VOLUME V

TITANIUM DEVELOPMENT PROGRAM

G. D. LINDENEAU
D. H. LOVE
J. K. NEARY
H. A. BUEHLER
G. E. FOELSCH

GENERAL DYNAMICS/CONVAIR
A Division of General Dynamics Corporation

Contract AF 33(600)34876
ASD Project 7-576

FINAL ENGINEERING REPORT
December 1957 - May 1961

Approved for public release; distribution is unlimited.

Typical airframe structures, fuselage frames, wing leading edge, bleed air ducts, tail cone, shear and compression panels of Ti-4Al-3Mo-2V and Ti-13V-11Cr-3Al were subjected to test loads in increasing of 100°F from room temperature to maximum temperatures of 800°F and 900°F depending on the part.

Fabrication Branch
Manufacturing Technology Laboratory
United States Air Force Aeronautical Systems Division
Wright-Patterson Air Force Base, Ohio
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FINAL TECHNICAL ENGINEERING REPORT
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Typical airframe structures, fuselage frames, wing leading edge, bleed air ducts, tail cone, shear and compression panels of Ti-4Al-3Mo-1V and Ti-13V-11Cr-3Al were subjected to test loads in increasing of 100° F from room temperature to maximum temperatures of 800° F and 900° F depending on the part.

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Typical airframe structures, fuselage frames, wing leading edge, bleed air ducts, tail cone, shear panels and compression panels of Ti-4Al-3Mo-1V and Ti-13V-11Cr-3Al were subjected to test loads in increasing increments of 100 degrees from room temperature to maximum temperatures of 800 F and 900 F, depending on the part. Riveted and resistance welded construction was evaluated in the fuselage frame and wing leading edge. Other components were either fusion welded, resistance welded, riveted or brazed. Components were subjected to static and repeated loadings with the exception of compression panels which had axial and side loads supplied. All components satisfactorily withstood static test loads. Under repeated load test, the resistance welded fuselage frame and wing leading edge, although adequate, did not perform as well as the riveted versions. Repeated load tests of resistance welded shear panels showed marginal results. Other components performed satisfactorily under repeated load conditions.

Tests of spotwelded construction in the fuselage frame and wing leading edge demonstrated the need of large margins in spotweld strengths at ends of members joined by spotwelding. Although Ti-4Al-3Mo-1V is not an optimum weldable alloy, the spotwelded assemblies of these components were considered adequate from repeated load tests at elevated temperature even though they did not perform as well as riveted construction. For example, the riveted fuselage frame withstood approximately 200% more repeated loads than the one of spotwelded construction.

Air ducts in fusion welded, seam welded and riveted and brazed configurations satisfactorily withstood static and repeated test load requirements. The seam welded construction sustained the highest pressure in the burst tests.

All resistance welded shear panels sustained design static test loads. The repeated load tests indicate that much more data is needed. The tests were not conclusive and fell short of expectations. Notch factors due to spotwelding need further investigation.

Three types of compression panels in Ti-4Al-3Mo-1V and three types of compression panels in Ti-13V-11Cr-3Al withstood combined compression load and side load from pressure in excess of design loads. Panels in Ti-13V-11Cr-3Al exhibited a brittle type of failure - probably due to low elongation in the material.
NOTICES

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FOREWORD

This is the fifth of five volumes comprising the Final Technical Engineering Report covering the work performed under Contract AF 33(600)34876. This work was conducted by General Dynamics/Convair, A Division of General Dynamics during the period, December 1957 to May 1961. This volume describes the structural testing and evaluation of the test assemblies made from the selected alloys of titanium. The manuscript was released 31 May 1961 for publication as an ASD Technical Report.

This contract was initiated under ASC Manufacturing Methods Project 7-576 "Titanium Development Program." It was administered under the direction of Mr. R. T. Jameson, ASRCTF, project engineer, Fabrication Branch of the Manufacturing Technology Laboratory (ASRCT), Aeronautical Systems Division, Wright-Patterson Air Force Base, Ohio.

Program work was conducted by the Applied Manufacturing Research, Operating Controls and Methods Department under the direction of A. P. Langlois, the project director, with the assistance of the Engineering Department. S. R. Carpenter was the engineering coordinator for the program; J. F. Murphy was the Applied Manufacturing Research project leader. Others who have contributed heavily to this program are C. W. Alesch, G. D. Lindeneau, D. H. Love, J. K. Neary, H. A. Buehler, G. F. Foelsch, S. D. Green, and R. D. Woodward.

The primary objective of the Air Force Manufacturing Methods Program is to increase producibility and improve the quality and efficiency of fabrication of aircraft, missiles and components thereof. This report is disseminated in order that methods and equipment developed may be made available throughout industry, thereby reducing costs or increasing capabilities, resulting in "More Air Force Per Dollar."

Your comments are solicited on the potential use of the information contained in this report as it applies to your present or future production programs. Suggestions concerning additional manufacturing methods development required on this or other subjects will be appreciated.

PUBLICATION REVIEW

This report has been reviewed and is approved.

FOR THE COMMANDER:

Charles F. H. Bege
Chief, Manufacturing Technology Laboratory
Directorate of Materials & Processes
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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

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<td>A-88</td>
<td>Fuselage Canted Bulkhead - Riveted Assembly - Web Fatigue Failure after 140,629 Total Cycles</td>
<td>105</td>
</tr>
<tr>
<td>A-89</td>
<td>Fuselage Canted Bulkhead - Riveted Assembly - Web Fatigue Failure after 140,629 Total Cycles</td>
<td>106</td>
</tr>
<tr>
<td>A-90</td>
<td>Fuselage Canted Bulkhead - Riveted Assembly - Inner Flange Fatigue Failure after 140,629 Cycles</td>
<td>107</td>
</tr>
<tr>
<td>A-91</td>
<td>Fuselage Canted Bulkhead - Riveted Assembly - Inner Flange Fatigue Failure after 140,629 Cycles</td>
<td>108</td>
</tr>
<tr>
<td>A-92</td>
<td>Fuselage Canted Bulkhead - Riveted Assembly - Inner Flange Fatigue Failure after 140,629 Cycles</td>
<td>109</td>
</tr>
</tbody>
</table>
This report was prepared in order to present the results of the static and fatigue tests of the spotwelded and riveted bulkheads. These specimens were manufactured according to the procedures developed during previous phases of the Titanium Development Project.

Deflection, permanent set, and strain data are presented for the room-temperature static tests. Deflection and permanent set data are presented for the elevated-temperature static tests.

The objectives were to compare the characteristics of the spotwelded and riveted bulkheads.
CONVAIR - SD

TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

A. FUSELAGE CANTED BULKHEAD - STATIC AND FATIGUE TESTS

II. SUMMARY

Two bulkhead specimens, one riveted and one spotwelded, were tested. Each specimen was static tested to 128% design ultimate load at 800 F and then fatigue tested to failure.

The spotwelded assembly failed after 37,200 cycles of 44.5% design ultimate skin shear load at 800 F. The riveted assembly sustained 92,629 cycles of 44.5% design ultimate load, plus 5,000 cycles of 53.2% design ultimate, and failed after 33,000 cycles of 66.6% design ultimate load. The riveted specimen was overheated and repaired in one area after 12,003 cycles at 800 F.
A. FUSELAGE CANTED BULKHEAD - STATIC AND FATIGUE TESTS

III. DESCRIPTION OF TEST SPECIMENS AND METHOD OF TESTING

1. Test Specimens:

Two canted bulkhead specimens were manufactured from 4Al-3Mo-1V titanium alloy at Convair-San Diego. One specimen was a spotwelded assembly, as shown in Figure A-1 (page 7) and the other riveted, as shown in Figure A-2 (page 9). Photographs before and after assembly are shown in Figures A-3 through A-6 (pages 11 through 14).

The first skin on the riveted bulkhead exhibited delayed cracking around the holes drilled for attachment of the shear load fixtures. There were approximately 72 hours between the drilling operation and observance of the cracks. Crack photographs are shown in Figures A-7, A-8, and A-9 (pages 15, 16 and 17).

It is believed that excessive hydrogen content and drill heat may have contributed to the cracking. Hydrogen analysis was made on this skin and found to have approximately 350 PPM hydrogen content. A comparison of the skins is shown in Table A-1 (page 18).

The skin was replaced prior to test.

2. Test Procedure:

The test specimens were tested in the special quartz lamp oven shown in Figures A-10 through A-14 (pages 19 through 23). The specimen temperatures were controlled by thermocouples attached to the bulkhead web. The thermocouple signal was fed into the Research, Inc. heat programmer, Figure A-15 (page 24), and matched with a calibrated signal from the function generator drum, Figure A-16 (page 25). The programmer forwarded a power demand signal to the ignitrons, Figure A-17 (page 26), which in turn controlled the power to the oven.
Figure A-2 - FUSELAGE CANTED BULKHEAD; Riveted Assembly - Engineering Drawing 29-01004, Sheet 2 of 2
Figure A-6 - FUSELAGE CANTED BULKHEAD; Spot Welded Assembly.
Figure A-8 — FUSELAGE CANTED BULKHEAD; Delayed Cracking after Drilling Operation.
Figure A-9 - FUSELAGE CANTED BULKHEAD; Delayed Cracking after Drilling Operation.
<table>
<thead>
<tr>
<th>SKIN NO.</th>
<th>SPEC NO.</th>
<th>YIELD STRENGTH (psi)</th>
<th>ULTIMATE STRENGTH (psi)</th>
<th>ELONGATION (% in 2&quot;)</th>
<th>HYDROGEN CONTENT (ppm)</th>
<th>REMARKS</th>
</tr>
</thead>
<tbody>
<tr>
<td>R-1</td>
<td>1</td>
<td>183,333</td>
<td>207,990</td>
<td>6.0</td>
<td>335</td>
<td>This skin exhibited delayed cracking after drilling for load points.</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>183,878</td>
<td>209,112</td>
<td>5.0</td>
<td>350</td>
<td></td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>184,816</td>
<td>212,941</td>
<td>4.5</td>
<td>399</td>
<td></td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>186,578</td>
<td>198,421</td>
<td>5.5</td>
<td>393</td>
<td></td>
</tr>
<tr>
<td></td>
<td>5</td>
<td>174,736</td>
<td>199,210</td>
<td>5.5</td>
<td>344</td>
<td></td>
</tr>
<tr>
<td></td>
<td>6</td>
<td>176,903</td>
<td>202,284</td>
<td>5.5</td>
<td>350</td>
<td></td>
</tr>
<tr>
<td>R-2</td>
<td>1</td>
<td>176,847</td>
<td>201,970</td>
<td>5.0</td>
<td>94</td>
<td>This skin was subjected to the test program outlined for the riveted bulkhead prior to analysis.</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>172,636</td>
<td>201,243</td>
<td>4.5</td>
<td>87</td>
<td></td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>173,529</td>
<td>201,225</td>
<td>4.5</td>
<td>96</td>
<td></td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>160,426</td>
<td>188,862</td>
<td>6.0</td>
<td>91</td>
<td></td>
</tr>
<tr>
<td></td>
<td>5</td>
<td>160,047</td>
<td>188,277</td>
<td>6.0</td>
<td>97</td>
<td></td>
</tr>
<tr>
<td></td>
<td>6</td>
<td>167,151</td>
<td>197,383</td>
<td>4.5</td>
<td>96</td>
<td></td>
</tr>
<tr>
<td>S-1</td>
<td>1</td>
<td>166,666</td>
<td>193,502</td>
<td>3.5</td>
<td>106</td>
<td>This skin was subjected to the test program outlined for the spot welded bulkhead prior to analysis.</td>
</tr>
<tr>
<td></td>
<td>2</td>
<td>177,428</td>
<td>192,857</td>
<td>4.0</td>
<td>107</td>
<td></td>
</tr>
<tr>
<td></td>
<td>3</td>
<td>160,335</td>
<td>192,737</td>
<td>3.5</td>
<td>105</td>
<td></td>
</tr>
<tr>
<td></td>
<td>4</td>
<td>160,919</td>
<td>191,827</td>
<td>4.5</td>
<td>108</td>
<td></td>
</tr>
<tr>
<td></td>
<td>5</td>
<td>167,151</td>
<td>197,383</td>
<td>4.5</td>
<td>112</td>
<td></td>
</tr>
<tr>
<td></td>
<td>6</td>
<td>167,543</td>
<td>191,812</td>
<td>5.5</td>
<td>117</td>
<td></td>
</tr>
</tbody>
</table>
Figure A-10 — FUSELAGE CANTED BULKHEAD; Static and Fatigue Test Set Up Showing the Specimen in Position in the Oven.
Figure A-11 — FUSELAGE CANTED BULKHEAD; Static and Fatigue Test Set Up Showing the Specimen in Position in the Oven.
Figure A-12 — FUSELAGE CANTED BULKHEAD; Static and Fatigue Test Set Up Showing the Specimen in Position in the Oven.
Figure A-13 — FUSELAGE CANTED BULKHEAD; Static and Fatigue Test Set Up Showing the Specimen in Position in the Oven.
Figure A-16 — HEAT PROGRAMMER CONTROL BOARD AND FUNCTION GENERATOR DRUM; With a Typical Heat Program Curve.
III. 2. **Test Procedure:** (Cont'd)

The specimen was reinforced at the fixed end with suitable doublers and angles, Figure A-18 (page 28), in order to assure sufficient strength at the specimen-fixture attachment point. The flanges were laterally supported at three points.

The specimen was loaded through fixtures attached to the skin to provide a skin shear load at each load point. Each load point was integrated into a whippletree and lever system in order that all loads could be applied with one hydraulic cylinder, Figure A-12 (page 21). The schematic diagram for design ultimate load is shown in Figure A-19 (page 29). The design ultimate reaction load parallel to Buttock Line 00.0 is shown as 1432 pounds. This reaction load on this line was measured during the static tests.

Eight strain gages (for room temperature static tests), eight calibrated dial indicators (for all static tests), and fifteen thermocouples were installed on the specimen. The locations of the strain gages and thermocouples are shown in Figures A-1 and A-2 (pages 7 and 9), and the deflection points are shown in Figure A-20 (page 30).

The following test schedule was performed in the order shown on both the spotwelded and riveted bulkheads:

a. **Static Tests** -

   Room Temperature to 80% design ultimate
   200 F to 66.6% design ultimate or design limit
   300 F to 66.6% design ultimate
   400 F to 66.6% design ultimate
   500 F to 66.6% design ultimate
   600 F to 66.6% design ultimate
   700 F to 66.6% design ultimate
   800 F to 66.6% design ultimate
   900 F to 66.6% design ultimate
   800 F to 128% design ultimate

Note: Strain data were taken at load increments in the room temperature tests. Deflection data were taken at load increments in all static tests.
Figure A-18 - FUSELAGE CANTED BULKHEAD; Specimen Reinforcement at the Fixed End.
CONVAIR, SAN DIEGO

Converted Sta. 67238
Canted Fuselage Bulkhead

NET LOAD "B"

28,980 In. #

1367#

781#

60.8 (23.7)

Resistant

23.7 In. (#6-16)

"P" REACTIONS

43,800 In. #

1704#

660#

33.3 In. (#0-6)

29.9

Point "O" (1432#) Free to Rotate About "P"

Figure A-19. SCHEMATIC DIAGRAM FOR DESIGN ULTIMATE TEST LOAD

29
Note:
Arrow indicates Positive Deflection On Curves.

Figure A-20 — DEFLECTION-POINT LOCATIONS.
III. 2. **Test Procedure**: (Cont'd)

b. **Fatigue Tests** -

   Room Temperature - 2,500 cycles - 44.5% design ultimate
   200 F - 2,500 cycles - 44.5% design ultimate
   400 F - 2,500 cycles - 44.5% design ultimate
   600 F - 2,500 cycles - 44.5% design ultimate
   800 F - 40,000 cycles - 44.5% design ultimate

c. **Additional Fatigue Tests - Riveted Bulkhead Only** -

   800 F - 52,000 cycles - 44.5% design ultimate
   800 F - 5,000 cycles - 53.4% design ultimate
   800 F - 33,000 cycles - 66.6% design ultimate

   Fatigue tests were conducted at a rate of 40 load cycles per minute.
A. FUSELAGE CANTED BULKHEAD - STATIC AND FATIGUE TESTS

IV. TEST RESULTS

The deflection and permanent set results are plotted in Figures A-21 through A-60 (pages 34 through 73). Figures A-61 through A-63 (pages 74 through 76), Spotwelded Bulkhead, and Figures A-64 through A-66 (pages 77 through 79), Riveted Bulkhead, show comparative deflection curves at various temperatures. Figures A-67 through A-72 (pages 80 through 85) show a comparison of the two bulkhead specimens at the indicated temperatures.

Room temperature strain gage data are plotted vs load in Figure A-73 (page 86), Riveted Bulkhead. Figures A-74 through A-76 (pages 87 through 89) show the reaction load parallel to buttock line 00.00 for both specimens.

A brief summary of the spotwelded bulkhead test results is given in Tables A-2 and A-3 (pages 90 and 91), and of the riveted bulkhead tests, in Tables A-4 and A-5 (pages 92 and 93).

Specimen failure photographs are shown in Figures A-77 through A-83 (pages 94 through 100), Spotwelded, and Figures A-84 through A-92 (pages 101 through 109), Riveted.

A comparison of the deflection characteristics of the two assemblies at various temperatures showed them to be very similar at all load levels. It was also noted that the deflection of each specimen was similar at all temperatures up to 800 F. A change in structural stiffness was noted at 900 F and 800 F after 900 F.

The indicated stresses at room temperature were relatively low. This, combined with the low restraining bar load, shows the assembly, including the fixed end, was more rigid than anticipated. The specimen did not fail at 128% design ultimate, but had some slight permanent web buckles. The deflection curves showed the specimens were approaching yield strength at 100% design ultimate. The spotwelded specimen had one crack in the spotweld after the static tests.

The riveted bulkhead had a better fatigue life than the spotwelded assembly.
Figure A-21 DEFLECTION AND PERMANENT SET AT ROOM TEMPERATURE; Static Load Test
Figure A-22. DEFLECTION AND PERMANENT SET AT ROOM TEMPERATURE; Static Load Test
TEST NO. 2(a) - SPOT WELDED BULKHEAD ASSEMBLY

Figure A-23  DEFLECTION AND PERMANENT SET AT 200 F; Static Load Test
Figure A-24. DEFLECTION AND PERMANENT SET AT 200 F; Static Load Test
TEST NO. 2 (b) - SPOT WELDED BULKHEAD ASSEMBLY

Figure A-25  DEFLECTION AND PERMANENT SET AT 300 F; Static Load Test
Figure A-26. DEFLECTION AND PERMANENT SET AT 300 F;
Static Test Load
Figure A-27 — DEFLECTION AND PERMANENT SET AT 400F; Static Load Test.
TEST No. 2 (c) SPOT WELDED BULKHEAD ASSEMBLY

Figure A-28 DEFLECTION AND PERMANENT SET AT 400 F;
Static Load Test
Figure A- 29 — DEFLECTION AND PERMANENT SET AT 500°F; Static Load Test.
CONVAIR, SAN DIEGO

TEST NO. 2(d)-SPOT WELDED BULKHEAD ASSEMBLY

Figure A-30 DEFLECTION AND PERMANENT SET AT 500 F; Static Load Test

43
Figure A-31  DEFLECTION AND PERMANENT SET AT 600 F; Static Load Test
CONVAIR, SAN DIEGO

TEST NO. 2(e) - SPOT WELDED BULKHEAD ASSEMBLY

Figure A-32  DEFLECTION AND PERMANENT SET AT 600 F  Static Load Test
Figure A-33  DEFLECTION AND PERMANENT SET AT 700 F; Static Load Test.
Figure A-34  DEFLECTION AND PERMANENT SET AT 700 F;
Static Load Test
Figure A-35 – DEFLECTION AND PERMANENT SET AT 800°F; Static Load Test
Figure A-36  DEFLECTION AND PERMANENT SET AT 800 F;
Static Load Test

49
Figure A-37  DEFLECTION AND PERMANENT SET AT 900 F; Static Load Test
Figure A-38  DEFLECTION AND PERMANENT SET AT 900 F;
Static Load Test
TEST NO. 3 - SPOT WELDED BULKHEAD ASSEMBLY
READINGS WERE DISCONTINUED AT 100% ULTIMATE
DESIGN LOAD. LOADING WAS CONTINUED TO 128%
ULTIMATE DESIGN LOAD WITHOUT FAILURE.

FIGURE A-39. DEFLECTION AND PERMANENT
SET AT 800°F; After 900°F - 128% Design Ultimate
Static Test
TEST NO. 3 - SPOT WELDED BULKHEAD ASSEMBLY
READINGS WERE DISCONTINUED AT 100% ULTIMATE
DESIGN LOAD LOADING WAS CONTINUED TO 128%
ULTIMATE DESIGN LOAD WITHOUT FAILURE.

SEE FIGURE A-20 FOR DEFLECTION
POINTS AND DIRECTION.

Figure A-40  DEFLECTION AND PERMANENT SET AT 800 F;
After 900 F - 128% Design Ultimate Static Test
Figure A-41  DEFLECTION AT ROOM TEMPERATURE; Static Load Test
CONVAIR, SAN DIEGO

TEST NO. 1 RIVETED BULKHEAD ASSEMBLY

Figure A-42 DEFLECTION AT ROOM TEMPERATURE;
Static Load Test

SEE FIGURE A-20 FOR DEFLECTION POINTS AND DIRECTION.

Deflection in inches

PERCENT DESIGN ULTIMATE LOAD

DEFLECTION IN INCHES

0.025 0.050 0.075 0.1 0.2 0.3 0.4

0 10 20 30 40 50 60 70 80 90 100

POINT #4
POINT #5
POINT #6
POINT #7
POINT #8

55
Figure A-43  DEFLECTION AT 200 F; Static Load Test
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TEST NO. 2A RIVETED BULKHEAD ASSEMBLY

Figure A-44  DEFLECTION AT 200 F; Static Load Test

SEE FIGURE A-20 FOR DEFLECTION POINTS AND DIRECTION.
Figure A-45 — DEFLECTION AT 300F; Static Load Test.
CONVAIR, SAN DIEGO

TEST NO. 2B RIVETED BULKHEAD ASSEMBLY

Figure A-46  DEFLECTION AT 300 F; Static Load Test
TEST NO. 2C RIVETED BULKHEAD ASSEMBLY

![Graph](image)

- **PERCENT DESIGN - ULTIMATE LOAD**
- **DEFLECTION IN INCHES**

Legend:
- ○ POINT #1
- Δ POINT #2
- ■ POINT #3

DEFLECTION

SEE FIGURE A-29 FOR DEFINITION POINTS AND DIRECTION.

Figure A-47  DEFLECTION AT 400 F; Static Load Test
TEST NO. 2C RIVETED BULKHEAD ASSEMBLY

DEFLECTION AT 400 F; Static Load Test

Figure A-48
TEST NO. 2 D RIVETED BULKHEAD ASSEMBLY

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DEFLECTION AT 500 F: Static Load Test

Figure A-49

SEE FIGURE A-20 FOR DEFLECTION POINTS AND DIRECTION.
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TEST NO. 2D RIVETED BULKHEAD ASSEMBLY

Figure A-50  DEFLECTION AT 500 F; Static Load Test

63
FEST NO. 21: RIVETED END LOAD ASSEMBLY

PERCENT DESIGN - ULTIMATE LOAD

DEFLECTION

SEE FIGURE A-20 FOR DEFLECTION POINTS AND DIRECTION.

NOTE: STATIC LOAD TEST

AT 600°F
Figure A-52  DEFLECTION AT 600 F; Static Load Test
Figure A-54  DEFLECTION AT 700 F; Static Load Test
Figure A-55 — DEFLECTION AT 800°F; Static Load Test.
Figure A-56  DEFLECTION AND PERMANENT SET AT 800 F;  
Static Load Test
Figure A-57 — DEFLECTION AT 900°F; Static Load Test.
Figure A-58  DEFLECTION AND PERMANENT SET AT 900 F;
Static Load Test
TEST NO. 3 - RIVETED BULKHEAD ASSEMBLY
(Readings Were Discontinued at 100% Ultimate Design Load. Loading Was Continued to 128% Ultimate Design Load Without Failure.

Figure A-59 — DEFLECTION AT 800°F; After 900°F - 128% Design Ultimate Static Test.
TEST NO. 3 - RIVETED BULKHEAD ASSEMBLY
(Readings Were Discontinued At 100% Ultimate Design Load. Loading Was Continued to 128% Ultimate Design Load Without Failure.

Figure A-60 — DEFLECTION AT 800°F AFTER 900°F; 128% Design Ultimate Static Test.
FIGURE A-61. DEFLECTION AT POINT #1 COMPARED AT SIX TEMPERATURES; FROM ROOM TEMPERATURE TO 900°F.
CONVAIR, SAN DIEGO

SPOT WELDED BULKHEAD ASSEMBLY

LEGEND
- - - - DEFLECTION CURVE AT ROOM TEMPERATURE
- - - - DEFLECTION CURVE AT 300°F
- - - - DEFLECTION CURVE AT 500°F
- - - - DEFLECTION CURVE AT 700°F
- - - - DEFLECTION CURVE AT 900°F
- - - - DEFLECTION CURVE AT 800°F, after 900°F

DEFLECTION CURVES AT SIX TEMPERATURES; From Room Temperature To 900°F

FIGURE A-62. DEFLECTION AT POINT #5 COMPARED AT SIX TEMPERATURES; From Room Temperature To 900°F
FIGURE A-63. DEFLECTION AT POINT #7 COMPARED AT SIX TEMPERATURES; FROM ROOM TEMPERATURE TO 900°F
FIGURE A-64. DEFLECTION AT POINT #1
COMPARED AT SIX TEMPERATURES; From
Room Temperature To 900F
RIVETED BULKHEAD ASSEMBLY

LEGEND

- - - - = DEFLECTION AT R. T.
- - - - - - = DEFLECTION AT 300°F
- - - - - - = DEFLECTION AT 500°F
- - - - - - = DEFLECTION AT 700°F
- - - - - - = DEFLECTION AT 900°F
- - - - - = DEFLECTION AT 800°F, after 900°F

DEFLECTION IN INCHES

% OF DESIGN ULTIMATE LOAD

FIGURE A-65. DEFLECTION AT POINT #5
COMPARED AT SIX TEMPERATURES; From
Room Temperature To 900°F
FIGURE A-66. DEFLECTION AT POINT #7 COMPARED AT SIX TEMPERATURES; FROM ROOM TEMPERATURE TO 900F
Figure A-67. Comparison of deflections between riveted and spot welded bulkhead assemblies; At Room Temperature
FIGURE A-68. COMPARISON OF DEFLECTIONS BETWEEN RIVETED AND SPOT WELDED BULKHEAD ASSEMBLIES AT 300F
FIGURE A-69. COMPARISON OF DEFLECTIONS BETWEEN RIVETED AND SPOT WELDED BULKHEAD ASSEMBLIES; AT 500°F
FIGURE A-70. COMPARISON OF DEFLECTIONS BETWEEN RIVETED AND SPOT WELDED BULKHEAD ASSEMBLIES; At 700F
FIGURE A-71. COMPARISON OF DEFLECTIONS BETWEEN RIVETED AND SPOT WELDED BULKHEAD ASSEMBLIES; AT 900F
FIGURE A-72. COMPARISON OF DEFLECTIONS BETWEEN RIVETED AND SPOT WELDED BULKHEAD ASSEMBLIES; AT 800°F AFTER 900°F
FIGURE A-73. STRAIN INDICATIONS VS % ULTIMATE LOAD ON THE RIVETED BULKHEAD ASSEMBLY; Room-Temperature Static Test
FIGURE A-74. COMPARISON OF RESTRAINING-BAR REACTION LOAD WITH AND WITHOUT WEB LATERAL RESTRAINT
FIGURE A-75. BAR REACTION LOAD
VS % ULTIMATE LOAD; AT SIX TEMPERATURES

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SPOT WELDED BULKHEAD ASSEMBLY

800F AFTER 900F
(FAILURE OR 128% ULTIMATE)

ROOM TEMP.

% OF DESIGN ULTIMATE LOAD

LOAD (lbs.)

0 100 200 300 400 500 600

0 10 20 30 40 50 60 70 80 90 100
FIGURE A-76. RESTRAINING BAR REACTION LOAD VS % ULTIMATE LOAD AT SIX TEMPERATURES
### TABLE A-2 - FUSELAGE CANTED BULKHEAD - SPOTWELDED

#### STATIC TEST OUTLINE AND RESULTS SUMMARY

<table>
<thead>
<tr>
<th>TEST</th>
<th>TEMPERATURE (F)</th>
<th>MAX. LOAD (% ULT.)</th>
<th>DEFLECTION PLOTS</th>
<th>STRAIN PLOTS</th>
<th>FAILURE PHOTOS</th>
<th>REMARKS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Static</td>
<td>R.T.</td>
<td>80.0</td>
<td>(1)</td>
<td>(1)</td>
<td>81</td>
<td>Severe elastic buckles in the inner and outer flanges - spotweld failure. Lateral restraints added to flanges. See Figure 74 for comparison of restraining bar load without and with lateral restraint on specimen flanges. Spotweld repaired - see Figure 81. Three-bolt repair only on second stiffener from free end.</td>
</tr>
<tr>
<td>Static</td>
<td>200</td>
<td>66.6</td>
<td>21 &amp; 22</td>
<td></td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>300</td>
<td>66.6</td>
<td>23 &amp; 24</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>400</td>
<td>66.6</td>
<td>25 &amp; 26</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>500</td>
<td>66.6</td>
<td>27 &amp; 28</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>600</td>
<td>66.6</td>
<td>29 &amp; 30</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>700</td>
<td>66.6</td>
<td>31 &amp; 32</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>800</td>
<td>66.6</td>
<td>33 &amp; 34</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>900</td>
<td>66.6</td>
<td>35 &amp; 36</td>
<td>-</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
<td>Static</td>
<td>800</td>
<td>128.0</td>
<td>37 &amp; 38</td>
<td>-</td>
<td>77</td>
<td>Small crack in spotweld at opposite end of stiffener repaired above. See repair only in Figure 77. Web and flanges severely buckled at load but only a small amount remained after load removed. Test discontinued since the specimen reached the limit of the flange restraints.</td>
</tr>
</tbody>
</table>

**Note:** (1) Data not presented because of flange and web deformations - insufficient lateral support.
<table>
<thead>
<tr>
<th>TEST</th>
<th>TEMP (F)</th>
<th>LOAD (% ULT.)</th>
<th>FATIGUE CYCLES AT CONDITION</th>
<th>TOTAL NO. FATIGUE CYCLES</th>
<th>FAILURE PHOTOS FIGURE NOS.</th>
<th>REMARKS</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fatigue</td>
<td>R.T.</td>
<td>44.5</td>
<td>2500</td>
<td>2500</td>
<td></td>
<td>No additional failures</td>
</tr>
<tr>
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<td>200</td>
<td>44.5</td>
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<tr>
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<td>400</td>
<td>44.5</td>
<td>2500</td>
<td>7500</td>
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<tr>
<td>Fatigue</td>
<td>600</td>
<td>44.5</td>
<td>2500</td>
<td>10,000</td>
<td>77,78,80</td>
<td>Specimen inspected - crank in spotweld at inboard flange and first stiffener from free end - Ref Figure 78. Repair shown in Figure 81. Also stiffener failed as shown in Figure 77.</td>
</tr>
<tr>
<td>Fatigue</td>
<td>800</td>
<td>44.5</td>
<td>3795</td>
<td>13,795</td>
<td></td>
<td>Crack started across second stiffener from free end at second spot.</td>
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<td>Fatigue</td>
<td>800</td>
<td>44.5</td>
<td>4095</td>
<td>17,890</td>
<td></td>
<td>Above crack completely across stiffener allowing inner flange to roll.</td>
</tr>
<tr>
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<td>800</td>
<td>44.5</td>
<td>19,310</td>
<td>37,200</td>
<td>79 thru 84</td>
<td>Major specimen failure - would not carry load. Inner flange and web failed. Plus additional single spot failures.</td>
</tr>
<tr>
<td>TEST</td>
<td>TEMP (F)</td>
<td>MAX. LOAD (% UTL.)</td>
<td>DEFLECTION PLOTS FIGURE NO.</td>
<td>STRAIN PLOTS FIGURE NO.</td>
<td>FAILURE PHOTOS FIGURE NO.</td>
<td>REMARKS</td>
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<td>-------------------</td>
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<td>------------------------</td>
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<td>---------------------------</td>
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<td>R.T.</td>
<td>80.0</td>
<td>41 &amp; 42</td>
<td>73</td>
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<td>No failures noted</td>
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<td>200</td>
<td>66.6</td>
<td>43 &amp; 44</td>
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<td>No failures noted</td>
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<td>500</td>
<td>66.6</td>
<td>49 &amp; 50</td>
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<td>51 &amp; 52</td>
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<tr>
<td>Static</td>
<td>700</td>
<td>66.6</td>
<td>53 &amp; 54</td>
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<td>66.6</td>
<td>57 &amp; 58</td>
<td>-</td>
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<td>128.0</td>
<td>59 &amp; 60</td>
<td>-</td>
<td></td>
<td>No failures noted. Similar buckles to those on spot welded bulkhead.</td>
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<td>TEST</td>
<td>TEMP (F)</td>
<td>MAX. LOAD (% ULT.)</td>
<td>NO. OF FATIGUE CYCLES AT CONDITION</td>
<td>TOTAL NO. FATIGUE CYCLES</td>
<td>FAILURE PHOTOS</td>
<td>FIGURE NO.</td>
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<td>R.T.</td>
<td>44.5</td>
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<td>4000</td>
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Figure A-77 — SPOT WELDED FUSELAGE CANTED BULKHEAD ASSEMBLY;
Bolt and Washer Repair for Spot Weld Crack, During Static Test.
Fatigue Failure Occurred After 10,000 Cycles (-33 Stiffener).
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Spot Weld Crack After 10,000 Fatigue Cycles (-31 Stiffener).
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Spot Weld Crack After 37,200 Cycles.
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Spot Weld Crack After 37,200 Cycles.
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Final Failure at 37,200 Cycles.
Figure A-83 — SPOT WELDED FUSELAGE CANTED BULKHEAD ASSEMBLY;
Final Failure at 37,200 Cycles.
Figure A-85 - RIVETED FUSELAGE CANTED BULKHEAD ASSEMBLY
After Overheating at 22, 629 Cycles.
Figure A-87 — RIVETED FUSELAGE CANTED BULKHEAD ASSEMBLY; After Overheating at 22,629 Cycles.
Figure A-88 — RIVETED FUSELAGE CANTED BULKHEAD ASSEMBLY;
Web Fatigue Failure After 140,629 Total Cycles.
Figure A-90 - RIVETED FUSELAGE CANTED BULKHEAD ASSEMBLY;
Inner Flange Fatigue Failure After 140,629 Cycles.
Figure A-91 — RIVETED FUSELAGE CANTED BULKHEAD ASSEMBLY;
Inner Flange Fatigue Failure After 140,629 Cycles.
Figure A-92 — RIVETED FUSELAGE CANTERED BULKHEAD ASSEMBLY
Inner Flange Fatigue Failure After 140,629 Cycles.
TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

A. FUSELAGE CANTED BULKHEAD - STATIC AND FATIGUE TESTS

V. SUMMARY OF RESULTS

1. The riveted and spotwelded assemblies had similar deflection characteristics in the static tests.

2. Both assemblies sustained 128% design ultimate load at 800 F without a major failure.

3. The riveted canted bulkhead had a better fatigue life than the spotwelded bulkhead.
VI. CONCLUSIONS AND DISCUSSION

From a structural design viewpoint the element tests on the titanium bulkheads were the most significant, for they furnished evidence of the superiority of titanium over aluminum for certain applications. This observation is possible because of the opportunity to compare the test bulkheads with actual aluminum construction presently used on the F-106, a typical advanced design Mach 2 Interceptor. The bulkhead test specimens were fabricated to the loft lines of the F-106 canted frame at Station 672.38. This is a major bulkhead that supports the aft fin spar for which the loads and design parameters are well defined.

The F-106 aluminum designed bulkhead weighs 26.9 pounds, not including the fin spar attach fitting. Approximately six pounds of this weight is for skin splices, miscellaneous clips, and the weight of a small portion of the frame that was designed to resist other design loads. In contrast, weight of the part that was fabricated from titanium for the test program was 8.1 pounds (the total bulkhead, less the fin spar attach fitting, would weigh 16.2 pounds). It can be seen that the weight saving resulting from using titanium, for the particular part in question, is approximately 29% assuming that the margin of safety is the same for both parts.

In the F-106 aft engine compartment heat was significant, and the design of all frames in the area was predicated on 272 F for 40 hours duration. The properties of 2024 T4 aluminum alloy are approximately 88% of room temperature properties. There is a somewhat similar reduction for 4Al-3Mo-1V titanium; however, the properties of aluminum deteriorate much more rapidly with increase in temperature. In the event of an engine shroud failure or fire in the engine compartment, the titanium structure would have an additional fail safe type of advantage over the aluminum construction.

Either the spot welded or riveted structure seems to be adequate from a fatigue standpoint for the particular part in question.
VI. CONCLUSIONS AND DISCUSSION (Cont'd)

The F-106 preliminary flight loads spectra give the following lateral
gust load factors:

- 12,769 cycles at 25% limit load
- 2,361 cycles at 50% limit load
- 218 cycles at 75% limit load
- 19 cycles at 100% limit load
- 1 cycle at 125% limit load

The above figures are preliminary data from a fatigue analysis being con-
ducted on the F-106. The number of cycles is predicated on an interceptor
life of 4000 flight hours.

The spot welded specimen sustained 37,200 cycles of 66.7% limit load
at various elevated temperatures. Since most fatigue damage occurs as a
result of a high number of cycles at low load levels, it is obvious that the
fatigue life of the titanium specimens would have been adequate for the
bulkhead in question.

The stress at the point of failure was calculated to be 44,300 lbs/sq. in.
at ultimate load. The stress level during the fatigue cycle varied from 0 to
19,200 lbs/sq. in. at an average temperature of slightly less than 800 F.
The only available information on fatigue for this alloy shows a maximum
fatigue stress of approximately 50,000 lbs/sq. in. for a life of 40,000 cycles.
This was with a notched specimen ($K_t = 3.5$) at room temperature with the
load ratio = 0.6 (maximum stress minus mean stress divided by the mean
stress), Ref: Titanium Engineering Bulletin No. 8, Titanium Metals Corp. of
America, 233 Broadway, New York. The difference between the 19,700
lbs/sq. in. and the 50,000 lbs/sq. in. could be due to several reasons. The
temperature undoubtedly reduced the fatigue life, the load ratio tested to
(1.0) is more severe, and the notch factor was not accurately known. The
failure occurred where a reinforcing angle ended and local stresses were
probably much higher than the calculated stresses. However, the riveted
bulkhead did not suffer a fatigue failure in this area. If we neglect scatter
(usually very large in this type of test) it would seem that the spot welds
have a much larger notch effect than do rivets, for this particular alloy.
B. WING LEADING EDGE SECTIONS - STATIC AND FATIGUE TESTS

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   1. Test Specimens
   2. Test Procedure
   3. Test Loads

IV. TEST RESULTS AND DISCUSSIONS

V. SUMMARY OF RESULTS

VI. CONCLUSIONS AND DISCUSSION

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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

B. WING LEADING EDGE SECTIONS - STATIC AND FATIGUE TESTS

I. INTRODUCTION

This report was prepared to present the results of static and fatigue tests, at temperature, of titanium wing leading edges. The Wing Leading Edge Assembly is basically a conversion to titanium of the F-106A Interceptor Part 8-18205, Leading Edge Assembly. This part was redesigned to take advantage of the improvement in properties of titanium over aluminum at higher temperatures. In this case, Ti-4Al-3Mo-1V titanium alloy replaced 7075-T6 aluminum alloy.

Two titanium Wing Leading Edge Assemblies were tested; one statically, one in fatigue. The static specimen was tested in 20% steps to limit load at room temperature, 200 F, 300 F, 400 F, 500 F, 600 F, 700 F, 800 F and 900 F. A failure test was then conducted at 800 F. The fatigue specimen was tested at 66.7% of limit load for 160,000 cycles; 2500 cycles each at room temperature, 200 F, 400 F and 600 F. and 150,000 cycles at 800 F.

These tests were conducted to determine:

1. The load carrying characteristics of a titanium Wing Leading Edge at various temperatures up through 900 F.

2. The ultimate strength of the assembly at 800 F.

3. The fatigue life of the assembly at 800 F.

4. The comparative strengths of the spot welded and riveted halves.
TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

B. WING LEADING EDGE SECTIONS - STATIC AND FATIGUE TESTS

II. SUMMARY

The statically tested specimen of the Wing Leading Edge Assembly withstood all tests with no apparent failure. The same assembly specimen, when tested to failure at 800 F, failed at 189.5% of limit load (126.3% of design ultimate). The upper skin failed as a column near the attachment fixture, pulling several spotwelds and rivet heads.

The fatigue specimen showed many cracks in the internal structure of the spotwelded portion; the riveted portion had only a few popped rivet heads.
III. DESCRIPTION OF TEST SPECIMENS AND METHOD OF TESTING

1. Test Specimens:

Two test specimens were manufactured according to Convair Engineering Drawing 29-01007, Figure B-1 (page 121). The sheet metal parts were made from Ti-4Al-3Mo-1V alloy. The skins on one half of each specimen were spotwelded to the ribs. The skins on the other half were riveted using countersunk monel rivets. Therefore, each of the two test specimens was effectively two specimens; one spotwelded and one riveted. The upper skin was reduced in thickness between the ribs by chemical etching.

2. Test Procedure:

The test specimens were attached to a rigid fixture in a manner similar to an actual installation, using 3/16-inch bolts with one inch spacing.

The test load was distributed over 66 points of application to simulate air loads using a whipple tree system shown in Figures B-2 and B-3 (pages 123 and 124). The load was applied by eyebolts through the skin and ribs into steel blocks cushioned by pieces of asbestos blanket.

The maximum static load for all conditions, except the failing load, was 6650 pounds, which is limit load. The load was applied in 20 per cent increments and deflections taken. Permanent set was measured at 10 per cent load after each of the increments. Deflections were taken at the mid-point of the front edge and at the quarter points, i.e., at points halfway between the mid-point and the edges. Eight strain gages were placed as shown in Table B-1 (page 125) for the room temperature static test.

The complete static load sequence was run at room temperature, 200 F, 300 F, 400 F, 500 F, 600 F, 700 F, 800 F, and 900 F. A similar load sequencing was used during the failure test, which was conducted at 800 F after 900 F.
Figure B-1 - WING LEADING-EDGE ASSEMBLY;
Engineering Drawing 29-01007
3. ALL FASTENERS INSTALLED PER CONVAIR 22-0001
2. WELD PER MIL-W-4520, EXCEPT PENETRATION TO BE 35-7600.
1. THIS PART DESIGNED FOR DYNAMIC TESTING PER CONVAIR 0-5030

NOTES:
Figure B-2 — FRONT VIEW OF TEST SET UP; With Oven Open
Figure B-3 — SIDE DETAIL OF TEST SET UP; Showing Load Attachment Points.
TABLE B-1 - ROOM-TEMPERATURE STRAIN GAGE DATA - STATIC TEST OF
WING LEADING-EDGE ASSEMBLY

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ATTACHMENT SIDE

PT. 8 SEE NOTE BELOW
PT. 5 SHOWN
PT. 6 BOTTOM SIDE

PT. #7 ON RIB SIDE BY JOGGLE

PT. 1 SHOWN, PT. 2 BOTTOM SIDE
PT. 3 SHOWN, PT. 4 BOTTOM SIDE
RIBS REF.

DIAL 1 SPOTWELDED
DIAL 2 RIVETED
DIAL 3

NOTE: POINT 8 WAS BETWEEN TWO SPOT WELDS

TOP VIEW
III. 2. **Test Procedure**: (Cont'd)

The fatigue test load was \( \frac{2}{3} \) of limit load (4433 lbs.). This load was applied and removed about 33 times per minute. Loadings were applied 2500 times at room temperature, 200 F, 400 F, and 600 F. At 800 F, the loadings were applied 150,000 times. All loads were applied hydraulically using a hydraulic cylinder and pump.

The loading cylinder was placed in a calibrated Baldwin Lima Hamilton standard universal testing machine prior to test. Pressure was applied to the cylinder by the hand pump. A pressure vs load curve was thereby obtained. By using the same pressure gage and cylinder, the loads could be duplicated accurately during subsequent testing.

Heat was applied to the specimen by means of quartz heat lamps in a contoured, reflective oven, Figure B-2 (page 123). The temperature was controlled by a Research, Inc. heat programmer. Four channels of heating were used: two above and two underneath the specimen. A channel consists of a lamp bank, a controller, and a feedback thermocouple attached to the specimen under the lamp bank. The accuracy of the temperature is dependent only on the accuracy of the thermocouple.

3. **Test Loads**:  

Test load for the static test was design limit load which is defined in Convair Report S-Gen-84 "Titanium Development Program" as 6650 pounds with a uniform distribution. This was based on condition 1610 from Convair Report ZS-8-136 "Static Test Load Summary 106A."

Fatigue load was 66.6% of limit load.
The static test Wing Leading Edge Assembly failed at 12,600 pounds which is 189.5% of limit load or 126.3% of design ultimate. Failure occurred in the upper skin which failed as a column, pulling out spotwelds, popping rivet heads, and bulging outward, Figures B-4 and B-5 (pages 128 and 129).

Strain gage readings taken during the room temperature static test are to be found in Table B-1 (page 125).

Deflection and set curves for the various test temperatures are shown in Figures B-6 through B-16 (pages 130 through 140). The data from the midpoint were used. This point approximates the average of the three. The quarter point falling in the middle of the riveted section was about 5% higher, the spotwelded, about 5% less.

The fatigue test Wing Leading Assembly withstood a total of 160,000 cycles of $2/3$ limit load. However, the spotwelded half suffered considerable damage. The first spotweld failures occurred at 29,855 cycles. The failure was heard at that time, but damage could not be seen. At 72,000 cycles, the failure of the spotwelds could be seen. They were mostly internal structure welds. By 160,000 cycles, there was extensive internal damage as well as damage to the skin as shown in Figures B-17 and B-18 (pages 141 and 142).

The first rivet head popped in the skin at 55,338 cycles. At 160,000 cycles, the riveted half had lost several rivet heads in the upper skin as shown in Figure B-19 (page 143). There was also some damage to the ribs next to the spotwelded half as shown at the right in Figures B-17 and B-18.
Figure B-4 — WING LEADING-EDGE ASSEM; Top View of Specimen Showing Rivet Failures.
Figure B-5 — WING LEADING-EDGE ASSEM; End View of Specimen Showing Spotweld Failures.
Figure B-6 — Deflection and Permanent Set at Room Temperature.
Figure B-7 — DEFLECTION AND PERMANENT SET AT 200F.
WING LEADING-EDGE ASSEMBLY

DEFLECTION CURVE

PERMANENT-SET CURVE

% OF LIMIT LOAD (2/3 DESIGN ULTIMATE)

DEFLECTION IN INCHES

(DEFLECTION & SET WERE MEASURED AT CENTER OF FRONT EDGE)

Figure B-8 — DEFLECTION AND PERMANENT SET AT 300F.
WING LEADING-EDGE ASSEMBLY

% OF LIMIT LOAD (2/3 DESIGN ULTIMATE)

DEFLECTION IN INCHES

PERMANENT-SET CURVE

DEFLECTION CURVE

(DEFLECTION & SET WERE MEASURED AT CENTER OF FRONT EDGE)

Figure B-10 — DEFLECTION AND PERMANENT SET AT 500F.
WING LEADING-EDGE ASSEMBLY

DEFLECTION CURVE

PERMANENT-SET CURVE

% OF LIMIT LOAD (2/3 DESIGN ULTIMATE)

DEFLECTION IN INCHES

(DEFLECTION & SET WERE MEASURED AT CENTER OF FRONT EDGE)

Figure B-11 — DEFLECTION AND PERMANENT SET AT 600°F.
Figure B-12 — DEFLECTION AND PERMANENT SET AT 700°F.
Figure B-13 — DEFLECTION AND PERMANENT SET AT 800°F.
Figure B-14 — DEFLECTION AND PERMANENT SET AT 900°F.
Figure B-16 — DEFLECTION AND PERMANENT SET DURING FAILURE TEST AT 800°F.
Figure B-17 — SPOT WELDED SECTION OF WING LEADING-EDGE ASSEMBLY; Oblique View Showing Fatigue Failures.
Figure B-18 — SPOT WELDED SECTION OF WING LEADING-EDGE ASSEMBLY:
Rear View Showing Fatigue Failures.
V. SUMMARY OF RESULTS

1. The load carrying characteristics of the Wing Leading Edge Assembly, as determined by the deflection and set curves, are not materially affected by temperatures up through 900 F, although deflections increased with temperature.

2. The ultimate strength of the Wing Leading Edge Assembly was 126.3% of design ultimate load.

3. The spotwelded section failed first, with failure progressing to the riveted side.

4. After 160,000 cycles of 66.7% limit load, the 29-01007 Leading Edge Assembly would still withstand the load.

5. The riveted portion was in good condition except for several popped rivet heads and cracks in the rib adjacent to the spotwelded portion.

6. The spotwelded portion had many internal failures and was gradually transferring more and more of its load through the skins into the riveted section, causing the failures in the adjacent riveted rib as mentioned above.
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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

B. WING LEADING EDGE SECTIONS - STATIC AND FATIGUE TESTS

VI. CONCLUSIONS AND DISCUSSION

The leading edge assembly which was fabricated from titanium for the structural tests weighed approximately 17.28 pounds. This is almost the identical weight of the presently used aluminum structure on the F-106 (calculated to be 16.1 pounds). The titanium skin was chemically milled between the ribs so that the weight of the specimen would approach the weight of the original design.

The condition that the specimen was tested to is a 7g limit, steady state pull up, at subsonic speed, at sea level. Temperature was not a design parameter. The specimen showed a margin of safety of 26.3 even though it was tested at 800 F. The fatigue life of the specimen far exceeded any reasonable design requirements. With these facts in mind, it can be reasonably assumed that a substantial weight saving could have been experienced had the original design been based on titanium. Leading edge designs in the near future will be influenced by aerodynamic heating generated by Mach 3.0 and above. Titanium as a material for leading edges has been demonstrated to be a very useful material under these operating conditions.
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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

C. DUCT-ENGINE BLEED AIR - STATIC, FATIGUE AND BURST TESTS

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<td>Duct-Engine Bleed Air - Riveted and Brazed Static Specimen - Burst Test Failure</td>
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I. INTRODUCTION

Six titanium alloy ducts similar to the F-106 engine bleed air ducts were manufactured by three methods, two specimens by each method. One of each type was subjected to static tests at 800 F and the other, to fatigue tests at 800 F. After completion of the above tests, each duct was burst-tested at room temperature.

II. SUMMARY

The program objective was to determine the structural integrity of air ducts, made from titanium alloy by three different methods. Since all specimens met or exceeded initial test requirements, the objective of the program was accomplished.
C. DUCT-ENGINE BLEED AIR - STATIC, FATIGUE AND BURST TESTS

III. DESCRIPTION OF TEST SPECIMENS AND METHOD OF TESTING

1. Test Specimens:

Six engine bleed air ducts were manufactured from Ti-4Al-3Mo-1V titanium alloy. These specimens simulated the engine bleed air ducts used in the Convair F-106 interceptor. The production ducts are manufactured from type 321 or 347 corrosion resistant steel, using fusion and seam welded construction.

The three manufacturing methods used on the titanium test specimens were:

a. Fusion butt welded with seam welded end flanges, Figure C-1 (page 153).

b. Seal welded, Figure C-2 (page 155).

c. Riveted and brazed, Figure C-3 (page 157).

It is to be noted that all specimens had fusion butt welds in some areas.

Details of the specimen manufacture are discussed in Volume IV, this report.

The test ducts were assembled using solid end caps. Sealed tubes were attached to the cap and inserted into the ducts in order to reduce the internal volume, Figure C-4 (page 159).

The caps were held in place with Marmon clamps and sealed with asbestos and steel gaskets, Figure C-5 (page 160).

A preliminary leak test was accomplished by submerging the duct in water and applying 90 PSIG internal air pressure.Leaks in the duct were repaired prior to test. Difficulty was experienced in obtaining a satisfactory
Figure C-1 - ENGINE-BLEED AIR DUCT; Butt Welded Assembly
Engineering Drawing 29-01005, Sheet 1 of 3
This face must be flat within .005 TTR.

**Detail B-05**

- Break sharp corners
- This face must be flat within .005 TTR.

**Detail A-04**

- Break sharp corners
- Grind down as shown under -9 1/2 flanges for seam welding

**Section D = D-04, 024**

- Weld burn
- View H-06
- View H-07
- 0.40 stock size (new)
- Burn down flanges in welding
- 0.40 stock size (new)
Figure C-2 - ENGINE-BLEED AIR DUCT; Seam Welded Assembly
Engineering Drawing 29-01005, Sheet 2 of 3
FUSION WELD PER O-05025

-23

BUTT WELD PER O-05025

3.0 OD

-21

2.03

1.18 TYP

-9

3.03 DUCT OD FOR 6 SIZE TO FIT -9

SEAM WELD PER MIL-W-6858A

CONTINUOUS FUSION WELD (TYP) PER O-05025

8.16

Figure
Figure C-2  Page 155

<table>
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<tr>
<th>Part No.</th>
<th>Description</th>
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<th>Test Date</th>
<th>Test Code</th>
<th>Test Method</th>
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<tr>
<td></td>
<td>DUCT-ENGINE SLEEVE</td>
<td>AIR 43I TITANIUM</td>
<td>SEMI WELDED</td>
<td>29-01005</td>
<td>CONV AIR</td>
</tr>
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</table>
Figure C-3 - ENGINE-BLEED AIR DUCT; Brazed Assembly
Engineering Drawing 29-01005, Sheet 3 of 3
Figure C-4 – ENGINE-BLEED AIR DUCT; Disassembled View of Test Specimen.
Figure C-5 — ENGINE BLEED AIR DUCT; View of Assembled Test Specimen.
III. 1. **Test Specimens**: (Cont'd)

Seal at the cap. It was necessary to polish the duct flanges, on a flat surface, using coarse and then fine emery paper. The cap and duct flange were then lapped together.

2. **Test Procedure**:

One specimen of each type was static tested and the other was fatigue tested.

The static and fatigue tests were run at 800°F in a steel box oven. The oven had three resistance wire heating elements supported approximately 1/2 inch from the surface of the specimen, Figure C-6 (page 162). The oven was wrapped with insulation and placed in a second steel container in order to contain the specimen in the event of an explosive failure.

Thermocouples were spot welded to the surface of the specimen in three locations. Each thermocouple controlled the heating element adjacent to it. The power was controlled by, and the temperature recorded by, a three-channel Brown Controller and Recorder. The 800°F test temperature was reached in two hours and the specimen soaked at that temperature for an additional two hours prior to the start of the test.

Test pressures were obtained with bottled, dry nitrogen and controlled by means of a gas regulator and a calibrated bourdon tube pressure gage. A schematic diagram of the pressurization system is shown in Figure C-7 (page 163).

A pair of bourdon tube pressure switches, counter, and relay were wired to automatically cycle the pressure from 15 PSIG to the maximum pressure for the fatigue test. The cycle rate was maintained at 40 cycles per minute. A schematic diagram of the cycling system is shown in Figure C-8 (page 164).

Burst tests were conducted on all specimens after completion of the static and fatigue tests. The burst tests were conducted at room temperature using hydraulic oil and a motor driven pump as the pressurization sources.
Figure C-6 — ENGINE-BLEED AIR DUCT; Specimen Installed in Oven.
Note Nitrogen Inlet Coil for Preheating Gas.
FIGURE C-7. ENGINE-BLEED AIR DUCT;
Schematic Diagram Of Pressurization & Heating Systems - Static Test
FIGURE C-8. ENGINE-BLEED AIR DUCT;
Schematic Diagram Of Pressurization
Cycling & Heating System - Fatigue Test
III. DESCRIPTION OF TEST SPECIMENS AND METHOD OF TESTING (Cont'd)

3. **Test Loads:**

   The static tests were conducted in the following order on one duct of each type:
   
   a. 245 PSIG at 800 F for 3 minutes
   
   b. 370 PSIG at 800 F for 3 minutes
   
   c. 550 PSIG at 800 F for 1 minute

   The fatigue tests were conducted in the following order on the other duct of each type:
   
   a. Static Proof 245 PSIG at 800 F for 3 minutes
   
   b. Static Proof 370 PSIG at 800 F for 3 minutes
   
   c. 100,000 cycles 245 PSIG at 800 F
   
   d. 120,000 cycles 310 PSIG at 800 F
   
   e. 30,000 cycles 370 PSIG at 800 F

   In the burst test, each specimen was subjected to an increasing hydraulic pressure at room temperature to duct failure. In some cases, a high flow was required at high pressure in order to compensate for end cap leakage.

4. **Test Results and Discussion:**

   All of the bleed air ducts satisfactorily completed the test schedule specified on the specimen drawings (reference Figures C-1 through C-3). It is to be noted that the fatigue test was changed from room temperature to 800 F.

   All but one specimen failed in the hydraulic burst test at room temperature. Typical burst failures occurred in the portions of the ducts that had been joined by fusion welds. The burst test results are presented in Table C-1 (page 166). Photographs of the specimen failures are shown in Figures C-9 through C-14 (pages 167 through 172).
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<th>Specimen Drawing</th>
<th>Assembly Type Previous Test</th>
<th>Burst Pressure (psi)</th>
<th>Remarks</th>
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<tr>
<td>Basic</td>
<td>Figure C-1</td>
<td>Butt Fusion Weld Fatigue</td>
<td>340</td>
<td>3&quot; long crack in longitudinal weld near large end. See Figure C-9.</td>
</tr>
<tr>
<td>Basic</td>
<td>&quot;</td>
<td>&quot;</td>
<td>200</td>
<td>Split on longitudinal weld. See Figure C-10</td>
</tr>
<tr>
<td>-1</td>
<td>Figure C-2</td>
<td>Seam Welded Fatigue</td>
<td>---</td>
<td>No Failure - Reached 1140 psi and cap leaks equalled hydraulic pump flow output.</td>
</tr>
<tr>
<td>-1</td>
<td>&quot;</td>
<td>&quot;</td>
<td>1100</td>
<td>See Figure C-11.</td>
</tr>
<tr>
<td>-3</td>
<td>Figure C-3</td>
<td>Riveted and Brazed Fatigue</td>
<td>740</td>
<td>Cracked but sustained 1200 psig at high flow. See Figure C-13.</td>
</tr>
<tr>
<td>-3</td>
<td>&quot;</td>
<td>&quot;</td>
<td>600</td>
<td>Repair Weld Cracked - would sustain higher pressure at high flow. See Figure C-14.</td>
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</table>
Figure C-9 — ENGINE-BLED AIR DUCT; Burst Test Failure of Butt, Fusion-Welded Fatigue Test Specimen.
Figure C-10 — ENGINE-BLEED AIR DUCT; Burst Test Failure of Butt, Fusion-Welded Static Test Specimen.
CONVAIR, SAN DIEGO

Figure C-12 — ENGINE-BLEED AIR DUCT; Riveted and Brazed Test Specimen.
Figure C-13 — ENGINE-BLEED AIR DUCT; Burst Test Failure of Riveted and Brazed Fatigue Test Specimen.
III. 4. Test Results and Discussion: (Cont’d)

The 29-01005 basic specimens (all fusion welded - Figures C-9 and C-10) failed in the room temperature burst test at lower pressures than the same specimens sustained at 800 F. The static test specimen failed at 200 PSIG and the fatigue specimen failed at 340 PSIG. Both specimens failed in the longitudinal fusion welded seam. These apparent premature failures cannot be fully explained. It is possible that the elevated temperature permitted the stress to be more evenly distributed over the weld area, and the stress concentration points due to local discontinuities were more effective at room temperature. Secondly, it is possible that the temperature cycle caused a redistribution of the residual stresses.

The 29-01005-1 seam welded duct reached 1100 PSIG in the burst test, which is 200% of the burst value required by the drawing. The static specimen failed in the fusion welded elbow, Figure C-11. The fatigue specimen did not fail. The end cap leakage equaled the pump output flow at 1140 PSIG.

The 29-01005-3 riveted and brazed specimens both had small failures at 740 and 600 PSIG, respectively, and would sustain higher pressures at high flow. A photograph of this specimen type is shown in Figure C-12 (page 170) and enlarged photographs of the failures are shown in Figures C-13 and C-14 (pages 171 and 172).
IV. SUMMARY OF RESULTS

1. All specimen types sustained the static or fatigue test schedules.

2. The seam welded specimens sustained the highest pressure in the burst tests.

3. The riveted and brazed ducts failed in excess of the required 555 PSIG burst pressure. One specimen failed in the fusion weld area.

4. The fusion welded specimens failed at a room temperature burst test pressure lower than the 800 F pressures applied in the static and/or fatigue tests.
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### D. SHEAR PANEL - ELEVATED-TEMPERATURE STATIC AND FATIGUE TEST

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<td>D-31</td>
<td>Static Failure of 29-01014 Panel - Unstiffened Side</td>
<td>222</td>
</tr>
<tr>
<td>D-32</td>
<td>Static Failure of 29-01014 Panel - Stiffened Side</td>
<td>223</td>
</tr>
<tr>
<td>D-33</td>
<td>Fatigue Failure of 29-01014 Panel - Unstiffened Side</td>
<td>224</td>
</tr>
<tr>
<td>D-34</td>
<td>Fatigue Failure of 29-01014 Panel - Stiffened Side</td>
<td>225</td>
</tr>
</tbody>
</table>
I. INTRODUCTION

Test panels, representative of typical shear panel applications in supersonic aircraft structures were tested. Flight conditions would be expected to impose combined shear loads and aerodynamic heating, up to 900 F.

The objectives were to conduct static and repeated load tests to:

Determine if the panels would withstand a predetermined stress level of 34,600 lbs/sq. in. at room temperature, 200 F and 100 F increments thereafter to 900 F.

Determine the change in deflection under load due to temperature variations from room temperature up to 900 F.

Determine the ultimate static failure strength of the panels at 800 F.

Obtain deflection normal to the panel surface when statically loaded at 800 F.

Determine the fatigue life of the panels at 800 F.
II. SUMMARY

Static and repeated load tests were conducted on 23" x 23" flat and stiffened shear panels mounted in a rhomboid shear frame. Panels were tested at temperatures up to 900 F in order to determine the static strength, load-deflection characteristics, and fatigue life at elevated temperatures.

All panels withstood a predetermined shear stress of 39,200 lbs/sq. in. at temperatures up to 900 F. The panels were then statically and fatigue loaded to destruction at 800 F.

Static and fatigue failure results, along with load-deflection curves, are presented herein.
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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

D. SHEAR PANEL - ELEVATED-TEMPERATURE STATIC AND FATIGUE TEST

III. TEST SPECIMENS

Two each of the following Ti-4Al-3Mo-1V titanium alloy shear panels were tested to destruction. One each was statically tested and one was fatigue tested.

<table>
<thead>
<tr>
<th>Test Panel</th>
<th>29-01010-1, -3 &amp; -5</th>
<th>(Figure D-1, page 183)</th>
</tr>
</thead>
<tbody>
<tr>
<td>29-01011</td>
<td>(Figure D-2, page 185)</td>
<td></td>
</tr>
<tr>
<td>29-01013</td>
<td>(Figure D-3, page 187)</td>
<td></td>
</tr>
<tr>
<td>29-01014</td>
<td>(Figure D-4, page 189)</td>
<td></td>
</tr>
</tbody>
</table>
Figure D-3 - RIGIDIZED-GRID SHEAR TEST PANEL -
Engineering Drawing 29-01013
Figure D-4 - SHEAR TEST PANEL - Engineering Drawing 29-01014
3. Hydrogen content not to exceed 200 ppm after etching.
2. Spotweld per MIL-W-6850.
1. Spotweld patterns shown are typical symmetrically around assembly.

Note:
IV. TEST SET-UP

1. **Load and Deflection:**

   The specimens were mounted in a rhomboid shear frame as shown in Figures D-5, D-6 and D-7 (pages 193, 195 and 196). In order to produce, as nearly as possible, pure shear in the specimen, the sides of the frame were designed to allow less than 0.003" deflection at center span. Load was applied to the shear frame by a hydraulic actuator and was monitored by a Baldwin-Lima-Hamilton SR-4 load cell. Diagonal deflection in the direction of load as well as deflection of the panel normal to the surface was measured by dial indicators. Deflection point locations are shown in Figure D-8 (page 197).

2. **Heating:**

   Heat was applied by tubular infrared lamps mounted at two inch centers and having a maximum heating capacity of 530 BTU/min/sq. ft. (see Figure D-9). Power to the lamps necessary to produce the correct steady-state temperature was controlled by a Research, Incorporated heat programmer utilizing thermocouple numbers 1 and 2 as control thermocouples. (See Figure D-8 for thermocouple location.)

   Heat was applied to the top and bottom of the shear frame. However, the specimen was heated on the bottom unstiffened side only. The two bottom lamp banks used to heat both the panel and frame are shown in Figure D-9 (page 198). The specimen and frame were placed over the bottom lamp bank, Figure D-10 (page 199), and a lamp bank was placed over the top to heat the frame only. The specimen and jig work were completely enclosed in a stainless steel oven to minimize heat loss and edge cooling effects. Figure D-11 (page 200).
LEGEND

⊙ Deflection Point Location
■ TC#1 Thermocouple Location - Specimen Heat-Lamp Control
■ TC#2 Thermocouple Location - Jig Heat-Lamp Control

Figure D-8—DEFLECTION POINT AND THERMOCOUPLE LOCATION.
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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

D. SHEAR PANEL - ELEVATED-TEMPERATURE
STATIC AND FATIGUE TEST

V. TEST PROCEDURE

1. Static Tests:

   a. Room Temperature -

      Load was applied in increments until a calculated shear stress of 34,600 lbs/sq. in. was obtained. Load was reduced to a tare of 2500 pounds after each increment in order to determine permanent set. Deflections were recorded at each increment.

   b. Elevated Temperature -

      The specimen and jig work were heated at a rate of 18 F/min. Preliminary testing indicated that this heat rate maintained a temperature differential between specimen and jig work of less than 50 F. Load was applied at 200 F and at each 100 F increment thereafter through 900 F. At each temperature increment, load was applied and deflections recorded in the same manner as was done at room temperature.

      After completion of tests at 900 F, temperature was reduced to 800 F at a rate of 18 F per minute. The specimen was then loaded in increments to failure with deflections recorded at each increment.

2. Fatigue Tests:

   The specimen and jig work were maintained at 800 F and load was applied at an approximate rate of 30 cycles per minute.

   Throughout the repeated load tests, the magnitude of load, as indicated by the SR-4 load cell, was monitored and recorded on a Sanborn oscillographic recorder. Applied loads test conditions and results are shown in Table D-1 (page 202) and Figures D-12 through D-34 (pages 203 through 225).
<table>
<thead>
<tr>
<th>DRAWING NUMBER</th>
<th>TYPE OF TEST</th>
<th>NUMBER OF CYCLES</th>
<th>TEMPERATURE (°F)</th>
<th>SKIN THICKNESS (IN)</th>
<th>CORROSION THICKNESS (IN)</th>
<th>EFFECTIVE THICKNESS (IN)</th>
<th>EFFECTIVE WIDTH (IN)</th>
<th>APPLIED LOAD (LBS)</th>
<th>SHEAR STRESS (LBS/IN²)</th>
<th>FIGURE SHOWN DEFLECTION</th>
<th>FIGURE SHOWN FAILURE</th>
<th>PANEL WEIGHT (LBS)</th>
<th>REMARKS</th>
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<tbody>
<tr>
<td>29-0101-1</td>
<td>STATIC DEFLECTION</td>
<td>—</td>
<td>AMB  - 900</td>
<td>.025</td>
<td>—</td>
<td>.025</td>
<td>14.5</td>
<td>0</td>
<td>—</td>
<td>—</td>
<td>—</td>
<td>.605</td>
<td></td>
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<tr>
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<td>.025</td>
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<td>—</td>
<td>—</td>
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<td>.605</td>
<td>SHEAR FAILURE ALONG FIRST ROW OF SPOT WELDS</td>
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<td>.025</td>
<td>14.5</td>
<td>16,900</td>
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<td>CRACK PROPAGATED FROM CORNER RESULTING IN A SECONDARY STATIC SHEAR FAILURE</td>
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<td>.032</td>
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<tr>
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<td>—</td>
<td>.032</td>
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<tr>
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<td>—</td>
<td>.040</td>
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<td>0-25,000</td>
<td>20,500</td>
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<td>.040</td>
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<td>AMB  - 100</td>
<td>.020</td>
<td>.016</td>
<td>.025</td>
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<td>.020</td>
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<td>.025</td>
<td>12.0</td>
<td>24,000</td>
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<td>1.760</td>
<td>—</td>
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<tr>
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<td>STATIC DEFLECTION</td>
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<td>AMB  - 900</td>
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<td>.016</td>
<td>.030</td>
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</tr>
<tr>
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<td>.020</td>
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<td>.016</td>
<td>.030</td>
<td>10.5</td>
<td>22,500</td>
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<td>—</td>
<td>1.160</td>
<td>SHEAR FAILURE ALONG ROW OF SPOT WELDS AT EDGE</td>
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<td>30</td>
<td>1.530</td>
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<td>.032</td>
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<td>.032</td>
<td>14.5</td>
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<td>.032</td>
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<td>—</td>
<td>—</td>
<td>1.530</td>
<td>SHEAR FAILURE ALONG FIRST ROW OF SPOT WELDS AT EDGE AND BETWEEN STIFFENERS</td>
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</table>
NOTE: SEE FIGURE #8 FOR DEFLECTION POINT LOCATION

Figure D-14  29-01010-3 UNSTIFFENED SHEAR PANEL; Load Versus Deflection
Figure D-15 — STATIC FAILURE OF TEST PANEL 29-01010-3.

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Figure D-16 - FATIGUE FAILURE OF TEST PANEL 29-01010-3.
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Figure D-17 29-01010-5 UNSTIFFENED SHEAR PANEL; Load Versus Deflection

LEGEND
60000 LB. LOAD
56000 LB. LOAD
46000 LB. LOAD
40000 LB. LOAD
30000 LB. LOAD
20000 LB. LOAD
10000 LB. LOAD

DEFLECTION POINTS:
SEE FIGURE #8 FOR DEFLECTION POINT LOCATION

DEFLECTION IN INCHES

NOTE:
DEFLECTION POINT LOCATION

208
LEGEND:

- 47500 LB. LOAD
- 45000 LB. LOAD
- 42500 LB. LOAD
- 40000 LB. LOAD
- 37500 LB. LOAD
- 35000 LB. LOAD
- 32500 LB. LOAD
- 30000 LB. LOAD
- 27500 LB. LOAD
- 25000 LB. LOAD
- 22500 LB. LOAD
- 20000 LB. LOAD
- 15000 LB. LOAD
- 10000 LB. LOAD

NOTE: SEE FIGURE #8 FOR DEFLECTION POINT LOCATION

Figure D-20  29-01011 SHEAR PANEL - CORRUGATED; Load Versus Deflection
Figure D-21 — STATIC FAILURE OF 29-01011 PANEL; Unstiffened Side.
LEGEND

46000 LB. LOAD
44000 LB. LOAD
40000 LB. LOAD
32000 LB. LOAD
28000 LB. LOAD
24000 LB. LOAD
20000 LB. LOAD
15000 LB. LOAD
12500 LB. LOAD
10000 LB. LOAD

DEFLECTION POINTS

DEFLECTION IN INCHES

NOTE: SEE FIGURE #8 FOR DEFLECTION POINT LOCATION

Figure D25  29-01013 TEST PANEL - RIGIDIZED GRID; Load Versus Deflection
Figure D-27 - STATIC FAILURE OF 29-01013 PANEL. Stiffened Side.
Figure D-28 — FATIGUE FAILURE OF 29-01013 PANEL; Unstiffened Side.
VI. TEST LOADS

1. **Static Tests:**

   All panels were loaded to the same stress level at room temperature, 200°F and at each 100°F increment thereafter through 900°F. This stress level equals 75% design limit stress at 900°F (design limit stress at 900°F equals 46,100 PSI).

2. **Fatigue Tests:**

   The magnitude of the applied fatigue load was 4/9 of the ultimate failing load as determined by previous static tests.

VII. TEST RESULTS

1. **Static Tests:**

   The ultimate static loads as well as references to failure photographs and load-deflection data are presented in Table D-1 (page 202).

2. **Fatigue Tests:**

   Applied load, number of cycles and references to failure photographs are presented in Table D-1.

VIII. DISCUSSION

   As shown in Table D-1 the 29-01014 stiffened panel withstood the greatest load (62,000 pounds). In addition, this panel had the least normal deflection under load (reference Figure D-30). However, the fatigue life of the 29-01014 panel was noticeably less than that of the other stiffened panels.
IX. CONCLUSIONS

All specimens withstood an applied shear stress of 39,200 lbs/sq. in. at room temperature, 300 F and 100 F increments thereafter through 900 F.

Variations of deflection due to temperature variations were considered negligible and are therefore not presented.

Ultimate static failing load, deflection under load, fatigue life and weights of panels are presented in tabular form in this report.
The titanium shear panel load is given as

\[ P = \frac{\tau tw}{0.707} \]

The shear stress \( \tau \) is the pertinent unknown for a given ultimate applied static load in each panel; \( w \), the effective width of the panel, is taken as the distance between the inboard rows of the spot welds or 12.78 inches in each of the six shear panels tested. The small bent effect of the sheet and doublers, at the corners, is neglected. The following table shows the result of the static portion of the tests, all of which were run at a temperature of 800 F.

<table>
<thead>
<tr>
<th>Panel No.</th>
<th>Specimen</th>
<th>Shear Load (lbs) ( \times 2 )</th>
<th>&quot;q&quot; (lbs/in) Shear Load/12.78</th>
<th>&quot;t&quot; (ins.)</th>
<th>&quot;( \tau )&quot; (lbs/sq. in.)</th>
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</thead>
<tbody>
<tr>
<td>1</td>
<td>29-01010-1</td>
<td>24000</td>
<td>1878</td>
<td>0.025</td>
<td>75,100</td>
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<tr>
<td>2</td>
<td>29-01010-3</td>
<td>35350</td>
<td>2766</td>
<td>0.032</td>
<td>86,400</td>
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<tr>
<td>3</td>
<td>29-01010-5</td>
<td>43800</td>
<td>3427</td>
<td>0.040</td>
<td>85,700</td>
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<tr>
<td>4</td>
<td>29-01011-1</td>
<td>38250</td>
<td>2993</td>
<td>0.020 + 0.016</td>
<td>83,100</td>
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<tr>
<td>5</td>
<td>22-01013-1</td>
<td>35820</td>
<td>2803</td>
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<td>77,900</td>
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<td>6</td>
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<td>43800</td>
<td>3427</td>
<td>0.032</td>
<td>107,100</td>
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</tbody>
</table>

The average ultimate shear stress for the first five panels is 81,600 lbs/sq. in. This compares favorably with the ultimate allowable shear stress (at test temperature) of 76,000 lbs/sq. in. Ref: Properties of Ti-4Al-3Mo-1V - Titanium Metals Corporation of America, 233 Broadway, New York. Panel No. 6 failed at a very high calculated stress because of the heavy stiffeners which acted as a vieren deel truss and reacted some of the applied load.
X. STRUCTURAL DISCUSSION (Cont'd)

Panel No. 5 actually failed in static tension. In this specimen the rigidized grid reacts shear, but is much too flexible in tension, in a diagonal direction, to help the face sheet react the applied load. The tension stress at rupture in the face sheet was $50,700 \times (1.414) (12.78) (0.020) (140,000 \text{ lbs/sq. in.})$. The allowable ultimate tensile stress at this temperature is $147,000 \text{ lbs/sq. in.}$ It seems probable that the spot welding necessary for this type of construction reduced the parent metal allowable to $95\%$ of the unwelded material allowable.

It is interesting to note that this alloy of titanium acts very similarly to the stainless steels. The ultimate shear stress is approximately $57\%$ of the ultimate tensile stress at room temperature. This percentage drops slightly to approximately $51\%$ at $800 \text{ F}$. The aluminums and the chrom-moly steels shear strengths are approximately $60\%$ of the ultimate tensile strengths, at room temperature, however, this percentage seems to increase slightly with increase in temperature.

From the graphs of the deflections, it is seen that the first three panels were in tension field from the tare load; however, at ultimate load the deflections were not greater than plus/minus $0.15$ inch. These three specimens had several deflection nodes, while the rigidized panels had only one. The corrugated panel (Panel No. 4) seems to be shear resistant up to a shear flow of approximately $1650 \text{ lbs/inch}$, at which time one large buckle appeared. The deflection at the center of the panel was over $0.40$" at ultimate load. The rigidized grid is similar to Panel No. 4 in that only one buckle appeared. It grew with increase in load to a maximum deflection of $0.30$ inch. The specimen did not seem to be shear resistant at the tare load. Panel No. 6 was shear resistant up to a shear flow of $2995 \text{ lbs/in}$. An unsymmetrical buckle then appeared and remained to failure. The deflections were small (less than $0.15$ inch at failure). The cost weight-wise of Panel No. 6 is prohibitive unless there are large compressive loads that would necessitate the heavy stiffeners.

The fatigue tests show that much data is missing, if fatigue life is to be predicted accurately. The tests were not conclusive and fell short of expectations. The notch factors due to the spot welding needs much investigating.
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E. COMPRESSION PANEL - ELEVATED-TEMPERATURE STATIC TEST

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<th>Figure</th>
<th>Description</th>
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<td>Engineering Drawing No. 29-01012 - Test Panel - Corrugation Uninterrupted - Ti-4Al-3Mo-1V Titanium Alloy</td>
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<td>E-3</td>
<td>Engineering Drawing No. 29-01008 - Test Panel Corrugation - Ti-4Al-3Mo-1V Titanium Alloy</td>
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<td>E-4</td>
<td>Engineering Drawing No. 30488 - Specimen Installation 29-01009</td>
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<td>Specimen Installation 29-01009 Showing Stiffener Attachment</td>
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<td>Engineering Drawing No. 30489 - Specimen Installation 29-01012</td>
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<td>Specimen Installation 29-01012 Showing Stiffener Attachment</td>
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I. INTRODUCTION

Test specimens represented typical wing or fuselage compression panels of supersonic aircraft structures which would, under flight conditions, be subjected to combined compressive load, pressurization, and aerodynamic heating.

The objectives of this test were:

To determine the effect of limit load and pressure on the specimens when applied at room temperature, 200 F and 100 F increments thereafter to 800 F.

To determine the ultimate compressive load that the specimen would withstand when pressurized to design limit pressure and maintained at a constant temperature of 800 F.
E. COMPRESSION PANEL - ELEVATED-TEMPERATURE STATIC TEST

II. SUMMARY

Static compression load and pressurization were conducted on 26" x 31-1/2" 4Al-3Mo-1V titanium alloy stiffened panels. Panels were heated by infrared lamps from the unstiffened side, pressurized to limit pressure from the stiffened side and loaded axially in compression. All specimens withstood limit load and pressure at room temperature, 200 F and each 100 F increment through 900 F. The panels were then maintained at 800 F, pressurized to limit pressure and axially loaded to failure.
E. COMPRESSION PANEL - ELEVATED-TEMPERATURE STATIC TEST

III. TEST SPECIMENS

The test specimens were 26-1/2" x 31" stiffened compression panels fabricated from Ti-4Al-3Mo-1V titanium alloy. The following specimens were tested to failure:

Test Part 29-01009 - Figure E-1 (page 239)
Test Part 29-01012 - Figure E-2 (page 241)
Test Part 29-01008 - Figure E-3 (page 243)
Figure E-1 - COMPRESSION SHEAR TEST PANEL -
Engineering Drawing 29-01009
<table>
<thead>
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<th>S. No.</th>
<th>Description</th>
</tr>
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<tr>
<td>1</td>
<td>Weld per MIL-W-5629, except penetration to be 40-60%</td>
</tr>
</tbody>
</table>

**NOTES:**
Figure E-2 - COMPRESSION TEST PANEL; Uninterrupted Corrugation - Engineering Drawing 29-01012
Figure E-2  Page 241

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<tr>
<th>CONVVAIR</th>
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</tr>
<tr>
<td>CORROSION</td>
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<tr>
<td>UN-INTERRUPTED</td>
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<tr>
<td>441-340-1UV</td>
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<tr>
<td>TITANIUM ALLOY</td>
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<tr>
<th>LIST</th>
<th>MATERIALS</th>
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<tbody>
<tr>
<td>CONVVAIR</td>
<td>Titanium</td>
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</table>
Figure E-3 - COMPRESSION TEST PANEL; Corrugation - Engineering Drawing 29-01008
CONVAIR - SD

TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

E. COMPRESSION PANEL - ELEVATED-TEMPERATURE STATIC TEST

IV. TEST SET-UP

The specimens were loaded in compression by a Baldwin-Southwork, 400,000 pound, universal test machine. The vertical sides of the specimens were clamped in the test fixture in a manner which prevented long column buckling at the edges. This clamping action, however, reacted a negligible amount of vertical load from the skin. The test fixture was designed so that air pressure was applied to the stiffened side of the panel during loading. In addition, the fixture provided support for the specimen sub-structure. Normal skin deflection at the geometric center of the panel, as well as vertical movement of the compression heads of the machine were measured by dial indicators. These deflections were taken in an attempt to predict the failure or indicate local buckling before failure occurred.

Installation was as follows:

1. 29-01009 Test Panel:

A load block applied a compressive load to the -9 stringers and the -7 skin as shown in Figure E-4 (page 247), section B-B. The -13 and -15 stiffeners were supported at the end through slotted holes as shown in Figure E-4, sections A-A and C-C. This can also be seen in the installation photograph, Figure E-5 (page 249). The supporting holes were slotted so that only shear load would be reacted to the fixture.

2. 29-01012 Test Panel:

The specimen was mounted as shown in Figure E-6 (page 251). The -23 web was attached to the test fixture as shown in Figure E-6, sections A-A and C-C and also in Figure E-7 (page 253). The specimen was first mounted so that all of the compressive load was applied to the -7 skin as shown in Figure E-6, section D-D. However, since the skin was not of sufficient section to distribute the load into the center of the panel, local buckling occurred at the loaded edges as shown in Figure E-8 (page 254). After the specimen was removed from the jig, a crack was observed in the -23 web as also shown in Figure E-8. A repair was made by spot welding.
MACHINE INSTRUCTIONS:

1. MACHINE LOADED ENDS OF 29-01009-9 STRINGER FLAT AND PARALLEL.

2. MACHINE LOADING SURFACES OF -7 LOAD BLOCK FLAT AND PARALLEL.

3. INSTALL 29-01009 PANEL IN TEST FIXTURE WITH EXCEPTION OF -9 LOAD BLOCK.

4. WITH SPECIMEN INSTALLED, MACHINE LOADED SURFACE OF -11 STRAP, 29-01 29-7 SKIN AND -7 LOAD BLOCK FLAT AND PARALLEL.

5.

Figure E-4    Page 247
Figure E-6 - INSTALLATION FOR COMPRESSION TEST
PANEL 29-01012 - Engineering Drawing 30489
MACHINE INSTRUCTIONS

1. Machine single end of 29-0102-25 cap flat & parallel
2. Machine loaded surfaces of -7 load block flat & parallel
3. Install 29-0102-25 in test fixture with secured on -7 load block
4. With specimen installed, machine common loaded surface of 29-0102-25 skin, -7 strap
   -7 load block flat & parallel

SECTION 12 - 12
FULL SCALE

SECTION 13 - 13
FULL SCALE
IV. 2. **29-01012 Test Panel:** (Cont'd)

an .020" doubler on both sides of the web as shown in Figure E-9 (page 256). The .07 skin was straightened and a .090" thick frame was welded along the edge. The repaired specimen was remounted in the fixture and a load block added to apply a compressive load to the -25 cap as shown in Figure E-6, section B-B. In addition, a retaining bar was placed against the skin at the loaded ends to prevent column buckling (see Figure E-6, section B-B).

3. **29-01018 Test Panel:**

The specimen was mounted in the test fixture as shown in Figures E-10 and E-11 (pages 257 and 259). The -21 web was attached to the load fixture as shown in Figure E-10, sections A-A and C-C. Figures E-12 and E-13 (pages 260 and 261) show the specimen and test fixture as mounted in the compression heads.

Thermocouples were mounted on the heated side of the panel at three locations as shown in Figure E-14 (page 262). Heat was applied to the unstiffened side of the panel by tubular, quartz, infrared lamps having a maximum heating capacity of 700 BTU/min. /sq. ft., Figure E-15 (page 263). The lamp bank produced a constant thermal flux over the entire heated surface of the specimen. No attempt was made to heat the test fixture or compensate for edge cooling and chimney effect, caused by natural convection inside the lamp bank. Power to the lamps necessary to produce the correct specimen temperature was controlled as a time-temperature function by a Research, Incorporated heat programmer, utilizing thermocouple No. 2 as the control thermocouple. See Figure E-14 for thermocouple locations.
Figure E-10 - INSTALLATION OF COMPRESSION TEST PANEL 29-01008
Engineering Drawing 30487
SECTION A-A
FULL SCALE

29-0100B-35
SCREW
29-0100B-21
CHANNEL
29-0100B-19
SPAR CAP

29-0100B SKIN
29-0100B SPAR
-7 LOAD BLOCK
PRESSURE BOX
-9 LOAD BLOCK

SECTION 13-13
FULL SCALE

MACHINE INSTRUCTIONS
1. MACHINE LOADED ENDS OF 29-0100B-19 SPAR CAP FLAT AND PARALLEL.
2. MACHINE LOADING SURFACES OF -7 LOAD BLOCK FLAT AND PARALLEL.
3. INSTALL 29-0100B PANEL IN TEST FIXTURE WITH EXCEPTION OF -7 LOAD BLOCK.
4. WITH SPECIMEN INSTALLED, MACHINE LOADING SURFACE OR -11 STEM, 29-0100B-7 SKIN 29-0100B-9 SKIN, AND -7 LOAD BLOCK FLAT AND PARALLEL.
Figure E-12 — SPECIMEN AND TEST FIXTURE; Mounted In Compression Heads of Test Machine.
Figure E-13 — PANEL TEST SET UP; A General View.
Figure E-14— COMPRESSON TEST PANEL; Showing Locations of Thermocouples.
Figure E.15 - INFRARED LAMP BANK.
E. COMPRESSION PANEL - ELEVATED-TEMPERATURE STATIC TEST

V. TEST PROCEDURE

The following procedure was followed for all specimens tested:

Limit load and pressure were applied at room temperature, 200 F, and at increments of 100 F thereafter through 800 F.

With a temperature of 800 F maintained and limit pressure applied, load was increased until failure occurred.

VI. TEST LOADS

The following design limit loads and pressures at 800 F were calculated prior to testing:

<table>
<thead>
<tr>
<th>Specimen No.</th>
<th>Design Limit Pressure (PSIG)</th>
<th>Design Limit Load (lb)</th>
</tr>
</thead>
<tbody>
<tr>
<td>29-01009</td>
<td>10</td>
<td>28,000</td>
</tr>
<tr>
<td>29-01012</td>
<td>4</td>
<td>36,700</td>
</tr>
<tr>
<td>29-01008</td>
<td>10</td>
<td>37,800</td>
</tr>
</tbody>
</table>

VII. TEST RESULTS

1. 29-01009 Test Panel:

The 29-01009 panel failed with limit pressure of 10 psi applied and a compressive load of 79,000 pounds. Failure is shown in Figures E-16 and E-1' (pages 266 and 267). The failure was due to compressive buckling of the -7 stringers and also the -13 and -15 stiffeners.
Figure E-17 — STATIC FAILURE OF 29-01009 PANEL; Oblique View, 10 psi at 79,000 lbs & 300F.
VII. TEST RESULTS (Cont'd)

2. 29-01012 Test Panel:

After the initial repair was made, the specimen failed with limit pressure of 4 PSIG applied and a compressive load of 72,000 pounds. The primary failure was due to compressive buckling of the -7 skin, Figure E-18 (page 269), followed by a secondary failure of the weld between the -23 web and -25 caps, Figure E-19 (page 270). See Figure E-20 (page 271) for an over-all view showing general location of the weld failure.

3. 29-01008 Test Panel:

The 29-01008 panel failed with limit pressure of 10 PSI and a compressive load of 36,500 pounds. Failure was due to compressive buckling of the -19 spar followed by a secondary tension failure in the -35 attaching screws. See Figures E-21 and E-22 (pages 272 and 273).

VIII. DISCUSSION

Panel deflection and compression head movement, as determined by dial indicators, did not indicate any local buckling or failure prior to the ultimate failure of the panel. Test data indicated that the effect of temperature upon deflection of the panel when loaded to limit load was negligible. Therefore, this deflection data is not presented in this report.

IX. CONCLUSIONS

The specimens withstood design limit load and pressure at room temperature, 200 F, and increments of 100 F thereafter through 800 F. No failure was evident.

All panels failed at loads exceeding design ultimate.
Figure E-18 — STATIC FAILURE OF 29-01012 PANEL;
Due to Skin Buckling, 4 psi at 36,000 lbs & 800°F.
Figure E-21 — STATIC FAULT RE OF 29-01008 PANEL; Spar and Skin Buckling, 10 psi at 86, 00 lb & 800°F.
Figure E-22 — STATIC FAILURE OF 29-01008 PANEL; 10 psi at 86,500 lbs & 800°F.
E. COMPRESSION PANEL - ELEVATED-TEMPERATURE STATIC TEST

X. STRUCTURAL DISCUSSION

The results of the compression test specimens and the method of test set-ups make it difficult to determine which is the most efficient panel design. The panel failing loads and weights are given below. All ultimate tests were run at 800 F.

<table>
<thead>
<tr>
<th>Panel No.</th>
<th>Specimen</th>
<th>Compression Load (lbs.)</th>
<th>Pressure Load (lbs/sq. in.)</th>
<th>Weight (lbs.)</th>
</tr>
</thead>
<tbody>
<tr>
<td>1</td>
<td>29-01008</td>
<td>86,500</td>
<td>10</td>
<td>13.6</td>
</tr>
<tr>
<td>2</td>
<td>29-01009</td>
<td>79,000</td>
<td>10</td>
<td>10.7</td>
</tr>
<tr>
<td>3</td>
<td>29-01012</td>
<td>72,000</td>
<td>4</td>
<td>8.3</td>
</tr>
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</table>

In Panels 1 and 3 the pressure load is reacted by the corrugated spar webs in tension. These loads are beamed to the spar webs by the face sheet corrugations and the face sheets. This test set-up resulted in stresses due to pressure which are 90 degrees removed from the primary compression load stresses. In these two panels the pressure is partially stabilizing the structure, however, this effect is negligible at ultimate load. In contrast, the pressure load on Panel No. 2 is directly adding to the stresses on the inboard leg of the stringer material. At failure of the -9 stringers in crippling, 72% of the load was due to primary compression and 28% was from bending due to pressure. This type of panel is probably best for resisting compression loads. Intercostaling of the skin is somewhat difficult at the stringer spacing shown, and the production costs are probably a little higher.

In Panel No. 3 design and production would be difficult at rib stations or bulkheads where the cross members must have continuity through the sine wave spar webs. Intercostaling of the skin would probably be accomplished by the skin corrugations in bending, which would seem to be too flexible for good design.
X. STRUCTURAL DISCUSSION (Cont'd)

Panel No. 1 has most of the disadvantages of Panel No. 3. In addition to a weight penalty, the only desirable feature of Panel No. 1 is the ability to remove the skins and corrugations from the substructure by means of screws or other fasteners.

1. Discussion of Stress and Allowables:

In Panel No. 2 the -9 stiffener and its effective skin has the following calculated properties:

\[
\begin{align*}
A &= .1749 \text{ sq. in.} \\
\bar{Z} &= .386 \text{ in. (from inside skin surface)} \\
I &= .0327 \text{ in}^4
\end{align*}
\]

The compressive load/stiffener = 7900 lbs.
The bending mom./stiffener = 770 in.-lbs.

Then,

\[
f_c = \frac{7900}{.1749} + \frac{777 (1.12 - .386)}{.0327}
\]

\[
= 62,800 \text{ lbs/sq. in.}
\]

(Outboard element of -9 stiffener)

The allowable calculator from the formula \( KE \left(\frac{t}{6}\right)^2 \) gives very close results, if the element is considered fixed at the ends and simply supported at the edges. This assumption sets \( K = 3.62 \).

Then,

\[
F_c = 3.62 \left(13.4 \times 10^6 \right) \left(\frac{.025}{.70}\right)^2 = 61,900 \text{ lbs/sq. in.}
\]

("E" at 800 F)
X. 1. Discussion of Stress and Allowables: (Cont'd)

In Panel No. 3 the radius of gyration, $\rho$, of the skin is

$$\frac{t}{\sqrt{12}} = 0.00777 \text{ in.}$$

$$\frac{L'}{\rho} = \left[ \frac{.5}{2 (.00777)} \right] = 32.2$$

(For $c = 4$; or fully fixed at each corrugation)

Then,

$$F_c = F_{cy} \left[ 1 - \frac{F_{cy}}{4 \pi^2 E} \left( \frac{L'}{e} \right)^2 \right]$$

(Johnson formula with $F_{cy}$ substituted for $F_{co}$)

$$F_c = 119,000 \left[ 1 - \frac{119,000 (32.2)^2}{4 \pi^2 13.4 \times 10^6} \right] = 91,000 \text{ lbs/sq. in.}$$

(Note: All values at 800 F)

Assuming that -25 plate and 1 inch of -23 web is effective at $F_{cy}$ stress levels, then the calculated compression load carrying ability of 29-01012 panel is:

$$28.5 (.020) 91,000 + 2 (.040) 1.3 (119,000) + 2 (.020) 1.0 (119,000) = 69,040 \text{ lbs.}$$

This calculation compares favorably with the ultimate compression load of 72,000 pounds that failed the panel.
# CONVAIR - SD

TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

F. TAIL CONE STATIC AND FATIGUE TESTS

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<td>Vertical Deflections and Sets, 800 F Static Test</td>
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<td>F-21</td>
<td>Horizontal Deflections and Sets, 800 F Static Test</td>
<td>309</td>
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<td>F-22</td>
<td>Vertical Deflections and Sets, 900 F Static Test</td>
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<td>Horizontal Deflections and Sets, 900 F Static Test</td>
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TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

F. TAIL CONE - STATIC AND FATIGUE TESTS

I. INTRODUCTION

The Fuselage Tail Cone is a conversion to Ti-4Al-3Mo-1V alloy of the F-102A Interceptor, Part No. 8-73490, Tail Cone Assembly. The original part contained 2024-T31 clad aluminum alloy, type 321 stainless steel, commercially pure titanium, and some titanium alloy.

The objectives of the program were to determine:

1. The load carrying characteristics of a titanium fuselage tail cone assembly at various temperatures through 900 F.

2. The fatigue strength of the assembly at 800 F.
II. SUMMARY

A titanium Fuselage Tail Cone Assembly was tested statically to limit load and in fatigue at several loads from 66.6% limit load to design ultimate at temperature.

In the static test, load was applied in 20 per cent steps up to limit load at room temperature, 200 F, 300 F, 400 F, 500 F, 600 F, 700 F, 800 F, and 900 F with no apparent failures.

The fatigue test consisted of 2,500 cycles each at room temperature, 200 F, 400 F, and 600 F; and 100,000 cycles at 800 F at 66.6% limit load, 50,000 cycles at limit load, 25,000 at 1-1/4 limit load and 17,375 at 150% limit load (design ultimate), at 800 F with some minor structural failures.
III. TEST SPECIMEN

The test specimen was manufactured according to Engineering Drawings 29-01001 and 29-01002, Figures F-1 and F-2 (pages 285 and 287).

The specimen was made entirely from Ti-4Al-3Mo-1V alloy except for the fairing tips which were spun from type 321 stainless steel.
Figure F-1 - FUSELAGE TAIL CONE ASSEMBLY -
Engineering Drawing 29-01001
Figure F-2 - FAIRING INSTALLATION; Fuselage Tail Cone Assembly - Engineering Drawing 29-01002
IV. TEST PROCEDURES

The test specimen was attached to a steel plate by four bolts, simulating an actual installation.

Test loads were applied through 108 points uniformly distributed over the surface of the specimen as shown in Figure F-3 (page 290). The loads were applied to the specimen skin by eyebolts through the skin into 1/2" x 1" steel blocks cushioned by pieces of asbestos blanket.

The static test load (limit load) was applied in 20 per cent increments and deflections taken. Permanent set was measured at 10 per cent load after each increment. Deflections were taken at four points on the skin. These points were 3-1/2" forward of the exit nozzle: one at each end of the flight vertical and horizontal axes. The purpose was to detect diameter changes. The complete load sequence was run at room temperature, 200 F, 300 F, 400 F, 500 F, 600 F, 700 F, 800 F and 900 F.

The first fatigue test load was 2/3 limit load. This load was applied 2,500 times at each of the following temperatures: room temperature, 200 F, 400 F, and 600 F. The same load was then applied 100,000 times at 800 F. Full limit load was applied 50,000 times at 800 F. 125% limit load (83.3% design ultimate) was applied 25,000 times at 800 F. 150% limit load (design ultimate) was applied 17,375 times at 800 F.

During the fatigue test, limit load was applied at the rate of 50 times per minute. Full limit load was applied at 30 times per minute, 125% limit load at 25 times per minute, and 150% limit load (design ultimate) at 20 times per minute.

Heat was applied by a conical oven within the specimen, simulating heat from a jet engine, as shown in Figure F-4 (page 291). The specimen was covered with an asbestos blanket, Figure F-5 (page 292), to reduce heat loss and help maintain an even temperature distribution. Quartz infrared lamps were used to provide heat. They were controlled by a Research, Incorporated heat programmer. Four channels of heating were
Figure F-3 — FUSELAGE TAIL CONE IN TEST FIXTURE;
With Whippletrees Attached.
Figure F-5 — OVERALL VIEW SHOWING ASBESTOS BLANKET COVERING IN PLACE.
CONVAIR - SD

IV. TEST PROCEDURES (Cont'd)

used: one each at the top and bottom and two sharing the center portion. A channel consists of a lamp bank, a controller, and a thermocouple attached to the specimen under the lamp bank. The accuracy of the temperature is dependent only on the accuracy of the thermocouple.

V. TEST LOADS

Load for the static test was design limit load which is defined in Convair Report S-GEN-84 "Titanium Development Program" as a combination of maneuver shear and moment and internal pressure (condition 3). Fatigue load was 66.6% of limit load for the first 110,000 cycles and was raised subsequently on instructions from Convair Structures Group in order to obtain failures in the specimen structure.

VI. TEST RESULTS AND DISCUSSION

The deflection and set data from the static test may be found in Figures F-6 through F-23 (pages 294 through 311). There was no apparent damage after the static test.

During the fatigue test, the Fuselage Tail Cone Assembly withstood a total of 202,375 cycles of loads which ran from 2/3 limit load to 150% limit load (design ultimate) at 800 F. The assembly would still carry the load, although three internal ribs had failed and another was about 80% failed. The three failed ribs are shown in Figure F-24 (page 312), and the partially failed one in Figure F-25 (page 313). Figure F-26 (page 314) shows their relative locations.

Several cracks started in the skin adjacent to rivets near the nozzle end of the Tail Cone during the 125% limit load testing at 800 F. These are shown in an over-all view in Figure F-27 (page 315). Figures F-28, F-29, and F-30 (pages 316, 317 and 318) show details of these cracks; left, middle, and right, respectively, as compared to Figure F-27. The cracks were located in a lightly loaded area (48 pounds per load point) while diametrically opposite there were no cracks with up to 101 pounds per load point. An investigation showed that the cracked skin had a hydrogen content of about 190 PPM. This was about the highest of all skins used on the Tail Cone. The load points adjacent to these cracks were then moved.
Figure F-6  ROOM-TEMPERATURE STATIC TEST RESULTS

FUSELAGE TAIL CONE ASSEMBLY

DEFLECTIONS

PERMANENT SETS

ΔD

#1 & #3

1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN, 3 1/2 INCHES FORWARD OF EXIT NOZZLE

2. ΔD REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE
FUSELAGE TAIL CONE ASSEMBLY

PERMANENT SETS

DEFLECTIONS

ΔD #1 #3
100

#1 ΔD

#3

PERCENT OF LIMIT LOAD

80

60

40

20

0

0 .02 .04 .06 .08 .10 .12 .14 .16 .18 .20

DEFLECTION IN INCHES

NOTES

1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN, 3 1/2 INCHES FORWARD OF EXIT NOZZLE
2. Δ D REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE

Figure F-8 200°F STATIC TEST RESULTS
FUSELAGE TAIL CONE ASSEMBLY

DEFLECTION
PERMANENT SETS

DEFLECTION

#4
#2
Δ D

Δ D

NOTES

1. HORIZONTAL DEFLECTIONS AND SETS TAKEN AT FAIRING, 3 1/2 INCHES FORWARD OF EXIT NOZZLE

2. D REPRESENTS A DECREASE IN DIAMETER OF THE NOZZLE

PERCENT OF LIMIT LOAD

DEFLECTION IN INCHES

Figure F-9 200°F STATIC TEST RESULTS
FUSELAGE TAIL CONE ASSEMBLY

DEFLECTIONS

PERMANENT SETS

DEFLECTION IN INCHES

NOTES

1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN, 3 1/2 INCHES FORWARD OF EXIT NOZZLE

2. ΔD REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE

FIGURE F-10. 300°F STATIC TEST RESULTS
FIGURE F-11. 300°F STATIC TEST RESULTS
FIGURE F-14. 500°F STATIC TEST RESULTS

NOTES
1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN 3 1/2 INCHES FORWARD OF THE EXIT NOZZLE
2. ΔD REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE
FIGURE F-16 600°F STATIC TEST RESULTS
FUSELAGE TAIL CONE ASSEMBLY

NOTES
1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN, 3 1/2 INCHES FORWARD OF EXIT NOZZLE
2. $\Delta D$ REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE

FIGURE F-18. 700°F STATIC TEST RESULTS
CONVAIR, SAN DIEGO

FUSELAGE TAIL CONE ASSEMBLY

1. HORIZONTAL DEFLECTIONS AND SETS TAKEN AT FAIRING, 3 1/2 INCHES FORWARD OF EXIT NOZZLE
2. ΔD REPRESENTS A DECREASE IN DIAMETER OF THE NOZZLE

FIGURE F-19. 700°F STATIC TEST RESULTS
1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN, 3 1/2 INCHES FORWARD OF EXIT NOZZLE

2. ΔD REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE

FIGURE F-20. 800°F STATIC TEST RESULTS
CONVAIR, SAN DIEGO

Figure F-21. 800°F Static Test Results

NOTES
1. Horizontal deflections and sets taken at fairing, 3 1/2 inches forward of exit nozzle.
2. ΔD represents a decrease in diameter of the nozzle.
FIGURE F-22. 900°F STATIC TEST RESULTS

1. VERTICAL DEFLECTIONS AND SETS TAKEN AT SKIN, 3 1/2 INCHES FORWARD OF EXIT NOZZLE

2. $\Delta D$ REPRESENTS AN INCREASE IN DIAMETER OF THE NOZZLE
CONVAIR, SAN DIEGO

DEFLECTION #4
ΔD #4 #2
DEFLECTION #2
ΔD

NOTES
1. HORIZONTAL DEFLECTIONS AND SETS TAKEN AT FAIRING, 3 1/2 INCHES FORWARD OF EXIT NOZZLE
2. ΔD REPRESENTS A DECREASE IN DIAMETER OF THE NOZZLE

FIGURE F-23. 900°F STATIC TEST RESULTS
Figure F-24 — DETAIL VIEW SHOWING THE THREE FAILED RIBS AND TORN OUT SECTION.
Figure F-25 — DETAIL VIEW SHOWING PARTIALLY FAILED RIB.
Figure F-27 — OVERALL VIEW SHOWING SKIN CRACKS AND THEIR LOCATION RELATIVE TO THE FAILED RIBS AND TORN OUT SECTION.
Figure F-29 — DETAIL VIEW SHOWING SKIN CRACKS; Original Load-Point Holes are Away From
Right Bottom Enlarged Holes on Bottom Flap.
VI. TEST RESULTS AND DISCUSSION (Cont'd)

to the rib where holes were drilled between rivets putting the load directly into the rib itself and testing continued.

After a total of 202,375 cycles, one of the loading blocks (1/2" x 1") pulled out a 2-1/2" x 3" piece of skin. This skin failure is shown in detail in Figure F-24. Figures F-26 and F-27 show its location relative to the other failures. Testing was discontinued at this point.

VII. CONCLUSIONS

1. The load carrying characteristics of the Fuselage Tail Cone Assembly, as determined by the deflection/set curves, are not materially affected by temperatures up through 900 F although deflections did increase slightly with temperature.

2. The Fuselage Tail Cone Assembly withstood 202,375 cycles of load, including 17,375 at design ultimate at 800 F, without major structural failure.
TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

G. PLATE STRINGER COMPRESSION PANELS

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# TITANIUM DEVELOPMENT PROGRAM

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I. INTRODUCTION

The type of test structures reported herein were skin and stringer combinations which represent sections of an airframe wing skin. Three different test configurations of these structures were fabricated. Each of the test structures represented varying degrees of difficulty of fabrication and strength. This report presents the results of testing the three configurations of skin and stringer combinations as edge compression members. Flight parameters for a type of future aircraft were duplicated, as closely as possible. These flight parameters are axial compressive load, internal pressure and temperature.
Three configurations were selected for evaluation of Titanium Alloy B-120 VCA as fabricated in typical wing structure. The three specimens were designed to be representative of typical plate stringer geometry. Each specimen consisted of three continuous 18.0 inch bays subjected to edge compression and internal pressure.

The analysis to predict their strength assumed each bay to act as a simple beam column having an end column fixity equal to 1.0. The material properties used in the analysis were based on available typical processing results obtained from Convair data for room temperature. The elevated temperature effects were derived by using a percentage deterioration factor estimated from data in DMIC Report 110.

The ultimate strength test loads have been correlated with the analytical method and the test data fitted on the design curves reasonably well.

The test unit strength-weights of the configurations tested based on the weight per square foot for a load of 10,000 pounds per chord inch are: 3.0 lbs/sq. ft, 3.29 lbs/sq. ft., and 2.52 lbs/sq. ft. for the -1, -3, and -5 specimens, respectively. These values do not reflect the fact that the -1 carried 4.7 PSIG at failure, the -3 carried 4.9 PSIG and the -5 carried 0 PSIG at failure. The theoretical strength-density for a 10,000 pound load per chord inch in combination with 9 PSIG pressure is 3.8 lbs/sq. ft, 4.67 lbs/sq. ft., and 3.09 lbs/sq. ft. for the -1, -3, and -5 specimens, respectively.

From a strength-weight standpoint the -5 specimen appears to be superior and its relative weight decreases as the internal pressure increases. The -3 specimen appears to be the least efficient and its relative weight increases as the pressure increases.

The 3.09 lbs/sq. ft. is equivalent to an aluminum panel operating at a gross average stress of 46,700 PSI which indicates that the titanium structure is essentially as efficient at 600 F as the aluminum is at room temperature for the load intensity compared.
III. DESCRIPTION OF TEST SPECIMENS AND METHOD OF TESTING

1. Test Specimens:

Three skin and stringer combination specimens were fabricated of the same thickness skin but with three different stringer shapes. The material used for all parts was Titanium alloy B-120 VCA. All of the specimens were of the same length. The three stringer shapes were "I", "J", and "Y" sections. The "I" section stringer was fabricated by fusion welding the top and bottom flanges to the web section. The shape was then spot-welded to the specimen skin. The "J" section was fabricated in two sections. One section was constructed similar to the shape of a channel section but of unequal flange width. The other section was constructed similar to a channel section. These two parts were made into an assembly by spotwelding the two parts back-to-back. The section was then spot-welded to the skin. The "Y" section was constructed from three parts. Two of the parts were constructed from the same shape by welding two sections back-to-back. These two sections were the same shape as the channel used for the "J" section. The third part of this stringer shape was a doubler cap spotwelded to the lower flanges of the "Y" section. This assembly was then spotwelded to the specimen skin. The specimen containing the fusion welded "I" section contained seven stringers, constructed on 2.00 inch centers, and was designated as -1 specimen on the manufacturing drawing shown in Figure G-1 (page 327). The finished test specimen is shown in Figure G-2a (page 329). The specimens containing the "J" and "Y" sections contained five stringers, constructed on 2.50 inch centers and were designated -3 and -5 specimens respectively on the manufacturing drawing shown in Figure G-1. The finished test specimens are shown in Figures G-2b and G-2c, respectively, (page 329).

At four locations along the length of the specimens structural shapes were spotwelded to the back flanges of the stringers. These structural shapes represented wing rib caps and were used to react pressure loads on the structure as well as restrain the skin and stringers from buckling. These rib cap shapes were located symmetrically about the center of the specimens and on 18.00 inch centers.
TYP POR -17
TYP POR -35

FOR -5 "MLV"
TYP FOR -17 TO -35 ASSY

SECTION LOCATED
LIKE 3-18, PAP -1

TYPE 13"

APPROX
FILL LEVEL OF
SEALING COMP
SEE NOTE 4

FOR -5 ONLY
**Notes:**

1. Spotweld per MIL-W-68568A except radiographic examination not needed for class "A" welds.

2. Fusion weld per Convair spec 0-6502S.

3. Add .002 in to each end of .05 - .25, A - .35 sub-assembly to provide gap fit.

4. Use DC 0.035 compound as sealant for -1, -3, -.5 ends.

5. Distortion of panel after straightening not to exceed limits shown.

6. Trimming & grinding of -1, -3, -.5 to be performed under test lab.

---

**Table:**

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<th>MATL</th>
<th>QTY</th>
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<td>TYPE &quot;A&quot;</td>
<td>AGED Ti 3Al 13V 11G (B120YCA)</td>
<td>1</td>
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<td>TYPE &quot;18&quot;</td>
<td></td>
<td>1</td>
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<tr>
<td>TYPE &quot;4&quot;</td>
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<td>1</td>
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---

**Diagram:**

- SKIN SIDE
- ALLOWABLE PANEL DISTORTION DIAMETR (NO SCALE)
- SYMBOL DENOTES SPOTWELD
- SYMBOL DENOTES NAS 503M, UNIVERSAL HEAD BARE MONEL RIVETS, 5/32 DIA

---

**Legend:**

- "A" type symbol denotes spotweld.
### Table:

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</table>

**Notes:**
- Performed under test lab direction.
- EAL's (EAL number) shown in D/A box.
- -5 ENDS
- R105S to provide gap for straightening.
- F-LAMINATION

**Figure G-1:**

- Panel-Plate Stringer
- ConAIR

**Page 327**
Figure G-2 — TEST SPECIMENS; Viewed from the Stringer Side.
TITANIUM DEVELOPMENT PROGRAM

Volume V - Structural Evaluations of Titanium Alloy Assemblies

III. DESCRIPTION OF TEST SPECIMENS AND METHOD OF TESTING (Cont'd)

2. Test Program:

Each of the test specimens was subjected to a specific test program. This program contained the parameter of pressure and temperature and axial compressive load representative of those to be encountered in high speed flight. The test program is outlined in Table G-1, below.

<table>
<thead>
<tr>
<th>Condition</th>
<th>Temperature</th>
<th>Pressure (PSIG)</th>
<th>Axial Compressive Load (lbs)</th>
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<tr>
<td>I</td>
<td>Room Temp.</td>
<td>9</td>
<td>79,500 57,700 126,000</td>
</tr>
<tr>
<td>II</td>
<td>200 F</td>
<td>9</td>
<td>73,300 51,300 113,334</td>
</tr>
<tr>
<td>III</td>
<td>400 F</td>
<td>9</td>
<td>73,300 51,300 113,334</td>
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<td>600 F</td>
<td>9</td>
<td>73,300 51,300 113,334</td>
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<tr>
<td>V</td>
<td>800 F</td>
<td>9</td>
<td>60,700 45,300 96,667</td>
</tr>
<tr>
<td>VI</td>
<td>900 F</td>
<td>9</td>
<td>60,700 45,300 96,667</td>
</tr>
<tr>
<td>VII</td>
<td>600 F</td>
<td>9*</td>
<td>Failure Failure Failure</td>
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</table>

* During the test of the -5 specimen to failure, the internal pressure was reduced to 0 PSIG.

During each of the above test conditions the axial compressive load was increased in 20% increments up to the load shown. For the test to failure the load was increased in 20% increments to the load shown and then in 10% increments to failure.

3. Test Setup and Methods:

a. Axial Compressive Load -

The specimens were tested in a 400,000 pound Baldwin Southwark Universal Test Machine. The compressive load was applied by the loading head of the machine and reacted by the fixed head of the machine. The
III. 3. a. **Axial Compressive Load** - (Cont'd)

The reaction head of the testing machine was adjusted flat and parallel to the loading head within plus/minus .001 inch. The specimens also had the stringer ends of the assemblies ground flat and parallel within plus/minus .001 inch. For setup work the specimens and test fixture were supported by the test machine columns. However, during testing the specimens were held in the test machine by the load on the ground ends of the specimens. The test assembly in the test machine is shown in Figures G-3 and G-4 (pages 332 and 333).

b. **Pressure Load**

To apply internal pressure simulating pressurization of a wing fuel tank, a special test fixture pressure box was constructed. The pressure box was constructed of 8.00 inch steel channel sections. The ends of the pressure box were set in from the ends of the side members. These ends were set in so that the end seal on the box was 4.00 inches from the end of the specimen. This pressure box was mounted on the specimen in a manner that allowed pressure to be applied to the stringer side of the specimen. To react the pressure load the specimens were tied to the pressure box through the rib caps on the back of the stringers. The method of attaching these rib caps is shown in Figures G-5 and G-6 (pages 334 and 336). This type of attachment allows the pressure to be reacted by tension in the tieback straps. To minimize the effect of these straps carrying part of the compressive load into the pressure box, the straps were made of thin layers of stainless steel. To react the pressure load on the edge of the specimen skin a special retainer was constructed. The retainer was constructed of 6.00 inch channel sections and matched the pressure box. The retainer was mounted on the skin side of the test assemblies and connected to the pressure box by bolts through the flange of the pressure box. This bolt attachment was outside the area of the specimen and, therefore, made no direct connection to the test specimen.

The assembly of the pressure box and retainer resulted in a clamping action on the specimen skin. This clamping action was adjusted to produce a light fit between the skin and the fixture. In addition to reacting the skin pressure load the clamping of the skin also prevented local buckling at the edge of the skin by affording a straight ridge guide. Also, since the attachment of the two fixture parts was a light fit the test specimen was allowed to develop the full compressive strain with only an infinitesimally small amount of strain being fed into the fixture through friction. In
Figure G-3 — TEST SPECIMEN AND FIXTURE ASSEMBLY IN THE TEST MACHINE: View From The Treated Side.
Figure G-4 — TEST SPECIMEN AND FIXTURE ASSEMBLY IN THE TEST MACHINE; Viewed From The Back of The Pressure Box.
Figure G-5a—TYPICAL TEST SET UP; Cross Section.
Figure G-5b — TYPICAL TEST SET UP; Side View.
Figure G-6 — TYPICAL TIEBACK STRAP INSTALLATION;
For Reacting Pressure Load.
addition to the light fit, the flat surfaces of the pressure box and retainer flanges were machined to produce a narrow contact area between the fixture and the specimen. This contact area was 3/16 inch wide and extended the full length of the fixture as well as across the full width of the ends. The resultant effect of the machining was a reduction of the contact areas.

During the test of the -1 specimen the distance between the contact areas on the fixture was 16.00 inches. This width allowed the same spacing between the edge stringer and the pressure box as existed between the stringers. For the test of the -3 and -5 specimens special bars were attached to the pressure box and retainer which reduced the distance between the contacts to 15.00 inches. These spacer bars also had the machined contact areas the same as the pressure box flange. By reducing the distance between the contact areas, the effective width of the fixture was reduced. This reduction allowed the same spacing between the edge stringer and the box as between the stringers for the -3 and -5 specimens.

c. Heat Source -

The flat skin side of the specimens only was heated during testing. Heating was accomplished by using the special pressure box retainer as a mounting for the heating devices. The heating devices were twenty-eight 2500 watt infrared heat lamps. The heat lamps were built into the pressure box retainer by welding two 6.00 inch steel channel sections to the top of the retainer. This channel section then provided a mounting for the heater clips and buss bars. The heat lamps were extended through holes, located on 2.00 inch centers, in the web of the retainer channels. Gold plated stainless steel reflectors were mounted around the inside of the retainer. Gold plated reflectors were also used across the open side of the retainer to enclose the heat lamps and produce a partial oven effect. Figure G-7 (page 338) shows the heat lamps as well as the reflectors installed in the retainer. The top and bottom of the retainer were not sealed to produce a complete oven effect. These ends were left open so the chimney effect of the heating could be reduced by venting some of the hot gases that accumulated at the top of the oven when the whole assembly was vertical in the test machine.
Figure G-7 — SPECIAL PRESSURE BOX RETAINER; Showing the Installation of the Heat Lamps and Gold-Plated Reflectors.
The heat lamps were divided into three bays for control purposes. Each heat bay covered one-third of the total effective test area. The temperature of the specimen was controlled by controlling the electric power to the heaters. To control this power the heat lamps were connected to three control channels of a twelve channel Research, Inc. Heat Control Programmer. In the center of each of the three heat bays a thermocouple was spotwelded to the test specimen skin. This thermocouple was spotwelded to the skin between the stringers so that the heat sink effect would be negligible. The thermocouple was then connected to the heat programmer to provide a control feedback voltage. The control voltage was summed with a voltage representing the desired temperature. The resultant different voltage or error signal was then used to control the power output of three channels of 480 KVA ignitron power controllers. The heat lamps in each of the control bays provided a uniform heat flux over the entire surface of the heat bay. No attempt was made to apply uniform temperature over the heat bay.

d. Instrumentation -

Each of the test specimens was instrumented with thermocouples for measuring the temperature distribution across the test panel, deflection wires for determining the deflection of the panel normal to the compressive load, and strain gages to determine the strain distribution as well as indicate the start of buckling. The locations and numbers of each type of instrumentation are presented as follows:

e. Deflections -

Twenty-seven deflection wires were attached to the -1 specimen and twenty-five wires were attached to the -3 and -5 specimens. Ten of the deflections were reference points and measured deformation of the test fixture. The remaining deflection locations determined the deflections along the length as well as across the width of the specimens. Each of the deflection wires was connected to a cantilever beam strain gaged deflection indicator. The deflection indicators are shown in Figure G-8 (page 340). In addition, the movement of the loading head of the test machine was recorded. This deflection was indicated on a 1.00 inch travel dial indicator. All deflections except the dial indicator were recorded on a remote indicator. The locations of the deflection points are shown in Figures G-9 and G-10 (pages 341 and 342).
Figure G-8 — DEFLECTION BEAM SET UP; Adjacent to the Test Machine.
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Figure G-9 — DEFLECTION LOCATIONS; 20-01015-1 Panel.
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Figure G-10 – DEFLECTION LOCATIONS; 29-01015-3 & -5 Panels.

MIDPOINT BETWEEN STRINGERS (TYP)

7.50
8.00
7.50
8.00
III. 3. **Test Setup and Methods:** (Cont'd)

f. **Thermocouples**

Chromel-alumel thermocouple wires were attached to each of the test specimens; i.e., twenty on -1 and fourteen on -3 and -5. These thermocouples were resistance spotwelded to the specimen to form a split junction. This type of junction is formed by spotwelding each wire individually to the skin at a maximum distance of 1/16 inch apart. The thermocouple therefore contains two junctions; i.e., one junction of Chromel-Titanium and the other junction of Titanium-Alumel. Since the titanium sheet is used as a connecting part of the electric circuit the effect of the two junctions resolves to the output of a Chromel-Alumel junction. This type of thermocouple installation provides the most accurate temperature of the skin surface since the skin is part of the electric circuit. Each of the thermocouples was connected to a 150 F reference junction and then connected to a remote indicator. The locations of the thermocouples on each specimen are shown in Figures G-11 and G-12 (pages 344 and 345).

g. **Strain Gages**

Ten strain gages were attached to each of the test specimens. The strain gages were attached by resistance spotwelding. Five of the strain gages were attached to the specimen skin on the centerline of the stringers. For the -1 and -5 specimens the second five were attached to the back flange of the stringer on the centerline. The -3 specimen stringer back flange was shaped in the form of the bottom of a "J". Therefore, the second five strain gages were attached to the bottom leg of the "J" section .20 inches from the centerline of the stringer. The locations of these gages are shown in Figures G-13, G-14 and G-15 (pages 346, 347 and 348). Each of the strain gages was wired as a single legged bridge and was connected to a remote indicator.

h. **Data Recording**

All instrumentation was connected to a remote recorder. This recorder was the Data Acquisition and Interpretation System (DAISY I) shown in Figure G-16 (page 349, 350). This recording system has the capability of recording 400 channels of data simultaneously with a maximum sampling rate of four samples per second for each of the 400 channels. The data gathered by this system was digitized and placed on tape for readout after completion of the test. After completion of the test the data from each instrumentation device was plotted from the tape on an X-Y plotter.
MIDPOINT BETWEEN STRINGERS (TYP)

NOTE:
1. THERMOCOUPLES ON STRINGERS TO BE LOCATED ON CENTERLINE OF STRINGER WEB.
2. MIDPOINT DATA THERMOCOUPLES LOCATED ON CENTER OF STRINGER SPACING.
3. ALL THERMOCOUPLES TO BE CHROMEL-ALUMEL WIRE.
4. JUNCTION TO BE SPLIT-WELDED JUNCTION.
5. MAXIMUM SPACING BETWEEN WIRES - 0.06 IN.

Figure G-11 — THERMOCOUPLE LOCATIONS; 29-01015-1 Panels.
NOTE:
1. FOR INSTALLATION NOTES SEE FIGURE

Figure G-12 — THERMOCOUPLE LOCATIONS; 29-01015-3 & 29-01015-5 Panels.
NOTE:
1. EACH STRAIN GAGE WIRED FOR SINGLE LEG BRIDGE.

Figure G-13 — STRAIN GAGE LOCATIONS; 29-01015-1 Panels.
NOTE:
1. EACH STRAIN GAGE WIRED FOR SINGLE LEG BRIDGE.

Figure G-14 STRAIN GAGE LOCATIONS; 29-01015-3 Panels
NOTE:
1. EACH STRAIN GAGE WIRED FOR SINGLE LEG BRIDGE.

Figure G-15 — STRAIN GAGE LOCATIONS; 29-01015-5 Panels.
IV. DISCUSSION OF TEST RESULTS

1. Specimen 29-01015-1:

The test specimen sustained all of the imposed test conditions up to 900 F without adverse effects. The temperature was then returned to 600 F. The test load was then increased in increments toward failure. When the load reached 100,000 pounds the internal pressure leakage became excessive and resulted in a gradual loss of pressure. As the test load was increased the pressure loss became greater and resulted in a linear drop in pressure to 4.9 PSIG at failure. The test specimen failed at 149,000 pounds compressive load. The specimen failure occurred in the approximate center of the lower bay. Failure occurred as a result of the buckling of the stringer inner flange. In addition, four of the seven stringers showed shear failures of the stringer webs. The details of this failure are shown in Figure G-17 (page 352).

The results of this test were projected onto the Stress Ratio Diagram shown in Figure G-18 (page 353). The loss of internal pressure was taken into account in projecting these results. The test points on the diagram indicate that the specimen exceeded design expectations with a margin of safety of plus 1.46.

2. Specimen 29-01015-3:

This test specimen sustained all of the imposed test conditions up to 900 F without adverse effects. When the specimen was loaded to failure excessive pressure leakage started at approximately 75,000 pounds load. As the test load was increased beyond this point the internal pressure leakage increased until failure occurred. At failure the internal pressure was 4.9 PSIG. The test specimen failed at 122,000 pounds compressive load. The specimen failure occurred in the approximate center of the upper bay. Failure occurred as a result of buckling of the stringer inner flange and shear of the stringer web. This failure is shown in Figure G-19 (page 354). After the test unit was failed a manufacturing defect was observed. This defect was the lack of 7.0 inches of spotweld directly
A/ LOCAL BUCKLING - REFERENCE NACATN 3782 p. 29
INNER FLANGE AND WEB CRITICAL

\[ \sigma_{cr} = \frac{K_{w} \pi^2 E_{c}}{12 (1 - V_e^2) b_w^2} \]
\[ t_{w} / t_f = \frac{t_w}{t_f} = .04 / .06 = .667 \]

Temperature

\[ E_c \quad \quad \sigma_{cr} \]
R.T. \quad 17.5 \times 10^6 \quad 124000 psi
600° \quad 16.0 \times 10^6 \quad 113000 psi
900 \quad 14.8 \times 10^6 \quad 104500 psi

B/ COLUMN ALLOWABLE

\[ F_c = F_{cc} \left(1 - \frac{F_{cc}}{4 \pi^2 E_c} \right) \]
USE \( F_{cc} = \sigma_{cr} \); \( l^2 = 18 \)

Temperature

\[ E_c \quad \quad F_c \]
R.T. \quad 17.5 \times 10^6 \quad 98000 psi
600° \quad 16.0 \times 10^6 \quad 89400 psi
900 \quad 14.8 \times 10^6 \quad 82500 psi

Figure G-18 PANEL DESIGN ALLOWABLES; Panel 29-01015-1
Figure G-19 — FAILURE OF PANEL 29-01015-3; Showing Stringer, Web and Back Flange Failures.

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IV. 2. Specimen 29-01015-3: (Cont'd)

under the rib cap attachment between the center and the upper bays. These spotwelds were the stringer to skin attachment. The stringer that was not spotwelded was adjacent to the center stringer. The failure of the specimen occurred in this region. However, since the failure did not occur directly through the center of the unspotwelded section, the effect of the lack of spotwelds is unpredictable.

The results of this test were projected onto the Stress Ratio Diagram shown in Figure G-20 (page 356). The loss of pressure was also taken into account in this projection. The test points on the diagram indicate that the specimen exceeded design expectations with a margin of safety of plus .185.

3. Specimen 29-01015-5:

This specimen sustained all of the required conditions. During the failure test on this specimen no internal pressure was applied. The test specimen failed at 198,800 pound compressive load. This specimen failure occurred in the approximate center of the upper bay. Failure occurred as a result of buckling of the skin and shear failure of the stringer web. As a result of these failures the stringer inner flange also failed. The details of this failure are shown in Figure G-21 (page 357). The results of this test were also projected onto the Stress Ratio Diagram in Figure G-22 (page 358). The test points on this diagram indicate that this panel also exceeded design expectations with a margin of safety of plus .073.

4. Discussion:

During the tests of each compression specimen deflections normal to the skin was recorded. These deflections included the deflection of the specimen as well as the movement of the fixture as well as the test machine relative to the deflection indicator rack. The deflection data was reduced to remove all the extraneous deflections and leave only the true deflections relative to the test fixture. After data reduction the deflection data proved to be small and within the range of test data scatter. Since this data was small and showed no trends relative to the test temperature, this data is not recorded in this report.
A. LOCAL BUCKLING
INNER FLANGE CRITICAL

\[ \sigma_{cr} = KE_c \left(\frac{d}{b}\right)^2 \]

\[ = 0.64 \times \left(\frac{0.40}{0.36}\right)^2 \times 0.00790E_c \]

\[ \sigma_{cr} = KE_c \left(\frac{d}{b}\right)^2 \]

Temperature \( E_c \)

R. T. \( 17.5 \times 10^6 \) 138000 psi

600° \( 16.0 \times 10^6 \) 126000 psi

900° \( 14.8 \times 10^6 \) 117000 psi

B. COLUMN ALLOWABLE

\[ F_c = F_{cc} \left( \frac{1 - F_{cc} (L/E)^2}{4 \pi^2} \right) \]

\[ F_{cc} = \sigma_{cr} \times \frac{1}{1 + \text{in.}} \]

\[ \Theta^2 = 0.2105 \]

Temperature \( E_c \)

R. T. \( 17.5 \times 10^6 \) 95500 psi

600° \( 16.0 \times 10^6 \) 87200 psi

900° \( 14.8 \times 10^6 \) 81000 psi

**Figure G-20. Panel Design Allowables; Panel 29-01015-3**
Figure G-21 — FAILURE OF PANEL 29-01015-5; Showing Buckled Skin and Back Flanges, As Well As Shear Failures of Webs.
A. LOCAL BUCKLING
INNER FLANGE CRITICAL

\[ \sigma_{cr} = KE_{c} \left( \frac{t}{b} \right)^{2} \]

\[ = 0.64 \times \left( \frac{0.06}{0.33} \right)^{2} E = 0.0088 \text{ksi} \]

TEMPERATURE \( E_{c} \)          \( \sigma_{cr} \)  \( b = 0.62 - 0.040 = 0.58 \)
R. T     \( 17.5 \times 10^{6} \)      120000 psi
600°     \( 16.0 \times 10^{6} \)      109500 psi
900°     \( 14.8 \times 10^{6} \)      101300 psi

B. COLUMN ALLOWABLE \( F_{x} \)

\[ F_{c} = F_{cc} \left[ 1 - F_{cc} \left( \frac{1}{e} \right)^{2} \right] \]

\[ F_{cc} = \sigma_{cr} ; t^{1} = 18 \]

\[ C^{2} = 0.362 \]

TEMPERATURE \( F_{c} \)
R. T     10100 psi
600°     92500 psi
900°     85500 psi

Figure G-22  PANEL DESIGN ALLOWABLES; Panel 29-01015-5
IV. 4. Discussion: (Cont'd)

The strain gages located on the stringer sections showed that all stringers on all specimens were loading evenly and uniformly up to the point where a stringer strain started decreasing disproportionately indicating an imminent failure. The strain indicated by the strain gages compared closely to the analytic strain based on the compressive modulus as determined by coupons from the test assemblies. Figure G-23 (page 360) shows the average strain versus applied compressive load for each specimen during the failure tests at 600 F. By taking into account the change of the compression modulus \(E_c\) with temperature the modulus calculated from the indicated strain is within 8.2% of the theoretical adjusted value, Figure G-24 (page 361). Table G-2, below, shows the percentage difference between these values.

<table>
<thead>
<tr>
<th>Specimen Dash No.</th>
<th>Arbitrary Specimen Load lbs.</th>
<th>Gross Area in/ln</th>
<th>Strain Indicated (E) x10^-6</th>
<th>Coupon (E) @ 600 F</th>
<th>% Diff.</th>
</tr>
</thead>
<tbody>
<tr>
<td>-1</td>
<td>102,620</td>
<td>1.81</td>
<td>3866 14,600,000 15,900,000</td>
<td>-8.2</td>
<td></td>
</tr>
<tr>
<td>-3</td>
<td>90,600</td>
<td>1.71</td>
<td>3227 16,400,000 15,600,000</td>
<td>+5.2</td>
<td></td>
</tr>
<tr>
<td>-5</td>
<td>128,000</td>
<td>2.11</td>
<td>4035 15,000,000 15,200,000</td>
<td>-1.33</td>
<td></td>
</tr>
</tbody>
</table>

The thermocouples located on the skin and the inner stringer flanges showed that the temperature transport properties of the three stringer configurations were approximately the same up to 600 F. At 600 F the temperature differences start diverging indicating that beyond this point the stringer conductivity shape factor is becoming more effective in determining the transport properties. The skin temperature over the stringer centerline versus the difference between this temperature and the inner flange temperature are shown in Figure G-25 (page 362). The graphs of this figure indicate that anyone of the three types of stringers will conduct approximately the same amount of heat to the inner structure up to 600 F. Therefore, the selection of a stringer design cannot be determined by the temperature transport properties of these configurations.
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Figure G-23
AVERAGE STRAIN VS. APPLIED COMPRESSION LOAD FOR ALL SPECIMENS
Figure G-24  PREDICTED TEMPERATURE EFFECT ON MODULUS OF ELASTICITY: Ti Alloy B-120 VCA, $E = 17.5 \times 10^6$ PSI
Figure G-25  SKIN TEMPERATURE VS. STRINGER INTERNAL-FLANGE TEMPERATURE

△ TEMPERATURE THROUGH STRINGER - °F
(REF. FIGURE III-11 & III-12)
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Elyria, Ohio                              

Lear, Inc., Learcal Division              
Attn: D.W. Dressel,                       
    Factory Manager                      
3171 S. Bundy Drive                      
Santa Monica, California

Lockheed Aircraft Corporation
Attn: H. Caldwell, Mfg. Manager           
P.O. Box 511                              
Burbank, California

Lockheed Aircraft Corporation
Attn: H. Fletcher Brown                   
Manufacturing Manager                    
Marietta, Georgia

Lycoming Division                        
AVCO Manufacturing Corporation           
Attn: Mr. W.A. Panks                     
    Superintendent, Mfg. Engrg.           
Stratford, Connecticut

Marquardt Aircraft Company               
16555 Saticoy Street                     
Attn: Mr. John D. Liefeld,               
    Director of Manufacturing            
Van Nuys, California

Marquardt Aircraft Company               
Attn: Mr. Eugene L. Klein,               
    Chief Manufacturing Engr.            
Ogden, Utah

McDonnel Aircraft Corporation            
Attn: A.F. Hartwig, Chief                
    Industrial Engineer                  
Lambert                                  
St. Louis Municipal Airport              
St. Louis 3, Missouri

North American Aviation, Inc.            
Attn: W. E. Fore, Plant Engineer         
International Airport                    
Los Angeles 45,                           
California
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CORPORATIONS (Cont'd)

Northrop Aircraft, Inc.
1001 E. Broadway
Hawthorne, California

Parker Aircraft Company
Attn: Robert H. Davies, Vice Pres.
17352 Euclid Avenue
Cleveland 12, Ohio

Pratt & Whitney
United Aircraft Corporation
Attn: Mr. W.P. Gwinn
Hartford 3, Connecticut

Radioplane Company
Attn: F.D. Murphy, Manager
Engineering Adm. Department
8000 Woodley Avenue
Van Nuys, California

Rem-Cru Titanium, Inc.
Attn: Dr. W.C. Finally
Midland, Pennsylvania

Republic Aviation Corporation
Attn: Adolph Kastelowitz, Chief, Mfg. Engineering
Farmingdale, Long Island, N.Y.

Rohr Aircraft Corporation
P.O. Box 878
Chula Vista, California

Ryan Aeronautical Company
Attn: Lawrence M. Limbach
Vice President, Manufacturing
2701 Harbor Drive
San Diego 12, California

Sikorsky Aircraft Division
United Aircraft Corporation
Attn: Alex Sperber, Factory Manager
North Main Street
Stratford, Connecticut

Solar Aircraft Company
Attn: J.A. Logan, Manager
Facilities Division
2200 Pacific Highway
San Diego 12, California

Solar Aircraft Corporation
Attn: William Dixon, Manager
Production Engineering Division
1900 Bell Avenue
Des Moines 5, Iowa

Sperry Gyroscope Company - Div. of Sperry Rand Corporation
Attn: G.A. Richroath,
Vice Pres. of Manufacturing
Great Neck, New York

Temco Aircraft Corporation
Attn: V.N. Ferguson, Mfg. Manager
P.O. Box 6191
Dallas, Texas

Temco Aircraft Corporation
Engineering Library
Department 413D
P.O. Box 6191
Dallas, Texas

Temco Aircraft Corporation
Attn: E.F. Bushring
Plant Manager
P.O. Box 1056
Greenville, Texas
CORPORATIONS (Cont'd)

Thiokol Chemical Corporation
Reaction Motors Division
Attn: Contracts Department
Danville, New Jersey

Thompson Products, Inc.
Attn: Emil F. Gibian, Staff
Director, Ind. Engineering
23555 Euclid Avenue
Cleveland 17, Ohio

Titanium Metals Corporation
Attn: Mr. W. Minkler
233 Broadway
New York 17, New York

Westinghouse Electric Corporation
P.O. Box 228 ACT Division
Attn: Mr. E.C. Sedlack,
    Works Manager
Kansas City, Missouri

Westinghouse Electric Corporation
Air Arm Division
Friendship International Airport
Attn: Mr. F.E. Tighe, Manager
    Manufacturing Engineering
    and Tools
Baltimore 27, Maryland
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<td>Under repeated load test, resistance welded fuselage frame and wing leading edge, although adequate, were not equal to those riveted. Repeated loading of resistance welded shear panels gave marginal results. Other components were satisfactory under repeated loads.</td>
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