WRDC-TR-89-3030
Volume II

DESIGN DEVELOPMENT AND DURABILITY VALIDATION OF POSTBUCKLED COMPOSITE AND METAL PANELS

VOLUME II - TEST RESULTS

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NOVEMBER 1989

Final Report for Period September 1984 - April 1989

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REPORT DOCUMENTATION PAGE

Form Approved OMB No. 0704-0188

1a. REPORT SECURITY CLASSIFICATION: UNCLASSIFIED	1b. RESTRICTIVE MARKINGS: N/A
2a. SECURITY CLASSIFICATION AUTHORITY: N/A 2b. DECLASSIFICATION/DOWNGRADING SCHEDULE: N/A	3. DISTRIBUTION/AVAILABILITY OF REPORT: APPROVED FOR PUBLIC RELEASE: DISTRIBUTION UNLIMITED
4. PERFORMING ORGANIZATION REPORT NUMBER(S): NOR 89-112	5. MONITORING ORGANIZATION REPORT NUMBER(S): WRDC-TR-89-3030, VOL. II
6a. NAME OF PERFORMING ORGANIZATION: NORTHROP CORPORATION; 6b. OFFICE SYMBOL (If applicable):	7a. NAME OF MONITORING ORGANIZATION: FLIGHT DYNAMICS LABORATORY (WRDC/FIBE) WRIGHT RESEARCH AND DEVELOPMENT CENTER
6c. ADDRESS (City, State, and ZIP Code): HAWTHORNE, CA 90250-3277	7b. ADDRESS (City, State, and ZIP Code): WPAFB, OH 45433-6533
8a. NAME OF FUNDING/SPONSORING ORGANIZATION: FLIGHT DYNAMICS LABORATORY; 8b. OFFICE SYMBOL (If applicable): WRDC/FIBE	9. PROCUREMENT INSTRUMENT IDENTIFICATION NUMBER: F33615-84-C-3220
8c. ADDRESS (City, State, and ZIP Code): WPAFB, OH 454?3-6553	10. SOURCE OF FUNDING NUMBERS

PROGRAM ELEMENT NO.	PROJECT NO.	TASK NO	WORK UNIT ACCESSION NO.
62201F	2401	01	80

11. TITLE (Include Security Classification)
DESIGN DEVELOPMENT AND DURABILITY VALIDATION OF POSTBUCKLED COMPOSITE AND METAL PANELS. VOLUME II - TEST RESULTS.

12. PERSONAL AUTHOR(S)
R. B. DEO, N. M. BHATIA

13a. TYPE OF REPORT	13b. TIME COVERED	14. DATE OF REPORT (Year, Month, Day)	15. PAGE COUNT
FINAL	FROM 9-84 TO 4-89	NOVEMBER 1989	75

16. SUPPLEMENTARY NOTATION

17. COSATI CODES

FIELD	GROUP	SUB-GROUP
11	04	

18. SUBJECT TERMS (Continue on reverse if necessary and identify by block number)
BUCKLING, COMBINED LOADS, COMPOSITE PANELS, COMPRESSION, CURVED PANELS, DESIGN, DURABILITY, DAMAGE TOLERANCE, FATIGUE, IMPACT, METAL PANELS, POSTBUCKLING, REPAIR, SHEAR

19. ABSTRACT (Continue on reverse if necessary and identify by block number)
The objective of this program was to develop design procedures and durability validation methods for curved metal and composite panels designed to operate in the postbuckling range under the action of combined compression and shear loads. This research and technology effort was motivated by the need to develop design and life prediction methodologies for such structures.

The program was organized in four tasks. In Task I, Technology Assessment, a complete review of the available test data was conducted to establish the strength, durability, and damage tolerance characteristics of postbuckled metal and composite panels and to identify data gaps that need to be filled. Task II, Data Base Development, was comprised of static and fatigue tests required to fill in the data gaps identified in Task I. New rigorous static analysis methods aimed at improving the accuracy of the existing semi-empirical analysis and life prediction techniques were developed in Task III. Task IV consisted of

20. DISTRIBUTION/AVAILABILITY OF ABSTRACT: ☒ UNCLASSIFIED/UNLIMITED ☐ SAME AS RPT. ☐ DTIC USERS	21. ABSTRACT SECURITY CLASSIFICATION: UNCLASSIFIED
22a. NAME OF RESPONSIBLE INDIVIDUAL: GEORGE P. SENDECKYJ	22b. TELEPHONE (Include Area Code): (513) 255-6104; 22c. OFFICE SYMBOL: WRDC/FIBE

UNCLASSIFIED

19. ABSTRACT (Continued)

Technology consolidation where the results of this program were incorporated in the Preliminary Design Guide developed under Contract F33615-81-C-3208 to provide a comprehensive design guide for postbuckled aircraft structures. The comprehensive design guide was also exercised in this task, on an actual aircraft fuselage section to illustrate the methodology and demonstrate weight and cost trade-offs.

This final report consists of the following five volumes:

Volume I - Executive Summary

Volume II - Test Results

Volume III - Analysis and Test Results

Volume IV - Design Guide Update

Volume V - Automated Data Systems Documentation

PREFACE

The work documented in this report was performed by Northrop Corporation, Aircraft Division, Hawthorne, California, under Contract F33615-84-C-3220 sponsored by the Air Force Wright Aeronautical Laboratories, Flight Dynamics Laboratory, WRDC/FIBE. The work was performed in the period from September 1984 through April 1989. The Air Force Program Monitor was Dr. G. P. Sendeckyj.

The following Northrop personnel contributed to the performance of the contract in their respective areas of responsibility:

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TABLE OF CONTENTS

SECTION		PAGE
1.	INTRODUCTION	1
2.	DESIGN OF TEST PANELS	3
	2.1 DESIGN CRITERIA	3
	2.2 ANALYSIS OF CURVED PANELS FOR COMBINED LOADING	5
	2.2.1 Test Panel Configurations	5
	2.2.2 Composite Panel Analysis Under Shear Loads	7
	2.2.3 Composite Panel Analysis Under Compression Loads	8
	2.2.4 Combined Loading Interaction Curves for Composite Panels	9
	2.2.5 Metal Panel Analysis Under Shear Loads	12
	2.2.6 Metal Panel Analysis Under Compression Loads	12
3.	TEST PLAN	17
	3.1 STATIC TESTING	17
	3.2 FATIGUE TESTING	20
4.	TEST RESULTS	23
	4.1 TEST PROCEDURES	23
	4.2 COMPOSITE PANEL STATIC TEST DATA	23
	4.3 METAL PANEL STATIC TEST DATA	23
	4.4 COMPOSITE PANEL FATIGUE TEST DATA	24
	4.5 METAL PANEL FATIGUE TEST DATA	24
	REFERENCES	49
	APPENDIX A - COMPOSITE PANEL ANALYSIS FOR SHEAR LOADS	51
	APPENDIX B - COMPOSITE PANEL ANALYSIS FOR COMPRESSION LOADS	61
	APPENDIX C - METAL PANEL ANALYSIS FOR SHEAR LOADS	65

LIST OF ILLUSTRATIONS

FIGURE		PAGE
1	Composite Test Panel Configuration	6
2	7075-T6 Aluminum Test Panel Configuration. Stringers are AND10138-1206 Z-Sections. Rings are AND10138-1306 Z-Sections	7
3	Initial Buckling Interaction Curve for Composite Panels	10
4	Combined Loading Failure Envelope for Composite Panels	11
5	Initial Buckling Interaction Curve for Metal Panels	15
6	Failure Envelope for Metal Panels	16
7	Strain Gage Layout and LVDT Locations for Static Test Panels	19
8	Strain Gage Layout for Fatigue Test Panels	21
9	Photograph of the Test Fixture	22

LIST OF TABLES

TABLE		PAGE
1	Range of Compression and Shear Loads in Realistic Aircraft Fuselage Panels	4
2	Graphite/Epoxy Material Properties	5
3	Program Test Matrix (RTD Environment)	18
4	Test Data Summary for Composite Panel GR-1	25
5	Test Data Summary for Composite Panel GR-2	26
6	Test Data Summary for Composite Panel GR-3	27
7	Test Data Summary for Composite Panel GR-4	28
8	Test Data Summary for Composite Panel GR-5	29
9	Test Data Summary for Composite Panel GR-6	30
10	Test Data Summary for Composite Panel GR-7	31
11	Test Data Summary for Composite Panel GR-8	32
12	Test Data Summary for Panel AL-1	33
13	Test Data Summary for Panel AL-2	34
14	Test Data Summary for Panel AL-3	35
15	Test Data Summary for Panel AL-4	36
16	Test Data Summary for Panel AL-5	37
17	Test Data Summary for Panel AL-6	38
18	Test Data Summary for Panel AL-7	39
19	Test Data Summary for Panel AL-8	40
20	Test Data Summary for Panel AL-9	41
21	Test Data Summary for Panel AL-10	42
22	Test Data Summary for Panel AL-11	43

LIST OF TABLES (Continued)

TABLE		PAGE
23	Test Data Summary for Panel AL-12	44
24	Fatigue Failure Modes for Composite Panels	45
25	Fatigue Failure Modes for Aluminum Panels	46
25	Fatigue Failure Modes for Aluminum Panels (Cont'd.)	47

SECTION 1

INTRODUCTION

The overall objective of the test program was to develop a data base on the static and fatigue behavior of metal and composite panels loaded under uniaxial compression and shear well into the postbuckling range. The test program was tailored to fill in the data gaps that were identified in the technology assessments conducted in References 1 and 2. The specific objectives of the static tests were to develop static strength and displacement data for verification of the semiempirical design methodology and the energy method based analysis described in Reference 3. The fatigue test objectives were to obtain applied load versus life and failure mode data for use in formulating fatigue analyses for postbuckled metal and composite panels under combined loads.

Selection of the test specimen configuration, and the design criteria were based on the geometric and loading conditions encountered in actual aircraft fuselage construction. The following Sections detail the design of the test specimens, a rationale for the selected test matrix and the test data obtained. Correlation of the test results with analyses is given in Reference 3.

SECTION 2

DESIGN OF TEST PANELS

The composite and metal curved panels tested in this program were identical in configuration to the shear panels tested in Reference 2. Their design for combined shear and compression loading was rechecked, however, in light of the design criteria outlined below. The purpose of these calculations was to determine if any changes to the web or the stiffener geometries were necessary to avoid negative margins with respect to the design loads selected. The panels have a radius of curvature of 45 inches which is representative of an aircraft aft fuselage component. The metal and composite panels are both designed to satisfy the same design criteria so that their relative efficiencies could be compared.

2.1 DESIGN CRITERIA

Typical compression and shear loads on postbuckled components of an aircraft fuselage structure can have a relatively wide range of values depending on the panel location and type of aircraft. In order to establish the range of compression and shear loads encountered in fuselage design practice, load distributions for several realistic aircraft fuselage structures were surveyed. Table 1 shows the maximum load intensities under compression and shear that can be expected in realistic aircraft fuselage panels. It should be noted that the maximum compression and shear loads shown in Table 1 do not occur in combination. The maximum compression load is generally seen by lower fuselage panels where the shear loads are considerably lower than the maximum value. The maximum shear loading, on the other hand, acts on fuselage side panels were the compression loads due to fuselage bending are considerably smaller than the maximum value. It is evident from Table 1 that the maximum compression load intensities range between 1,000 lb/in and 2,000 lb/in,

TABLE 1. RANGE OF COMPRESSION AND SHEAR LOADS IN REALISTIC AIRCRAFT FUSELAGE PANELS.

AIRCRAFT	PANEL RADIUS OF CURVATURE	PANEL LOADS, LB/IN *	
		MAXIMUM AXIAL, N_x	MAXIMUM SHEAR, N_{xy}
F-15 +	> 40 in.	- 2000	700
F/A-18 +	> 40 in.	- 1800	700
Twin Engine + Supersonic V/STOL	> 40 in.	- 1000	700
Twin Engine Subsonic V/STOL	> 40 in.	- 1000	450

* Loads do not act simultaneously

\+ Reference 4

whereas the maximum shear load intensities range between 400 lb/in to 700 lb/in with the worst case combination determined by panel location on the fuselage. Postbuckled designs at the higher end of this load range were investigated in a Navy sponsored program (Reference 4) where the F/A-18 fuselage maximum loads have been used for test panel design. These design loads correspond to panels located in the lower fuselage section. In order to expand the combined loading design data base, the test panels in this program are designed to investigate the other extreme of the load range shown in Table 1. The design ultimate loads selected are based on side panel loading conditions for a supersonic fighter aircraft. The maximum compression load intensity for these panels is 1,000 lb/in and the maximum shear load intensity is 700 lb/in. Thus, for the program test panels a limit load intensity of 660 lb/in in compression and 600 lb/in in shear was selected as the design goal. The initial buckling load requirement for the panel skins was set at approximately 30 percent of the design limit load, with no rupture or collapse of the panel to occur prior to the design ultimate load (1.5 times the design limit load).

Analysis of the Reference 2 shear panels for the above design criteria was performed and is detailed in the following paragraphs.

2.2 ANALYSIS OF CURVED PANELS FOR COMBINED LOADING

The composite test panel configuration is shown in Figure 1 and Figure 2 shows the metal panel configuration. These panels were analyzed for initial buckling loads and ultimate strengths for various combinations of compression and shear loads. The semi-empirical analysis methodology developed in Reference 2 for panels loaded in shear or compression only in conjunction with interaction rules available from combined load data generated in Reference 4 were used for the analysis.

2.2.1 Test Panel Configurations

The composite panel, Figure 1, consisted of three cocured hat stiffeners and two cocured J-section frames. The panel edges were thickened for load introduction purposes. The panel configuration gives two identical test bays in addition to load introduction bays so that postbuckling deformation can fully develop without undue restraints. The materials used in panel fabrication were Hercules AS4/3501-6 graphite/epoxy tape and A370-5H/3501-6 biwoven graphite/epoxy. The mechanical properties for these materials are given in Table 2. The fabrication procedures for the composite panels are identical to those well established in Reference 2.

TABLE 2. GRAPHITE/EPOXY MATERIAL PROPERTIES

PROPERTY	AS/3501-6	A370-5H/3501-6 (FABRIC)
E_1^c, psi	18.7×10^6	10.0×10^6
E_2^c, psi	1.87×10^6	9.2×10^6
G_{12}, psi	0.85×10^6	0.9×10^6
ν_{12}	0.3	0.055

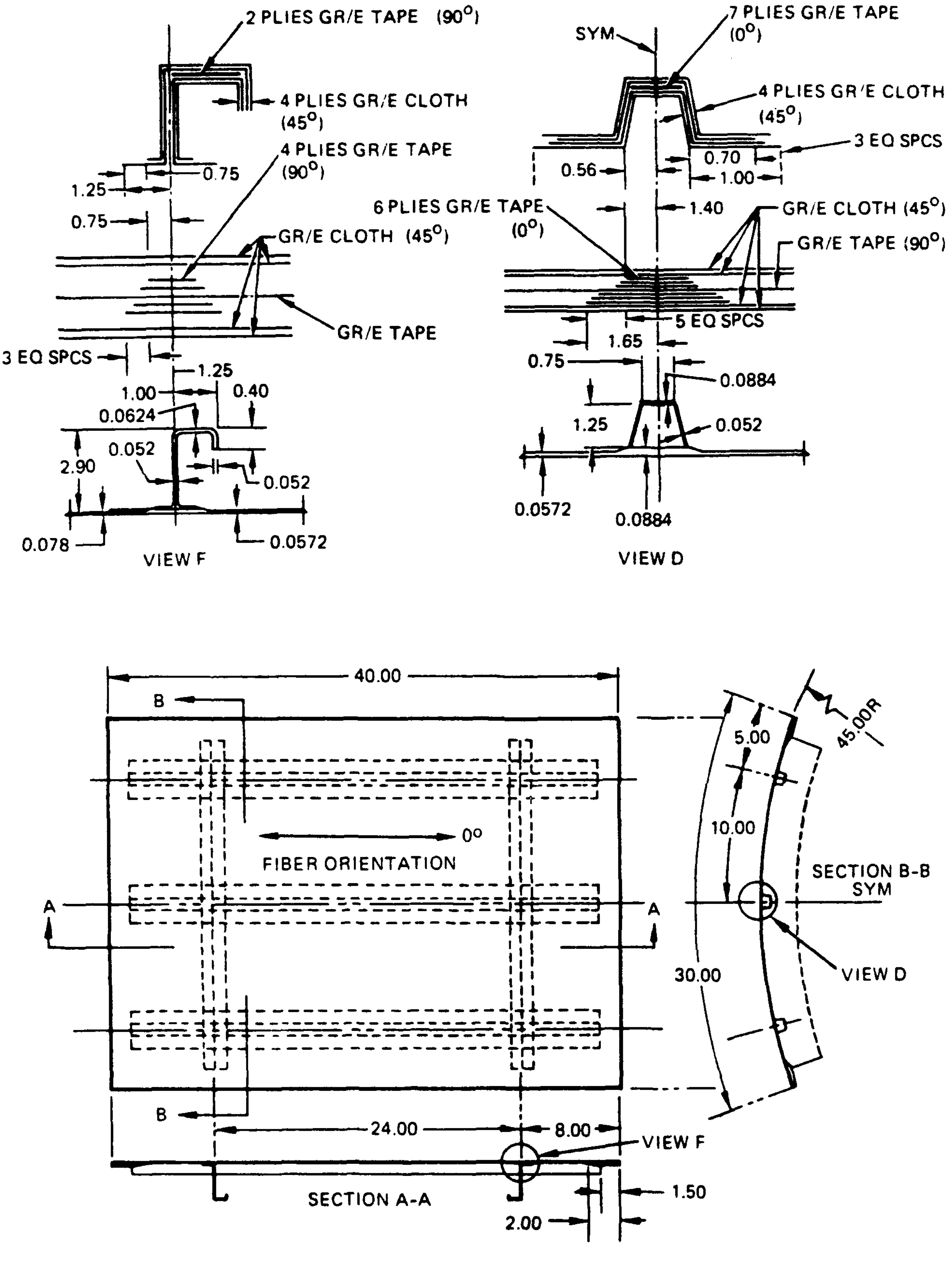


Figure 1. Composite Test Panel Configuration

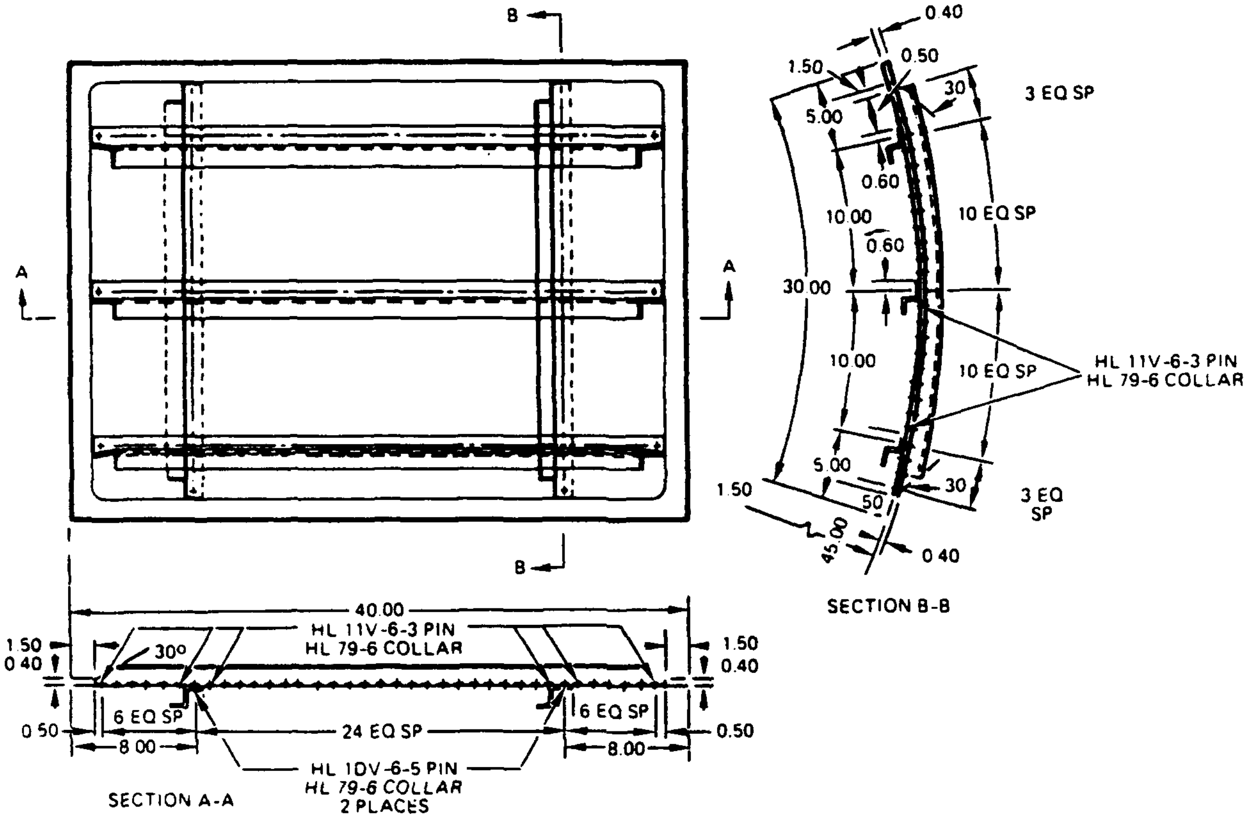


Figure 2. 7075-T6 Aluminum Test Panel Configuration. Stringers are AND10138-1206 Z-Sections. Rings are AND10138-1306 Z Sections.

The metal panels, Figure 2, are stiffened with bolted Z-section stringers and frames. The panel edges were reinforced with a bonded doubler for load introduction purposes. As in the case of composite panels, the metal panel configuration provides two identical test bays. The metal Z-section edges are rounded to avoid fatigue crack initiation in the skins due to the sharp stiffener flange corner bearing on the skin.

2.2.2 Composite Panel Analysis Under Shear Loads

Analysis of the composite panel under shear loading alone was performed in Reference 2 using program TENWEB. The analysis details are given in Reference 2 and the program run is included in Appendix A. The salient features of panel analysis under shear loading are given below.

Initial Buckling Analysis. The composite panel web is 5 plies thick with a ($\underline{45}_2$/90/$\underline{45}_2$) layup, where the underscore denotes woven reinforcement. A buckling analysis of the panel using program BUCLASP-2 showed that for a stiffener spacing of 10 inches and a frame spacing of 24 inches, the initial shear buckling load of the web $N^{o}_{xy,cr}$, was 200 lb/in for boundary conditions intermediate between simply supported and fixed. These calculations are documented in Reference 2.

The actual measured average shear buckling strain for the skin was $\gamma_{xy,cr}$ = 1184 μin/in. This translates to a buckling shear flow of 284 lb/in. Thus, the initial buckling shear flow used in the present calculations with program TENWEB was 284 lb/in.

Failure Analysis. Failure analysis of the panel was performed using the modified tension field theory and semi-empirical failure criteria given in Reference 2. The compression failure strain value used in the calculations was ϵ_{cu} = 0.015. As shown in Appendix A, for the panel and stiffener configuration selected, an ultimate shear flow of N^{ult}_{xy} = 850 lb/in gives a minimal margin of safety of 1.0 percent on the ring. Thus, the predicted failure load for the panel ranges between 850 lb/in and 875 lb/in. This load corresponds to a design limit shear flow of between 570-584 lb/in. The predicted failure mode is forced crippling of the ring. It was demonstrated in Reference 2 that such a failure mode results in separation of skin and the ring, and the forced crippling strain, therefore, is also a measure of the latter failure strain. The predicted angle of diagonal tension is 39.7°.

2.2.3 Composite Panel Analysis Under Compression Loads

Program CRIP developed in Reference 2 was used to analyze the panel shown in Figure 1 for compression loading. In addition, to analyze for the stiffener web separation mode of failure, the semi-empirical criteria developed in Reference 2 was used.

A computer run of CRIP for the panel configuration given in Figure 1 is given in Appendix B. The salient features of the results are summarized below.

Initial Buckling Analysis. The initial buckling strain calculated for the skin using program SS8 was $\epsilon_{x,cr} = 570\ \mu in/in$ which corresponds to a running axial load $N^{o}_{x,cr} = 264$ lb/in.

Failure Analysis. The compression failure modes analyzed for were Euler buckling of the panel as a whole, stiffener crippling and stiffener/web separation. The Euler buckling strain for the panel calculated in program CRIP was $\epsilon^{E} = 0.0166$ in/in corresponding to a running load $N^{E}_{x} = 2453$ lb/in.

The total load at stiffener crippling, P^{cc}, was 42,680 lb corresponding to a running load $N^{cc}_{x} = 1,423$ lb/in for the 30-inch wide panel.

In order to calculate the stiffener/web separation failure load, the following equation was used

$$\epsilon_{ss} = 0.4498\ \epsilon^{sk}_{cr} \left(\frac{\epsilon_{cu}}{\epsilon^{sk}_{cr}}\right)^{0.72715}$$

where, ϵ_{ss} is the stiffener/web separation strain, $\epsilon_{cu} = 0.015$ is the compression ultimate strain, and $\epsilon^{sk}_{cr} = 0.00056$ is the skin buckling strain. The calculated stiffener/web separation strain ϵ_{ss} was 0.00277 in/in. The corresponding running load for stiffener/web separation was $N_{x,ss} = 1288$ lb/in.

Thus, the predicted failure mode for the panel under compression loading alone was stiffener/web separation.

2.2.4 Combined Loading Interaction Curves for Composite Panels

The initial buckling interaction curve and failure envelope for the composite test panel are shown in Figures 3 and 4, respectively. For the initial buckling interaction curve a parabolic law was assumed. Buckling loads under combined loading were also calculated using program SS8 (Reference 5) and the resulting interaction curve is shown in Figure 3 for comparison.

In the final failure of the panel, the critical modes are stiffener and ring forced crippling under shear loading and stiffener web separation and stiffener crippling under compression loading. Under combined loading the axial strains in the stiffener due to the shear and compression loads add up and the total stiffener load is that caused by the total strain. In terms of running loads N_x and N_{xy}, the total stiffener load at crippling is given by:

$$P^{cc} = (EA)_s \epsilon_s$$

where $(EA)_s$ is the stiffener axial stiffness and ϵ_s is the total strain in the stiffener obtained from Equation 37 in Reference 6 and is expressed as:

$$\epsilon_s = \frac{-N_x h_s}{\left[(EA)_s + w\, t_w E_{ws}\right]} - \frac{k\, N_{xy} \cot\alpha}{t_w\left[\frac{(EA)_s}{h_s t_w} + 0.5(1-k)E_{ws}R_s\right]}$$

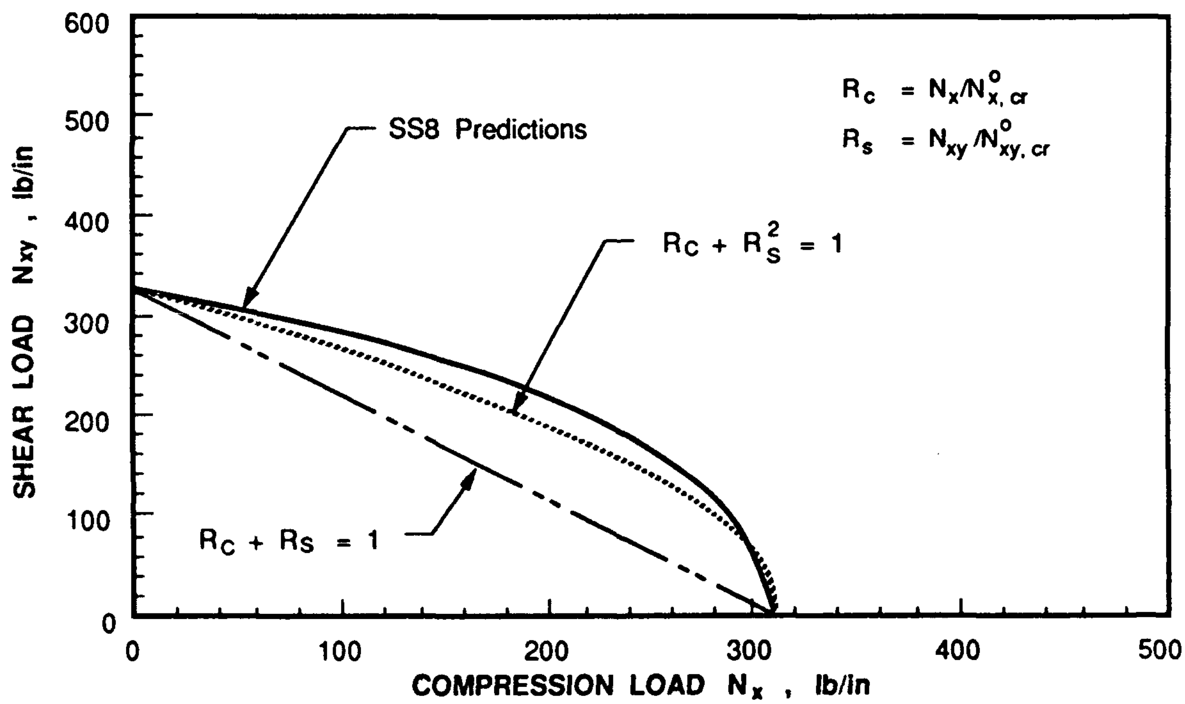


Figure 3. Initial Buckling Interaction Curve for Composite Panels.

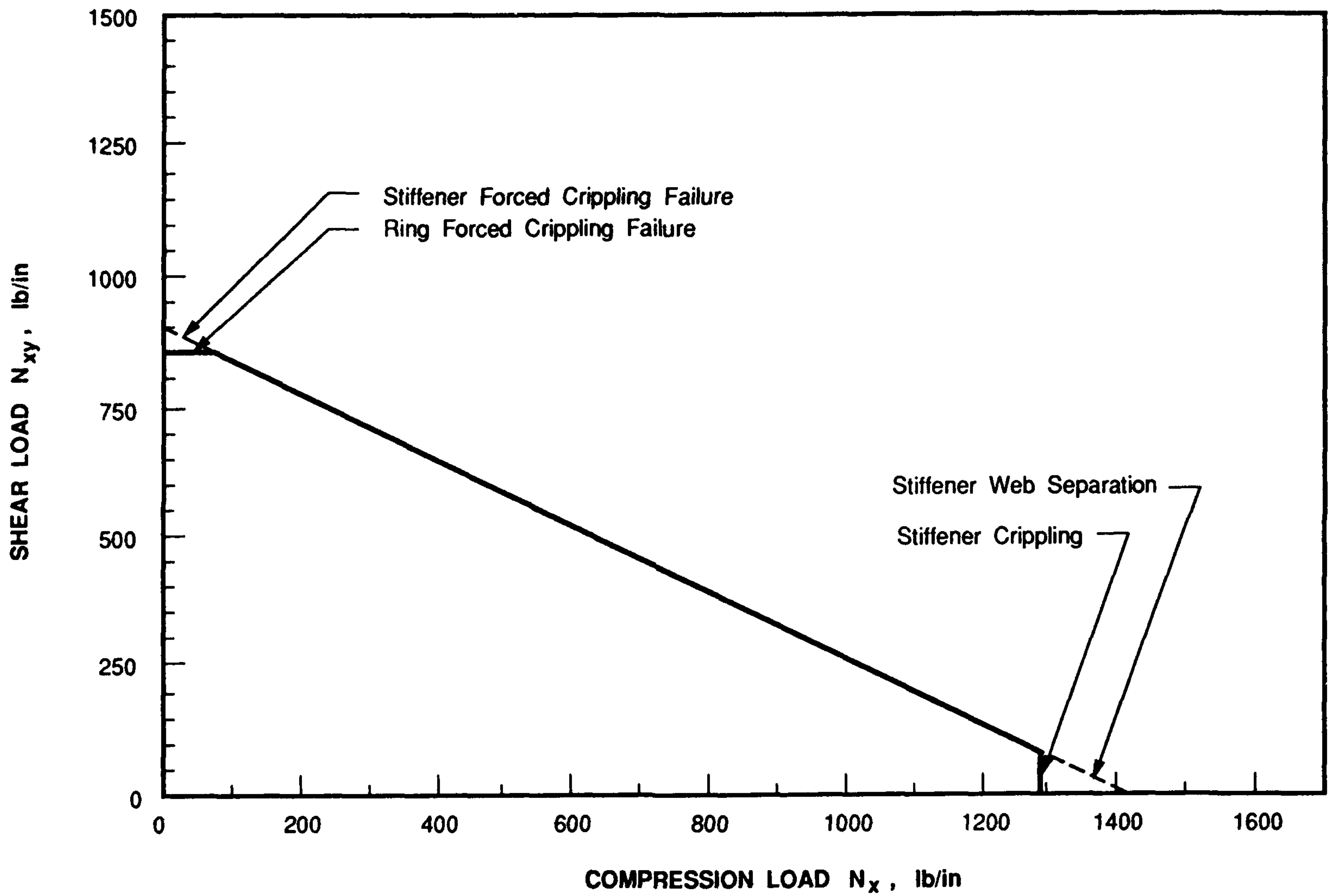


Figure 4. Combined Loading Failure Envelope for Composite Panels.

This expression, i.e., the linear superposition of the two strains, is reflected in the failure load interaction diagram of Figure 4. The combination of shear and compression loads required to cause failure by stiffener crippling is shown as a linear interaction in Figure 4. In the program test panels ring forced crippling is the predicted failure mode under shear loads, and stiffener/web separation is predicted to cause panel failure under compression loads. These modes are included in the interaction diagram of Figure 4 to define the composite test panel design envelope.

2.2.5 Metal Panel Analysis Under Shear Loads

The metal panels were analyzed for shear loading using program TENWEB. The program run for this analysis is given in Appendix C. The results of this analysis are summarized below.

Initial Buckling Analysis. Initial shear buckling load for the metal panel was calculated using Figure C9.4 of Reference 7 as $N^{o}_{xy,cr} = 446$ lb/in. This value was compared with actual test data obtained for these panels in Reference 2. The average measured initial buckling load was lower than the calculated value and equaled 325 lb/in. Thus, for the metal shear panel analysis $N^{o}_{xy,cr} = 325$ lb/in was used.

Failure Analysis. The predicted failure mode from program TENWEB based on the tension field theory was forced crippling of the stiffener. The predicted failure load was approximately 875 lb/in. The predicted angle of diagonal tension was 40°.

2.2.6 Metal Panel Analysis Under Compression Loads

Analysis of the metal panel under compression loading was accomplished using the classical methods documented in Reference 7. A summary of the analysis is presented in the following paragraphs.

Initial Buckling Analysis. The initial buckling load for the metal panel under compression loading was calculated using the following parameters:

Stiffener spacing	b_s	=	10 inches
Web thickness	t_w	=	0.063 inch
Panel length	L	=	24 inches

For local buckling of web in between stiffeners, the buckling stress F_{cr} for a curved panel in compression is given in Reference 7 as:

$$F_{cr} = \frac{K_c \pi^2 E}{(12(1-\nu^2)} \frac{(t_w)^2}{b_s}$$

where

K_c = 12 (obtained from Figure C9.1 in Reference 7)

F_{cr} = 4606 psi

The stiffener area A_s = 0.338 inch and the corresponding running load is given by:

$$N^o_{x,cr} = F_{cr} \cdot t_{eq} = \frac{A_s + b_s t_w}{b_s} \cdot F_{cr} = 446 \text{ lb/in}$$

Failure Analysis. The panel failure modes interrogated in the analysis were Euler buckling of the panel and stiffener crippling. The stiffener crippling stress F_{cs} was calculated using Figure C7.9 of Reference 7 as 52 ksi. The effective web width at the time of stiffener crippling, w, was obtained from.

$$w = 1.9 t_w \sqrt{\frac{E}{F_{cs}}} \quad \text{(Reference 7)}$$

$$= 1.72 \text{ inches}$$

Thus, the total load at panel failure P_{ult} was

$$P_{ult} = F_{cs} (A_s + w \cdot t_w) = 52,000 (0.338 + 1.72 \times 0.063)$$

$$= 23,210 \text{ lb}$$

ence, the ultimate failure load per unit width N_x^{ult} is:

$$N_x^{ult} = \frac{P_{ult}}{b_s} = 2321 \text{ lb/in}$$

The Euler buckling stress was calculated using

$$F_{cr} = \frac{\pi^2 E I_e}{L_e^2 A_t}$$

where, L_e is the effective length of the panel, A_t is the total area of the panel and I_e is the panel moment of inertia about the neutral axis.

Because the frame spacing for design purposes is assumed to be 24 inches, the effective length "L" for Euler buckling is 12 inches assuming fully fixed ends. Thus, calculated Euler buckling stress of the panel is:

$$F_{cr}^{E} = 232 \text{ ksi}$$

The actual Euler buckling stress will be lower than the value determined above due to plasticity effects. However, this stress is considerably in excess of the stiffener crippling stress and, therefore, the predicted failure mode is stiffener crippling.

<u>Combined Loading Interaction Curves for Metal Panel</u>. The initial buckling load interaction curve for the metal panel was obtained using a parabolic law and is shown in Figure 5.

The failure load interaction diagram for the metal panel is shown in Figure 6 and was obtained assuming a linear interaction for stiffener crippling.

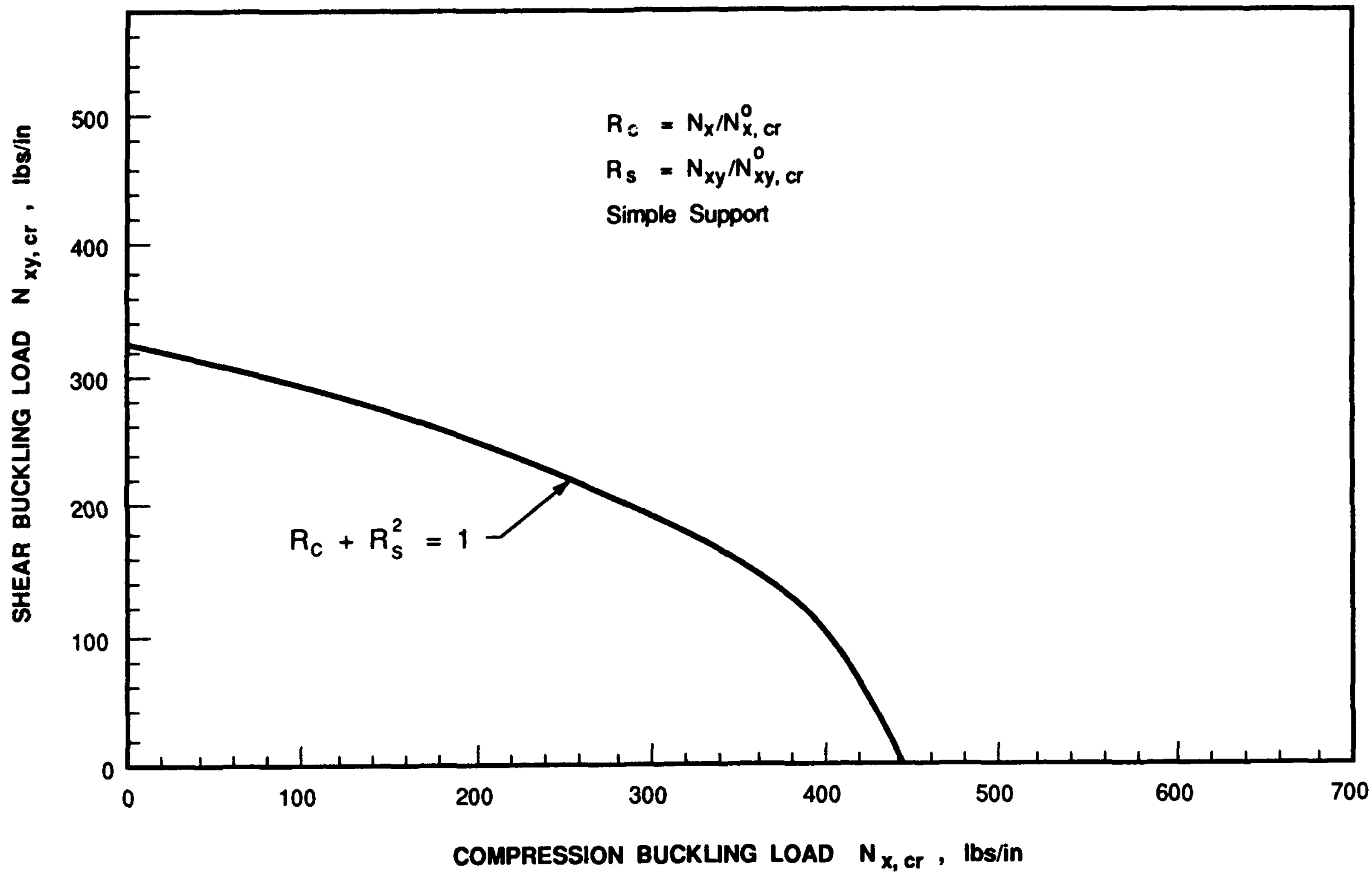


Figure 5. Initial Buckling Interaction Curve for Metal Panels.

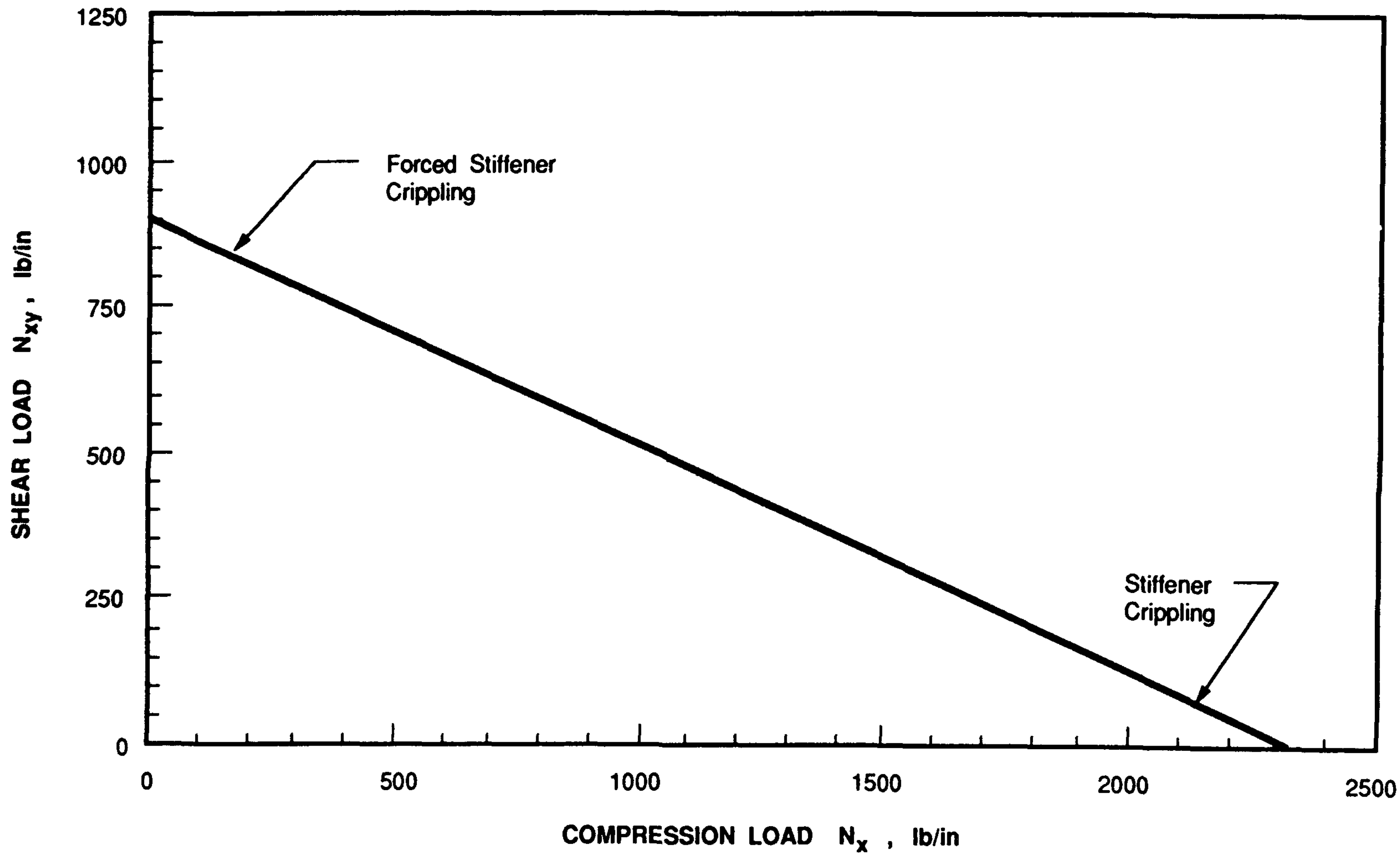


Figure 6. Failure Envelope for Metal Panels.

SECTION 3

TEST PLAN

The selection of a suitable test matrix for the program was made after a careful examination of the existing experimental data and the overall objectives of the program. As mentioned in Section 1, curved metal and composite panels subjected to combined loads were tested in this program. The overall test matrix is shown in Table 3. These tests were intended to provide data for verification of semi-empirical and non-empirical analysis methodologies, and for metal panel fatigue life prediction methodology development.

A total of 20 panel tests (8 composite panels and 12 metal panels) were conducted. Greater emphasis was placed on testing metal panels since no experimental data existed for such panels subjected to fatigue loading. Each test condition is replicated at least twice to demonstrate repeatability of the test and to obtain more reliable test data for analysis verification. Two values of the axial compression to shear load ratios N_x/N_{xy} were investigated in the static tests to establish validity of the analysis procedures in a wide range of load ratios. For the fatigue tests, the load ratio was variable and the two load amplitudes were keyed to their respective R-ratios and the panel static strength.

3.1 STATIC TESTING

The composite and metal panel static tests were conducted to determine the initial buckling and postbuckling combined loading interaction curves, and the effect of order of shear and compression load application on initial buckling and postbuckling strength. Tests conducted in a previous program (Reference 2) provided the initial buckling and failure load data under shear loading only, thus facilitating development of the interaction curves.

TABLE 3. PROGRAM TEST MATRIX (RTD ENVIRONMENT).

PANEL NO.	MATERIAL	TYPE OF TEST	LOAD RATIO N_x/N_{xy}	R-RATIO * FOR FATIGUE TESTS	MAXIMUM FATIGUE LOAD, % STATIC STRENGTH	STATIC STRAIN SURVEYS TO INITIAL BUCKLING (N_x/N_{xy})
GR1 GR2	AS4 AND A370-5H/ 3501-6 GRAPHITE-EPOXY	STATIC	2.0	--	--	0, 0.5, 1.0 ∞, 2.0
GR3 GR4		STATIC	0.5	--	--	0, 0.5, 1.0, ∞, 2.0
GR5 GR6		FATIGUE	-- --	R_x = 10, R_{xy} = -1.0	70	0, 0.5, 1.0, 2.0
GR7 GR8		FATIGUE	-- --	R_x = 10, R_{xy} = -1.0	70	0, 0.5, 1.0, 2.0
AL-1 AL-2	7075-T6 ALUMINUM	STATIC	0.5 2.0	-- --	-- --	0, 0.5, 1.0, ∞, 2.0
AL-3 AL-4		STATIC	2.0 0.5	-- --	-- --	0, 0.5, 1.0, ∞, 2.0
AL-5 AL-6		FATIGUE	-- --	R_x = 10, R_{xy} = -1.0	66	0, 0.5, 1.0, 2.0
AL-7 AL-8		FATIGUE	-- --	R_x = 10, R_{xy} = -1.0	54	0, 0.5, 1.0, 2.0
AL-9 AL-10		FATIGUE	-- --	R_x = 10, R_{xy} = -1.0	50	0, 0.5, 1.0, 2.0
AL-11 AL-12		FATIGUE	-- --	R_x = 10, R_{xy} = -1.0	60	0, 0.5, 1.0, 2.0

* R_x DENOTES R-RATIO FOR COMPRESSION LOAD

R_{xy} DENOTES R-RATIO FOR SHEAR LOAD

All static test panels were instrumented with back-to-back strain gages to determine the strain distribution within the panels. The back-to-back gages permitted determination of the bending and membrane strains. The static test specimens, were extensively instrumented to obtain strain distribution throughout the panels for non-empirical analysis verification. The strain gage layout for these panels is shown in Figure 7. In order to monitor the out-of-plane displacements, transducers at locations shown in Figure 7 were also utilized. In addition to the strain gages and the displacement transducers, the Moire' fringe technique was used to monitor the buckle patterns. The initial buckling load was obtained using the appropriate back-to-back strain gage data. All static tests to failure were preceded by strain surveys to initial buckling loads for a range of N_x/N_{xy} load ratios.

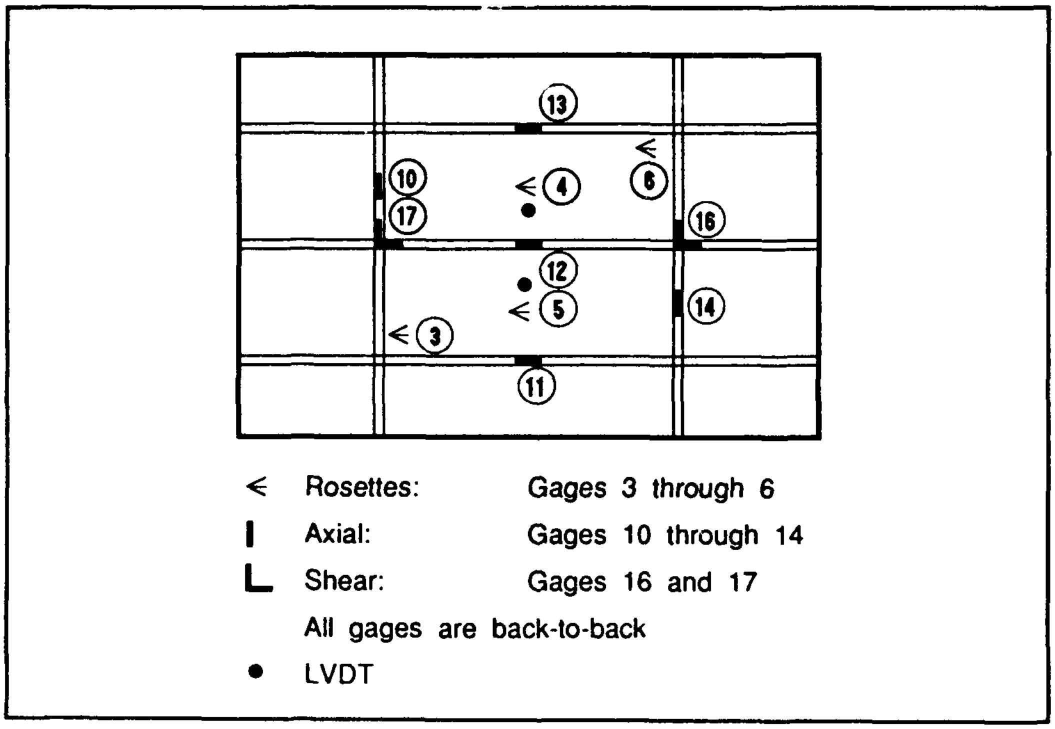


Figure 7. Strain Gage Layout and LVDT Locations for Static Test Panels.

3.2 FATIGUE TESTING

Four composite panels and eight metal panels were tested under constant amplitude fatigue loading. The maximum fatigue loads as a percentage of the static strength are given in Table 3. All panels were subjected to a maximum of 100,000 cycles of constant amplitude fatigue loading. The panels were inspected periodically to determine the change in the initial buckling load and map any damage or fatigue crack growth. The compression loading was applied at an R-ratio of 10, whereas, the shear loading was fully reversed (R = -1). The metal and composite fatigue test panels were instrumented as shown in Figure 8. Static strain surveys were conducted during the course of the fatigue tests to determine the influence of cyclic loading on the initial buckling loads. The panels that survived the 100,000 cycle fatigue test were residual strength tested.

The composite and metal panels were tested in a specially designed combined loading test fixture. The loading concept to introduce combined shear and axial loads in cylindrically curved panels was developed and verified under Northrop's IR&D plans. A photograph of the test fixture is shown in Figure 9. The fixture consists of a triangular cross-section hollow tube. Two sides of this tube are flat "dummy" panels and the third side houses the test panel. The shear load is introduced by twisting the tube between flat, parallel platens, and the axial load by axial displacement of these platens.

This test fixture was built and an aluminum panel tested to verify the accuracy of the combined loading concept experimentally. The test panel was identical to the metal shear panels tested under Air Force Contract F33615-81-C-3208, Reference 2. The test results showed no interaction between shear and compression loading.

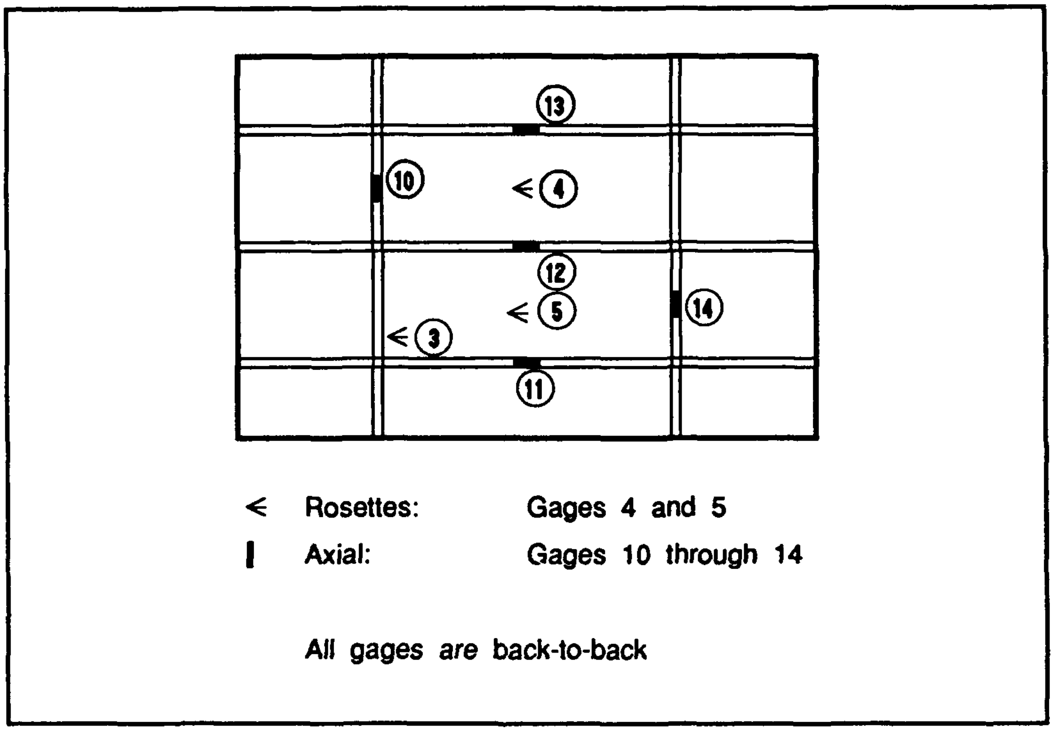


Figure 8. Strain Gage Layout for Fatigue Test Panels.

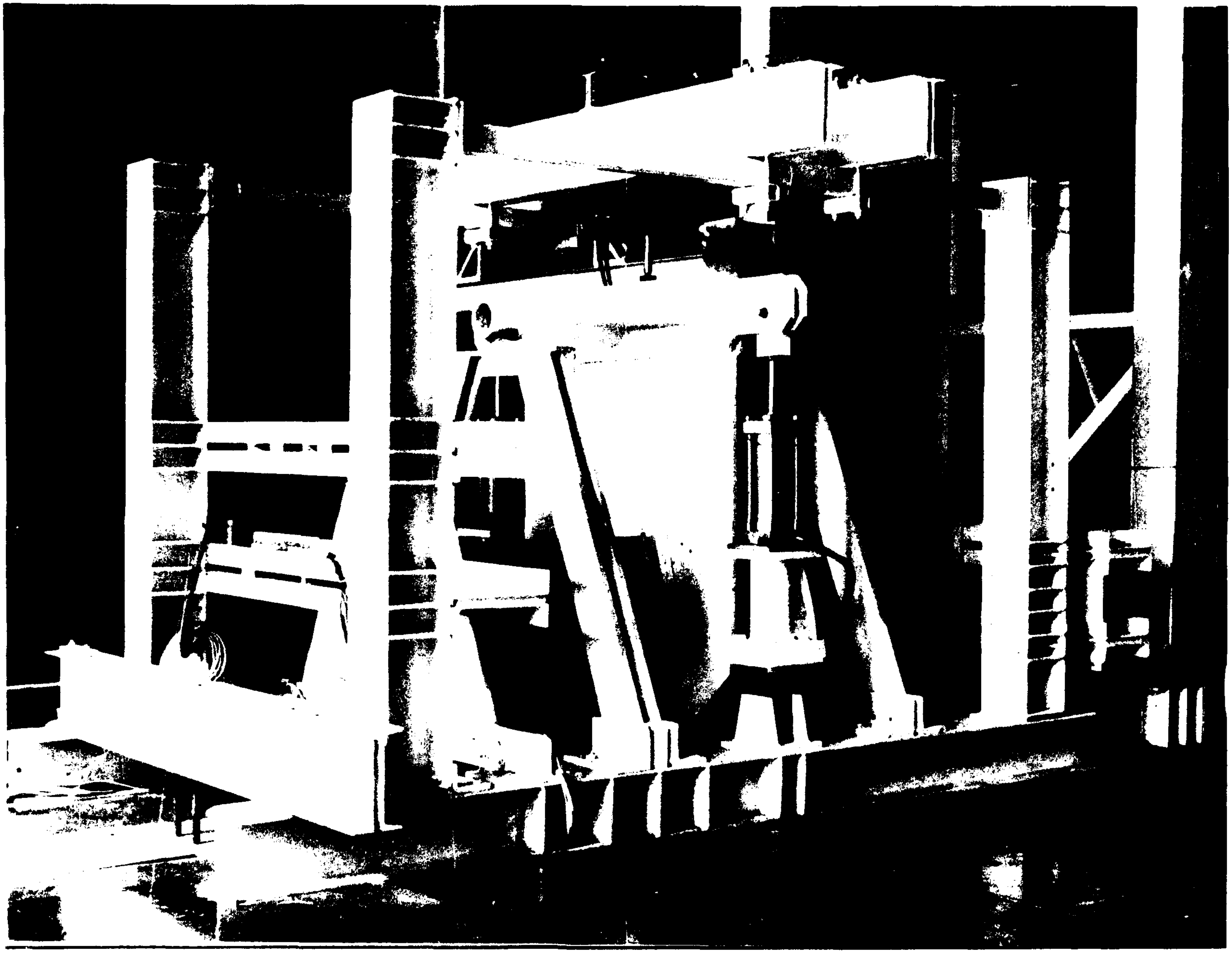


Figure 9. Photograph of the Test Fixture.

SECTION 4

TEST RESULTS

4.1 TEST PROCEDURES

As a first step in all the static and fatigue tests, a pair of calibration runs were conducted to establish the actuator load to panel running load conversion factors. One of the calibration runs was under uniaxial compression alone and the other under torque loading only. Multiple strain surveys up to a load level just beyond the skin buckling load were conducted on all test articles. On the fatigue test panels additional strain surveys were conducted after every 25K to 30K cycles of loading. The surviving fatigue specimens were residual strength tested at a load ratio equal to the fatigue test load ratio. A majority of the static and fatigue tests were videotaped to document buckling mode shapes and changes mode shapes in the postbuckling load range.

4.2 COMPOSITE PANEL STATIC TEST DATA

Static strength and static strain survey data for composite panels GR-1 through GR-8 are summarized in Tables 4 through 11. The static test results are compared with predictions in Reference 3. The failure modes for the static test articles are described in the respective data tables, i.e., Tables 4 through 7. All static tested composite panels failed by separation of the skin from the stiffeners at the intersection of the stringers and frames.

4.3 METAL PANEL STATIC TEST DATA

Data for all metal panel static tests are summarized in Tables 12 through 23. The static test results are compared with predictions in Reference 3. The failure modes for metal panels tested for static strength are described in Tables 12 through 15. The metal panel failure mode under compression dominated loading (N_x/N_{xy} = 2.0) was by stiffener crippling, whereas for the shear dominated loading case (N_x/N_{xy} = 0.5) permanent set in the skin concurrent with stiffener crippling was the observed failure mode.

4.4 COMPOSITE PANEL FATIGUE TEST DATA

The composite panel fatigue test data are summarized in Table 24. The two panels tested at N_x/N_{xy} ratio of 2 experienced no fatigue failure after 100,000 cycles of constant amplitude fatigue loading. The residual static strength data for these panels are included in Tables 8 and 9. The static failure mode was primarily skin-stiffener separation. Panels tested at N_x/N_{xy} ratio of 0.5 failed during fatigue cycling. The fatigue failure mode in these panels (GR-7 and GR-8) was by skin/stiffener separation accompanied by skin rupture at the outer corner where the stiffener intersects the frame.

4.5 METAL PANEL FATIGUE TEST DATA

The metal panel fatigue test data and the failure modes are summarized in Table 25. The dominant fatigue failure mode observed in these panels was independent of the N_x/N_{xy} ratio. The basic fatigue failure mode in the metal panel was crack initiation and subsequent crack propagation in the skin. The crack initiation site for panels tested at N_x/N_{xy} ratio of approximately 2 was at the junction of the skin and the stiffener. The crack propagated initially along the stiffener direction. After a certain length, the crack branched and grew toward the centerline of the bay in the diagonal direction. For panels tested at N_x/N_{xy} ratio of approximately 0.5, cracks initiated at the edges of the fastener holes in the skin. The subsequent crack growth pattern was similar to that of panels tested at N_x/N_{xy} ratio of 2.

TABLE 4. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-1.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS[5], GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	2.62	0	241	0	0	0.0906	246	0	0	0.0925
∞	③	Calibration	0	18.97	0	186	∞	0.0098	0	205	∞	0.0108
0	②	Buckling	2.93/3.33	0	296	0	0	0.0906	324	0	0	0.0925
0.5	①	Buckling	2.20/2.77	10.63	224	137	0.61	--	223	137	0.61	--
1.0	②	Buckling	1.91/2.33	7.56	218	97	0.35	--	279	98	0.35	--
2.0	③	Buckling	1.35/1.61	13.84	228	178	0.78	--	226	179	0.79	--
2.0	①	Buckling	1.61/1.59	19.38	216	250	1.15	--	214	250	1.17	--
2.0	①	Buckling	1.62	26.97	151	348	2.30	--	150	349	2.32	--
∞	①	Buckling	0	32.17	0	415	∞	0.0098	0	416		0.0108
2.0	①	Failure[6]	3.77	63.42	639	407	0.64	--	635	408	0.64	--

<u>NOTES:</u>

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay
   N_x = Et ϵ_{xy}, Average Compression Load in Bay
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
5. Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in.
6. Failure Mode: Stringer / Frame Separation from Skin at Intersection Accompanied by Skin Rupture

* See Figure 7 for Strain Gage Nomenclature

TABLE 5. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-2.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	3.25	0	356	0	0	0.1095	294	0	0	0.0905
∞	③	Calibration	0	22.6	0	252	∞	0.0111	0	235	∞	0.0104
0	②	Strain	3.6	0	394	0	0	0.1095	326	0	0	0.0905
0.5	①	Survey to	2.6	7.6	285	84	0.3	--	235	79.0	0.34	--
1.0	①	Buckling	2.2	13.1	241	145	0.6	--	199	136	0.68	--
2.0	①	↓	1.86	22.5	204	250	1.22	--	168	234	1.39	--
∞	③		0	26.4	0	293	∞	--	0	275	∞	--
2.0	①	Loaded to Failure	1.3	23.1	142	256	1.80	--	118	240	2.03	--

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{avg}, Average Shear Load in Bay
   N_x = Et ϵ_{avg}, Average Compression Load in Bay
   $\Delta N/\Delta P$ = Calibration Ratios of Load in Bay Per Applied Load
   Aluminum : $E_x = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
   Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in. (Et = 0.2019×10^6, Gt = 0.2402×10^6)

* See Figure 7 for Strain Gage Nomenclature

TABLE 6. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-3.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS[5], GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	3.35	0	296	0	0	0.08836	294	0	0	0.08776
∞	③	Calibration	0	28.17	0	363		0.01289	0	364		0.01292
0	②	Buckling	3.58	0	316	0	0	0.08836	314	0	0	0.0878
0.5	①	Buckling	2.54	10.63	224	137	0.61	--	223	137	0.61	--
0.5	②	Buckling	3.18	7.56	218	97	0.35	--	279	98	0.35	--
0.5	③	Buckling	2.58	13.84	228	178	0.78	--	226	179	0.79	--
1.0	①	Buckling	2.44	19.38	216	250	1.15	--	214	250	1.17	--
2.0	①	Buckling	1.71	26.97	151	348	2.30	--	150	349	2.32	--
∞	①	Buckling	0	32.17	0	415	∞	0.01289	0	416	∞	0.01292
0.5	①	Failure[6]	7.23	31.58	639	407	0.64	--	635	408	0.64	--

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay
   N_x = Et ϵ_{xy}, Average Compression Load in Bay
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
5. Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in.
6. Failure Mode: Stringer / Frame Separation from Skin at Intersection Accompanied by Skin Rupture

* See Figure 7 for Strain Gage Nomenclature

TABLE 7. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-4.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, kib	COMPRESSION[3] LOAD, P_c, kib	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	3.36	0	313	0	0	0.0931	317	0	0	0.0942
∞	③	Calibration	0	22.3	0	269	∞	0.0121	0	257	∞	0.0115
0	②	Buckling	3.7	0	344	0	0	0.0931	349	0	0	0.0942
1.0	①	Buckling	2.5	19.3	233	234	1.0	- -	236	222	0.94	- -
2.0	①	Buckling	1.8	27.1	168	328	1.95	- -	170	312	1.84	- -
∞	③	Buckling	0	33.0	0	399	∞	0.0121	0	380	∞	0.0115
0.5	①	Buckling	3.2	13.7	298	166	0.56	- -	301	158	0.52	- -
0.5	①	Failure	7.8	33.1	726	401	0.55	- -	735	381	0.52	- -

<u>NOTES :</u>

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{avg}, Average Shear Load in Bay
   N_x = Et ϵ_{avg}, Average Compression Load in Bay
   ΔN/ΔP = Calibration Ratios of Load in Bay Per Applied Load
   Aluminum : $E_x = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
   Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in. (Et = 0.2019×10^6, Gt = 0.2402×10^6)

* See Figure 7 for Strain Gage Nomenclature

TABLE 8. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-5.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR LOAD, P_t, kip [2]	COMPRESSION LOAD, P_c, kip [3]	UPPER BAY LOADS [4], GAGE 4*				LOWER BAY LOADS, GAGE 5			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	2.74	0	251	0	0	0.0916	241	0	0	0.088
∞	③	Calibration	0	31.9	0	325	∞	0.0102	0	355	∞	0.0111
0	②	Buckling	2.85	0	261	0	0	0.0916	251	0	0	0.088
0.5	①	Buckling	2.48	10.34	227	106	0.46	--	218	115	0.53	--
1.0	①	Buckling	1.87	15.86	171	162	0.95	--	165	176	1.07	--
2.0	①	Buckling	1.64	24.98	150	255	1.7	--	144	277	1.92	--
∞	③	Buckling	0	32.97	0	336	∞	0.0102	0	366	∞	0.0111
2.0	①	After 31 K_c Fatigue	1.46	25.0	134	255	1.9	--	128	278	2.17	--
2.0	①	After 50 K_c Fatigue	1.51	25.0	138	257	1.86	--	132	280	2.12	--
2.0	①	After 79.5 K_c Fatigue	1.47	25.1	135	256	1.9	--	129	279	2.16	--
0	②	After 100 K_c Fatigue	3.05		280	0	0	--	269	0	0	--
2.0	①	After 100 K_c Fatigue	1.65	25.2	151	257	1.7	--	145	280	1.93	--
2.0	①	After 100 K_c Fatigue to Failure	3.83	70.61	351	720	2.05	--	337	784	2.33	--

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = $Gt\,\gamma_{avg}$, Average Shear Load in Bay
   N_x = $Et\,\varepsilon_{avg}$, Average Compression Load in Bay
   ΔN/ΔP = Calibration Ratios of Load in Bay Per Applied Load
   Aluminum : $E_x = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
   Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in. ($Et = 0.2019 \times 10^6$, $Gt = 0.2402 \times 10^6$)

* See Figure 8 for Strain Gage Nomenclature

TABLE 9. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-6.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	2	Prebuckling	3.12	0	302	0	0	0.0967	267	0	0	0.0855
∞	3	Calibration	0	32.1	0	308	∞	0.0096	0	327	∞	0.0102
0	2	Buckling	3.33	0	322	0	0	0.0967	285	0	0	0.0855
0.5	1	Buckling	2.75	11.31	266	109	0.41	--	235	115	0.49	--
1.0	1	Buckling	2.33	19.12	225	184	0.82	--	199	195	0.98	--
2.0	1	Buckling	1.92	29.7	186	285	1.53	--	164	303	1.85	--
∞	3	Buckling	0	35.4	0	340	∞	--	0	361	∞	--
2.0	1	After 31 Kc Fatigue	1.48	25.14	143	241	1.69	--	127	256	2.02	--
2.0	1	After 50 Kc Fatigue	1.48	25.2	143	242	1.69	--	127	257	2.02	--
2.0	1	After 75 Kc Fatigue	1.48	25.14	143	241	1.69	--	127	256	2.02	--
0	2	After 100 Kc Fatigue	2.83	0	274	0	0	--	242	0	0	--
2.0	1	After 100 Kc Fatigue	1.47	25.2	143	242	1.69	--	127	256	2.02	--
2.0	1	After 100 Kc Fatigue to Failure	4.03	73.66	390	707	1.813	--	345	361	2.18	--

NOTES:

1. (1) Combined Shear and Axial Loads
   (2) Shear Load Applied First and Held Constant Compression Load Applied Next
   (3) Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{avg}, Average Shear Load in Bay
   N_x = Et ε_{avg}, Average Compression Load in Bay
   ΔN/ΔP = Calibration Ratios of Load in Bay Per Applied Load
   Aluminum : $E_x = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
   Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in. (Et = 0.2019×10^6, Gt = 0.2402×10^6)

* See Figure 8 for Strain Gage Nomenclature

TABLE 10. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-7.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR LOAD,[2] P_t, klb	COMPRESSION LOAD,[3] P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	3.19	0	281	0	0	0.0881	269	0	0	0.0843
∞	③	Calibration	0	32.1	0	329	∞	0.0102	0	346	∞	0.0108
0	②	Buckling	3.27	0	288	0	0	0.0881	276	0	0	0.0843
0.5	①	Buckling	2.65	11.32	233	115.5	0.5	--	223	122	0.55	--
1.0	①	Buckling	2.24	9.54	197	97	0.49	--	189	103	0.55	--
2.0	①	Buckling	1.72	29.73	152	303	1.99	--	145	321	2.21	--

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{avg}, Average Shear Load in Bay
   N_x = Et ε_{avg}, Average Compression Load in Bay
   ΔN/ΔP = Calibration Ratios of Load in Bay Per Applied Load
   Aluminum : $E_x = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
   Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in. (Et = 0.2019×10^6, Gt = 0.2402×10^6)

* See Figure 8 for Strain Gage Nomenclature

TABLE 11. TEST DATA SUMMARY FOR COMPOSITE PANEL GR-8.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR [2] LOAD, P_t, kib	COMPRESSION [3] LOAD, P_c, kib	UPPER BAY LOADS [4], GAGE 4 *				LOWER BAY LOADS, GAGE 5			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	2.77	0	248	0	0	0.0895	262	0	0	0.0946
∞	③	Calibration	0	34.4	0	353	∞	0.0103	0	368	∞	0.0107
0	②	Buckling	2.99	0	268	0	0	0.0895	283	0	0	0.0946
0.5	①	Buckling	2.79	11.4	250	117	0.47	- -	264	122	0.46	- -
∞	①	Buckling	0	32.66	0	336	∞	0.0107	0	349	∞	0.0107

NOTES :

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder; Torque Arm = 74 in. Upper Bay Load/Lower Bay Load
3. P_c = Axial Compression Load Upper Bay Load/Lower Bay Load
4. N_{xy} = Gt γ_{avg}, Average Shear Load in Bay
   N_x = Et ε_{avg}, Average Compression Load in Bay
   $\Delta N/\Delta P$ = Calibration Ratios of Load in Bay Per Applied Load
   Aluminum : $E_x = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
   Graphite / Ep: $E_x = 3.35 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .0572 in. (Et = 0.2019×10^6, Gt = 0.2402×10^6)

* See Figure 8 for Strain Gage Nomenclature

TABLE 12. TEST DATA SUMMARY FOR PANEL AL-1.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS[5], GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	2	Prebuckling	3.04	0	313	0	0	0.1029	319	0	0	0.1049
∞	3	Calibration	0	21.15	0	528	∞	0.025	0	549	∞	0.026
0		Buckling	4.04	0	416	0	0	0.1029	424	0	0	0.1049
1.0		Buckling	3.01	12.33	310	308	0.99	--	316	321	1.01	--
2.0	1	Buckling	2.13	17.32	219	433		--	223	450	2.02	--
∞	3	Buckling	0	23.63	0	591	1.98	0.025	0	614		0.026
0.5	1	Buckling	3.64	7.59	375	190	0.51	--	382	197	0.52	--
0.5	2	Buckling	3.64	7.43	375	186	0.5	--	382	193	0.51	--
0.5	3	Buckling	3.63	7.51	374	188	0.5	--	381	195	0.51	--
0.5	1	Failure[5]	8.24	16.99	848	425	0.5	--	864	442	0.51	--

<u>NOTES:</u>

1. 1 Combined Shear and Axial Loads
   2 Shear Load Applied First and Held Constant Compression Load Applied Next
   3 Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
5. Failure Mode: Concurrent Stiffener Crippling and Permanent Set in Skin

* See Figure 7 for Strain Gage Nomenclature

TABLE 13. TEST DATA SUMMARY FOR PANEL AL-2.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR LOAD,[2] P_t, klb	COMPRESSION LOAD,[3] P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS[5], GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	3.13	0	326	0	0	0.1042	--	--	--	--
∞	③	Calibration	0	15.03	--	--	--	--	0	- 357		0.0237
0	②	Buckling	3.50	0	365	0	0	--	365	0	0	--
1.0	①		2.80	10.50	292	249	0.85	--	292	- 249	0.85	--
2.0	①	Buckling	1.67	13.81	206	386	1.87	--	206	- 386	1.87	--
∞	③		0	15.03	0	356		--	0	- 356		--
2.0	①	Failure[5]	5.5	47.02	672	1360	2.02	--	672	1360	2.02	--

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   $\Delta N/\Delta P$ = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, $t = .063$ in.
5. Failure Mode: Concurrent Stiffener Crippling and Permanent Set in Skin

* See Figure 7 for Strain Gage Nomenclature

TABLE 14. TEST DATA SUMMARY FOR PANEL AL-3.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS[5], GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	2	Prebuckling	4.06	0	426	0	0	0.1068	444	0	0	0.1093
∞	3	Calibration	0	26.11	0	583		0.0223	0	574		0.0220
0	2	Buckling	4.09	Ↄ	437	0	0	--	447	0	0	--
0.5	1	Buckling	3.45	8.37	368	187	0.51	--	377	184	0.49	--
1.0	1	Buckling	2.88	13.99	308	312	1.01	--	315	308	1.98	--
∞	3	Buckling	0	26.80	0	598		--	0	590		--
2.0	1	Buckling	2.05	19.79	219	441	2.01	--	219	429	1.96	--
2.0	3	Buckling	2.07	19.68	221	439	1.99	--	226	433	1.92	--
2.0	2	Buckling	1.98	19.73	211	440	2.09	--	216	434	2.0	--
2.0	1	Buckling	2.0	19.52	214	435	2.03	--	219	430	1.96	--
2.0	1	Failure[5]	5.08	48.22	543	1075	1.98	--	555	1061	1.91	--

NOTES:

1. 1 Combined Shear and Axial Loads
   2 Shear Load Applied First and Held Constant Compression Load Applied Next
   3 Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = $Gt\,\gamma_{xy}$, Average Shear Load in Bay, lb/in
   N_x = $Et\,\varepsilon_{xy}$, Average Compression Load in Bay, lb/in
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
5. Failure Mode: Concurrent Stiffener Crippling and Permanent Set in Skin

* See Figure 7 for Strain Gage Nomenclature

TABLE 15. TEST DATA SUMMARY FOR PANEL AL-4.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR LOAD, P_t, kib [2]	COMPRESSION LOAD, P_c, kib [3]	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS[5], GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	2.67	0	275	0	0	0.1030	282	0	0	0.1056
∞	③	Calibration	0	15.58	0	299	∞	0.0192	0	367	∞	0.0236
0	①	Buckling	3.90	0	402	0	0	0.1030	412	0	0	0.1056
0.5	①	Buckling	3.22	7.12	332	137	0.41	- -	340	168	0.49	- -
0.5	②	Buckling	3.21	7.73	331	148	0.45	- -	339	182	0.54	- -
0.5	③	Buckling	3.25	7.07	335	136	0.41	- -	343	167	0.49	- -
1.0	①	Buckling	2.70	12.11	278	233	0.84	- -	285	286	1.0	- -
2.0	①	Buckling	1.96	17.72	202	340	1.68	- -	207	418	2.02	- -
∞	①	Buckling	0	24.9/30.0	0	576		0.0192	0	588		0.0236
0.5	①	Failure [5]	8.43	18.83	868	362	0.42	- -	890	444	0.50	- -

<u>*NOTES:*</u>

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ϵ_{xy}, Average Compression Load in Bay, lb/in
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.
5. Failure Mode: Concurrent Stiffener Crippling and Permanent Set in Skin

* See Figure 7 for Strain Gage Nomenclature

TABLE 16. TEST DATA SUMMARY FOR PANEL AL-5.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	3.76	0	401	0	0	0.1067	418	0	0	0.1112
∞	③	Calibration	0	21.03	0	483		0.0229	0	512		0.0243
0	②	Pre-Fatigue	3.88	0	414	0	0	0.1067	431	0	0	0.1112
2.0	①	Buckling	1.83	16.57	195	379	1.94	--	203	403	1.99	--
∞	③		0	26.37	0	604		0.0229	0	641		0.0243
2.0	①	Maximum Fatigue Load	3.35	30.0	357	687 (56%)	1.92	--	373	729	1.95	--
2.0	①	Buckling After 25K Cycles (2.6 in. Crack) in Upper Bay)	1.72	15.30	184	350	1.90		191	372	1.95	

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ϵ_{xy}, Average Compression Load in Bay, lb/in
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 17. TEST DATA SUMMARY FOR PANEL AL-6.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	3.88	0	401	0	0	0.1035	375	0	0	0.0968
∞	③	Calibration	0	22.22	0	418	∞	0.0188	0	511	∞	0.0230
0	②	Buckling	4.18	0	433	0	0	0.1035	405	0	0	0.0968
2.0	①	Buckling	2.79	20.97	--	--	--	--	270	482	1.79	--
2.0	①	Buckling	3.03	23.07	314	434	1.38	--	--	--	--	--
∞	③	Buckling	0	29.34	0	--	--	0.0188	0	675	∞	0.0230

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   $\Delta N/\Delta P$ = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 18. TEST DATA SUMMARY FOR PANEL AL-7.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	3.68	0	398	0	0	0.1082	401	0	0	0.1090
∞	③	Calibration	0	20.45	0	427		0.0209	0	494		0.0242
0	②	Buckling	4.19	0	453	0	0	0.1082	457	0	0	0.1090
2.0	①	Buckling	2.50	20.91	271	437	1.61	- -	273	506	1.85	- -
	③	Buckling	0	29.97	0	626		0.0209	0	725		0.0242

<u>NOTES:</u>

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 19. TEST DATA SUMMARY FOR PANEL AL-8.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	ΔN/ΔP
0	②	Prebuckling	2.78	0	275	0	0	0.0990	287	0	0	0.1032
∞	③	Calibration	0	20.81	0	417	∞	0.0200	0	476	∞	0.0229
0	②	Buckling	3.80	0	376	0	0	0.0990	392	0	0	0.1032
2.0	①	Buckling	2.08	18.09	206	362	1.76	- -	215	414	1.93	- -
∞	③	Buckling	0	25.50/29.07	0	581	∞	0.0200	0	584	∞	0.0229

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   ΔN/ΔP = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 20. TEST DATA SUMMARY FOR PANEL AL-9.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR [2] LOAD, P_t, klb	COMPRESSION [3] LOAD, P_c, klb	UPPER BAY LOADS [4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	2.62	0	277	0	0	0.1057	283	0	0	0.1080
∞	③	Calibration	0	19.19	0	391	∞	0.0204	0	441	∞	0.0230
0	②	Buckling	3.89	0	411	0	0	0.1057	420	0	0	0.1080
0.5	①	Buckling	3.66	8.11	387	165	0.43	- -	395	187	0.47	- -
∞	③	Buckling	0	30.6/33.9	0	692	∞	0.0204	0	703	∞	0.0230

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ϵ_{xy}, Average Compression Load in Bay, lb/in
   $\Delta N/\Delta P$ = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 21. TEST DATA SUMMARY FOR PANEL AL-10.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR[2] LOAD, P_t, klb	COMPRESSION[3] LOAD, P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	3.47	0	357	0	0	0.1029	357	0	0	0.1029
∞	③	Calibration	0	15.45	0	313	∞	0.0203	0	373	∞	0.0241
0	②	Buckling	3.56	0	366	0	0	0.1029	366	0	0	0.1029
0.5	①	Buckling	3.13	7.02	322	143	0.44	- -	322	169	0.52	- -
∞	③	Buckling	0	21.8/27.3	0	555	∞	0.0203	0	525	∞	0.0241

NOTES:

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   $\Delta N/\Delta P$ = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, $t = .063$ in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 22. TEST DATA SUMMARY FOR PANEL AL-11.

NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR LOAD[2], P_t, klb	COMPRESSION LOAD[3], P_c, klb	UPPER BAY LOADS[4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	2	Prebuckling	3.46	0	359	0	0	0.1038	388	0	0	0.1121
∞	3	Calibration	0	15.58	0	321	∞	0.0206	0	365	∞	0.0234
0	2	Buckling	4.19	0	435	0	0	0.1038	470	0	0	0.1121
0.5	1	Buckling	3.35/3.43	7.4-7.7	348	154	0.44	- -	385	181	0.47	- -
∞	3	Buckling	0	27.1/29.0	0	597	∞	0.0206	0	634	∞	0.0234

NOTES:

1. 1 Combined Shear and Axial Loads
   2 Shear Load Applied First and Held Constant Compression Load Applied Next
   3 Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ϵ_{xy}, Average Compression Load in Bay, lb/in
   $\Delta N/\Delta P$ = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 23. TEST DATA SUMMARY FOR PANEL AL-12.

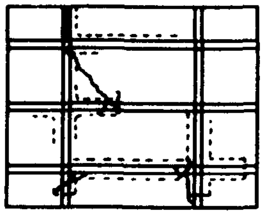
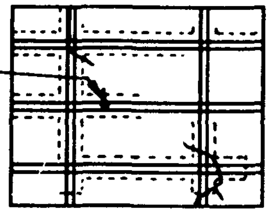
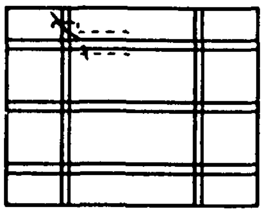
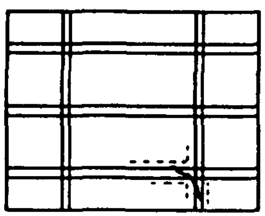
NOMINAL LOAD RATIO (N_x/N_{xy})	LOAD SEQUENCE	LOAD LEVEL TO	SHEAR [2] LOAD, P_t, kib	COMPRESSION [3] LOAD, P_c, kib	UPPER BAY LOADS [4], GAGE 4 *				LOWER BAY LOADS, GAGE 5 *			
					N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$	N_{xy} lb/in	N_x lb/in	N_x/N_{xy} ACTUAL	$\Delta N/\Delta P$
0	②	Prebuckling	2.7	0	274	0	0	0.1004	287	0	0	0.1051
∞	③	Calibration	0	19.1	0	395	∞	0.0207	0	449	∞	0.0235
						0						
0	②	Buckling	4.13	0	415	472	0	0.1004	434	0	0	0.1051
∞	③	Buckling	0	22.8	0	130	∞	- -	0	536	∞	0.0235
0.5	①	Buckling	3.03	6.29	304		0.43	0.0207	318	148	0.46	0.0235

<u>NOTES:</u>

1. ① Combined Shear and Axial Loads
   ② Shear Load Applied First and Held Constant Compression Load Applied Next
   ③ Compression Load Applied First and Held Constant Shear Load Applied Next
2. P_t = Torque Load Per Cylinder
3. P_c = Axial Compression Load
4. N_{xy} = Gt γ_{xy}, Average Shear Load in Bay, lb/in
   N_x = Et ε_{xy}, Average Compression Load in Bay, lb/in
   $\Delta N/\Delta P$ = Applied Cylinder Load to Running Load Conversion Factor
   Aluminum: $E_c = 10.6 \times 10^6$ psi, $G_{xy} = 4.2 \times 10^6$ psi, t = .063 in.

* See Figure 8 for Strain Gage Nomenclature

TABLE 24. FATIGUE FAILURE MODES FOR COMPOSITE PANELS.

PANEL No.	FATIGUE CRACKS*	MAX FATIGUE LOAD, lbs/in		$\frac{N_x}{N_{xy}}$	$\frac{N_x}{N_x^{CR}}$	$\frac{N_{xy}}{N_{xy}^{CR}}$	MAX. FATIGUE LOAD, % STATIC STRENGTH	FATIGUE HISTORY	STATIC OR FATIGUE FAILURE
		N_x	N_{xy}						
GR-5	STATIC FAILURE MODE 	538	218	2.47	1.82	1.45	0.69	RUN OUT AT 100,000 CYCLES	RESIDUAL STATIC STRENGTH: N_x = 833 lb/in N_{xy} = 353 lb/in
GR-6	STATIC FAILURE MODE STIFF. FAILURE 	590	239	2.70	1.68	1.41	0.76	RUN OUT AT 100,000 CYCLES	RESIDUAL STATIC STRENGTH: N_x = 810 lb/in N_{xy} = 356 lb/in
GR-7		334	530	0.63	2.29	2.26	0.78	FATIGUE FAILURE AS SHOWN AT 38,743 CYCLES	FATIGUE
GR-8		286	521	0.55	2.13	2.03	0.77	FATIGUE FAILURE AS SHOWN AT 62,095 CYCLES	FATIGUE

* ——— SKIN RUPTURE .... STIFFENER/SKIN DISBONDS

TABLE 25. FATIGUE FAILURE MODES FOR ALUMINUM PANELS.

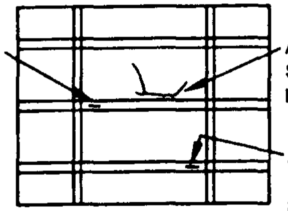
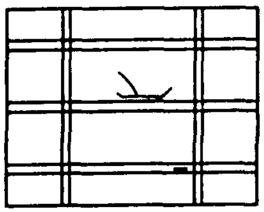
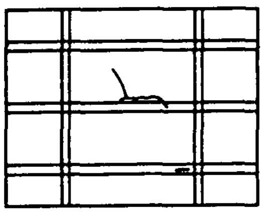
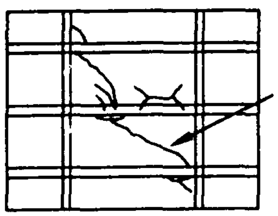
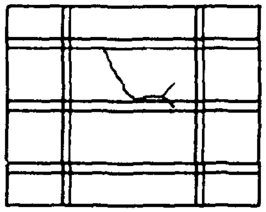
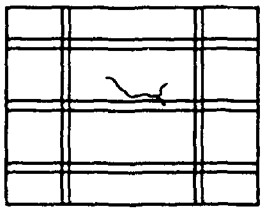
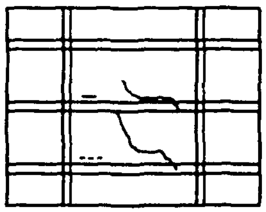
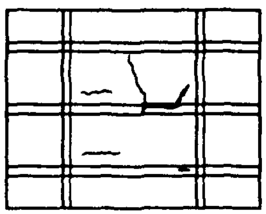
PANEL No.	FATIGUE CRACKS	MAX FATIGUE LOAD, lbs/in N_x	MAX FATIGUE LOAD, lbs/in N_{xy}	$\frac{N_x}{N_{xy}}$	$\frac{N_x}{N_x^{CR}}$	$\frac{N_{xy}}{N_{xy}^{CR}}$	MAX. FATIGUE LOAD, % STATIC STRENGTH	FATIGUE HISTORY	STATIC OR FATIGUE FAILURE
AL-5		640	375	1.70	1.70	1.92	59	CRACK INITIATION AT 24,530 CYCLES	FATIGUE TEST STOPPED, AFTER 41,740 CYCLES WITH CRACK PATTERN SHOWN
AL-6		686	376	1.83	1.64	1.33	63	CRACK INITIATION AT 22,500 CYCLES	FATIGUE TEST STOPPED AFTER 38,905 CYCLES WITH CRACK PATTERN SHOWN
AL-7		605	304	1.99	1.42	1.17	56	RUN OUT AT 100K CYCLES. CRACKS INITIATED AFTER FATIGUE LOAD WAS INCREASED	FATIGUE FAILURE AFTER 162K CYCLES AT HIGHER LOADS
AL-8		627	362	1.73	1.77	1.65	58	CRACK INITIATION AT 16,604 CYCLES	FATIGUE TEST STOPPED AT 27,900 CYCLES WITH CRACK PATTERN SHOWN

TABLE 25. FATIGUE FAILURE MODES FOR ALUMINUM PANELS (CONT'D.)

PANEL No.	FATIGUE CRACKS	MAX FATIGUE LOAD, lbs/in		$\frac{N_x}{N_{xy}}$	$\frac{N_x}{N_x^{CR}}$	$\frac{N_{xy}}{N_{xy}^{CR}}$	MAX. FATIGUE LOAD, % STATIC STRENGTH	FATIGUE HISTORY	STATIC OR FATIGUE FAILURE
		N_x	N_{xy}						
AL-9		302	745	0.41	1.82	1.83	0.86	CRACK INITIATION AT 10,635 CYCLES	FATIGUE TEST STOPPED AFTER 12,273 CYCLES WITH CRACK PATTERN SHOWN
AL-10		225	542	0.41	1.63	1.64	0.64	CRACK INITIATION AT 22,300 CYCLES	FATIGUE TEST STOPPED AFTER 42,971 CYCLES WITH CRACK PATTERN SHOWN
AL-11		253	608	0.42	1.72	1.69	0.73	CRACK INITIATION AT 13,540 CYCLES	FATIGUE TEST STOPPED AFTER 25,120 CYCLES WITH CRACK PATTERN SHOWN
AL-12		280	648	0.43	2.15	2.13	0.80	CRACK INITIATION AT 16,543 CYCLES	FATIGUE TEST STOPPED AT 24,100 CYCLES WITH PATTERN SHOWN

REFERENCES

1. Deo, R.B., "Design Development and Durability Validation of Postbuckled Composite and Metal Panels," AFWAL-TR-85-3077, Final Report for Period September 1984 through September 1985 on Air Force Contract F33615-84-C-3220.

2. Deo, R.B., Agarwal, B.L., and Madenci, E., "Design Methodology and Life Analysis of Postbuckled Metal and Composite Panels," AFWAL-TR-85-3096, Vol. I. Final Report on Contract F33615-81-C-3208, December 1985.

3. Deo, R.B., Kan, H.P., and Bhatia, N.M., "Design Development and Durability Validation of Postbuckled Composite and Metal Panels," Final Report Volume III on AFWAL Contract F33615-84-C-3220.

4. Ogonowski, J.M., and Sanger, K.B., "Postbuckling of Curved and Flat Stiffened Composite Panels Under Combined Loads," Report No. NADC-81097-60, December 1984.

5. Wilkins, D.J., "Anisotropic Curved Panel Analysis," General Dynamics, Convair Aerospace Division Report FZM-5567, May 1973.

6. Deo, R.B., Kan, H.P., and Bhatia, N.M., "Design Development and Durability Validation of Postbuckled Composite and Metal Panels, Volume III - Analysis and Test Results," WRDC-TR-89-3030, Volume III, Contract F33615-84-C-3220, November 1989.

7. Bruhn, E.F., "Analysis and Design of Flight Vehicle Structures," S.R. Jacobs and Associates, Inc., 1973.

APPENDIX A

COMPOSITE PANEL ANALYSIS FOR SHEAR LOADS

Interactive analysis of the composite panel under shear loads using program TENWEB is presented in this Appendix.

```
A>b:tenweb
File name missing or blank - Please enter name
UNIT 6? CON
 YOU ARE EXECUTING COMPOSITE TENSION FIELD PROGRAM.      ""GOOD LUCK""
 INPUT NO OF MATERIALS USED IN PANEL CONSTRUCTION
UNIT 5? CON
2

  INPUT - PANEL RADIUS,RING SPACING, STRINGER SPACING
45. 24. 10.

 INPUT MATERIAL LAMINA PROPERTIES. LONGITUDINAL DIRECTION IS ALONG STRINGER AXIS

 INPUT LAMINA PROPERTIES FOR MAT NO  1  "EL,ET,GLT,NULT"
10.E6 9.2E6 .9E6 .055

 INPUT LAMINA PROPERTIES FOR MAT NO  2  "EL,ET,GLT,NULT"
18.7E6 1.87E6 .85E6 .3

  2 MATERIALS ARE USED IN PANEL CONSTRUCTION. THE LAMINA PROPERTIES ARE AS FOLLOWS

                   EL         ET          GLT     NULT
 MATERIAL NO 1  .1000E+08  .9200E+07  .9000E+06 .055
 MATERIAL NO 2  .1870E+08  .1870E+07  .8500E+06 .300

 INPUT WEB LAMINATE PROPERTIES
  ENTER NO OF LAYERS IN THE LAMINATE
5

INPUT MATERIAL KIND FOR    5 LAYERS
2*1 2 2*1

INPUT THICKNESS FOR    5 LAYERS
2*.013 .0052 2*.013

INPUT ORIENTATION FOR    5 LAYERS
2*45 90 2*45

  INPUT STRINGER CONFIGURATION-- 1 FOR HAT, 2 FOR I SECTION
1

  INPUT RING CONFIGURATION-- 1 FOR HAT, 2 FOR I SECTION
2

  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
Y

    11111111244444444444442111111111
             2          2
              2        2
               2      2
                233332

 DO YOU WISH TO INPUT NEW ELEMENT WIDTHS? INPUT YES OR NO
Y

 INPUT ELEMENT WIDTHS
1. 1.3 .75 1.12

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  1 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
6*1 3*2 2*1
```

```
INPUT THICKNESS FOR   11 LAYERS
6*.013 3*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
6*45 90 2*0 2*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  2 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
4

INPUT MATERIAL KIND FOR    4 LAYERS
4*1

INPUT THICKNESS FOR    4 LAYERS
4*.013

INPUT ORIENTATION FOR    4 LAYERS
4*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  3 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
2*1 7*2 2*1

INPUT THICKNESS FOR   11 LAYERS
2*.013 7*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
2*45 7*0 2*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  4 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
2*1 7*2 2*1

INPUT THICKNESS FOR   11 LAYERS
2*.013 7*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
2*45 3*0 90 3*0 2*45

    STIFFENER PROPERTIES
    --------------------



    1111111124444444444444211111111
             2          2
              2        2
               2      2
                233332

    ELEMENT NUMBERS          1         2         3         4
    ELEMENT WIDTHS =       1.000     1.300      .750     1.120
    ELEMENT THICKNESS =     .120      .052      .088      .088
    ELEMENT MODULAS =       .47E+07   .31E+07   .96E+07   .89E+07
         EA= .31E+07
         EI= .90E+06
```

```
        YBAR= .355
  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
Y


    11112222366667777
            3
            3
            3
            3    5
            3    5
            344445

 DO YOU WISH TO INPUT NEW ELEMENT WIDTHS? INPUT YES OR NO
Y

 INPUT ELEMENT WIDTHS
.75 .75 2.9 1. .4 .75 .75

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  1 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
8

INPUT MATERIAL KIND FOR    8 LAYERS
3*1 3*2 2*1

INPUT THICKNESS FOR    8 LAYERS
3*.013 3*.0052 2*.013

INPUT ORIENTATION FOR    8 LAYERS
3*45 3*0 2*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  2 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
4*1 5*2 2*1

INPUT THICKNESS FOR   11 LAYERS
4*.013 5*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
4*45 5*0 2*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  3 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
4

INPUT MATERIAL KIND FOR    4 LAYERS
4*1

INPUT THICKNESS FOR    4 LAYERS
4*.013

INPUT ORIENTATION FOR    4 LAYERS
4*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  4 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
```

```
6

INPUT MATERIAL KIND FOR    6 LAYERS
2*1 2*2 2*1

INPUT THICKNESS FOR    6 LAYERS
2*.013 2*.0052 2*.013

INPUT ORIENTATION FOR    6 LAYERS
2*45 2*0 2*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  5 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
4

INPUT MATERIAL KIND FOR    4 LAYERS
4*1

INPUT THICKNESS FOR    4 LAYERS
4*.013

INPUT ORIENTATION FOR    4 LAYERS
4*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  6 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
4*1 5*2 2*1

INPUT THICKNESS FOR   11 LAYERS
4*.013 5*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
4*45 5*0 2*45

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  7 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
8

INPUT MATERIAL KIND FOR    8 LAYERS
3*1 3*2 2*1

INPUT THICKNESS FOR    8 LAYERS
3*.013 3*.0052 2*.013

INPUT ORIENTATION FOR    8 LAYERS
3*45 3*0 2*45

    STIFFENER PROPERTIES
    --------------------


    11112222366667777
            3
            3
            3
            3    5
            3    5
            344445
```

```
   ELEMENT NUMBERS          1         2         3         4         5         6         7
   ELEMENT WIDTHS =        .750      .750     2.900     1.000      .400      .750      .750
   ELEMENT THICKNESS =     .081      .104      .052      .062      .052      .104      .081
   ELEMENT MODULAS =       .61E+07   .70E+07   .31E+07   .57E+07   .31E+07   .70E+07   .61E+07
        EA= .27E+07
        EI= .35E+07
        YBAR= .689
  INPUT - ULTIMATE SHEAR FLOW, BUCK'ING SHEAR FLOW
900 200


   WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .353E+07,  451E+07,  .422E+07,  .5378,  .0572
   PANEL RADIUS=     45.0
   STRINGER SPACING=   10.0
   RING SPACING=   24.0
   ULTIMATE SHEAR FLOW=  900.0
   BUCKLING SHEAR FLOW=  200.0


   DIAGONAL TENSION ANGLE ALPHA= 39.84  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .003031
 STRINGER STRAIN=  -.003105
 RING STRAIN=  -.003724
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.015 .015

   STRINGER FORCED CRIPPLING STRAIN=  .0033
   RING FORCED CRIPPLING STRAIN=  .0035
   MAXIMUM STRINGER STRAIN= -.0035
   MAXIMUM RING STRAIN= -.0042
   AVERAGE STRINGER STRAIN= -.0022
   AVERAGE RING STRAIN=    -.0027
   STRINGER MARGIN OF SAFETY=   -7.0  PERCENT
   RING MARGIN OF SAFETY=  -18.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
900 284


   WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .353E+07,  .451E+07,  .422E+07,  .5378,  .0572
   PANEL RADIUS=     45.0
   STRINGER SPACING=   10.0
   RING SPACING=   24.0
   ULTIMATE SHEAR FLOW=  900.0
   BUCKLING SHEAR FLOW=  284.0


   DIAGONAL TENSION ANGLE ALPHA= 39.81  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .002837
 STRINGER STRAIN=  -.002482
 RING STRAIN=    .002761
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.015  015

   STRINGER FORCED CRIPPLING STRAIN=  .0029
   RING FORCED CRIPPLING STRAIN=  .0031
   MAXIMUM STRINGER STRAIN= -.0030
```

```
    MAXIMUM RING STRAIN= -.0033
    AVERAGE STRINGER STRAIN= - 0017
    AVERAGE RING STRAIN=   -.0020
    STRINGER MARGIN OF SAFETY=   -1.0  PERCENT
    RING MARGIN OF SAFETY=   -7.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
700 284


    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .353E+07,  .451E+07,  .422E+07,  .5378,  .0572
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=  24.0
    ULTIMATE SHEAR FLOW=  700.0
    BUCKLING SHEAR FLOW=  284.0


    DIAGONAL TENSION ANGLE ALPHA= 38.77  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .002085
 STRINGER STRAIN=  -.001587
 RING STRAIN=  -.001549
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.015 .015

    STRINGER FORCED CRIPPLING STRAIN=  .0026
    RING FORCED CRIPPLING STRAIN=  .0027
    MAXIMUM STRINGER STRAIN= -.0020
    MAXIMUM RING STRAIN= -.0019
    AVERAGE STRINGER STRAIN= -.0011
    AVERAGE RING STRAIN=   -.0011
    STRINGER MARGIN OF SAFETY=   29.0  PERCENT
    RING MARGIN OF SAFETY=   39.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
800 284


    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .353E+07,  .451E+07,  .422E+07,  .5378,  .0572
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=  24.0
    ULTIMATE SHEAR FLOW=  800.0
    BUCKLING SHEAR FLOW=  284.0


    DIAGONAL TENSION ANGLE ALPHA= 39.40  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .002460
 STRINGER STRAIN=  -.002026
```

```
 RING STRAIN=  -.002132
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.015 .015

    STRINGER FORCED CRIPPLING STRAIN=  .0028
    RING FORCED CRIPPLING STRAIN=  .0029
    MAXIMUM STRINGER STRAIN= -.0025
    MAXIMUM RING STRAIN= -.0026
    AVERAGE STRINGER STRAIN= -.0014
    AVERAGE RING STRAIN=   -.0016
    STRINGER MARGIN OF SAFETY=   11.0  PERCENT
    RING MARGIN OF SAFETY=   11.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
850 284


    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .353E+07,  .451E+07,  .422E+07,  .5378,  .0572
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=   24.0
    ULTIMATE SHEAR FLOW=  850.0
    BUCKLING SHEAR FLOW=  284.0


    DIAGONAL TENSION ANGLE ALPHA= 39.62  DEGREES


 WEB DIAGONAL TENSION STRAIN=   .002648
 STRINGER STRAIN=  -.002252
 RING STRAIN=  -.002441
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.015 .015

    STRINGER FORCED CRIPPLING STRAIN=  .0029
    RING FORCED CRIPPLING STRAIN=  .0030
    MAXIMUM STRINGER STRAIN= -.0027
    MAXIMUM RING STRAIN= -.0030
    AVERAGE STRINGER STRAIN= -.0016
    AVERAGE RING STRAIN=   -.0018
    STRINGER MARGIN OF SAFETY=    4.0  PERCENT
    RING MARGIN OF SAFETY=    1.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW875 284
875 284


    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .353E+07,  .451E+07,  .422E+07,  .5378,  .0572
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=   24.0
    ULTIMATE SHEAR FLOW=  875.0
    BUCKLING SHEAR FLOW=  284.0
```

```
   DIAGONAL TENSION ANGLE ALPHA= 39.72  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .002743
 STRINGER STRAIN=  -.002366
 RING STRAIN=  -.002600
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.015 .015

   STRINGER FORCED CRIPPLING STRAIN=  .0029
   RING FORCED CRIPPLING STRAIN=  .0030
   MAXIMUM STRINGER STRAIN= -.0029
   MAXIMUM RING STRAIN= -.0031
   AVERAGE STRINGER STRAIN= -.0017
   AVERAGE RING STRAIN=   -.0019
   STRINGER MARGIN OF SAFETY=    1.0  PERCENT
   RING MARGIN OF SAFETY=   -2.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
 Y
Stop - Program terminated.
```

APPENDIX B

COMPOSITE PANEL ANALYSIS FOR COMPRESSION LOADS

Interactive analysis of the composite panel under compression loads using program CRIP is presented in this Appendix.

```
File name missing or blank - Please enter name
UNIT 6? con

 YOU ARE EXECUTING COMPRESSION PANEL PROGRAM.            ""GOOD LUCK""

 INPUT NO OF MATERIALS USED IN PANEL CONSTRUCTION
UNIT 5? con
2

  INPUT - PANEL RADIUS,PANEL LENGTH, STRINGER SPACING
45. 24. 10.

 INPUT MATERIAL LAMINA PROPERTIES. LONGITUDINAL DIRECTION IS ALONG STRINGER AXIS

 INPUT LAMINA PROPERTIES FOR MAT NO  1  "EL,ET,GLT,NULT"
10.e6 9.2e6 .9e6 .055

 INPUT LAMINA PROPERTIES FOR MAT NO  2  "EL,ET,GLT,NULT"
18.7e6 1.87e6 .85e6 .3

  2 MATERIALS ARE USED IN PANEL CONSTRUCTION. THE LAMINA PROPERTIES ARE AS FOLLOWS

                    EL          ET          GLT     NULT
 MATERIAL NO 1  .1000E+08  .9200E+07  .9000E+06 .055
 MATERIAL NO 2  .1870E+08  .1870E+07  .8500E+06 .300
  INPUT STRINGER CONFIGURATION-- 1 FOR HAT, 2 FOR I SECTION
1


    1111111124444444444444211111
             2         2
              2       2
               2     2
                233332

 INPUT ELEMENT WIDTHS
1. 1.3 0.75 1.12

 INPUT ELEMENT ALLOWABLE ULTIMATE STRAINS
.015 .012 .012 .015

  INPUT LAMINATE PROPERTIES FOR STIFFENER ELEMENT    1
  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
6*1 3*2 2*1

INPUT THICKNESS FOR   11 LAYERS
6*.013 3*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
6*45 90 2*0 2*85

  INPUT LAMINATE PROPERTIES FOR STIFFENER ELEMENT    2
  ARE LAMINATE PROPERTIES OF THIS ELEMENT IDENTICAL TO ANY OF THE PREVIOUS ELEMENT??
   IF YES INPUT THE ELEMENT NUMBER OTHERWISE INPUT 0
0

  ENTER NO OF LAYERS IN THE LAMINATE
4
```

```
INPUT MATERIAL KIND FOR    4 LAYERS
4*1

INPUT THICKNESS FOR    4 LAYERS
4*.013

INPUT ORIENTATION FOR    4 LAYERS
4*45

  INPUT LAMINATE PROPERTIES FOR STIFFENER ELEMENT    3
  ARE LAMINATE PROPERTIES OF THIS ELEMENT IDENTICAL TO ANY OF THE PREVIOUS ELEMENT??
   IF YES INPUT THE ELEMENT NUMBER OTHERWISE INPUT 0
0

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
2*1 7*2 2*1

INPUT THICKNESS FOR   11 LAYERS
2*.013 7*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
2*45 7*0 2*45

  INPUT LAMINATE PROPERTIES FOR STIFFENER ELEMENT    4
  ARE LAMINATE PROPERTIES OF THIS ELEMENT IDENTICAL TO ANY OF THE PREVIOUS ELEMENT??
   IF YES INPUT THE ELEMENT NUMBER OTHERWISE INPUT 0
0

  ENTER NO OF LAYERS IN THE LAMINATE
11

INPUT MATERIAL KIND FOR   11 LAYERS
2*1 7*2 2*1

INPUT THICKNESS FOR   11 LAYERS
2*.013 7*.0052 2*.013

INPUT ORIENTATION FOR   11 LAYERS
2*45 3*0 90 3*0 2*45

    STIFFENER PROPERTIES
    --------------------



    11111111244444444444444211111111
            2           2
             2         2
              2       2
               233332

    ELEMENT NUMBERS          1         2         3         4
    ELEMENT WIDTHS =       1.000     1.300      .750     1.120
    ELEMENT THICKNESS =     .120      .052      .088      .088
    ELEMENT MODULAS =        .470E+07   .306E+07   .960E+07   .892E+07
         EA=  .306E+07
         EI=  .899E+06
         YBAR= .355

 INPUT WEB LAMINATE PROPERTIES
  ENTER NO OF LAYERS IN THE LAMINATE
5
```

```
INPUT MATERIAL KIND FOR    5 LAYERS
2*1 2 2*1

INPUT THICKNESS FOR    5 LAYERS
2*.013 .0052 2*.013

INPUT ORIENTATION FOR    5 LAYERS
2*45 90 2*45

  WEB PROPERTIES ARE AS FOLLOWS
  -----------------------------

    T= .0572   EX=  .353E+07   EY=  .451E+07   GXY=  .422E+07   NUXY= .538
    AIJ=  .320E+06  .220E+06  .408E+06  .104E+05  .104E+05  .242E+06
    DIJ=  .930E+02  .649E+02  .932E+02  .313E+01  .313E+01  .711E+02

  SUMMARY OF THE RESULTS
  ----------------------

  EULER BUCKLING STRAIN=   .016552
  SKIN BUCKLING STRAIN=   .001356
  BUCKLING STRAIN OF STIFFENER ELEMENTS ARE AS FOLLOWS
      .013141   .016576   .044671   .021562
  CRIPPLING STRAIN OF STIFFENER ELEMENTS ARE AS FOLLOWS
      .013141   .016576   .044671   .021562
  FAILURE LOAD DUE TO EULER BUCKLING=   .7359E+05
  FAILURE LOAD DUE TO STIFFENER CRIPPLING=   .4268E+05
          94.  PERCENT OF THE LOAD IS CARRIED IN THE STIFFENERStop - Program terminated.

A>
```

APPENDIX C

METAL PANEL ANALYSIS FOR SHEAR LOADS

The interactive computer output describing the analysis of metal panel under shear loads is presented in this Appendix. Program TENWEB was used for the analysis.

```
B:TENWEB
File name missing or blank - Please enter name
UNIT 6? CON

 YOU ARE EXECUTING COMPOSITE TENSION FIELD PROGRAM.      ""GOOD LUCK""

 INPUT NO OF MATERIALS USED IN PANEL CONSTRUCTION
UNIT 5? CON
1

  INPUT - PANEL RADIUS,RING SPACING, STRINGER SPACING
45. 24. 10.

 INPUT MATERIAL LAMINA PROPERTIES. LONGITUDINAL DIRECTION IS ALONG STRINGER AXIS

 INPUT LAMINA PROPERTIES FOR MAT NO  1  "EL,ET,GLT,NULT"
10.7E6 10.7E6 4.E6 .33

  1 MATERIALS ARE USED IN PANEL CONSTRUCTION. THE LAMINA PROPERTIES ARE AS FOLLOWS

                    EL          ET          GLT     NULT
 MATERIAL NO 1   .1070E+08   .1070E+08   .4000E+07 .330

 INPUT WEB LAMINATE PROPERTIES
  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.063

INPUT ORIENTATION FOR    1 LAYERS
0

  INPUT STRINGER CONFIGURATION-- 1 FOR HAT, 2 FOR I SECTION
2

  INPUT RING CONFIGURATION-- 1 FOR HAT, 2 FOR I SECTION
2

  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
Y



    11112222366667777
            3
            3
            3
            3    5
            3    5
            344445

 DO YOU WISH TO INPUT NEW ELEMENT WIDTHS? INPUT YES OR NO
Y

 INPUT ELEMENT WIDTHS
 .5625 0. 1 25 1.125 0. 0  .5625

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  1 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
```

```
1

INPUT THICKNESS FOR    1 LAYERS
.094

INPUT ORIENTATION FOR    1 LAYERS
0

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  3 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.094

INPUT ORIENTATION FOR    1 LAYERS
0

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  4 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.094

INPUT ORIENTATION FOR    1 LAYERS
0

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  7 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.094

INPUT ORIENTATION FOR    1 LAYERS
0

    STIFFENER PROPERTIES
    --------------------


    11112222366667777
            3
            3
            3
            3    5
            3    5
            344445

   ELEMENT NUMBERS         1        2        3        4        5        6        7
   ELEMENT WIDTHS =       .563     .000    1.250    1.125     .000     .000     .563
   ELEMENT THICKNESS =    .094     .000     .094     .094     .000     .000     .094
```

```
    ELEMENT MODULAS =       .11E+08  .00E+00  .11E+08  .11E+08  .00E+00  .00E+00  .11E+08
         EA= .35E+07
         EI= .10E+07
         YBAR= .625
  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
Y



    11112222366667777
            3
            3
            3
            3    5
            3    5
            344445

 DO YOU WISH TO INPUT NEW ELEMENT WIDTHS? INPUT YES OR NO
Y

 INPUT ELEMENT WIDTHS
.5625 0. 1.375 1.125 0. 0. .5625

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  1 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.125

INPUT ORIENTATION FOR    1 LAYERS
0

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  3 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.125

INPUT ORIENTATION FOR    1 LAYERS
0

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  4 INPUT YES OR NO
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.125

INPUT ORIENTATION FOR    1 LAYERS
0

 DO YOU WISH TO INPUT  LAMINATE CONFIGURATION FOR ELEMENT  7 INPUT YES OR NO
```

```
Y

  ENTER NO OF LAYERS IN THE LAMINATE
1

INPUT MATERIAL KIND FOR    1 LAYERS
1

INPUT THICKNESS FOR    1 LAYERS
.125

INPUT ORIENTATION FOR    1 LAYERS
0

    STIFFENER PROPERTIES
    --------------------



    11112222366667777
            3
            3
            3
            3    5
            3    5
            344445

    ELEMENT NUMBERS        1        2        3        4        5        6        7
    ELEMENT WIDTHS =       .563     .000     1.375    1.125    .000     .000     .563
    ELEMENT THICKNESS =    .125     .000     .125     .125     .000     .000     .125
    ELEMENT MODULAS =      .11E+08  .00E+00  .11E+08  .11E+08  .00E+00  .00E+00  .11E+08
         EA= .48E+07
         EI= .17E+07
         YBAR= .688
  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
900 200

    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .107E+08,  .107E+08,  .400E+07,  .3300,  .0630
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=   24.0
    ULTIMATE SHEAR FLOW=  900.0
    BUCKLING SHEAR FLOW=  200.0

    DIAGONAL TENSION ANGLE ALPHA= 40.40  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .002074
 STRINGER STRAIN=  -.003400
 RING STRAIN=  -.003423
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.004 .004

    STRINGER FORCED CRIPPLING STRAIN=  .0025
    RING FORCED CRIPPLING STRAIN=  .0027
    MAXIMUM STRINGER STRAIN= -.0028
    MAXIMUM RING STRAIN= -.0028
    AVERAGE STRINGER STRAIN= -.0015
    AVERAGE RING STRAIN=   -.0015
    STRINGER MARGIN OF SAFETY=  -12.0  PERCENT
    RING MARGIN OF SAFETY=   -3.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N
```

```
 DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
900 200

    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .107E+08,  .107E+08,  .400E+07,  .3300
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=   24.0
    ULTIMATE SHEAR FLOW=  900.0
    BUCKLING SHEAR FLOW=  200.0

    DIAGONAL TENSION ANGLE ALPHA= 40.40  DEGREES

 WEB DIAGONAL TENSION STRAIN=   .002074
 STRINGER STRAIN=  -.003400
 RING STRAIN=  -.003423
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
 0065 .0065

    STRINGER FORCED CRIPPLING STRAIN=  .0030
    RING FORCED CRIPPLING STRAIN=  .0033
    MAXIMUM STRINGER STRAIN= -.0028
    MAXIMUM RING STRAIN= - 0028
    AVERAGE STRINGER STRAIN= -.0015
    AVERAGE RING STRAIN=   -.0015
    STRINGER MARGIN OF SAFETY=    6.0  PERCENT
    RING MARGIN OF SAFETY=   16.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
900 226

    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .107E+08,  .107E+08,  .400E+07,  .3300,  .0630
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=   24.0
    ULTIMATE SHEAR FLOW=  900.0
    BUCKLING SHEAR FLOW=  226.0

    DIAGONAL TENSION ANGLE ALPHA= 40.31  DEGREES

 WEB DIAGONAL TENSION STRAIN=   .002059
 STRINGER STRAIN=  -.003084
 RING STRAIN=  -.003027
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
 .0065 .0065

    STRINGER FORCED CRIPPLING STRAIN=  .0029
    RING FORCED CRIPPLING STRAIN=  .0032
    MAXIMUM STRINGER STRAIN= -.0025
    MAXIMUM RING STRAIN= -.0024
    AVERAGE STRINGER STRAIN= -.0013
    AVERAGE RING STRAIN=   -.0013
    STRINGER MARGIN OF SAFETY=   17.0  PERCENT
```

```
    RING MARGIN OF SAFETY=   31.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
N
  DO YOU WISH TO INPUT NEW STRINGER CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT
N

  DO YOU WISH TO INPUT NEW RING CONFIGURATION OR MODIFY ELEMENT LAMINATE PROPERTIES? INPUT YES
N

  INPUT - ULTIMATE SHEAR FLOW, BUCKLING SHEAR FLOW
900 350

    WEB PROPERTIES EX,EY,GXY,NUXY,THICKNESS= .107E+08,  .107E+08,  .400E+07,  .3300,  .0630
    PANEL RADIUS=    45.0
    STRINGER SPACING=   10.0
    RING SPACING=   24.0
    ULTIMATE SHEAR FLOW=  900.0
    BUCKLING SHEAR FLOW=  350.0


    DIAGONAL TENSION ANGLE ALPHA= 39.63  DEGREES


 WEB DIAGONAL TENSION STRAIN=    .001991
 STRINGER STRAIN=  -.001967
 RING STRAIN=  -.001722
  INPUT - ALLOWABLE YIELD STRAIN FOR -STRINGER AND RING
.0065 .0065

    STRINGER FORCED CRIPPLING STRAIN=  .0024
    RING FORCED CRIPPLING STRAIN=  .0027
    MAXIMUM STRINGER STRAIN= -.0013
    MAXIMUM RING STRAIN= -.0012
    AVERAGE STRINGER STRAIN= -.0009
    AVERAGE RING STRAIN=   -.0007
    STRINGER MARGIN OF SAFETY=   81.0  PERCENT
    RING MARGIN OF SAFETY=  128.0  PERCENT

  ARE YOU HAPPY WITH THE DESIGN ? INPUT-YES OR NO
Y
Stop - Program terminated.

A>
```


U S Government Printing Office 1992—648-127/62364