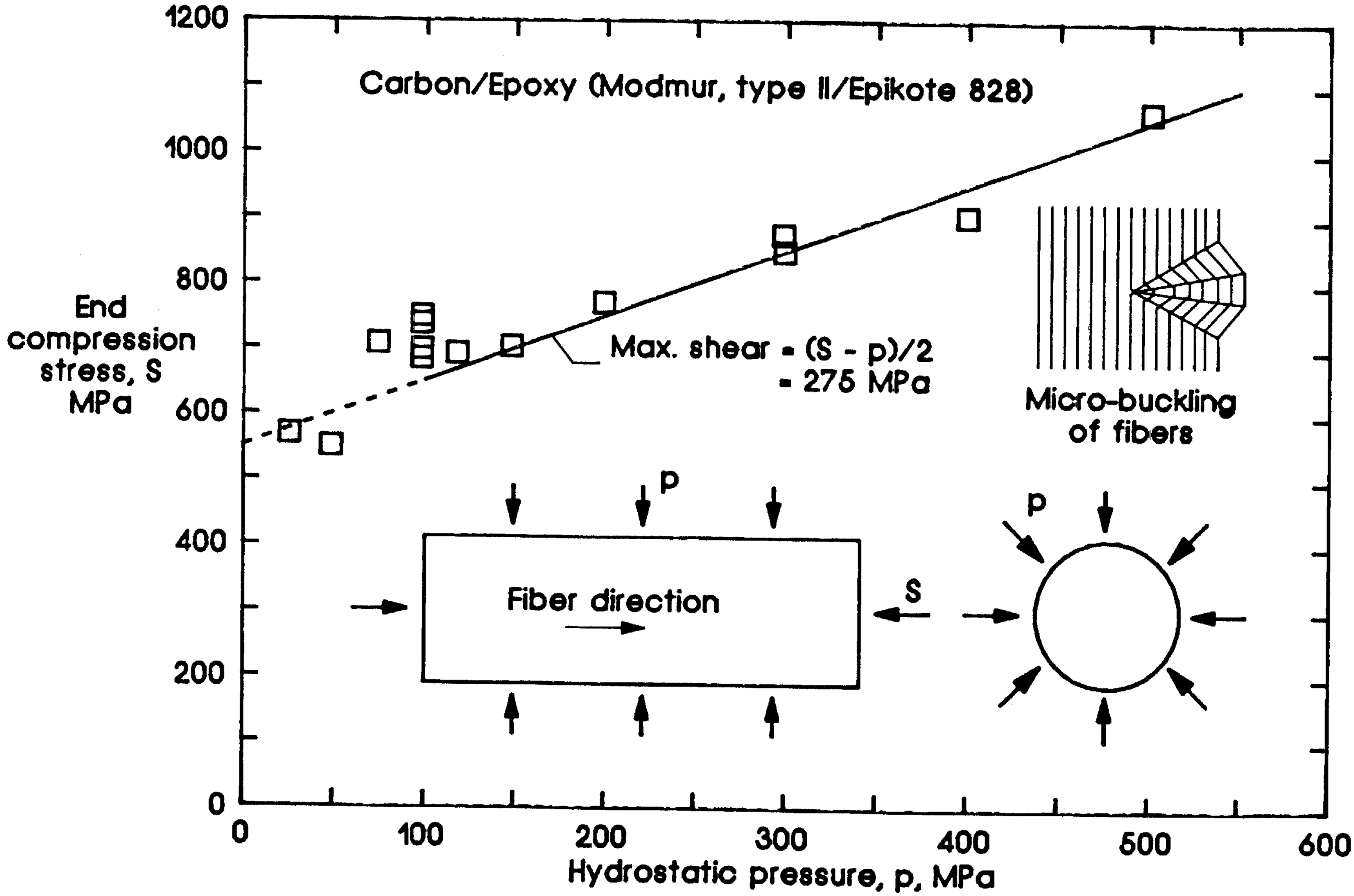
$$P = 2\pi\int_0^{r_c} p(r)\,r\,dr$$

where

n_o = 4.52 GPa from indentation measurements
= 4.69 GPa from elastic constants
= 3.98 GPa from contact radii measurements (previous investigation)

SLIDEX.ANN

FAILURE UNDER COMPRESSION AND HYDROSTATIC PRESSURE



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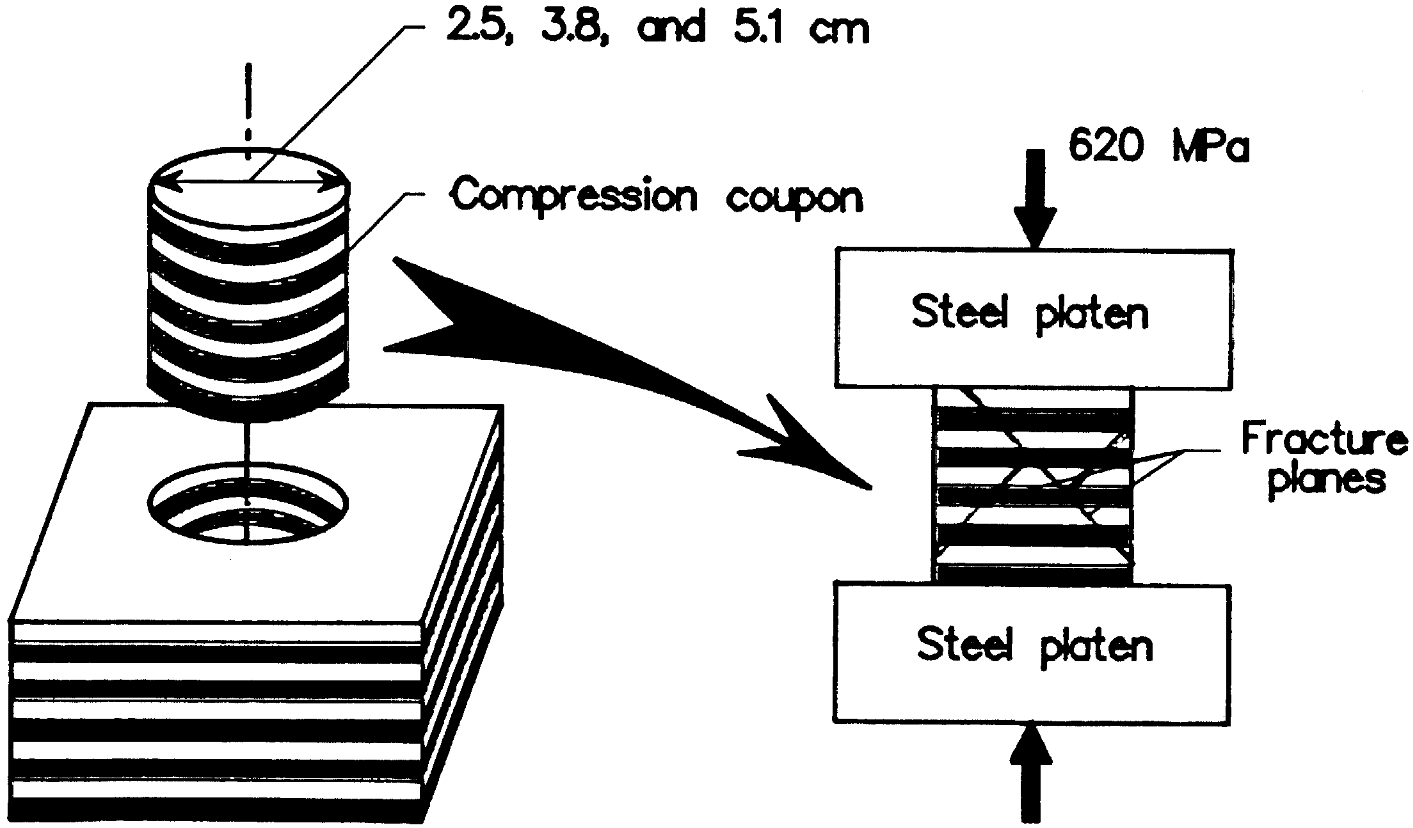
COMPRESSION TEST FOR SHEAR STRENGTH



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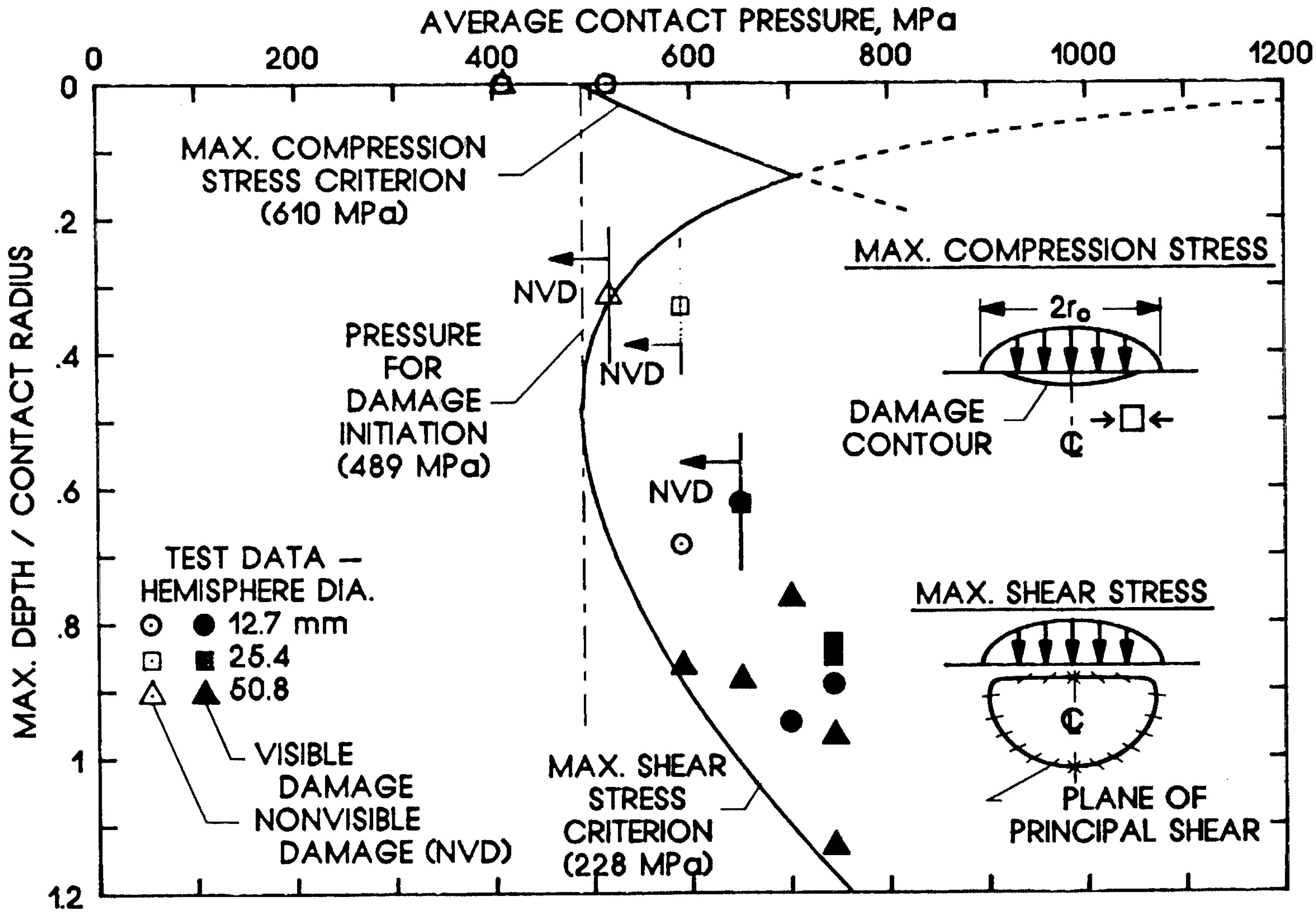
EFFECT OF IMPACTER RADIUS ON DAMAGE DEPTH IN DEPLIED LAYERS



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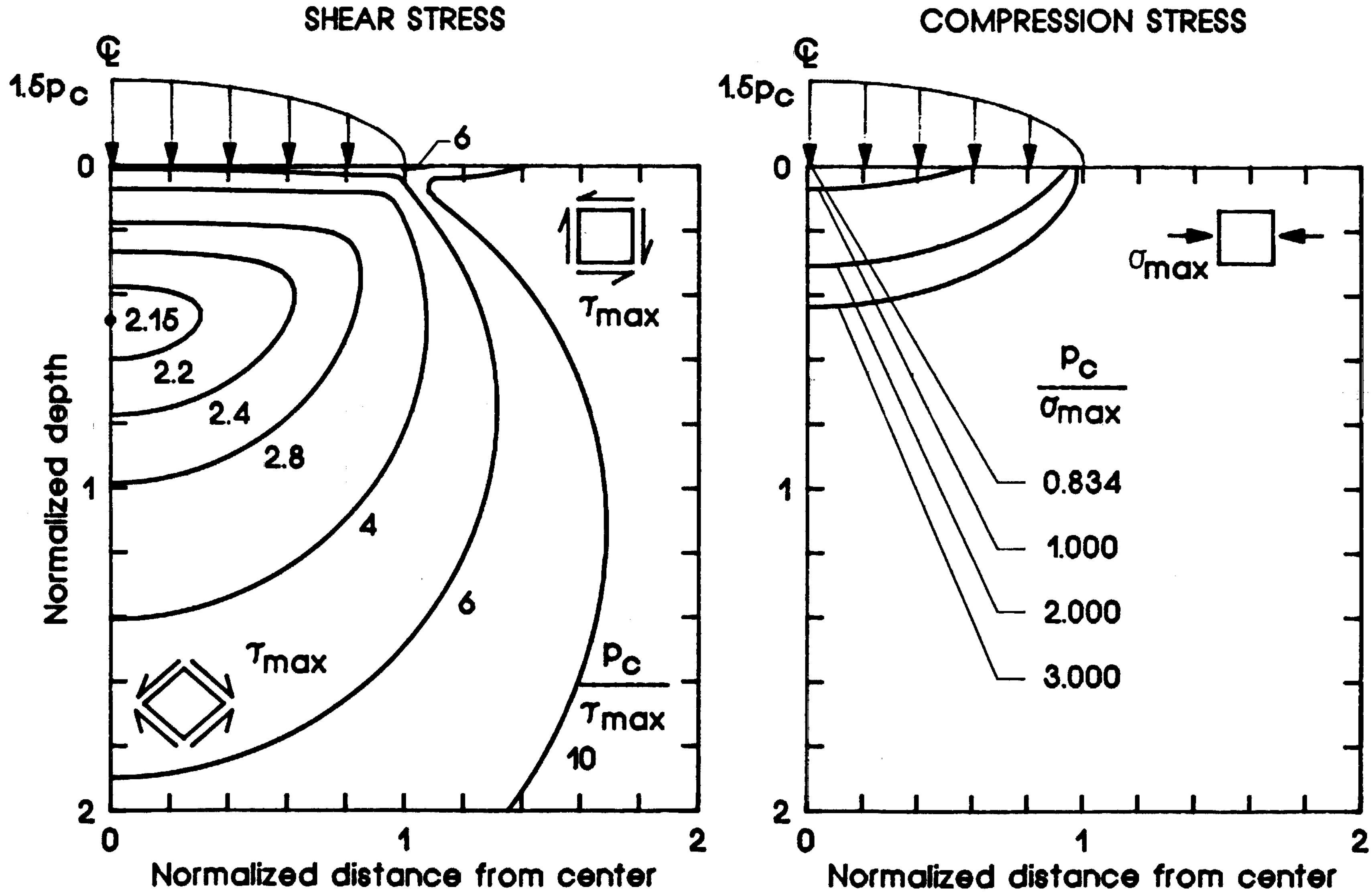
MAXIMUM STRESS CONTOURS - LOVE'S SOLUTION



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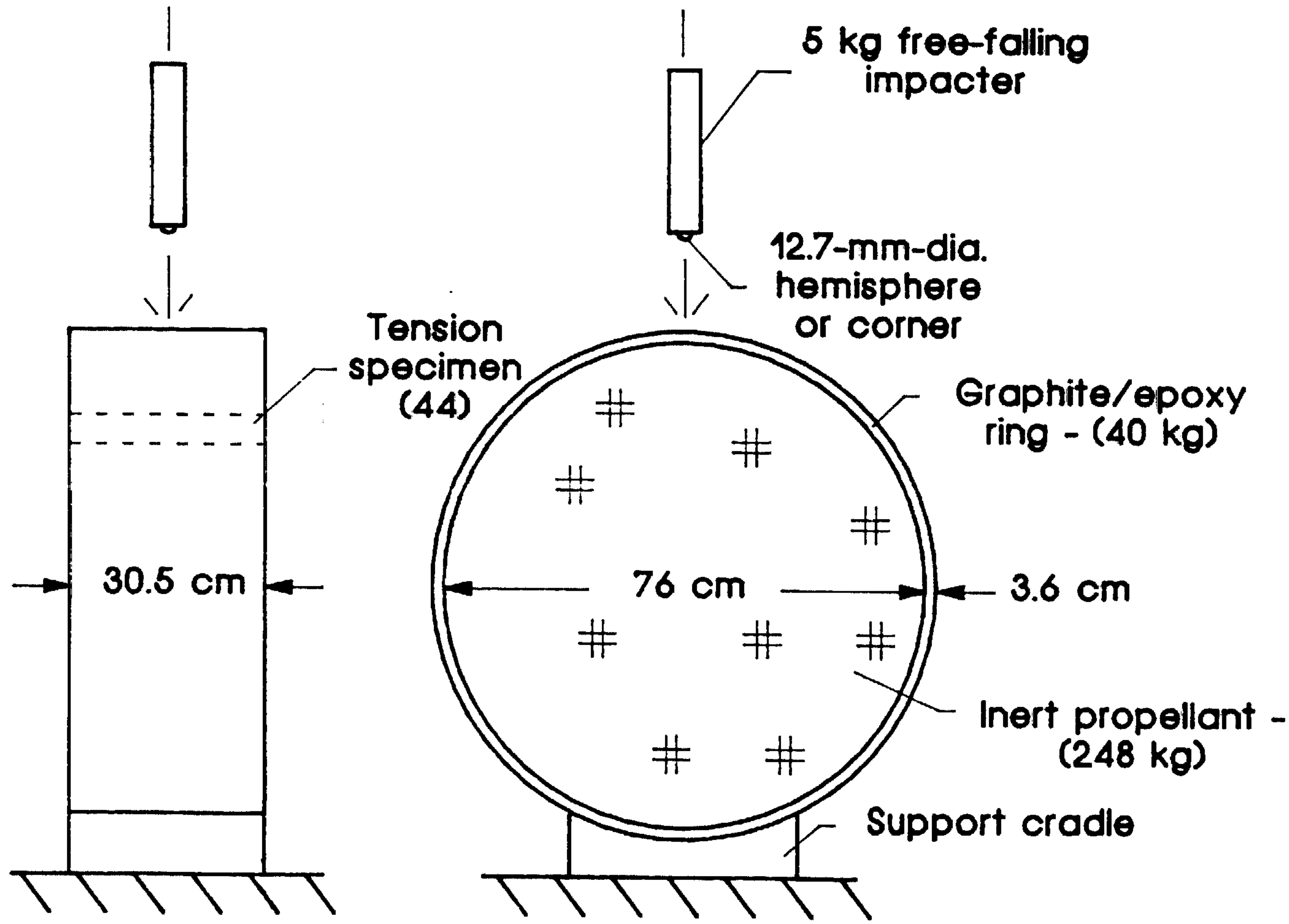
CONCLUSIONS

- FIBERS BROKE BELOW CONTACT SITE AT CRITICAL CONTACT PRESSURE, INDEPENDENT OF INDENTER RADIUS.
- DAMAGE VISIBLE WHEN CRITICAL CONTACT PRESSURE WAS EXCEEDED.
- SIZE OF DAMAGE INCREASED WITH INCREASING CONTACT PRESSURE AND INDENTER RADIUS.
- SIZE OF DAMAGE WAS PREDICTED USING THEORY OF ELASTICIY AND MAXIMUM SHEAR STRESS CRITERIA.

PART II

STRENGTH AFTER LOW-VELOCITY IMPACTS -- ANALYSIS & EXPERIMENTS

IMPACT TESTS



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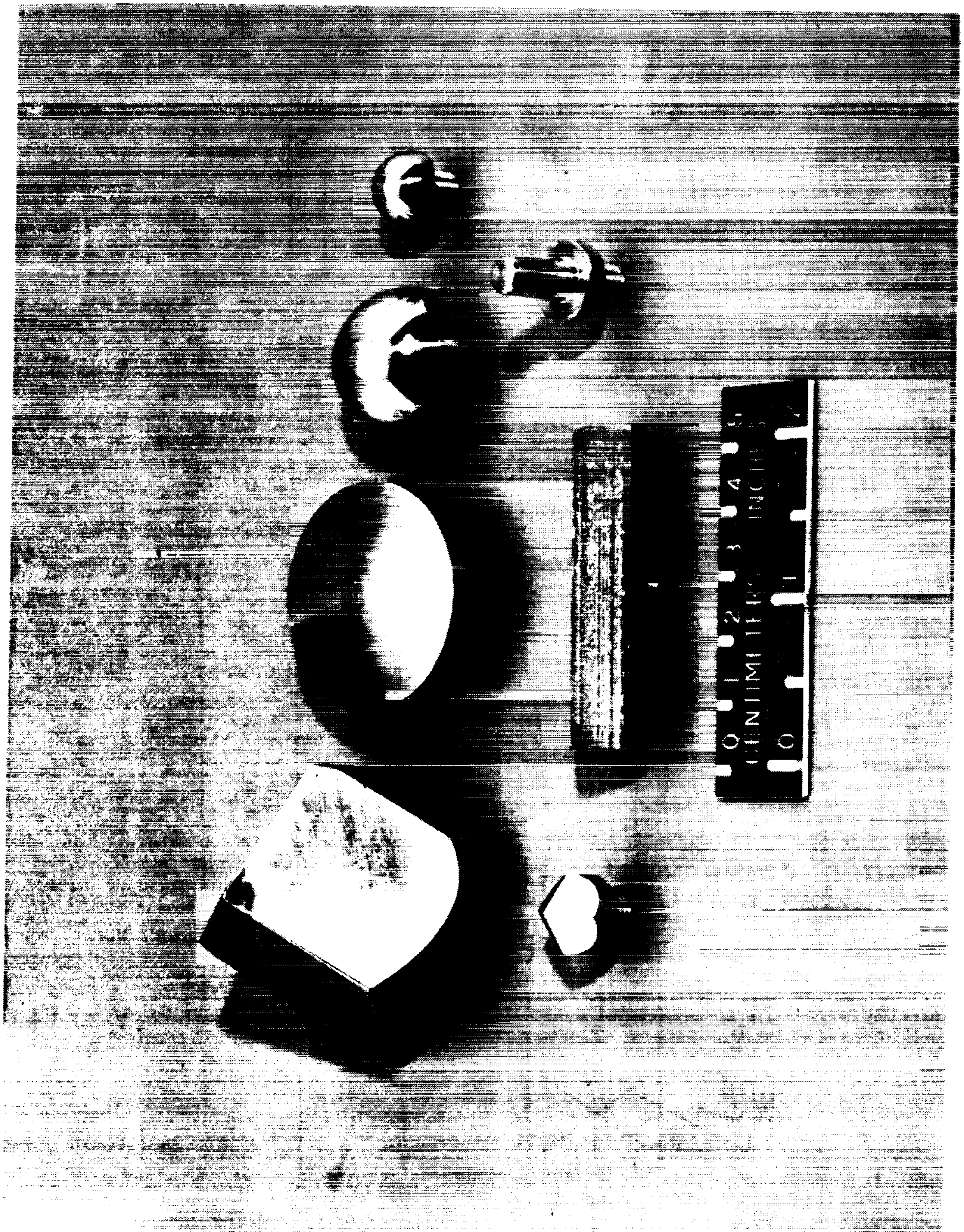
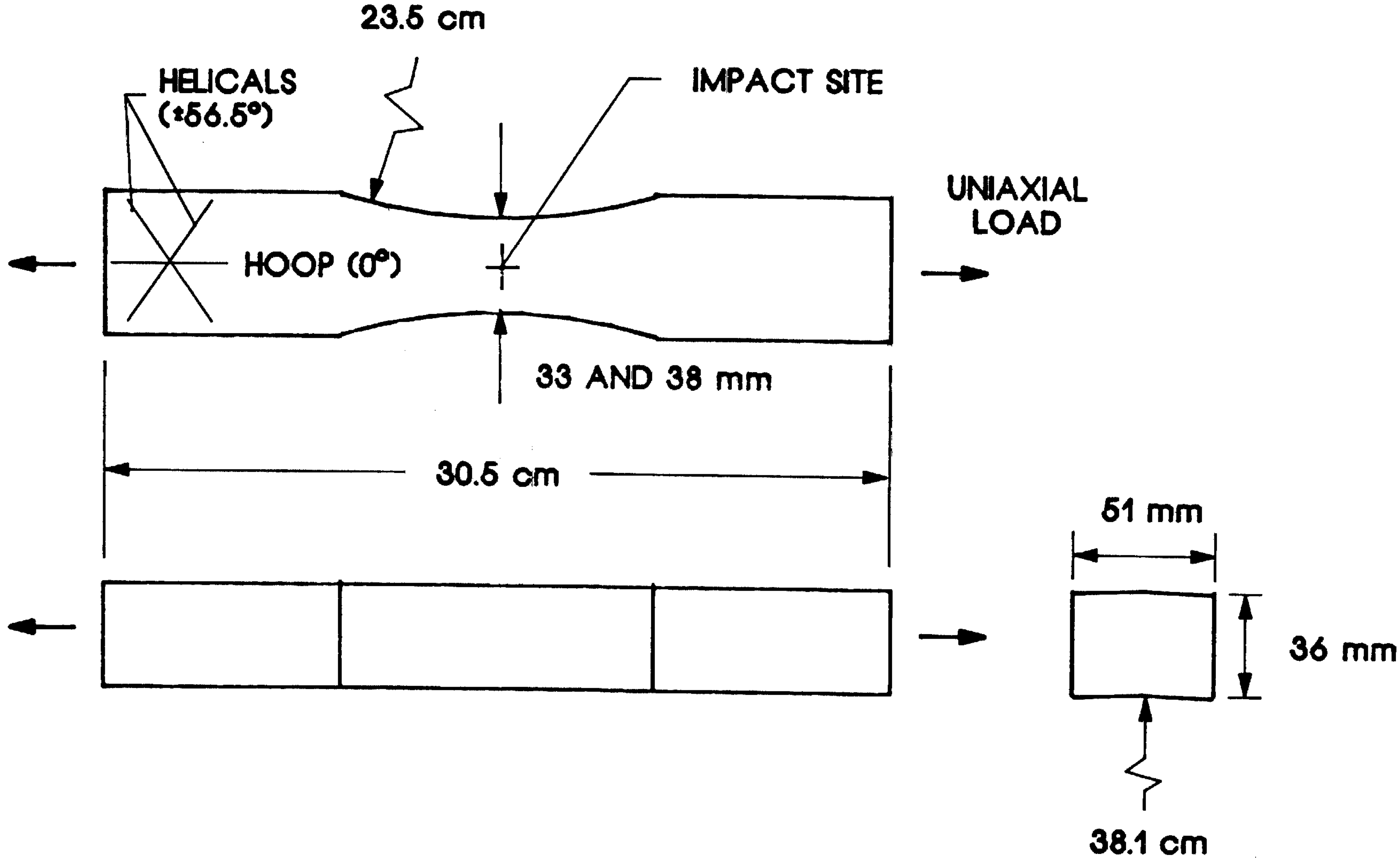


NASA
L-84-5453

EXPERIMENTAL APPROACH -- RESIDUAL STRENGTH TESTS



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ANALYTICAL APPROACH

IMPACT FORCE — ENERGY BALANCE ANALYSIS

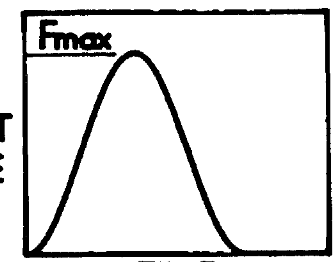
$$0.4\, R_1^{-1/3}\, n_o^{-2/3}\, F_{max}^{5/3} + 0.5\, k_b^{-1}\, F_{max}^2 - KE_{eff} = 0$$

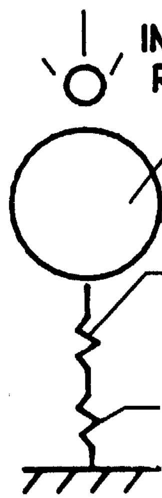
WHERE

$$K_{eff} = Mv_1^2/2$$

AND

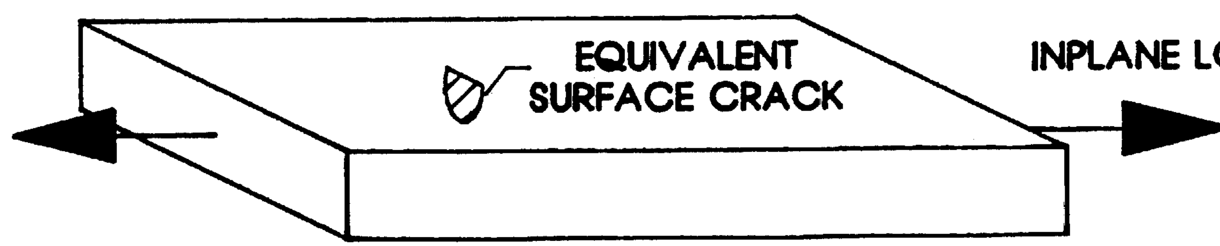
$$M = 1/[1/m_1 + 1/(m_2/4)]$$





STRENGTH — ANALYSIS OF SURFACE CRACKS

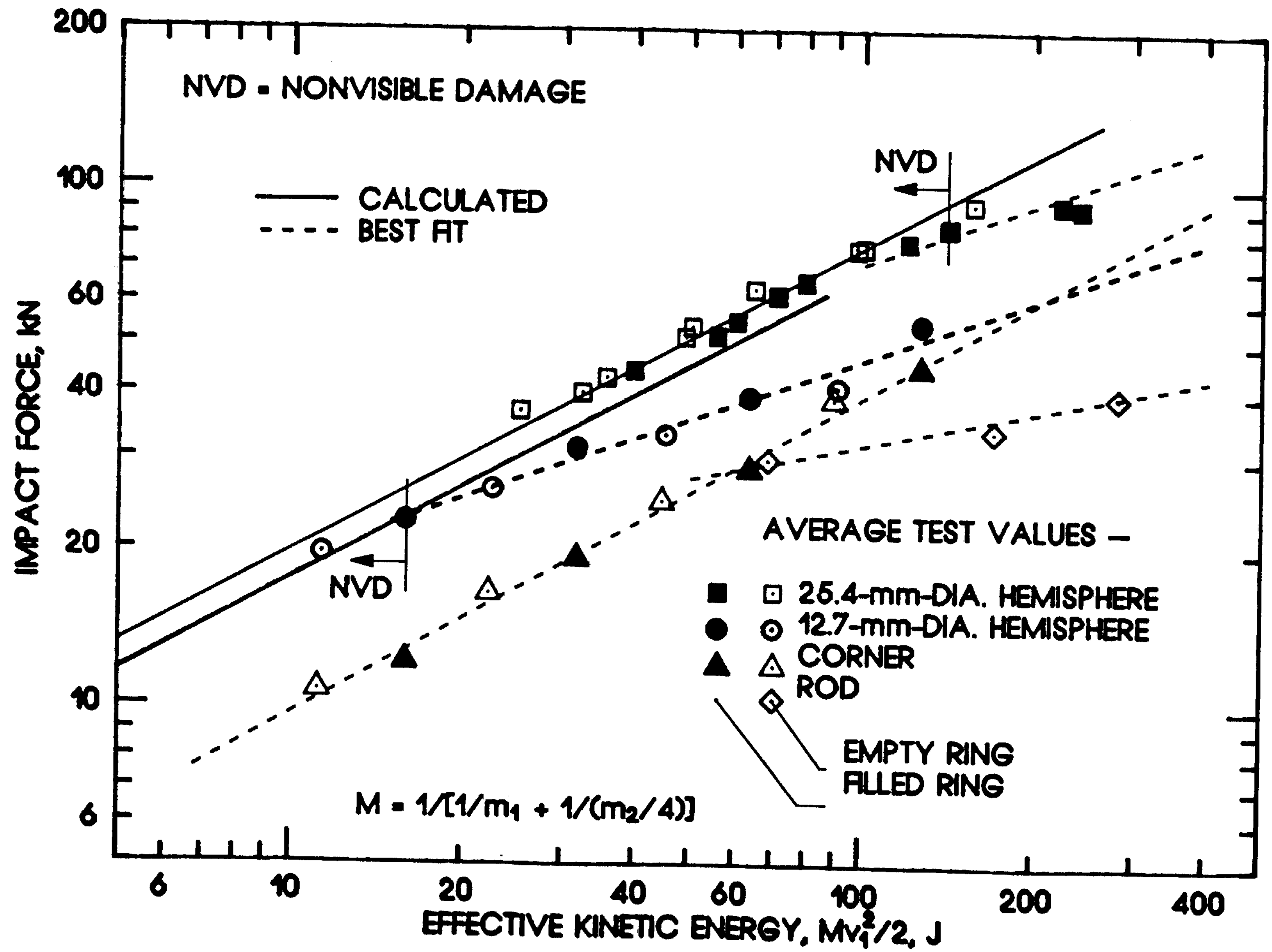
STRESS INTENSITY FACTOR FOR ISOTROPIC PLATE



C-5

\WORKSHOP\SLIDES\SLIDE5

EFFECT OF IMPACTER SHAPE ON IMPACT FORCE



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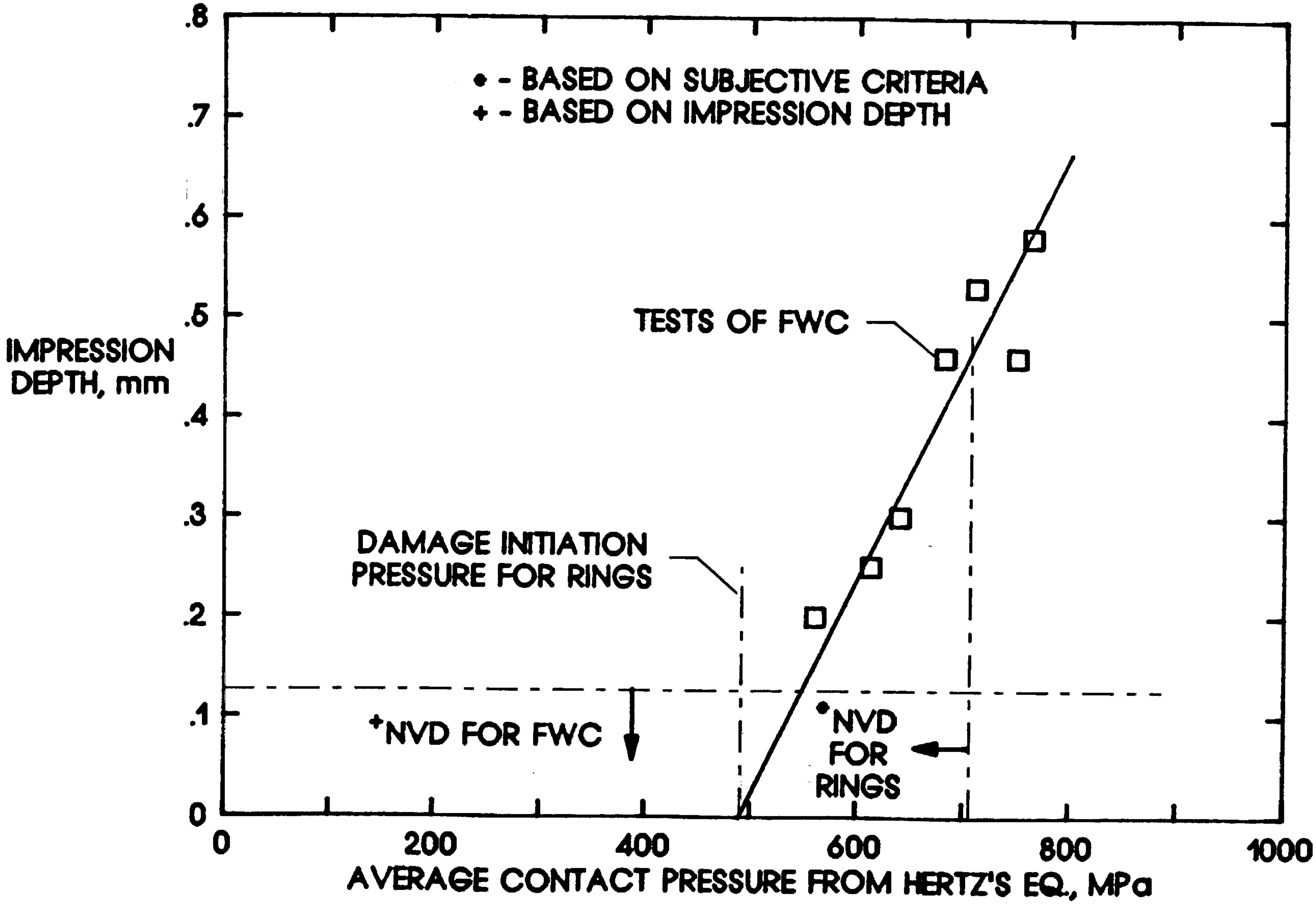
COMPARISON OF DETECTABLE IMPACT DAMAGE

25.4-mm-DIA. IMPACTER



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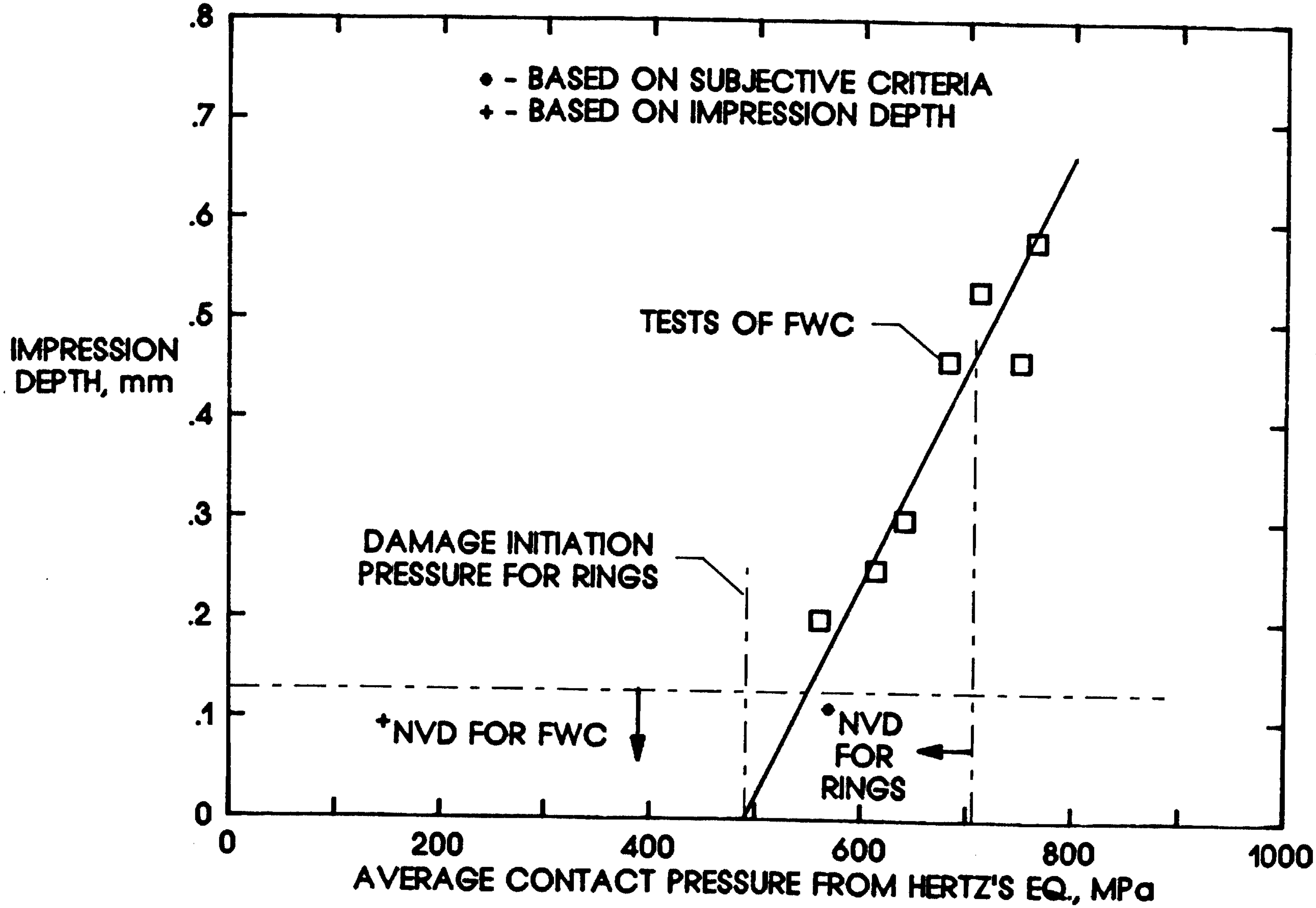
IMPRESSION DEPTHS FOR FWC

25.4-mm-DIA. IMPACTER



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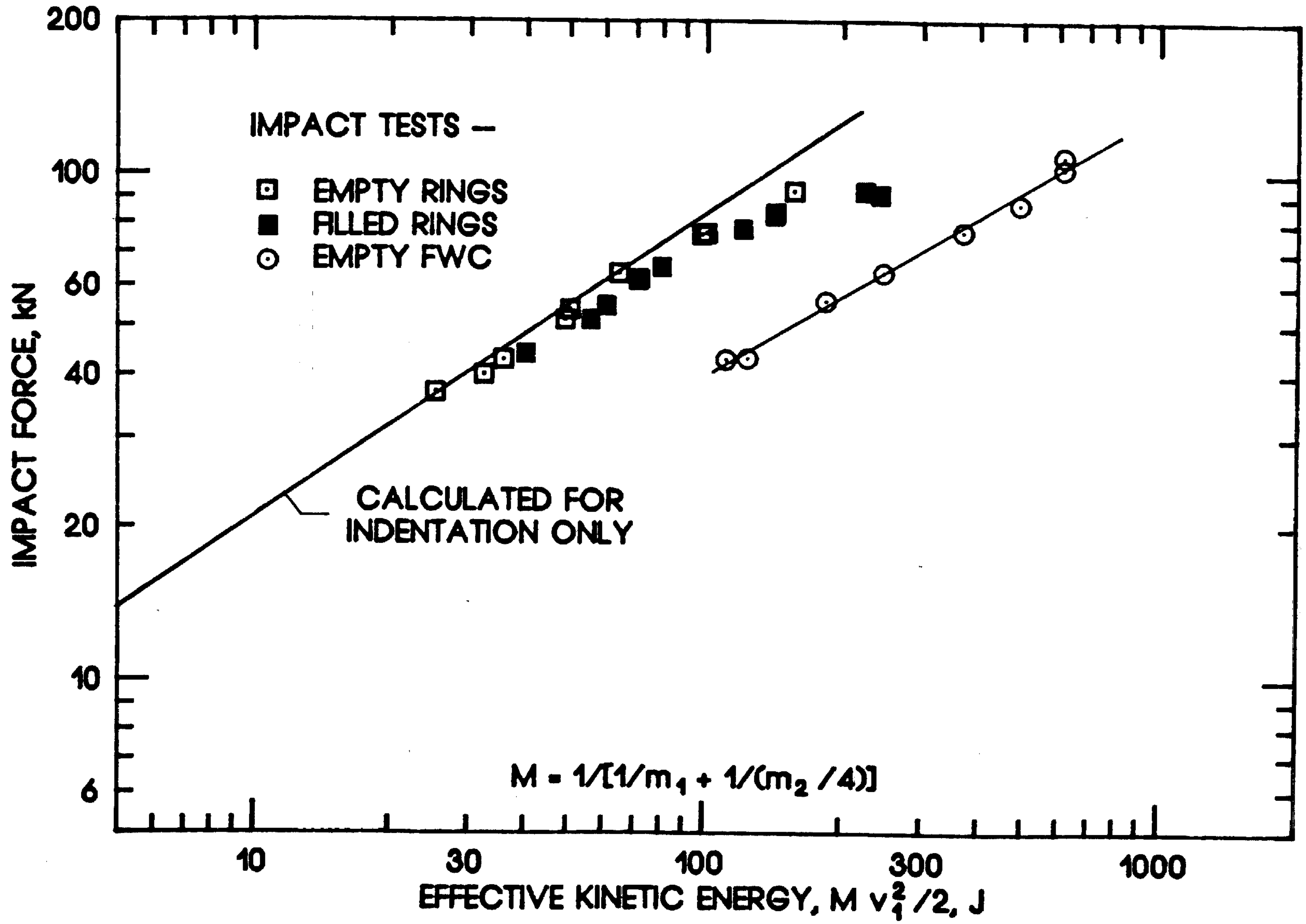
IMPACT FORCES FOR RINGS AND FWC

25.4-mm-DIA. HEMISPHERE INDENTER



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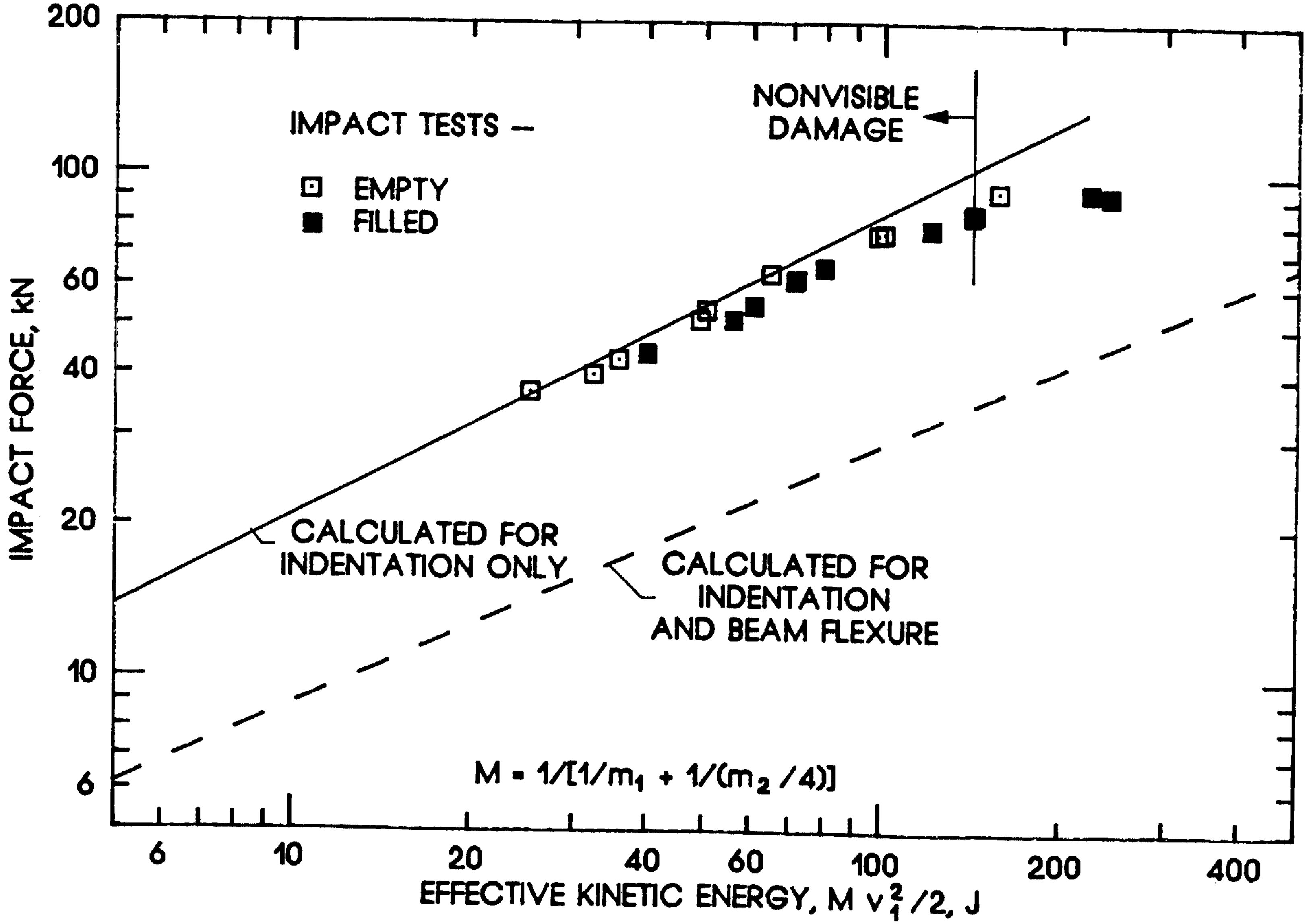
IMPACT FORCE VERSUS EFFECTIVE KINETIC ENERGY

25.4-mm-DIA. HEMISPHERE INDENTER



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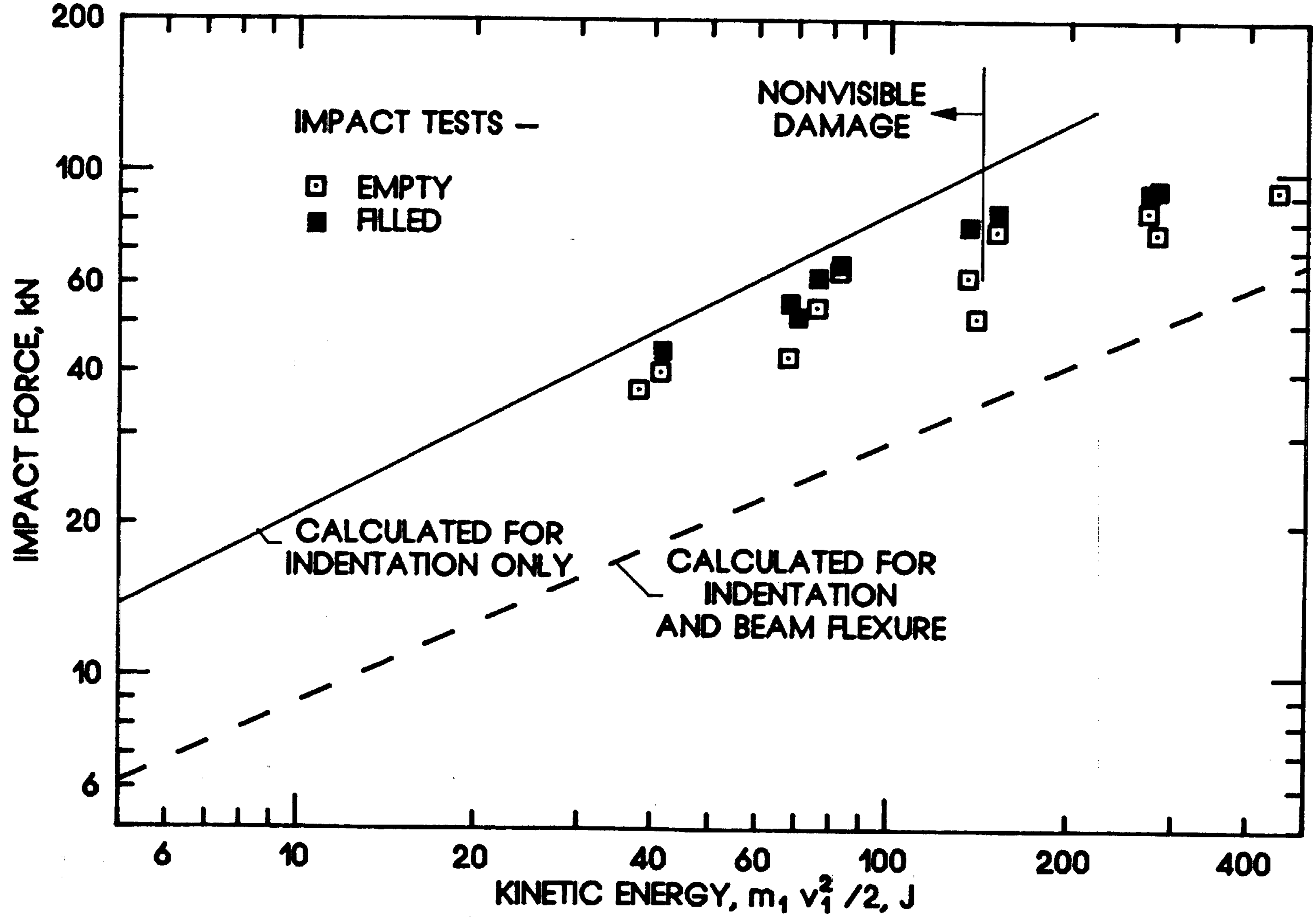
EFFECT OF MASS & VELOCITY ON IMPACT FORCE

25.4-mm-DIA. HEMISPHERE INDENTER



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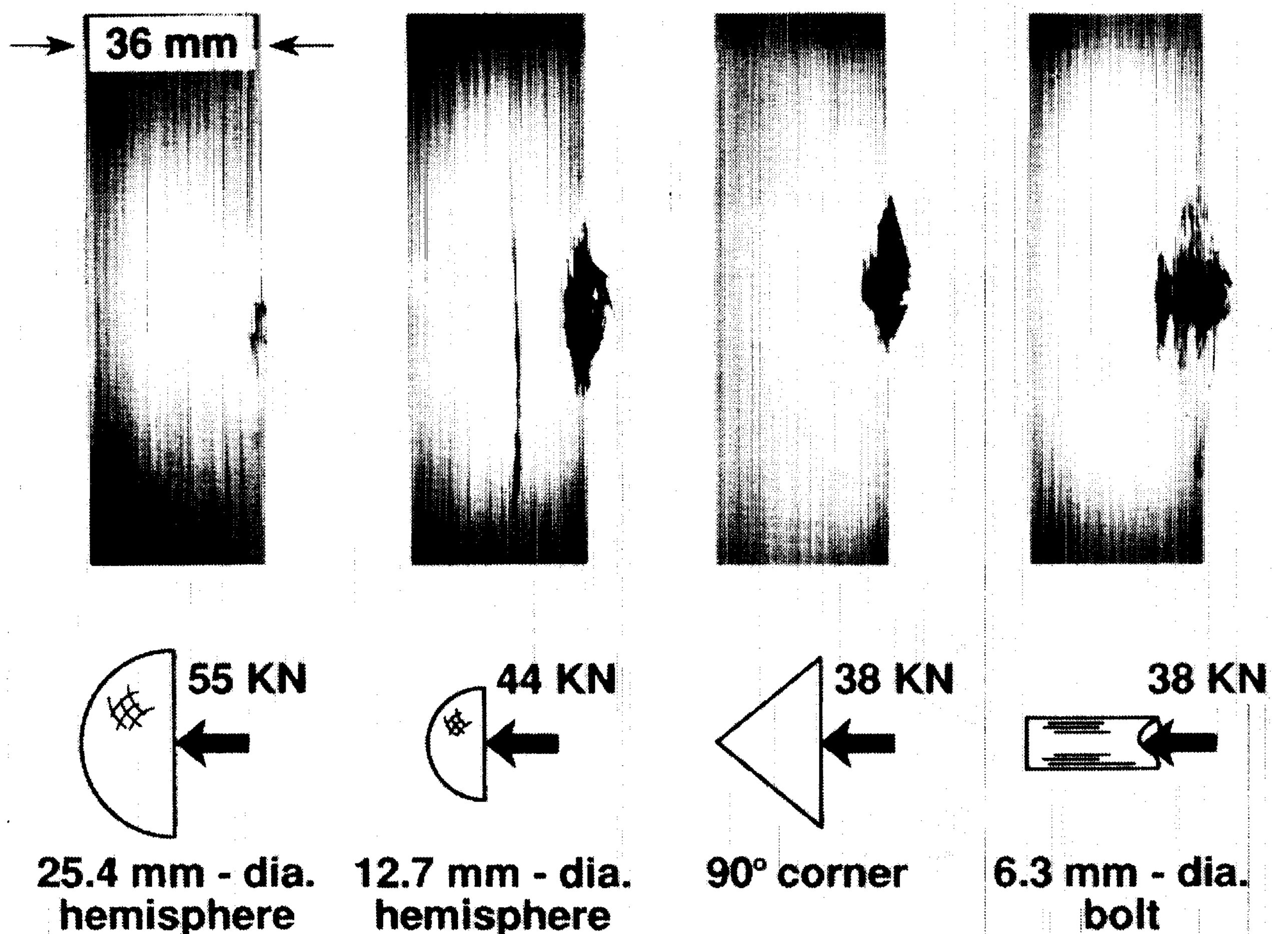
RADIOGRAPHS OF IMPACT DAMAGE FOR VARIOUS INDENTERS



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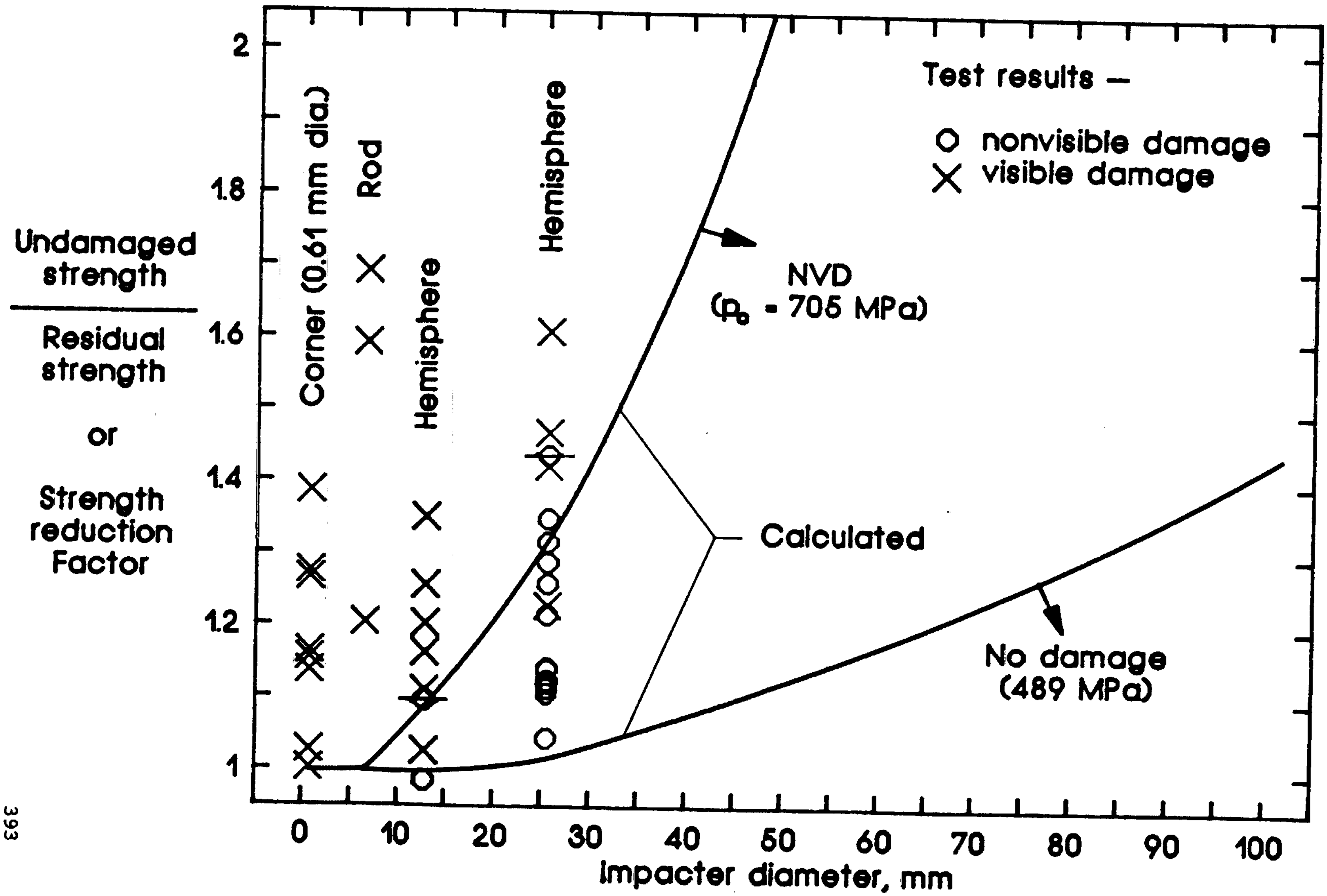
STRENGTH REDUCTION FACTOR VERSUS IMPACTER DIAMETER



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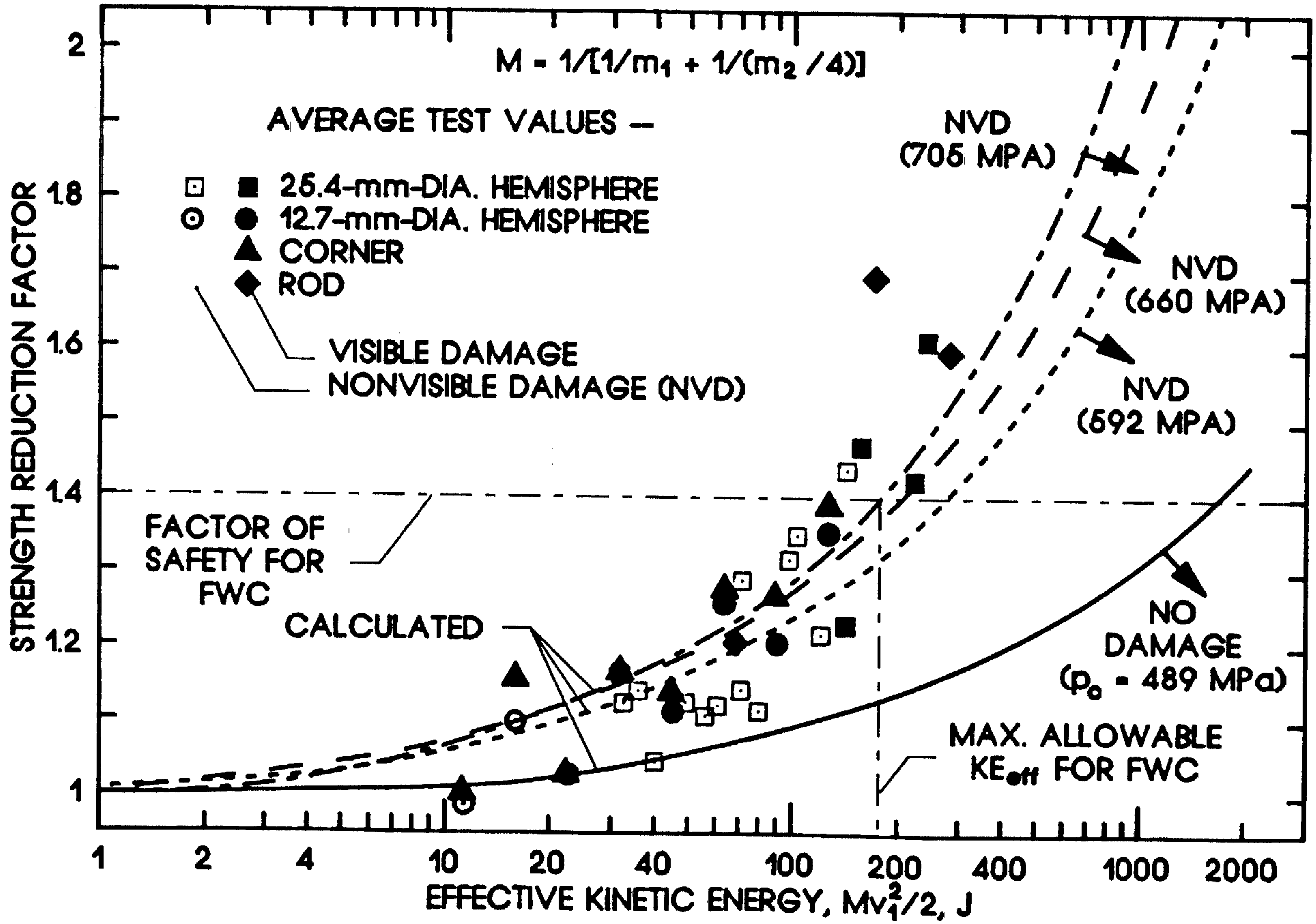
STRENGTH REDUCTION FACTOR FOR NONVISIBLE DAMAGE CRITERIA



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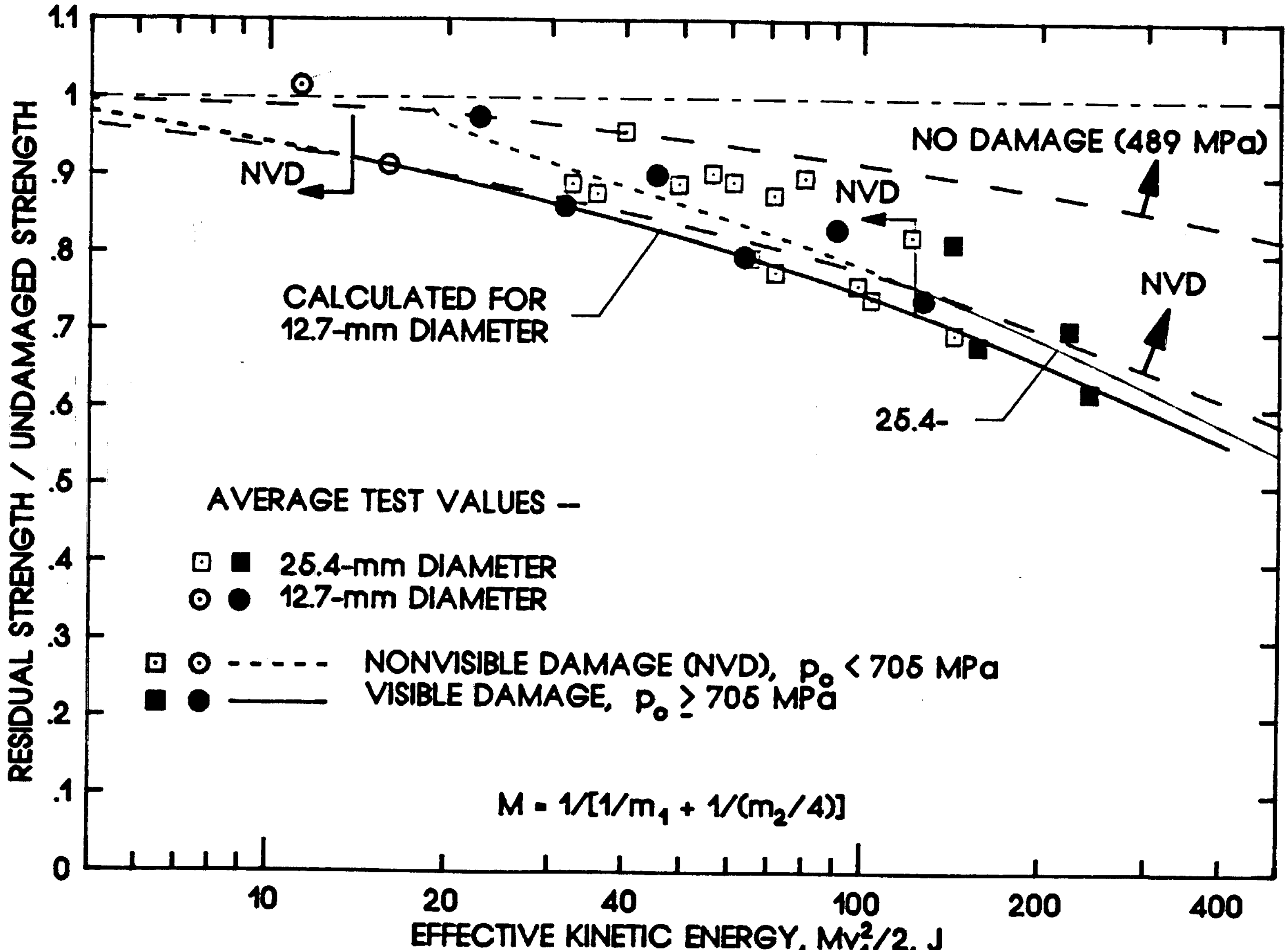
TENSION STRENGTH FOR HEMISPHERE INDENTERS



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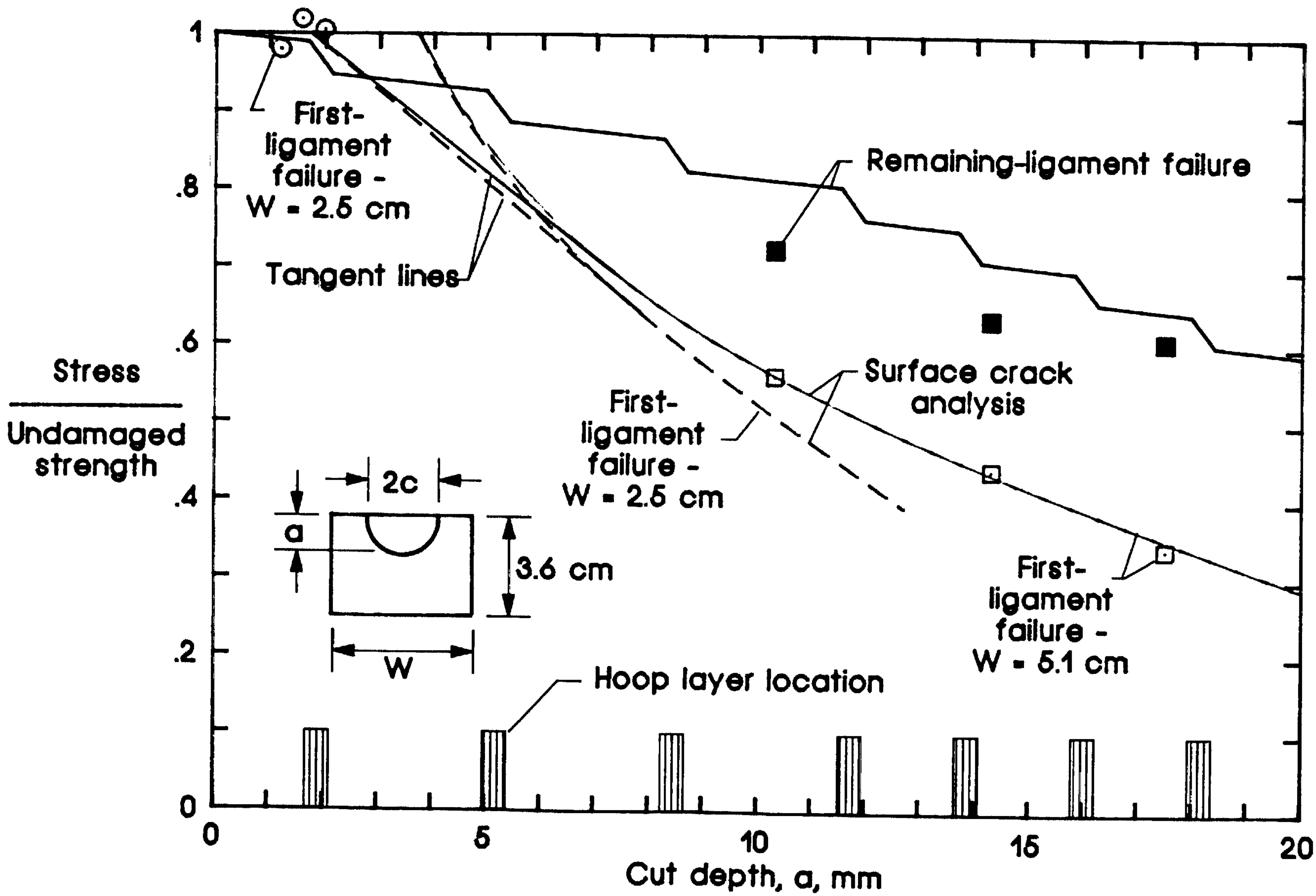
STRENGTHS WITH SURFACE CUTS

$c/a = 1.0$



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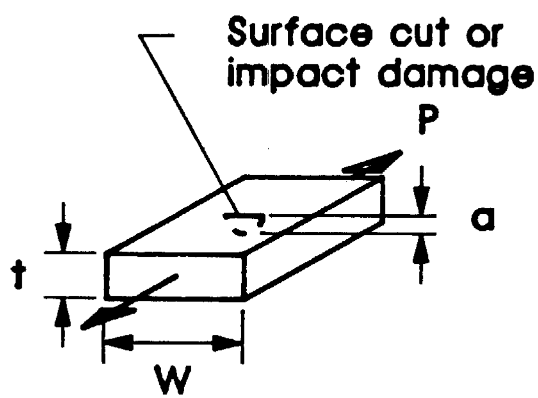
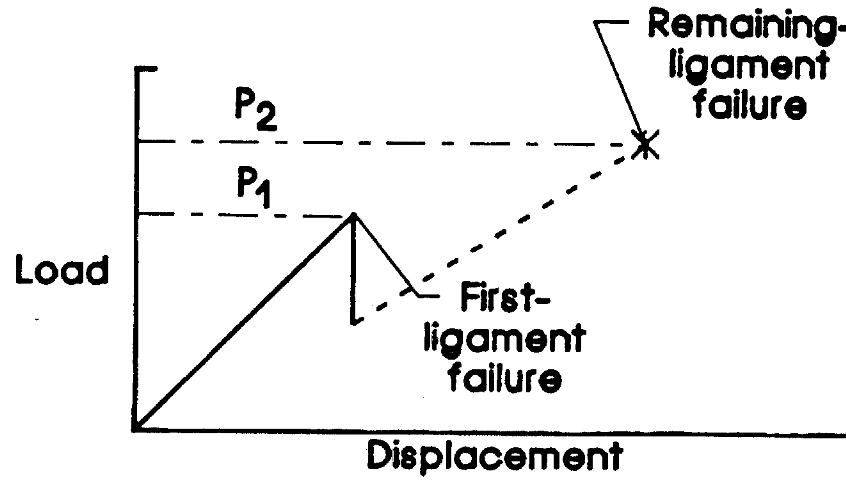
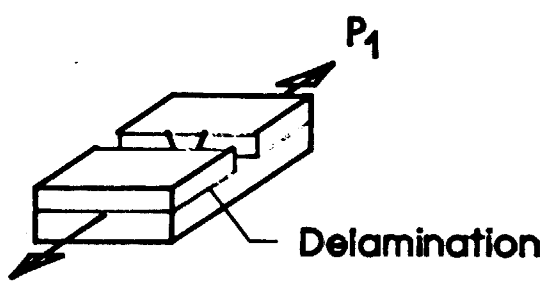
TWO-PART FAILURE WITH SURFACE DAMAGE

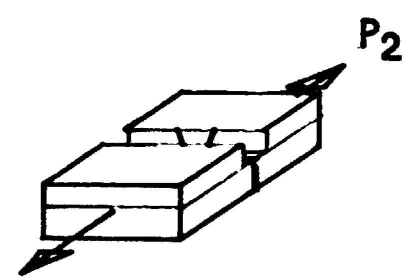


BEFORE FAILURE





FIRST-LIGAMENT FAILURE



REMAINING-LIGAMENT FAILURE

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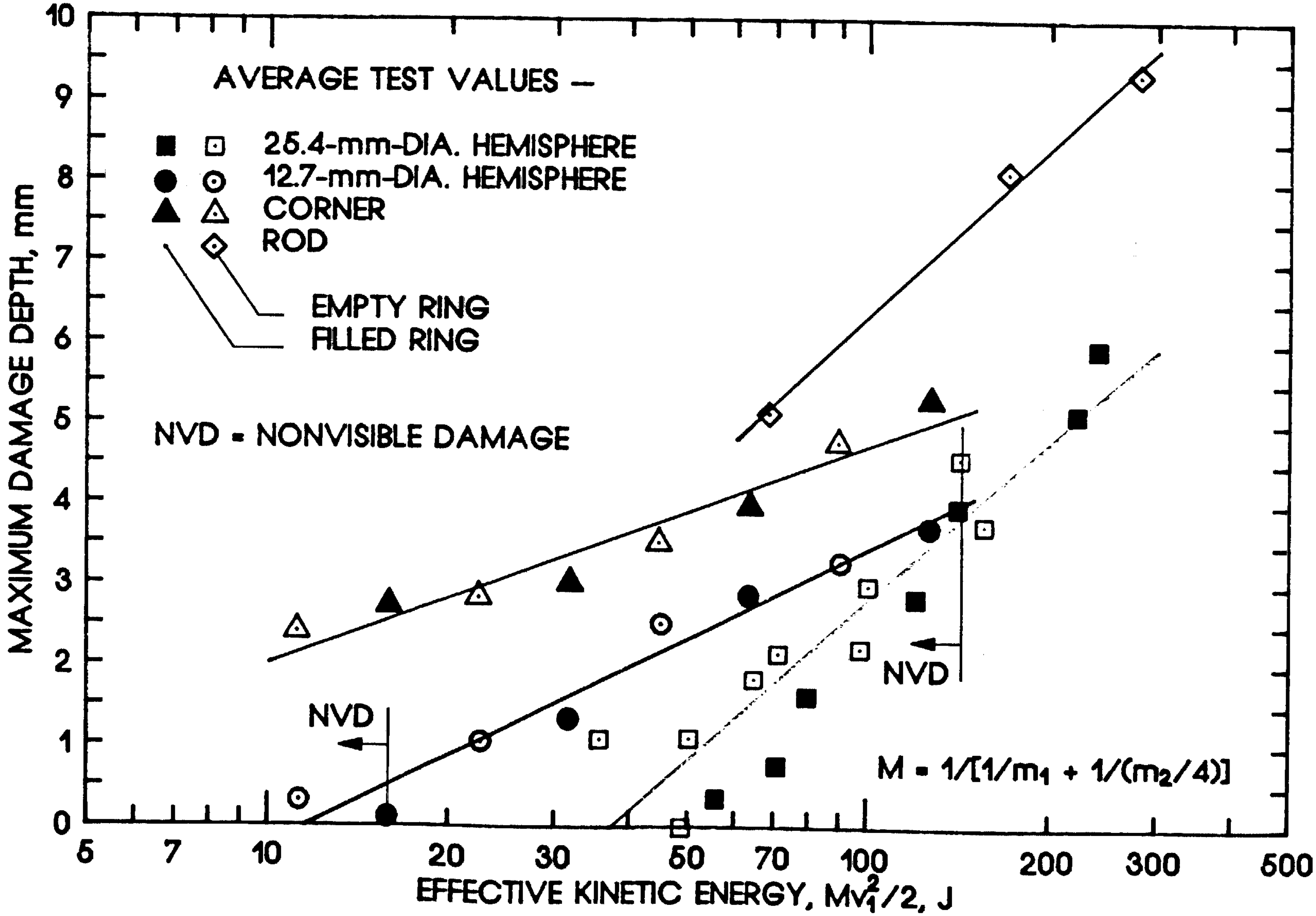
EFFECT OF IMPACTER SHAPE ON DAMAGE DEPTH IN RADIOGRAPHS



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CONCLUSIONS

- IMPACT FORCE AND IMPACT PARAMETERS WERE CORRELATED QUITE WELL WITH AN ENERGY BALANCE ANALYSIS EXCEPT WHEN DAMAGE WAS EXTENSIVE.

- SURFACE CRACK ANALYSIS GAVE GOOD PREDICTIONS OF TENSION STRENGTHS.

- FOR BARELY VISIBLE DAMAGE --

  1. STRENGTHS WERE LOWER FOR BLUNT IMPACTERS THAN FOR SHARP IMPACTERS
  2. FACTOR OF SAFETY WAS INDEPENDENT OF IMPACTER SHAPE.

CERTIFICATION OF COMPOSITES FOR AIRCRAFT

John Lincoln

Aeronautical Systems Division,

Wright-Patterson Air Force Base

CERTIFICATION OF COMPOSITES FOR AIRCRAFT

BASIS FOR USAF AIRCRAFT

- MIL-STD-1530A THE USAF AIRCRAFT STRUCTURAL INTEGRITY PROGRAM (ASIP)
  - PROGRAM FOR FULL SCALE DEVELOPMENT OF METAL AND COMPOSITE STRUCTURE

- COMPOSITE STRUCTURES CAUSE SOME SHIFTING OF EMPHASIS IN THE

TASKS OF ASIP →

I DESIGN INFORMATION

II DESIGN ANALYSES AND DEVELOPMENT TESTS

III FULL SCALE TESTING

IV FORCE MANAGEMENT DATA PACKAGE

V FORCE MANGEMENT

TASK II

MATERIALS AND JOINT ALLOWABLES

B-BASIS

DERIVED FROM SUITABLE COUPONS WITH ENVIRONMENTAL CONDITIONING-END OF LIFE MOSITURE CONDITION WITH SEVERE BASING SCENARIO

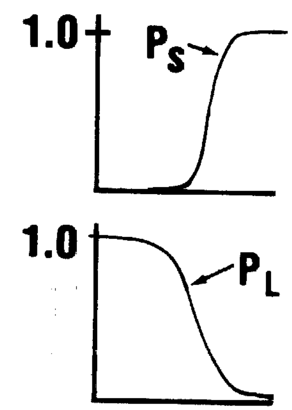
RISK OF USING B-BASIS ALLOWABLE

STRENGTH ⟶ $P_S(x) = 1 - \exp\left[-\left(\frac{X}{\beta_S}\right)^{\alpha_S}\right]$

$\alpha_S = 20$

LOAD ⟶ $P_L(y) = \exp\left[-\left(\frac{Y}{\beta_L}\right)^{\alpha_L}\right]$

$\alpha_L = 6$



PROBABILITY OF FAILURE IN ONE LIFETIME FOR A SINGLE AIRCRAFT $= \iint_R P_S' \, P_L' \, dx\,dy$

R = REGION WHERE LOAD EXCEEDS STRENGTH

PROBABILITY OF FAILURE IN ONE LIFETIME FOR A SINGLE AIRCRAFT $= 1.5 \times 10^{-3}$

TASK II ANALYSES

- STRESS ANALYSIS
- DAMAGE TOLERANCE ANALYSIS
- DURABILITY ANALYSIS
- NUCLEAR WEAPONS EFFECTS ANALYSIS
- NON-NUCLEAR WEAPONS EFFECTS ANALYSIS

TASK II TESTS

DESIGN DEVELOPMENT TESTS

BUILDING BLOCK APPROACH

- COUPONS
- ELEMENTS
- SUBCOMPONENTS
- COMPONENTS

DESIGN DEVELOPMENT TESTS

BATTLE DAMAGE TESTS

- INCLUDES ENVIRONMENTALLY CONDITIONED TEST SPECIMENS
- BUILDING BLOCK APPROACH
- DAMAGE BASED ON SPECIFIED THREAT FOR WEAPON SYSTEM
- DAMAGE STRUCTURE MUST MEET IN-FLIGHT EVIDENT RESIDUAL STRENGTH REQUIREMENT
- REPAIRABILITY REQUIREMENTS FOR EACH WEAPON SYSTEM

DESIGN DEVELOPMENT TESTS

STRENGTH TESTS

- INCLUDES ENVIRONMENTALLY CONDITIONED TEST SPECIMENS
- BUILDING BLOCK APPROACH
- B-BASIS KNOCKDOWN FOR SUBCOMPONENT AND COMPONENT TESTS
- STRAIN GAGED FOR FULL SCALE TEST CORRELATION
- TESTS TO INTERROGATE FOR ENVIRONMENTALLY INDUCED FAILURE MODES

DESIGN DEVELOPMENT TESTS

DAMAGE TOLERANCE TESTS

- INCLUDES ENVIRONMENTALLY CONDITIONED TEST SPECIMENS
- BUILDING BLOCK APPROACH
- DAMAGE
  - HIGH ENERGY IMPACT
  - SCRATCHES
  - DELAMINATIONS
- UPPER BOUND SPECTRUM
- MINIMUM-TWO LIFETIME TEST AND MEET RESIDUAL STRENGTH REQUIREMENTS
- SENSITIVITY TESTS

INITIAL FLAW/DAMAGE ASSUMPTIONS

FLAW/DAMAGE TYPE	FLAW/DAMAGE SIZE
SCRATCHES	SURFACE SCRATCH 4.0 INCHES IN LENGTH AND 0.02 INCH DEEP
DELAMINATION	INTERPLY DELAMINATION EQUIVALENT TO A 2.0-INCH DIAMETER CIRCLE WITH DIMEMSIONS MOST CRITICAL TO ITS LOCATION
IMPACT DAMAGE	DAMAGE FROM A 1.0-INCH DIAMETER HEMISPHERICAL IMPACTOR WITH 100 FT-LB OF KINETIC ENERGY OR WITH THAT KINETIC ENERGY REQUIRED TO CAUSE A DENT 0.10 INCH DEEP WHICHEVER IS LEAST.

RESIDUAL STRENGTH REQUIREMENTS

PXX*	DEGREE OF INSPECTABILITY	TYPICAL INSPECTION INTERVAL	MAGNIFICATION FACTOR
P_{FE}	IN-FLIGHT EVIDENT	ONE FLIGHT**	100
P_{GE}	GROUND EVIDENT	ONE DAY (TWO FLIGHTS)**	100
P_{WV}	WALK-AROUND VISUAL	TEN FLIGHTS**	100
P_{SV}	SPECIAL VISUAL	ONE YEAR	50
P_{DM}	DEPOT OR BASE LEVEL	1/4 LIFETIME	20
P_{LT}	NON-INSPECTABLE	ONE LIFETIME	20

*P_{XX} = MAXIMUM AVERAGE INTERNAL MEMBER LOAD THAT WILL OCCUR ONCE IN M TIMES THE INSPECTION INTERVAL. WHEN P_{DM} OR P_{LT} IS DETERMINED TO BE LESS THAN THE DESIGN LIMIT LOAD, THE DESIGN LIMIT LOAD SHOULD BE THE REQUIRED RESIDUAL STRENGTH LOAD LEVEL. P_{XX} NEED NOT BE GREATER THAN 1.2 TIMES THE MAXIMUM LOAD IN ONE LIFETIME, IF P_{XX} IS GREATER THAN DESIGN LIMIT LOAD.

**MOST DAMAGING DESIGN MISSION

DESIGN DEVELOPMENT TESTS

DURABILITY TESTS

- INCLUDES ENVIRONMENTALLY CONDITIONED TEST SPECIMENS
- BUILDING BLOCK APPROACH
- DAMAGE
  - LOW ENERGY IMPACT (TOOL DROP, HAIL, RUNWAY DEBRIS)
- UPPER BOUND SPECTRUM
- MINIMUM-TWO LIFETIME TEST AND MEET RESIDUAL IMPAIRMENT REQUIREMENTS
- SENSITIVITY TESTS

LOW ENERGY IMPACT
(TOOL DROP)

ZONE	DAMAGE SOURCE	DAMAGE LEVEL	REQUIREMENTS
1 HIGH PROBABILITY OF IMPACT	• 0.5" DIAMETER SOLID IMPACTOR • LOW VELOCITY • NORMAL TO SURFACE	VISIBLE (0.1" DEEP DENT) 6 FT-LB MAX	• NO FUNCTIONAL IMPAIRMENT OR OR STRUCTURAL REPAIR REQUIRED FOR TWO DESIGN LIFETIMES AND AND NO WATER INTRUSION • NO VISIBLE DAMAGE FROM A SINGLE 4 FT-LB IMPACT
2 LOW PROBABILITY OF IMPACT	SAME AS ZONE 1	SAME AS ZONE 1	• NO FUNCTIONAL IMPAIRMENT AFTER TWO DESIGN LIFETIMES AND NO WATER INTRUSION - AFTER FIELD REPAIR IF DAMAGE IS VISIBLE

LOW ENERGY IMPACT
(HAIL AND RUNWAY DEBRIS)

ZONE	DAMAGE SOURCE	DENSITY	REQUIREMENTS
ALL VERTICAL AND UPWARD FACING HORIZONTAL SURFACES	HAIL • 0.8" DIAMETER • SP. GR = 0.9 • 90 FT/SEC • NORMAL TO HORIZONTAL SURFACES • 45° ANGLE TO VERTICAL SURFACES	UNIFORM DENSITY 0.8" ON CENTER	• NO FUNCTIONAL IMPAIRMENT OR STRUCTURAL REPAIR REQUIRED FOR TWO DESIGN LIFETIMES AND NO WATER INTRUSION • NO VISIBLE DAMAGE
STRUCTURE IN PATH OF DEBRIS	RUNWAY DEBRIS • 0.5" DIAMETER OBJECT • SP. GR = 3.0 • VELOCITY APPROPRIATE TO WEAPON SYSTEM	—	• NO FUNCTIONAL IMPAIRMENT FOR TWO DESIGN LIFETIMES AND NO WATER INTRUSION - AFTER FIELD REPAIR IF DAMAGE IS VISIBLE

TASK III

FULL SCALE TESTING

- **STATIC TESTS** - REQUIRED FOR COMPOSITE STRUCTURE
- **DURABILITY TESTS** - NOT REQUIRED FOR COMPOSITE STRUCTURE IF DESIGN DEVELOPMENT TESTS MEET OBJECTIVES
- **DAMAGE TOLERANCE TESTS** - NOT REQUIRED FOR COMPOSITE STRUCTURE IF DESIGN DEVELOPMENT TESTS MEET OBJECTIVES

FULL SCALE TESTING

STATIC TESTS

- CRITICAL ENVIRONMENTALLY INDUCED FAILURE MODE

NO	YES
• TEST AT ROOM TEMPERATURE WITHOUT SPECIAL MOISTURE CONDITIONING	• TEST WITH ENVIRONMENTALLY CONDITIONED ARTICLE
• TEST TO 150% DESIGN LIMIT LOAD CONDITION	• TEST TO 150% DESIGN LIMIT LOAD CONDITION
• STRAIN GAGING TO CORRELATE WITH DESIGN DEVELOPMENT TESTS	• STRAIN GAGING TO CORRELATE WITH DESIGN DEVELOPMENT TESTS

TASK IV

**FORCE MANAGEMENT
DATA PACKAGE**

- LOADS/ENVIRONMENTAL SPECTRUM SURVEY

AND

- INDIVIDUAL AIRCRAFT TRACKING PROGRAM
  - MUST BE CAPABLE OF RECORDING HIGH LOAD EVENTS THAT ARE POTENTIALLY DAMAGING TO COMPOSITE STRUCTURES

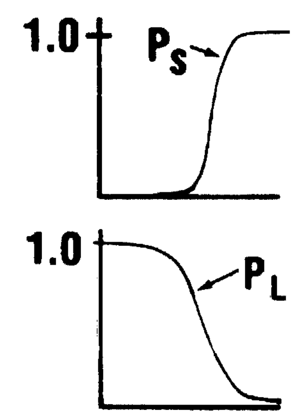
RISK OF USING B-BASIS ALLOWABLE

STRENGTH ⟶ $P_s(x) = 1 - \exp\left[-\left(\frac{X}{\beta_s}\right)^{\alpha_s}\right]$

$\alpha_s = 20$

LOAD ⟶ $P_L(y) = \exp\left[-\left(\frac{Y}{\beta_L}\right)^{\alpha_L}\right]$

$\alpha_L = 6$



PROBABILITY OF FAILURE IN ONE LIFETIME FOR A SINGLE AIRCRAFT $= \iint_R P_s' \, P_L' \, dxdy$

R = REGION WHERE LOAD EXCEEDS STRENGTH

PROBABILITY OF FAILURE IN ONE LIFETIME FOR A SINGLE AIRCRAFT $= 1.5 \times 10^{-3}$

86-(MCS)#80/11A-51

CONCLUSIONS

- LABORATORY PROGRAMS HAVE PROVIDED THE CONFIDENCE TO USE COMPOSITES IN FLIGHT CRITICAL STRUCTURE OF NEW WEAPON SYSTEMS
- THE AIRCRAFT STRUCTURAL INTEGRITY PROGRAM CURRENTLY CONTAINS THE NECESSARY ELEMENTS FOR FULL SCALE DEVELOPMENT OF COMPOSITE STRUCTURES

Damage Tolerance of Composites Criteria and Evaluation

NASA Workshop on Impact Damage to Composites
NASA Langley Research Center
Hampton, Virginia

March 19-20, 1991

Ray Horton
Boeing Commercial Airplane Group
Seattle, Washington

A1891.01 s

Presentation Areas

- Air Force criteria development
- Boeing Commercial Airplane criteria
- Air Force contract test approach

A1891.02 s

Air Force Damage Tolerance Criteria Development

Approach

- Basic objectives
- Flaw selection
- Loading requirement
- Industry and Air Force
  - Participation
  - Concurrence

A1891.03 s

Basic Objectives

- Safety enhancement
- Maximum retention of elements of metals requirements
- Realistic compliance requirements
- Minimum impact on design / weight / cost

Composite Versus Metals Properties

Condition		Composite behavior relative to metals
Load-strain relationship		More linear strain to failure
Notch sensitivity	Static	Greater sensitivity
	Fatigue	Less sensitivity
Transverse properties		Weaker
Mechanical properties variability		Slightly higher
Sensitivity to aircraft hygrothermal environment		Greater
Damage growth mechanism		In-plane delamination instead of through-thickness cracks

Manufacturing Flaw / Damage Sources

- Materials
  - Material specification controlled—all
- Layup / processing
  - Process specification controlled—most
- Assembly
  - Assembly specification controlled—some

In-Service
Flaw / Damage Sources

- Servicing — fueling, inspection and maintenance access, handling / moving
  - Access damage — access attachment — fastener holes — exposed edges gouges, scratches
  - Impact damage — service carts, work stands, toolbox, tools

- Flight / taxi
  - Flight: hail, bird and detached fairing— impact / penetration damage
  - Taxi: runway debris, blown tires— impact / penetration damage

- Repair — processing and assembly
  - Similar to manufacturing consideration

Concerns and Considerations

- Damage not included
  - Battle damage
  - Engine disintegration damage
  - Lightning damage
  - Bird impact damage
  - Hail damage
- Flaws not included
  - Porosity
  - Fastener hole cracks

Lightning Damage
Should Not Be Considered for Damage Tolerance Requirements

- Rationale
  - Lightning protection needed for other reasons (i.e., fuel ignition etc.)
  - Lightning strike damage will be evaluated in protection system development; protection system should preclude damage that is not visual or significant when not a known event
  - Strike assumptions are conservative, as well as testing techniques
  - Inspection of damage will require assessment for both protection and structural. Significant damage visually detectable
  - Strike usually a known event
  - Damage definition difficult
  - Impact and penetration should blanket this case

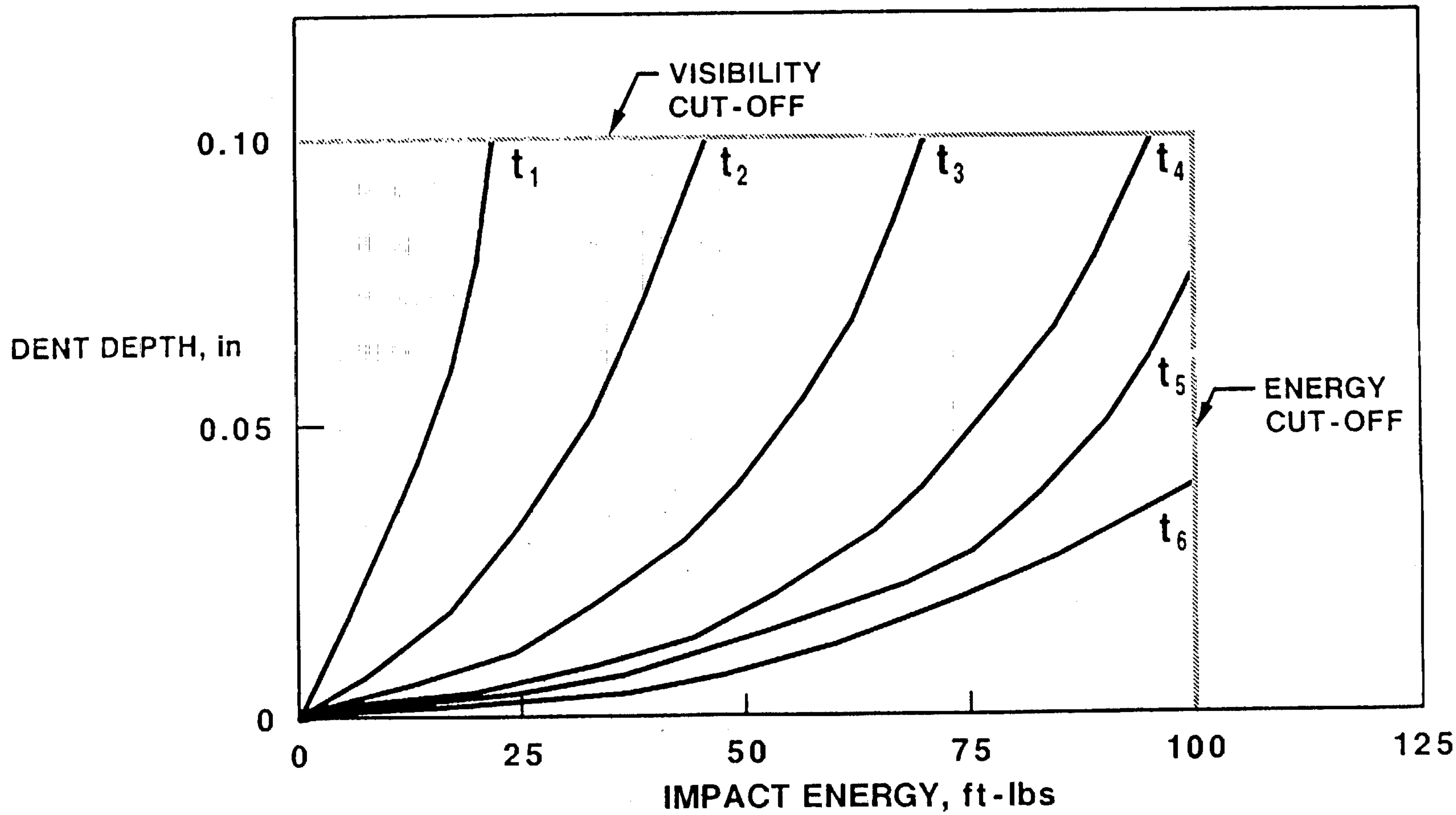
Initial Flaw / Damage Assumptions

Flaw / Damage Type	Flaw / Damage Size (1)
Scratches	Assume the presence of a surface scratch that is 4.0 in long and 0.02 in deep
Delamination	Assume the presence of an interply delamination that has an area equivalent to a 2.0-in diameter circle with dimensions most critical to its location (2)
Impact Damage	Assume the presence of damage caused by the impact of a 1.0-in diameter hemispherical impactor with 100 ft-lb of kinetic energy or with that kinetic energy required to cause a dent 0.10 in deep, whichever is least

(1) For limited access areas such as the interior of the wing, the contractor shall have the option of proposing an inspection procedure before closeout which will allow the assumed damage area size to be reduced

(2) This requirement also accounts for delamination that might occur and be nondetected as a result of in-service repair

Impact Damage Assumptions



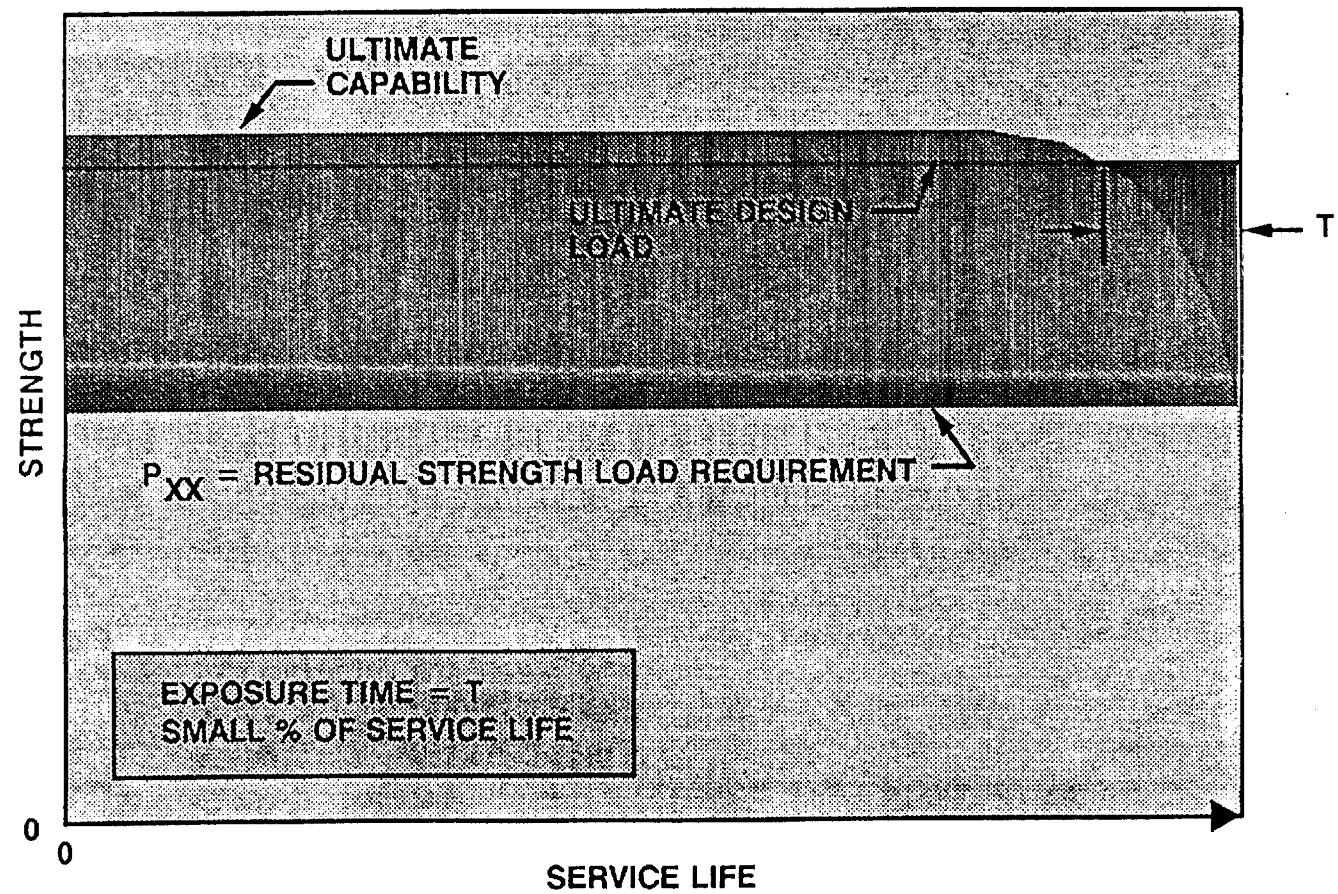
Residual Strength Load Requirements

P_{XX} *	Degree of Inspectability	Typical Inspection Interval	Period of Unrepaired Service Usage	Magnification Factor, M
P_{FE}	In-flight evident	One flight	Return to base	100
P_{GE}	Ground evident	One flight	Two flights of most damaging design mission	100
P_{WV}	Walk-around visual	Ten flights	Five inspection intervals	100
P_{SV}	Special visual	One year	Two inspection intervals	50
P_{DM}	Depot or base level	1/4 lifetime	Two inspection intervals	20
P_{LT}	Noninspectable	One lifetime	Two lifetimes	20

*P_{XX} = Maximum internal member load that will occur once in M times the appropriate inspection interval

Where P_{DM} or P_{LT} is determined to be less than design limit load, the design limit load shall be the required residual strength load level. P_{XX} need not be greater than 1.2 times the maximum load in one lifetime, if greater than design limit load

Impact Damage and Requirement
Metal Structure



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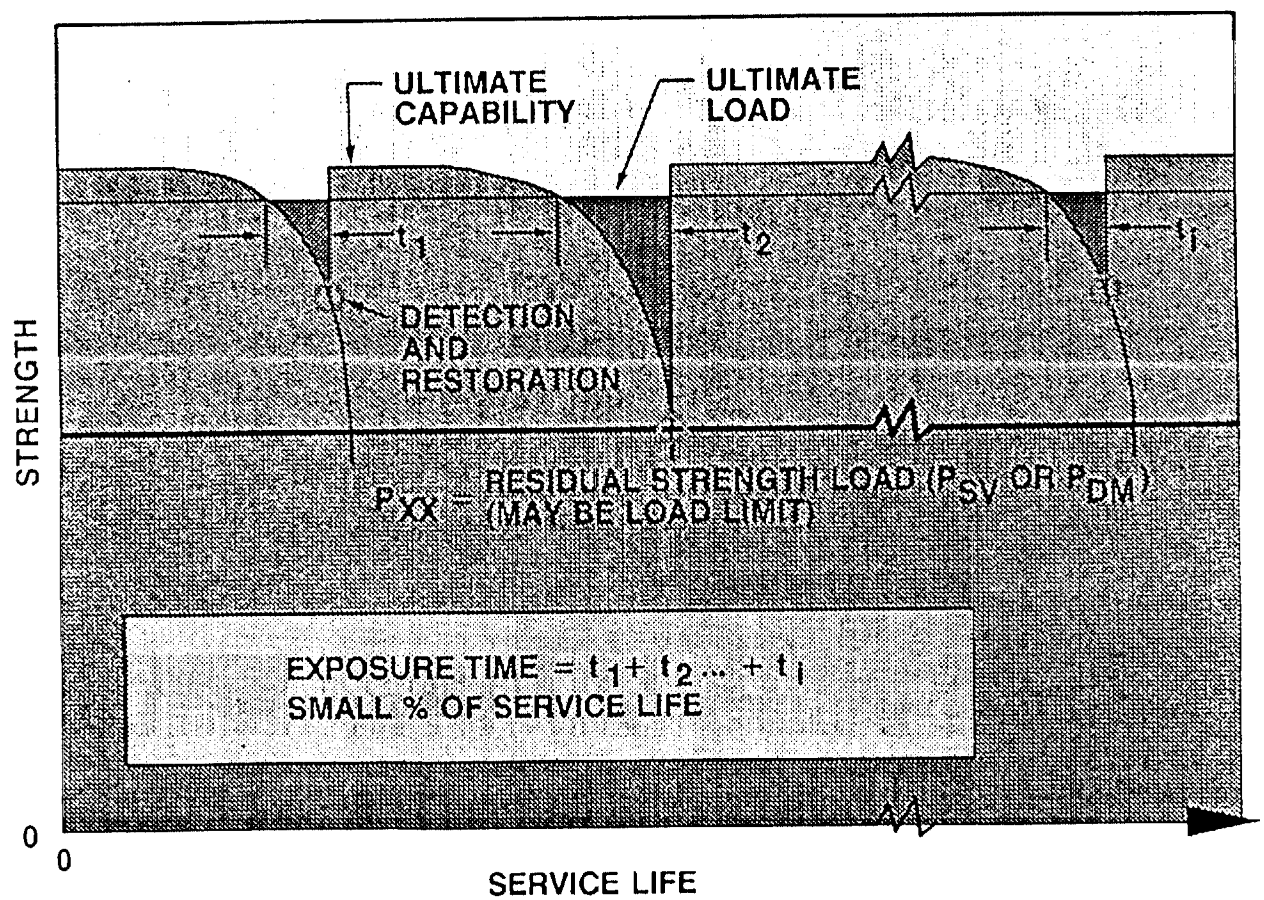
Damage Growth Detection and Restoration Cycle for Metal Structure



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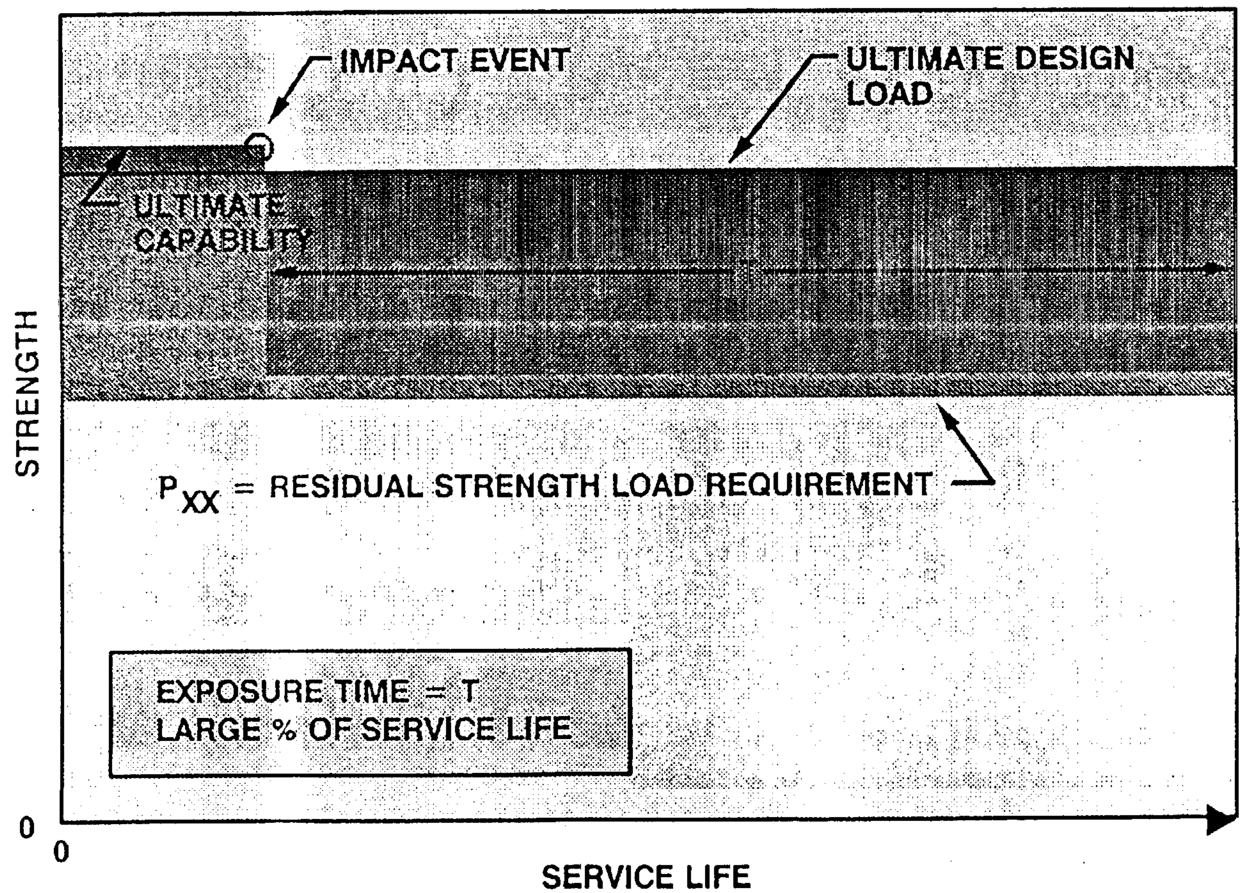
Impact Damage and Requirement
Composite Structure



Conservative Assumptions

- That an undetected 100 ft lb impact is a roque event
- That the occurrence is at a most critical location
- That the occurrence will account for worst case environmental conditions
- That with the above combined conditions, the airplane will experience an exceptionally high load

Industry and Air Force Participation and Concurrence

- Air Force
  - Reviewed approach
  - Concurrence
    - Flaw selection
    - Compliance requirements
- AIA
  - Two reviews
  - General acceptance
  - Compliance requirement realistic

FAA Advisory Circular AC-107A

"It should be shown that impact damage that can be realistically expected from manufacturing and service, but not more than the established threshold of detectability for the selected inspection procedure, will not reduce the structural strength below ultimate load capability. This can be shown by analysis supported by test evidence, or by tests at the coupon, element or subcomponent level"

Boeing Commercial Airplane Design Criteria

Damage Tolerance Design Philosophy

Composite primary structure shall meet the safety standards and the economic maintenance standards of current transport airplanes.

Strength, durability, and damage tolerance allowables and stiffness properties shall include the effects of adverse environments such as moisture, fuel, hydraulic fluid, and temperature.

Internal loads caused by environmentally induced strains shall be combined with ultimate, residual strength, and operating loads.

Structure shall be designed to be resistant to damage from normal handling in manufacturing or airline environments. Equivalence to service-demonstrated successful applications shall be obtained.

Repaired structure shall meet the requirements for static strength, durability, damage tolerance, and stiffness. Fixed structure shall be designed for on-airplane repair.

A1891.04 s1h

Boeing Commercial Airplane Design Criteria

Ultimate Strength

Accidental Impact Damage

It shall be shown that impact damage that can be realistically expected from manufacturing and service, but not more than the established threshold of detectability for the selected inspection procedure, will not reduce the structural strength below ultimate load capability. (FAA AC 20-107A, 6.g.)

Composite primary structure shall be capable of sustaining ultimate load with isolated impact damage, inflicted by any likely energy source that cannot be seen visually. To assure that this criteria is met, the structure shall be designed to sustain ultimate load after being impacted with the minimum energy source (up to 1,200 in-lb) that is required to produce impact damage that is visible from a distance of 5 feet with the unaided eye under normal structural inspection conditions.

A1891.05 s1h

Boeing Commercial Airplane Design Criteria

Ultimate Strength

Ground hail

Structure exposed to ground hail shall sustain ultimate load and not require immediate structural repair following impacts as specified below, spaced 12-in apart at critical locations.

Structure type	Surface position	Hail diameter, in	Impact energy, in-lb
Fixed primary	Horizontal	2.5	500
	Vertical	2.5	300
Removeable primary	Horizontal	1.5	70
	Vertical	1.5	40

Lightning strike

Primary structure exposed to lightning strike shall sustain ultimate load and not require immediate structural repair following the maximum energy strike occurring at the considered location once in the life of the airframe.

A1891.06 s1h

Boeing Commercial Airplane Design Criteria

Damage Growth and Residual Strength

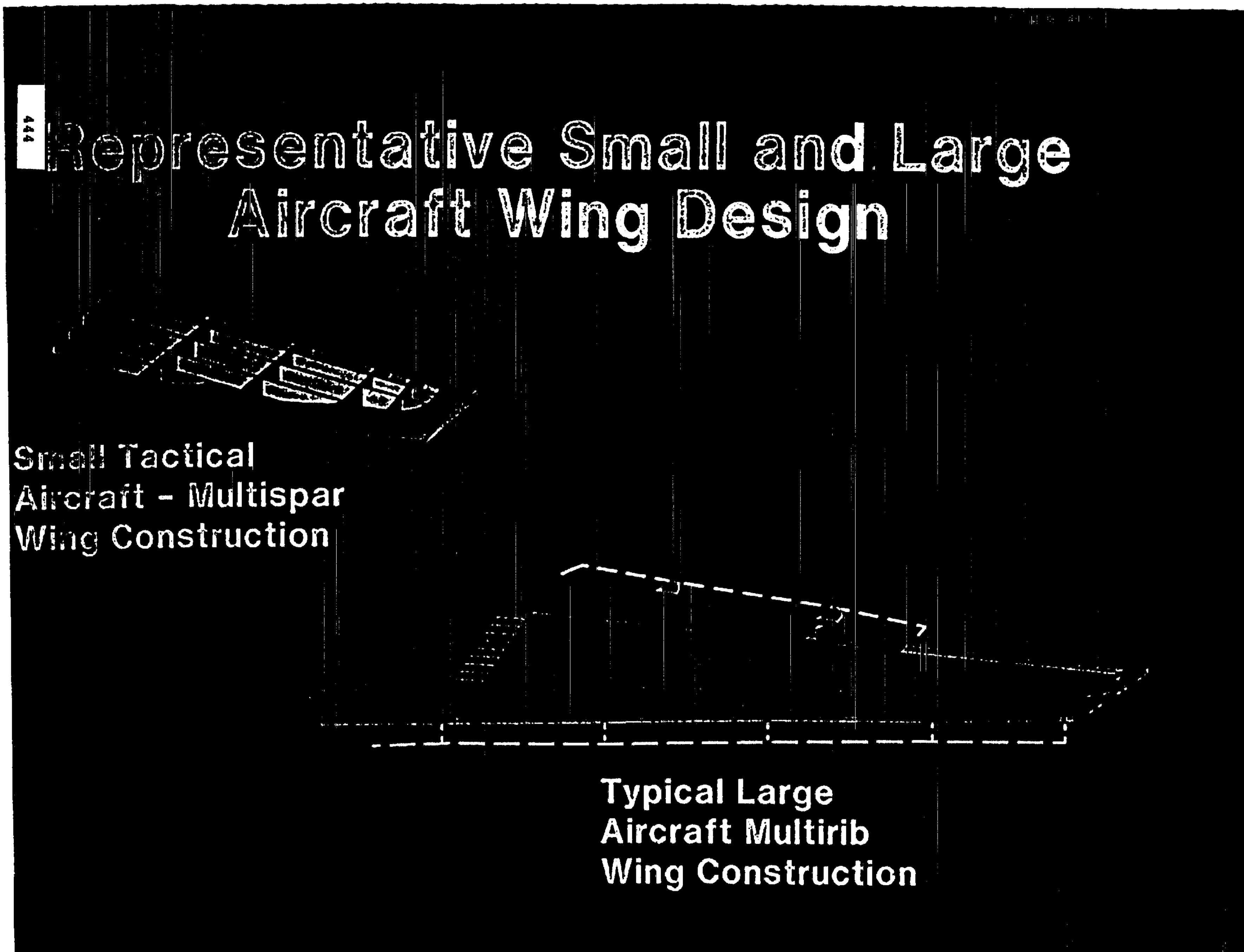
Composite primary structure shall be designed to preclude detrimental damage growth under operating loads following isolated impact damage at the energy levels defined for ultimate strength requirements with accidental impact damage.

Composite primary structure shall be designed damage tolerant, i.e., it shall be shown by analysis or test that fatigue, corrosion, or accidental damage will be detected by the structural inspection plan and prevent catastrophic failure during the operational life of the airframe.

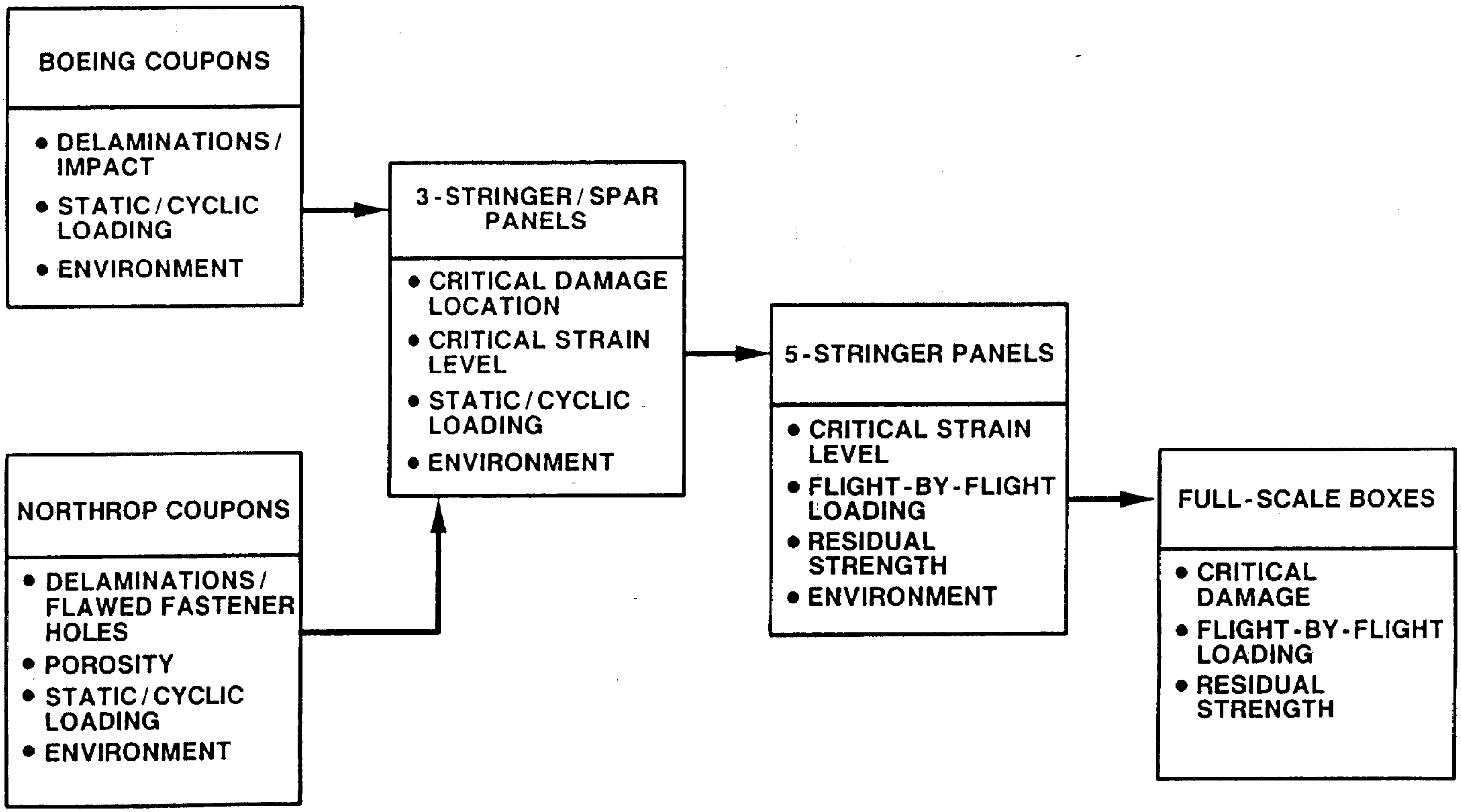
Primary composite structure shall satisfy regulatory residual strength requirements (FAR 25.571(b)) following failure or obvious partial failure of a single structural unit (e.g., a stiffener with two skin bays, a chord with one skin bay, or similar combinations).

A1891.07 s1h

AIR FORCE DAMAGE TOLERANCE REQUIREMENTS VALIDATION PROGRAM


444
Representative Small and Large
Aircraft Wing Design
Small Tactical
Aircraft - Multispar
Wing Construction
Typical Large
Aircraft Multirib
Wing Construction

Building-Block Test Approach



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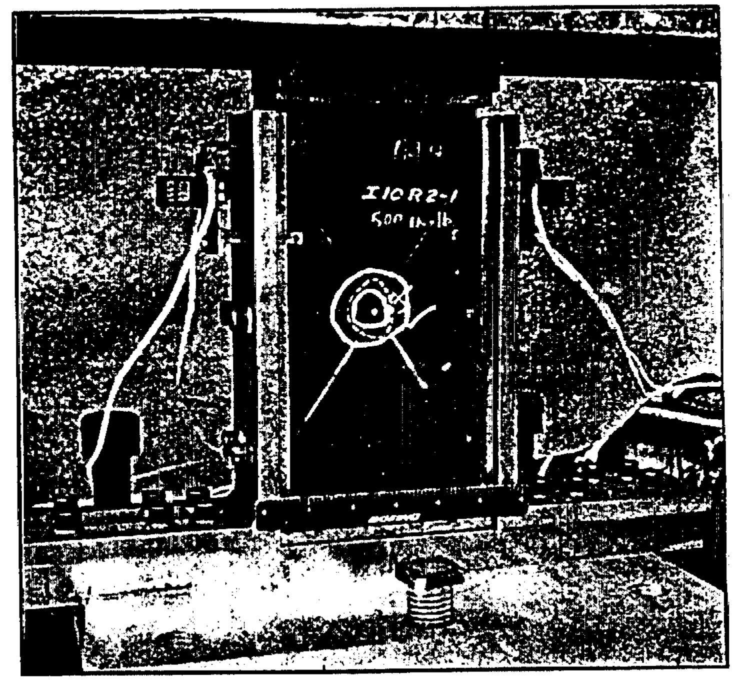
Coupon Basic Property Test Matrix

Specimen description	Static ▷1			Fatigue			
				Const amplitude			Sprectrum
	RT dry	RT wet	-75°F dry	RT dry	RT wet	-75°F dry	RT dry
Delam-ination	●	●	●	●	●	●	●
200-in-lb impact damage	●			●			
350-in-lb impact damage	●			●			●

Material: AS6/2220-3

Specimen: 0.25 x 5.0 x 10.0

▷1 R = 10



Test Setup

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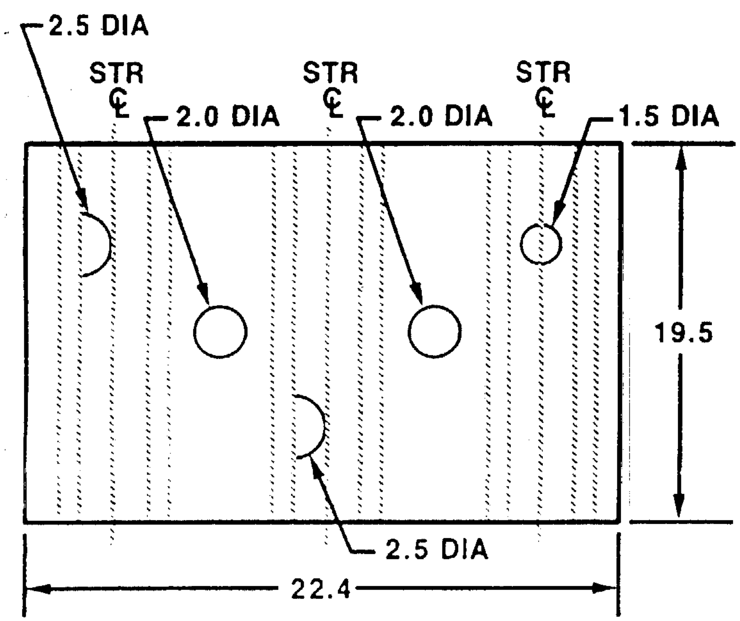
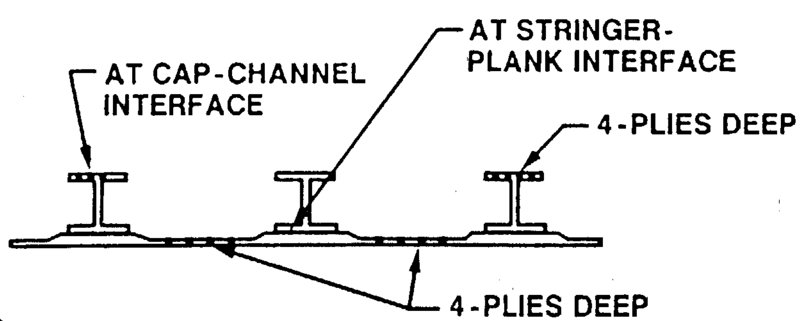
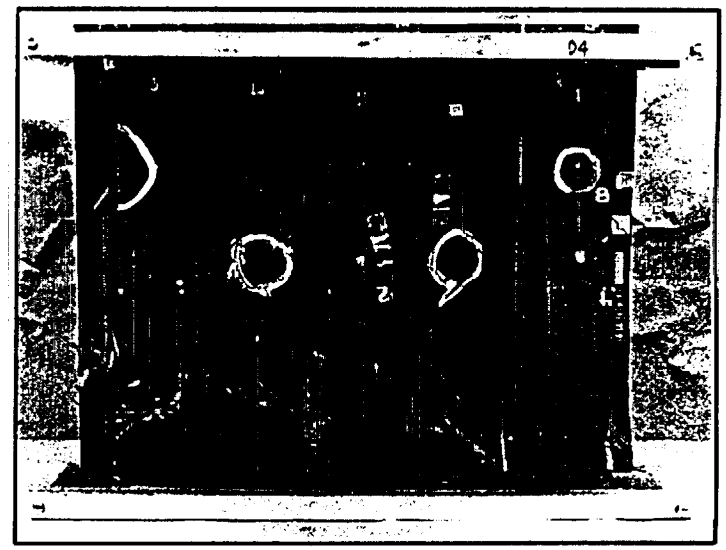
Location of Delaminations in 3-Stringer Panels



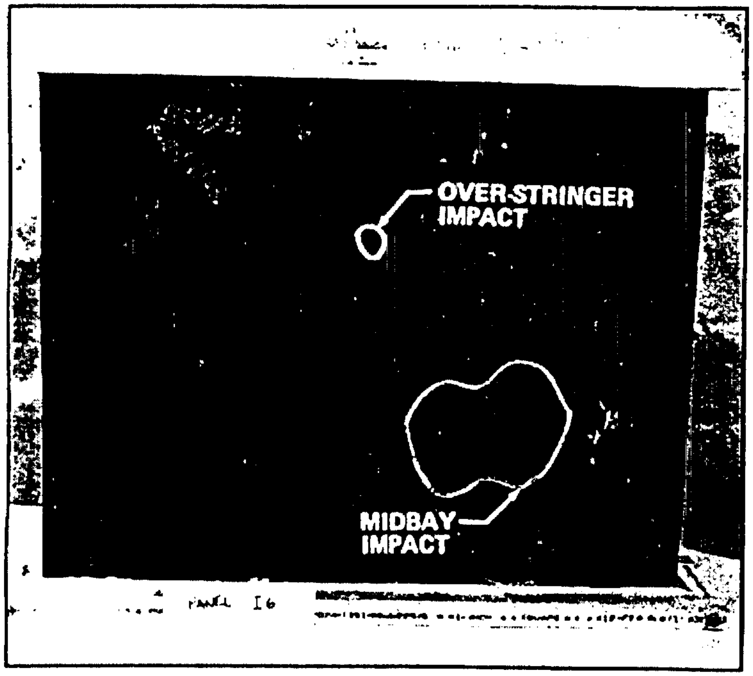


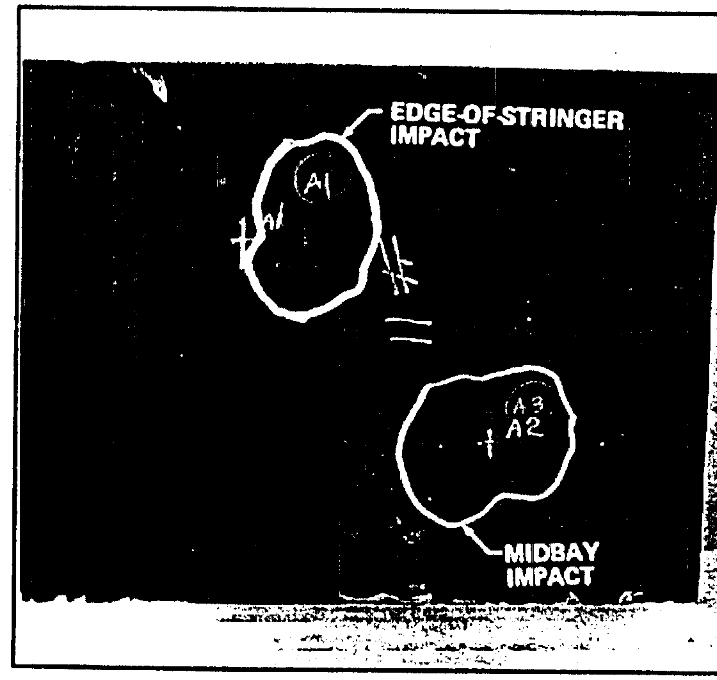


Panel After Failure

G32662B 060

Impact Damage on 3-Stringer Panels
Ultrasonic Pulse Echo Readings





1200-in-lb impacts
1-in-dia tip

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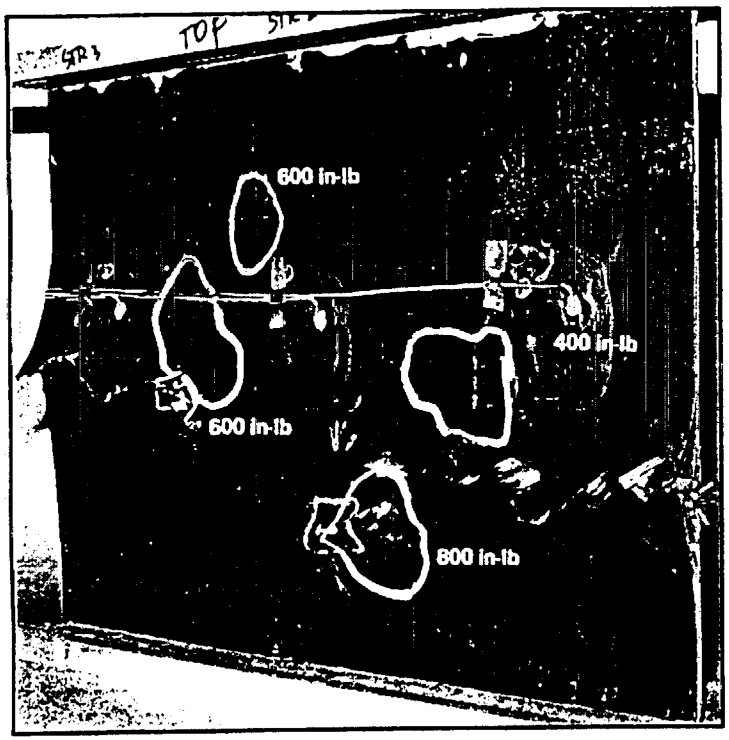
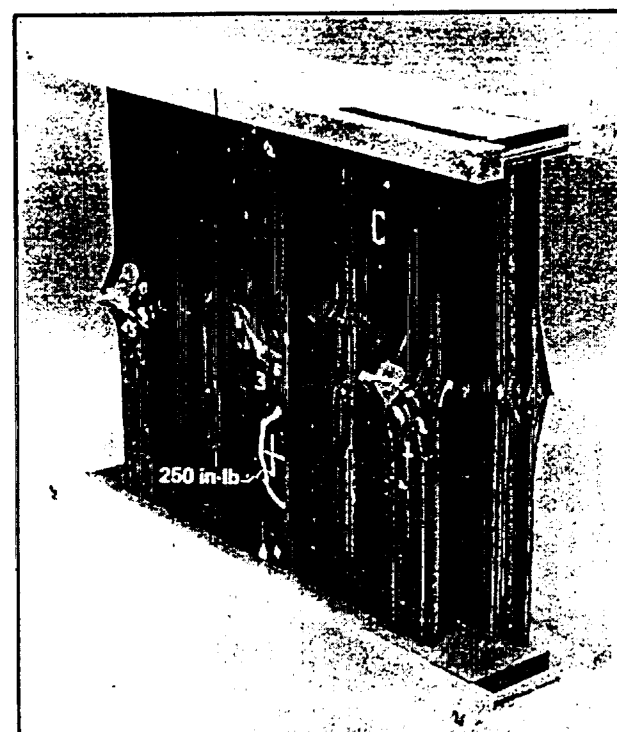
Impact Damage on 3-Stringer Panel

Ultrasonic Pulse Echo Readings



Multiple Impact
Damage



Internal Stringer Flange
Impact Damage

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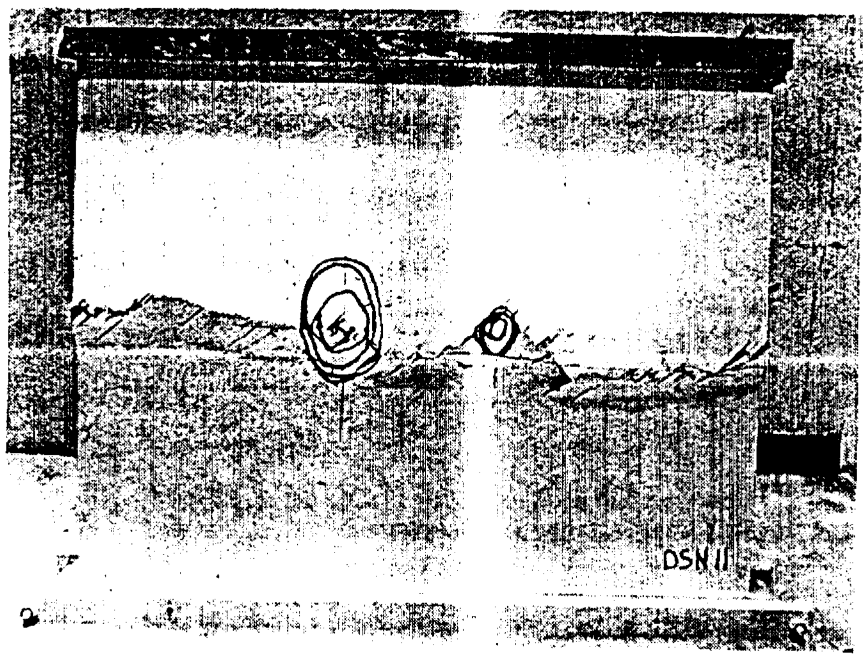
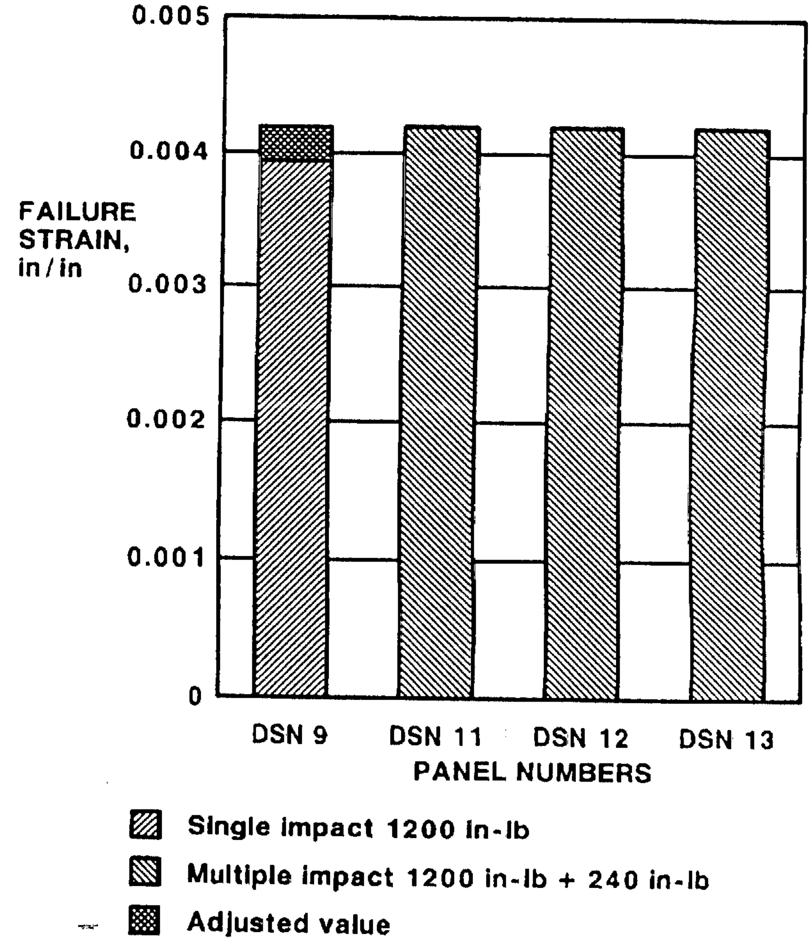
Multiple Impact Static Test Results



Failed Panel



- AS4/3501-6

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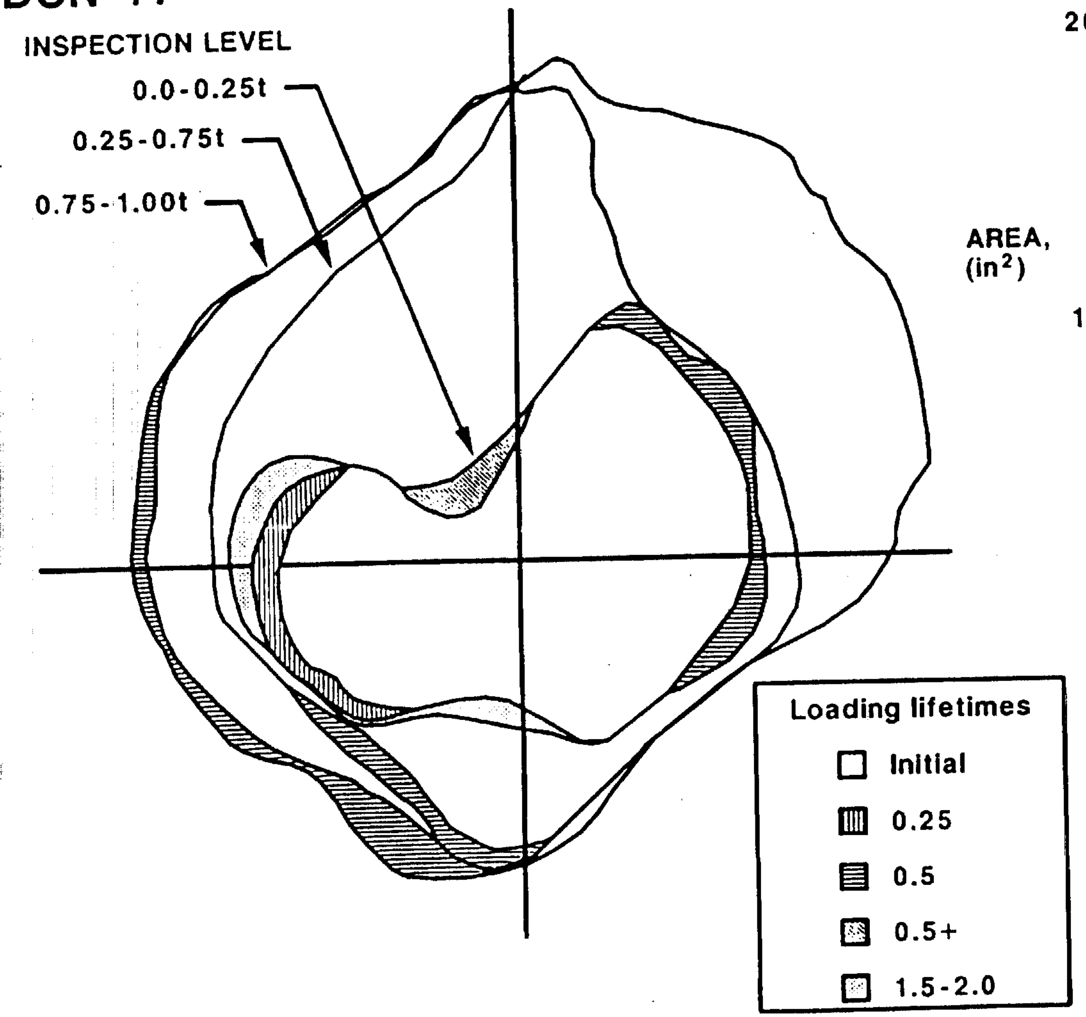
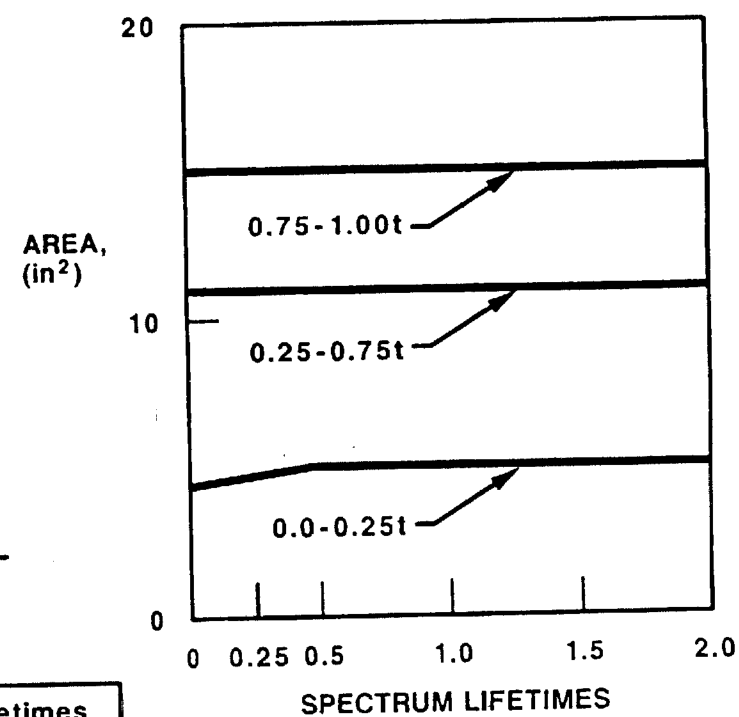
Damage Growth Pattern
DSN 17





- 10/80/10 Skin layup
- AS4/3501-6

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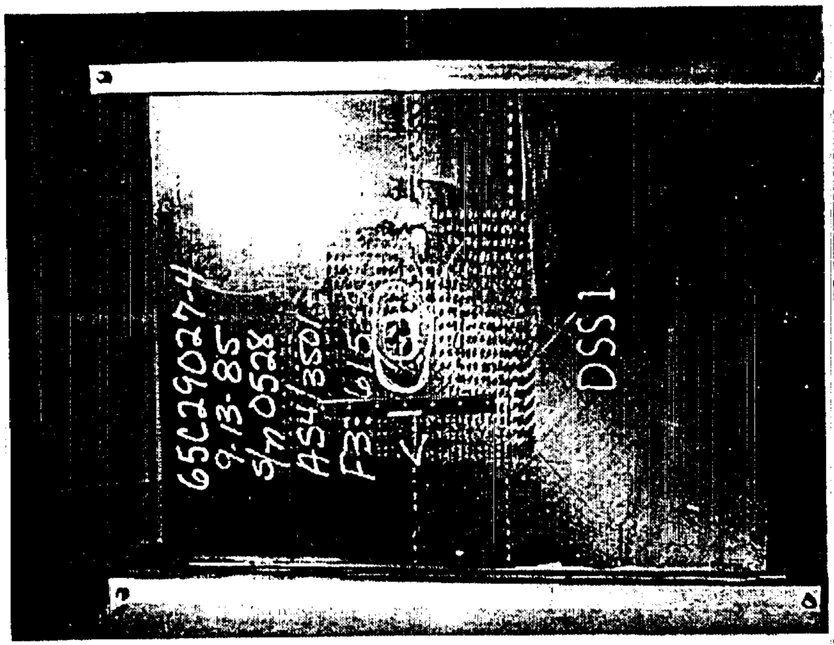
Effect of Stitching



Stitched Panel

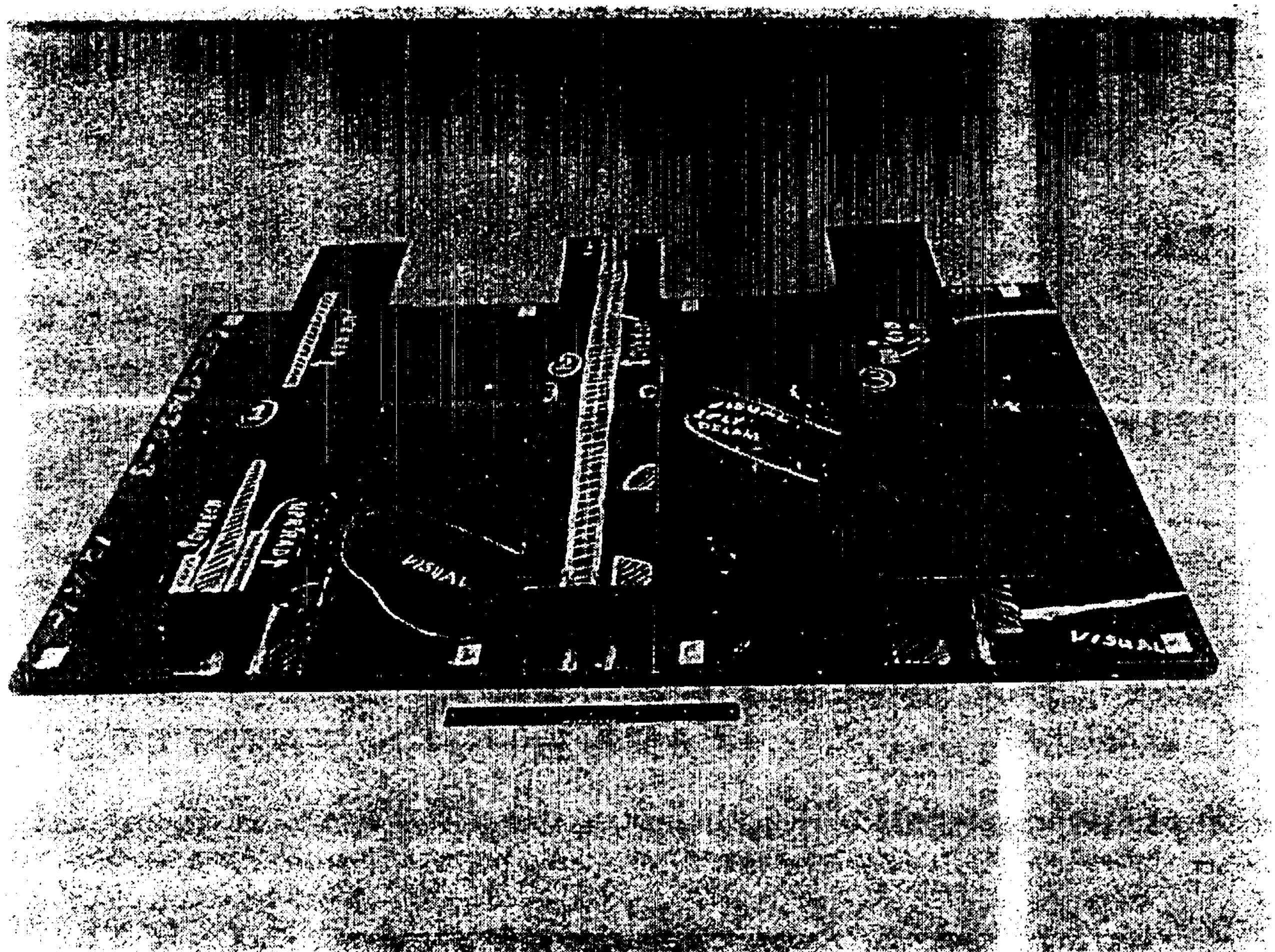
Stitching
- 0.25 inch pitch and row spacing
- Thru skin, planking, and stringer flange edge
- Kevlar thread

Other
- 10 / 80 / 10 Skin layup
- 1200 In-lb impact
- Tested wet at 180 F°

Results
- Failed at 114% of control

THERMOPLASTIC - PEEK APC - 2

First Panel After Initial Consolidation Showing Void Areas



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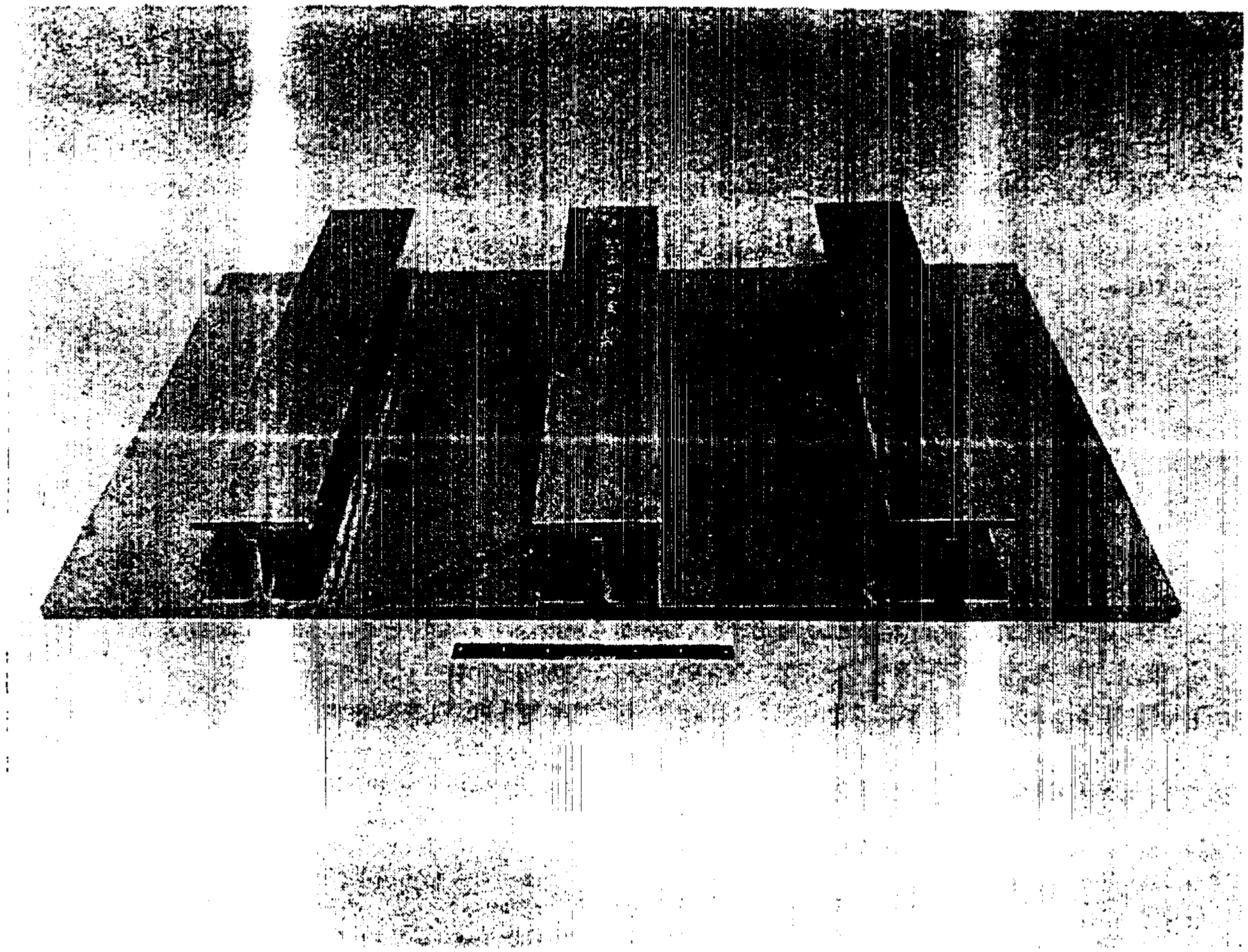
THERMOPLASTIC - PEEK APC - 2

Panel After Tool Rework and Reconsolidation



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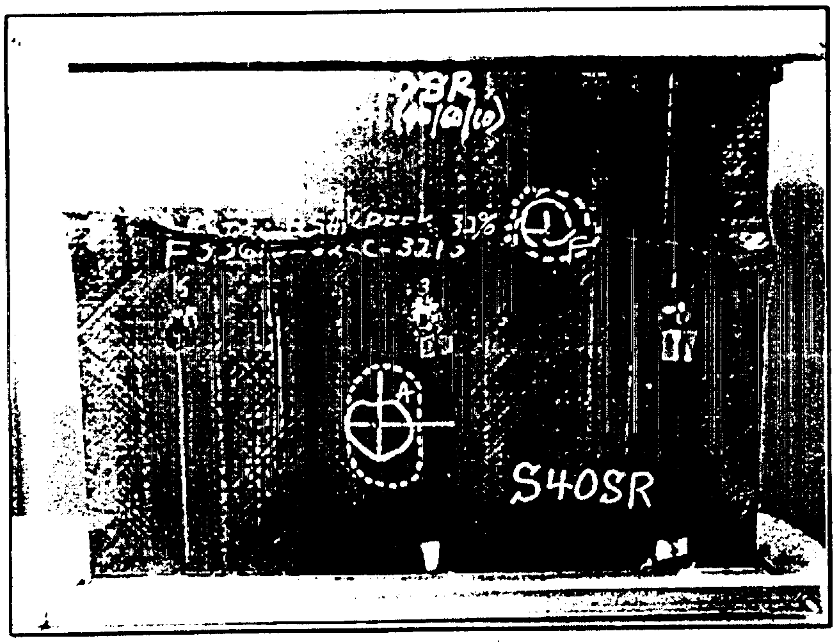
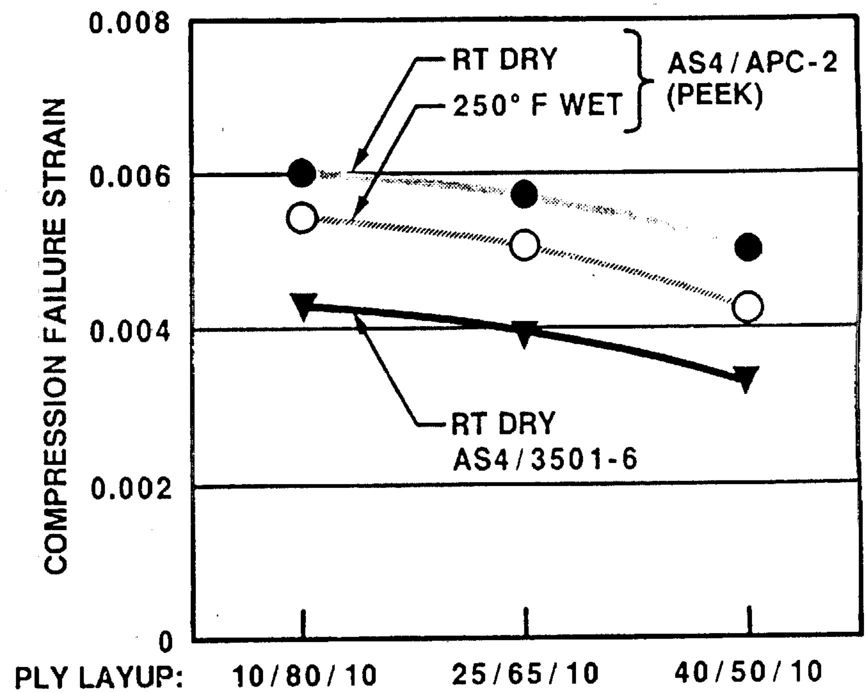
Comparison of Post-Impact Failure Strain — AS4/APC-2 (PEEK) Versus AS4/3501-6



Failed Panel



100 ft-lb impact

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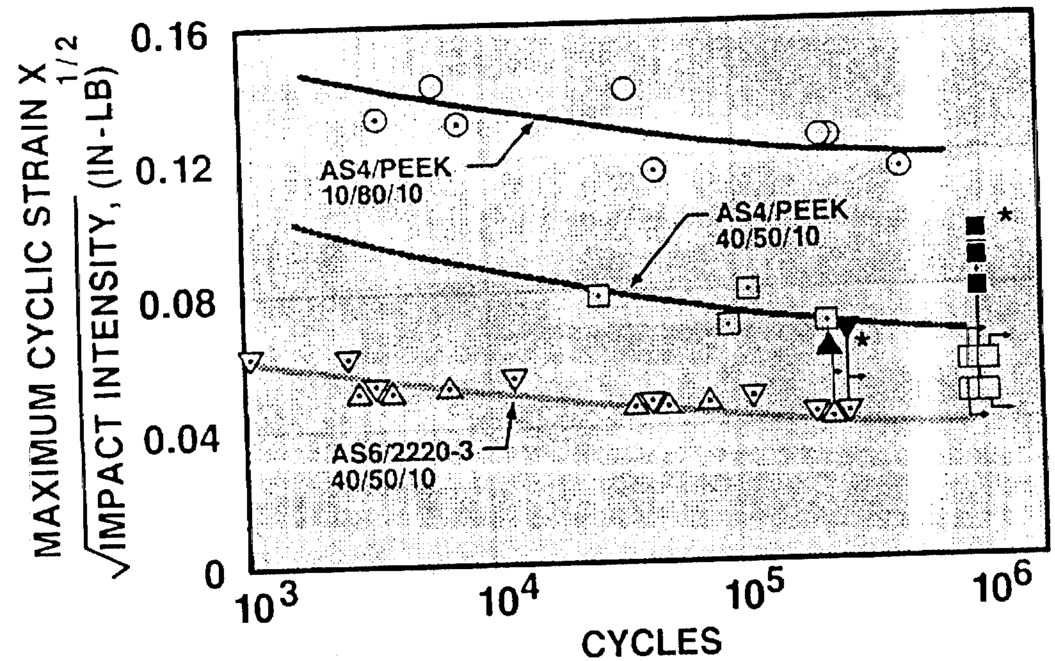
Fatigue Test Results of Impacted Laminates

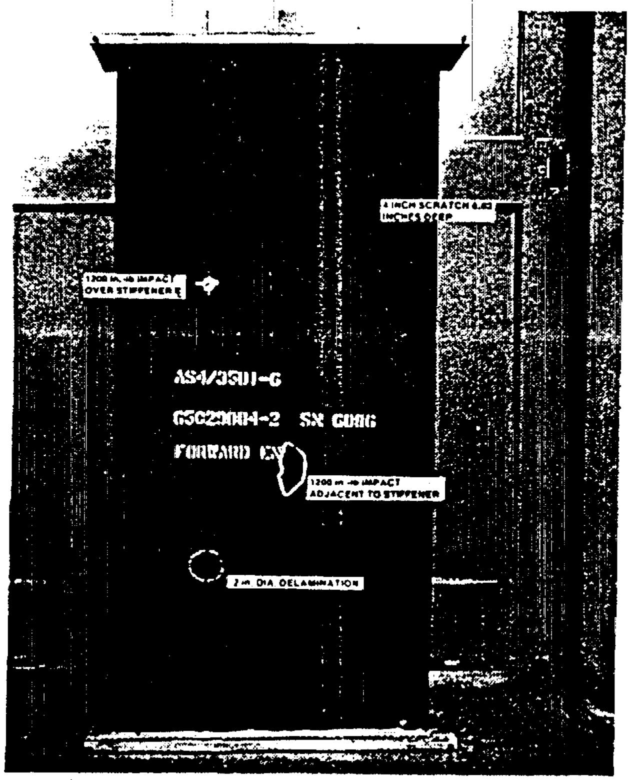


	Material	Layup	Impact in-lb
○	AS4/PEEK	10/80/10	250
⊙	AS4/PEEK	10/80/10	500
◇	AS4/PEEK	40/50/10	250
◇ (dotted)	AS4/PEEK	40/50/10	500
△	AS6/2220-3	40/50/10	200
▽	AS6/2220-3	40/50/10	350

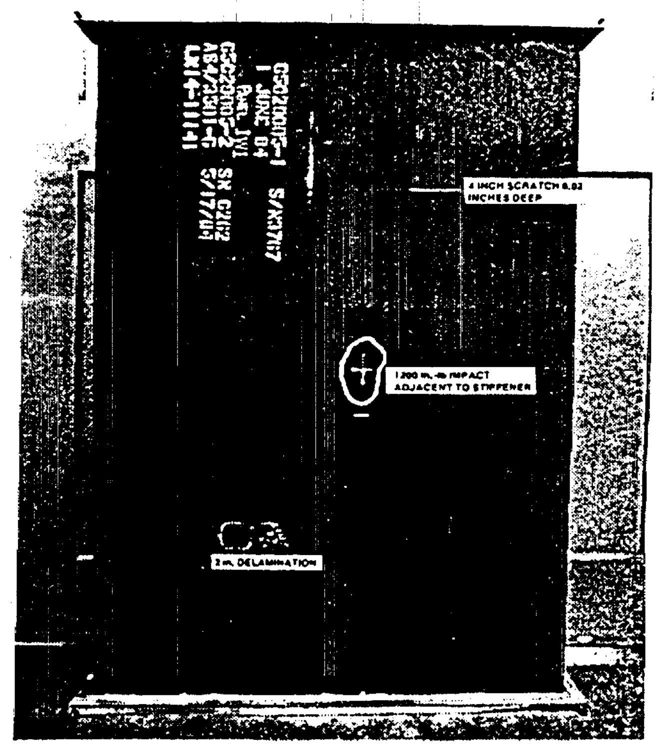
Notes:
- R = 10
- 70°F, dry
- 0.25 X 5.0 X 10.0
- *Residual static strength

Task II 5-Stringer Test Panels

Prior to Test



BLADE-STIFFENED PANEL



I-STIFFENED PANEL

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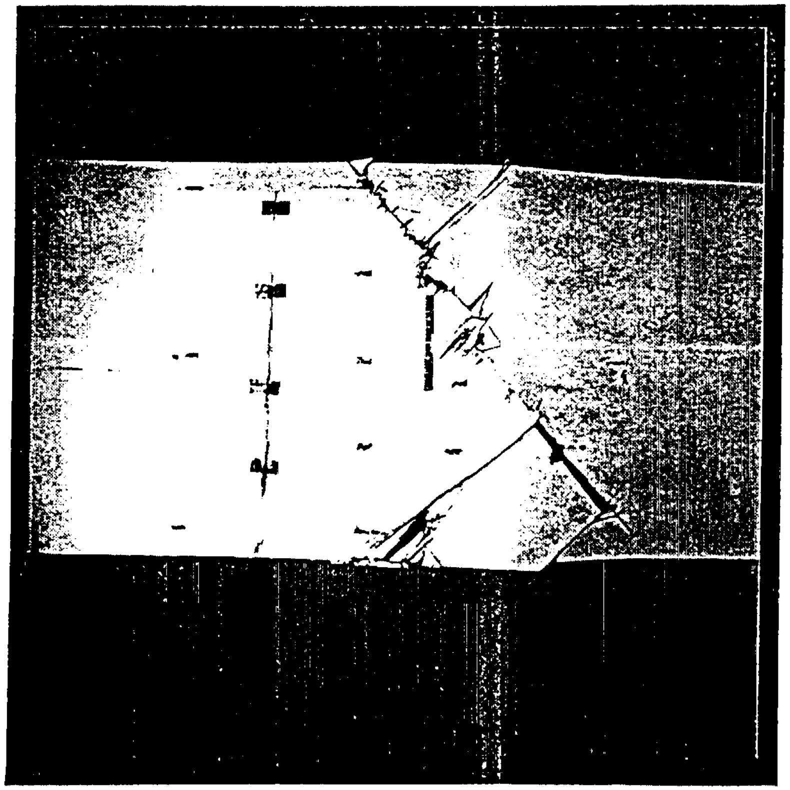
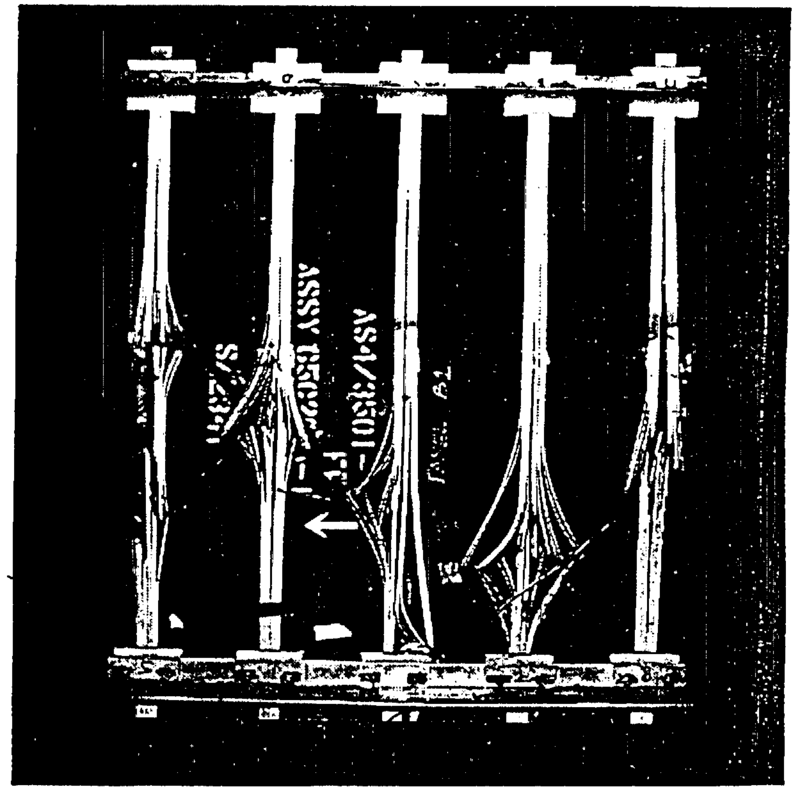
Typical Blade-Stiffened Panel Failure

5-Stringer Panel



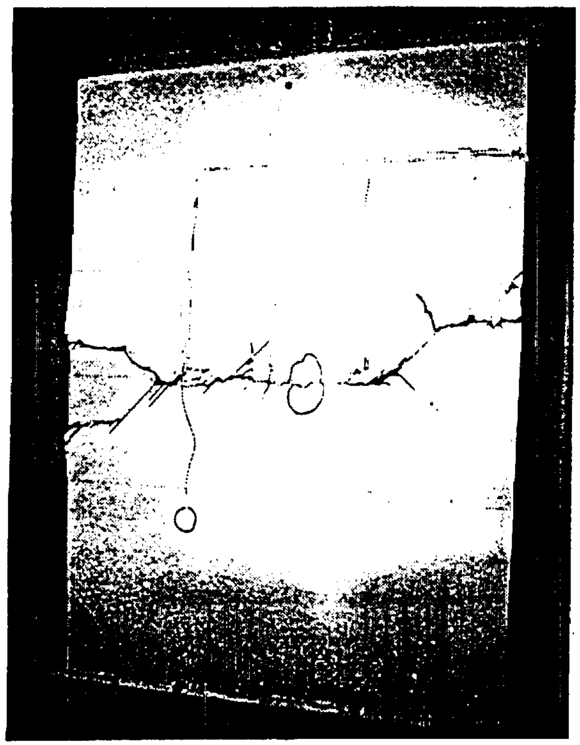
SKIN SIDE



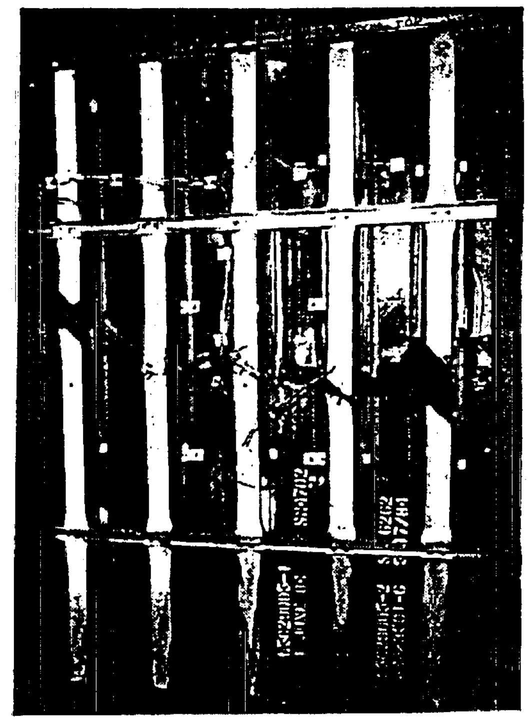
STRINGER SIDE

Typical I-Stiffened Panel Failure

5-Stringer Panel



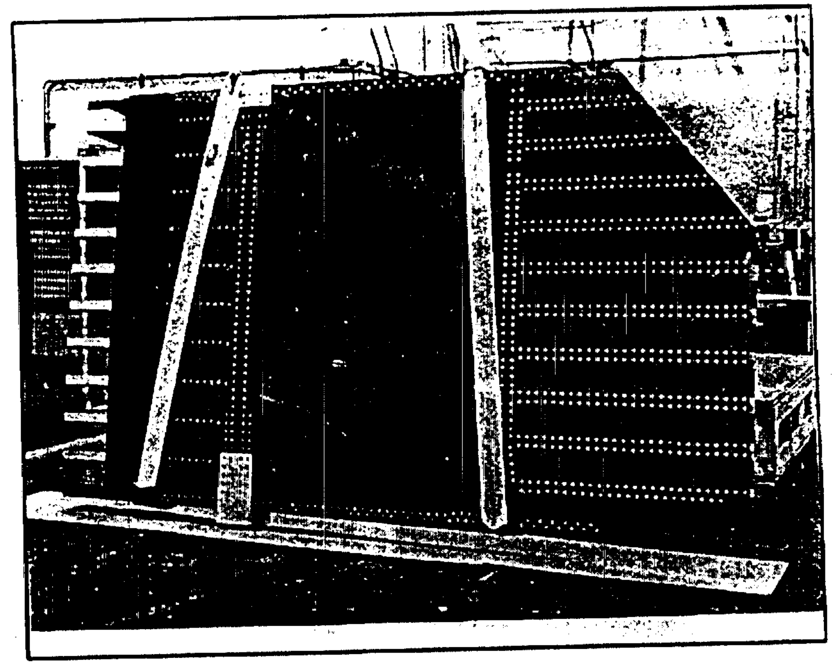
SKIN SIDE



STIFFENER SIDE

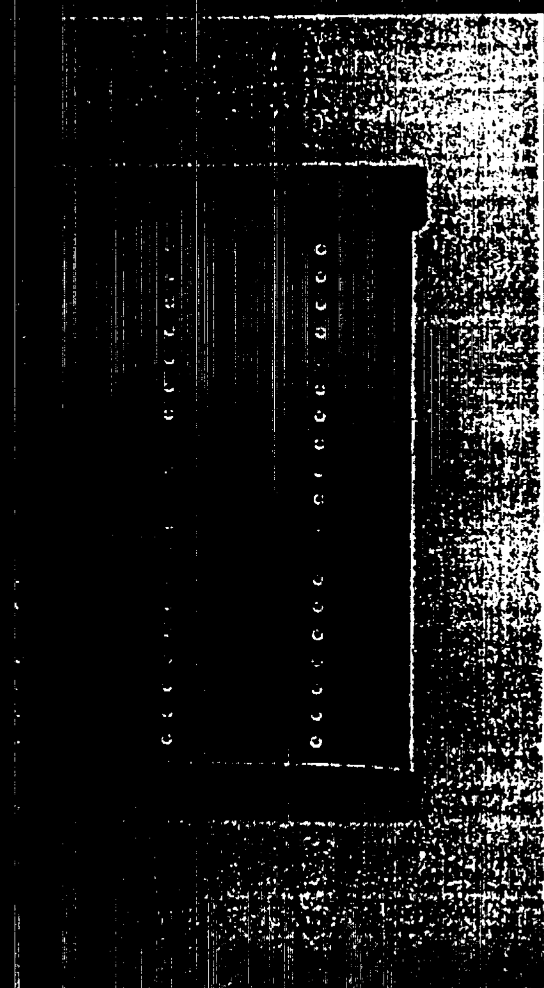
Northrop Wing Box Test

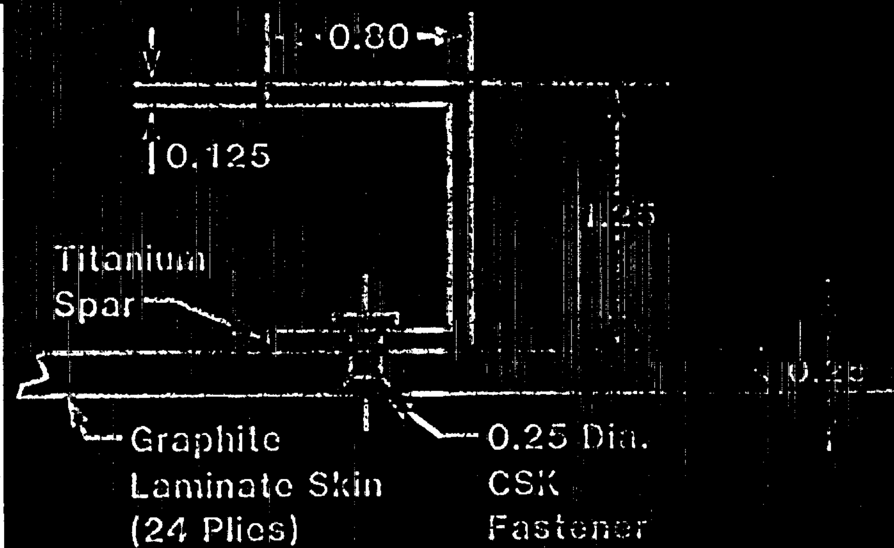
- Conventional composite fighter wing design
- Multispar, hard skin
- Mechanically fastened upper skin to spar
- Harsh load cycling requirement



Northrop Fighter Wing Test Box

G32839B 060





- Panel Size: 18.0 in [illegible]
- Stringer Spacing: [illegible]
- Skin Buckling Design [illegible]
- Material: AS4/3501-6
- Nominal Ply Thickness [illegible]
- Ply Layup Orient[illegible]
[±45/90/0/(±[illegible])]

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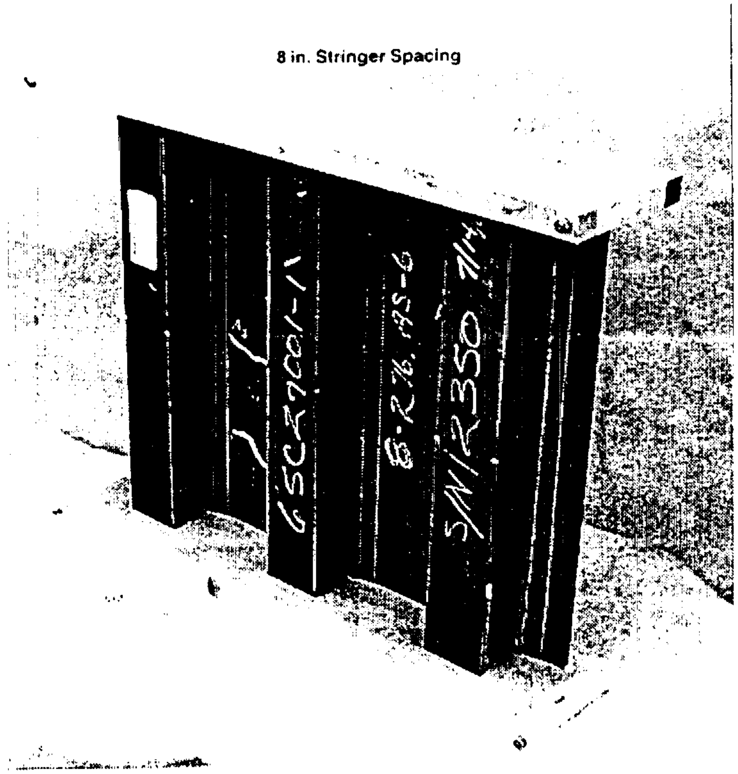
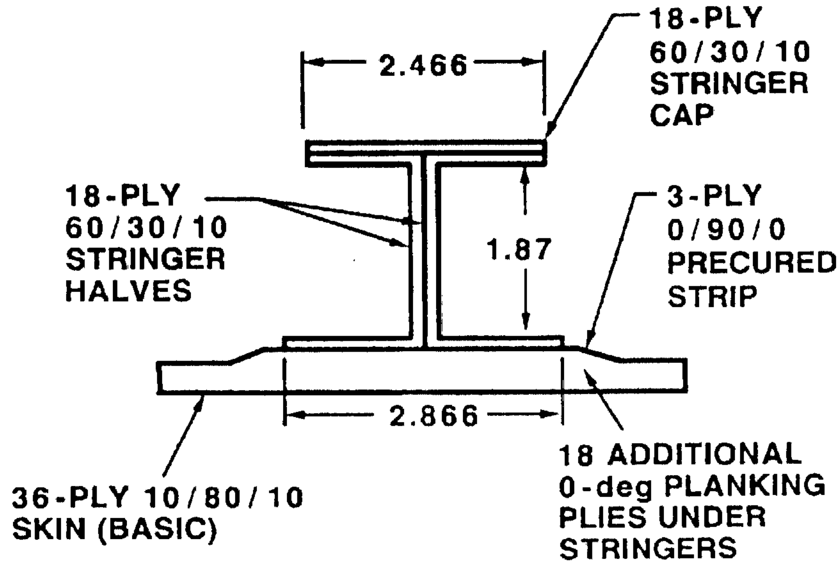
3-Stringer Panel Design





- Panel size: 22.4 in x 19.5 in
- Stringer spacing: 8 in
- Design load: 25k/in
- Ultimate design strain: 0.005
- Material: AS4/3501-6
- Nominal ply thickness: 0.0073

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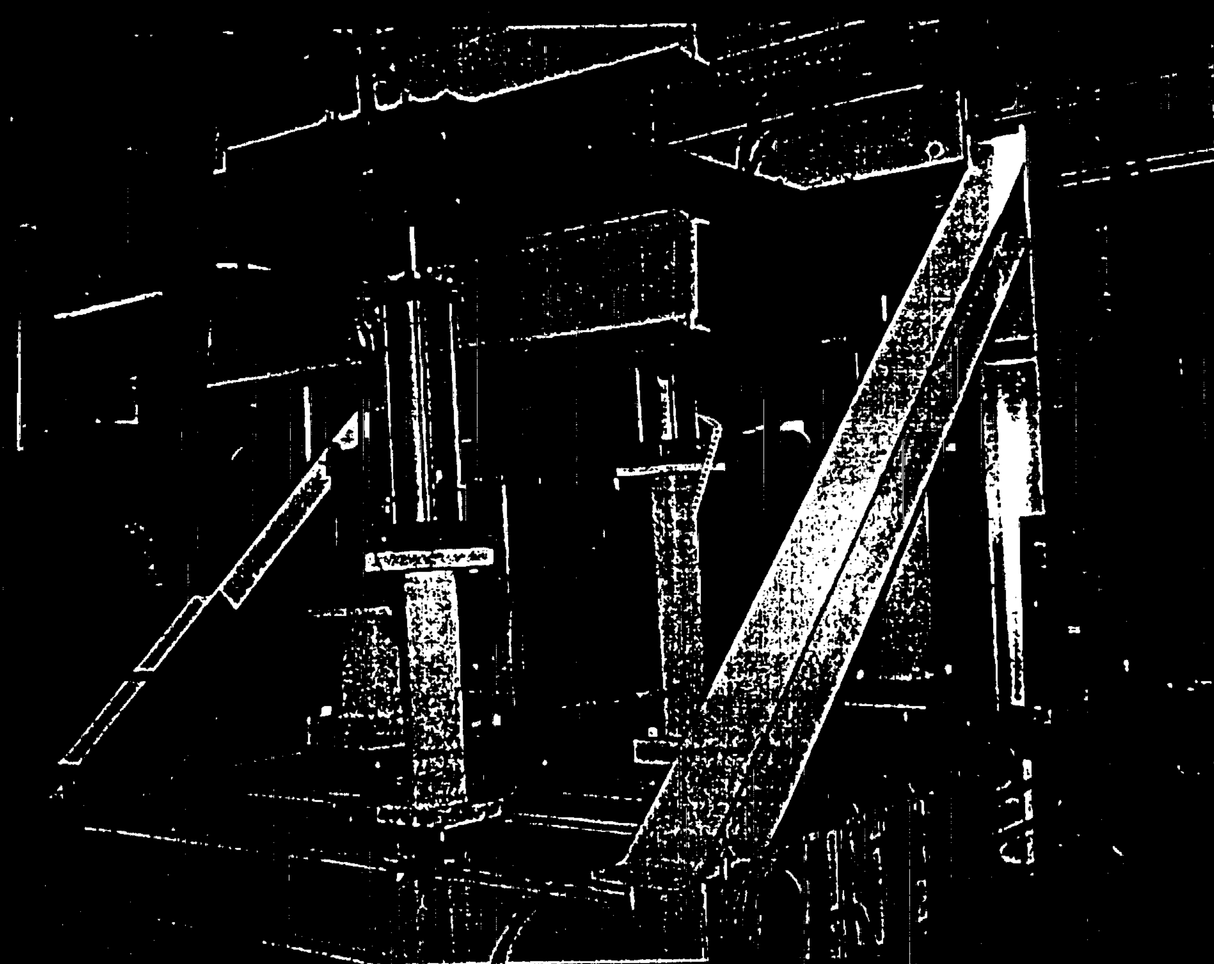
Multispar Wing Box Test



Multispar Wing Box Failure Summary

Failure occurred after 355 flight cycles (0.07 lifetimes) at a maximum spectrum load.

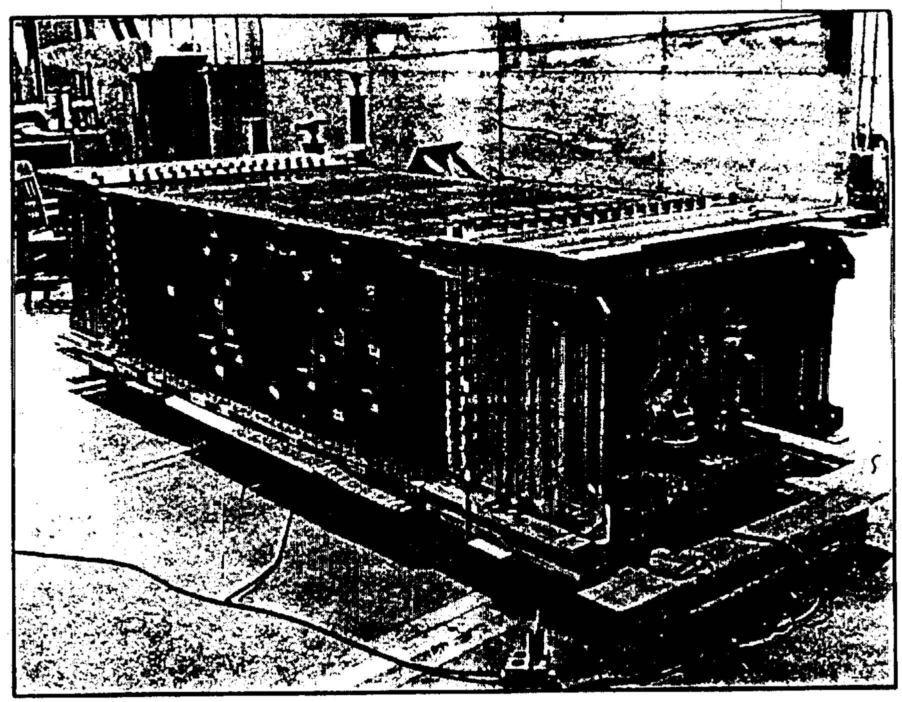
The box had been designed to the following upper skin strain levels without consideration for damage tolerance -

- Design ultimate strain - 0.004
- Design limit strain - 0.00267
- Maximum spectrum strain - 0.00325

The box would require redesign to meet damage tolerance requirements.

Boeing Wing Box Test

- Advanced design composite transport wing
- Multirib, soft skin
- Co-cured stiffeners with integral reinforcing planks
- Moderate load cycling requirement



Boeing Transport Wing Test Box

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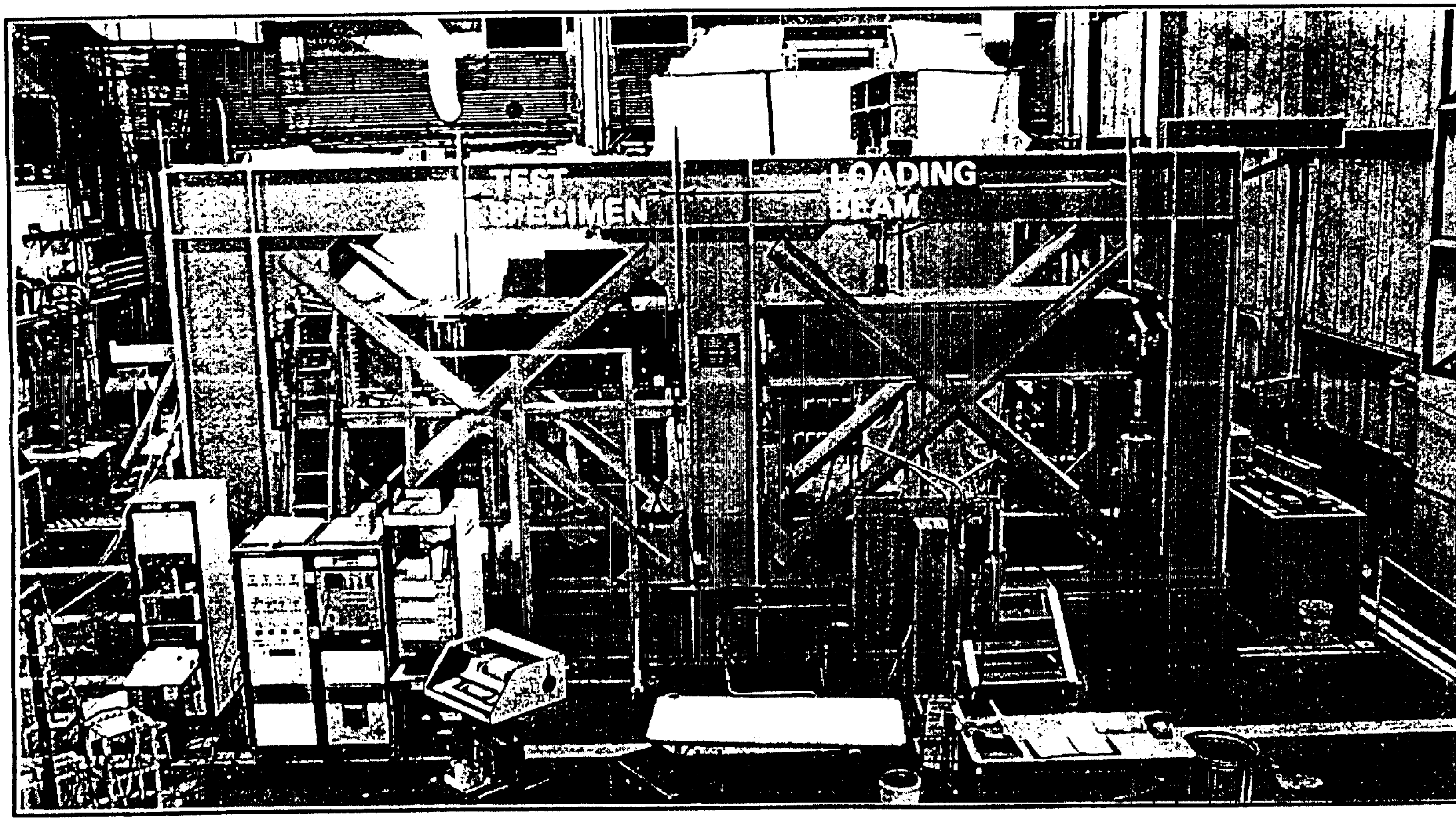
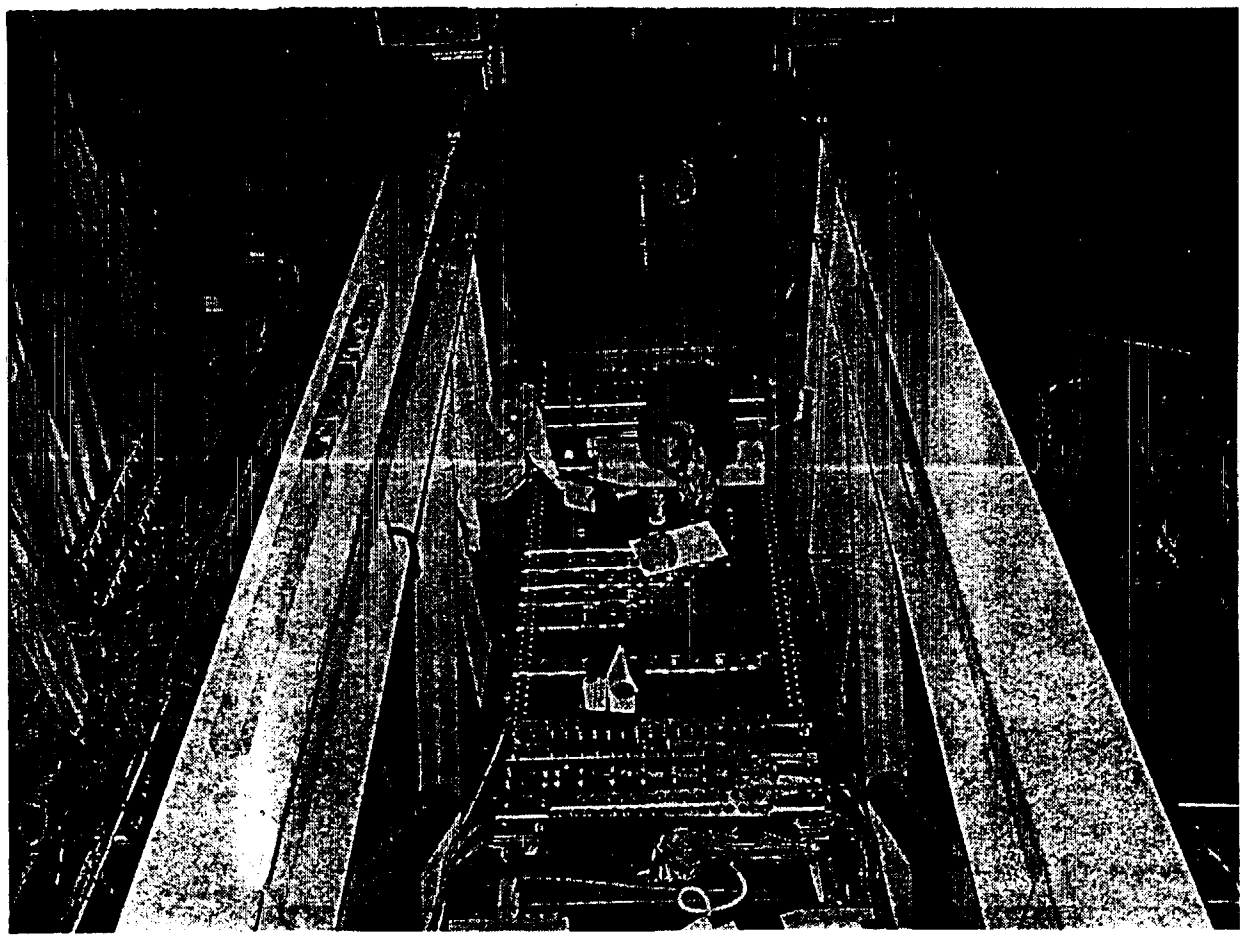
Side Elevation View of Test Setup



G31690 054

Multirib Wing Box in Loading Fixture

Installation of Strain Gages



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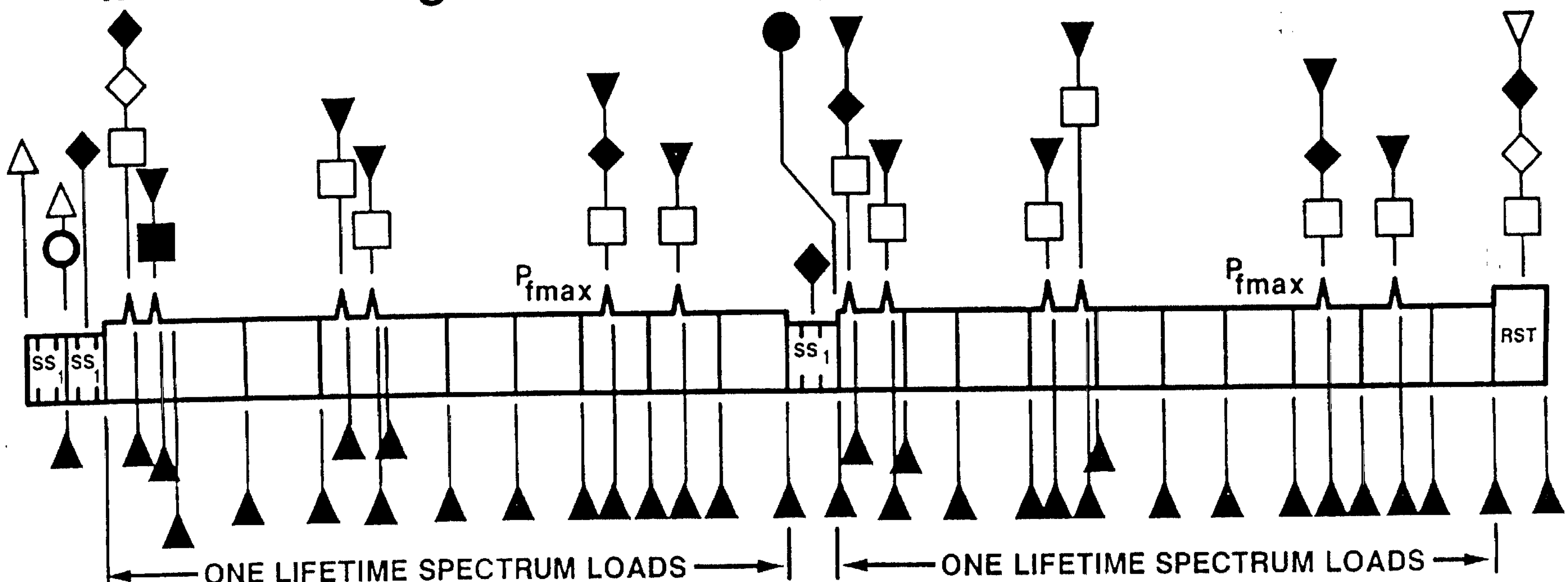
Multirib Wing Box Test Sequence



■ Strain survey load
- Pure bending
- Pure torque
- Combined

□ Strain survey during load at 10% intervals

○ Impact damage in five places

● Impact damage in one place

◆ Moire fringe photographs

◇ 16 mm motion pictures

▼ Video

▽ High speed video

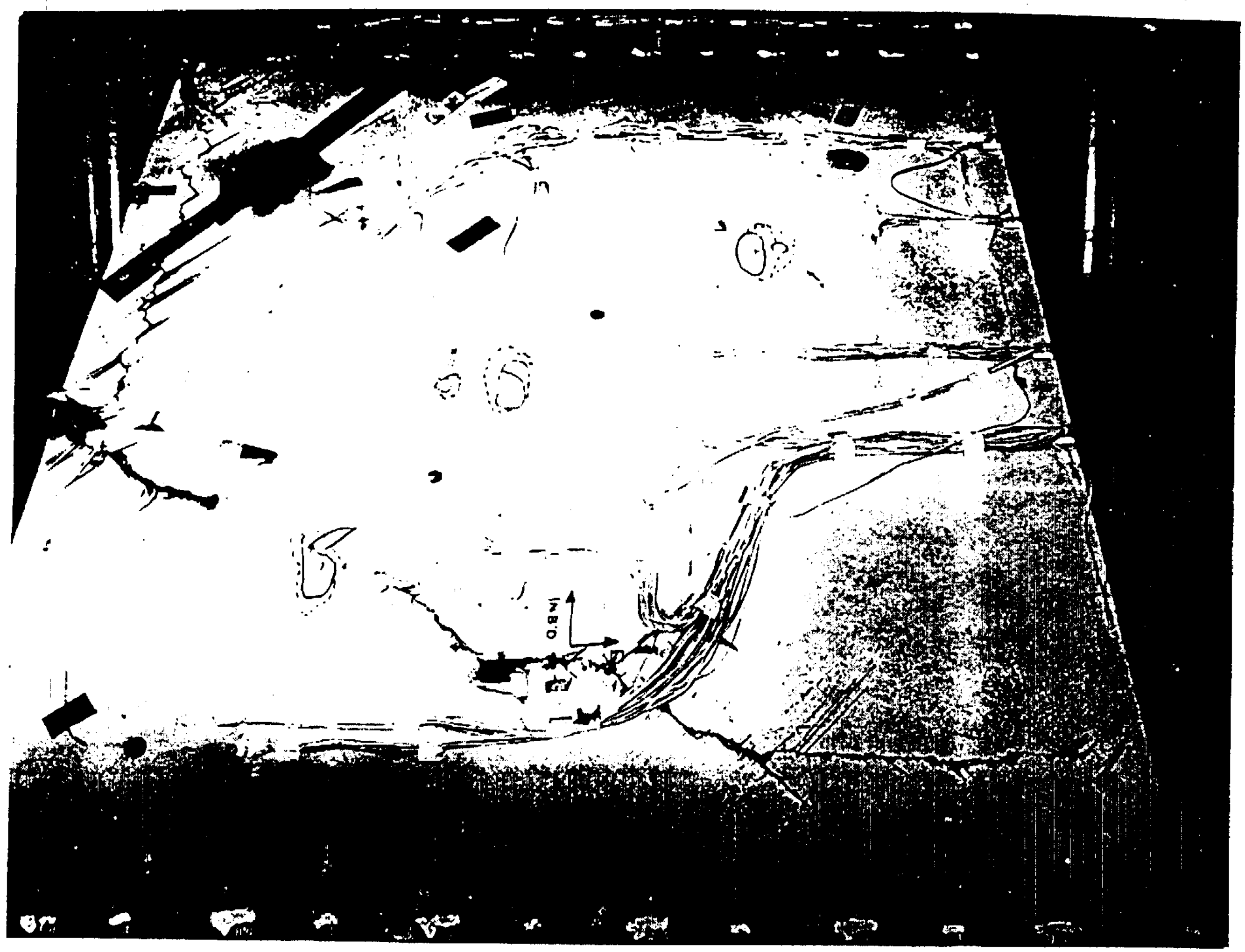
△ Still photographs

▲ Pulse echo inspection

RST=Residual Strength Test

P_{fmax} $\epsilon = -0.0032$ in/in

Upper Surface Panel Compression Failure



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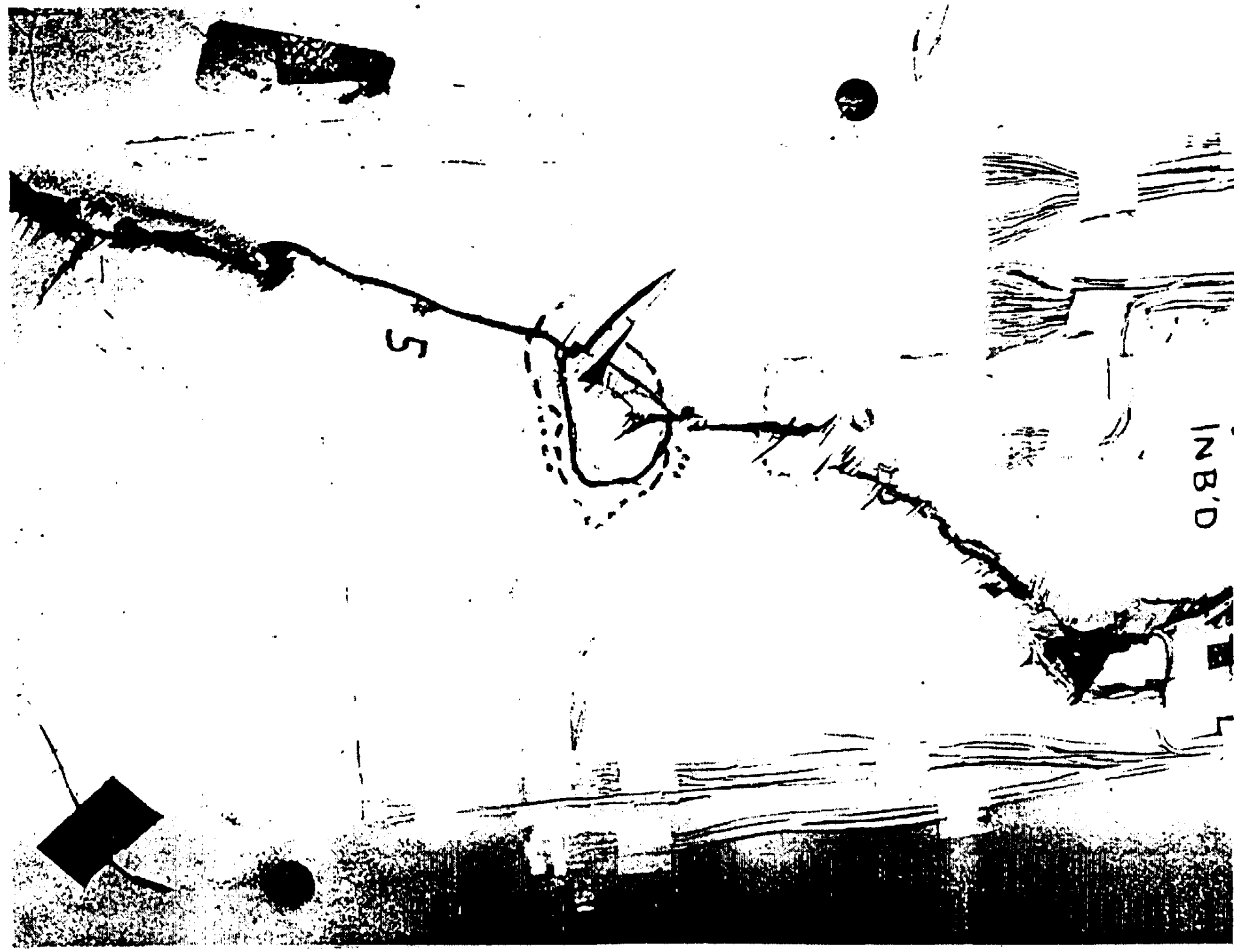
Upper Surface Panel Compression Failure Close-up



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Conclusions

- Failure was in the test box upper surface panel and initiated at an induced impact damage location

- The strain at failure was 4200 μ in/in, at 105% of the design goal

- The test met all qualification requirements

Air Force Damage Tolerance Program Results

- Establishment of damage tolerance specification requirements.
- Development and use of a building-block test approach to demonstrate specification compliance.
- Evaluation of a tough thermoplastic material to demonstrate superior damage tolerance properties.
- Development of analysis methods to predict damage characteristics.
- Testing to investigate methods and materials to improve damage tolerance.
- Assessment of the characteristic ability of the representative composite fighter and transport structure to resist battle damage.

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15. Supplementary Notes

Primarily viewgraphs.

16. Abstract

This document is a compilation of the slides presented at the NASA Workshop on Impact Damage to Composites held on March 19 and 20, 1991, at Langley Research Center, Hampton, Virginia. The Mechanics of Materials Branch of NASA Langley Research Center sponsored the workshop. The objective of the workshop was to review technology for evaluating impact-damage tolerance of composite structures and identify deficiencies. Research, development, design methods, and design criteria were addressed. Actions to eliminate technology deficiencies were developed. A list of those actions and a list of attendees is also included.

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