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C-5 A FUSELAGE STRESS ANALYSIS

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This report describes the work performed to provide analytical predictions that can be used to address cracking problems that have arisen in particular areas of the C-5A forward fuselage section. The specific objective of the present study was to investigate the various factors in as much detail as possible that are likely to contribute to cracking of the fuselage skin. Three distinct activities were undertaken to establish the cause of skin cracking: review and evaluation of available data, creation and execution of test plans, and performance of static and dynamic stress analysis; results from tests were used to validate analytical models. Recommendations for a follow-on study and possible fixes based on results obtained to date are presented.

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C-5A Fuselage Stress Analysis

This report describes the work performed to provide analytical predictions that can be used to address cracking problems that have arisen in particular areas of the C-5A forward fuselage section. The specific objective of the present study was to investigate various factors in as much detail as possible that are likely to contribute to cracking of the fuselage skin. Three distinct activities were undertaken to establish the cause of skin cracking: review and evaluation of available data, creation and execution of test plans, and performance of static and dynamic stress analysis; results from tests were used to validate analytical models. Recommendations for a follow-on study and possible fixes based on results obtained to date are presented.

Review and evaluation of available data

A preliminary review of fractographic, modal response, and flight test data indicate that cracking problems may be the result of more than just acoustic fatigue due to engine noise. For example, visual examination of most of the cracks showed evidence of stress corrosion. Also, the fuselage skin is subjected to cyclical loading due to internal pressurization. The stresses around the fasteners due to stress concentration are likely to be high enough to propagate fatigue fracture. Furthermore, the means of fastening the forward fuselage fairing attachment channel to the fuselage skin also appears to be contributing to the problem. The lower channel is not backed up by a stringer as is the upper channel - most of the forward fuselage skin cracks are at the lower channel fastener row and none are at the upper channel. Clearly, the C-5A fuselage skin cracking problem is the result of a combination of factors: high cycle-low stress fatigue, stress corrosion, corrosion fatigue, modal bending stresses and high stress concentration factors. The earlier study performed by Lockheed to investigate the cracking of the skin at water line (WL) 295 did not address all of the above factors. In the study reported here, more detailed tests and analyses were performed to investigate the causes of the crack problem.

Creation and execution of test plans

In view of the preliminary observation, several tests were planned and performed by the Wright Laboratory with the consulting assistance of CSA subcontractor - to determine the effects of various factors listed above.

- A modal survey test was performed by WL/FIBG to identify structural parameters and associated skin strain levels in the areas where forward fuselage skin cracks have occurred; the data from this test were used for validating finite element analysis.

- Material characteristics tests were performed by WL/MLSA to determine if operational usage and environmental exposure of the skin may be contributing to the problem.

- Coupon tests were performed by WL/MLSA to establish the material strength of skin material, residual strength of cracked skin, and fatigue life characteristics of the fuselage skin under different environmental conditions.
- Acoustic tests were performed by WL/FIBG to determine the effects of acoustic pressure on fuselage skin response.
- Videoholographic tests were conducted by WL/FIBG to define the modal response of the fuselage section.
- Test plans were prepared for the upcoming flight test to gather strain and acoustic data.

Structural analysis

The main objective was to analytically predict the causes of fuselage skin cracking. Both static and dynamic analyses were performed. First, normal modes analysis was performed using the finite element method. The test results from the videohologhrapy were used to validate the FE model. The FE model created for obtaining normal modes was subsequently used for static stress analysis.

Finite Element Model: Since the fuselage section is symmetric, only half of the section was modeled. Furthermore, only the segment starting from fuselage station 604 (FS604) through FS684 was modeled. The model primarily consisted of beams and shell elements. The frames and stringers were modeled as beams and the fuselage skin panels were modeled with shell elements. The cargo floor, troop floor, and fairing panel were modeled with QUAD4 and BAR elements. In the model there are several ROD, SHEAR and RIGID elements which were included to replicate the Lockheed model. The FE model is shown in Figure 1.

Normal Modes Analysis: Normal modes for the FE model were obtained for all frequencies below 550 Hz. There were 411 modes within this range. In order to identify the modes that are likely to contribute to the response, CSA's MSET computer program was used to find the modal strain energies in panels located near WL 295. The lowest mode with any significant modal strain energy in the panel at WL 295 had a frequency of 502 Hz. The lower modes generally involved the frames, stringers, fairing panel, cargo floor, or troop floor. Moreover, the modal survey tests performed by FIBG personnel did not show participation of any global modes in the peak response of the panel, which was measured at approximately 450 Hz, near WL 295. Clearly, the local bending modes of the fuselage skin panel contributed most to the response. The videoholography of the interior of the fuselage skin showed that the local panel modes near WL 295 range from 375-405 Hz for the lowest bending mode to 940-980 Hz for the 6×1 (six bulls eyes) mode. Thus to capture the four

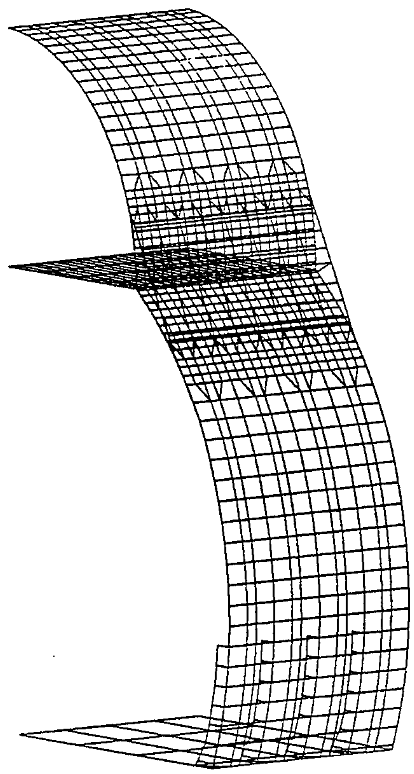

Figure 1: Finite element model of the fuselage section

Bending modes	Frequency (Hz)		
	FE model	Flat plate cccc	Test
First (1×1)	391	396	250 - 325
Second (2×1)	441	434	375 - 400
Third (3×1)	523	503	575 - 600
Fourth (4×1)	643	608	775 - 800
Fifth (5×1)	797	751	950 - 975
Sixth (6×1)	961	930	> 1000

Table 1: Bending modes of skin panel at WL 295

lowest bending modes of the panel at WL295 would have required computation of all modes up to 800 or 900 Hz. To get even the lowest four bending modes of the panel at WL 295, the highest frequency of the interest for normal modes would have to be increased to 800 - 900 Hz range. Furthermore, the FE mesh at WL 295 would have had to be refined to capture these higher modes. However, the test results indicated that the peak response at WL 295 is largely dependent on local modes, hence only a small segment of the fuselage section would be adequate to characterize the peak response.

Subsequently, a segment of the FE model containing the fuselage section from FS624 through FS684 and Stringer (ST) 41 through ST47 was used for further refinement. The fasteners that connect the fairing angle and the fuselage skin were modeled as BARs to accurately represent the connections between the two; a fastener hole was also modeled at FS660. The natural frequencies for this smaller FE model are listed in Table 1 along with the theoretical natural frequencies of a clamped flat plate of the same size (20" x 6.2"); the frequencies observed in the videoholography test are also shown. For fourth and higher modes, the test results show that the stiffness of the panel to be higher than that predicted by the FE analysis or theoretical solution. The results from the FE analysis are close to the clamped flat plate frequency for all modes. The slight difference is due to two differences: the fuselage panel is curved with 143" radius and the titanium doubler is not accounted for in the flat plate solution.

Static Analysis: The objective of the static stress analysis was to obtain the stresses near a typical fastener hole and to see how frame forces change when cracks propagate in the fuselage skin. The finite element models created and verified in the normal modes analysis were used to perform the static analysis. Specifically, the fine mesh segment model was inserted in the symmetric half FE model of the fuselage section (Figure 1), and the symmetric boundary conditions were applied. The transition from the fine mesh to the coarse mesh was achieved by using RSPLINEs.

Three analyses were performed; each analysis had two load cases, one with only uniform internal pressure of 8.3 psi and the other with 100 lb/ft^2 of cargo load in addition to the internal pressure. The initial analysis was a baseline analysis with the aforementioned two load cases. The second analysis had three simulated cracks: one 8" wide from FS620 to FS628; another 4" wide crack from FS644 to FS648; and one 0.14" crack at the fastener hole along WL295. The third analysis also had three simulated cracks: one 16" wide extending from FS616 to FS632; another 2" wide crack from FS644 to FS646; and one 0.15" crack at the fastener hole at about 45 degrees from WL295. The layout of the cracks were based on two criteria: to select cracks of various lengths, and to minimize the interaction between two cracks (i.e. to prevent one crack from modifying the stress field around the other.)

The uniform internal pressure load was applied to the fuselage skin panels; the fairing panel was not loaded. In the aircraft, the internal pressure will result in bulkhead pressure, which was simulated by applying an equivalent load along the aft edge of the FE model at FS684 and constraining the model in the fore-aft direction at the fore end (FS604). For all cases, symmetric boundary conditions were applied at the plane of symmetry. Additionally, the troop floor at FS604 was vertically supported. This boundary condition did not violate the symmetry condition.

Baseline Static Analysis

Internal pressure: The results from the baseline static stress analysis with only uniform internal pressure were as expected. In particular, the stress level in the fuselage skin panel in the hoop or circumferential direction remote from the fastener hole ranged from 10,000 to 12,000 psi. The theoretical value of the hoop stress in a thin-walled cylinder of 143" radius and 0.068" thickness when subjected to uniform radial pressure of 8.3 psi is 17,454 psi. Evidently, the frames and titanium doublers are picking up significant amount of load. The theoretical value of the meridional stress for an equivalent thin-walled cylinder is 8,727 psi; in the fuselage skin remote from the hole, meridional stresses ranged from 6,500 to 7,500 psi; the rest, apparently, is carried by the stringers. The major principal stress near the fastener hole was 19,100 psi, a jump of 75% from the stress observed in the areas remote to the fastener hole. Figure 2 shows a plot of the maximum principal stress along WL 295 near a typical fastener, which in this case was modeled at FS660.

Internal pressure with cargo load: When the cargo load is present, the major principal stress near the fastener hole increases from 19,100 psi to 19,860 psi and the location of the point where the maximum occurs shifts by 30 degrees from the horizontal. Additionally, as expected, the stresses in the fuselage skin near the cargo floor are high. Overall, the presence of cargo load influences the WL295 stresses only

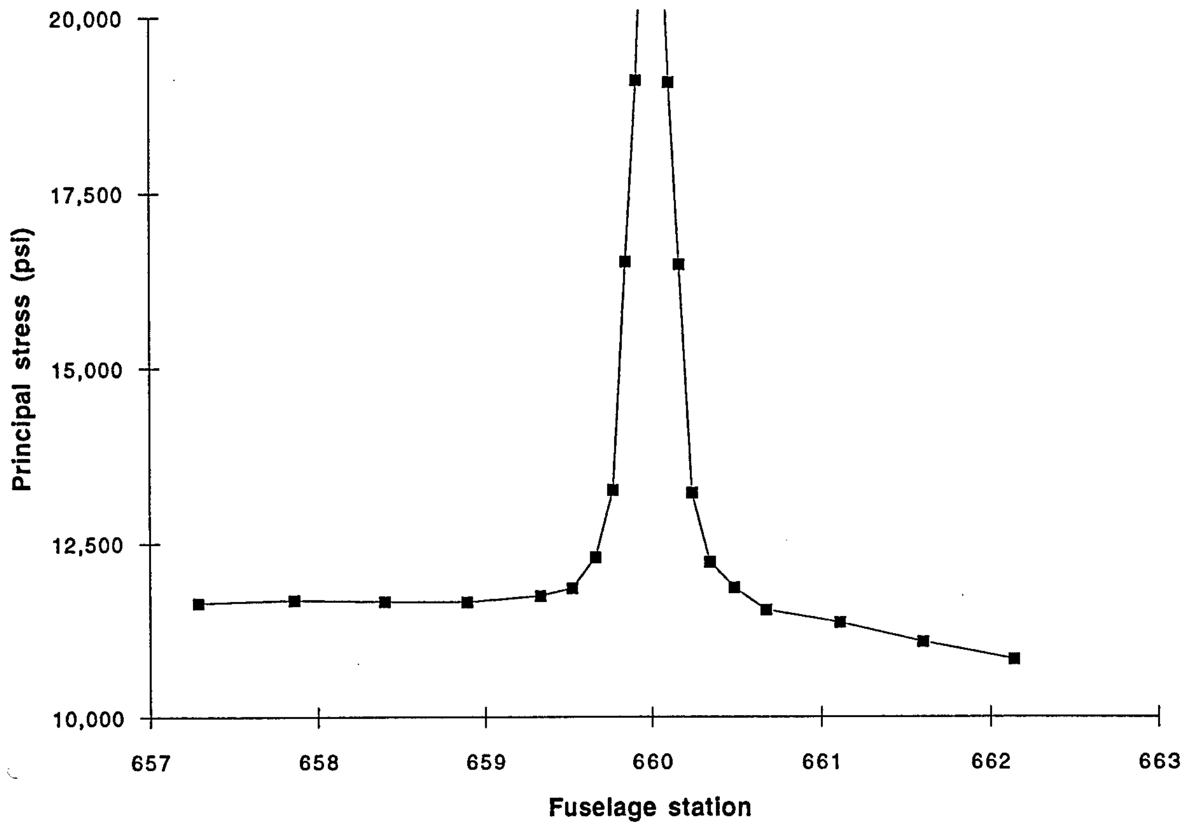

Figure 2: Maximum principal stress along WL 295 near a typical fastener at FS660

marginally.

Static Analysis with First Crack Pattern

Figure 3 shows the crack pattern along WL295 that was used to assess the effects of typical cracks on frames. Specifically, three cracks were modeled: one 8" crack along WL295 centered at FS624 was modeled to determine its effect on the frame at FS624; one 4" crack was modeled from FS644 to FS648 to obtain its effects on the FS644 frame; and one 0.14" crack was modeled at the fastener hole to determine the stresses in the fuselage skin ahead of the crack when a crack initiates and propagates from the fastener hole along WL295.

Internal pressure: For the internal pressure load only, the principal stress ahead of the crack near the fastener hole was 15,770 psi; the stress at the same location for the baseline analysis was 13,210 psi, a jump of 20%. Also, the maximum principal stress at the fore edge of the fastener hole increased from 19,100 psi to 20,430 psi. Table 2 shows the forces in the frames under various conditions. Note that an 8" crack causes the bending moment in the FS624 frame at WL295 to increase from 7,503 in-lb to 15,713 in-lb, or 109%. The axial force increases from 5,350 lb to 8,092 lb, or 51%. The 4" crack near the frame at FS644 increases the bending moment by 57% and the axial force by 11%.

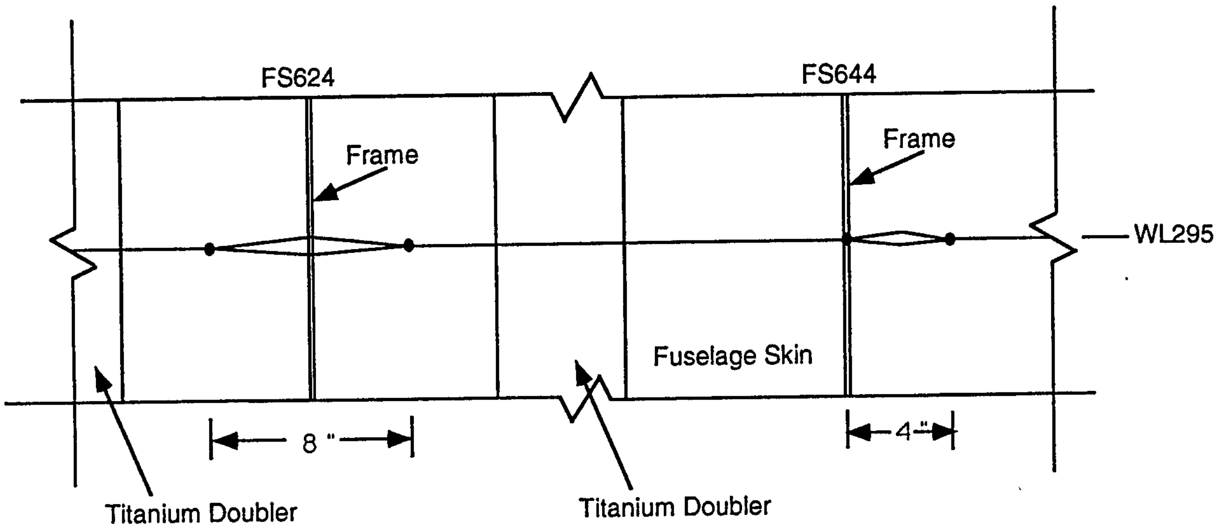

Figure 3: Cracks at WL 295; first crack pattern

Cargo load	Frame	Crack	Bending moment (in-lb)		Axial force (lb)	
			With crack	Baseline	With crack	Baseline
none	FS624	8"	15,713	7,503	8,092	5,350
none	FS644	4"	6,316	4,030	8,115	7,320
100 psf	FS624	8"	12,417	4,893	8,200	5,618
100 psf	FS644	4"	-1,589	-3,594	10,479	9,793

Table 2: Moments and forces in FS624 and FS644 frames at WL 295 for first set of cracks

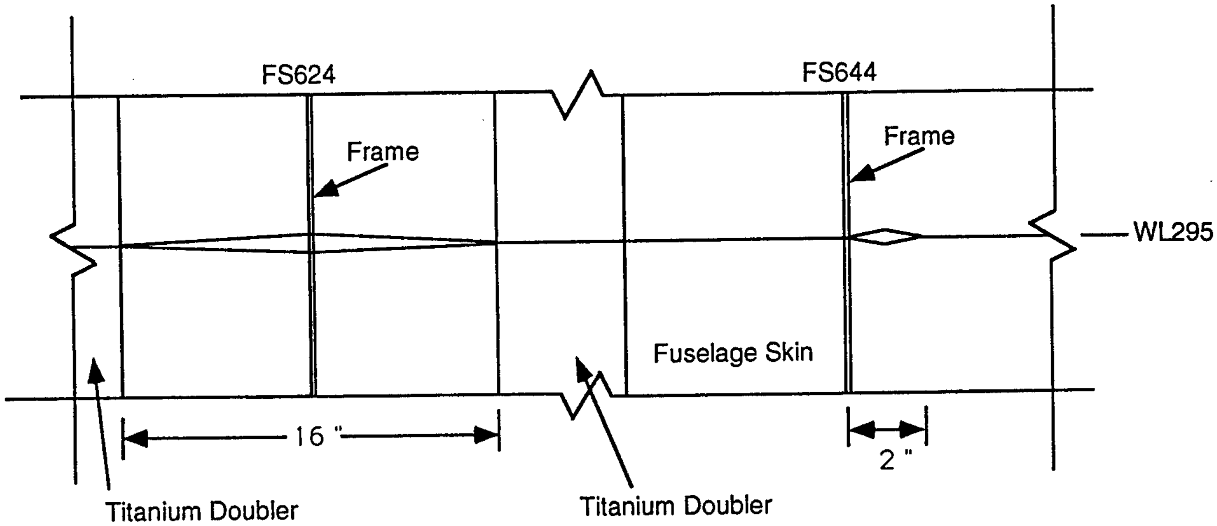

Figure 4: Cracks at WL 295; second crack pattern

Internal pressure with cargo load: The maximum principal stress at the fore edge of the fastener hole, for this load case, rose from 20,380 psi to 21,530 psi; the principal stress at the aft edge dropped from 19,860 psi to 15,490, however. Table 2 shows the forces in the frames when the cargo load is present. The presence of an 8" crack caused the bending moment in the FS624 frame at WL295 to increase from 4,893 in-lb to 12,417 in-lb, an increase of 154%. The axial force increased from 5,618 lb to 8,200 lb, an increase of 46%. The forces in the FS644 frame did not change significantly, however.

Static Analysis with Second Cracks Pattern

Figure 4 shows the cracks along WL295 that were modeled to assess the effects of typical cracks on frames. Specifically, three cracks were modeled: one 16" crack along WL295 centered at FS624 was modeled to determine its effect on the frame at FS624; a 2" crack was modeled from FS644 to FS646 to obtain its effects on the frame at FS644; and one 0.15" crack was modeled at the fastener hole to determine the stresses in the fuselage skin ahead of the crack when a crack initiates and propagates from the fastener hole at an angle making 45 degrees with WL295.

Internal pressure: For the internal pressure load only, the maximum principal stress along WL295 rose from 19,110 psi to 19,700 psi; the stresses at the aft end of the hole along WL295 dropped from 19,080 psi to 16,130 psi. Table 3 shows the forces in the frames under various conditions. Note that a 16" crack increases the bending moment in the FS624 frame at WL295 from 7,503 in-lb to 22,777 in-lb, or 204%. The axial force increases from 5,350 lb to 10,599 lb, or 98%. The 2" crack near the frame at FS644 increases the bending moment by 25% and the axial force by 5%.

Internal pressure with cargo load: The maximum principal stress at the fore edge of the fastener hole, for this load case, rose from 20,380 psi to 20,540 psi; the principal stress at the aft edge dropped from 19,860 psi to 19,180, however. Table 3 shows the forces in the frames when the cargo load is present. A 16" crack caused the bending moment in the FS624 frame at WL295 to increase from 4,893 in-lb to

Cargo	Frame	Crack load	Bending moment (in-lb)		Axial force (lb)	
			With crack	Baseline	With crack	Baseline
none	FS624	16"	22,777	7,503	10,599	5,350
none	FS644	2"	5,040	4,030	7,675	7,320
100 psf	FS624	16"	19,396	4,893	10,688	5,618
100 psf	FS644	2"	-2,654	-3,594	10,089	9,793

Table 3: Moments and forces in FS624 and FS644 frames at WL 295 for second set of cracks

19,396 in-lb, or 296%. The axial force increased from 5,618 lb to 10,688 lb, or 90%. The forces in the FS644 frame did not change significantly, however.

Summary and Recommendations

The results from the structural analysis and test results indicate that several factors are responsible for cracking problems in the C-5A fuselage skin. Specifically, the stress corrosion, corrosion fatigue, and high stress concentration factors are most likely causes. The results from the acoustic survey test showed that the overall sound pressure were not high enough to cause the skin cracking. The upcoming flight test will provide more detail on the effects of ground and flight operation. The results from the static stress analyses showed that the stress levels near the fastener hole are high enough to contribute to the corrosion deterioration of the skin around the fastener. Specifically, bending at the fastener head may be causing the edge of the head to "dig in" to the skin, setting up conditions for corrosion to cause a flaw that can easily grow when subjected to flight induced stresses.

So far, it appears that no simple solution exists for the correction of the skin cracking problem. Although most of the cracks occur along WL 295, it appears prudent to develop a repair that applies to all cracks or to develop a set of repairs covering all cracks. The current repair involves putting titanium doublers on both sides of the cracks. The drawback of this repair is that it requires more holes be made in the skin, thus creating the potential for more crack sites. Other possible repairs include unloading WL 295 fasteners by designing a new structure for supporting the external fairing and routing the loads down to the next lower stringer. Also, the cracks occurring at other stringers could be eliminated by putting doublers on the opposite side of the skin. These repairs will not address the corrosion damage that already exists in these areas. Another repair that addresses all of the concerns is a

Boron Epoxy patch. The current fasteners should be replaced with flush countersunk fasteners to remove some of the existing corrosion and provide a flat surface to bond a multiple layer Boron Epoxy patch. Such a patch will reduce the stresses at the crack tip and carry the load across the crack. Also, the patch would inhibit further crack growth, as well as crack growth from existing corrosion pits. Furthermore, since no new holes are required, new potential crack initiation sites would not be created. It appears that seven layers of Boron are needed in multiple directions to carry the hoop tension and longitudinal stresses from pressurization. An extension of the FEA work to investigate the Boron Epoxy repair is recommended.