CHAPTER

57

WINGS

Subject/Page	Date
57-EFFECTIVE PAGES	
1 thru 20	Jul 10/2016
57-CONTENTS	
1	Mar 10/2014
2	Mar 10/2014
3	Mar 10/2014
4	Mar 10/2015
5	Nov 10/2015
6	Jul 10/2015
7	Mar 10/2016
8	Nov 10/2015
9	Nov 10/2015
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2	Nov 10/2014
3	Nov 10/2014
4	Nov 10/2014
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105	Nov 10/2012
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112	Jul 10/2015
113	Nov 10/2012
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118	Nov 10/2012
57-00-03 ALLOWABLE DAMAGE 1 (cont)	
119	Nov 10/2012
120	Nov 10/2012
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1	Nov 10/2012
2	Nov 10/2012
57-10-01 IDENTIFICATION 2	
1	Nov 10/2012
2	Nov 10/2012
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R 101	Jul 10/2016
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103	Nov 10/2012
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110	Nov 10/2012
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208	Nov 10/2014
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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-10-03 REPAIR 4	
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204	Nov 10/2014
205	Nov 10/2014
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208	Nov 10/2014
209	Nov 10/2014
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201	Nov 10/2014
202	Nov 10/2014
203	Nov 10/2014
57-10-03 REPAIR 7 (cont)	
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3	Mar 10/2014
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201	Nov 10/2012
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211	Mar 10/2016
212	Mar 10/2016
213	Mar 10/2016
214	Mar 10/2016
215	Mar 10/2016

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-10-10 REPAIR 2 (cont)	
216	Mar 10/2016
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2	Nov 10/2013
3	Nov 10/2013
4	Nov 10/2012
5	Nov 10/2012

A = Added, R = Revised, D = Deleted, O = Overflow

57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-20-01 IDENTIFICATION 2 (cont)	
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207	Nov 10/2012
208	Nov 10/2012
209	Nov 10/2012

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page		Date
57-20-03 REPAIR 4 (cont)		
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O	3	Jul 10/2016
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	5	Jul 10/2015
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	2	Jul 10/2015
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57-20-10 ALLOWABLE DAMAGE 1 (cont)		
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57-20-10 ALLOWABLE DAMAGE 2		
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O	102	Jul 10/2016
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	118	Nov 10/2012
57-20-10 REPAIR 1		
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A = Added, R = Revised, D = Deleted, O = Overflow

57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

CHAPTER 57
WINGS

Subject/Page	Date
57-20-90 IDENTIFICATION 1	
1	Jul 10/2015
2	Jul 10/2015
3	Nov 10/2012
4	Nov 10/2012
57-20-90 IDENTIFICATION 2	
1	Nov 10/2012
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4	Nov 10/2012
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57-20-90 ALLOWABLE DAMAGE 2	
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107	Nov 10/2012
57-20-90 ALLOWABLE DAMAGE 2 (cont)	
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57-20-90 REPAIR 1	
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57-20-90 REPAIR 3 (cont)	
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3	Mar 10/2013
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R 106	Jul 10/2016
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103	Nov 10/2013
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109	Nov 10/2013
110	Nov 10/2012
111	Nov 10/2012

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

CHAPTER 57
WINGS

Subject/Page		Date
57-30-01 ALLOWABLE DAMAGE 2 (cont)		
	112	Nov 10/2012
	113	Nov 10/2012
	114	Nov 10/2012
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	26	Nov 10/2012

A = Added, R = Revised, D = Deleted, O = Overflow

57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-41-01 IDENTIFICATION 4	
1	Nov 10/2012
O 2	Jul 10/2016
3	Nov 10/2012
4	Nov 10/2012
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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page		Date
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	2	Nov 10/2012
O	3	Jul 10/2016
O	4	Jul 10/2016
O	5	Jul 10/2016
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	7	Nov 10/2012
	8	Nov 10/2012
	9	Nov 10/2012
	10	Nov 10/2012
57-41-02 ALLOWABLE DAMAGE 1		
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	102	Nov 10/2012
	103	Nov 10/2012
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	108	Nov 10/2012
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	112	Nov 10/2012
	113	Nov 10/2012
	114	Nov 10/2012
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	103	Nov 10/2012
	104	Nov 10/2012
	105	Nov 10/2015
	106	Nov 10/2012
	107	Nov 10/2012
	108	Nov 10/2012
57-41-02 ALLOWABLE DAMAGE 2 (cont)		
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	112	Nov 10/2012
	113	Nov 10/2012
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57-41-02 REPAIR 1		
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57-41-02 REPAIR 2		
	201	Nov 10/2012
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57-42-01 IDENTIFICATION 1		
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	3	Mar 10/2014
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	5	Nov 10/2012
	6	Mar 10/2014
O	7	Jul 10/2016
	8	Nov 10/2012
	9	Mar 10/2014
O	10	Jul 10/2016
	11	Nov 10/2012
	12	Mar 10/2014
O	13	Jul 10/2016
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	21	Nov 10/2012
	22	Nov 10/2012
57-42-01 ALLOWABLE DAMAGE 1		
	101	Nov 10/2012
	102	Nov 10/2012
	103	Mar 10/2014
	104	Mar 10/2014
	105	Nov 10/2012
	106	Nov 10/2015
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	202	Nov 10/2015
	203	Nov 10/2012
	204	Nov 10/2012
	205	Nov 10/2012
	206	Nov 10/2012
57-42-01 REPAIR 2		
	201	Nov 10/2012
	202	Nov 10/2015
	203	Nov 10/2012
	204	Nov 10/2012
	205	Nov 10/2012
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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
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57-42-01 REPAIR 4	
201	Mar 10/2013
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57-42-01 REPAIR 5	
201	Nov 10/2012
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57-42-02 IDENTIFICATION 1	
1	Jul 10/2015
O 2	Jul 10/2016
57-42-02 IDENTIFICATION 1 (cont)	
3	Nov 10/2012
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6	Nov 10/2012
57-42-02 ALLOWABLE DAMAGE 1	
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107	Nov 10/2013
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57-42-02 REPAIR 1	
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O 2	Jul 10/2016
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4	Nov 10/2012
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57-43-02 IDENTIFICATION 1 (cont)	
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57-43-02 ALLOWABLE DAMAGE 1	
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57-43-02 REPAIR 2	
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57-43-90 ALLOWABLE DAMAGE 1	
101	Nov 10/2012
102	Nov 10/2012
103	Nov 10/2012
104	Nov 10/2015
105	Nov 10/2012
106	Nov 10/2012

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
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107	Nov 10/2012
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57-43-90 REPAIR 1	
201	Nov 10/2012
202	Nov 10/2012
57-51-01 IDENTIFICATION 1	
1	Nov 10/2012
O 2	Jul 10/2016
3	Nov 10/2012
4	Nov 10/2012
O 5	Jul 10/2016
O 6	Jul 10/2016
O 7	Jul 10/2016
8	Nov 10/2012
9	Nov 10/2012
O 10	Jul 10/2016
O 11	Jul 10/2016
O 12	Jul 10/2016
O 13	Jul 10/2016
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57-51-01 ALLOWABLE DAMAGE 1	
101	Nov 10/2015
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106	Nov 10/2015
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110	Nov 10/2015
111	Nov 10/2015
O 112	Jul 10/2016
113	Nov 10/2015
114	Nov 10/2015
57-51-01 ALLOWABLE DAMAGE 1 (cont)	
115	Nov 10/2015
116	Nov 10/2015
117	Nov 10/2015
118	Nov 10/2015
119	Nov 10/2015
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57-51-01 REPAIR 1	
201	Nov 10/2012
202	Nov 10/2012
203	Nov 10/2012
204	Nov 10/2012
205	Nov 10/2014
206	Nov 10/2014
207	Nov 10/2014
O 208	Jul 10/2016
57-51-01 REPAIR 2	
201	Nov 10/2015
202	Nov 10/2015
203	Nov 10/2015
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57-51-02 IDENTIFICATION 1	
1	Jul 10/2015
O 2	Jul 10/2016
3	Nov 10/2012
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9	Nov 10/2012
10	Nov 10/2012
57-51-02 IDENTIFICATION 1 (cont)	
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57-51-02 ALLOWABLE DAMAGE 1	
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57-51-02 REPAIR 1	
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209	Jul 10/2015
210	Jul 10/2015
57-51-02 REPAIR 2	
201	Jul 10/2015
202	Jul 10/2015
203	Nov 10/2015

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-51-02 REPAIR 2 (cont)	
204	Jul 10/2015
205	Jul 10/2015
206	Jul 10/2015
207	Jul 10/2015
208	Jul 10/2015
209	Jul 10/2015
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203	Nov 10/2015
R 204	Jul 10/2016
O 205	Jul 10/2016
O 206	Jul 10/2016
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57-51-02 REPAIR 4	
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4	Mar 10/2013
57-51-14 ALLOWABLE DAMAGE 1	
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105	Nov 10/2012
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110	Nov 10/2012
111	Nov 10/2012
112	BLANK
57-51-90 IDENTIFICATION 1	
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5	Nov 10/2012
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57-51-90 ALLOWABLE DAMAGE 1	
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105	Nov 10/2012
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57-53-01 IDENTIFICATION 1	
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O 2	Jul 10/2016
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4	Nov 10/2012
57-53-01 IDENTIFICATION 1 (cont)	
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57-53-01 IDENTIFICATION 3	
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O 2	Jul 10/2016
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2	Nov 10/2012
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57-53-01 ALLOWABLE DAMAGE 1	
101	Nov 10/2012
102	Nov 10/2012

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57-EFFECTIVE PAGES

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CHAPTER 57
WINGS

Subject/Page	Date
57-53-01 ALLOWABLE DAMAGE 1 (cont)	
103	Nov 10/2012
104	Nov 10/2012
105	Nov 10/2012
106	Nov 10/2012
107	Nov 10/2012
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111	Nov 10/2012
112	BLANK
57-53-01 ALLOWABLE DAMAGE 2	
R 101	Jul 10/2016
102	Mar 10/2015
103	Nov 10/2015
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57-53-01 ALLOWABLE DAMAGE 3	
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O 105	Jul 10/2016
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112	Nov 10/2012
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57-53-01 ALLOWABLE DAMAGE 3 (cont)	
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57-53-01 REPAIR 2	
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57-53-01 REPAIR 3	
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57-53-01 REPAIR 6	
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57-53-01 REPAIR 7	
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57-53-01 REPAIR 8	
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57-53-01 REPAIR 9	
201	Nov 10/2014
202	Nov 10/2014

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57-EFFECTIVE PAGES

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Subject/Page	Date
57-53-01 REPAIR 9 (cont)	
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57-53-01 REPAIR 10	
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57-53-01 REPAIR 11	
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57-53-01 REPAIR 12	
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212	Jul 10/2015
57-53-01 REPAIR 14	
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5	Nov 10/2012
57-53-02 IDENTIFICATION 1 (cont)	
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O 9	Jul 10/2016
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O 2	Jul 10/2016
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O 2	Jul 10/2016
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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
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113	Nov 10/2014
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57-53-02 ALLOWABLE DAMAGE 2	
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O 108	Jul 10/2016
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207	Nov 10/2012
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57-53-02 REPAIR 10	
201	Nov 10/2012
202	Nov 10/2015

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
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57-53-02 REPAIR 11	
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23	Nov 10/2012
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57-53-70 IDENTIFICATION 2 (cont)	
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27	Nov 10/2012
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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

CHAPTER 57
WINGS

Subject/Page	Date
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102	Jul 10/2015
103	Jul 10/2015
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109	Nov 10/2012
110	Nov 10/2012
111	Nov 10/2012
112	Nov 10/2012
57-53-70 REPAIR 1	
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207	Mar 10/2016
208	Jul 10/2015
209	Mar 10/2016
210	BLANK
57-53-70 REPAIR 2	
201	Mar 10/2015
202	Mar 10/2015
203	Nov 10/2015
204	Nov 10/2015
205	Nov 10/2015
206	Mar 10/2015
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210	BLANK
57-53-70 REPAIR 3	
201	Nov 10/2015
202	Nov 10/2015
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214	Nov 10/2015
57-53-70 REPAIR 4	
201	Mar 10/2016
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221	Mar 10/2016
57-53-70 REPAIR 4 (cont)	
222	BLANK
57-53-71 IDENTIFICATION 1	
1	Jul 10/2015
O 2	Jul 10/2016
3	Nov 10/2012
4	Nov 10/2012
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6	Nov 10/2012
7	Nov 10/2012
8	Nov 10/2012
57-53-71 ALLOWABLE DAMAGE 1	
101	Nov 10/2012
102	Nov 10/2012
103	Nov 10/2012
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107	Nov 10/2012
108	BLANK
57-53-71 REPAIR 1	
201	Nov 10/2012
202	Nov 10/2012
203	Nov 10/2012
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205	Jul 10/2015
206	Nov 10/2012
57-60-00 GENERAL	
1	Nov 10/2012
2	BLANK
57-60-01 IDENTIFICATION 1	
1	Jul 10/2015
2	Jul 10/2015
3	Jul 10/2015
4	Jul 10/2015
5	Jul 10/2015
6	Jul 10/2015

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-60-01 IDENTIFICATION 1 (cont)	
7	Jul 10/2015
8	Jul 10/2015
9	Jul 10/2015
10	Jul 10/2015
11	Jul 10/2015
12	Jul 10/2015
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20	Jul 10/2015
57-60-01 IDENTIFICATION 2	
1	Nov 10/2012
2	Nov 10/2012
3	Nov 10/2012
4	Nov 10/2012
5	Mar 10/2013
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57-60-01 ALLOWABLE DAMAGE 1	
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57-60-01 ALLOWABLE DAMAGE 1 (cont)	
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57-60-01 REPAIR 1	
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211	Nov 10/2012
O 212	Jul 10/2016
O 213	Jul 10/2016
214	BLANK
57-60-01 REPAIR 2	
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202	Nov 10/2012
203	Nov 10/2012
204	Nov 10/2012
57-60-01 REPAIR 2 (cont)	
205	Nov 10/2012
206	Mar 10/2013
57-60-02 IDENTIFICATION 1	
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5	Nov 10/2012
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14	Mar 10/2013
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20	Mar 10/2013
57-60-02 IDENTIFICATION 2	
1	Nov 10/2012
2	Nov 10/2012
3	Nov 10/2012
4	BLANK
57-60-02 ALLOWABLE DAMAGE 1	
101	Nov 10/2012
102	Nov 10/2012
103	Nov 10/2012
104	Nov 10/2012
105	Nov 10/2012
O 106	Jul 10/2016
107	Nov 10/2012

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57-EFFECTIVE PAGES

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

Subject/Page	Date
57-60-02 ALLOWABLE DAMAGE 1 (cont)	
108	Nov 10/2012
109	Nov 10/2012
110	Nov 10/2012
111	Nov 10/2012
112	Nov 10/2012
113	Nov 10/2012
114	Nov 10/2012
57-60-02 ALLOWABLE DAMAGE 2	
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57-60-02 REPAIR 1	
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57-60-02 REPAIR 2	
201	Nov 10/2012
202	BLANK
57-60-70 ALLOWABLE DAMAGE 1	
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102	BLANK
57-60-70 REPAIR 1	
201	Nov 10/2012
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57-60-90 IDENTIFICATION 1	
1	Jul 10/2014
2	Nov 10/2012
3	Nov 10/2012
4	Nov 10/2012
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6	BLANK
57-60-90 IDENTIFICATION 2	
1	Nov 10/2012
2	Nov 10/2012
3	Nov 10/2012
4	BLANK
57-60-90 ALLOWABLE DAMAGE 1	
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57-60-90 ALLOWABLE DAMAGE 2	
101	Nov 10/2012
102	Nov 10/2012
103	Nov 10/2012
104	Nov 10/2012
57-60-90 ALLOWABLE DAMAGE 2 (cont)	
105	Nov 10/2012
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57-60-90 REPAIR 1	
201	Nov 10/2012
202	BLANK
57-60-90 REPAIR 2	
201	Nov 10/2012
202	BLANK
57-70-00 GENERAL	
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57-70-01 IDENTIFICATION 1	
1	Nov 10/2012
2	Nov 10/2012
3	Nov 10/2012
4	Nov 10/2012
57-70-01 IDENTIFICATION 2	
1	Nov 10/2012
O 2	Jul 10/2016
3	Nov 10/2012
4	Nov 10/2012
5	Nov 10/2012
6	Nov 10/2012
7	Nov 10/2012
8	Nov 10/2012
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14	Nov 10/2012
15	Nov 10/2012
16	Nov 10/2012
17	Nov 10/2012
18	BLANK

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57-EFFECTIVE PAGES

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CHAPTER 57
WINGS

Subject/Page	Date
57-70-01 ALLOWABLE DAMAGE 1	
101	Mar 10/2016
102	Nov 10/2012
103	Nov 10/2012
104	Nov 10/2012
105	Nov 10/2012
106	Nov 10/2012
107	Nov 10/2012
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119	Mar 10/2013
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121	Mar 10/2016
122	Mar 10/2016
57-70-01 REPAIR 1	
201	Nov 10/2012
202	Nov 10/2012
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57-70-02 IDENTIFICATION 1	
1	Nov 10/2012
2	Nov 10/2012
3	Nov 10/2012
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57-70-02 IDENTIFICATION 2	
1	Nov 10/2012
O 2	Jul 10/2016
3	Nov 10/2012
57-70-02 IDENTIFICATION 2 (cont)	
4	Nov 10/2012
5	Nov 10/2012
6	Nov 10/2012
7	Nov 10/2012
8	Nov 10/2012
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10	Nov 10/2012
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57-70-02 ALLOWABLE DAMAGE 1	
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57-70-02 REPAIR 1	
201	Nov 10/2012
202	Nov 10/2012
203	Nov 10/2015
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205	Nov 10/2012
206	BLANK
57-70-90 ALLOWABLE DAMAGE 1	
101	Nov 10/2012
102	BLANK
57-70-90 REPAIR 1	
201	Nov 10/2012
202	BLANK

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57-EFFECTIVE PAGES

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CHAPTER 57
WINGS

SUBJECT	CHAPTER SECTION SUBJECT
WINGS - GENERAL	**57-00-00**
GENERAL - Wings	
WING STRINGERS	**57-00-03**
ALLOWABLE DAMAGE 1 - Wing Stringers	
CENTER WING SKIN	**57-10-01**
IDENTIFICATION 1 - Wing Center Section Upper Skin Identification	
IDENTIFICATION 2 - Wing Center Section Lower Skin Identification	
ALLOWABLE DAMAGE 1 - Wing Center Section Upper Skin Panels	
ALLOWABLE DAMAGE 2 - Wing Center Section Lower Skin Panels	
REPAIR 1 - Wing Center Section Upper Skin Panels	
REPAIR 2 - Wing Center Section Lower Skin Panels	
CENTER WING STRINGERS	**57-10-03**
ALLOWABLE DAMAGE 1 - Wing Center Section Stiffeners	
REPAIR 1 - Upper Zee Stringer on the Center Wing	
REPAIR 2 - Upper Vent Stringer on the Center Wing Adjacent to a Padup Area	
REPAIR 3 - Upper Vent Stringer on the Center Wing Not Adjacent to a Padup Area	
REPAIR 4 - Lower Splice Stringer on the Center Wing Adjacent to a Padup	
REPAIR 5 - Lower Splice Stringer on the Center Wing Not Adjacent to a Padup Area	
REPAIR 6 - Lower Zee Stringer on the Center Wing Adjacent to a Padup Area	
REPAIR 7 - Lower Zee Stringer on the Center Wing Not Adjacent to a Padup Area	
CENTER WING SPARS	**57-10-10**
IDENTIFICATION 1 - Wing Center Section Front Spar	
IDENTIFICATION 2 - Wing Center Section Rear Spar for Airplane Line Numbers 1 Through 4514	
IDENTIFICATION 3 - Wing Center Section Rear Spar for Airplane Line Numbers 4515 and On	
ALLOWABLE DAMAGE 1 - Wing Center Section Front Spar	
ALLOWABLE DAMAGE 2 - Wing Center Section Rear Spar	
REPAIR 1 - Wing Center Section Front Spar	
REPAIR 2 - Wing Center Section Rear Spar	
REPAIR 3 - Wing Center Section Rear Spar Vertical Pressure Deck Web Repair	

57-CONTENTS

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CHAPTER 57
WINGS

SUBJECT	CHAPTER SECTION SUBJECT
CENTER WING BEAMS	**57-10-13**
IDENTIFICATION 1 - Wing Center Section Spanwise Beams	
IDENTIFICATION 2 - Wing Center Section Support Beams	
ALLOWABLE DAMAGE 1 - Wing Center Section Spanwise Beams	
ALLOWABLE DAMAGE 2 - Wing Center Section Support Beams	
REPAIR 1 - Wing Center Section Spanwise Beams	
REPAIR 2 - Wing Center Section Support Beams	
OUTER WING SKIN	**57-20-01**
IDENTIFICATION 1 - Outer Wing Upper Inspar Skin	
IDENTIFICATION 2 - Outer Wing Lower Inspar Skin	
ALLOWABLE DAMAGE 1 - Wing Inspar Skin Inboard of WBL 616.75	
ALLOWABLE DAMAGE 2 - Wing Inspar Skin Outboard of WBL 616.75	
REPAIR 1 - Wing Skin Lower Surface - Corrosion Repair at the Fuel Tank Access Panel Clamp Ring	
OUTER WING STRINGERS	**57-20-03**
IDENTIFICATION 1 - Outer Wing Upper Stringers	
IDENTIFICATION 2 - Outer Wing Lower Stringers	
REPAIR 1 - Repair for Damage to an Upper Splice Stringer on the Outer Wing	
REPAIR 2 - Repair For Damage To an Upper Zee Stringer on the Outer Wing	
REPAIR 3 - Repair for Damage Between Ribs to a Vent Stringer on the Outer Wing	
REPAIR 4 - Repair for Damage At a Rib to a Vent Stringer on the Outer Wing	
REPAIR 5 - Repair For Damage to a Lower Zee Stringer on the Outer Wing	
OUTER WING RIBS	**57-20-09**
IDENTIFICATION 1 - Outer Wing Ribs	
ALLOWABLE DAMAGE 1 - Outer Wing Ribs	
REPAIR 1 - Outer Wing Rib Repairs	
OUTER WING SPARS	**57-20-10**
IDENTIFICATION 1 - Outer Wing Front Spar Structure	
IDENTIFICATION 2 - Outer Wing Rear Spar	
ALLOWABLE DAMAGE 1 - Outer Wing Front Spar	

57-CONTENTS

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SUBJECT	CHAPTER SECTION SUBJECT
ALLOWABLE DAMAGE 2 - Outer Wing Rear Spar	
ALLOWABLE DAMAGE 3 - Outer Wing Spar Web at Fuel Boost Pump	
REPAIR 1 - Outer Wing Spars	
OUTER WING ATTACHMENT FITTINGS	**57-20-90**
IDENTIFICATION 1 - Outer Wing Front Spar Fittings	
IDENTIFICATION 2 - Outer Wing Rear Spar Fittings	
ALLOWABLE DAMAGE 1 - Outer Wing Front Spar Fittings	
ALLOWABLE DAMAGE 2 - Outer Wing Rear Spar Fittings	
REPAIR 1 - Outer Wing Front Spar Fittings	
REPAIR 2 - Outer Wing Rear Spar Fittings - Main Landing Gear Forward Trunnion Support Fitting Corrosion Repair	
REPAIR 3 - Main Landing Gear Forward Trunnion Support Fitting Corrosion Repair	
WING TIP SKIN	**57-30-01**
IDENTIFICATION 1 - Wing Tip Skin	
IDENTIFICATION 2 - Winglet Skin	
ALLOWABLE DAMAGE 1 - Wing Tip Skin	
ALLOWABLE DAMAGE 2 - Winglet Skin	
REPAIR 1 - External Repair of the Wing Tip Skin	
REPAIR 2 - Wing Tip Skin Flush Repair	
REPAIR 3 - Winglet Leading Edge Skin Flush Repair	
REPAIR 4 - Winglet Skin Repair	
WING TIP STRUCTURE	**57-30-02**
IDENTIFICATION 1 - Wing Tip Structure	
IDENTIFICATION 2 - Winglet Structure	
ALLOWABLE DAMAGE 1 - Wing Tip Structure	
ALLOWABLE DAMAGE 2 - Winglet Structure	
REPAIR 1 - Wing Tip Structure	
REPAIR 2 - Winglet Structure	
LEADING EDGE SKIN	**57-41-01**
IDENTIFICATION 1 - Wing Inboard Fixed Leading Edge Upper Skin Panels	

57-CONTENTS

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SUBJECT	CHAPTER SECTION SUBJECT
IDENTIFICATION 2 - Wing Inboard Fixed Leading Edge Lower Skin Panels	
IDENTIFICATION 3 - Wing Outboard Fixed Leading Edge Upper Skin Panels	
IDENTIFICATION 4 - Wing Outboard Fixed Leading Edge Lower Skin Panels	
ALLOWABLE DAMAGE 1 - Wing Fixed Leading Edge Panels	
REPAIR 1 - Wing Fixed Leading Edge Skin Panels Made of Composite Materials	
REPAIR 2 - Wing Inboard Fixed Leading Edge Skin - Flush Repair for Damage Between Ribs and Stringers	
REPAIR 3 - Wing Inboard Fixed Leading Edge Skin - Flush Repair for Damage at the Front Spar	
REPAIR 4 - Wing Inboard Fixed Leading Edge Skin - Flush Repair for Damage at a Rib	
REPAIR 5 - Wing Inboard Fixed Leading Edge Skin - Flush Repair for Damage at a Stringer	
REPAIR 6 - Wing Leading Edge Skin - Erosion/ Edge Strip Repair	
WING FIXED LEADING EDGE STRUCTURE	**57-41-02**
IDENTIFICATION 1 - Wing Inboard Fixed Leading Edge Structure	
IDENTIFICATION 2 - Wing Outboard Fixed Leading Edge Structure	
ALLOWABLE DAMAGE 1 - Wing Inboard Fixed Leading Edge Structure	
ALLOWABLE DAMAGE 2 - Wing Outboard Fixed Leading Edge Structure	
REPAIR 1 - Wing Inboard Fixed Leading Edge Structure	
REPAIR 2 - Wing Outboard Fixed Leading Edge Structure	
WING LEADING EDGE SLAT SKIN	**57-42-01**
IDENTIFICATION 1 - Wing Leading Edge Slat Skin	
ALLOWABLE DAMAGE 1 - Wing Leading Edge Slat Skin	
REPAIR 1 - Leading Edge Slat Skin - Flush Repair Between the Ribs Forward of the Nose Beam	
REPAIR 2 - Leading Edge Slat Skin - Flush Repair Aft of the Nose Beam	
REPAIR 3 - Leading Edge Slat Cove Skin	
REPAIR 4 - Outboard Leading Edge Slat Skin Trailing Edge Wedge	
REPAIR 5 - Leading Edge Slat Skin Splice Repair - Slats 1 and 8	
REPAIR 6 - Leading Edge Slat Trailing Edge Wedge Lower Skin Wear Repair	
REPAIR 7 - Leading Edge Slat Skin Splice Repair - Slats 2 thru 7	
REPAIR 8 - Leading Edge Slat Nose Skin External Doubler Repair at the Actuator Rib	

57-CONTENTS

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CHAPTER 57
WINGS

SUBJECT	CHAPTER SECTION SUBJECT
WING LEADING EDGE SLAT STRUCTURE	**57-42-02**
IDENTIFICATION 1 - Outboard Wing Leading Edge Slat Structure	
ALLOWABLE DAMAGE 1 - Wing Leading Edge Slat Structure	
REPAIR 1 - Wing Leading Edge Slat Structure at a Nose Beam	
LEADING EDGE SLAT AND FLAP STRUCTURE	**57-43-02**
IDENTIFICATION 1 - Inboard and Outboard Wing Leading Edge - Krueger Flap Structure	
ALLOWABLE DAMAGE 1 - Wing Leading Edge Krueger Flap Structure	
REPAIR 1 - Inboard Wing Leading Edge Krueger Flap Structure - External Repair Between the Ribs	
REPAIR 2 - Wing Leading Edge Krueger Flap Structure - External Repair at Ribs	
LEADING EDGE SLAT AND FLAP ATTACHMENT FITTINGS	**57-43-90**
IDENTIFICATION 1 - Inboard Wing Leading Edge (Krueger) Flap Fittings	
ALLOWABLE DAMAGE 1 - Inboard Wing Leading Edge (Krueger) Flap Fittings	
REPAIR 1 - Inboard Wing Leading Edge (Krueger) Flap Fittings	
WING FIXED TRAILING EDGE SKIN PANELS	**57-51-01**
IDENTIFICATION 1 - Wing Fixed Trailing Edge Skin Panels	
ALLOWABLE DAMAGE 1 - Wing Fixed Trailing Edge Skin Panels	
REPAIR 1 - Wing Fixed Trailing Edge Skin Panels Made of Composite Materials	
REPAIR 2 - Wing Fixed Trailing Edge Panel Conductive Strip	
WING TRAILING EDGE STRUCTURE	**57-51-02**
IDENTIFICATION 1 - Wing Fixed Trailing Edge Structure	
ALLOWABLE DAMAGE 1 - Wing Fixed Trailing Edge Structure	
REPAIR 1 - Wing Fixed Trailing Edge Support Beam	
REPAIR 2 - Wing Inboard Fixed Trailing Edge Deflection Control Rib Turnbuckle Lug Repair	
REPAIR 3 - Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Damage Repair	
REPAIR 4 - Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair	
MAIN LANDING GEAR BEAM	**57-51-14**
IDENTIFICATION 1 - Main Landing Gear Beam and Stabilizer Structure	
ALLOWABLE DAMAGE 1 - Main Landing Gear Beam Installation	

57-CONTENTS

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

SUBJECT	CHAPTER SECTION SUBJECT
WING TRAILING EDGE FITTINGS	**57-51-90**
IDENTIFICATION 1 - Wing Trailing Edge Fittings	
ALLOWABLE DAMAGE 1 - Wing Trailing Edge Fittings	
TRAILING EDGE FLAP SKIN	**57-53-01**
IDENTIFICATION 1 - Wing Trailing Edge Inboard Main Flap Skin	
IDENTIFICATION 2 - Wing Trailing Edge Inboard Aft Flap Skin	
IDENTIFICATION 3 - Wing Trailing Edge Outboard Main Flap Skin	
IDENTIFICATION 4 - Wing Trailing Edge Outboard Aft Flap Skin	
ALLOWABLE DAMAGE 1 - Wing Inboard Trailing Edge Main Flap	
ALLOWABLE DAMAGE 2 - Wing Inboard Trailing Edge Aft Flap	
ALLOWABLE DAMAGE 3 - Wing Outboard Trailing Edge Main Flap Skin	
ALLOWABLE DAMAGE 4 - Wing Outboard Trailing Edge Aft Flap	
ALLOWABLE DAMAGE 5 - Wing Outboard Trailing Edge Aft Flap Skin	
REPAIR 1 - Wing Trailing Edge - Main Flap Lower Skin	
REPAIR 2 - Wing Trailing Edge Aft Flap Leading Edge Skin Blind Fastener Repair	
REPAIR 3 - Outboard Trailing Edge Main Flap Wedge	
REPAIR 4 - Inboard and Outboard Aft Flap Trailing Edge Wedge	
REPAIR 6 - Wing Trailing Edge Aft Flap Leading Edge Skin Solid Fastener Repair	
REPAIR 7 - Inboard Main Flap Trailing Edge Wedge	
REPAIR 8 - Inboard Main Flap Trailing Edge Wedge Repair of Disbond Less Than 75 Square Inches	
REPAIR 9 - Inboard Main Flap Trailing Edge Wedge Repair of Disbonds That Are 200 Square Inches or Less	
REPAIR 10 - Inboard Main Flap Trailing Edge Wedge Large Disbond Repair	
REPAIR 11 - Outboard Aft Flap Trailing Edge Wedge Repair of Disbonds Less Than 75 Square Inches	
REPAIR 12 - Outboard Aft Flap Trailing Edge Wedge Repair of Disbonds That Are 200 Square Inches or Less	
REPAIR 13 - Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair	
REPAIR 14 - Outboard Aft Flap Trailing Edge Wedge	
WING TRAILING EDGE FLAP STRUCTURE	**57-53-02**
IDENTIFICATION 1 - Wing Trailing Edge Inboard Main Flap Structure	

57-CONTENTS

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SUBJECT	CHAPTER SECTION SUBJECT
IDENTIFICATION 2 - Wing Trailing Edge Inboard Aft Flap Structure	
IDENTIFICATION 3 - Wing Trailing Edge Outboard Main Flap Structure	
IDENTIFICATION 4 - Wing Trailing Edge Outboard Aft Flap Structure	
ALLOWABLE DAMAGE 1 - Wing Inboard Trailing Edge Main Flap Structure	
ALLOWABLE DAMAGE 2 - Wing Inboard Trailing Edge Aft Flap Structure	
ALLOWABLE DAMAGE 3 - Wing Outboard Trailing Edge Main Flap Structure	
ALLOWABLE DAMAGE 4 - Wing Outboard Trailing Edge Aft Flap Structure	
REPAIR 1 - Wing Inboard Trailing Edge Main Flap Structure	
REPAIR 2 - Wing Inboard Trailing Edge Aft Flap Structure	
REPAIR 3 - Wing Outboard Trailing Edge Main Flap Structure	
REPAIR 4 - Wing Outboard Trailing Edge Aft Flap Structure	
REPAIR 5 - Inboard Flap Aft Flap Track Oversized Bearing Bore Repair	
REPAIR 6 - #1 Outboard Flap Aft Flap Track Oversized Bearing Bore Repair	
REPAIR 7 - #2 Outboard Flap Aft Flap Track Oversized Bearing Bore Repair	
REPAIR 8 - #3 Outboard Flap Aft Flap Track Oversized Bearing Bore Repair	
REPAIR 9 - #4 Outboard Flap Aft Flap Track Oversized Bearing Bore Repair	
REPAIR 10 - Wing Trailing Edge Main Flap Track Numbers 1, 2, 3, 6, 7, and 8	
REPAIR 11 - Wing Trailing Edge Main Flap Track Numbers 4 and 5	
REPAIR 14 - Outboard Aft Flap Inboard Closure Rib Crack Repair	
TRAILING EDGE FLAP SUPPORT FAIRING SKIN PANELS	**57-53-70**
IDENTIFICATION 1 - Fairing Skins - Wing Outboard Trailing Edge Flap Supports - Numbers 1 and 8	
IDENTIFICATION 2 - Fairing Skins - Wing Outboard Trailing Edge Flap Supports - Numbers 2 and 7	
IDENTIFICATION 3 - Fairing Skins - Wing Inboard Trailing Edge Flap Supports - Numbers 3 and 6	
ALLOWABLE DAMAGE 1 - Flap Support Fairing Skin, Wing Trailing Edge Flap	
REPAIR 1 - Flap Support Fairing Skin, Wing Trailing Edge Flap	
REPAIR 2 - Flap Support Fairing - Numbers 3 and 6 Inboard Trailing Edge Flap	
REPAIR 3 - Flap Support Forward Fairing Skin, Wing Trailing Edge Flap	
REPAIR 4 - Wing Inboard Trailing Edge Flap Support Aft Fairing Skin	

57-CONTENTS

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SUBJECT	CHAPTER SECTION SUBJECT
TRAILING EDGE FLAP SUPPORT FAIRING STRUCTURE	**57-53-71**
IDENTIFICATION 1 - Flap Support Fairings Structure	
ALLOWABLE DAMAGE 1 - Wing Trailing Edge Flap Track Fairing Support Structure	
REPAIR 1 - Wing Trailing Edge Flap Track Fairing Support Structure	
AILERONS	**57-60-00**
GENERAL - Aileron Diagram	
AILERON SKIN	**57-60-01**
IDENTIFICATION 1 - Aileron Skin	
IDENTIFICATION 2 - Aileron Tab Skin	
ALLOWABLE DAMAGE 1 - Aileron Skin	
ALLOWABLE DAMAGE 2 - Aileron Tab Skin	
REPAIR 1 - Aileron Skin	
REPAIR 2 - Aileron Tab Skin	
AILERON STRUCTURE	**57-60-02**
IDENTIFICATION 1 - Aileron Structure	
IDENTIFICATION 2 - Aileron Tab Structure	
ALLOWABLE DAMAGE 1 - Aileron Structure	
ALLOWABLE DAMAGE 2 - Aileron Tab Structure	
REPAIR 1 - Aileron Structure	
REPAIR 2 - Aileron Tab Structure	
AILERON FAIRINGS	**57-60-70**
ALLOWABLE DAMAGE 1 - Wing Aileron Fairing	
REPAIR 1 - Wing Aileron Fairing	
AILERON TAB FITTINGS	**57-60-90**
IDENTIFICATION 1 - Aileron Fittings	
IDENTIFICATION 2 - Aileron Tab Fittings	
ALLOWABLE DAMAGE 1 - Aileron Hinge Fittings	
ALLOWABLE DAMAGE 2 - Aileron Tab Hinge Fittings	
REPAIR 1 - Aileron Hinge Fittings	
REPAIR 2 - Aileron Tab Hinge Fittings	

57-CONTENTS

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CHAPTER 57
WINGS

SUBJECT	CHAPTER SECTION SUBJECT
SPOILERS	**57-70-00**
GENERAL - Spoiler Diagram	
SPOILER SKIN	**57-70-01**
IDENTIFICATION 1 - Inboard Spoiler Skin	
IDENTIFICATION 2 - Outboard Spoiler Skins	
ALLOWABLE DAMAGE 1 - Inboard and Outboard Spoiler Skins	
REPAIR 1 - Inboard and Outboard Spoiler Skins	
SPOILER STRUCTURE	**57-70-02**
IDENTIFICATION 1 - Inboard Spoiler Structure	
IDENTIFICATION 2 - Outboard Spoiler Structure	
ALLOWABLE DAMAGE 1 - Inboard and Outboard Spoiler Structure	
REPAIR 1 - Spoiler End Rib	
SPOILER ATTACHMENT FITTINGS	**57-70-90**
ALLOWABLE DAMAGE 1 - Wing Spoiler Fittings	
REPAIR 1 - Wing Spoiler Fittings	

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1. General

A. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

B. This chapter contains information on identification, allowable damage, and repairs to the structural components of the wing, leading edge slats, trailing edge flaps, and the ailerons.

C. The fastener and seal requirements specified in the repairs for Lightning Zone 2 areas are very important to keep the safety of the aircraft. Refer to Figure 2/GENERAL for definition of the Lightning Zone areas. For fastener installation requirements for repairs in all lightning zone areas, refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01.

D. The design conditions that have CDCCLs are specified and controlled by Special Federal Aviation Regulation (SFAR) 88. The CDCCLs can be seen in Section 9 of the Maintenance Planning Data (MPD) document. CDCCLs are a means to identify special design configurations that will prevent a fuel tank ignition for the serviceable life of the airplane. CDCCLs are mandatory and cannot be changed or removed without the approval of the applicable FAA office or regulatory agency that approves the airplane model Type Certificate. A very important configuration to prevent fuel tank ignitions is put in some of the fuel system installations. A dangerous condition can start if the ignition prevention configuration is changed. The ignition prevention configuration must not change and the procedures must be followed to make sure that the CDCCL requirements are obeyed. Parts or procedures that are not specified in the applicable CDCCL must only be used with approval from the applicable FAA office or regulatory agency that approved the airplane model Type Certificate.

WARNING: OBEY THE MANUFACTURER'S PROCEDURES WHEN YOU DO MAINTENANCE THAT HAS AN EFFECT ON A CDCCL. IF YOU DO NOT OBEY THE PROCEDURES, IT CAN INCREASE THE RISK OF A SOURCE OF FUEL TANK IGNITION. INJURIES TO PERSONNEL, AND DAMAGE TO EQUIPMENT CAN OCCUR IF THERE IS A FIRE OR EXPLOSION.

E. Procedures in Chapter 51 or Chapter 57 related to a Maintenance Planning Data (MPD) document Airworthiness Limitation (AWL) are identified by the CDCCL text and AWL note specified below.

(1) Entire procedures that are related to a Maintenance Planning Data (MPD) document Airworthiness Limitation (AWL) for SFAR88 are identified by the CDCCL text and AWL note that follows. This note is found in the General paragraph of each applicable procedure. All of the data in these procedures is very important to the safety requirements of the given AWL.

- CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements. All content in this procedure is CDCCL data. It is very important to the safety requirements of the given AWL.
- NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(2) Parts of a procedure that are related to an MPD AWL are identified by the CDCCL text and AWL note that follows. This note is found in the General paragraph of each applicable procedure. If a procedure has this CDCCL text and AWL note in the General paragraph, then there are steps or paragraphs that are very important to the safety requirements of the given AWL.

- CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank

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must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

- NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

The steps or paragraphs that are very important to the safety requirements of the given AWL will have the CDCCL text and AWL note that follows:

- NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).
- NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

These steps or paragraphs must be followed to obey the CDCCL requirements. If you do not follow these steps or paragraphs, then you can increase the risk of fuel tank ignition. All other instructions in the procedure that are not identified with the CDCCL text and AWL note are not applicable to the CDCCL requirement.

2. References

Reference	Title
51-20-05, GENERAL	Repair Sealing
51-60-00, GENERAL	Control Surface Balance Procedures
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS

3. Sealing

A. The outer wing and center wing structure between the front and rear spars is sealed to form integral fuel tanks.

B. Repairs to the wing involving the fuel tank area must be sealed. Refer to 51-20-05, GENERAL for sealing requirements and processes.

4. Control Surface Balancing

A. Refer to 51-60-00, GENERAL for general data on control surface balancing.

5. Fastener Installation Lightning Protection Requirements

A. See Figure 2/GENERAL for the Wingbox Fuel Tank Lightning Zone 2 areas. All wingbox fuel tank areas that are not Lightning Zone 2 are Lightning Zone 3.

B. See SRM section LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01.

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Wing Station Diagram
Figure 1

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Wing Lightning Zone 2 Area Diagram
Figure 2

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ALLOWABLE DAMAGE 1 - WING STRINGERS

1. Applicability

A. This subject gives the allowable damage limits for the center and outer wing stringers shown in Wing Stringer Locations, Figure 101/ALLOWABLE DAMAGE 1.

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OUTER WING STRINGERS

A–A

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Wing Stringer Locations
Figure 101 (Sheet 1 of 2)

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CENTER WING STRINGERS

B-B

G00007 S0006594011_V1

Wing Stringer Locations
Figure 101 (Sheet 2 of 2)

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2. General

A. Refer to Wing Stringers, Figure 102/ALLOWABLE DAMAGE 1 for the wing stringer shapes.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

(1) Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limits that is applicable to each type of stringer.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRINGER	**PARAGRAPH**
Zee Stringers	4.A
Splice Stringers	4.B
Vent Stringers	4.C
Outer Wing Stringer S-1U	4.D
Outer Wing Stinger S-21U	4.D
Outer Wing Stinger S-1L	4.D
Outer Wing Stringer S-14L and S-1L	4.D

C. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE ROTARY PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Rotary peen or shot peen the reworked areas. Refer to SOPM 20-10-03.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(3) Do the step that follows for the interior surfaces.

(a) Apply one layer of BMS 10-20, Type II primer to the conversion coated, reworked area of the stringer. Refer to AMM 28-11-00/701.

(4) Do the steps that follow for the exterior surfaces.

(a) Fill the space caused from the removal of damaged material at a skin splice with BMS 5-95 sealant. Refer to 51-20-05.

(b) Apply BMS 10-20, Type II primer to the reworked area. Refer to AMM 28-11-00/701.

(c) Apply BMS 10-20, Type II primer to the exterior surface of the BMS 5-95 sealant that was used to fill the space on the outer surface. Refer to SOPM 20-44-04.

(d) Apply an applicable finish to the BMS 10-20, Type II primer. Refer to AMM 51-21-00/701.

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ZEE STRINGER

SKIN SPLICE STRINGER

VENT STRINGER

NOTES

- TYPICAL ALUMINUM STRINGER SECTIONS ARE SHOWN.

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Wing Stringers
Figure 102 (Sheet 1 of 3)

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OUTER WING STRINGER S-1U

OUTER WING STRINGER S-21U

F58985 S0006594014_V1

Wing Stringers
Figure 102 (Sheet 2 of 3)

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OUTER WING STRINGER S-1L

OUTER WING STRINGER S-14L

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Wing Stringers
Figure 102 (Sheet 3 of 3)

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-09	COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Zee Stringers

(1) Free Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , C , D , E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

(2) Skin Attachment Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and G .

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(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

B. Skin Splice Stringers

(1) Free Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , C , D , E , and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

(2) Skin Attachment Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

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1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , C , D , E , G and H .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

C. Vent Stringers

(1) Skin Attachment Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , C , D , E , and G .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

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4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a MS21141-()()P blind bolt installed wet with BMS 5-26 or BMS 5-45 sealant.

(3) Horizontal Web

(a) Cracks:

1) Remove the damage at the fuel system cutouts as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details I and J .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details E , G , I , and J .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(4) Vertical Stiffening Flange

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail K .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details K and L .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

D. Outer Wing Stringers S-1U, S-21U, S-1L, and S-14L

(1) Free Flanges

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details C , D , E , and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

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6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

(2) Skin Attachment Flanges

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and G .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Vertical Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details E and F .

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Detail F

2) They are a maximum of 0.25 inch in diameter

3) They are a minimum of 1.0 inch away from the edge of a hole or other damage

4) They are a minimum of 2D (D = the diameter of the damage) away from a material edge

5) They are cold worked with the Class 2 (low interference) procedure as given in 51-40-09

6) They are filled with a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = THE WIDTH OF THE MATERIAL REMOVED
= 10 PERCENT OF THE WIDTH OF THE FLANGE MAXIMUM

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

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Allowable Damage Limits
Figure 103 (Sheet 1 of 8)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

D

A–A

F59202 S0006594017_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 8)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

B–B

F59206 S0006594018_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 8)

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A_1 = NET AREA = TOTAL FLANGE OR WEB AREA - HOLES MADE AT THE TIME OF MANUFACTURE.

A_2 = AREA REMOVED FOR NICKS, SCRATCHES, GOUGES, CORROSION, HOLES, AND PUNCTURES.

THE MAXIMUM LOSS PERMITTED IN THE CROSS-SECTIONAL AREA IS DEFINED AS FOLLOWS:

$$\frac{A_2}{A_1} \leq 0.10$$

THE MAXIMUM LOSS IN THE CROSS-SECTIONAL AREA IS FOR DAMAGE THAT IS IN ALL 1.5 INCH LENGTHS ALONG THE STRINGER.

REMOVAL OF DAMAGED MATERIAL
IN A STRINGER FLANGE OR WEB

F

C-C

F59245 S0006594019_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 8)

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A_1 = NET FLANGE AREA = TOTAL FLANGE AREA - HOLES MADE AT THE TIME OF MANUFACTURE.

A_2 = AREA REMOVED FOR NICKS, SCRATCHES, GOUGES, AND CORROSION.

THE MAXIMUM LOSS PERMITTED IN THE CROSS-SECTIONAL AREA IS DEFINED AS FOLLOWS:

$$\frac{A_2}{A_1} \leq 0.10$$

THE MAXIMUM LOSS IN THE CROSS-SECTIONAL AREA IS FOR DAMAGE THAT IS IN ALL 1.5 INCH LENGTHS ALONG THE STRINGER.

REMOVAL OF DAMAGED MATERIAL IN A STRINGER FLANGE

G

F59269 S0006594020_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 8)

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REMOVAL OF CORROSION AT A SKIN SPLICE

H

SKIN PANEL SPLICE
E–E

F59285 S0006594021_V1

Allowable Damage Limits
Figure 103 (Sheet 6 of 8)

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FUEL SYSTEM CUTOUT IN VENT STRINGER

I

AREA OF THE REMOVED DAMAGE

X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

DAMAGE REMOVAL AT THE EDGE OF A FUEL SYSTEM CUTOUT
OVAL SHAPE HOLE IS SHOWN, DAMAGE REMOVAL ON THE CIRCULAR
FUEL SYSTEM CUTOUT IS PERMITTED

J

F59685 S0006594022_V1

Allowable Damage Limits
Figure 103 (Sheet 7 of 8)

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X = THE WIDTH OF THE MATERIAL REMOVED
= 10 PERCENT OF THE WIDTH OF
THE FLANGE MAXIMUM

REMOVAL OF DAMAGED
MATERIAL ON AN EDGE

K

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

L

F–F

F59485 S0006594023_V1

Allowable Damage Limits
Figure 103 (Sheet 8 of 8)

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IDENTIFICATION 1 - WING CENTER SECTION UPPER SKIN IDENTIFICATION

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F81220 S0006594028_V1

Wing Center Section Upper Skin Panel Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A3000	Wing Center Section Upper Skin Panel Installation
111A3101	Wing Center Section Upper Aft Skin Panel
111A3102	Wing Center Section Upper Forward Skin Panel

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

F81225 S0006594030_V1

Wing Center Section Upper Skin Panels Identification
Figure 2

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Upper Forward Skin Panel	0.375 (9.53)	7055-T7751 plate as given in BMS 7-307	
[2]	Upper Aft Skin Panel	0.375 (9.53)	7055-T7751 plate as given in BMS 7-307	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 2 - WING CENTER SECTION LOWER SKIN IDENTIFICATION

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F81144 S0006594033_V1

Wing Center Section Lower Skin Panel Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A4000	Wing Center Section Lower Skin Panel Installation
111A4101	Wing Center Section Aft Lower Skin Panel
111A4102	Wing Center Section Forward Lower Skin Panel

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

F81137 S0006594035_V1

Wing Center Section Lower Skin Panels Identification
Figure 2

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Forward Lower Skin Panel	0.75 (19.05)	2324-T39 plate as given in BMS 7-254	
[2]	Aft Lower Skin Panel	0.75 (19.05)	2324-T39 plate as given in BMS 7-254	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING CENTER SECTION UPPER SKIN PANELS

1. Applicability

A. Allowable Damage 1 gives the allowable damage limits for the upper skin panels of the wing center section shown in Wing Center Section Upper Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.

B. Allowable Damage 1 is applicable to:

(1) Airplanes that have not had winglets installed as part of the production of the airplane or retrofitted by the Supplemental Type Certificate (STC) ST00830SE.

(2) Airplanes equipped with blended winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets.

(3) Airplane line numbers 778 and on retrofitted by the Supplemental Type Certificate (STC) ST00830SE with blended winglets, except airplane line numbers 779, 787 and 836.

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

C. Allowable Damage 1 is not applicable:

(1) If there is damage to a stringer that is not repaired in the same area.

(2) To airplane line numbers 1 thru 777, 779, 787, and 836 retrofitted with winglets. Refer to the (APB) Supplemental Type Certificate (STC) ST00830SE for the applicable data.

(3) To airplanes retrofitted with Supplemental Type Certificate Split Scimitar winglets or Scimitar winglet provisions. Contact Aviation Partners Boeing (APB) for support regarding this data.

2. General

A. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for the possible sources of the abrasive materials you can use to remove the damage.

(3) Refer to 51-30-05 for the possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the upper skin panels.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM 51-21-00/701.

(4) Apply BMS 5-81, Type II secondary fuel barrier sealant to the exterior surfaces. Refer to AMM 28-11-00.

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F81350 S0006594039_V1

Wing Center Section Upper Skin Panel Location
Figure 101

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PLAN VIEW, UPPER SKINS - WING CENTER SECTION

G17456 S0006594040_V1

Wing Center Section Allowable Damage Zones
Figure 102 (Sheet 1 of 3)

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ZONE 4

ZONE 2

G91330 S0006594041_V1

Wing Center Section Allowable Damage Zones
Figure 102 (Sheet 2 of 3)

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G91335 S0006594042_V1

Wing Center Section Allowable Damage Zones
Figure 102 (Sheet 3 of 3)

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
AMM 28-11-00	Integral Fuel Tanks
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-10-03	SHOT PEENING

4. Allowable Damage Limits

A. Zone 1

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A and B if:

1) The maximum decrease in depth is 0.02 inch (0.51 mm)

2) The maximum decrease in cross-sectional area is:

a) 0.020 $inch^2$ (12.90 mm^2) between adjacent stringers or between a spar and a stringer

b) 0.20 $inch^2$ (129.03 mm^2) between the front spar and the rear spar.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Zone 2

(1) Damage is not permitted.

C. Zone 3

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A and B if:

1) The maximum decrease in depth is 0.03 inch (0.76 mm).

2) The maximum decrease in cross-sectional area is:

a) 0.020 $inch^2$ (12.90 mm^2) between adjacent stringers or between a spar and a stringer

b) 0.20 $inch^2$ (129.03 mm^2) between the front spar and the rear spar.

D. Zone 4

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(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A and B if:

1) The maximum decrease in depth is 0.04 inch (1.02 mm)

2) The maximum decrease in cross-sectional area is:

a) 0.020 $inch^2$ (12.90 mm^2) between adjacent stringers or between a spar and a stringer

b) 0.20 $inch^2$ (129.03 mm^2) between the front spar and the rear spar.

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

A

A–A

G18261 S0006594043_V2

Wing Center Section Upper Skin Allowable Damage
Figure 103 (Sheet 1 of 2)

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CLEANUP OF DAMAGE THAT IS ALONG A LINE PERPENDICULAR TO THE STIFFENERS (REFER TO PARAGRAPH 5 FOR EXAMPLE CALCULATIONS)

B

B–B

G18259 S0006594044_V1

Wing Center Section Upper Skin Allowable Damage
Figure 103 (Sheet 2 of 2)

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5. **Example Calculations**

A. Calculate the damage depths and areas shown in Wing Center Section Upper Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Detail B to find if the damage is permitted.

NOTE: You can make a conservative estimate of the area loss as described below if you can not accurately calculate the area loss. In the example below, the cross sectional area loss is estimated to be 1/2 the width X depth.

(1) Find the allowable damage depth (X) for each damage location shown.

(a) The damage depth at Damage A is 0.015 inch (0.381 mm).

1) This is less than 0.040 inch (1.02 mm) and is permitted.

(b) The damage depth at Damage B is 0.025 inch (0.635 mm).

1) This is less than 0.030 inch (0.762 mm) and is permitted.

(c) The damage depth at Damage C is 0.030 inch (0.762 mm).

1) This is equal to 0.030 inch (0.762 mm) and is permitted.

(2) Calculate the cross-sectional area that has decreased between the stringers (W/2 times X, with W = damage width and X = damage depth). Then find if the decreased area is permitted.

(a) The area that has decreased at Damage A is:

1) 1.6 inches/2 X 0.015 inch = 0.012 inch^2 (40.64 mm/2 X 0.381 mm = 7.74 mm^2)

a) This is less than 0.020 inch^2 (12.90 mm^2) and is permitted.

(b) The area that has decreased at Damage B is:

1) 1.5 inches/2 X 0.025 inch = 0.0188 inch^2 (38.1 mm/2 X 0.635 mm = 12.09 mm^2).

a) This is less than 0.020 inch^2 (12.90 mm^2) and is permitted.

(c) The area that has decreased at Damage C is:

1) 1.2 inches/2 X 0.030 inch = 0.018 inch^2 (30.48 mm/2 X 0.762 mm = 11.61 mm^2).

a) This is less than 0.020 inch^2 (12.90 mm^2) and is permitted.

(3) Calculate the total cross sectional area that has decreased between the front and rear spars. Then find if that decreased area is permitted.

(a) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

1) (0.012 + 0.0188 + 0.018) inch^2 = 0.0488 inch^2 [(7.74 + 12.09 + 11.61) mm^2 = 31.44 mm^2)

a) This is less than 0.200 square inch (128.85 mm^2) and is permitted.

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ALLOWABLE DAMAGE 2 - WING CENTER SECTION LOWER SKIN PANELS

1. Applicability

A. This subject gives the allowable damage limits for the lower skin panels of the wing center section shown in Wing Center Section Lower Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.

B. Allowable Damage 2 is applicable to airplanes that have not had winglets installed

C. Allowable Damage 2 is not applicable to:

(1) Airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets.

(2) Airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

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Wing Center Section Lower Skin Panel Location
Figure 101

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BOTTOM VIEW, LOWER SKINS - WING CENTER SECTION

G18990 S0006594047_V1

Wing Center Section Allowable Damage Zones
Figure 102 (Sheet 1 of 3)

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G19549 S0006594048_V1

Wing Center Section Allowable Damage Zones
Figure 102 (Sheet 2 of 3)

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G19563 S0006594049_V1

Wing Center Section Allowable Damage Zones
Figure 102 (Sheet 3 of 3)

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2. General

A. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasive materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the lower skin panels.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM 51-21-00/701.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-10-03	SHOT PEENING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Zone 1

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Lower Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A and B .

1) Refer to Table 101/ALLOWABLE DAMAGE 2 for the maximum depth, maximum loss of area, and maximum total loss of area.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Zone 2

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(1) Damage is not permitted.

Table 101:

ZONE 1	MAXIMUM DEPTH OF THE DAMAGE AFTER CLEANUP IN INCHES	MAXIMUM LOSS OF CROSS SECTIONAL AREA BETWEEN ADJACENT STRINGERS OR BETWEEN A SPAR AND A STRINGER IN INCHES²	TOTAL MAXIMUM LOSS OF CROSS SECTIONAL AREA BETWEEN THE FRONT SPAR AND THE REAR SPAR IN INCHES²
FORWARD LOWER SKIN (FRONT SPAR TO S-9)	0.025	0.040	0.400
AFT LOWER SKIN (S-9 TO S-5)	0.035	0.060	0.400
AFT LOWER SKIN (S-5 TO REAR SPAR)	0.046	0.065	0.400

NOTE: Allowable damage limits are not applicable if there is damage to a stiffener which is not repaired in the same area.

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REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

A

A–A

G18187 S0006594051_V1

Wing Center Section Lower Skin Allowable Damage
Figure 103 (Sheet 1 of 2)

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CLEANUP OF DAMAGE THAT IS ALONG A LINE PERPENDICULAR TO THE STIFFENERS (REFER TO PARAGRAPH 5 FOR EXAMPLE CALCULATIONS)

B

B–B

G18185 S0006594052_V1

Wing Center Section Lower Skin Allowable Damage
Figure 103 (Sheet 2 of 2)

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5. **Example Calculations**

A. Calculate the damage depths and areas shown in Wing Center Section Lower Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail B . Refer Table 101/ALLOWABLE DAMAGE 2 to find if the damage is permitted.

NOTE: For the examples that follow, the calculations will be shown for damage in Zone 1 between the front spar and Stringer S-9. You can make a conservative estimate of the area as described below if you can not accurately calculate the area loss. In the example below, the cross sectional area loss is estimated to be 1/2 the width X depth.

(1) Find the allowable damage depth (X) for each damage location shown.

(a) The damage depth at Damage A is 0.015 inch.

1) This is less than 0.025 inch and is permitted.

(b) The damage depth at Damage B is 0.020 inch.

1) This is less than 0.025 inch and is permitted.

(c) The damage depth at Damage C is 0.025 inch.

1) This is equal to 0.025 inch and is permitted.

(2) Calculate the cross-sectional area that has decreased between the stringers (W/2 times X, with W = damage width and X = damage depth). Then find if the decreased area is permitted.

(a) The area that has decreased at Damage A is:

1) 1.6 inches/2 X 0.015 inch = 0.012 $inch^2$.

a) This is less than 0.040 $inch^2$ and is permitted.

(b) The area that has decreased at Damage B is:

1) 2.5 inches/2 X 0.020 inch = 0.025 $inch^2$.

a) This is less than 0.040 $inch^2$ and is permitted.

(c) The area that has decreased at Damage C is:

1) 3.0 inches/2 X 0.025 inch = 0.037 $inch^2$.

a) This is less than 0.040 $inch^2$ and is permitted.

(3) Calculate the total cross-sectional area that has decreased between the front and rear spars. Then find if that decreased area is permitted.

(a) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

1) [0.012 + 0.025 + 0.037] $inch^2$ = 0.074 $inch^2$

2) This is less than 0.400 $inch^2$ and is permitted.

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REPAIR 1 - WING CENTER SECTION UPPER SKIN PANELS

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

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Wing Center Section Upper Skin Panel Repairs
Figure 201

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REPAIR 2 - WING CENTER SECTION LOWER SKIN PANELS

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

F81234 S0006594057_V2

Wing Center Section Lower Skin Panel Repairs
Figure 201

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ALLOWABLE DAMAGE 1 - WING CENTER SECTION STIFFENERS

G97215 S0006594075_V1

Wing Center Section Stiffener Allowable Damage
Figure 101

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REPAIR 1 - UPPER ZEE STRINGER ON THE CENTER WING

1. Applicability

A. Repair 1 is applicable for damage to an upper zee stringer on the center wing. A typical zee stringer is shown in Typical Zee Stringer Section, Figure 201/REPAIR 1.

B. Repair 1 is not applicable to upper zee stringers that have:

(1) A free flange that is thicker than 0.140 inch

(2) A skin attachment flange that is thicker than 0.180 inch.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. This repair is a category B repair. This category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 1. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to 51-00-06 for the repair categories and definitions.

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F69383 S0006594078_V1

Typical Zee Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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4. Repair Instructions

WARNING: MAKE SURE THERE IS SUFFICIENT AIR IN THE WORK AREA. MAKE SURE YOU OBEY ALL OTHER PRECAUTIONS. THIS WILL PREVENT INJURY TO PERSONS AND DAMAGE TO THE AIRCRAFT.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 1. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 1.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	2	Use 7075-O material, 0.050 inch in thickness. Heat treat to T6 condition after you form the part
[2]	Angle	2	Use 7075-O material, 0.050 inch in thickness. Heat treat to T6 condition after you form the part
[3]	Strap	1	Use 7075-T6 material, 0.080 inch in thickness
[4]	Strap	1	Use 7075-T6 material, 0.125 inch in thickness
[5]	Filler	1	Make the part from the same type of extrusion as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

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H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [5] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(5) Apply a layer of BMS 5-81, Type II secondary fuel barrier sealant to the upper surface of the upper wing skin.

(a) Apply the BMS 5-81 sealant to the heads of the initial fasteners that you replaced when you installed the repair.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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NOTE: THE UPPER WING SKIN IS NOT SHOWN.

DETAIL I

SECTION A-A

F62128 S0006594080_V2

Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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SECTION B-B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30R8 COLLAR.

REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30M8 COLLAR.

F62129 S0006594081_V2

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

WARNING: MAKE SURE THERE IS SUFFICIENT AIR IN THE WORK AREA. MAKE SURE YOU OBEY ALL OTHER PRECAUTIONS. THIS WILL PREVENT INJURY TO PERSONS AND DAMAGE TO THE AIRCRAFT.

A. Inspect the repair area as follows: (Figure 203/REPAIR 1)

(1) Do a Low Frequency Eddy Current (LFEC) Inspection at 20,000 flight cycles or 40,000 flight hours after the repair installation, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(a) Inspect the stringer vertical web flange from inside the wing tank at the stringer repair area.

1) Inspect at and between all vertical web flange fasteners common to the repair parts and the stringer.

(2) After the initial inspection, do the LFEC Inspection at each 36,000 flight cycles interval.

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1840099 S0000323928_V1

Upper Zee Stringer on the Center Wing Inspection
Figure 203

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REPAIR 2 - UPPER VENT STRINGER ON THE CENTER WING ADJACENT TO A PADUP AREA

1. Applicability

A. Repair 2 is applicable to damage to an upper vent stringer on the center wing that is adjacent to a padup area. A typical upper vent stringer is shown in Typical Vent Stringer Section, Figure 201/ REPAIR 2.

B. Repair 2 is applicable at locations where there is not sufficient clearance between the repair parts and the stringer padup area.

F69409 S0006594083_V1

Typical Vent Stringer Section
Figure 201

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. This repair is a category B repair. This category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 2. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of these repairs. Refer to 51-00-06 for the repair categories and definitions.

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D. Repair 3 is an alternative to Repair 2. Use Repair 3 for damage to an upper vent stringer on the center wing that is not adjacent to a padup area. Use this repair when there is sufficient clearance between the repair parts and the stringer padup area.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

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CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 2. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 2.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	As Necessary	Use 7075-O material with a thickness that is the same as the depth of the recess. Heat treat to T6 condition after you form the part. If there is not a recess, do not use the part
[2]	Angle	2	Use 7075-O material, 0.063 inch in thickness. Heat treat to the T6 condition after you form the part
[3]	Angle	2	Use 7075-O material, 0.080 inch in thickness. Heat treat to the T6 condition after you form the part
[4]	Plate	1	Use 7075-T6 material, 0.050 inch in thickness
[5]	Plate	1	Use 7075-T6 material, 0.063 inch in thickness
[6]	Plate	1	Use 7075-T6 material, 0.080 inch in thickness
[7]	Stringer	1	Make the part from the same type of extrusion as the initial stringer. Use 7055-T77511 material. As an al- ternative, you can machine the part from 7075-T6 material

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

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(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 2.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

O. Apply a layer of BMS 5-81, Type II secondary fuel barrier sealant to the upper surface of the upper wing skin. Refer to AMM 28-11-00/801

(1) Apply the BMS 5-81 sealant to the heads of the initial fasteners that you replaced when you installed the repair.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 3)

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Layout of the Repair Parts
Figure 202 (Sheet 2 of 3)

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B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR.

REPAIR FASTENER LOCATION. INSTALL A MS21141-08()P BLIND BOLT.

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Layout of the Repair Parts
Figure 202 (Sheet 3 of 3)

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5. **Inspection Instructions**

WARNING: MAKE SURE THERE IS SUFFICIENT AIR IN THE WORK AREA. MAKE SURE YOU OBEY ALL OTHER PRECAUTIONS. THIS WILL PREVENT INJURY TO PERSONS AND DAMAGE TO THE AIRCRAFT.

A. Inspect the repair area as follows: (Figure 203/REPAIR 2)

(1) Do a Low Frequency Eddy Current (LFEC) Inspection at 20,000 flight cycles or 40,000 flight hours after the repair installation, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(a) Inspect the stringer vertical web flange from inside the wing tank at the stringer repair area.

1) Inspect at and between all vertical web flange fasteners common to the repair parts and the stringer.

(2) After the initial inspection, do the LFEC Inspection at each 36,000 flight cycles interval.

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DETAIL I

SECTION C–C

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Upper Vent Stringer on the Center Wing Adjacent to a Padup Area Inspection
Figure 203

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REPAIR 3 - UPPER VENT STRINGER ON THE CENTER WING NOT ADJACENT TO A PADUP AREA

1. Applicability

A. Repair 3 is applicable for damage to a vent stringer on the center wing that is not adjacent to padup area. A typical vent stringer is shown in Typical Vent Stringer Section, Figure 201/REPAIR 3.

B. Repair 3 is applicable at locations where there is sufficient clearance between the repair parts and the stringer padup area.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. This repair is a category B repair. This category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 3. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of these repairs. Refer to 51-00-06 for the repair categories and definitions.

D. Repair 2 is an alternative to Repair 3. Use Repair 2 for damage to a upper vent stringer on the center wing that is adjacent to a padup area. Use this repair when there is not sufficient clearance between the repair parts and the stringer padup.

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Typical Vent Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
57-10-01, REPAIR 2	Wing Center Section Lower Skin Panels
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 3. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 3.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

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Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	As Necessary	Use 7075-O material that is the same thickness as the depth of the recess. Heat treat to T6 condition after you form the part. If there is not a recess, do not use the part
[2]	Angle	2	Use 7075-O material, 0.063 inch in thickness. Heat treat to T6 condition after you form the part
[3]	Angle	2	Use 7075-O material, 0.080 inch in thickness. Heat treat to T6 condition after you form the part
[4]	Plate	1	Use 7075-T6 material, 0.050 inch in thickness
[5]	Plate	1	Use 7075-T6 material, 0.063 inch in thickness
[6]	Plate	1	Use 7075-T6 material, 0.080 inch in thickness
[7]	Filler	1	Make the part from the same type of extrusion as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 3.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

O. Apply a layer of BMS 5-81, Type II secondary fuel barrier sealant to the upper surface of the upper wing skin. Refer to AMM 28-11-00/801

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(1) Apply the BMS 5-81 sealant to the heads of the initial fasteners that you replaced when you installed the repair.

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A-A

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC COLLAR.

REPAIR FASTENER LOCATION. INSTALL A MS21141-08()P BLIND BOLT.

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Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

WARNING: MAKE SURE THERE IS SUFFICIENT AIR IN THE WORK AREA. MAKE SURE YOU OBEY ALL OTHER PRECAUTIONS. THIS WILL PREVENT INJURY TO PERSONS AND DAMAGE TO THE AIRCRAFT.

A. Inspect the repair area as follows: (Figure 203/REPAIR 3)

(1) Do a Low Frequency Eddy Current (LFEC) Inspection at 20,000 flight cycles or 40,000 flight hours after the repair installation, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(a) Inspect the stringer vertical web flange from inside the wing tank at the stringer repair area.

1) Inspect at and between all vertical web flange fasteners common to the repair parts and the stringer.

(2) After the initial inspection, do the LFEC Inspection at each 36,000 flight cycles interval.

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Upper Vent Stringer on the Center Wing Not Adjacent to a Padup Area
Figure 203

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REPAIR 4 - LOWER SPLICE STRINGER ON THE CENTER WING ADJACENT TO A PADUP

1. Applicability

A. Repair 4 is applicable for damage to a lower splice stringer on the center wing adjacent to a padup. A typical splice stringer is shown in Typical Splice Stringer Section, Figure 201/REPAIR 4.

B. Repair 4 is applicable at locations where there is not sufficient clearance between the repair parts and the stringer padup area.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. This repair is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 4. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of these repairs. Refer to 51-00-06 for the repair categories and definitions.

D. The Repair 5 is an alternative to Repair 4. Use Repair 5 for damage to a lower splice stringer on the center wing that is not adjacent to a padup area. Use this repair when there is sufficient clearance between the repair parts and the stringer padup.

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Typical Splice Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-09	COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 4. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 4.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

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Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	2	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[2]	Channel	2	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[3]	Angle	2	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[4]	Channel	2	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[5]	Plate	2	Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part
[6]	Plate	2	Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part
[7]	Stringer	1	Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy. As an alternative, you can machine the part from 2024-T42 aluminum alloy material

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 4.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

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(2) Do a leak test to make sure there are no leaks in the fuel tank.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 3)

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A–A

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Layout of the Repair Parts
Figure 202 (Sheet 2 of 3)

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B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

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Layout of the Repair Parts
Figure 202 (Sheet 3 of 3)

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5. **Inspection Instructions**

WARNING: MAKE SURE THERE IS SUFFICIENT AIR IN THE WORK AREA. MAKE SURE YOU OBEY ALL OTHER PRECAUTIONS. THIS WILL PREVENT INJURY TO PERSONS AND DAMAGE TO THE AIRCRAFT.

A. Inspect the repair area as follows: (Figure 203/REPAIR 4)

(1) Do a Low Frequency Eddy Current (LFEC) Inspection at 20,000 flight cycles or 40,000 flight hours after the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(a) Inspect the stringer vertical web flange from inside the wing tank at the stringer repair area.

1) Inspect at and between all vertical web flange fasteners common to the repair parts and the stringer.

(2) Do a Surveillance Visual Inspection at 20,000 flight cycles or 40,000 flight hours after the repair, whichever comes first.

(a) Inspect the wing skin adjacent to the repaired stringer from inside the wing tank.

(3) After the initial inspection, do the LFEC Inspection and the Surveillance Visual Inspection at each 4,000 flight cycles.

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DETAIL I

SECTION A–A

1853647 S0000326994_V1

Lower Splice Stringer on the Center Wing Adjacent to a Padup Area
Figure 203

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REPAIR 5 - LOWER SPLICE STRINGER ON THE CENTER WING NOT ADJACENT TO A PADUP AREA

1. Applicability

A. Repair 5 is applicable for damage on a lower splice stringer on the center wing that is not adjacent to a padup area. A typical splice stringer is shown in Typical Splice Stringer Section, Figure 201/ REPAIR 5.

B. Repair 5 is applicable to locations where there is sufficient clearance between the repair parts and the stringer padup area.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. This repair is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 5. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of these repairs. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for the repair categories and definitions.

D. Repair 4 is an alternative to Repair 5. Use Repair 4 for damage to a lower splice stringer on the center wing that is adjacent to a padup area. Use this repair when there is not sufficient clearance between the repair parts and the stringer padup.

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Typical Splice Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-09	COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 5. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 5.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

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Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	1	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[2]	Channel	1	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[3]	Angle	1	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[4]	Channel	1	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[5]	Plate	1	Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part
[6]	Plate	1	Use 2024-O material, 0.224 inch in thickness. Heat treat to T42 condition after you form the part
[7]	Filler	1	Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy. As an alternative, you can machine the part from 2024-T42 aluminum alloy material

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 5.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

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(2) Do a leak test to make sure there are no leaks in the fuel tank.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

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Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

WARNING: MAKE SURE THERE IS SUFFICIENT AIR IN THE WORK AREA. MAKE SURE YOU OBEY ALL OTHER PRECAUTIONS. THIS WILL PREVENT INJURY TO PERSONS AND DAMAGE TO THE AIRCRAFT.

A. Inspect the repair area as follows: Figure 203/REPAIR 5

(1) Do a Low Frequency Eddy Current (LFEC) Inspection at 20,000 flight cycles or 40,000 flight hours after the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(a) Inspect the stringer vertical web flange from inside the wing tank at the stringer repair area.

1) Inspect at and between all vertical web flange fasteners common to the repair parts and the stringer.

(2) Do a Surveillance Visual Inspection at 20,000 flight cycles or 40,000 flight hours after the repair, whichever comes first.

(a) Inspect the wing skin adjacent to the repaired stringer from inside the wing tank.

(3) After the initial inspection, do the LFEC Inspection and the Surveillance Visual Inspection at each 4,000 flight cycles.

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DETAIL I

SECTION A–A

1854721 S0000327084_V1

Lower Splice Stringer on the Center Wing Not Adjacent to a Padup Area
Figure 203

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REPAIR 6 - LOWER ZEE STRINGER ON THE CENTER WING ADJACENT TO A PADUP AREA

1. Applicability

A. Repair 6 is applicable for damage to a lower zee stringer on the center wing that is adjacent to a padup area. A typical zee stringer is shown in Typical Zee Stringer Section, Figure 201/REPAIR 6.

B. Repair 6 is applicable at locations where there is not sufficient clearance between the repair parts and the stringer padup area.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 6 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 6. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for repair categories and definitions.

D. Repair 7 is an alternative to Repair 6. Use Repair 7 for damage to a lower zee stringer on the center wing that is not adjacent to a padup area. Use this repair when there is sufficient clearance between the repair parts and the stringer padup.

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Typical Zee Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-09	COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 6. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 6.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

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Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	2	Use 2024-O material, 0.050 inch in thickness. Heat treat to T42 condition after you form the part
[2]	Angle	2	Use 2024-O material, 0.050 inch in thickness. Heat treat to T42 condition after you form the part
[3]	Angle	2	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[4]	Angle	2	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[5]	Angle	2	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[6]	Angle	2	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[7]	Stringer	1	Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy. As an alternative, you can machine the part from 2024-T42 aluminum alloy material

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 6.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

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(2) Do a leak test to make sure there are no leaks in the fuel tank.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 3)

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Layout of the Repair Parts
Figure 202 (Sheet 2 of 3)

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B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

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Layout of the Repair Parts
Figure 202 (Sheet 3 of 3)

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5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do the inspection of the stringer vertical web flange from inside the wing tank at the stringer repair area.

(a) Do the inspection at and between all vertical web flange fasteners common to the repair parts and the stringer, as shown in Figure 203/REPAIR 6.

B. After you do the initial inspection, do the inspection again at each 10,000 flight cycles interval. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

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Inspection Instructions
Figure 203

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REPAIR 7 - LOWER ZEE STRINGER ON THE CENTER WING NOT ADJACENT TO A PADUP AREA

1. Applicability

A. Repair 7 is applicable for damage to a lower zee stringer on the center wing that is not adjacent to a padup area. A typical zee stringer is shown in Typical Zee Stringer Section, Figure 201/REPAIR 7.

B. Repair 7 is applicable at locations where there is sufficient clearance between the repair parts and the stringer padup area.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 7 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 7. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of this repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for the repair categories and definitions.

D. Repair 6 is an alternative to Repair 7. Use Repair 6 for damage to a lower zee stringer on the center wing that is adjacent to a padup area. Use this repair when there is not sufficient clearance between the repair parts and the stringer padup.

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Typical Zee Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-09	COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS ON OR NEAR THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE IN A STABLE POSITION. IF YOU DO NOT OBEY, AN UNSATISFACTORY REPAIR CAN BE THE RESULT.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

CAUTION: BE CAREFUL WHEN YOU CUT THE INITIAL STRINGER. IF YOU DO NOT OBEY, YOU CAN CAUSE DAMAGE TO THE STRUCTURE THAT IS ADJACENT TO THE REPAIR AREA.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 7. Refer to 51-10-02.

(1) Remove the full cross-section of the stringer.

(2) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 7.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 through 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

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Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	1	Use 2024-O material, 0.050 inch in thickness. Heat treat to T42 condition after you form the part
[2]	Angle	1	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[3]	Angle	1	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[4]	Angle	1	Use 2024-O material, 0.063 inch in thickness. Heat treat to T42 condition after you form the part
[5]	Angle	1	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[6]	Angle	1	Use 2024-O material, 0.080 inch in thickness. Heat treat to T42 condition after you form the part
[7]	Filler	1	Make the part from the same type of extrusion as the initial stringer. Use 2224-T3511 aluminum alloy

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 7.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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B–B

FASTENER SYMBOLS

┼ INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NX()K()Y HEX DRIVE BOLT AND A BACC30AC8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

✦ REPAIR FASTENER LOCATION. INSTALL A BACB30NX8K HEX DRIVE BOLT AND A BACC30X8 COLLAR. INSTALL THE FASTENER IN A TRANSITION FIT HOLE. COLD WORK THE HOLE WITH THE CLASS 1 (HIGH INTERFERENCE) PROCEDURE. REFER TO SRM 51-40-09.

F65412 S0006594114_V2

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection at 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do the inspection of the stringer vertical web flange from inside the wing tank at the stringer repair area.

(a) Do the inspection at and between all vertical web flange fasteners common to the repair parts and the stringer, as shown in Figure 203/REPAIR 7.

B. After you do the initial inspection, do the inspection again at each 10,000 flight cycles interval. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

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LFEC INSPECTION AREA

1841605 S0000324492_V1

Inspection Instructions
Figure 203

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IDENTIFICATION 1 - WING CENTER SECTION FRONT SPAR

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

A

F73231 S0006594118_V1

Wing Center Section Front Spar Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A2000	Front Spar Installation - Wing Center Section
111A2001	Front Spar Assembly - Wing Center Section

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3. Applicability

A. IDENTIFICATION 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. Refer to Table 2 for List of Materials in Figure 2.

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Web	0.250 (6.35)	2024-T351 plate as given in QQ-A-250/4	
[2]	Upper Chord		BAC1514-3221 7150-T77511 extrusion as given in BMS 7-306	
[3]	Lower Chord		BAC1514-3222 2224-T3511 extrusion as given in BMS 7-255	
[4]	Stabilizer Strap	0.040 (1.02)	7075-T6 clad sheet as given in QQ-A-250/13	
[5]	Stiffener		BAC1506-3045 7150-T77511 extrusion as given in BMS 7-306	
[6]	Stiffener		BAC1517-2794 7150-T77511 extrusion as given in BMS 7-306	
[7]	Stiffener		BAC1517-2793 7150-T77511 extrusion as given in BMS 7-306	
[8]	Bulkhead Chord		BAC1514-3131 2024-T42 extrusion as given in QQ-A-200/3	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

VIEW LOOKING AFT

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F73233 S0006594120_V1

Wing Center Section Front Spar Identification
Figure 2

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IDENTIFICATION 2 - WING CENTER SECTION REAR SPAR FOR AIRPLANE LINE NUMBERS 1 THROUGH 4514

NOTES

- IDENTIFICATION 2 IS APPLICABLE TO AIRPLANE LINE NUMBERS 1 THROUGH 4514.
- REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

A

F77556 S0006594123_V2

Wing Center Section Rear Spar Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A1000	Rear Spar Installation - Wing Center Section
111A1001	Rear Spar Assembly - Wing Center Section

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

REAR VIEW

CENTER SECTION REAR SPAR

A

W16184 S0000165548_V2

Wing Center Section Rear Spar Identification
Figure 2 (Sheet 1 of 2)

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FOR AIRPLANES THAT HAVE BEEN MODIFIED
BY SB 737-57-1253

SECTION A-A

W16185 S0000165547_V1

Wing Center Section Rear Spar Identification
Figure 2 (Sheet 2 of 2)

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Lower Chord		BAC1514-3220 2224-T3511 extrusion as given in BMS 7-255	
[2]	Upper Chord		BAC1514-3331 7150-T77511 extrusion as given in BMS 7-306	
[3]	Web		2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the grain direction.	
[4]	Lower Left and Right Webs		Glass Fiber Reinforced Plastic (GFRP) laminate as given in BMS 8-139, Type 181, 4 plies (Optional: BMS 8-79, Class I, Type 181, 4 plies)	All airplane line numbers except 3464
[4]	Lower Left and Right Webs		Glass Fiber Reinforced Plastic (GFRP) laminate as given in BMS 8-79, Style 1581, Grade A, 4 plies (Optional: BMS 8-79, Class III, Style 7781, Grade A, 4 plies)	Airplane line number 3464 only
[5]	Lower Center Web	0.063 (1.6)	2024-T3 clad sheet as given in QQ-A-250/5	
[6]	Support Fitting		7050-T7451 plate as given in BMS 7-323 and AMS 4050	
[7]	Stiffener		BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306	
[8]	Stiffener		BAC1506-4364 7150-T77511 extrusion as given in BMS 7-306	
[9]	Stiffener		BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306	
[10]	Stiffener		BAC1506-4363 7150-T77511 extrusion as given in BMS 7-306	All airplane line numbers except 3464
[10]	Stiffener		7050-T7451 plate as given in AMS 4050. Refer to the production drawings for the grain direction.	Airplane line number 3464 only
[11]	Stiffener		BAC1517-2776 7150-T77511 extrusion as given in BMS 7-306.	
[12]	Stiffener LBL 6.15		BAC1517-2776 7150-T77511 extrusion as given in BMS 7-306 or BAC1517-2780 altered to match cross section of BAC1517–2276. As an optional BAC1518–1152 7055–T77511 extrusion as given in BMS 7-308	Airplane line numbers 1 thru 439
[12]	Stiffener LBL 6.15		2024-T351 plate as given in QQ-A-250/4 or 2024-T351 extruded bar as given in QQ-A-200/3, machined to BAC1517-2776 shape.	Airplane line numbers 440 and on
[13]	Stiffener		BAC1503-4223 7075-T73511 extrusion as given in QQ-A-200/11	
[14]	Stiffener		BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306	
[15]	Stiffener		BAC1514-1160 7150-T77511 extrusion, altered, as given in BMS 7-306	
[16]	Stiffener		BAC1517-2779 7150-T77511 extrusion as given in BMS 7-306	
[17]	Stiffener		BAC1517-4362 7150-T77511 extrusion as given in BMS 7-306	All airplane line numbers except 3464

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[17]	Stiffener		7050-T7451 plate as given in AMS 4050. Refer to the production drawings for the grain direction.	Airplane line number 3464 only
[18]	Stiffener		BAC1517-2778 7150-T77511 extrusion as given in BMS 7-306	
[19]	Stiffener		BAC1506-436 7150-T77511 extrusion as given in BMS 7-306	
[20]	Stiffener		BAC1506-4364 7150-T77511 extrusion as given in BMS 7-306	
[21]	Stiffener		BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306	
[22]	Stiffener		7050-T7451 plate as given in AMS 4050	
[23]	Stiffener		BAC1517-2778 7150-T77511 extrusion as given in BMS 7-306	All airplane line numbers except 3464
[23]	Stiffener		7050-T7451 plate as given in AMS 4050. Refer to the production drawings for the grain direction.	Airplane line number 3464 only
[24]	Stiffener		7050-T7451 plate as given in AMS 4050	
[25]	Vertical Pressure Web	0.150 (3.81)	2024-T3 bare sheet as given in QQ-A-250/4	
[26]	Stiffener, RBL 6.14		BAC1517-2778 7150-T77511 extrusion as given in BMS 7-306 or BAC1517-2779 altered to match cross section of BAC1517-2778. As an optional BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306 or BAC1518-1152 7055-T77511 extrusion as given in BMS 7-308.	Airplane line numbers 1 thru 439
[26]	Stiffener, RBL 6.14		2024-T351 plate as given in QQ-A-250/4 or 2024-T3511 extruded bar as given in QQ-A-200/3, machined to BAC1517-2776 shape.	Airplane line numbers440 and on
[27]	Angle (2)		7050-T7451 plate as given in BMS 7-323 Type III, machined to make an angle.	Airplane line numbers 1 thru 321 that have been modified by SB 57–1253

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 3 - WING CENTER SECTION REAR SPAR FOR AIRPLANE LINE NUMBERS 4515 AND ON

NOTES

- IDENTIFICATION 3 IS APPLICABLE TO AIRPLANE LINE NUMBERS 4515 AND ON.
- REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

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Wing Center Section Rear Spar Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A1000	Rear Spar Installation - Wing Center Section
111A1001	Rear Spar Assembly - Wing Center Section

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

REAR VIEW

CENTER SECTION REAR SPAR

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2169492 S0000479293_V1

Wing Center Section Rear Spar Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Lower Chord		BAC1514-3220 2224-T3511 extrusion as given in BMS 7-255	
[2]	Upper Chord		BAC1514-3331 7150-T77511 extrusion as given in BMS 7-306	
[3]	Web		2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the grain direction.	
[4]	Lower Left and Right Webs	0.040 in. (1.02 mm)	2024-T3 clad sheet as given in QQ-A-250/5	
[5]	Lower Center Web	0.063 in. (1.60 mm)	2024-T3 clad sheet as given in QQ-A-250/5	
[6]	Support Fitting		7050-T7451 plate as given in BMS 7-323 and AMS 4050	
[7]	Stiffener		BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306	
[8]	Stiffener		BAC1506-4364 7150-T77511 extrusion as given in BMS 7-306	
[9]	Stiffener		BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306	
[10]	Stiffener		BAC1506-4363 7150-T77511 extrusion as given in BMS 7-306	
[11]	Stiffener		BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306	
[12]	Stiffener		2024-T351 plate as given in QQ-A-250/4	
[13]	Stiffener		BAC1503-4223 7075-T73511 extrusion as given in QQ-A-200/11	
[14]	Stiffener		BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306	
[15]	Stiffener		BAC1514-1160 7150-T77511 extrusion, altered, as given in BMS 7-306	
[16]	Stiffener		BAC1517-2779 7150-T77511 extrusion as given in BMS 7-306	
[17]	Stiffener		BAC1506-4363 7150-T77511 extrusion as given in BMS 7-306	
[18]	Stiffener		BAC1517-2779 7150-T77511 extrusion as given in BMS 7-306	
[19]	Stiffener		BAC1506-4361 7150-T77511 extrusion as given in BMS 7-306	
[20]	Stiffener		BAC1506-4364 7150-T77511 extrusion as given in BMS 7-306	
[21]	Stiffener		BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306	
[22]	Stiffener		7050-T7451 plate as given in AMS 4050	
[23]	Stiffener		BAC1517-2780 7150-T77511 extrusion as given in BMS 7-306	
[24]	Stiffener		7050-T7451 plate as given in BMS 7-323	
[25]	Vertical Pressure Web	0.150 in. (3.81 mm)	2024-T3 bare sheet as given in QQ-A-250/4	
[26]	Stiffener		2024-T3511 extruded bar as given in QQ-A-200/3, machined to BAC1517-2776 shape.	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING CENTER SECTION FRONT SPAR

1. Applicability

A. ALLOWABLE DAMAGE 1 gives the allowable damage limits for the wing center section front spar as shown in Wing Center Section Front Spar Location, Figure 101/ALLOWABLE DAMAGE 1.

B. ALLOWABLE DAMAGE 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

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NOTE: ALL PARTS ARE MADE OF ALUMINUM.

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Wing Center Section Front Spar Location
Figure 101

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the spar chords or the web.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Chords (Except the Bulkhead Chord)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, C, D and E.

(3) Dents are not permitted.

(4) Holes and punctures are not permitted.

B. Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B and F.

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(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, E and F.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail G.

(4) Holes and Punctures:

(a) Damage is permitted if:

1) The damage is a maximum of 0.25 inch diameter after the damage is removed.

2) The damage is in an area away from a padup.

3) The damage is a minimum of 1.00 inch away from a hole, an edge, or other damage.

4) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet.

a) Install the rivet wet with BMS 5-26 sealant.

C. Stiffeners

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, C, D and E.

(3) Dents are not permitted.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) The damage is a maximum of 0.25 inch diameter after the damage is removed.

2) The damage is a minimum of 0.30 inch from the edge of the part.

3) The damage is a minimum of 0.75 inch from a hole or other damage.

4) The damage is to a maximum of 4 holes for each stiffener.

5) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet.

a) Install the rivet wet with BMS 5-26 sealant.

D. Strap

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B and H.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove edge damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, E and H.

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail G.

(4) Holes and punctures are not permitted.

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E. Bulkhead Chord

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B and H.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, E and H.

(3) Dents are not permitted.

(4) Holes and punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

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Allowable Damage Limits
Figure 102 (Sheet 1 of 5)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON A
SURFACE AT AN EDGE OF AN EXTRUDED PART

C

A–A

F79557 S0006594131_V2

Allowable Damage Limits
Figure 102 (Sheet 2 of 5)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF AN EXTRUDED PART

D

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

B–B

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Allowable Damage Limits
Figure 102 (Sheet 3 of 5)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A WEB

F

F72326 S0006594133_V2

Allowable Damage Limits
Figure 102 (Sheet 4 of 5)

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DENT THAT IS PERMITTED

G

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE OF A FORMED PART

H

F72328 S0006594134_V2

Allowable Damage Limits
Figure 102 (Sheet 5 of 5)

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ALLOWABLE DAMAGE 2 - WING CENTER SECTION REAR SPAR

1. Applicability

A. ALLOWABLE DAMAGE 2 is applicable to all airplanes which include those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE blended winglet (not the split scimitar configuration), except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. This subject gives the allowable damage limits for the wing center section rear spar as shown in Figure 101/ALLOWABLE DAMAGE 2.

C. Allowable Damage 2 is applicable to:

(1) Single cracks in the Vapor Barrier no more than 16 inches long.

(2) More than one crack in the Vapor Barrier on the left or right of BL 0. The sum of the cracks on either side of BL 0 must not be more than 20 inches long .

(3) Airplanes that have initial GFRP (Glass Fabric Reinforced Plastic) rear spar vapor barrier webs as shown in Figure 101/ALLOWABLE DAMAGE 2.

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NOTE: REFER TO PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

A

F73386 S0006594136_V2

Wing Center Section Rear Spar Allowable Damage
Figure 101 (Sheet 1 of 2)

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NOTES

- REFER TO PARAGRAPH 2 OR 4, AS APPLICABLE FOR THE ALLOWABLE DAMAGE LIMITS.

1 MORE THAN ONE CRACK IS PERMITTED ON THE LEFT OR RIGHT OF BL 0.0. THE SUM OF THE LENGTHS OF THE CRACKS ON EACH SIDE MUST NOT BE MORE THAN 20 INCHES. NO UNREPAIRED DAMAGE IS PERMITTED TO THE ADJACENT ALUMINUM STRUCTURE. ALL DAMAGE MUST BE REPAIRED AS GIVEN IN THE APPLICABLE SRM INSTRUCTIONS.

2 CRACKS THAT TERMINATE AT THE INITIAL FASTENER HOLES DO NOT NEED TO BE STOP DRILLED.

VAPOR BARRIER

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

AFT VIEW FACING FORWARD

B

436084 S0000141442_V3

Wing Center Section Rear Spar Allowable Damage
Figure 101 (Sheet 2 of 2)

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Do the steps that follow for the parts made of aluminum.

(1) Remove the damage as necessary.

(a) Refer to 51-10-02 for the inspection and removal of damage.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(a) Flap peen or shot peen the reworked areas of the spar chords or the upper web.

1) Refer to 51-20-06 for shot peen intensity and shot number.

2) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

(b) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(c) Apply a layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

C. Do the steps that follow for the lower left or right webs made of Glass Fabric Reinforced Plastic (GFRP).

(1) Refer to 51-30-05 for the tools.

(2) Do the steps that follow to clean the surface.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(a) Remove loose dirt, unwanted grease, or oil. Be careful not to spread the dirt over a larger area than necessary. Refer to SOPM 20-30-99 for the applicable solvents.

(b) Wet the surface with solvent and wipe or scrub with a wiper, sponge, cheesecloth or brush. Repeat this step as necessary to remove the dirt.

(c) Rinse the surface with fresh solvent and wipe with a clean sponge or cheesecloth.

(d) Wipe off the excess solvent and let the surface dry.

(e) Abrade the surface.

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WARNING: DO NOT BREATHE THE DUST THAT IS MADE WHEN YOU ABRADE OR CUT CURED COMPOSITE MATERIALS, OR LET THE DUST GET IN YOUR EYES OR ON YOUR SKIN. USE EYE PROTECTION AND PROTECTIVE CLOTHING. MAKE SURE THAT YOU USE RESPIRATORY EQUIPMENT WHEN YOU WORK IN CLOSED LOCATIONS. USE A VACUUM TABLE OR PORTABLE VACUUM TO REMOVE THE DUST WHILE YOU ABRADE OR CUT. IF YOU DO NOT OBEY, INJURY TO PERSONS CAN OCCUR.

CAUTION: DO NOT DAMAGE THE FIBERS WHEN YOU MAKE THE SURFACES ROUGH WITH ABRASIVE PAPER. IF YOU DO NOT OBEY, YOU WILL CAUSE A DECREASE IN THE STRENGTH OF THE PART.

1) Protect the undamaged area of the web from the abrasive material. Mask around the damage area as necessary. Use material that is resistant to the abrasive material, such as vinyl tape, or Teflon tape.

2) Abrade (sand) the surface with one of the steps that follow to remove the coating.

NOTE: If you use an orbital driven sander, set the sander below 20,000 revolutions per minute (RPM).

a) Sand the surface with 80 grit or finer abrasive paper for extensive sanding.

b) Sand the surface with 150 grit or finer abrasive paper for final sanding.

c) Finish the surface with 1/0 or finer garnet paper or cloth or fine/ultrafine abrasive fabric pad (Scotch-Brite).

(f) Repeat Paragraph 2.C.(2)(a) through 2.C.(2)(d) to re-clean the repair area.

(3) Do one of the steps that follow to seal the damage that is not more than one ply deep and that agrees with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 2.

(a) Make an interim seal on the side common to the damage.

1) Refer to Paragraph 2.C.(2)/ALLOWABLE DAMAGE 2 for the clean up procedure.

NOTE: Remove contamination from an area 1.50 inches (38 mm) minimum in all directions of the crack.

2) Apply aluminum foil tape (speed tape).

3) Keep a record of the location.

4) Make sure the tape is in satisfactory condition at each 600 flight cycles interval, or more frequently.

(b) Make a permanent seal.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to SOPM 20-44-04.

3) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Do either the steps in Paragraph 2.C.(4)(a)/ALLOWABLE DAMAGE 2 or in Paragraph 2.C.(4)(b)/ALLOWABLE DAMAGE 2 to seal the cracks that are not more than 16.0 in. (406.4 mm) long. These cracks must be in Zone 1 as given in Figure 101 (Sheet 2). Repairs to these cracks are Category B repairs. Refer to 51-00-06, GENERAL for the definitions of the different repair categories.

(a) Make an interim seal on the wheel well side of the web only.

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1) Perform a 0.25 inch diameter stop drill at the crack ends. Refer to 51-10-02, GENERAL for the stop drill procedures.

2) Use vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04, REPAIR GENERAL.

3) Refer to Paragraph 2.C.(2)/ALLOWABLE DAMAGE 2 for the clean up procedure.

NOTE: Remove contamination from 1.50 inches (38 mm) on either side of the crack and 3.00 inches (76.2 mm) beyond the end of the crack.

4) Apply aluminum foil tape (speed tape) 3.00 inches (76.2 mm) beyond the end of the crack.

5) Keep a record of the crack length and the location.

6) Do a visual inspection of the repair area at 1200 flight cycle intervals or more frequently.

a) Make sure that crack has not grown past the interim seal.

b) If the crack has grown past the interim seal, refer to Paragraph 4./ ALLOWABLE DAMAGE 2 for allowable damage limits.

c) If the total crack length meets the allowable damage limits in Paragraph 4./ ALLOWABLE DAMAGE 2, refer to Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2 for repair instructions.

d) If the total crack length is more than the allowable limits given in Paragraph 4./ALLOWABLE DAMAGE 2, do a permanent repair as given in 51-70-04, REPAIR 10 to remove the crack. As an alternative, you can repair part of the crack so that the crack length is less than or equal to the maximum allowable damage limit. Apply the interim seal over the crack that is left as given in Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2.

(b) Make an interim seal on the wheel well side of the web only.

1) Perform a 0.25 inch diameter stop drill at the crack ends. Refer to 51-10-02, GENERAL for the stop drill procedures.

2) Use vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04, REPAIR GENERAL.

3) Refer to Paragraph 2.C.(2)/ALLOWABLE DAMAGE 2 for the clean up procedure.

NOTE: Remove contamination from 1.50 inches (38 mm) one either side of the crack and 3.00 inches (76.2 mm) beyond the end of the crack.

4) Apply BMS 5-45 or BMS 5-95 sealant (class and type optional) plus one ply of BMS 9-3, style 1581 fiberglass cloth on the cracked areas as follows.

a) Cut a piece of fiberglass cloth to fit 1.00/1.50 inches (25/38 mm).

b) Layout the fiberglass cloth on a clean work surface.

c) Impregnate the cloth with BMS 5-45 or BMS 5-95 sealant.

d) Apply sealant .03/.10 inch (0.76/2.5 mm) thick X 1.00/1.50 inches (25/38 mm) on either side of the crack and 3.00 inches (76.2 mm) beyond the end of the crack.

e) Apply and press the impregnated fiberglass cloth lightly on vapor barrier. Make sure you have sealant around the edge of the fiberglass cloth.

f) Apply an additional .03/.10 inch (0.76/2.5 mm) thick layer of sealant over the cloth. Make sure the sealant completely covers the fiberglass cloth. Refer to 51-20-05, GENERAL.

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5) Finish with BMS 10-79, Type 3 or BMS 10-103, Type 1 primer and BMS 10-60 enamel.

6) Keep a record of the crack length and the repair location.

7) Do a visual inspection of the repair area at 1200 flight cycle intervals or more frequently.

   a) Make sure that crack has not grown past the interim seal.

   b) If the crack has grown past the interim seal, refer to Paragraph 4./ALLOWABLE DAMAGE 2 for allowable damage limits.

   c) If the total crack length meets the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 2, refer to Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2 for repair instructions.

   d) If the total crack length is more than the allowable limits given in Paragraph 4./ALLOWABLE DAMAGE 2, do a permanent repair as given in 51-70-04, REPAIR 10 to remove the crack. As an alternative, you can repair part of the crack so that the crack length is less than or equal to the maximum allowable damage limit. Apply the interim seal over the crack that is left as given in Paragraph 2.C.(4)/ALLOWABLE DAMAGE 2.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-05, GENERAL	Repair Sealing
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-04, REPAIR 10	Repair of Damage That is More Than 0.50 Inch in Solid Laminates
51-70-04, REPAIR GENERAL	Repair Procedures for Wet Layup Materials
51-70-08	RESIN SWEEP-FAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Chords (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , and F .

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(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Upper Web (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , and G .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , E , F , and G.

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Detail H .

(4) Holes and Punctures are not permitted.

C. Lower Center Web (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , and G.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , E , F and G.

(3) Dents:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Detail H.

(4) Holes and Punctures:

(a) Damage is permitted if:

1) The damage is a maximum of 0.25 inch diameter after the damage is removed.

2) The damage is a minimum of 0.5 inch from all padups.

3) The damage is a minimum of 1.00 inch from any hole, edge of part, or other damage.

4) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet.

a) Install the rivet wet with BMS 5-26 sealant.

D. Lower Left Web and Lower Right Web (GFRP)

NOTE: Refer to Definitions of the Damage Size, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Cracks:

(a) The damage is permitted if they are:

1) Less than or equal to the dimensions given in Paragraph 2.C. and as shown in Figure 101, Sheet 2.

2) Sealed as given in Paragraph 2.C.

(2) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

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(a) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

1) A maximum of one ply in depth.

NOTE: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 5.0 inches in length.

(c) A maximum of 0.25 inch in width.

(d) A minimum of 0.50 inch away from the edge of a fastener hole.

(e) A minimum of 0.50 inch away from the edge of other damage. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) The damage is filled with a 2117-T3 or 2117-T4 aluminum protruding head rivet installed wet with BMS 5-26 sealant.

(5) Delaminations are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies.

2) Are sealed as given in Paragraph 2.

E. Stiffeners (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details , A , B , D and I.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F and I.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

F. Support Fitting (Aluminum)

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(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , and B.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the edge damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , E , and J .

(b) Bushing damage is not permitted.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

F79574 S0006594137_V1

Allowable Damage Limits
Figure 102 (Sheet 1 of 6)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON A
SURFACE AT AN EDGE OF AN EXTRUDED PART

C

F79577 S0006594138_V3

Allowable Damage Limits
Figure 102 (Sheet 2 of 6)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF AN EXTRUDED PART

D

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

F79578 S0006594139_V2

Allowable Damage Limits
Figure 102 (Sheet 3 of 6)

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REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

F

C–C

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

G

F79579 S0006594140_V1

Allowable Damage Limits
Figure 102 (Sheet 4 of 6)

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DENT THAT IS PERMITTED

H

REMOVAL OF DAMAGED MATERIAL ON AN EDGE OF A FORMED PART

I

F79580 S0006594141_V3

Allowable Damage Limits
Figure 102 (Sheet 5 of 6)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

J

F79581 S0006594142_V1

Allowable Damage Limits
Figure 102 (Sheet 6 of 6)

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SIZE DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

SIZE DEFINITIONS FOR
DENT DAMAGE

B

SIZE DEFINITIONS FOR
EDGE DAMAGE

C

F80406 S0006594143_V1

Definitions of the Damage Size
Figure 103

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REPAIR 1 - WING CENTER SECTION FRONT SPAR

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

A

F73381 S0006594146_V2

Wing Center Section Front Spar Repair
Figure 201

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REPAIR 2 - WING CENTER SECTION REAR SPAR

1. Applicability

A. This repair is applicable to damage to the wing center section rear spar fiberglass vapor barrier as shown in Figure 201.

B. The 3 Bay Doubler Repair as given in Paragraph 4 and the Full Cross Brace Repair as given in Paragraph 5 are applicable to airplanes with line numbers 1 through 1639 only. For airplanes with line numbers 1640 and on, contact The Boeing Company for alternative repair instructions, or you can use the wet layup repair as given in 51-70-04, REPAIR 10.

C. The Patch Repair as given in Paragraph 6 is applicable to airplanes with line numbers 1 through 1994 and 2319 through 4514 only. The Patch Repair is also applicable to airplanes with retrofitted flat panel webs instead of stiffening bead webs. For all other airplanes, contact The Boeing Company for alternative repair instructions, or you can use the wet layup repair as given in 51-70-04, REPAIR 10.

NOTE: The alternative wet layup repair as given in 51-70-04, REPAIR 10 is applicable to all line numbers.

2. General

A. The 3 Bay Doubler Repair as given in Paragraph 4 and the Full Cross Brace Repair as given in Paragraph 5 are permanent repairs. Refer to 51-00-06, GENERAL to find the different categories of repairs.

B. The Full Cross Brace Repair as given in Paragraph 5 addresses repair action to be taken for cracks in the rear spar vapor barrier web on an individual bay by bay basis. Cracks inside any individual bay which do not extend closer to 2.0 inches to adjacent structure require repair only to the affected bay. Cracks in the individual bay closer than 2.0 inches to an adjacent bay also require repair to the adjacent bay.

C. The Patch Repair as given in Paragraph 6 is an interim repair. This interim repair is limited to cracks up to 18 inches in length and up to 1/16 inch in width. If damage is more than 18 inches in length and/or 1/16 inch in width, the repair is not applicable. When you accomplish this interim repair, do a detailed visual inspection every 3,000 flight cycles.

NOTE: If additional damage is found during detailed visual inspection and the combined damage is more than the limits of this interim repair, then you must replace this interim repair with a permanent repair as given in 51-70-04, REPAIR 10.

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F76006 S0006594148_V2

Wing Center Section Rear Spar Repair
Figure 201

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-70-04, REPAIR 10	Repair of Damage That is More Than 0.50 Inch in Solid Laminates
AMM 28-11-00	FUEL TANKS
AMM 51-31-00	SEALS AND SEALING
BAC 5000	General Sealing
SOPM 20-30-03	General Cleaning Procedures

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(Continued)

Reference	Title
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions For The 3 Bay Doubler Repair As Given In Figure 203

A. Get access to the repair area.

B. If there is a crack, then drill a 0.25 inch (6.35 mm) diameter stop hole at the crack ends as given in 51-10-02, GENERAL.

NOTE: A stop hole is not necessary if the crack ends at a fastener hole.

C. Fabricate the repair parts as shown in Figure 203 and as given in Table 201.

(1) Make the surface finish to all cut surfaces 125 microinches Ra, or smoother.

(2) Make the internal corner radius 0.50 inch (12.70 mm) minimum. Make the external corner radius 0.25 inch (6.35 mm) minimum.

D. Assemble the repair parts and drill the fastener holes as shown in Figure 203.

Table 201:

REPAIR MATERIAL FOR FIGURE 203			
PART NUMBER	PART	QUANTITY	MATERIAL
[1]	Repair Doubler	1	Use 2024-T3 clad sheet that is 0.040 inch (1.02 mm) thick
[2]	Filler	3	Use 2024-T3 clad sheet with thickness as necessary to limit any gaps to a maximum of 0.005 inch (0.13 mm).

E. Do a penetrant inspection of the joggle locations to make sure that there are no cracks.

F. Remove the repair parts.

G. Remove all nicks, scratches, burrs, and sharp edges from the repair parts.

H. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the vapor barrier. Refer to 51-20-01, GENERAL.

I. Apply one layer of BMS 10-11, Type I primer to the repair parts and to the bare surfaces of the vapor barrier. Refer to SOPM 20-41-02.

J. Install the repair parts wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Apply a fillet seal of BMS 5-95 sealant to the edges of the part [1] repair doubler.

K. Install the fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

NOTE: It is permitted to use 1/64 inch diameter or 1/32 inch diameter oversize fasteners, if necessary. Make sure there is a minimum of 1.5D (D = fastener diameter) edge margin.

(1) Install the fasteners with the fastener head on the opposite side of the part [1] repair doubler.

5. Repair Instructions For The Full Cross Brace Repair As Given In Figure 204

A. Get access to the repair area.

B. If there is a crack, then drill a 0.25 inch (6.35 mm) diameter stop hole at the crack ends as given in 51-10-02, GENERAL.

NOTE: A stop hole is not necessary if the crack ends at a fastener hole.

C. Fabricate the repair parts as shown in Figure 204 and as given in Tables 202 through 214.

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(1) Make the surface finish to all cut surfaces 125 microinches Ra, or smoother.

(2) Follow all radius callouts as described in Figure 204.

D. Assemble the repair parts and drill the fastener holes as shown in Figure 204.

Table 202:

REPAIR MATERIAL FOR BAY 1L AND BAY 1R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-1	-2	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 34.4 inches
[2]	-3	-4	Crossbrace	1	
[3]	-5	-6	Crossbrace	1	
[4]	-7	-8	Joint Angle	1	Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches
[5]	-9	-10	Joint Angle	1	
[6]	-11	-12	Strap	1	Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 34.4 inches
[7]	-13	-14	Strap	1	
[8]	-15	-16	Strap	1	
[9]	-17	-18	Strap	1	Use 7075-T6 bare sheet that is 0.036 inch thick X 4.0 inches X 34.4 inches
[10]	-19	-20	Strap	1	
[11]	-21	-22	Strap	1	
[12]	-23	-24	Strap	1	Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 34.4 inches
[13]	-25	-26	Strap	1	
[14]	-27	-28	Strap	1	

*[1] These part numbers are for reference only.

Table 203:

REPAIR MATERIAL FOR BAY 2L AND BAY 2R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-29	-30	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 20.1 inches
[2]	-31	-32			
[3]	-33	-34			
[4]	-35	-36	Joint Angle	1	Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches
[5]	-39	-40			
[6]	-41	-42	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch X 4.0 inches X 20.1 inches
[7]	-43	-44			
[8]	-45	-46			

*[1] These part numbers are for reference only.

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Table 204:

REPAIR MATERIAL FOR BAY 3L AND BAY 3R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-47	-48	Crossbrace	1	Use 7075-T6 in BAC1503-100146 extruded angle X 8.5 inches
[2]	-49	-50	Crossbrace	1	Use 7075-T6 in AND10134-1403 extruded angle X 18.5 inches
[3]	-51	-52	Crossbrace	1	Use 7075-T6 in BAC1503-100146 extruded angle X 10.5 inches
[4]	-53	-54	Joint Angle	1	Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches
[5]	-55	-56	Joint Angle	1	
[6]	-57	-58	Strap	1	Use 7075-T6 bare sheet that is 0.063 inch X 4.0 inches X 33.2 inches
[7]	-59	-60	Strap	1	
[8]	-61	-62	Strap	1	
[9]	-63	-64	Strap	1	Use 7075-T6 bare sheet that is 0.036 inch thick X 4.0 inches X 33.2 inches
[10]	-65	-66	Strap	1	
[11]	-67	-68	Strap	1	
[12]	-69	-70	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 33.2 inches.
[13]	-71	-72	Strap	1	
[14]	-73	-74	Strap	1	

*[1] These part numbers are for reference only.

Table 205:

REPAIR MATERIAL FOR BAY 4L AND BAY 4R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-75	-76	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 29.0 inches
[2]	-77	-78	Crossbrace	1	
[3]	-81	-82	Crossbrace	1	
[4]	-83	-84	Strap	1	Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 29.0 inches
[5]	-85	-86	Strap	1	
[6]	-87	-88	Strap	1	

*[1] These part numbers are for reference only.

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Table 206:

REPAIR MATERIAL FOR BAY 5L					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-89	-	Crossbrace	1	Use 7075-T6 in AND10134-1403 extruded angle X 5.0 inches
[2]	-91	-	Crossbrace	1	Use 7075-T6 in AND10134-1403 extruded angle X 11.5 inches
[3]	-93	-	Crossbrace	1	Use 7075-T6 in BAC1503-100146 extruded angle X 5.0 inches
[4]	-95	-	Joint Angle	1	Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches
[5]	-97	-	Joint Angle	1	
[6]	-99	-	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches width X 21.6 inches length
[7]	-101	-	Strap	1	
[8]	-103	-	Strap	1	
[9]	-105	-	Strap	1	
[10]	-107	-	Strap	1	
[11]	-109	-	Strap	1	

*[1] These part numbers are for reference only.

Table 207:

REPAIR MATERIAL FOR BAY 5R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-	-181	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 16.2 inches
[2]	-	-183	Crossbrace	1	
[3]	-	-185	Strap	1	Use7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 16.2 inches
[4]	-	-187	Strap	1	
[5]	-	-189	Strap	1	
[6]	-	-191	Strap	1	Use 7075-T6 bare sheet that is 0.036 inch thick X 4.0 inches X 16.2 inches

*[1] These part numbers are for reference only.

Table 208:

REPAIR MATERIAL FOR BAY 6L AND BAY 6R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-111	-112	Crossbrace	1	Use 7075-T6 in AND10134-2001 extruded angle X 24.0 inches
[2]	-113	-114	Crossbrace	1	
[3]	-115	-116	Crossbrace	1	

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Table 208: (Continued)

REPAIR MATERIAL FOR BAY 6L AND BAY 6R					
ITEM	PART NUMBER 111A1152 *[1] LH	RH	PART	QUANTITY	MATERIAL
[4]	-117	-118	Joint Angle	1	Use 7075-T6 AND10134-2001 extruded angle X 6.0 inches
[5]	-119	-120	Joint Angle	1	
[6]	-121	-122	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 24.0 inches
[7]	-123	-124	Strap	1	
[8]	-125	-126	Strap	1	

*[1] These part numbers are for reference only.

Table 209:

REPAIR MATERIAL FOR BAY 7L AND BAY 7R					
ITEM	PART NUMBER 111A1152 *[1] LH	RH	PART	QUANTITY	MATERIAL
[1]	-127	-128	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 27.7 inches
[2]	-129	-130	Crossbrace	1	
[3]	-131	-132	Crossbrace	1	
[4]	-133	-134	Joint Angle	1	Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches
[5]	-135	-136	Joint Angle	1	
[6]	-137	-138	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 27.7 inches
[7]	-139	-140	Strap	1	
[8]	-141	-142	Strap	1	
[9]	-249	-250	Strap	1	
[10]	-251	-252	Strap	1	
[11]	-253	-254	Strap	1	
[12]	-143	-144	Bracket	2	Use 7075-T6 AND10134-2001 extruded angle X 6.0 inches

*[1] These part numbers are for reference only.

Table 210:

REPAIR MATERIAL FOR BAY 8L					
ITEM	PART NUMBER 111A1152 *[1] LH	RH	PART	QUANTITY	MATERIAL
[1]	-145	-	Crossbrace	1	Use 7075-T6 in AND10134-1403 extruded angle X 22.0 inches
[2]	-147	-	Crossbrace	1	
[3]	-149	-	Crossbrace	1	
[4]	-151	-	Joint Angle	1	Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches
[5]	-153	-	Joint Angle	1	

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Table 210: (Continued)

REPAIR MATERIAL FOR BAY 8L					
ITEM	PART NUMBER 111A1152 *[1]		PART	QUANTITY	MATERIAL
	LH	RH			
[6]	-155	-	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 22.0 inches
[7]	-157	-	Strap	1	
[8]	-159	-	Strap	1	

*[1] These part numbers are for reference only.

Table 211:

REPAIR MATERIAL FOR BAY 8R					
ITEM	PART NUMBER 111A1152 *[1]		PART	QUANTITY	MATERIAL
	LH	RH			
[1]	-	-193	Crossbrace	1	Use 7075-T6 in AND10134-1403 extruded angle X 32.2 inches
[2]	-	-195	Crossbrace	1	
[3]	-	-197	Crossbrace	1	
[4]	-	-199	Crossbrace	1	
[5]	-	-201	Crossbrace	1	
[6]	-	-231	Crossbrace	1	
[7]	-	-203	Joint Angle	1	Use 7075-T6 in AND10134-2001 extruded angle X 6.0 inches
[8]	-	-205	Joint Angle	1	
[9]	-	-207	Joint Angle	1	
[10]	-	-209	Joint Angle	1	
[11]	-	-211	Strap	1	Use 7075-T6 bare sheet that is 0.063 inch thick X 4.0 inches X 32.2 inches
[12]	-	-213	Strap	1	
[13]	-	-215	Strap	1	
[14]	-	-217	Strap	1	
[15]	-	-219	Strap	1	
[16]	-	-221	Strap	1	
[17]	-	-223	Strap	1	Use 7075-T6 bare sheet that is 0.125 inch thick X 4.0 inches X 32.2 inches
[18]	-	-225	Strap	1	
[19]	-	-227	Strap	1	
[20]	-	-233	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 32.2 inches
[21]	-	-235	Strap	1	
[22]	-	-237	Strap	1	
[23]	-	-239	Strap	1	Use 7075-T6 bare sheet that is 0.025 inch thick X 4.0 inches X 32.2 inches
[24]	-	-241	Strap	1	
[25]	-	-243	Strap	1	

*[1] These part numbers are for reference only.

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Table 212:

REPAIR MATERIAL FOR BAY 9L AND BAY 9R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-161	-162	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 22.6 inches
[2]	-163	-164	Crossbrace	1	
[3]	-165	-166	Crossbrace	1	
[4]	-167	-168	Joint Angle	1	Use 7075-T6 in AND10134-1401 extruded angle X 6.0 inches
[5]	-169	-170	Joint Angle	1	
[6]	-171	-172	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 22.6 inches
[7]	-173	-174	Strap	1	
[8]	-175	-176	Strap	1	

*[1] These part numbers are for reference only.

Table 213:

REPAIR MATERIAL FOR BAY 10L					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-177	-	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 22.0 inches
[2]	-179	-	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 22.0 inches

*[1] These part numbers are for reference only.

Table 214:

REPAIR MATERIAL FOR BAY 10R					
	PART NUMBER 111A1152 *[1]				
ITEM	LH	RH	PART	QUANTITY	MATERIAL
[1]	-	-245	Crossbrace	1	Use 7075-T6 in AND10134-1401 extruded angle X 15.7 inches
[2]	-	-247	Strap	1	Use 7075-T6 bare sheet that is 0.10 inch thick X 4.0 inches X 15.7 inches.

*[1] These part numbers are for reference only.

E. Remove the repair parts.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and the metal parts of initial structure.

G. Apply a chemical conversion coating to the repair parts and the metal parts of initial structure. Refer to 51-20-01, GENERAL.

H. Apply one layer of BMS 10-11, Type I primer to the repair parts and the metal parts of initial structure. Refer to SOPM 20-41-02.

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I. Clean the vapor barrier web assembly.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(1) Remove loose dirt, unwanted grease, or oil. Be careful not to spread the dirt over a larger area than necessary. Refer to SOPM 20-30-99 for the applicable solvents.

(2) Wet the surface with solvent and wipe or scrub with a wiper, sponge, cheesecloth or brush. Repeat this step as necessary to remove the dirt.

(3) Rinse the surface with fresh solvent and wipe with a clean sponge or cheesecloth.

(4) Wipe off the excess solvent and let the surface dry.

(5) Abrade the surface.

WARNING: DO NOT BREATHE THE DUST THAT IS MADE WHEN YOU ABRADE OR CUT CURED COMPOSITE MATERIALS, OR LET THE DUST GET IN YOUR EYES OR ON YOUR SKIN. USE EYE PROTECTION AND PROTECTIVE CLOTHING. MAKE SURE THAT YOU USE RESPIRATORY EQUIPMENT WHEN YOU WORK IN CLOSED LOCATIONS. USE A VACUUM TABLE OR PORTABLE VACUUM TO REMOVE THE DUST WHILE YOU ABRADE OR CUT. IF YOU DO NOT OBEY, INJURY TO PERSONS CAN OCCUR.

CAUTION: DO NOT DAMAGE THE FIBERS WHEN YOU MAKE THE SURFACES ROUGH WITH ABRASIVE PAPER. IF YOU DO NOT OBEY, YOU WILL CAUSE A DECREASE IN THE STRENGTH OF THE PART.

(a) Protect the undamaged area of the web from the abrasive material. Mask around the crack a minimum of 2 inches from the crack. Use material that is resistant to the abrasive material, such as vinyl tape, or Teflon tape.

(b) Abrade (sand) the surface with one of the steps that follow to remove the coating.

NOTE: If you use an orbital driven sander, set the sander below 20,000 revolutions per minute (RPM).

1) Sand the surface with 80 grit or finer abrasive paper for extensive sanding.

2) Sand the surface with 150 grit or finer abrasive paper for final sanding.

3) Finish the surface with 1/0 or finer garnet paper or cloth or fine/ultrafine abrasive fabric pad (Scotch-Brite).

(6) Repeat steps (1) through (4) above to re-clean the repair area.

J. Install the repair parts with BMS 5-45 or BMS 5-95 sealant between the mating surfaces. Refer to Figure 204.

K. Install the fasteners with the fastener head on the opposite side of the repair parts.

NOTE: It is permitted to use 1/64 inch diameter or 1/32 inch diameter oversize fasteners, if necessary. Make sure there is a minimum of 2.0D (D = fastener diameter) edge/end margin.

L. Apply a fillet of BMS 5-45 or BMS 5-95 sealant to all part edges as given in the Figure 204 instructions. Refer to 51-20-05, GENERAL and AMM SUBJECT 51-31-00.

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M. Apply a fillet of BMS 5-45 or BMS 5-95 sealant to all fasteners as given in the Figure 204 instructions. Refer to 51-20-05, GENERAL and AMM SUBJECT 51-31-00.

N. Fill the stop drilled hole with BMS 5-45 or BMS 5-95 sealant flush to -0.00/+0.03 inch. Refer to 51-20-05, GENERAL and AMM SUBJECT 51-31-00.

O. Apply BMS 5-45 or BMS5-95 (class and type optional) and one ply of BMS 9-3,style 1581 fiberglass cloth over the part of the crack that you can see. Use the steps that follow:

(1) Cut a piece of BMS 9-3, style 1581 fiberglass cloth to fit 1.00/1.50 inches (25/38 mm) in all directions beyond the crack.

(2) Layout the fiberglass cloth on a clean work surface.

(3) Impregnate the cloth with BMS 5-45 or BMS5-95 sealant.

(4) Apply sealant .03/.10 inch (0.76/2.5 mm) thick X 1.00/1.50 inches (25/38 mm) in all directions over the crack.

(5) Apply and press the impregnated fiberglass cloth lightly on the vapor barrier. Make sure you have sealant around the edge of the fiberglass cloth.

(6) Apply an additional .03/.10 inch (0.76/2.5 mm) thick layer of sealant over the cloth. Make sure the sealant completely covers the fiberglass cloth. Refer to 51-20-05, GENERAL

P. Apply a layer of BMS 10-11, Type I primer over the vapor barrier web assembly, all repair parts, and the sealant. Refer to SOPM 20-41-02.

Q. Apply a layer of BMS 5-81 coating as given in the Figure 204 instructions to all vapor barrier web locations after installation and fillet sealing of the repair parts. Refer to AMM SUBJECT 28-11-00.

6. Repair Instructions for Fiberglass Fabric Repair as given in Figure 202

A. Get access to the repair area.

B. If there is a crack, then drill a 0.25 inch (6.35 mm) diameter stop hole at the crack ends as given in 51-10-02, GENERAL.

NOTE: A stop hole is not necessary if the crack ends at a fastener hole or ends under existing structure.

C. Do the steps that follow to clean the surface area.

(1) Remove loose dirt, unwanted grease or oil. Be careful not to spread the dirt over a larger area than necessary.

(2) Clean the damaged area with a clean cloth moistened with solvent. Refer to SOPM 20-30-03 for applicable solvents.

(3) Abrade the surface to remove surface gloss with 120 grit, or finer. Use Aluminum Oxide abrasive paper or cloth. Do not sand into fibers.

(a) Clean into the surface with a soft cloth moist with cleaning solvent. Refer to SOPM 20-30-03 for applicable solvents.

(b) Do a visual inspection to ensure the bonding surface is free from dirt, corrosion, grease and unwanted material.

D. Fabricate the repair parts as given in Table 215.

(1) Install the repair parts with BMS 5-95 sealant ensuring the patch extends two inches in all directions past the crack.

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NOTE: Ensure the fabric/patch is completely coated with the sealant.

Do not remove any structure to install the above repair. If the crack runs under a stiffener or other structure, stop the patch at the structure and start on the other side of the structure. There is no need to go under structure common to the web or remove any fasteners to install this repair.

Table 215:

REPAIR MATERIAL FOR FIGURE 202			
PART NUMBER	**PART**	**QUANTITY**	**MATERIAL**
[1]	Repair Patch	1	Use BMS 9-3, Style 120 or Style 1581 fiberglass fabric, 2-inches wide.
[2]	Sealant	1	Use BMS 5-95, Class C-20 or Class F-1 Sealant. Refer to SOPM 20-50-19 for application instructions and BAC5000 for sealant properties such as cure time.

E. Keep a record of the crack length and the repair location.

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382597 S0000134405_V3

Wing Center Section Rear Spar
Figure 202 (Sheet 1 of 2)

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NOTES

1 MORE THAN ONE CRACK IS PERMITTED ON THE LEFT OR RIGHT OF BL 0.0. NO UNREPAIRED DAMAGE IS PERMITTED TO THE ADJACENT ALUMINUM STRUCTURE. ALL DAMAGE MUST BE REPAIRED AS GIVEN IN THE APPLICABLE SRM INSTRUCTIONS.

2 CRACKS THAT TERMINATE AT THE INITIAL FASTENER HOLES DO NOT NEED TO BE STOP DRILLED.

VAPOR BARRIER

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

AFT VIEW FACING FORWARD

B

2476068 S0000580866_V1

Wing Center Section Rear Spar
Figure 202 (Sheet 2 of 2)

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FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A BACR15BB5A()C RIVET.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AS INTITIAL FASTENER. IT IS PERMITTED TO INSTALL A 1/64 INCH OR 1/32 INCH DIAMETER OVERSIZE FASTENER. MAKE SURE THERE IS A MINIMUM OF 1.5D (D = FASTENER DIAMETER) EDGE MARGIN AND A 4D - 6D DISTANCE BETWEEN FASTENERS.

382745 S0000134568_V1

Wing Center Section Vapor Barrier Repair
Figure 203

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CONFIGURATION FOR LINE NUMBERS 1-1639 IS SHOWN
LINE NUMBERS 1640 AND ON ARE SIMILAR
(VIEW LOOKING AFT)

J85149 S0000178830_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 1 of 43)

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(VIEW LOOKING AFT)

A

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

BAY 1L IS SHOWN, BAY 1R IS OPPOSITE

B

J85147 S0000178826_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 2 of 43)

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NOTE: REFER TO THE TABLE 202 FOR BAY 1L AND BAY 1R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

C

J85151 S0000180952_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 3 of 43)

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NOTE: REFER TO THE TABLE 202 FOR BAY 1L AND BAY 1R MATERIALS

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 4 of 43)

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(VIEW LOOKING AFT)

D

BAY 2L IS SHOWN, BAY 2R IS OPPOSITE

E

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85499 S0000181658_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 5 of 43)

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NOTE: REFER TO THE TABLE 203 FOR BAY 2L AND BAY 2R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

F

J85768 S0000181673_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 6 of 43)

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NOTE: REFER TO THE TABLE 203 FOR BAY 2L AND BAY 2R MATERIALS

J86440 S0000181688_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 7 of 43)

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(VIEW LOOKING AFT)

G

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

BAY 3L IS SHOWN, BAY 3R IS SIMILAR

H

J85502 S0000181659_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 8 of 43)

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NOTE: REFER TO THE TABLE 204 FOR BAY 3L AND BAY 3R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

I

J85815 S0000181678_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 9 of 43)

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NOTE: REFER TO THE TABLE 204 FOR BAY 3L AND BAY 3R MATERIALS

J86816 S0000181695_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 10 of 43)

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NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85519 S0000181660_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 11 of 43)

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NOTE: REFER TO THE TABLE 205 FOR BAY 4L
AND BAY 4R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

L

J86053 S0000181674_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 12 of 43)

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NOTE: REFER TO THE TABLE 205 FOR BAY 4L AND BAY 4R MATERIALS

J86817 S0000181690_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 13 of 43)

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(VIEW LOOKING AFT)

M

BAY 5L

N

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85541 S0000181661_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 14 of 43)

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NOTE: REFER TO THE TABLE 206 FOR BAY 5L MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

0

J85850 S0000181676_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 15 of 43)

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NOTE: REFER TO THE TABLE 206 FOR BAY 5L MATERIALS

J86764 S0000181694_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 16 of 43)

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(VIEW LOOKING AFT)

P

BAY 5R

Q

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85554 S0000181662_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 17 of 43)

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NOTE: REFER TO THE TABLE 207 FOR BAY 5R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

R

J87018 S0000181685_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 18 of 43)

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L-L

M-M

NOTE: REFER TO THE TABLE 207 FOR BAY 5R MATERIALS

J87041 S0000181682_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 19 of 43)

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NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

BAY 6L IS SHOWN, BAY 6R IS OPPOSITE

T

J85581 S0000181663_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 20 of 43)

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NOTE: REFER TO THE TABLE 208 FOR BAY 6L AND BAY 6R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

U

J86112 S0000181680_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 21 of 43)

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NOTE: REFER TO THE TABLE 208 FOR BAY 6L AND BAY 6R MATERIALS

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 22 of 43)

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(VIEW LOOKING AFT)

V

BAY 7L IS SHOWN, BAY 7R IS OPPOSITE

W

(2 LOCATIONS)
R-R

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85594 S0000181664_V4

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 23 of 43)

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NOTE: REFER TO THE TABLE 209 FOR BAY 7L AND BAY 7R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

X

J86624 S0000181687_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 24 of 43)

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NOTE: REFER TO THE TABLE 209 FOR BAY 7L AND BAY 7R MATERIALS

J86704 S0000181689_V5

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 25 of 43)

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NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85606 S0000181665_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 26 of 43)

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NOTE: REFER TO THE TABLE 210 FOR BAY 8L MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

AA

J86072 S0000181681_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 27 of 43)

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NOTE: REFER TO THE TABLE 210 FOR BAY 8L MATERIALS

J86957 S0000181696_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 28 of 43)

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NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85782 S0000181666_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 29 of 43)

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EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

DD

J86674 S0000181686_V3

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 30 of 43)

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NOTE: REFER TO THE TABLE 211 FOR BAY 8R MATERIALS

J86868 S0000181691_V3

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 31 of 43)

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NOTE: REFER TO THE TABLE 211 FOR BAY 8R MATERIALS

J85155 S0000178829_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 32 of 43)

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(VIEW LOOKING AFT)

EE

NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

BAY 9L IS SHOWN, BAY 9R IS OPPOSITE

FF

J85834 S0000181668_V3

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 33 of 43)

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NOTE: REFER TO THE TABLE 212 FOR BAY 9L AND BAY 9R MATERIALS

EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

GG

J86342 S0000181684_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 34 of 43)

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Y–Y

Z–Z

NOTE: REFER TO THE TABLE 212 FOR BAY 9L
AND 9R MATERIALS

J87361 S0000181697_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 35 of 43)

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NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85843 S0000181671_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 36 of 43)

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EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

JJ

AA–AA

NOTE: REFER TO THE TABLE 213 FOR BAY 10L MATERIALS

J85837 S0000181692_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 37 of 43)

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NOTE: FIGURE 204 ADDRESSES REPAIR ACTION TAKEN FOR CRACKS IN THE REAR SPAR VAPOR BARRIER WEB ON AN INDIVIDUAL BAY BY BAY BASIS. CRACKS INSIDE ANY INDIVIDUAL BAY WHICH DO NOT EXTEND CLOSER THAN TWO INCHES TO ADJACENT STRUCTURE REQUIRE REPAIR ONLY TO THE AFFECTED BAY. CRACKS IN AN INDIVIDUAL BAY WHICH ARE CLOSER THAN TWO INCHES TO AN ADJACENT BAY REQUIRE REPAIR TO THE ADJACENT BAY OR BAYS ALSO.

J85855 S0000181672_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 38 of 43)

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EXPLODED VIEW OF THE REPAIR PARTS AND ADJACENT PARTS

NOTE: REFER TO THE TABLE 214 FOR BAY 10R MATERIALS

J86038 S0000181683_V2

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 39 of 43)

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AND10134-1401 JOINT ANGLE
AND10134-1401 CROSSBRACE

NOTE: HOLES IN ALL JOINT ANGLES ARE CLASS 2 AND CLASS 3 AS SHOWN

TYPICAL JOINT ANGLE

AND10134-2001 JOINT ANGLE
AND10134-1403 CROSSBRACE

AND10134-2001 JOINT ANGLE
AND10134-2001 CROSSBRACE

DETAILS FOR JOINT ANGLES

NN

J85153 S0000178827_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 40 of 43)

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Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 41 of 43)

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NOTES

- PROVIDE 2D MINIMUM EDGE MARGIN AT ALL LOCATIONS UNLESS SHOWN DIFFERENTLY.
- PROVIDE 4-6D FASTENER SPACING AT NEW LOCATIONS.
- INSTALL HI-LOKS WITH WET BMS 5-95.
- ALL BEND RADII TO BE 0.38 TO 0.50 UNLESS SHOWN DIFFERENTLY.
- PARTS TO BE TRIMMED WITH THE DIMENSIONS AND A TOLERANCE OF -0.000 TO +0.050.
- ALL DIMENSIONS ARE IN INCHES.

1 PERMITTED CRACKS ARE MORE THAN 2.0 INCHES FROM AN ADJACENT BAY

2 IF REPAIRED BEFORE AS GIVEN IN FIGURE 203, THEN CONTACT THE BOEING COMPANY FOR INSTRUCTIONS

3 USE THIS DIMENSION TO ORDER THE MATERIAL AND FABRICATE THE PART (DIMENSION INCLUDES EXCESS)

4 GAP BETWEEN THE ENDS OF STRAPS AND ADJACENT STRUCTURE OR PARTS TO BE 0.020 TO 0.080

5 MAKE SURE NEW PARTS FIT AROUND FASTENERS AND STRUCTURES WITH A 0.020 TO 0.080 GAP

6 CHAMFER ANGLE TO CLEAR RADIUS BEFORE PROTECTIVE TREATMENT IS APPLIED (0.120 TO 0.180 BY 45°)

7 RADIUS PARTS ARE 0.12 TO 0.18 R TO FIT INTO RADIUS OF ADJACENT PART

8 CAUTION: THE BMS 5-81 COATING MAKES A FLAMMABLE VAPOR BARRIER. MAKE SURE THE VAPOR BARRIER IS NOT DAMAGED.

APPLY BMS 10-11, TYPE 1, PRIMER AND BMS 5-81 COATING AS SHOWN IN BAC 5527 OVER FORWARD SURFACE OF VAPOR BARRIER WEB ASSEMBLY AND ALL SURFACES OF NEWLY ATTACHED PARTS TO A THICKNESS OF 0.015 TO 0.025. BMS 5-81 COATING IS TO EXTEND 1.0 INCH MINIMUM ONTO SURFACES OF COMPONENTS ATTACHED TO THE VAPOR BARRIER WEB

9 FILLET SEAL (BMS 5-95) ALL EDGES OF REPAIR PARTS, SURROUNDING PARTS, AND AFFECTED FASTENERS ON THE FORWARD SIDE OF THE VAPOR BARRIER WEB ASSEMBLY

J85160 S0000178828_V1

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 42 of 43)

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FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL AND REPAIR FASTENER LOCATIONS:
- HOLE; 0.1635 TO 0.1645 DIA
- EDGE MARGIN; 0.3270 MIN
- HEX DRIVE BOLT; BACB30VT5
- COLLAR; BACC30BL5

REPAIR FASTENER LOCATION:
- HOLE; 0.1660 TO 0.1710 DIA
- EDGE MARGIN; 0.3320 MIN
- HEX DRIVE BOLT; BACB30VT5HK()
- COLLAR; BACC30BL5

INITIAL AND REPAIR FASTENER LOCATIONS:
- HOLE; 0.1895 TO 0.1905 DIA
- EDGE MARGIN; 0.3790 MIN
- HEX DRIVE BOLT; BACB30VT6K
- COLLAR; BACC30BL6

INITIAL FASTENER LOCATION:
- HOLE; 0.2495 TO 0.2505 DIA
- EDGE MARGIN; 0.4990 MIN
- HEX DRIVE BOLT; BACB30VT8K
- COLLAR; BACC30BL8

INITIAL AND REPAIR FASTENER LOCATIONS:
- HOLE; 0.1635 TO 0.1645 DIA
- EDGE MARGIN; 0.3270 MIN
- HEX DRIVE BOLT; BACB30MB5
- COLLAR; BACC30BH5

INITIAL FASTENER LOCATION:
- HOLE; 0.1895 TO 0.1905 DIA
- EDGE MARGIN; 0.3790 MIN
- HEX DRIVE BOLT; BACB30MB6
- COLLAR; BACC30BH6

REPAIR FASTENER LOCATION:
- HOLE; 0.1920 TO 0.2030 DIA
- EDGE MARGIN; 0.29 MIN
- HEX DRIVE BOLT; BACB30VT5HK()
- COLLAR; BACC30BL5
- WASHER; BACW10CT5J OR BACW10EG5J (HEAD SIDE)

J88485 S0000184336_V4

Wing Center Section Rear Spar Web Cross Brace Repair
Figure 204 (Sheet 43 of 43)

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REPAIR 3 - WING CENTER SECTION REAR SPAR VERTICAL PRESSURE DECK WEB REPAIR

1. Applicability

A. REPAIR 3 is applicable to damage on the vertical pressure deck web of the wing center section rear spar at STA 663 in BL 32 to BL 45 bays. Refer to Figure 201/REPAIR 3.

B. REPAIR 3 is applicable to airplane line numbers 1 through 3457 which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

C. REPAIR 3 is applicable to the vertical pressure deck web with 0.040 in. (1.02 mm) and 0.045 in. (1.14 mm) thick pockets.

2. General

A. This repair is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 6./REPAIR 3. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for the repair categories and definitions.

B. This repair has two alternatives:

Alternative 1: Use this repair if there are damages on one bay of the vertical pressure deck web, BL 32 to BL 37. Refer to Paragraph 4./REPAIR 3 and Figure 203/REPAIR 3.

Alternative 2: Use this repair if there are damages on two bays of the vertical pressure deck web, BL 32 to BL 45. Refer to Paragraph 5./REPAIR 3 and Figure 204/REPAIR 3.

C. Maintain a minimum of 2D edge margin on all fastener and all web pocket chem-milled radius.

D. Maintain 4D to 6D fastener spacing.

E. Make sure that there is sufficient edge margin from the edge of chem-milled pocket and do not drill fastener holes into chem-milled radius.

F. The pull-up on all parts must be 0.005 in. (0.127 mm) or less.

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2153653 S0000458140_V1

Wing Center Section Rear Spar Vertical Pressure Deck Web Location
Figure 201

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-01	SHEET METAL MATERIALS
51-40-03	FASTENER SUBSTITUTION
AMM 51-21-00	INTERIOR AND EXTERIOR FINISHES
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
737 NDT Part 6, 51-00-00, Procedure 17	Open Hole Inspection of Titanium and Other Low Conductivity Materials
737 NDT Part 6, 51-00-00, Procedure 23	Aluminum Part Surface Inspection (Impedance Plane Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)
737 NDT Part 6, 51-00-26	Aluminum Part Subsurface Inspection - Multilayer

4. Repair Instructions for Alternative 1: Damages on One Bay, BL 32 to BL 37

A. Get access to the damaged area.

B. Cut and remove the damaged part of the web within the 0.040 in. (1.02 mm) or 0.045 in. (1.14 mm) pocket as shown in Figure 203/REPAIR 3. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

(1) It is acceptable to cut out all or part of the web in the pocket.

(2) If there are more than one cutout in the pocket, then the edges of the cutouts must be a minimum of 2.00 in. (50.8 mm) apart.

(3) Maintain a minimum of 0.50 in. (12.7 mm) radius on the cutout.

(4) If the damage extend out of the pocket, do Alternative 2 repair as given in Paragraph 5./REPAIR 3 if applicable or contact The Boeing Company for the repair instructions.

C. Do a High Frequency Eddy Current (HFEC) inspection of the cutout to make sure there are no cracks. Refer to 737 NDT Part 6, 51-00-00, Procedure 4.

D. Make the repair parts. Refer to Table 201/REPAIR 3 and Figure 203/REPAIR 3.

(1) Make the corner radius of all the repair parts 0.25 in. (6.35 mm).

Table 201: Repair Material

PART	DESCRIPTION	QUANTITY	MATERIAL
[1]	Repair Web	1	Use 0.10 in. (2.54 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4
[2]	Filler	1	Use 0.10 in. (2.54 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4

E. Assemble the repair parts and drill the fastener holes.

F. Disassemble the repair parts.

G. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the web.

H. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the web. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

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I. Apply one layer of BMS 10-11, Type I primer to the repair parts and to the bare surfaces of the web. Refer to SOPM 20-41-02.

J. Install the repair parts with BMS 5-95 sealant.

K. Shim gaps. The thickness of the shim cannot be more than 0.02 in. (0.51 mm). Refer to SHEET METAL MATERIALS, 51-30-01.

L. Install the fastener with BMS 5-95 sealant.

M. Apply a fillet seal. Refer to REPAIR SEALING, 51-20-05.

N. Apply a finish to the repair area. Refer to AMM SUBJECT 51-21-00.

O. Reinstall the items that you removed for access.

5. Repair Instructions for Alternative 2: Damages on Two Bays, BL 32 to BL 45

A. Get access to the damaged area.

B. Cut and remove the damaged part of the web at the cut line as shown in Figure 204/REPAIR 3.

NOTE: Be careful not to damage the chords and other structures.

(1) Cut the web into smaller pieces for easy removal, if necessary.

C. Do a detailed visual inspection of the repair area to make sure there are no damages. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02. It is optional to do a High Frequencey Eddy Current (HFEC) inspection on the cut edge as given in 737 NDT Part 6, 51-00-00, Procedure 4 or 737 NDT Part 6, 51-00-00, Procedure 23.

D. Make the repair parts. Refer to Table 202/REPAIR 3 and Figure 204/REPAIR 3.

(1) Make the corner radius of all the repair parts 0.25 in. (6.35 mm).

Table 202: Repair Material

PART	DESCRIPTION	QUANTITY	MATERIAL
[1]	Repair Web	1	Use 0.15 in. (3.81 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4 *[1]
[2]	Repair Web	1	Use 0.10 in. (2.54 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4
[3]	Filler	1	Use 0.10 in. (2.54 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4
[4]	Filler	1	Use 0.10 in. (2.54 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4

*[1] It is acceptable to fabricate the Part 1 Repair Web from 0.16 in. (4.06 mm) thick 2024-T3 bare sheet as given in QQ-A-250/4 that is machined to the required thickness of 0.15 in. (3.81 mm).

E. Assemble the repair parts and drill the fastener holes.

F. Disassemble the repair parts.

G. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts, the web, and the floor beam fitting.

H. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the web and the floor beam fitting. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

I. Apply one layer of BMS 10-11, Type I primer to the repair parts and to the bare surfaces of the web. Refer to SOPM 20-41-02.

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J. Install the repair parts with BMS 5-95 sealant.

K. Shim gaps. The thickness of the shim cannot be more than 0.02 in. (0.51 mm). Refer to SHEET METAL MATERIALS, 51-30-01.

L. Install the fastener with BMS 5-95 sealant.

M. Apply a fillet seal. Refer to REPAIR SEALING, 51-20-05.

N. Apply a finish to the repair area. Refer to AMM SUBJECT 51-21-00.

O. Reinstall the items that you removed for access.

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NOTES

- ALL DIMENSIONS ARE IN INCHES (mm).
- D = DIAMETER OF THE FASTENER.

1 IT IS ACCEPTABLE TO TAPER THE UPPER 1.00 INCH (25.4 mm) OF THE EDGE OF THE PART 2 FILLER TO ALIGN WITH THE 0.09 INCH (2.29 mm) THICK COMPRESSION CHORD AS AN ALTERNATIVE TO INSTALL SHIM AT THE COMPRESSION CHORD. THE MAXIMUM MATERIAL REMOVAL IS 0.01 INCH (0.25 mm). USE 100:1 TAPER RATIO.

2 IT IS ACCEPTABLE TO TAPER THE UPPER 1.00 INCH (25.4 mm) OF THE EDGE OF THE PART 3 FILLER TO ALIGN WITH THE 0.09 INCH (2.29 mm) THICK COMPRESSION CHORD AS AN ALTERNATIVE TO INSTALL SHIM AT THE COMPRESSION CHORD. THE MAXIMUM MATERIAL REMOVAL IS 0.01 INCH (0.25 mm). USE 100:1 TAPER RATIO.

3 THE CUTOUT OF ALL OF THE WEB IN THE 0.040 INCH (1.02 mm) OR 0.045 INCH (1.14 mm) POCKET IS SHOWN. IT IS ACCEPTABLE TO CUT OUT ALL OR PART OF THE WEB IN THE POCKET.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER. IF AN OVERSIZE IS NECESSARY, INSTALL UP TO 1/64 INCH DIAMETER OVERSIZE. FASTENER SUBSTITUTION AS GIVEN IN SRM 51-40-03 IS PERMITTED.

REPAIR FASTENER LOCATION. INSTALL A BACB30NX6K HEX DRIVE BOLT WITH A BACC30BH6 COLLAR.

2153689 S0000462983_V2

Notes and Fastener Symbols
Figure 202

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Rear Spar Vertical Pressure Deck Web Repair for Alternative 1
Figure 203 (Sheet 1 of 2)

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2154687 S0000465324_V1

Rear Spar Vertical Pressure Deck Web Repair for Alternative 1
Figure 203 (Sheet 2 of 2)

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2154742 S0000458086_V1

Rear Spar Vertical Pressure Deck Web Repair for Alternative 2
Figure 204 (Sheet 1 of 2)

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Rear Spar Vertical Pressure Deck Web Repair for Alternative 2
Figure 204 (Sheet 2 of 2)

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6. Inspection Requirements

A. For Alternative 1, do the supplemental inspections as given in Table 203/REPAIR 3 and Figure 205/REPAIR 3.

Table 203: Category B Repair Inspection Requirements for Alternative 1

CATEGORY B REPAIR INSPECTION REQUIREMENTS				
INSPECTION THRESHOLD	REPEAT INSPECTION			
	METHOD	INTERVAL	STRUCTURE DETAIL	REFERENCE
56,000 flight cycles after repair installation	High Frequency Eddy Current (HFEC) Inspection	36,000 flight cycles	Initial web from the aft side (arrow 1)	737 NDT Part 6, 51-00-00, Procedure 4 or 737 NDT Part 6, 51-00-00, Procedure 23
			Part 2 Filler not common to the Part 1 Repair Web from the forward side (arrow 2)	
			Part 1 Repair Web from the forward side (arrow 4)	
			Horizontal flange of the upper compression chord from the aft side (arrow 5)	
			Horizontal flange of the lower attach angle from the forward side (arrow 6)	
			12 fastener holes on the vertical flange of the upper compression chord common to the initial web and the Part 1 Repair Web (arrow 7)	737 NDT Part 6, 51-00-00, Procedure 17
	Low Frequency Eddy Current (LFEC) Inspection	36,000 flight cycles	Part 2 Filler common to the Part 1 Repair Web from the forward side (arrow 3)	737 NDT Part 6, 51-00-26

B. For Alternative 2, do the supplemental inspections as given in Table 204/REPAIR 3 and Figure 206/REPAIR 3.

Table 204: Category B Repair Inspection Requirements for Alternative 2

CATEGORY B REPAIR INSPECTION REQUIREMENTS				
INSPECTION THRESHOLD	REPEAT INSPECTION			
	METHOD	INTERVAL	STRUCTURE DETAIL	REFERENCE

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Table 204: Category B Repair Inspection Requirements for Alternative 2 (Continued)

CATEGORY B REPAIR INSPECTION REQUIREMENTS				
56,000 flight cycles after repair installation	High Frequency Eddy Current (HFEC) Inspection	36,000 flight cycles	Initial web and Part 1 Repair Web from the aft side (arrow 1)	737 NDT Part 6, 51-00-00, Procedure 4 or 737 NDT Part 6, 51-00-00, Procedure 23
			Part 3 Filler not common to the Part 2 Repair Web from the forward side (arrow 2)	
			Part 2 Repair Web from the forward side (arrow 4)	
			Horizontal flange of the upper compression chord from the aft side (arrow 5)	
			Vertical flange of the upper compression chord not common to the initial web or the Part 1 Repair Web from the aft side (arrow 6)	
			Horizontal flange of the lower attach angle from the forward side (arrow 7)	
			11 fastener holes on the vertical flange of the upper compression chord common to the initial web and the Part 2 Repair Web, the Part 1 Repair Web, or the Part 4 Filler (arrow 8)	737 NDT Part 6, 51-00-00, Procedure 17
	Low Frequency Eddy Current (LFEC) Inspection	36,000 flight cycles	Part 3 Filler common to the Part 2 Repair Web from the forward side (arrow 3)	737 NDT Part 6, 51-00-26

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Inspection Requirements for Alternative 1
Figure 205

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2155026 S0000458091_V1

Inspection Requirements for Alternative 2
Figure 206 (Sheet 1 of 2)

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SECTION B-B

SECTION C-C

2155032 S0000470292_V1

Inspection Requirements for Alternative 2
Figure 206 (Sheet 2 of 2)

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IDENTIFICATION 1 - WING CENTER SECTION SPANWISE BEAMS

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F73198 S0006594152_V1

Wing Center Section Spanwise Beam Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A5000	Spanwise Beam Installation - Wing Center Section
111A5001	Spanwise Beam Assembly - No. 1, Wing Center Section
111A5002	Spanwise Beam Assembly - No. 2, Wing Center Section

3. Applicability

A. IDENTIFICATION 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. Refer to Table 2 for List of Materials in Figure 2.

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Integral Web Number 1		7050-T7451 plate that is machined to different thicknesses	
[2]	Number 8 Stringer		BAC1518-1168 7055-T77511 extrusion as given in BMS 7-308	Line Number 1 thru 1537
[2]	Number 8 Stringer		BAC1518-1266 7055-T77511 extrusion as given in BMS 7-308	Line Number 1538 and on
[3]	Number 6 Stringer		BAC1517-2739 2224-T3511 extrusion as given in BMS 7-255	
[4]	Tension Fitting		7050-T7451 plate	
[5]	Backup Fitting		7050-T7451 plate	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

C. Refer to Table 3 for List of Materials in Figure 3.

Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Integral Web Number 2		7050-T7451 plate that is machined to different thicknesses	
[2]	Number 16 Stringer		BAC1518-1168 7055-T77511 extrusion as given in BMS 7-308	
[3]	Number 11 Stringer		BAC1517-2738 2224-T3511 extrusion as given in BMS 7-255	
[4]	Tension Fitting		7050-T7451 plate	
[5]	Backup Fitting		7050-T7451 plate	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

F72825 S0006594154_V1

Wing Center Section Spanwise Beam Number 1 Identification
Figure 2

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

F72859 S0006594156_V1

Wing Center Section Spanwise Beam Number 2 Identification
Figure 3

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IDENTIFICATION 2 - WING CENTER SECTION SUPPORT BEAMS

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F72905 S0006594159_V1

Wing Center Section Support Beam Locations
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
111A7000	Lower Beam Installation - Wing Center Section
111A7001	Lower Beam Assembly - Wing Center Section

3. Applicability

A. IDENTIFICATION 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. Refer to Table 2 for List of Materials in Figure 2.

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Lower Beam Web		7050-T7451 plate that is machined to different thicknesses	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[2]	Horizontal Stiffener		BAC1506-4365 7075-T3511 extrusion	
[3]	Lower Chord		BAC1506-4274 7075-T73 extrusion	
[4]	Front Spar Attach Stiffener		BAC1514-2763 7075-T3511 extrusion	
[5]	Tension Fitting		7050-T7451 plate	
[6]	Tension Fitting		7050-T7451 plate	
[7]	Rear Spar Attach Stiffener		BAC1505-101665 7075-T3511 extrusion	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE SUPPORT BEAM IS SHOWN,
RIGHT SIDE SUPPORT BEAM IS OPPOSITE

F73587 S0006594161_V1

Wing Center Section Support Beam Identification
Figure 2

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ALLOWABLE DAMAGE 1 - WING CENTER SECTION SPANWISE BEAMS

1. Applicability

A. ALLOWABLE DAMAGE 1 is applicable to the wing center section spanwise beams. Refer to Wing Center Section Spanwise Beam Allowable Damage, Figure 101/ALLOWABLE DAMAGE 1.

B. ALLOWABLE DAMAGE 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

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Wing Center Section Spanwise Beam Allowable Damage
Figure 101

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

2. General

WARNING: THE GASSES FROM THE FUEL IN THE FUEL TANK ARE DANGEROUS AND EXPLOSIVE. MAKE SURE YOU REMOVE ALL OF THE FUEL FROM THE FUEL TANKS. MAKE SURE THAT THE FUEL TANKS HAVE A GOOD FLOW OF AIR. USE RESPIRATORY EQUIPMENT AND WEAR PROTECTIVE CLOTHING WHEN YOU GO IN THE FUEL TANKS. DO NOT USE EQUIPMENT THAT IS A SOURCE OF HEAT OR SPARKS. IF YOU DO NOT OBEY, AN EXPLOSION CAN OCCUR AND CAUSE INJURY TO PERSONS AND DAMAGE TO THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

C. Remove the damage material as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasives and other materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the tools and equipment you can use to remove the damage.

D. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

E. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the areas that have rework. Refer to 51-20-01.

(3) Apply two layers of BMS 10-20, Type II primer to the areas that have rework. Refer to AMM 28-11-00/701.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-05	FASTENER HOLE SIZES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling
SOPM 20-10-03	SHOT PEENING

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(Continued)

Reference	Title
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. **Allowable Damage Limits**

A. Webs

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , and D .

(3) Dents are permitted between LBL 60.6 and RBL 60.6 only as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Detail E .

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 1.5 inch away from the edge of a hole, the edge of other damage, or the material edge.

(b) A maximum of 0.25 inch in diameter after they are drilled and filled with a 1/4 inch diameter or smaller rivet.

1) Use a 2117-T3 or 2177-T4 aluminum rivet.

2) Drill the hole to the applicable diameter as given in 51-40-05.

3) Install the rivet without sealant.

B. Stringers and Stiffeners

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Fittings

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Spanwise Beams Allowable Damage Details, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, C, and F.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

A–A

F75606 S0006594166_V1

Wing Center Section Spanwise Beams Allowable Damage Details
Figure 102 (Sheet 1 of 3)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

D

DENT THAT IS PERMITTED

E

F75607 S0006594167_V1

Wing Center Section Spanwise Beams Allowable Damage Details
Figure 102 (Sheet 2 of 3)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

F

F75610 S0006594168_V1

Wing Center Section Spanwise Beams Allowable Damage Details
Figure 102 (Sheet 3 of 3)

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ALLOWABLE DAMAGE 2 - WING CENTER SECTION SUPPORT BEAMS

1. Applicability

A. ALLOWABLE DAMAGE 2 is applicable to the wing center section support beams. Refer to Wing Center Section Support Beam Locations, Figure 101/ALLOWABLE DAMAGE 2.

B. ALLOWABLE DAMAGE 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

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LEFT SIDE SUPPORT BEAM IS SHOWN,
RIGHT SIDE SUPPORT BEAM IS OPPOSITE

A

F75613 S0006594170_V1

Wing Center Section Support Beam Locations
Figure 101

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damaged material as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasives and other materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the tools and equipment you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the areas that have rework. Refer to 51-20-01.

(3) Apply two layers of BMS 10-20, Type II primer to the areas that have rework. Refer to AMM 28-11-00/701.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-05	FASTENER HOLE SIZES
AMM 51-21-00	INTERIOR AND EXTERIOR FINISHES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Webs

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A, B, and F.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and F .

(3) Dents are permitted as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Detail G .

(4) Holes and Punctures are permitted if they are:

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(a) A minimum of 1.5 inch away from the edge of a hole, the edge of other damage, or the material edge

(b) A maximum of 0.25 inch in diameter after they are drilled and filled with a 1/4 inch diameter or smaller rivet.

1) Use a 2117-T3 or 2117-T4 aluminum rivet

2) Drill the hole to the applicable diameter as given in 51-40-05

3) Install the rivet without sealant.

B. Stiffeners

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A, B, and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Chord

(1) Cracks:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A, B, and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Fittings

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Center Section Support Beam Allowable Damage, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and D .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

F75614 S0006594171_V1

Wing Center Section Support Beam Allowable Damage
Figure 102 (Sheet 1 of 4)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

D

B–B

F75615 S0006594172_V1

Wing Center Section Support Beam Allowable Damage
Figure 102 (Sheet 2 of 4)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE BUT NOT MORE THAN 0.05 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

E

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

F

F75618 S0006594173_V1

Wing Center Section Support Beam Allowable Damage
Figure 102 (Sheet 3 of 4)

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DENT THAT IS PERMITTED

G

F75619 S0006594174_V1

Wing Center Section Support Beam Allowable Damage
Figure 102 (Sheet 4 of 4)

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REPAIR 1 - WING CENTER SECTION SPANWISE BEAMS

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

SPANWISE BEAM NUMBER 1 IS SHOWN,
SPANWISE BEAM NUMBER 2 IS ALMOST THE SAME

A

F75895 S0006594177_V2

Wing Center Section Spanwise Beam Repairs
Figure 201

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REPAIR 2 - WING CENTER SECTION SUPPORT BEAMS

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE SUPPORT BEAM IS SHOWN,
RIGHT SIDE SUPPORT BEAM IS OPPOSITE

A

F75894 S0006594179_V2

Wing Center Section Support Beam Repairs
Figure 201

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IDENTIFICATION 1 - OUTER WING UPPER INSPAR SKIN

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

G46146 S0006594184_V1

Outer Wing Upper Inspar Skin Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
112A0001	Functional Product Collector - Skin Panel
112A0002	Functional Product Collector - Skin Panel
112A3001	Panel Installation - Upper, Section 12, Wing
112A3050	Skin Splice Installation - Upper Forward, Wing
112A3401	Attach Strap Installation - Leading Edge to Wing, Upper
112A3500	Vortex Generator Installation - Upper Surface, Wing
737–8021	Installation - Upper Panel, Outboard*[1]
737–3601	Panel Installation - Machined, Upper Wingtip, Wing*[2]

*[1] Aviation Partners Boeing (APB) part as retrofitted by Supplemental Type Certificate (STC) ST00830SE for Unprovisioned Wing.

*[2] Aviation Partners Boeing (APB) part as retrofitted by Supplemental Type Certificate (STC) ST00830SE for Partially Provisioned Wing.

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3. Applicability

A. Identification 1 is applicable to all airplanes which include those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) winglet.

B. List of Materials:

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin Panel Number 1		7055-T7751 plate as given in BMS 7-307. Refer to the production drawings for the the machined thicknesses	
[2]	Skin Panel Number 2		7055-T7751 plate as given in BMS 7-307. Refer to the production drawings for the machined thicknesses	
[3]	Splice Plate	0.040 (1.02)	6013-T4 sheet as given in AMS 43476	
[4]	Vortex Generator		AND10133-0602 2024-T3511 extrusion. (Optional: BAC1503-100093)	
[5]	Wing Tip Upper Skin Panel	0.312 (7.937)	7050-T7451 plate. Refer to Figure 3 for the chem-mill thicknesses	
[5]	Wing Tip Upper Skin Panel	1.000 (25.4)	7050-T7451 plate as given in AMS4050. Refer to the production drawings for the machined thicknesses.	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.
[6]	Strap	0.100 (2.54)	7075-T6 sheet	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.
- 1 FOR ALL AIRPLANES NOT GIVEN IN 2
- 2 AVIATION PARTNERS BOEING (APB) PART AS RETROFITTED BY SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE FOR UNPROVISIONED WING.

2274737 S0000512530_V1

Outer Wing Upper Inspar Skin Identification
Figure 2

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BONDED UPPER WING TIP PANEL
(FOR CUM LINE NUMBER 1 THRU 777)

A–A

B–B

NOTE: ALL DIMENSIONS ARE IN INCHES.

G46985 S0006594189_V1

Chem-mill Thicknesses for Figure 2, Item [5]
Figure 3 (Sheet 1 of 2)

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BONDED UPPER WING TIP PANEL
(FOR CUM LINE 778 AND ON)

A–A

B–B

NOTE: ALL DIMENSIONS ARE IN INCHES.

L59855 S0006594190_V2

Chem-mill Thicknesses for Figure 2, Item [5]
Figure 3 (Sheet 2 of 2)

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IDENTIFICATION 2 - OUTER WING LOWER INSPAR SKIN

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

G47018 S0006594193_V1

Outer Wing Lower Inspar Skin Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
112A0001	Functional Product Collector - Skin Panel
112A0002	Functional Product Collector - Skin Panel
112A4001	Panel Installation - Lower, Section 12, Wing
112A4050	Skin Splice Installation - Lower Aft, Wing
112A4051	Skin Splice Installation - Lower Forward, Wing
112A4301	Splice Installation - BBL 70.85, Lower Panel
112A4601	Panel Installation - Honeycomb, Lower, Wingtip, Wing
737–8022	Installation - Lower Panel, Outboard*[1]
737–4601	Panel Installation - Machined, Lower Wingtip, Wing *[2]

*[1] Aviation Partners Boeing (APB) part as retrofitted by Supplemental Type Certificate (STC) ST00830SE for Unprovisioned Wing.

*[2] Aviation Partners Boeing (APB) part as retrofitted by Supplemental Type Certificate (STC) ST00830SE for Partially Provisioned Wing.

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3. Applicability

A. Identification 2 is applicable to all airplanes which include those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) winglet.

B. List of Materials

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin Panel Number 1		2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the machined thicknesses	
[2]	Skin Panel Number 2		2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the machined thicknesses	
[3]	Skin Panel Number 3		2324-T39 plate as given in BMS 7-254. Refer to the production drawings for the machined thicknesses	
[4]	Splice Plate (2)	0.040 (1.02)	6013-T4 sheet as given in AMS 4347	
[5]	Splice Plate	0.100 (2.54)	2024-T3 sheet as given in QQ-A-250/4	
[6]	Bonded Wing Tip Panel, Lower		Bonded panel as given in BAC5514-5101 with 1 ply of BMS 5-101, Type II, Grade 10 adhesive. Refer to Figure 3 for the chem-mill thicknesses	
	Face Sheet, Inner	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Face Sheet, Inner	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12	
	Skin	0.180 (4.57)	7075-T6 sheet as given in QQ-A-250/12	
	Core (2)		5052 aluminum honeycomb as given in BMS 4-4, Type 3-15NPA, Grade I	
[6]	Wing Tip Lower Skin Panel	1.000 (25.4)	2024-T351 plate as given in QQ-A-250/4. Refer to the production drawings for the machined thicknesses	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

1. FOR ALL AIRPLANES NOT GIVEN IN 2
2. AVIATION PARTNERS BOEING (APB) PART AS RETROFITTED BY SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE FOR UNPROVISIONED WING.

2274739 S0000512559_V1

Outer Wing Lower Inspar Skin Identification
Figure 2

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BONDED LOWER WING TIP PANEL
(FOR CUM LINE NUMBER 1 THRU 777)

A-A

NOTE: ALL DIMENSIONS ARE IN INCHES.

L59709 S0006594198_V1

Chem-Mill Thicknesses for Figure 2, Item [6]
Figure 3 (Sheet 1 of 4)

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BONDED LOWER WING TIP PANEL
(FOR CUM LINE NUMBER 778 AND ON)

A–A

NOTE: ALL DIMENSIONS ARE THICKNESSES IN INCHES.

L59819 S0006594199_V1

Chem-Mill Thicknesses for Figure 2, Item [6]
Figure 3 (Sheet 2 of 4)

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BONDED LOWER WING TIP PANEL
(FOR AIRPLANE CUM LINE NUMBER 789 AND ON WITH
FULL WINGLET PROVISION INSTALLATIONS)

A–A

L59852 S0006594200_V1

Chem-Mill Thicknesses for Figure 2, Item [6]
Figure 3 (Sheet 3 of 4)

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(FOR CUM LINE NUMBER 1 THRU 777)

L59853 S0006594201_V1

Chem-Mill Thicknesses for Figure 2, Item [6]
Figure 3 (Sheet 4 of 4)

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ALLOWABLE DAMAGE 1 - WING INSPAR SKIN INBOARD OF WBL 616.75

1. Applicability

A. This subject gives the allowable damage limits for the wing inspar skin inboard of the splice at WBL 616.75 shown in Wing Inspar Skin Location, Figure 101/ALLOWABLE DAMAGE 1.

B. Allowable Damage 1 is applicable to airplanes that have not had winglets installed.

C. Allowable Damage 1 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

NOTE: Allowable Damage 1 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G44352 S0006594205_V1

Wing Inspar Skin Location
Figure 101

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Refer to Table 101 or Table 102 and Paragraph 4.A for the allowable damage limits for the upper inspar skin.

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D. Refer to Table 103 or Table 104 and Paragraph 4.B for the allowable damage limits for the lower inspar skin.

E. Refer to Figure 103 (Sheet 1) for the definitions of skin areas.

F. If you have damage at a fuel tank door, do as follows before you remove the damage.

WARNING: IF YOU DO NOT SUPPLY GOOD AIRFLOW YOU WILL HAVE DANGEROUS AND EXPLOSIVE FUEL VAPORS IN THE WORK AREA WHICH CAN CAUSE INJURY TO PERSONS.

(1) Remove the fuel from the fuel tanks and make sure there is a good airflow in the damage area. Refer to AMM 28-11-00 and AMM 28-23-00 for the procedures.

(2) Remove the fuel tank access door. Refer to AMM 28-11-11.

G. Remove the damage as necessary. Refer to 51-10-02.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

H. After you remove the damage, do the procedures that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the skins of the lower inspar and fuel tank or vent door surrounds.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(2) If you removed a fastener on the lower inspar skin that does not attach to the inspar rib, backup fitting, or nacelle support fitting, then do as follows:

(a) Cold Work the fastener hole with the High Interference Cold Work method. Refer to 51-40-09.

NOTE: Do not Cold Work the fastener hole on the lower inspar skin at an inspar rib, backup fitting, or nacelle support fitting. Refer to 51-40-02 for fastener installation procedures.

(3) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(4) Apply the finishes to the surfaces of the reworked areas as follows.

NOTE: Make sure the aerodynamic smoothness on the external surfaces is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

WARNING: DO NOT APPLY A FINISH TO THE ELECTRICAL BOND SURFACE OF THE SKIN AT THE FUEL TANK ACCESS OR VENT DOOR SURROUND. IF THERE IS PRIMER ON THE ELECTRICAL BOND SURFACE OF THE DOOR, YOU MUST REMOVE THE PRIMER. THE ELECTRICAL BOND IS NECESSARY TO PREVENT POSSIBLE LIGHTNING DAMAGE AT THE FUEL TANK.

(a) Apply one layer of BMS 10-79, Type III primer. Refer to SOPM 20-44-04.

(b) Apply one layer of BMS 10-60, Type II enamel to the external skin surface. Refer to AMM 51-21-00/701.

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(c) Apply one layer of BMS 10-20, Type II primer to the internal skin surface. Refer to AMM 51-21-00/701.

Table 101:

ALLOWABLE DAMAGE LIMITS FOR THE UPPER SKIN FOR AIRPLANES THAT DO NOT HAVE WINGLETS											
ZONE (WING STATION)	MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR										TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO REAR SPAR (SQ. IN.)
	REAR SPAR		STIFFENER NO. 1 THROUGH STIFFENER NO. 6		STIFFENER NO. 7 THROUGH STIFFENER NO. 11		STIFFENER NO. 12 THROUGH STIFFENER NO. 21		FRONT SPAR		
	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	
WBL71-WS204	0.02	0.04	0.02	0.02	0.02	0.02	0.00	0.00	0.02	0.02	0.15
WS204-WS253	0.01	0.08	0.00	0.00	0.00	0.00	0.02	0.02	0.04	0.02	0.10
WS253-WS326	0.01	0.07	0.02	0.02	0.02	0.02	0.01	0.02	0.03	0.02	0.20
WS326-WS432	0.01	0.06	0.01	0.02	0.00	0.00	0.01	0.01	0.03	0.02	0.10
WS432-WS537	0.01	0.05	0.00	0.00	0.00	0.00	0.00	0.00	0.02	0.02	0.05
WS537-WS617	0.01	0.03	---	--	0.01	0.01	0.00	0.00	0.02	0.02	0.05
WS617-WBL616.7	0.01	0.02	---	--	0.01	0.02	0.01	0.02	0.01	0.02	0.05

Table 102:

ALLOWABLE DAMAGE LIMITS FOR THE UPPER SKIN FOR AIRPLANES THAT HAVE WINGLETS											
ZONE (WING STATION)	MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR										TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO REAR SPAR (SQ. IN.)
	REAR SPAR		STIFFENER NO. 1 THROUGH STIFFENER NO. 6		STIFFENER NO. 7 THROUGH STIFFENER NO. 11		STIFFENER NO. 12 THROUGH STIFFENER NO. 21		FRONT SPAR		
	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	
WBL71-WS204	0.02	0.04	0.01	0.01	0.01	0.02	0.00	0.00	0.02	0.02	0.15

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Table 102: (Continued)

ALLOWABLE DAMAGE LIMITS FOR THE UPPER SKIN FOR AIRPLANES THAT HAVE WINGLETS											
ZONE (WING STATION)	MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR										TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO REAR SPAR (SQ. IN.)
	REAR SPAR		STIFFENER NO. 1 THROUGH STIFFENER NO. 6		STIFFENER NO. 7 THROUGH STIFFENER NO. 11		STIFFENER NO. 12 THROUGH STIFFENER NO. 21		FRONT SPAR		
	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	
WS204-WS253	0.01	0.08	0.00	0.00	0.00	0.00	0.02	0.02	0.04	0.02	0.10
WS253-WS326	0.01	0.07	0.00	0.00	0.02	0.02	0.01	0.02	0.03	0.02	0.20
WS326-WS432	0.01	0.06	0.01	0.02	0.00	0.00	0.00	0.00	0.03	0.02	0.10
WS432-WS537	0.01	0.05	0.00	0.00	0.00	0.00	0.00	0.00	0.02	0.02	0.05
WS537-WS617	0.01	0.03	---	--	0.00	0.00	0.00	0.00	0.02	0.02	0.05
WS617-WBL616.7	0.01	0.02	---	--	0.01	0.02	0.01	0.02	0.01	0.02	0.05

Table 103:

ALLOWABLE DAMAGE LIMITS FOR THE LOWER SKIN FOR AIRPLANES THAT DO NOT HAVE WINGLETS									
ZONE (WING STATION)	MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR								TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO) REAR SPAR (SQ. IN.)
	REAR SPAR		STIFFENER NO. 1 THROUGH STIFFENER NO. 6		STIFFENER NO. 8 THROUGH STIFFENER NO. 14		FRONT SPAR		
	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	
WBL71-WS204	0.03	0.06	0.02	0.04	0.02	0.07	0.02	0.03	0.40
WS204-WS253	0.05	0.10	0.03	0.10	0.03	0.10	0.06	0.09	0.40
WS253-WS326	0.04	0.08	0.03	0.02	0.03	0.10	0.04	0.06	0.30
WS326-WS432	0.03	0.06	0.02	0.03	0.03	0.03	0.02	0.03	0.10

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Table 103: (Continued)

ALLOWABLE DAMAGE LIMITS FOR THE LOWER SKIN FOR AIRPLANES THAT DO NOT HAVE WINGLETS									
ZONE (WING STATION)	MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR								TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO) REAR SPAR (SQ. IN.)
	REAR SPAR		STIFFENER NO. 1 THROUGH STIFFENER NO. 6		STIFFENER NO. 8 THROUGH STIFFENER NO. 14		FRONT SPAR		
	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	
WS432-WS537	0.02	0.05	0.02	0.05	0.00	0.00	0.00	0.00	0.05
WS537-WS617	0.01	0.03	0.01	0.03	0 .00	0.00	0.00	0.00	0.05
WS617-WBL616.7	0.01	0.02	0.01	0.02	0.01	0.02	0.01	0.02	0.05

Table 104:

ALLOWABLE DAMAGE LIMITS FOR THE LOWER SKIN FOR AIRPLANES THAT HAVE WINGLETS									
ZONE (WING STATION)	MAXIMUM DEPTH OF DAMAGE AFTER CLEANUP AND MAXIMUM LOSS OF CROSS-SECTIONAL AREA AT A STIFFENER OR SPAR								TOTAL MAXIMUM LOSS OF CROSS-SECTIONAL AREA FROM FRONT SPAR TO) REAR SPAR (SQ. IN.)
	REAR SPAR		STIFFENER NO. 1 THROUGH STIFFENER NO. 6		STIFFENER NO. 8 THROUGH STIFFENER NO. 14		FRONT SPAR		
	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	MAX DEPTH (INCH)	MAX AREA LOSS (SQ. IN.)	
WBL71-WS204	0.03	0.06	0.02	0.04	0.01	0.07	0.02	0.03	0.40
WS204-WS253	0.05	0.10	0.03	0.10	0.03	0.10	0.06	0.09	0.40
WS253-WS326	0.04	0.08	0.03	0.02	0.03	0.10	0.04	0.06	0.30
WS326-WS432	0.03	0.06	0.02	0.03	0.03	0.03	0.02	0.03	0.10
WS432-WS537	0.02	0.05	0.02	0.05	0.00	0.00	0.00	0.00	0.05
WS537-WS617	0.01	0.03	0.01	0.03	0 .00	0.00	0.00	0.00	0.05
WS617-WBL616.7	0.01	0.02	0.00	0.00	0.00	0.00	0.01	0.02	0.02

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Table 105:

INITIAL FASTENER	MAXIMUM DEPTH OF MATERIAL REMOVAL FROM A FASTENER HEAD (INCH)			
	3/16 DIA	1/4 DIA	5/16 DIA	3/8 DIA
ALUMINUM RIVET WITH 82°/30° COUNTERSINK	0.01	0.01	0.01	0.01
BACB30VU()K() BACB30NW()K()	0.00	0.00	0.00	0.00
BACB30PT()K()	0.00	0.00	0.00	0.00

Table 106:

INITIAL FASTENER	FASTENER	MINIMUM THICKNESS OF THE SKIN THAT REMAINS AFTER CLEANUP	REPLACEMENT FASTENER
ALUMINUM RIVET WITH 82°/30° COUNTERSINK	3/16	0.12	BACB30NY6K()Y BOLT WITH A BACC30AC6 COLLAR
	1/4	0.20	BACB30NY8K()Y BOLT WITH A BACC30AC8 COLLAR
	5/16	0.20	BACB30NY10K()Y BOLT WITH A BACC30AC10 COLLAR
	3/8	0.24	BACB30NY12K()Y BOLT WITH A BACB30AC12 COLLAR
BACB30VU()K() BOLT WITH A BACC30BP COLLAR BACB30NW()K() BOLT WITH A BACC30M COLLAR	3/16	0.08	BACB30NW6K()X BOLT WITH A BACC30M6 COLLAR
	1/4	0.09	BACB30NW8K()X BOLT WITH A BACC30M8 COLLAR
	5/16	0.10	BACB30NW10K()X BOLT WITH A BACC30M10 COLLAR
	3/8	0.12	BACB30NW12K()X BOLT WITH A BACC30M12 COLLAR
BACB30PT()K() BOLT WITH A BACN10WM NUT	3/16	0.08	BACB30NW6K()X BOLT WITH A BACC30AC6 COLLAR
	1/4	0.09	BACB30NW8K()X BOLT WITH A BACC30AC8 COLLAR
	5/16	0.10	BACB30NW10K()X BOLT WITH A BACC30AC10 COLLAR
	3/8	0.12	BACB30NW12K()X BOLT WITH A BACC30AC12 COLLAR

NOTE: Refer to Table 107 if the minimum thickness of the skin that remains is less than is given for an aluminum rivet with a 82°/30°countersink.

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LEFT SIDE UPPER SKIN IS SHOWN,
RIGHT SIDE IS OPPOSITE

NOTE: THE UPPER INSPAR WING SKIN IS MADE OF MACHINED ALUMINUM.

G45276 S0006594210_V1

Wing Inspar Skin Zones
Figure 102 (Sheet 1 of 4)

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LEFT SIDE LOWER SKIN IS SHOWN,
RIGHT SIDE IS OPPOSITE

NOTE: THE LOWER INSPAR WING SKIN IS MADE OF MACHINED ALUMINUM.

G45280 S0006594211_V1

Wing Inspar Skin Zones
Figure 102 (Sheet 2 of 4)

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DAMAGE IS PERMITTED. REFER TO TABLES 101, 102, 103, OR 104 AND PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

DAMAGE IS NOT PERMITTED IN THIS AREA IF THE MAXIMUM DAMAGE DEPTH IS MORE THAN 0.02 INCH (0.51 mm).

SKIN ZONES AT A REAR SPAR AND RIB AREA
(TYPICAL)

A

426859 S0000139449_V2

Wing Inspar Skin Zones
Figure 102 (Sheet 3 of 4)

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DAMAGE IS PERMITTED. REFER TO TABLE 102 AND PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS ON THE SKIN. REFER TO SRM 57-20-01, REPAIR 1 FOR THE ALLOWABLE DAMAGE LIMITS ON THE FUEL TANK ACCESS DOOR SURROUND EDGE.

DAMAGE IS NOT PERMITTED

DAMAGE IS PERMITTED. REFER TO SRM 57-20-01, REPAIR 1 FOR THE ALLOWABLE DAMAGE LIMITS.

LOWER SKIN ZONES BETWEEN STIFFENER 6 AND STIFFENER 8
(FUEL TANK ACCESS DOOR AREA IS SHOWN)
(TYPICAL)

B

426860 S0000139453_V3

Wing Inspar Skin Zones
Figure 102 (Sheet 4 of 4)

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G45454 S0006594214_V1

Definitions of Skin Areas Used to Find the Allowable Damage Limits at a Stiffener or a Spar
Figure 103

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
51-40-09	COLD WORKING OF HOLES FOR FATIGUE IMPROVEMENT
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
AMM 28-11-00	Integral Fuel Tanks
AMM 28-11-11	Wing Fuel Tank Access Panel Removal/Installation
AMM 28-23-00	Fuel Tank Refueling
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-04-00	Ultrasonic
737 NDT Part 6, 51-00-00	Structures - General
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Allowable Damage Limits

A. Upper Skin (Except at Fastener Locations)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and G .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Lower Skin (Except at Fastener Locations)

(1) Cracks:

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(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and G .

(b) Remove the damage at the fuel tank access or vent door surround as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Upper and Lower Skin at a Fastener Location

(1) Refer to Table 106 to make sure you have a replacement fastener. Do not remove the damage if you do not have a replacement fastener.

(2) Remove the damage at an aluminum or titanium fastener.

NOTE: You must remove steel fasteners before you remove the damage. Do the procedures given in Paragraph 4.C.(3) for damage at a steel fastener.

(a) Sand or grind the aluminum fastener and the damaged skin around the fastener as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A .

CAUTION: DO NOT GRIND INTO A TITANIUM FASTENER. SMALL PARTICLES OF TITANIUM ARE FLAMMABLE. SAND INTO THE FASTENER MANUALLY OR WITH TOOLS. REFER TO SRM 51-10-02.

(b) Sand the titanium fastener and the damaged skin around the fastener as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(c) Do an inspection of the damage removal area.

1) Do a 10X visual inspection to make sure the damage is removed from under the surface.

2) Do an ultrasonic inspection to make sure the damage is removed from under the skin surface. Refer to 737 NDT Part 1, 51-04-00.

(3) If you have one of these three conditions, you must remove the fastener and the damage around the fastener area as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail B .

NOTE: Do not remove more than 75 percent of the fasteners at one time from skin attached to a structure.

(a) The fastener is made of steel.

(b) The damage is not fully removed from the aluminum or titanium fastener.

(c) The material removed from the aluminum or titanium fastener is more than is given in Table 105 .

(4) If, and after you remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail B , do an inspection of the area.

(a) Do an ultrasonic inspection to make sure the damage is removed from under the surface. Refer to 737 NDT Part 1, 51-04-00.

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(b) Do a High Frequency Eddy Current (HFEC) inspection of the fastener hole to make sure there are no cracks. Refer to 737 NDT Part 6, 51-00-00, Figure 4 .

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(5) Shot peen the damage area.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(6) If you removed the fastener, do as follows.

(a) Drill and countersink the fastener hole. Refer to FASTENER HOLE SIZES, 51-40-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(b) Cold Work the fastener if it is in the lower skin.

1) Use the High Interference Cold Work method. Refer to 51-40-09.

(c) Replace the fastener you removed with the applicable fastener given in Table 106 or Table 107.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

Table 107:

INITIAL FASTENER	FASTENER	MINIMUM THICKNESS OF THE SKIN THAT THAT REMAINS AFTER CLEANUP	REPLACEMENT FASTENER
ALUMINUM RIVET WITH 82°/30° COUNTERSINK	3/16	0.08	BACB30NW6K()Y BOLT WITH A BACC30AC6 COLLAR
	1/4	0.09	BACB30NW8K()Y BOLT WITH A BACC30R8 COLLAR
	5/16	0.10	BACB30NW10K()Y BOLT WITH A BACC30R10 COLLAR
	3/8	0.12	BACB30NW12K()Y BOLT WITH A BACC30R12 COLLAR

NOTE: Use the data in this table only if the minimum thickness of the skin that remains after cleanup is less than is given in Table 106 for the initial fastener given.

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REMOVAL OF DAMAGED MATERIAL AT
EDGES WHERE THE FASTENER EDGE
MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT
EDGES WHERE THE FASTENER EDGE
MARGINS HAVE AN OVERLAP

B

G58732 S0006594216_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 7)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH (2.54 mm)

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN

C

G58733 S0006594217_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 7)

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REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D

A–A

426863 S0000139457_V3

Allowable Damage Limits
Figure 104 (Sheet 3 of 7)

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NOTE: REFER TO FIGURE 104, DETAIL G AND PARAGRAPH 5, TO CALCULATE THE CROSS SECTIONAL AREA THAT IS REMOVED.

QUANTITY OF DAMAGE IN A CROSS SECTIONAL AREA AT A STIFFENER OR A SPAR

E

G58755 S0006594219_V1

Allowable Damage Limits
Figure 104 (Sheet 4 of 7)

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NOTE: REFER TO FIGURE 104, DETAIL G AND PARAGRAPH 5, TO CALCULATE THE CROSS SECTIONAL AREA THAT IS REMOVED.

1 INCLUDE THIS DAMAGE AS THE TOTAL DAMAGE FROM THE FRONT SPAR TO THE REAR SPAR.

QUANTITY OF DAMAGE BETWEEN THE FRONT SPAR AND THE REAR SPAR

F

G58754 S0006594220_V1

Allowable Damage Limits
Figure 104 (Sheet 5 of 7)

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REMOVAL OF DAMAGED MATERIAL AT STIFFENERS AND BETWEEN SPARS
(REFER TO PARAGRAPH 5 FOR AN EXAMPLE)

G

B–B

419075 S0000137382_V1

Allowable Damage Limits
Figure 104 (Sheet 6 of 7)

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ELECTRICAL BOND LIMITS FOR FUEL TANK ACCESS
OR VENT DOOR SURROUND SKIN, CLAMP RING, AND DOOR STRUCTURE

H

NOTES

2 REMOVAL OF IMPRESSIONS CAUSED BY THE KNITTED ALUMINUM GASKET IS NOT NECESSARY IF THE IMPRESSIONS ARE NOT MORE THAN 0.005 INCH (0.127 mm) DEEP. THIS IS APPLICABLE TO THE FUEL TANK ACCESS DOORS, CLAMP RINGS AND THE MILLED STEP OF THE SKIN.

D–D

G58771 S0006594223_V2

Allowable Damage Limits
Figure 104 (Sheet 7 of 7)

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THE REMOVAL OF MATERIAL AROUND THREE FASTENERS IN ALL GROUPS OF TEN IS PERMITTED TO A MAXIMUM DEPTH OF X

NOTE: FASTENERS ON A SURFACE ARE SHOWN. FASTENERS AT AN EDGE ARE APPLICABLE.

REMOVAL OF DAMAGE AT A FASTENER LOCATION WITH THE FASTENER INSTALLED

(A)

MAKE THE TAPER TO A MINIMUM OF 10X FOR THE UPPER SKIN AND A MINIMUM OF 20X FOR THE LOWER SKIN

REMOVE THE MATERIAL TO A MINIMUM RADIUS OF 1.0 INCH (25.4 mm), THEN TAPER AS SHOWN

X = THE MAXIMUM DEPTH OF MATERIAL REMOVED (REFER TO TABLE 101 OR TABLE 102 FOR THE UPPER SKIN OR TABLE 103 OR TABLE 104 FOR THE LOWER SKIN)

MAKE IT SMOOTH (TYPICAL)

A–A

THE REMOVAL OF MATERIAL AROUND THREE FASTENERS IN ALL GROUPS OF TEN IS PERMITTED TO A DEPTH OF X MAXIMUM

REMOVE THE MATERIAL TO A 0.50 INCH (12.7 mm) RADIUS MINIMUM (TYPICAL)

REMOVE THE INITIAL FASTENER BEFORE THE DAMAGED MATERIAL IS REMOVED. (REFER TO TABLE 106 FOR A REPLACEMENT FASTENER AFTER ALL THE DAMAGE IS REMOVED)

REMOVAL OF DAMAGE AT A FASTENER LOCATION WITH THE FASTENER REMOVED

(B)

REMOVE THE MATERIAL TO A MINIMUM RADIUS OF 1.00 INCH (25.4 mm), THEN TAPER AS SHOWN

X = THE MAXIMUM DEPTH OF MATERIAL REMOVED. (REFER TO TABLE 101 OR TABLE 102 FOR THE UPPER SKIN OR TABLE 103 OR TABLE 104 FOR THE LOWER SKIN)

MAKE IT SMOOTH (TYPICAL)

FASTENER HOLE ℄

FLAT AREA FOR THE REPLACEMENT FASTENER

Y = THE MINIMUM THICKNESS OF MATERIAL THAT REMAINS (REFER TO TABLE 106 FOR THE DIMENSION)

NOTE: IF Y IS LESS THAN IS GIVEN FOR AN ALUMINUM RIVET WITH A 30° TO 82° COUNTERSINK, REFER TO TABLE 107 FOR THE DIMENSION.

B–B

426870 S0000139461_V2

Allowable Damage Limits
Figure 105

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5. **Example to Calculate the Cross-Sectional Area Removed**

A. Calculate the damage depths and areas shown in Allowable Damage Limits, Figure 104/ ALLOWABLE DAMAGE 1, Detail G to find if the damage is permitted.

NOTE: You can make a conservative estimate of the area loss as described below if you can not accurately calculate the area loss. In the example below, the cross sectional area loss is estimated to be 1/2 the width X depth.

(1) Find the allowable damage depth (Y) for each damage location shown.

(a) The damage depth at Damage A is 0.01 inch (0.25 mm).

1) This is less than 0.02 inch (0.51 mm) and is permitted.

(b) The damage depth at Damage B is 0.01 inch (0.25 mm).

1) This is less than 0.02 inch (0.51 mm) and is permitted.

(c) The damage depth at Damage C is 0.015 inch (0.38 mm).

1) This is less than 0.02 inch (0.51 mm) and is permitted.

(2) Calculate the cross sectional area that is decreased between the stringers, then find if the decreased area is permitted.

(a) The area that has decreased at Damage A is:

1) 2.20 inches/2 X 0.01 inch = 0.01 $inch^2$ (7.10 mm^2)

a) This is less than 0.02 $inch^2$ (12.90 mm^2) and is permitted.

(b) The area that has decreased at Damage B is:

1) 2.40 inches/2 X 0.01 inch = 0.012 $inch^2$ (7.74 mm^2)

a) This is less than 0.02 $inch^2$ (12.90 mm^2) and is permitted.

(c) The area that has decreased at Damage C is:

1) 2.50 inches/2 X 0.015 inch = 0.019 $inch^2$ (12.26 mm^2)

a) This is less than 0.02 $inch^2$ (12.90 mm^2) and is permitted.

(3) Calculate the total cross sectional area that has decreased between the front and rear spars. After you do this, find if that decreased area is permitted.

(a) Add the areas that have decreased in Damage A, B, and C as the total area decreased between the front and rear spars.

1) [0.01 + 0.012 + 0.019] $inch^2$ = 0.041 $inch^2$ (26.45 mm^2).

a) This is less than 0.20 $inch^2$ (129.03 mm^2) and is permitted.

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ALLOWABLE DAMAGE 2 - WING INSPAR SKIN OUTBOARD OF WBL 616.75

1. Applicability

A. This subject gives the allowable damage limits for the wing inspar skin outboard of WBL 616.75 shown in Wing Inspar Skin Location, Figure 101/ALLOWABLE DAMAGE 2.

B. Allowable Damage 2 is applicable to airplanes that have not had winglets installed.

C. Allowable Damage 2 is not applicable to:

(1) Airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets.

(2) Airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Wing Inspar Skin Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G44982 S0006594227_V1

Wing Inspar Skin Outboard of the Splice at WBL 616.75
Figure 102 (Sheet 1 of 2)

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THE ACCESS DOOR SURROUND IN THE LOWER HONEYCOMB SKIN IS SHOWN

A

A–A

ACCESS DOOR SURROUND
(ALUMINUM SKIN)

G48517 S0006594228_V1

Wing Inspar Skin Outboard of the Splice at WBL 616.75
Figure 102 (Sheet 2 of 2)

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

(1) Apply a chemical conversion coating to the bare surfaces of the skin. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the internal surfaces of the reworked skins. Refer to SOPM 20-41-02.

(3) Apply a layer of BMS 10-79, Type III primer to the external surfaces of the reworked skins. Refer to SOPM 20-44-04.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and G .

1) Only one damage removal area is permitted for each 15 square inches (9679 square mm) of skin surface area.

(3) Dents:

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(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 2, Detail E if:

1) It is not more than 15 square inches (9679 square mm) for each square foot (0.09 square m) of skin surface area

2) The distance away from a hole, other damage, or part edge is 4D (D= the largest dimension of the damage).

(b) If the damage is more than the above conditions, then you are permitted to fill or rework the dent as given in 51-70-01.

1) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval, or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted in the outer facesheet of the honeycomb area up to 0.25 inch (6.35 mm) in diameter if:

1) It is a minimum of 1.0 inch (25.4 mm) away from a hole, other damage, or part edge

2) There is not more than one damage area for each 15 square inches (9679 square mm) of skin surface area

3) You make a temporary seal

a) Apply aluminum foil tape (speed tape).

b) Keep a record of the location.

c) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

d) Repair the damage at or before 5000 flight hours from the time the seal was made.

(b) The damage is not permitted in the edgebands.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail F .

B. Lower Skin - Access Door Surround (Aluminum Skin)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details C and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

G45019 S0006594229_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 5)

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Allowable Damage Limits
Figure 103 (Sheet 2 of 5)

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 2 INCHES (50.8 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 3d.

DELAMINATION THAT IS PERMITTED TO HONEYCOMB PANELS

F

G45193 S0006594231_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 5)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH (2.54 mm)

REMOVAL OF DAMAGED MATERIAL
AT AN EDGE OF A METAL SKIN

G

G48855 S0006594232_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 5)

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REMOVAL OF DAMAGED MATERIAL AT THE EDGE
OF THE ACCESS DOOR SURROUND

H

TYPICAL BLENDOUT AT ACCESS DOOR SURROUND
C–C

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Allowable Damage Limits
Figure 103 (Sheet 5 of 5)

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REPAIR 1 - WING SKIN LOWER SURFACE - CORROSION REPAIR AT THE FUEL TANK ACCESS PANEL CLAMP RING

1. Applicability

A. This repair is applicable to corrosion damage at:

(1) The edge of the wing skin at the opening for all of the fuel tank access doors.

(2) The machined surface on the wing skin at all door locations.

(3) The machined surface on the fuel tank access door, excluding the high impact main and center tank doors and the vented surge tank doors.

(4) The fuel tank access panel clamp ring common to the access door and the lower skin.

B. This repair is applicable to airplanes that have had winglets installed as part of the production of the airplane. Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets.

C. This repair is applicable to airplanes that have the Aviation Partners Boeing (APB) Supplemental Type Certificate (STC) ST00830SE blended winglet (not the split scimitar configuration) installed on the airplane, except for airplane line numbers 779, 787, and 836.

2. General

A. This repair is a Category A damage tolerant repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06.

Refer to AMM SUBJECT 28-10-00 for the location of the fuel tank access doors.

B. Make sure the aerodynamic smoothness around the fuel tank access doors is satisfactory or there will be a loss in airplane performance. Refer to AERODYNAMIC SMOOTHNESS, 51-10-01.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-06, GENERAL	Shot Peening
AMM 28-10-00	STORAGE
AMM 28-11-00 P/B 701	FUEL TANKS - CLEANING/PAINTING
AMM 28-11-11 P/B 401	WING FUEL TANK ACCESS PANELS - REMOVAL/INSTALLATION
AMM 28-11-31 P/B 401	CENTER TANK ACCESS DOOR - REMOVAL/INSTALLATION
AMM 28-26-00 P/B 201	DEFUELING - MAINTENANCE PRACTICES
SOPM 20-10-03	SHOT PEENING

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS HAVE GOOD FLOW OF CLEAN AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS ON, OR NEAR THE AIRPLANE.

A. Remove the fuel and fuel tank access door(s).

(1) Defuel all fuel tanks. Refer to AMM PAGEBLOCK 28-26-00/201.

(2) Do what follows, as applicable.

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(a) Do the main fuel tank and surge tank access door removal procedures. Refer to AMM PAGEBLOCK 28-11-11/401.

(b) Do the center tank access door removal procedure. Refer to AMM PAGEBLOCK 28-11-31/401.

B. Clean the area of the repair with the steps that follow.

(1) Clean the edge of the skin located at the opening for the fuel tank access door.

(2) Clean the access door.

(3) Clean the surfaces of the skin that touch the clamp ring.

C. Remove and discard the initial knitted aluminum gasket.

D. Remove the damaged material. Refer to 51-10-02, GENERAL.

(1) Remove the damage from the edge of the skin at the opening for the fuel tank access door as shown in Figure 201/REPAIR 1 and Figure 202/REPAIR 1.

(2) Remove the damage from the surface of the machined step in the skin and in the fuel tank access door as shown in Figure 201/REPAIR 1 and Figure 203/REPAIR 1.

(3) Remove the damage from the clamp ring as shown in Figure 205/REPAIR 1.

E. Do a 10X visual inspection to make sure that you removed all the damage.

F. Make the surface roughness of the damage removal area as follow:

(1) 63 microinches Ra or smoother for the skin.

(2) 125 microinches Ra or smoother for the door area and the clamp ring.

NOTE: Do not fill the areas where you removed material.

NOTE: CDCCL - Refer to the task: Airworthiness Limitation Precautions, TASK 28-00-00-910-801, for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-16.

CAUTION: DO NOT USE FILLERS OR OTHER MATERIALS THAT DO NOT MEET THE SPECIFICATIONS FOR THE REPAIR. REPAIRS THAT DO NOT MEET THE BLEND REQUIREMENTS COULD CAUSE A REDUCTION IN THE INITIAL ELECTRICAL BOND SURFACE AREA.

G. Do a visual inspection on the electrical bond surface area.

(1) If you cannot meet the blending requirements specified in this repair or you use non-conductive fillers, you cannot remove more than 50% of the initial electrical bond surface area. If you do meet the blending requirements specified in this repair, you may remove up to 100% of the initial electrical bond surface area.

(2) The initial bond surface area is the non-primed area that touches the gasket. Refer to Figure 201/REPAIR 1.

(3) The electrical bond surface area that remains is the non-primed area that touches the gasket where both the door and the skin under it are not damaged.

(4) The electrical bond is necessary to prevent lightning conducted current arcing in the fuel tank.

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(5) If the damage of the initial electrical bond surface area is more than 50% and you cannot meet the blending requirements specified in this repair, contact The Boeing Company. Refer to Figure 201/REPAIR 1 and Figure 204/REPAIR 1.

NOTE: CDCCL - Refer to the task: Airworthiness Limitation Precautions, TASK 28-00-00-910-801, for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-16.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

H. Flap peen or shot peen the areas of the damage removal.

(1) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(2) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

WARNING: DO NOT BREATHE THE FUMES OR ALLOW THE SOLVENTS TO TOUCH YOUR SKIN, EYES, OR CLOTHING. MAKE SURE THAT THE AIR SUPPLY TO THE AREA IS NOT BLOCKED. IF YOU DO NOT OBEY, YOU CAN CAUSE SKIN IRRITATION OR INJURY.

DO NOT USE SOLVENTS IN AREAS WITH EQUIPMENT THAT CAN PRODUCE HEAT OR SPARKS. IF YOU DO NOT OBEY, AN EXPLOSION CAN BE THE RESULT.

I. Clean the repair with a soft cloth that is moist with Series 88 solvent. Remove all of the contamination.

J. Apply a chemical conversion coating to the areas where you removed material. Refer to 51-20-01, GENERAL.

CAUTION: MAKE SURE THAT THE AREA IS FREE FROM PRIMER. IF PRIMER IS IN THESE AREAS AN UNSATISFACTORY BOND CAN BE THE RESULT.

K. Apply one layer of BMS 10-20, Type II primer to the repair area. Refer to Figure 201/REPAIR 1 and Figure 204/REPAIR 1. Refer to AMM PAGEBLOCK 28-11-00/701. Do not apply primer to:

(1) The lower surface of the access door

(2) The lower surface of the skin

(3) The upper surface of the clamp ring.

NOTE: If primer is in these areas, you must remove it.

NOTE: CDCCL - Refer to the task: Airworthiness Limitation Precautions, TASK 28-00-00-910-801, for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-16.

L. Install a new knitted aluminum gasket with part number 65C33095 or 65C33161. Install the gasket with the BMS 3-38, Aeroshell 14 grease or Mobilgrease 33 grease impregnated into the gasket continuously and evenly throughout the knit.

NOTE: CDCCL - Refer to the task: Airworthiness Limitation Precautions, TASK 28-00-00-910-801, for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-16.

M. Install the access door.

(1) Do the step that follow, as applicable:

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(a) Do the main fuel tank and surge tank access door installation procedure. Refer to AMM PAGEBLOCK 28-11-11/401.

(b) Do the center tank access door installation procedure. Refer to AMM PAGEBLOCK 28-11-31/401.

NOTE: To ensure good clamp-up during fuel door installation, it is acceptable to use fasteners with a shorter grip length to compensate for material removed during the repair.

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BOTTOM VIEW OF THE AREA
AROUND THE ACCESS DOOR

A

A–A

NOTES

- IF YOU NEED ALLOWABLE DAMAGE LIMITS FOR AREAS THAT ARE NOT SHOWN IN DETAIL A, GET INSTRUCTIONS FROM THE BOEING COMPANY.
- REFER TO FIGURE 204 FOR THE NECESSARY ELECTRICAL BOND SURFACE AREA.

1904251 S0000351052_V1

Fuel Tank Access Door
Figure 201

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BOTTOM VIEW
DAMAGE REMOVAL AT THE EDGE OF THE SKIN CUTOUT

A

A–A

NOTE: REFER TO FIGURE 204 FOR THE NECESSARY ELECTRICAL BOND SURFACE AREA.

1904258 S0000351054_V1

Damage Removal at the Edge of the skin Cutout
Figure 202

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NOTE: REFER TO FIGURE 204 FOR THE NECESSARY ELECTRICAL BOND SURFACE AREA.

1904261 S0000351055_V2

Damage Removal on the Milled Step in the Skin or Fuel Tank Access Door
Figure 203 (Sheet 1 of 2)

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MINIMUM THICKNESS REMAINING:

0.122 INCH (3.10 mm) IN THE MACHINED STEP IN THE SKIN AT THE 8 INCH BY 18 INCH DOOR LOCATIONS

0.150 INCH (3.81 mm) IN THE MACHINED STEP IN THE SKIN AT THE 10 INCH BY 18 INCH DOOR LOCATIONS

B–B
THE SURFACE OF THE MACHINED STEP IN THE SKIN

B–B
THE SURFACE OF THE MACHINED STEP IN
THE FUEL TANK ACCESS DOOR

NOTE: REFER TO FIGURE 204 FOR THE NECESSARY ELECTRICAL BOND SURFACE AREA.

2314696 S0000526386_V2

Damage Removal on the Milled Step in the Skin or Fuel Tank Access Door
Figure 203 (Sheet 2 of 2)

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NOTE: THE CLAMP RING IS NOT SHOWN.

BOTTOM VIEW OF THE WING

A

A–A

NOTES

- DO NOT APPLY THE PRIMER TO THE LOWER SURFACE OF THE SKIN OR ACCESS DOOR IN THE AREAS WITH THE DIMENSIONS SHOWN IN A–A.
- DO NOT APPLY THE PRIMER TO THE UPPER SURFACE OF THE CLAMP RING IN THE AREAS WITH THE DIMENSIONS SHOWN IN A–A.

1904263 S0000351059_V1

Requirements for Electrical Bonding
Figure 204

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1904267 S0000351058_V1

Damage Removal on the Clamp Ring
Figure 205

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IDENTIFICATION 1 - OUTER WING UPPER STRINGERS

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F73196 S0006594237_V1

Outer Wing Upper Stringers Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
110A1503	Wing Stringer Diagram, Upper 737-NG
112A3001	Panel Installation - Upper, Section 12, Wing
737–8010	Wing Reinforcement Mod, Winglet*[1]

*[1] Aviation Partners Boeing (APB) part as retrofitted by Supplemental Type Certificate (STC) ST00830SE.

3. Applicability

A. Identification 1 is applicable to all airplanes which include those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglets.

B. List of Materials

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
STRINGER	**DESCRIPTION**	**MATERIAL**	**DRAWING NUMBER**	**EFFECTIVITY**
U-1	Splice	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3201	
U-2	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3202	
U-3	Zee	BAC1518-1169 7055-T77511 Extrusion as given in BMS 7-308.	112A3203	
U-4	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3204	
U-5	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3205	
U-6	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3206	
U-7	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3207	
U-8	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3208	
U-9	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3209	
U-9 Reinforcement	Angle	7055-T77511 extrusion as given in BMS 7-308. Optional: 7055-T7751 plate as given in BMS 7-307. Optional: 7150-T6511 extrusion per BMS 7-257, Type 1. Optional: 7150-T6511 extruded bar per BMS 7-257, Type 1. Optional: 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)	737–8380	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.
U-10 and U-11	Vent	BAC1510-1313 7055-T77511 Extrusion as given in BMS 7-308.	112A3210	Cum Line 1 thru 777
	Vent	BAC1510-1378 7055-T77511 Extrusion as given in BMS 7-308.	112A3210	Cum Line 778 and on
U-10 Reinforcement	Angle	7055-T77511 extruded bar as given in BMS 7-308. Optional: 7150-T6511 extruded bar per BMS 7-257, Type 1. Optional: 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)	737–8390	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.
U-12 and U-13	Vent	BAC1510-1313 7055-T77511 Extrusion as given in BMS 7-308.	112A3212	Cum Line 1 thru 777
	Vent	BAC1510-1378 7055-T77511 Extrusion as given in BMS 7-308.	112A3212	Cum Line 778 and on
U-12 and U-13	Vent	BAC1510-1313 7055-T77511 Extrusion as given in BMS 7-308 (Optional: BAC1510-1378)	112A3212	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
STRINGER	**DESCRIPTION**	**MATERIAL**	**DRAWING NUMBER**	**EFFECTIVITY**
U-13 Reinforcement	Angle	7055-T77511 extruded bar as given in BMS 7-308. Optional: 7150-T6511 extruded bar per BMS 7-257, Type 1. Optional: 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)	737–8370	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.
U-14	Splice	BAC1518-1169 7055-T77511 Extrusion as given in BMS 7-308.	112A3214	
U-14 Reinforcement	Angle	7055-T77511 extrusion as given in BMS 7-308. Optional: 7055-T7751 plate as given in BMS 7-307. Optional: 7150-T6511 extrusion per BMS 7-257, Type 1. Optional: 7150-T6511 extruded bar per BMS 7-257, Type 1. Optional: 7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)	737–8360	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.
U-15	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3215	
U-16	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3216	
U-17	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3217	
U-18	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3218	
U-19	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3219	
U-20	Zee	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3220	
U-21	Splice	BAC1518-1152 7055-T77511 Extrusion as given in BMS 7-308.	112A3221	

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LEFT WING IS SHOWN, RIGHT WING IS OPPOSITE
PLAN VIEW OF OUTER WING UPPER SURFACE

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

1 STRINGER REINFORCEMENT INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE FOR UNPROVISIONED WING.

2277118 S0000513461_V1

Outer Wing Upper Stringers Identification
Figure 2 (Sheet 1 of 3)

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F73460 S0006594241_V1

Outer Wing Upper Stringers Identification
Figure 2 (Sheet 2 of 3)

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2277567 S0000513464_V1

Outer Wing Upper Stringers Identification
Figure 2 (Sheet 3 of 3)

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IDENTIFICATION 2 - OUTER WING LOWER STRINGERS

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F72916 S0006594243_V2

Outer Wing Lower Stringers Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
110A1504	Wing Stringer Diagram - Lower 737-NG
112A4001	Panel Installation - Lower, Section 12, Wing
737–8010	Wing Reinforcement Mod, Winglet*[1]

*[1] Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.

3. Applicability

A. Identification 2 is applicable to all airplanes which include those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglets.

B. List of Materials:

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
STRINGER	**DESCRIPTION**	**MATERIAL**	**DRAWING NUMBER**	**EFFECTIVITY**
L-1	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4201	
L-2	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4202	
L-3	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4203	
L-4	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4204	
L-5	Splice	BAC1506-4261 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-4563)	112A4205	
L-6	Zee	BAC1517-2869 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2731)	112A4206	For Cum Line 778 and on
L-6	Zee	BAC1517-2731 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2845 or BAC1517-2869)	112A4206	For Cum Line 1 thru 777
L-6 Reinforcement	Strap	301 Stainless steel as given in MIL-S-5059.	737–8480	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.
L-7	Splice	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4207	
L-8	Zee	BAC1517-2869 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2845 or BAC1517-2731)	112A4208	For Cum Line 778 and on
	Zee	BAC1517-2731 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2845 or BAC1517-2869)	112A4208	For Cum Line 1 thru 777
L-8 Reinforcement	Strap	301 Stainless steel as given in MIL-S-5059.	737–8460	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE for unprovisioned wing.
L-9	Splice	BAC1506-4563 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-4261)	112A4209	For Cum Line 778 and on
	Splice	BAC1506-4261 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-4563)	112A4209	For Cum Line 1 thru 777

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
STRINGER	DESCRIPTION	MATERIAL	DRAWING NUMBER	EFFECTIVITY
L-10	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4210	
L-11	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4211	
L-12	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4212	
L-13	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4213	
L-14	Zee	BAC1517-2703 2224-T3511 extrusion as given in BMS 7-255 (Optional: BAC1517-2846 or BAC1517-2868)	112A4214	

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LEFT WING IS SHOWN,
RIGHT WING IS OPPOSITE

PLAN VIEW OF OUTER WING LOWER SURFACE

NOTES

1 STRINGER REINFORCEMENT INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE FOR UNPROVISIONED WING.

2277124 S0000513465_V1

Outer Wing Lower Stringers Identification
Figure 2 (Sheet 1 of 3)

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F73222 S0006594247_V1

Outer Wing Lower Stringers Identification
Figure 2 (Sheet 2 of 3)

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2277605 S0000513482_V1

Outer Wing Lower Stringers Identification
Figure 2 (Sheet 3 of 3)

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REPAIR 1 - REPAIR FOR DAMAGE TO AN UPPER SPLICE STRINGER ON THE OUTER WING

1. Applicability

A. Repair 1 is applicable to damage to upper splice stringers on the outer wing. A typical upper splice stringer is shown in Figure 201 (Sheet 1).

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 1 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 1. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for repair categories and definitions.

F63927 S0006594250_V1

Typical Upper Splice Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS

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(Continued)

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
57-20-01	OUTER WING SKIN
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling
SOPM 20-10-03	SHOT PEENING

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 1. Refer to 51-10-02.

NOTE: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

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(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 1.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Channel	1	Use 7075 in the annealed condition that is 0.050 inch thick. Heat treat to the T6 condition after you bend the part
[2]	Angle	1	Use 7075 in the annealed condition that is 0.050 inch thick. Heat treat to the T6 condition after you bend the part
[3]	Channel	1	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[4]	Angle	1	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[5]	Angle	1	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[6]	Plate	2	Use 7075-T6 that is 0.190 inch thick
[7]	Filler	1	Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

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(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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NOTE: UPPER WING SKIN NOT SHOWN.

F61549 S0006594252_V1

Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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A–A

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NW()K()Y HEX DRIVE BOLT AND A BACC30R COLLAR. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30MY9K() HEX DRIVE BOLT AND A BACC30M COLLAR.

F61550 S0006594253_V2

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do the inspection of the stringer skin attachment flanges from outside the wing tank at the stringer repair area.

(a) Do the inspection of the skin attachment flange common to the repair parts, as shown in Figure 203/REPAIR 1.

(2) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the same inspection again at each 36,000 cycle interval.

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NOTE: UPPER WING SKIN NOT SHOWN.

DO THE LFEC INSPECTION OF STRINGER IN THIS AREA

1840193 S0000324264_V1

Inspection Instructions
Figure 203 (Sheet 1 of 2)

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1840143 S0000324265_V1

Inspection Instructions
Figure 203 (Sheet 2 of 2)

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REPAIR 2 - REPAIR FOR DAMAGE TO AN UPPER ZEE STRINGER ON THE OUTER WING

1. Applicability

A. Repair 2 is applicable to damage to upper zee stringers on the outer wing. A typical upper zee stringer section is shown in Figure 201/REPAIR 2.

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 2 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 2. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for repair categories and definitions.

F63930 S0006594255_V1

Typical Upper Zee Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS

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(Continued)

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
57-20-01	OUTER WING SKIN
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling
SOPM 20-10-03	SHOT PEENING

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 2. Refer to 51-10-02.

NOTE: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

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(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 2.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	2	Use 7075 in the annealed condition that is 0.050 inch thick. Heat treat to the T6 condition after you bend the part
[2]	Angle	2	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[3]	Strap	1	Use 7075-T6 that is 0.190 inch thick
[4]	Filler	1	Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

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(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [4] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 2.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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NOTE: THE UPPER WING SKIN IS NOT SHOWN.

F61543 S0006594257_V2

Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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A-A

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC COLLAR. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30MY10K() HEX DRIVE BOLT AND A BACC30M COLLAR.

F61546 S0006594258_V3

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do the inspection of the entire stringer skin attachment flange from outside the wing tank at the stringer repair area.

(a) Do the inspection of the entire skin attachment flange at all fasteners common to the repair parts and the stringer, as shown in Figure 203/REPAIR 2.

(2) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the same inspection again at each 36,000 flight cycle interval.

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NOTE: THE UPPER WING SKIN IS NOT SHOWN.

DO LFEC INSPECTION OF STRINGER IN THIS AREA

1841610 S0000324317_V1

Inspection Instructions
Figure 203 (Sheet 1 of 2)

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B–B

1841611 S0000324318_V1

Inspection Instructions
Figure 203 (Sheet 2 of 2)

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REPAIR 3 - REPAIR FOR DAMAGE BETWEEN RIBS TO A VENT STRINGER ON THE OUTER WING

1. Applicability

A. Repair 3 is applicable to damage between ribs to vent stringers on the outer wing. A typical vent stringer is shown in Figure 201 (Sheet 1).

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 3 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections as given in Paragraph 5./REPAIR 3. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of these repairs. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for the repair categories and definitions.

F63910 S0006594260_V1

Layout of the Repair Parts
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS

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(Continued)

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
57-20-01	OUTER WING SKIN
57-20-01, ALLOWABLE DAMAGE 1	Wing Inspar Skin Inboard of WBL 616.75
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling
SOPM 20-10-03	SHOT PEENING

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 3. Refer to 51-10-02.

NOTE: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

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(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 3.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	As Necessary	Use 7075 in the annealed condition that is as thick as the depth of the recess. Heat treat to the T6 con- dition after you bend the part. If there is not a recess, do not use the part
[2]	Angle	2	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[3]	Angle	2	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[4]	Plate	1	Use 7075-T6 that is 0.050 inch thick
[5]	Plate	2	Use 7075-T6 that is 0.090 inch thick
[6]	Plate	2	Use 7075-T6 that is 0.090 inch thick
[7]	Filler	1	Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy. As an alternative, you can machine the part from 7075-T6 aluminum alloy material

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

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(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 3.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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NOTE: UPPER SKIN IS NOT SHOWN.

A-A

F61547 S0006594262_V1

Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K() HEX DRIVE BOLT AND A BACC30AC COLLAR. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30VH11CD() BLIND BOLT.

F61548 S0006594263_V2

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do the inspection of the entire stringer skin attachment flange from outside the wing tank at the stringer repair area.

(a) Do the inspection of the entire skin attachment flange at all fasteners common to the repair parts and the stringer, as shown in Figure 203/REPAIR 3.

(2) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the same inspection again at each 36,000 flight cycle interval.

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NOTE: UPPER SKIN IS NOT SHOWN.

DO LFEC INSPECTION OF INITIAL STRINGER IN THIS AREA

DO LFEC INSPECTION OF INITIAL STRINGER IN THIS AREA

A–A

1840230 S0000323963_V1

Inspection Instructions
Figure 203 (Sheet 1 of 2)

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B–B

1840208 S0000324257_V1

Inspection Instructions
Figure 203 (Sheet 2 of 2)

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REPAIR 4 - REPAIR FOR DAMAGE AT A RIB TO A VENT STRINGER ON THE OUTER WING

1. Applicability

A. Repair 4 is applicable to damage at a rib to vent stringers on the outer wing. A typical vent stringer is shown in Figure 201 (Sheet 1).

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 4 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 4. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for repair categories and definitions.

F63908 S0006594265_V1

Typical Vent Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS

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(Continued)

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
57-20-01	OUTER WING SKIN
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling
SOPM 20-10-03	SHOT PEENING

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 4. Refer to 51-10-02.

NOTE: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

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(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

E. Make the repair parts. Refer to Table 201/REPAIR 4.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	As Necessary	Use 7075 in the annealed condition that is as thick as the depth of the recess. Heat treat to the T6 con- dition after you bend the part. If there is not a recess, do not use the part
[2]	Angle	4	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[3]	Angle	4	Use 7075 in the annealed condition that is 0.090 inch thick. Heat treat to the T6 condition after you bend the part
[4]	Filler	As Necessary	Use 7075-T6 that is as thick as the depth of the recess in the stringer. If there is not a recess, do not use the part
[5]	Plate	2	Use 7075-T6 that is 0.050 inch thick
[6]	Plate	2	Use 7075-T6 that is 0.090 inch thick
[7]	Plate	2	Use 7075-T6 that is 0.090 inch thick
[8]	Stringer	1	Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 7055-T77511 aluminum alloy

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

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(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [8] stringer and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 4.

(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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NOTE: INITIAL UPPER WING SKIN NOT SHOWN

A

F61658 S0006594267_V1

Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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A–A

B–B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K() HEX DRIVE BOLT AND A BACC30AC COLLAR. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30VH11CD() BLIND BOLT.

F61653 S0006594268_V2

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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5. **Inspection Instructions**

A. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do the inspection of the entire stringer skin attachment flange from outside the wing tank at the stringer repair area.

(a) Do the inspection of the entire skin attachment flange at all fasteners common to the repair parts and the stringer, as shown in Figure 203/REPAIR 4.

(2) Mark the area of the inspection so you can find it when you do a repeat inspection.

B. After you do the initial inspection, do the same inspection again at each 36,000 flight cycle interval.

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Inspection Instructions
Figure 203 (Sheet 1 of 2)

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A–A

B–B

DO LFEC INSPECTION OF STRINGER IN THIS AREA

1841038 S0000324401_V1

Inspection Instructions
Figure 203 (Sheet 2 of 2)

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REPAIR 5 - REPAIR FOR DAMAGE TO A LOWER ZEE STRINGER ON THE OUTER WING

1. Applicability

A. Repair 5 is applicable to damage to the lower zee stringers on the outer wing. A typical lower zee stringer is shown in Typical Lower Zee Stringer Section, Figure 201/REPAIR 5.

B. Repair 5 is not applicable to damage at a location where the thickness of the stringer is more than 0.250 inch.

C. Repair 5 is not applicable to damage common to stringers which are adjacent to the lower skin access holes (rail stringers).

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Repair 5 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 5. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for the repair categories and definitions.

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Typical Lower Zee Stringer Section
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-50-02	SUPPORT OF AIRPLANE FOR REPAIR
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL
57-20-01	OUTER WING SKIN
AMM 28-11-00 P/B 201	FUEL TANKS - MAINTENANCE PRACTICES
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
AMM 28-11-00/801	Fuel Tanks - Approved Repairs
AMM 28-11-11/401	Wing Fuel Tank Access Panels
AMM 28-11-31/401	Center Fuel Tank Access Panels
AMM 28-26-00/201	Defueling

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(Continued)

Reference	Title
SOPM 20-10-03	SHOT PEENING

4. Repair Instructions

WARNING: MAKE SURE THAT YOU REMOVE ALL THE FUEL FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. IF YOU DO NOT OBEY, YOU CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Defuel all fuel tanks. Refer to AMM 28-26-00/201.

B. Remove the fuel tank access doors as necessary.

(1) Do the main fuel tank access door removal procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door removal procedure. Refer to AMM 28-11-31/401.

CAUTION: MAKE SURE THE WING IS HELD IN THE JIG POSITION WHEN TWO OR MORE STRINGERS THAT ARE ADJACENT TO EACH OTHER ARE REPAIRED. IT IS IMPORTANT THAT THE AIRPLANE BE STABLE. IF YOU DO NOT OBEY, YOU CAN CAUSE AN UNSATISFACTORY REPAIR.

C. Hold the wing in the jig position when two or more stringers that are adjacent to each other are repaired. Refer to 51-50-02.

D. Cut and remove the damaged part of the stringer as shown in Layout of the Repair Parts, Figure 202/REPAIR 5. Refer to 51-10-02.

NOTE: If the skin in the outer wing area is damaged, refer to 57-20-01 for allowable damage and repair data.

(1) Remove the full cross-section of the stringer.

(2) Use care so you do not damage the structure adjacent to the repair area.

(3) Keep the fastener edge margins.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	1	Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part
[2]	Angle	1	Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part
[3]	Angle	1	Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part
[4]	Angle	1	Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part
[5]	Angle	1	Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part
[6]	Angle	1	Use 2024 in the annealed condition that is 0.063 inch thick. Heat treat to the T42 condition after you bend the part
[7]	Filler	1	Make the part from the same type of extrusion and the same dimensions as the initial stringer. Use 2224-T3511 aluminum alloy
[8]	Strap	2	Use 2024-T3 that is 0.090 inch thick

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E. Make the repair parts. Refer to Table 201/REPAIR 5.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen all repair parts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

F. Assemble the repair parts.

(1) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

G. Drill the fastener holes. Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(1) For repairs made in rib bays 9, 10, and 11, install the initial fasteners in interference fit holes. Refer to Location of Rib Bays 9, 10, and 11, Figure 203/REPAIR 5 for the location of rib bays 9, 10, and 11.

(2) Refer to 51-40-05 for the hole sizes. Refer to 51-40-02 thru 51-40-03 and 51-40-06 for other fastener data.

H. Disassemble the repair parts.

I. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the cut edges of the stringer. Refer to 51-10-02.

J. Apply a chemical conversion coating to the repair parts and to the cut edges of the stringer. Refer to 51-20-01.

K. Apply one layer of BMS 10-20, Type II primer to the chemical conversion coating. Refer to AMM 28-11-00/701.

L. Install the repair parts and seal the repair. Refer to 51-20-05.

(1) Apply BMS 5-45 sealant between the initial stringer and the upper wing skin.

(a) Apply the sealant a minimum of 6 inches from the location where the stringer was cut.

(2) Install the repair parts.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

(a) Apply BMS 5-45 sealant between the mating surfaces. Refer to 51-20-05.

(b) Install the fasteners wet with BMS 5-45 sealant. If the fastener heads are external to the fuel tank, then it is optional to install the fasteners wet with BMS 5-95. Refer to 51-20-05.

(3) Fill the space between the part [7] filler and the initial stringer with BMS 5-45 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 5.

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(4) Apply a fillet seal around the repair parts and the fasteners with BMS 5-45 sealant. Refer to 51-20-05.

NOTE: CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

M. Reinstall the fuel tank access doors as necessary.

(1) Do the main fuel tank access door installation procedure. Refer to AMM 28-11-11/401.

(2) Do the center tank access door installation procedure. Refer to AMM 28-11-31/401.

N. Do the task in AMM 28-11-00/801 of refueling a repaired fuel tank.

(1) Refuel the fuel tank.

(2) Do a leak test to make sure there are no leaks in the fuel tank.

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NOTE: THE LOWER WING SKIN IS NOT SHOWN.

SECTION A-A

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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SECTION B-B

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION

INITIAL FASTENER LOCATION. INSTALL A 1/32 INCH DIAMETER OVERSIZE BACB30NY()K()Y HEX DRIVE BOLT AND A BACC30AC COLLAR. IN RIB BAYS 9, 10, AND 11 INSTALL THE FASTENER IN AN INTERFERENCE FIT HOLE. REFER TO FIGURE 203 FOR THE LOCATION OF RIB BAYS 9, 10, AND 11. NOTE: IF THIS REPAIR IS IN LIGHTNING ZONE 2, THEN REFER TO 51-80-01 FOR ADDITIONAL FASTENER INSTALLATION REQUIREMENTS.

REPAIR FASTENER LOCATION. INSTALL A BACB30MY10K() HEX DRIVE BOLT.

F61720 S0006594273_V3

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Location of Rib Bays 9, 10, and 11
Figure 203

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5. **Inspection Instructions**

A. Two possible inspection options are given. If the exterior side of the lower skin panel is accessible then do the inspection given in Paragraph 5.B. If the exterior side of the lower skin panel is covered by fittings, brackets or other parts, it is necessary to do the inspection in Paragraph 5.C instead.

B. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do this inspection if the exterior side of the lower skin panel is accessible.

(2) Do the inspection of the entire stringer skin attachment flange from outside the wing tank at the stringer repair area.

(a) Do the inspection of the entire skin attachment flange at all fasteners common to the repair parts and the stringer, as shown in Figure 204/REPAIR 5

(3) Mark the area of the inspection so you can find it when you do a repeat inspection.

(4) After you do the initial inspection, do the same inspection again at each 12,000 flight cycle interval.

C. Do a Low Frequency Eddy Current (LFEC) inspection by 20,000 flight cycles/40,000 flight hours after you install the repair, whichever comes first. Refer to 737 NDT Part 6, 51–00–27.

NOTE: No airplane may be operated beyond the stated threshold until the applicable NDT inspection procedures are validated, published in the appropriate manual, and accepted by the FAA.

(1) Do this inspection if the exterior side of the lower skin panel is not accessible.

(2) Do the inspection of the stringer vertical web flange from inside the wing tank at the stringer repair area.

(a) Do the inspection between and adjacent to all vertical web flange fasteners common to the repair parts and the stringer, as shown in Figure 205/REPAIR 5.

(3) After you do the initial inspection, do the same inspection again at each 10,000 flight cycle interval.

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SECTION A-A

DO LFEC INSPECTION OF THE STRINGER IN THIS AREA

NOTE: THE LOWER WING SKIN IS NOT SHOWN.

1841581 S0000324616_V1

Inspection Instructions with External Access to Lower Skin Panel
Figure 204 (Sheet 1 of 2)

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SECTION B–B

DO LFEC INSPECTION OF THE STRINGER IN THIS AREA

1841592 S0000324620_V1

Inspection Instructions with External Access to Lower Skin Panel
Figure 204 (Sheet 2 of 2)

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SECTION A–A

DO LFEC INSPECTION OF STRINGER IN THIS AREA

1841246 S0000324617_V1

Inspection Instructions with No External Access to Lower Skin Panel
Figure 205

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IDENTIFICATION 1 - OUTER WING RIBS

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

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Outer Wing Rib Locations
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
112A0015	Functional Product Collector - Inspar Ribs and Wing Tip, Left Side
112A0016	Functional Product Collector - Inspar Ribs and Wing Tip, Right Side
112A5010	Side-Of-Body Installation, BBL 70.85
112A5020	Rib Installation - Inspar Rib 2, WBL 94.24
112A5030	Rib Installation - Inspar Rib 3, WSTA 156.25
112A5040	Rib Installation - Inspar Rib 4, WSTA 180.00
112A5050	Rib Installation - Inspar Rib 5, WSTA 204.25
112A5060	Rib Installation - Inspar Rib 6, WSTA 228.25
112A5070	Rib Installation - Inspar Rib 7, WSTA 253.00
112A5080	Rib Installation - Inspar Rib 8, WSTA 278.50
112A5090	Rib Installation - Inspar Rib 9, WSTA 302.50
112A5100	Rib Installation - Inspar Rib 10, WSTA 326.00
112A5110	Rib Installation - Inspar Rib 11, WSTA 353.00
112A5120	Rib Installation - Inspar Rib 12, WSTA 378.50

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
112A5130	Rib Installation - Inspar Rib 13, WSTA 405.50
112A5140	Rib Installation - Inspar Rib 14, WSTA 432.50
112A5150	Rib Installation - Inspar Rib 15, WSTA 458.50
112A5160	Rib Installation - Inspar Rib 16, WSTA 485.00
112A5170	Rib Installation - Inspar Rib 17, WSTA 512.00
112A5180	Rib Installation - Inspar Rib 18, WSTA 537.50
112A5190	Rib Installation - Inspar Rib 19, WSTA 564.50
112A5200	Rib Installation - Inspar Rib 20, WSTA 591.50
112A5210	Rib Installation - Inspar Rib 21, WSTA 617.00
112A5220	Rib Installation - Inspar Rib 22, WSTA 643.50
112A5230	Rib Installation - Inspar Rib 23, WSTA 667.50
112A5240	Rib Installation - Inspar Rib 24, WSTA 691.50
112A5250	Rib Installation - Inspar Rib 25, WBL 616.75
737–8025	Rib Installation - Inspar Rib 25, WBL 616.75*[1]
112A5260	Rib Installation - Inspar Rib 26, WSTA 736.50
112A5270	Rib Installation - Closure Rib 27, WBL 658.17
737–8024	Rib Installation - Closure Rib 27, WBL 658.17*[1]
737-5270	Rib Installation - Closure Rib 27, WBL 658.17*[1]
737–8502	Adapter Rib Assy, Wing Tip, WBL 658.17 *[1]

*[1] Aviation Partners Boeing (APB) part as retrofitted by Supplemental Type Certificate (STC) ST00830SE.

3. Applicability

A. Identification 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglets.

B. List of Materials:

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Side of Body Rib Assembly		Refer to Figure 3	
	Web, Machined		7150-T651 aluminum plate as given in BMS 7-256. Refer to Figure 3 for the machined thicknesses	
	Forward Web	0.500 (12.7)		
	Mid Web	0.500 (12.7)		
	Aft Web	0.700 (17.78)		

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Stiffener 1 (8)		BAC1520-2798 7150-T77511 aluminum extrusion as given in BMS 7-306	
	Stiffener 2 (3)		BAC1506-4454 7150-T77511 aluminum extrusion as given in BMS 7-306	
	Stiffener 3 (2)		BAC1505-100900 7150-T77511 aluminum extrusion as given in BMS 7-306	
	Stiffener 4		BAC1520-2797 7150-T77511 aluminum extrusion as given in BMS 7-306	
	Upper Rib Chord		BAC1520-2775 7075-T73 aluminum extrusion as given in QQ-A-200/11	
	Lower Rib Chord		BAC1520-2754 2024-T42 aluminum extrusion as given in QQ-A-200/3	
[2]	Rib, Machined		7050-T7451 aluminum plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses. Refer to Figure 4 for an example of a typical machined rib.	
			7050-T7451 aluminum plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined thicknesses. Refer to Figure 4 for an example of a typical machined rib *[2].	Cum Line 789 and on with winglet full provision installation *[2].
[3]	Rib, Machined		7050-T7451 aluminum plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined thicknesses. Refer to Figure 5 for an example of a typical machined rib.	
[4]	Rib 25, Retrofit		Refer to Figure 7.	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.
	Rib Post - Rear Spar		7050-T7451 aluminum plate per AMS 4050.	
	Rib Post - Front Spar		7050-T7451 aluminum plate per AMS 4050.	
	Chord - Upper		BAC1505-100637 7075-T6511 aluminum as given in QQ-A-200/11. Optional: 7075-T3511 aluminum extrusion as given in QQ-A-200/11. Optional Material - 7050-T7451 aluminum plate as given in BMS 7-323 Type I or III.	
	Chord - Lower		BAC1505-100637 7075-T6511 aluminum as given in QQ-A-200/11. Optional: 2024-T3511 aluminum extrusion as given in QQ-A-200/3. Optional: 2024-T351 aluminum plate as given in QQ-A-250/4.	
	Web		2024-T3 clad aluminum sheet as given in QQ-A-250/5	
	Seal Plate		2024-T3 bare aluminum sheet as given in QQ-A-250/4. Optional: 6061-T4 bare aluminum sheet as given in QQ-A-250/11. Optional: 7075-T6 bare aluminum sheet as given in QQ-A-250/12.	Limited

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Stiffener		AND10134-1006 stock extruded shape 7075-T76511 aluminum extruded shape as given in QQ-A-200/15 or 7075-T6511 aluminum extruded shape as given in QQ-A-200/11. Optional Materials: 7050-T7451 aluminum plate as given in AMS 4050 or 7075-T7351 aluminum plate as given in QQ-A-250/12 or 7075-T73510 aluminum extruded bar as given in QQ-A-250/11	
	Strap - Lower		BAC1505-100637 7075-T6511 aluminum as given in QQ-A-200/11. Optional: 2024-T3511 aluminum extrusion as given in QQ-A-200/3. Optional: 2024-T351 aluminum plate as given in QQ-A-250/4.	
	Stringer - End Fitting		7075-T7451 aluminum plate as given in AMS4050	
[5]	Inspar Rib 27		Refer to Fig 8. 7050–T7451 aluminum plate as given in BMS 7-323 Type I or Type III.	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.
[6]	Inspar Rib 27		Refer to Fig. 6. 7050-T7451 aluminum plate as given in BMS 7-323, Type 1 *[3]. Optional: 7050-T7451 aluminum plate as given in BMS 7-323, Type III *[4].	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.
[7]	Adapter Rib - Wing Tip		Refer to Fig 9. 7050-T7451 aluminum plate as given in AMS4050.	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.
[8]	Inspar Rib 27 Fwd & Aft Attach Fittings		7050-T7451 aluminum plate as given in AMS4050 for Boeing production part numbers; see Figure10.	
[9]	Inspar Rib 27 Fwd & Aft Attach Fittings		7050-T7451 aluminum plate as given in BMS 7-323, Type III for Aviation Partners Boeing (APB) part numbers; see Figure 11.	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.
[10]	Inspar Rib 27 Fwd & Aft Attach Fittings		7050-T7451 aluminum plate as given in BMS 7-323, Type III for Aviation Partners Boeing (APB) part numbers; see Figure 11.	For airplanes that have Aviation Partners Boeing (APB) parts as retrofitted by Supplemental Type Certificate (STC) ST008300SE.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

*[2] Applicable only to Inspar Rib 27 Closure Rib.

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*[3] APB Part No. 737-5271-5,-6,-7,-8.

*[4] APB Part No. 737-5271-7,-8 only.

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Outer Wing Rib Identification
Figure 2

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NOTES

- ALL DIMENSIONS ARE IN INCHES.

F85218 S0006594282_V1

Side of Body Rib Assembly
Figure 3 (Sheet 1 of 4)

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F85221 S0006594283_V1

Side of Body Rib Assembly
Figure 3 (Sheet 2 of 4)

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MID WEB OF THE SIDE OF BODY RIB

B

B–B

F85225 S0006594284_V1

Side of Body Rib Assembly
Figure 3 (Sheet 3 of 4)

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AFT WEB OF THE SIDE OF BODY RIB

C

C–C

F85226 S0006594285_V1

Side of Body Rib Assembly
Figure 3 (Sheet 4 of 4)

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PRODUCTION MACHINED RIB NO. 27 FOR WING TIP
(FOR PARTIAL OR NO WINGLET PROVISION INSTALLATION)

PRODUCTION MACHINED RIB NO. 27 FOR WINGLET
(FOR CUM LINE NUMBER 789 AND ON WITH WINGLET FULL
PROVISION INSTALLATION)

MACHINED RIB (TYPICAL)

2156796 S0000470595_V1

Typical Outer Wing Machined Rib for Figure 2, Item [2]
Figure 4

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MACHINED RIB, SHEAR TIED (TYPICAL)

F85437 S0006594287_V1

Typical Outer Wing Machined Rib for Figure 2, Item [3]
Figure 5

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NOTES

- RETROFIT RIB 27 INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE.
- ALL PART NUMBERS LISTED ARE AVIATION PARTNERS BOEING (APB) PART NUMBERS

AVIATION PARTNERS BOEING (APB)
RETROFIT RIB NO. 27 FOR
WINGLET - PARTIALLY PROVISIONED WING

2156853 S0000470766_V1

Aviation Partners Boeing (APB) Retrofit Rib No. 27 for Winglet
Figure 6

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NOTES

- RETROFIT RIB 25 INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE.
- ALL PART NUMBERS LISTED ARE AVIATION PARTNERS BOEING (APB) PART NUMBERS UNLESS NOTED.

AVIATION PARTNERS BOEING (APB) RETROFIT RIB NO. 25
FOR WINGLET - UNPROVISIONED WING

2157464 S0000470853_V1

Aviation Partners Boeing (APB) Retrofit Rib No. 25 for Winglet
Figure 7

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NOTES

- RETROFIT RIB 27 INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE.
- ALL PART NUMBERS LISTED ARE AVIATION PARTNERS BOEING (APB) PART NUMBERS UNLESS NOTED.

AVIATION PARTNERS BOEING (APB) RETROFIT RIB NO. 27
FOR WINGLET - UNPROVISIONED WING

2156969 S0000470855_V1

Aviation Partners Boeing (APB) Retrofit Rib No. 27 for Winglet
Figure 8

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NOTES

- ADAPTER RIB ASSEMBLY INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE.
- ALL PART NUMBERS LISTED ARE AVIATION PARTNERS BOEING (APB) PART NUMBERS UNLESS NOTED.

AVIATION PARTNERS BOEING (APB) RETROFIT RIB NO. 27
FOR WINGLET - UNPROVISIONED WING

2157240 S0000470856_V1

Aviation Partners Boeing (APB) Rib Assembly Wing Tip Adapter
Figure 9

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PRODUCTION ATTACH FITTINGS
(FOR PARTIAL OR NO WINGLET PROVISION)

2160794 S0000473803_V1

Production Rib No. 27 Attach Fittings
Figure 10 (Sheet 1 of 2)

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PRODUCTION ATTACH FITTINGS FOR WINGLETS
(FOR CUM NUMBER 789 AND ON WITH WINGLET FULL
PROVISION INSTALLATION)

2161071 S0000473806_V1

Production Rib No. 27 Attach Fittings
Figure 10 (Sheet 2 of 2)

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NOTES

- ATTACH FITTINGS INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE.
- ALL PART NUMBERS LISTED ARE AVIATION PARTNERS BOEING (APB) PART NUMBERS UNLESS NOTED.

AVIATION PARTNERS BOEING (APB) RETROFIT RIB NO. 27
ATTACH FITTINGS FOR WINGLET - UNPROVISIONED WING

2160743 S0000473811_V1

Aviation Partners Boeing (APB) Retrofit Rib No. 27 Attach Fittings
Figure 11 (Sheet 1 of 2)

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NOTES

- ATTACH FITTINGS INSTALLED AS GIVEN IN SUPPLEMENTAL TYPE CERTIFICATE (STC) ST00830SE.
- ALL PART NUMBERS LISTED ARE AVIATION PARTNERS BOEING (APB) UNLESS NOTED.

AVIATION PARTNERS BOEING (APB)
RETROFIT RIB NO. 27
ATTACH FITTINGS - PARTIALLY PROVISIONED WING

2161063 S0000473814_V1

Aviation Partners Boeing (APB) Retrofit Rib No. 27 Attach Fittings
Figure 11 (Sheet 2 of 2)

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ALLOWABLE DAMAGE 1 - OUTER WING RIBS

1. Applicability

A. ALLOWABLE DAMAGE 1 gives the allowable damage limits for the outer wing ribs shown in Outer Wing Rib Locations, Figure 101/ALLOWABLE DAMAGE 1.

B. ALLOWABLE DAMAGE 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

A

F85496 S0006594290_V1

Outer Wing Rib Locations
Figure 101 (Sheet 1 of 2)

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SIDE OF BODY RIB (ALUMINUM)

B

TYPICAL MACHINED RIB (ALUMINUM)

C

TYPICAL MACHINED RIB, SHEAR TIED (ALUMINUM)

D

F85532 S0006594291_V1

Outer Wing Rib Locations
Figure 101 (Sheet 2 of 2)

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2. General

WARNING: MAKE SURE THAT ALL THE FUEL IS REMOVED FROM THE TANKS. MAKE SURE THAT THE TANKS ARE SUPPLIED WITH A GOOD FLOW OF AIR. FUEL VAPORS CAN CAUSE AN EXPLOSION AND INJURY TO PERSONS NEAR OR ON THE AIRPLANE.

A. Remove the parts as necessary to get access to the damaged area.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

C. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of abrasives and other materials you can use to remove damage.

(3) Refer to 51-30-05 for possible sources of equipment and tools you can use to remove damage.

D. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(2) Apply a layer of BMS 10-20, Type II primer to the surface of the reworked area. Refer to SOPM 20-41-02.

(3) Apply BMS 5-26 or BMS 5-45 sealant to damaged fillet seals as given in 51-20-05.

(a) Refer to 51-20-05 for the sealing of tank end ribs.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Side of Body Rib

(1) Web

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A and B .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, C and F.

(c) Dents:

1) Refer to Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail D for the damage that is permitted.

(d) Holes and Punctures are not permitted.

(2) Chord

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(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Details A, B and E.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Details A, B, C and F.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(3) Stiffener

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Details A, B and E.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Detail A, B, C, E and F.

(c) Dents are not permitted:

(d) Holes and Punctures are not permitted.

B. Inspar Ribs 2 through 27

(1) Web

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Detail A, B, C and F.

(c) Dents:

1) Refer to Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail D for the damage that is permitted.

(d) Holes and Punctures are not permitted.

(2) Chord

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Details A, B and E.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Details A, B, C, E and F.

(c) Dents:

1) Refer to Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Detail D for the damage that is permitted.

(d) Holes and Punctures are not permitted.

(3) Stiffener

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ ALLOWABLE DAMAGE 1, Details A, B and E.

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(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 102/ALLOWABLE DAMAGE 1, Details A, B, C, E and F.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(4) Shear tie

(a) Damage is not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

NOTE: THE MAXIMUM CROSS-SECTIONAL AREA OF MATERIAL THAT IS REMOVED MUST NOT BE MORE THAN 4% OF THE INITIAL CROSS-SECTIONAL AREA.

FOR FINDING THE CROSS-SECTIONAL AREA, SEE G .

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

A–A

F85536 S0006594292_V1

Allowable Damage Limits
Figure 102 (Sheet 1 of 4)

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$\frac{W}{Y}$ MUST BE 30 OR MORE AT EACH POINT ALONG THE LENGTH OF THE DENT.

SIDE OF BODY RIB WEB:
Y = A MAXIMUM OF 0.05 INCH FOR THE WEB.

INSPAR RIBS 2 THRU 27 CHORD:
Y = A MAXIMUM OF 0.125 INCH.

INSPAR RIBS 2 THRU 27 WEB:
Y = A MAXIMUM OF 0.05 INCH

DENT THAT IS PERMITTED

D

F85754 S0006594293_V2

Allowable Damage Limits
Figure 102 (Sheet 2 of 4)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED

SIDE OF BODY RIB AND INSPAR RIBS 2 THRU 27:
X = A MAXIMUM OF 0.10 INCH.

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

E

NOTE: THE CROSS-SECTIONAL AREA OF THE DAMAGE WHICH HAS BEEN REMOVED ALONG ANY LINE A-B OR E-F MUST NOT BE MORE THAN 4 PERCENT OF THE INITIAL CROSS-SECTIONAL AREA.

IF THERE ARE NO INITIAL HOLES MADE BY BOEING, DO NOT USE "D" WHEN FINDING THE INITIAL CROSS-SECTIONAL AREA.

CROSS-SECTIONAL AREA REMOVAL

F

F85756 S0006594294_V2

Allowable Damage Limits
Figure 102 (Sheet 3 of 4)

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C–C

EXAMPLE: A_1 = INITIAL CROSS-SECTIONAL AREA
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE INITIAL HOLE CROSS-SECTIONAL AREA MADE BY BOEING.
= TH–TD

A_2 = CROSS-SECTIONAL AREA OF THE DAMAGE WHICH HAS BEEN REMOVED
= THE DEPTH OF THE DAMAGE REMOVED MULTIPLIED BY THE DIMENSION ACROSS THE DAMAGE REMOVED
= XI

$\left(\frac{A_2}{A_1}\right) \leq$ 4% MAXIMUM

F88959 S0006594295_V1

Allowable Damage Limits
Figure 102 (Sheet 4 of 4)

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REPAIR 1 - OUTER WING RIB REPAIRS

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

OUTER WING RIBS

A

F85492 S0006594298_V2

Outer Wing Rib Repairs
Figure 201

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IDENTIFICATION 1 - OUTER WING FRONT SPAR STRUCTURE

1. Locator

U40944 S0000194169_V1

Outer Wing Front Spar Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
111A2000	Front Spar Installation - Wing Center Section
112A0013	Collector - Functional, Product, Front Spar, Section 12, Left
112A2001	Spar Installation - Front, Section 12, Wing
112A2003	Spar Assembly - Front, Section 12, Wing
112A2004	Leading Edge Support Installation - Front, Section 12, Wing

3. Applicability

A. IDENTIFICATION 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. List of Materials:

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Angle Support		7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[2]	Actuator Support		7050-T7451 forged block as given in BMS 7-186. Refer to production drawing for the machined areas (Grain direction controlled part)	
[3]	Stiffener (10)		7050-T7451 plate as given in BMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[4]	Actuator Support		7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[5]	Chord, Upper		BAC1514-3096 7150-T77511 extrusion as given in BMS 7-306 BAC1514-3105 7150-T77511 extrusion as given in BMS 7-306	Cum Line 001 to 777 CUM LINE 778 and on
[6]	Stiffener		BAC1503-100017 7075-T6511 extrusion as given in QQ-A-250/11	
[7]	Angle Support	0.063 (1.60)	2024-T42 sheet as given in QQ-A-250/5	
[8]	Chord, Lower		BAC1514-3095 2224-T3511 extrusion as given in BMS 7-255	
[9]	Stiffener (5)		7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[10]	Stiffener		BAC1503-100130 7075-T6511 extrusion as given in QQ-A-200/11	
[11]	Web		2324-T39 plate as given in BMS 7-254. Refer to the production drawing for the machined areas	
[12]	Stiffener		BAC1503-100683 7075-T6511 extrusion as given in QQ-A-250/11	
[13]	Support Bracket (4)		BAC1509-100662 7075-T3511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined areas. (Optional: 7050-T7451 as given given in AMS 4050)	
[14]	Stiffener (2)		BAC1514-1655 7075-T6511 extrusion as given in QQ-A-250/11	
[15]	Stiffener		BAV1503-100624 7075-T6511 extrusion as given in QQ-A-200/11	
[16]	Bracket Assembly (4) Bracket Bracket	 0.050 (1.27)	 BAC1509-100662 7075-T73511 extrusion as given in QQ-A-200/11 (Optional: 7050-T7451 as given given in AMS 4050) 6013-T6 sheet as given in AMS 4347	
[17]	Stiffener (3)		BAC1503-100607 7075-T6511 extrusion as given in QQ-A-250/11	
[18]	Stiffener		BAC1503-100671 7075-T6511 extrusion as given in QQ-A-250/11	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[19]	Stiffener		BAC1514-447 7075-T6511 extrusion as given in QQ-A-250/11	Left side only
[20]	Stiffener(2)		BAC1503-100387 7075-T6511 extrusion as given in QQ-A-200/11.	
[21]	Stiffener (3)		BAC1503-100413 7075-T6511 extrusion as given in QQ-A-250/11	
[22]	Angle (2)		BAC1503-100796 7075-T73511 extrusion as given in QQ-A-200/11	
[23]	Stiffener		BAC1503-100787 7075-T6511 extrusion as given in QQ-A-200/11	
[24]	Stiffener		7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Optional: 7050-7451 as given in AMS 7–323, Type 1) (Grain direction controlled part).	
[25]	Rib Post (10)		7050-T7451 plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[26]	Rib Post (2)		7050-T7451 plate as given in BMS 7-323, Type 1 (Optional: 7050-T7451 plate as given in AMS 4050) (Grain direction controlled part)	
[27]	Rib Post (9)		7050-T7451 plate as given in BMS 7-323, Type 1 (Grain direction controlled part)	
[28]	Rib Post (1)		7050-T7451 plate as given in BMS 7-323, Type 1 (Grain direction controlled part)	Left side only
[29]	Slat Can (7)		D357-T6 investment casting as given in BMS 7-330. Refer to the production drawing for the machined areas	

*[1] Note: T = Pre-manufactured thicknesses in inches (millimeters).

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NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

G23695 S0006594305_V2

Outer Wing Front Spar Identification
Figure 2 (Sheet 1 of 2)

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Outer Wing Front Spar Identification
Figure 2 (Sheet 2 of 2)

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IDENTIFICATION 2 - OUTER WING REAR SPAR

1. Locator

U41091 S0000194885_V1

Outer Wing Rear Spar Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
112A0011	Collector - Functional, Product, Rear Spar, Section 12, Left
112A1001	Spar Installation - Rear Wing
112A1003	Spar Assembly - Rear Wing

3. Applicability

A. IDENTIFICATION 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. List of Materials

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Rib Post (8)		7050-T7451 machined plate as given in BMS 7-323, Type 1 Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[2]	Rib Post (2)		7050-T7451 machined plate as given in BMS 7-323, Type III Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[3]	Rib Post		7050-T7452 forged block as given in AMS 4050, Type III Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	YA001-YC003, YE001-YG999
			7050-T7452 forged block as given in BMS 7-214, Type II (Optional: 7050-T7452 forged block as given in AMS 4050, Type III) (Grain direction controlled part)	YC004-YD999
[4]	Stiffener		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	YA001-YC003 YE001-YG999
			7050-T7451 machined plate as given in BMS 7-323, Type I (Grain direction controlled part)	YC004-YD999
[5]	Rib Post (4)		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	YA001-YC003 YE001-YG999
			7050-T7451 machined plate as given in BMS 7-323, Type I (Grain direction controlled part)	YC004-YD999
[6]	Rib Post (5)		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[7]	Stiffener (9)		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[8]	Stiffener (7)		BAC1503-100917 7075-T6511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined thicknesses	
[9]	Stiffener (3)		BAC1503-100066 7075-T6511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined thicknesses	
[10]	Bracket Assembly			
	Bracket		7075-T7351 machined plate, Class A as given in QQ-A-250/12. Refer to the production drawing for the machined thicknesses	
	Clip	0.063 (1.6)	6013-T6 formed sheet as given in AMS 4347	
	Bracket	0.063 (1.6)	6013-T6 formed sheet as given in AMS 4347	
	Bracket	0.050 (1.27)	6013-T6 formed sheet as given in AMS 4347	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[11]	Stiffener (3)		BAC1514-735 7075-T6511 extrusion as given in QQ-A-200/11 Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[12]	Stiffener (3)		7075-T73511 extruded bar as given in QQ-A-200/11 Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[13]	Bracket (3)		7075-T7351 machined plate as given in QQ-A-250/12. Refer to the production drawing for the machined thicknesses	
[14]	Stiffener (4)		BAC1503-100161 7075-T6511 extrusion as given in QQ-A-200/11 Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[15]	Bracket		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined thicknesses (Grain direction controlled part)	
[16]	Bracket	0.050 (1.27)	6013-T6 formed sheet as given in AMS 4347	
[17]	Stiffener (2)		BAC1517-2769 7075-T6511 extrusion as given in QQ-A-200/11. Refer to the production drawing for the machined thicknesses	
[18]	Bracket	0.080 (2.03)	6013-T6 formed sheet as given in AMS 4347	
[19]	Chord, Upper		BAC1514-3102 7150-T77511 formed extrusion as given in BMS 7-306	
[20]	Web		2324-T39 machined plate as given in AMS 7-254. Refer to the production drawing for the machined thicknesses	
[21]	Chord, Lower		BAC1514-3103 2224-T3511 formed extrusion as given in BMS 7-255	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

VIEW IS LOOKING AFT

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G28207 S0006594311_V1

Outer Wing Rear Spar Identification
Figure 2

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G29409 S0006594312_V1

Outer Wing Rear Spar Identification
Figure 3

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

ALLOWABLE DAMAGE 1 - OUTER WING FRONT SPAR

1. Applicability

A. ALLOWABLE DAMAGE 1 is applicable to damage on the outer wing front spar as shown in Location of the Outer Wing Front Spar, Figure 101/ALLOWABLE DAMAGE 1.

B. ALLOWABLE DAMAGE 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

2. General

A. Refer to Outer Wing Front Spar, Figure 102/ALLOWABLE DAMAGE 1 for the different parts of the front spar assembly.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

C. Remove the damage.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasive and other materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(4) Make the surface texture roughness for all cut surfaces 125 microinches Ra or smoother.

D. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the chords, web, and rib posts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(3) Apply one layer of BMS 10-20, Type II primer to the conversion coated, reworked areas. Refer to AMM 28-11-00/701.

(4) Refer to 51-20-05 for sealing of fuel tanks.

(5) Refer to 51-20-05 for the installation of fasteners in fuel tanks.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	**PARAGRAPH**
UPPER CHORD	4.A
LOWER CHORD	4.A
WEB	4.B
STIFFENERS	4.C
RIB POSTS	4.C

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Table 101: (Continued)

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	PARAGRAPH
SUPPORT BRACKETS	4.D
ACTUATOR SUPPORTS	4.D

G11026 S0006594316_V1

Location of the Outer Wing Front Spar
Figure 101

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Outer Wing Front Spar
Figure 102 (Sheet 1 of 2)

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G11242 S0006594318_V1

Outer Wing Front Spar
Figure 102 (Sheet 2 of 2)

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Upper and Lower Chords

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E, and F.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage with one of the two procedures that follow:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and E .

2) Drill a hole through the part at the damage location:

NOTE: Do not drill out the damage if the damage has been blended out before.

a) The hole must be:

- A maximum of 0.25 inch in diameter
- A minimum of 1.5 inches from a fastener hole, the edge of the part, or other damage.

b) Fill the hole with a 2117-T3 or 2117-T4 aluminum rivet installed wet with BMS 5-26 sealant.

(3) Dents are permitted if:

(a) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail G

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(b) The edge of the damage is a minimum of 1.5 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) They are a maximum of 0.25 inch in diameter

(b) The edge of the damage is a minimum of 1.5 inches away from a fastener hole or other damage

(c) They are filled with a 2117-T3 or 2117-T4 aluminum rivet. Install the rivet with BMS 5-26 sealant.

C. Stiffeners and Rib Posts

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Support Brackets and Actuator Supports

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

C

G11472 S0006594319_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 4)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

D

A–A

G11480 S0006594320_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 4)

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NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

B–B

1347889 S0000241696_V2

Allowable Damage Limits
Figure 103 (Sheet 3 of 4)

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DAMAGE LIMITS IN A CROSS SECTION OF A SPAR CHORD

F

DENT THAT IS PERMITTED

G

1347890 S0000241699_V2

Allowable Damage Limits
Figure 103 (Sheet 4 of 4)

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ALLOWABLE DAMAGE 2 - OUTER WING REAR SPAR

1. Applicability

A. ALLOWABLE DAMAGE 2 is applicable to damage on the outer wing rear spar shown in Location of the Outer Wing Rear Spar, Figure 101/ALLOWABLE DAMAGE 2.

B. ALLOWABLE DAMAGE 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

2. General

A. Refer to Outer Wing Rear Spar, Figure 102/ALLOWABLE DAMAGE 2 for the different parts of the rear spar assembly.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

C. Remove the damage.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasive and other materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

D. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the chords, web, and rib posts.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(3) Apply one layer of BMS 10-20, Type II primer to the conversion coated, reworked areas. Refer to AMM 28-11-00/701.

(4) Refer to 51-20-05 for sealing of fuel tanks.

(5) Refer to 51-20-05 for the installation of fasteners in fuel tanks.

E. Refer to Table 101/ALLOWABLE DAMAGE 2 for the references for the allowable damage limits of the different types of structure.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	**PARAGRAPH**
UPPER CHORD	4.A
LOWER CHORD	4.A
WEB	4.B
STIFFENERS	4.C

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Table 101: (Continued)

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	**PARAGRAPH**
RIB POSTS	4.C
HINGE ASSEMBLY	4.D

G14540 S0006594325_V1

Location of the Outer Wing Rear Spar
Figure 101

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Outer Wing Rear Spar
Figure 102 (Sheet 1 of 2)

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VIEW IS LOOKING AFT

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE G15008 S0006594327_V1

Outer Wing Rear Spar
Figure 102 (Sheet 2 of 2)

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
AMM 28-11-00/701	Fuel Tanks - Cleaning and Painting
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Upper and Lower Chords

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , and F .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage with one of the two procedures that follows:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , and E .

2) Drill a hole through the part at the damage location:

NOTE: Do not drill out the damage if the damage has been blended out before.

a) The hole must be:

- A maximum of 0.25 inch in diameter
- A minimum of 1.5 inches from a fastener hole, the edge of the part, or other damage.

b) Fill the hole with a 2117-T3 or 2117-T4 aluminum rivet installed wet with BMS 5-26 sealant.

(3) Dents are permitted if:

(a) They agree with the conditions shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail G

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(b) The edge of the damage is a minimum of 1.5 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) They are a maximum of 0.25 inch in diameter

(b) The edge of the damage is a minimum of 1.5 inches away from a fastener hole or other damage

(c) They are filled with a 2117-T3 or 2117-T4 aluminum rivet. Install the rivet wet with 5-26 sealant.

C. Stiffeners and Rib Posts

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Hinge Assembly

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.05 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

C

G15013 S0006594328_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 4)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE WIDTH OF THE FLANGE

NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

D

A–A

G15017 S0006594329_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 4)

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NOTE: DO NOT REMOVE MORE THAN 4 PERCENT OF THE INITIAL CROSS SECTIONAL AREA.

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

B–B

1347892 S0000241705_V2

Allowable Damage Limits
Figure 103 (Sheet 3 of 4)

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DAMAGE LIMITS IN A CROSS SECTION OF A SPAR CHORD

F

DENT THAT IS PERMITTED

G

1347893 S0000241706_V2

Allowable Damage Limits
Figure 103 (Sheet 4 of 4)

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ALLOWABLE DAMAGE 3 - OUTER WING SPAR WEB AT FUEL BOOST PUMP

1. Applicability

A. ALLOWABLE DAMAGE 3 is applicable to damage on the outer wing spar at the fuel boost pump as shown in Figure 101/ALLOWABLE DAMAGE 3.

B. ALLOWABLE DAMAGE 3 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

DAMAGE IS PERMITTED IN THIS AREA. REFER TO FIGURE 102 FOR DETAIL.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

1945696 S0000367810_V1

Location of the Outer Wing Spar Web at the Fuel Boost Pump
Figure 101 (Sheet 1 of 2)

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DAMAGE IS PERMITTED IN THIS AREA. REFER TO FIGURE 102 FOR DETAIL.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

B

1945715 S0000367811_V1

Location of the Outer Wing Spar Web at the Fuel Boost Pump
Figure 101 (Sheet 2 of 2)

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2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 3 for the allowable damage limits.

B. Remove the damage. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02

(1) Remove the damage with a minimum 1.0 in. (25.4 mm) blend radius and the following taper ratios:

- 20:1 minimum to 50:1 maximum in the inboard/outboard direction.
- 10:1 minimum to 20:1 maximum in the up/down direction.

(2) Keep a surface finish 125 microinches Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

(3) Make sure that the surface of the spar is flat at the boost pump fastener and bonding jumper locations.

C. After you remove the damage, do the steps that follow:

(1) Clean the reworked areas with solvent. Refer to SOPM 20-30-03 for the applicable solvents and procedures.

(2) Do a Fluorescent Penetrant Inspection of the reworked areas to make sure that there is no other damage. Refer to SOPM 20-20-02.

NOTE: As an alternative, you can do a High Frequency Eddy Current (HFEC) inspection of the reworked area. Refer to 737 NDT Part 6, 51-00-00, Procedure 1 and 737 NDT Part 6, 51-00-00, Procedure 4.

(3) Flap peen or shot peen the reworked areas.

(a) Refer to SHOT PEENING, 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(4) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

(5) If you removed damage from the bonding jumper locations, mask off the 1.25 inch diameter area that did not have an initial layer of primer. Refer to Figure 103/ALLOWABLE DAMAGE 3.

(6) Apply one layer of BMS 10-20, Type II primer. Refer to AMM SUBJECT 28-11-00.

(7) If you removed the support angle (116A3939) to get access to the outer wing front spar web, do the steps that follow:

(a) The gap between the support angle and the spar web must be 0.005 in. (0.13 mm) or less. If the gap is more than 0.005 in. (0.13 mm), you must install shims.

1) Make the shims. Use 7075-T6 or 2024-T3 aluminum.

2) Apply a chemical conversion coating to the fabricated shims. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

3) Apply one coat of BMS 10-11, Type 1 primer to the fabricated shims.

4) Apply BMS 5-45 sealant between the angle and the spar web. Refer to REPAIR SEALING, 51-20-05.

(b) Install the angle using 1/64 inch oversize BACB30NX fasteners and BACC30X or BACC30BH collars. You are permitted to use alternative fasteners as given in FASTENER SUBSTITUTION, 51-40-03. The hole size must be a 0.001 in. (0.03 mm) to 0.0035 in. (0.09 mm) interference fit as given in FASTENER HOLE SIZES, 51-40-05.

(8) Polish the contact areas of the bonding jumper terminal with an abrasive pad to a smooth, uniform and bright finish.

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(9) If you removed the boost pump, reinstall the boost pump. Refer to AMM SUBJECT 28-22-41.

(a) Remove all masking before you install the boost pump.

(b) Apply a fillet seal with BMS 5-142 sealant to the surface outside the fuel tank where the pump housing touches the spar. Refer to AMM TASK 28-22-41-400-802.

(10) Install the bonding jumper. Refer to AMM SUBJECT 28-22-41.

NOTE: Replace the initial NAS1149D0416H washer with an equivalent washer for installation between the bonding jumper terminal and spar web. The replacement washer must be the same type as the initial and have the appropriate thickness to perform bolt installation as given in FASTENER INSTALLATION AND REMOVAL, 51-40-02.

NOTE: CDCCL - Refer to the task: Airworthiness Limitation Precautions, TASK 28-00-00-910-801, for important information on Critical Design Configuration Control Limitations (CDCCLs).

NOTE: This is applicable to Airworthiness Limitation 28-AWL-14.

(11) Refer to REPAIR SEALING, 51-20-05 for sealing of fuel tanks.

(12) Refer to REPAIR SEALING, 51-20-05 for the installation of fasteners in fuel tanks.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-20-13	SURFACE ROUGHNESS FINISH REQUIREMENTS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
AMM 28-11-00	FUEL TANKS
AMM 28-22-41	FUEL BOOST PUMP
AMM 28-22-41-400-802	Fuel Boost Pump Housing Installation (P/B 401)
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-30-03	General Cleaning Procedures
737 NDT Part 6, 51-00-00, Procedure 1	Fastener Holes in Aluminum Parts (Meter Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Allowable Damage Limits

A. Nicks, Gouges, Scratches, and Corrosion:

(1) Remove the damage as shown in Figure 102/ALLOWABLE DAMAGE 3, Details A , B and C. You must not remove more material from the spar than as follows:

(a) For the outer wing external front spar web at a fuel boost pump:

- Zone 1: 0.1 in. (2.54 mm) maximum depth, 0.20 in^2 (129.0 mm^2) maximum total cross section area loss.
- Zone 2: 0.1 in. (2.54 mm) maximum depth, 0.20 in^2 (129.0 mm^2) maximum total cross section area loss.

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(b) For the outer wing external rear spar web at a fuel boost pump

- Zone 1: 0.1 in. (2.54 mm) maximum depth, 0.40 in^2 (258.1 mm^2) maximum total cross section area loss.
- Zone 2: 0.1 in. (2.54 mm) maximum depth, 0.40 in^2 (258.1 mm^2) maximum total cross section area loss.

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Allowable Damage Limits - Outer Wing Spar Web at Fuel Boost Pump
Figure 102 (Sheet 1 of 2)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

A–A

1945726 S0000369242_V1

Allowable Damage Limits - Outer Wing Spar Web at Fuel Boost Pump
Figure 102 (Sheet 2 of 2)

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Masking Off - Fuel Boost Pump Bonding Strap Installed Area
Figure 103

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5. **Example to Calculate the Cross-Sectional Area Removed**

A. Calculate the damage depths and areas shown in Figure 104/ALLOWABLE DAMAGE 3, Detail A to find if you are permitted to use this procedure to remove the damage.

B. You can do a conservative calculation of the area loss as shown in the steps that follow. In this calculation, the cross sectional area loss is estimated as (width x depth). You can do an alternative calculation that does not assume that the damage removal cross section is in the shape of a rectangle.

NOTE: If you do an alternative calculation, then you must find the maximum area loss at a typical radial line at the damage location. To calculate the maximum area loss, use the maximum damage depths along the radial line within 1.0 in. (25.4 mm) of either side of the radial line at each location along the radial line.

NOTE: If the damage that you removed is more than the maximum allowed, contact The Boeing Company for repair instructions.

(1) Find the maximum damage depth (Y) for each damage location shown.

(a) At Damage A, the maximum damage depth is Ya1 at cross section A-A.

1) The damage depth Ya1 is 0.07 in. (1.8 mm).

- This is less than0.1 in. (2.54 mm) and is permitted.

(b) At Damage B, the maximum damage depth is Yb at cross section C-C.

1) The damage depth Yb is 0.08 in. (2.0 mm).

- This is less than 0.1 in. (2.54 mm) and is permitted.

(c) At Damage C, the maximum damage depth is Yc at cross section C-C.

1) The damage depth Yc is 0.09 in. (2.3 mm).

- This is less than 0.1 in. (2.54 mm) and is permitted.

(d) At Damage D, the maximum damage depth is Yd1 at cross section D-D.

1) The damage depth Yd1 is 0.06 in. (1.5 mm).

- This is less than 0.1 in. (2.54 mm) and is permitted.

(e) At Damage E, the maximum damage depth is Ye1 at cross section E-E.

1) The damage depth Ye1 is 0.05 in. (1.3 mm).

- This is less than 0.1 in. (2.54 mm) and is permitted.

(2) Calculate the maximum area loss of the cross section for each damage location.

(a) At Damage A:

1) For large damage locations, you must find the maximum area loss.

NOTE: The maximum area loss for the damage location might not occur at the maximum damage depth, nor at the widest radial line through the damage.

2) To calculate the area loss, use the damage width at a typical radial line, and the maximum damage depth within 1.0 in. (25.4 mm) of the typical radial line.

a) Damage A:

<1> Calculate the maximum area loss with the damage depth Ya1 (0.07 in. (1.8 mm)) and the damage width Xa2 (1.75 in. (44.4 mm)).

<2> (1.75 inches x 0.07 inch) = 0.12 in^2 (77.4 mm^2)

- This is less than 0.2 in^2 (129.0 mm^2) and is permitted.

(b) At Damage B and C:

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1) If Dimension Z is more than 1.0 in. (25.4 mm), then you must calculate the separate area loss at both Damage B and Damage C.

   a) Damage B:

      <1> Calculate the maximum area loss with the damage depth Yb (0.08 in. (2.0 mm)) and the damage width Xb (1.5 in. (38.1 mm)).

      <2> (1.5 inch x 0.08 inch) = 0.12 in^2 (77.4 mm^2)

      - This is less than 0.2 in^2 (129.0 mm^2) and is permitted.

   b) Damage C:

      <1> Calculate the maximum area loss with the damage depth Yc (0.09 in. (2.3 mm)) and the damage width Xc (1.75 in. (44.4 mm)).

      <2> (1.75 inch x 0.09 inch) = 0.16 in^2 (103.2 mm^2)

      - This is less than 0.2 in^2 (129.0 mm^2) and is permitted.

2) If Dimension Z is less than 1.0 in. (25.4 mm), then you must calculate the combined area loss at Damage B and Damage C.

   a) Calculate the separation between Damage B and Damage C as follows:

      NOTE: Refer to Flagnotes 3 and 4 on Detail A of Figure 104/ALLOWABLE DAMAGE 3 for an illustration of separation.

      <1> Draw a radial line centered between Damage B and Damage C.

      <2> Draw lines that intersect the radial line from the endpoints of the blend out boundaries for Damage B and Damage C. The lines must be perpendicular to the radial line.

      <3> If there is no overlap between Damage B and Damage C, then go to the next step. If there is overlap between Damage B and Damage C, then see Paragraph 5.B.(2)(c)/ALLOWABLE DAMAGE 3.

   b) Calculate the area loss at Damage B:

      <1> Calculate the maximum area loss with the damage depth Yb (0.08 in. (2.0 mm)) and the damage width Xb (1.5 in. (38.1 mm)).

      <2> (1.5 inch x 0.08 inch) = 0.12 in^2 (77.4 mm^2)

   c) Calculate the area loss at Damage C:

      <1> Calculate the maximum area loss with the damage depth Yc (0.09 in. (2.3 mm)) and the damage width Xc (1.75 in. (44.4 mm)).

      <2> (1.75 inch x 0.09 inch) = 0.16 in^2 (103.2 mm^2)

   d) Add the area loss at Damage B with the area loss at Damage C:

      <1> 0.12 in^2 (77.4 mm^2) +0.16 in^2 (103.2 mm^2) = 0.28 in^2 (180.6 mm^2)

      - This is more than 0.2 in^2 (129.0 mm^2) and is not permitted. Contact the Boeing Company for an approved repair.

(c) At Damage D and E:

1) If Dimension Z is less than 1.0 in. (25.4 mm), then go to the next step. If Dimension Z is more than 1.0 in. (25.4 mm), then see Paragraph 5.B.(2)(b)/ALLOWABLE DAMAGE 3.

2) Calculate the separation between Damage D and Damage E as follows:

   NOTE: Refer to Flagnotes 3 and 4 on Detail A of Figure 104/ALLOWABLE DAMAGE 3 for an illustration of separation.

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a) Draw a radial line centered between Damage D and Damage E.

b) Draw lines that intersect the radial line from the endpoints of the blend out boundaries for Damage B and Damage C. The lines must be perpendicular to the radial line.

c) If there is overlap between Damage D and Damage E, then go to the next step. If there is no overlap between Damage D and Damage E, then see Paragraph 5.B.(2)(b)/ALLOWABLE DAMAGE 3.

3) If there is overlap between Damage D and Damage E, and Dimension Z is less than 1.0 in. (25.4 mm), then calculate the cross-sectional area that you removed as follows. See Paragraph 5.B.(2)(b)/ALLOWABLE DAMAGE 3 if adjacent damage locations are separated along a radial line.

4) Calculate the area loss for the non-overlapped cross sections.

a) Damage D1:

<1> Calculate the maximum area loss with the damage depth Yd1 (0.06 in. (1.5 mm)) and the damage width Xd1 (0.9 in. (22.9 mm)).

<2> (0.9 inch x 0.06 inch) = 0.054 in^2 (34.84 mm^2)

b) Damage E1:

<1> Calculate the maximum area loss with the damage depth Ye1 (0.05 in. (1.3 mm)) and the damage width Xe1 (0.7 in. (17.8 mm)).

<2> (0.7 inch x 0.05 inch) = 0.035 in^2 (22.58 mm^2)

5) Calculate the area loss for the overlapped cross sections. Find the maximum area loss of the overlapped cross sections.

a) Damage D2:

<1> Calculate the maximum area loss with the damage depth Yd2 (0.04 in. (1.0 mm)) and the damage width Xd2 (0.3 in. (7.6 mm)).

<2> (0.3 inch x 0.04 inch) = 0.012 in^2 (7.74 mm^2)

b) Damage E2:

<1> Calculate the maximum area loss with the damage depth Ye2 (0.03 in. (0.8 mm)) and the damage width Xe2 (0.3 in. (7.6 mm)).

<2> (0.3 inch x 0.03 inch) = 0.009 in^2 (5.8 mm^2)

c) The maximum area loss for the overlapped cross sections is 0.012 in^2 (7.74 mm^2).

6) Calculate the total cross-sectional area that you removed. Calculate the sum of the areas of the non-overlapped cross sections with the maximum area loss of the overlapped cross sections.

a) (0.054 inch + 0.035 inch + 0.012 inch) = 0.10 in^2 (64.5 mm^2)

- This is less than 0.2 in^2 (129.0 mm^2) and is permitted.

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NOTES

- ALL SECTION VIEWS THROUGH DAMAGE LOCATIONS ARE ON A RADIAL LINE.

1 IF DIMENSION Z IS LESS THAN 1.0 INCH (25.4 mm), AREA LOSS DUE TO ADJACENT DAMAGE LOCATIONS MUST BE COMBINED.

2 YOU MUST CALCULATE THE MAXIMUM AREA LOSS USING:

- THE WIDTH OF THE DAMAGE AT A TYPICAL RADIAL LINE.
- THE MAXIMUM DAMAGE DEPTH WITHIN 1.0 INCH (25.4 mm) OF THE TYPICAL RADIAL LINE.

NOTE: THE MAXIMUM AREA LOSS FOR THE DAMAGE LOCATION MIGHT NOT OCCUR AT THE MAXIMUM DAMAGE DEPTH, NOR AT THE WIDEST RADIAL LINE THROUGH THE DAMAGE.

3 IF DIMENSION Z IS LESS THAN 1.0 INCH (25.4 mm) AND THERE IS SEPARATION BETWEEN THE DAMAGE LOCATIONS, THEN CALCULATE THE AREA LOSS SEPARATELY FOR EACH DAMAGE LOCATION. REFER TO PARAGRAPH 5.B.(2)(b).

4 IF DIMENSION Z IS LESS THAN 1.0 INCH (25.4 mm) AND THERE IS NO SEPARATION BETWEEN THE DAMAGE LOCATIONS, THEN CALCULATE THE AREA LOSS OF THE OVERLAPPED AND NON-OVERLAPPED SECTIONS FOR EACH DAMAGE LOCATION. REFER TO PARAGRAPH 5.B.(2)(c).

2009323 S0000395131_V1

Example of the area loss calculation
Figure 104 (Sheet 1 of 6)

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LEFT SIDE OUTER WING FRONT SPAR IS SHOWN
VIEW LOOKING AFT

A

1946246 S0000369244_V1

Example of the area loss calculation
Figure 104 (Sheet 2 of 6)

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2009584 S0000395147_V1

Example of the area loss calculation
Figure 104 (Sheet 3 of 6)

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2009601 S0000395148_V1

Example of the area loss calculation
Figure 104 (Sheet 4 of 6)

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1996257 S0000388964_V1

Example of the area loss calculation
Figure 104 (Sheet 5 of 6)

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D–D

E–E

2009617 S0000395151_V1

Example of the area loss calculation
Figure 104 (Sheet 6 of 6)

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REPAIR 1 - OUTER WING SPARS

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

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Outer Wing Spar Repair
Figure 201

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IDENTIFICATION 1 - OUTER WING FRONT SPAR FITTINGS

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

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G44255 S0006594338_V1

Outer Wing Front Spar Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
001A1002	Wing Right - Product Collector
112A0013	Collector - Functional, Product, Front Spar, Section 12, Left
112A0014	Collector - Functional, Product, Front Spar, Section 12, Right
112A2003	Front Spar Installation - Wing
112A2501	Terminal Fitting Installation - Front Spar, Wing BBL 70.85
112A2511	Terminal Fitting - Front Spar, Wing to Body Joint
112A7100	Nacelle Support Fitting Installation - Front Spar

3. Applicability

A. IDENTIFICATION 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. Refer to Table 2 for List of Materials in Figure 2.

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Nacelle Support Fitting, Outboard and Inboard Pitch Load (2)		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas.	
[2]	Nacelle Support Backup Fitting, Outboard Side Load		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[3]	Spar Fitting, Fixed Leading Edge, Canted Rib		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[4]	Actuator Support Fitting, Fixed Leading Edge at FSS 85.19		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[5]	Actuator Support Fitting at FSS 202.45		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[6]	Actuator Support Fitting, Fixed Leading Edge at FSS 322.71		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[7]	Actuator Support Fitting at FSS 451.97		7050-T7451 plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[8]	Nacelle Support Backup Fitting, Side Link		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas	
[9]	Nacelle Rib Post		7050-T73 forging as given in BMS 7-186. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[10]	Backup Fitting at IFSS 185		7050-T7451 machined plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[11]	Jack Fitting		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[12]	Backup Fitting at IFSS 132		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[13]	Terminal Fitting, Front Spar		7050-T74 forging as given in BMS 7-214. Refer to the production drawing for the machined areas (Grain direction controlled part) 7050-T7452 forging as given in BMS 7-214. Refer to the production drawing for the machined areas (Grain direction controlled part)	Cum Line Numbers 001 to 744 Cum Line Number 745 and on

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

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Outer Wing Front Spar Fitting Identification
Figure 2 (Sheet 1 of 2)

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Outer Wing Front Spar Fitting Identification
Figure 2 (Sheet 2 of 2)

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IDENTIFICATION 2 - OUTER WING REAR SPAR FITTINGS

1. Locator

U41087 S0000194906_V1

Outer Wing Rear Spar Fittings Location
Figure 1

2. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
001A1002	Wing Right - Product Collector
112A0011	Collector - Functional, Product, Rear Spar, Section 12, left
112A0012	Collector - Functional, Product, Rear Spar, Section 12, Right
112A1003	Spar Assembly - Rear, Wing
112A1501	Terminal Fitting Installation - Rear Spar, Wing BBL 70.85

3. Applicability

A. IDENTIFICATION 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line number 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

B. Refer to Table 2 for List of Materials in Figure 2.

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Support Fitting, Aileron Balance Panel		7050-T7451 machined plate as given in AMS 4050. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[2]	Support Fitting, Flap		7050-T7451 machined plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined areas (Grain direction controlled part)	
[3]	Support Bracket, Inboard Fixed Trailing Edge Panel		7050-T7451 machined plate as given in BMS 7-323, Type I. Refer to the production drawing for the machined areas. direction controlled part)	
[4]	Terminal Fitting		7050-T74 machined forging as given in BMS 7-214. Refer to the production drawing for the machined areas. direction controlled part)	
[5]	Backup Fitting, Main Landing Gear		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas. direction controlled part)	
[6]	Backup Fitting, Flap		7050-T7451 machined plate as given in BMS 7-323, Type III. Refer to the production drawing for the machined areas (Grain direction controlled part)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS LOOKING FORWARD

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

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Outer Wing Rear Spar Fittings Identification
Figure 2 (Sheet 1 of 2)

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H55636 S0006594347_V1

Outer Wing Rear Spar Fittings Identification
Figure 2 (Sheet 2 of 2)

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ALLOWABLE DAMAGE 1 - OUTER WING FRONT SPAR FITTINGS

1. Applicability

A. ALLOWABLE DAMAGE 1 gives the allowable damage limits for the outer wing front spar fittings shown in Location of the Outer Wing Front Spar, Figure 101/ALLOWABLE DAMAGE 1.

B. ALLOWABLE DAMAGE 1 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

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Location of the Outer Wing Front Spar
Figure 101

2. General

A. CDCCL – Refer to WINGS - GENERAL, 57-00-00 for important information on Critical Design Configuration Control Limitations (CDCCLs). Fasteners that penetrate the fuel tank must have protection against lightning strikes and electrical hot shorts. Refer to LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS , 51-80-01 for specific fastener selection and installation requirements.

NOTE: This is applicable to Airworthiness Limitation 28-AWL-12.

B. These data have been reviewed by the Boeing Organization Designation Authorization for fuel tank ignition prevention requirements, and no further approval from the FAA Oversight Office is required to satisfy the requirements of CDCCL 28-AWL-12 provided the repair or alteration is accomplished in accordance with these instructions.

C. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 and Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limits.

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D. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

E. After you remove the damage, do the procedures that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of all parts except the canted spar fitting.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked parts. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the surfaces of the backup fittings, rib post fittings, and terminal fittings. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Apply a layer of BMS 10-11, Type I primer to the surfaces of the pitch load, actuator support, and the canted spar fittings. Refer to SOPM 20-41-02.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
STRUCTURAL PART	**PARAGRAPH**
End Pads	4.A
Gussets	4.A
Webs	4.A
Attach Flanges	4.A
Pitch Load Fitting Lugs	4.B
Actuator Support Fitting Lugs	4.C

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-80-01	LIGHTNING, ELECTRICAL WIRING AND EQUIPMENT FAULT CURRENT REQUIREMENTS
57-00-00	WINGS - GENERAL

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(Continued)

Reference	Title
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-50-03	Bearing and Bushing Replacement

4. Allowable Damage Limits

A. End Pads, Gussets, and Attach Flanges

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E , F , and G .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Pitch Load Fitting Lugs

NOTE: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore to a maximum oversize diameter of 0.06 inch more than the initial bore diameter if:

- There is no damage on the edge of the lug
- You follow the bushing removal procedures as given in SOPM 20-50-03.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail A .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail A .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Actuator Support Fitting Lugs

NOTE: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore to a maximum oversize diameter of 0.06 inch more than the initial bore diameter if:

- There is no damage on the edge of the lug
- You follow the bushing removal procedures as given in SOPM 20-50-03.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail B .

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(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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NOTES

- ALL FITTINGS IDENTIFIED ARE MADE OF MACHINED ALUMINUM.

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Outer Wing Front Spar Fittings
Figure 102 (Sheet 1 of 5)

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G58795 S0006594354_V1

Outer Wing Front Spar Fittings
Figure 102 (Sheet 2 of 5)

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G58835 S0006594355_V1

Outer Wing Front Spar Fittings
Figure 102 (Sheet 3 of 5)

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G58931 S0006594356_V1

Outer Wing Front Spar Fittings
Figure 102 (Sheet 4 of 5)

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RIB POST FITTING

I

BACKUP FITTING

J

G58961 S0006594357_V1

Outer Wing Front Spar Fittings
Figure 102 (Sheet 5 of 5)

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G57861 S0006594358_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 5)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

G61184 S0006594359_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 5)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

A–A

D

THE REMOVAL OF MATERIAL AROUND THREE FASTENERS IN A GROUP OF TEN IS PERMITTED TO A DEPTH OF X MAXIMUM

REMOVE THE INITIAL FASTENERS BEFORE THE DAMAGED MATERIAL IS REMOVED. INSTALL THE SAME TYPE AND SIZE (UP TO THE FIRST OVERSIZE) FASTENERS AFTER THE REWORK IS COMPLETED

NOTES

- CDCCL - REFER TO WINGS-GENERAL, 57-00-00 FOR IMPORTANT INFORMATION ON CRITICAL DESIGN CONFIGURATION CONTROL LIMITATIONS (CDCCLS).
- THIS IS APPLICABLE TO AIRWORTHINESS LIMITATION 28-AWL-12.

REMOVAL OF CORROSION AROUND THE FASTENERS

E

B–B

G57865 S0006594360_V2

Allowable Damage Limits
Figure 103 (Sheet 3 of 5)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 1.0 INCH (25.4 mm)

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

F

C–C

G57867 S0006594361_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 5)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

G

G60754 S0006594362_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 5)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

(A)

G57871 S0006594363_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

B

G60765 S0006594364_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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C–C

D–D 1

NOTES

1 THESE ALLOWABLE DAMAGE LIMITS ARE APPLICABLE ONLY IF THERE IS NO DAMAGE IN THE BORE OF THE LUG.

G61652 S0006594365_V1

Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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ALLOWABLE DAMAGE 2 - OUTER WING REAR SPAR FITTINGS

1. Applicability

A. ALLOWABLE DAMAGE 2 gives the allowable damage limits for the outer wing rear spar fittings shown in Location of the Outer Wing Rear Spar Fittings, Figure 101/ALLOWABLE DAMAGE 2.

B. ALLOWABLE DAMAGE 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglet.

G53502 S0006594367_V1

Location of the Outer Wing Rear Spar Fittings
Figure 101

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VIEW IS LOOKING FORWARD

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: ALL PARTS ARE MADE OF MACHINED ALUMINUM.

G53505 S0006594368_V1

Location of the Outer Wing Rear Spar Fittings
Figure 102 (Sheet 1 of 2)

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NOTE: ALL PARTS ARE MADE OF MACHINED ALUMINUM.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G53511 S0006594369_V1

Location of the Outer Wing Rear Spar Fittings
Figure 102 (Sheet 2 of 2)

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of all parts except to the aileron balance hinge support fittings.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply a layer of BMS 10-20, Type II primer to the surfaces of the backup fittings. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Apply a layer of BMS 10-11, Type I primer to the surfaces of the support panel and aileron balance fittings. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-10-03	SHOT PEENING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-50-03	Bearing and Bushing Replacement

4. Allowable Damage Limits

A. Support Panel Fittings, Support Fittings, Terminal Fitting, and Backup Fittings

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

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(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Actuator Balance Hinge Support Fittings

NOTE: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore to a maximum oversize diameter of 0.06 inch more than the initial bore diameter if:

- There is no damage on the edge of the lug
- You follow the bushing removal procedures as given in SOPM 20-50-03.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F , G , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

G53518 S0006594370_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 5)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

REMOVAL OF CORROSION AROUND THE FASTENERS

E

G53524 S0006594371_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 5)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

F

C–C

G53526 S0006594372_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 5)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

G

G53528 S0006594373_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 5)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

H

G53529 S0006594374_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 5)

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REPAIR 1 - OUTER WING FRONT SPAR FITTINGS

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G97237 S0006594377_V2

Location of the Outer Wing Front Spar
Figure 201

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REPAIR 2 - OUTER WING REAR SPAR FITTINGS - MAIN LANDING GEAR FORWARD TRUNNION SUPPORT FITTING CORROSION REPAIR

1. Applicability

A. This repair is applicable to corrosion damage to the support fitting of the main landing gear forward trunnion. Refer to Figure 201/REPAIR 2 for the support fitting locations.

B. This repair is applicable to corrosion damage on the surfaces of the support fitting shown in Figure 201/REPAIR 2.

C. This repair is not applicable to crack damage to the support fitting. If you find a crack at a support fitting lug, replace the support fitting.

2. General

A. Repair 2 gives instructions for a permanent repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. This repair uses the interference fit procedure to install the replacement bushings for a support fitting that is shown in Figure 201/REPAIR 2.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

1317269 S0000239182_V1

Main Landing Gear Forward Trunnion Support Fitting Location
Figure 201

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-20-13	SURFACE ROUGHNESS FINISH REQUIREMENTS
AMM 32-11-83 P/B 401	MAIN LANDING GEAR FORWARD TRUNNION BEARING ASSEMBLY - REMOVAL/INSTALLATION
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-30-03	General Cleaning Procedures
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating
SOPM 20-50-03	Bearing and Bushing Replacement
737 NDT Part 6, 51-00-00, Procedure 1	Fastener Holes in Aluminum Parts (Meter Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the main landing gear forward trunnion bearing assembly as given in AMM PAGEBLOCK 32-11-83/401.

CAUTION: USE CARE WHEN YOU REMOVE THE BUSHINGS. THESE BUSHINGS WERE INSTALLED WITH AN INTERFERENCE FIT. IF YOU DO NOT, DAMAGE TO THE FITTING CAN OCCUR.

C. Remove the bushings from the support fitting where the corrosion damage occurs. Keep the removed bushings, if they are in satisfactory condition.

D. Do a detailed visual inspection of the damaged area. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02 for the inspection procedure.

CAUTION: USE CARE WHEN YOU WORK ON THE SURFACE OF THE FITTING HOLE. SOME TOOLS AND REPAIR PROCEDURES CAN CAUSE DAMAGE TO THE FITTING HOLE. THE RESULT CAN BE A HOLE THAT IS LARGER THAN THE MAXIMUM DIMENSION THAT IS PERMITTED.

E. If there is corrosion damage on the outer surface of the support fitting around the bore, do as follows:

(1) Remove the damaged material to a maximum depth of 0.210 in. (5.3 mm). Make the surface flat and parallel to the initial surface. Refer to Figure 202/REPAIR 2, Detail A, Detail B, and INSPECTION AND REMOVAL OF DAMAGE, 51-10-02

(2) Inspect the area for cracks after corrosion removal. Refer to 737 NDT Part 6, 51-00-00, Procedure 4.

(3) Spotface and/or chamfer the fitting with the requirements below:

(a) Chamfer the hole 0.030 in. (0.762 mm) and 0.040 in. (1.016 mm).

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(b) Transition fillet radius of the spot face to be a minimum of 0.250 in. (6.350 mm).

(c) Break all sharp edges from the transition radius and hand blend into the fitting radius.

(d) Spotface diameter to be such that the washer does not ride the radius of the spotface. Spotface diameter not to exceed 3.0 inches.

F. If there is corrosion damage on the inner surface of the bore(s) of the support fitting, do as follows:

(1) Get or make the tools to line ream the two adjacent support fitting bores as follows. Refer to tool drawing MIT115A5002.

(a) Use a long piloted reamer to keep the bores coaxial to each other. The pilot part of the reamer must be non-cutting to prevent damage to the bore.

(b) Use a guide bushing in the adjacent bore to support the reamer, and protect the surfaces of the bore.

(2) Line ream and hone the bore(s) to remove the damaged material. Remove the damaged material up to a maximum bore diameter (D1) of 1.620 in. (41.15 mm). Refer to Figure 202/REPAIR 2, Section A-A, INSPECTION AND REMOVAL OF DAMAGE, 51-10-02, and tool drawing MIT115A5002.

(a) Make sure that the bores are coaxial to each other.

(b) Make the surface finish of the bores equal to 63 microinches (1.6 micrometers) Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

G. Remove the sharp edges and burrs from the cut surfaces.

WARNING: KEEP THE SOLVENTS AWAY FROM SOURCES OF HEAT, FIRE, OR SPARKS. HEAT OR SPARKS CAN CAUSE AN EXPLOSION. DO NOT LET THE SOLVENTS GET INTO YOUR EYES OR ON YOUR SKIN OR CLOTHING. USE EYE PROTECTION. MAKE SURE THAT YOU USE AIRFLOW EQUIPMENT WHEN YOU DO WORK IN A CLOSED SPACE. IF YOU DO NOT OBEY, YOU CAN CAUSE INJURY TO PERSONS.

H. Clean the reworked areas of the support fitting with solvent. Refer to SOPM 20-30-03 for applicable solvents and procedures.

I. Do a penetrant inspection of the reworked areas to make sure that there is no other damage. Refer to SOPM 20-20-02. If you find more damage, do steps E through H, Paragraph 4./REPAIR 2 again.

NOTE: As an alternative, you can do a High Frequency Eddy Current (HFEC) inspection of the reworked area. Refer to 737 NDT Part 6, 51-00-00, Procedure 1 and 737 NDT Part 6, 51-00-00, Procedure 4.

J. Shot peen the outer surfaces of the support fitting where the material was removed, but do not apply the shot peen procedure to the bores of the support fitting.

(1) Keep 0.010A shot peening intensity and 2.0 coverage (manual procedure).

(2) Refer to SHOT PEENING, 51-20-06 for the shot number.

(3) Refer to SOPM 20-10-03 for shot peen procedure.

K. Make the surface finish of the shot peened surfaces of the support fitting equal to 125 microinches Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

L. Apply a chemical conversion coating to the bare surfaces of the support fitting. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

M. Apply a layer of BMS 10-11, Type I primer to the bare surfaces of the support fitting, but do not apply primer to the surfaces of the bores. Refer to SOPM 20-41-02.

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N. Make the repair parts. Refer to Table 201/REPAIR 2 and Figure 203/REPAIR 2.

(1) Make or get the part [1] bushing with one of the two alternatives that follow:

(a) If you do not ream the support fitting bore, make part [1] bushing. Refer to Paragraph 5.A./REPAIR 2 for the necessary outer diameter dimension. Also, you can use the initial bushing that you removed, if it is not damaged, or get and use part number 115A5230-2 bushing from The Boeing Company for the part [1] bushing. Make sure that the diameter of the support fitting bore is 1.5602 in. (39.629 mm) to 1.5610 in. (39.649 mm). Refer to Paragraph 5.C./REPAIR 2 to calculate the necessary length of the part [1] bushing.

(b) If you ream the support fitting bore, make the part [1] bushing as shown in Figure 203/REPAIR 2, Detail A. Refer to Paragraph 5.B./REPAIR 2 to calculate the necessary outer diameter of the part [1] bushing. Refer to Paragraph 5.C./REPAIR 2 to calculate the necessary length of the part [1] bushing.

(2) Make the part [2] and the part [3] washers to replace the initial washers. Make the part [2] washer as shown in Figure 203/REPAIR 2, Detail C, and the part [3] washer as shown in Figure 203/REPAIR 2, Detail D. Refer to Paragraph 5.C.(2)/REPAIR 2 for the part [2] washer, and Paragraph 5.C.(3)/REPAIR 2 for the part [3] washer to calculate the necessary washer thickness.

Table 201: Repair Material

ITEM	PART	QUANTITY	MATERIAL
[1]	Bushing	4	Use copper-beryllium (Cu – Be) tubing as given in AMS 4535. As an alternative, you can use copper-beryllium bar as given in AMS 4533. Refer to Figure 203/REPAIR 2, Detail A for the necessary dimensions.
[2]	Washer for the Fuse Pin Head Side	1	Use 15-5PH CRES with a heat treat of 150 to 170 KSI as given in AMS 5659. Refer to Figure 203/REPAIR 2, Detail C for the necessary dimensions.
[3]	Washer for the Fuse Pin Nut Side	1	Use 15-5PH CRES with a heat treat of 150 to 170 KSI as given in AMS 5659. Refer to Figure 203/REPAIR 2, Detail D for the necessary dimensions.

O. Apply the cadmium plating (Type II, Class 2) to the repair parts. It is optional to apply cadmium plating to the surface of the inner diameter of the part [1] bushing. Make the thickness of cadmium plating from 0.0003 in. (0.0076 mm) to 0.0005 in. (0.0127 mm). Refer to SOPM 20-42-05.

WARNING: USE PROTECTION WHEN YOU DO WORK WITH LIQUID NITROGEN. MAKE SURE THAT THE AREA HAS A GOOD FLOW OF CLEAN AIR. THE TEMPERATURE FOR LIQUID NITROGEN IS APPROXIMATELY MINUS 320°F (MINUS 196°C). IF YOU DO NOT OBEY, INJURY TO PERSONS CAN BE THE RESULT.

P. Use the shrink fit procedure to install the part [1] bushing with BMS 5-95 sealant as shown in Figure 202/REPAIR 2, Detail C. Use liquid nitrogen at -320°F (-196°C). Refer to SOPM 20-50-03.

(1) Make sure that the part [1] bushing is installed in the correct location.

(2) Hold the assembly in position until the assembly is at room temperature.

Q. Get or make the tools that follow to line ream the bushing through the pair of bushing holes to keep the bushing holes coaxial. Refer to tool drawing MIT115A5002.

(1) Use a long piloted reamer to keep the bushing holes coaxial. The pilot part of the reamer must be non-cutting to prevent damage to the bushing.

(2) Use a guide bushing to slip into the adjacent bushing to support the reamer, and protect the bushing holes.

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R. Line ream and hone the holes of the part [1] bushings to the final inner diameter (D2) of 1.2905 in. (32.78 mm) to 1.2915 in. (32.80 mm). Refer to tool drawing MIT115A5002 for reaming and honing holes.

(1) Make sure that the pairs of bushing holes are coaxial as given in Figure 202/REPAIR 2, Detail C, Section B-B. If necessary, hone the pair of the replacement bushings at the same time to keep them coaxial.

(2) Make sure the surface finish of the holes is equal to 32 microinches Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

S. If you install new bushings to the support fitting then:

(1) Machine the flange of the bushings to the dimension shown in Figure 202/REPAIR 2, Detail C, section B-B. It is not necessary to re-cad plate the bushing after you machine the bushing flange.

(2) Machine the chamfer of the bushing flange as shown in Figure 202/REPAIR 2, Detail C, section B-B.

T. Clean the bushing holes to make sure that all the honing debris and machining oil are removed. Refer to SOPM 20-30-03.

U. Install the main landing gear forward trunnion bearing assembly as given in AMM PAGEBLOCK 32-11-83/401, but:

(1) If you removed material from around the bores of the support fitting, do the procedures that follow:

(a) Install the part [2] repair washer with BMS 3-38 corrosion inhibiting compound at the head side of the fuse pin. The part [2] washer replaces the initial special washer. Discard the initial special washer. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01 and Figure 204/REPAIR 2. Make sure that the washer is concentric with the support fitting bore when you install the fuse pin.

(b) Install the part [3] repair washer with BMS 3-38 corrosion inhibiting compound at the nut side of the fuse pin. The part [3] washer replaces the initial special washer. Discard the initial special washer. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01 and Figure 204/REPAIR 2. Make sure that the washer is concentric with the support fitting bore when you install the fuse pin.

(c) Apply a fillet seal with BMS 5-95 sealant to the edges of the part [2] and the part [3] washers. Refer to REPAIR SEALING, 51-20-05.

V. Install the components that you removed for access.

5. Repair Parts Dimensions

A. If you do not ream the bore of the support fitting, make the diameter of the bore 1.5602 in. (39.629 mm) to 1.5610 in. (39.649 mm). Make the outer diameter of the part [1] bushing 1.5626 in. (39.690 mm) to 1.5638 in. (39.721 mm) for an interference fit.

NOTE: As an alternative, you can get and use a part number 115A5230-2 bushing from The Boeing Company for the part [1] bushing, or you can use the initial bushing that you removed, if it is in satisfactory condition.

B. If you ream the bore of the support fitting, make the outer diameter of the part [1] bushing 0.0016 in. (0.041 mm) to 0.0036 in. (0.091 mm) larger for an interference fit. Refer to Figure 203/REPAIR 2, Detail A, Section A-A to identify the outer diameter (D3) of the part [1] bushing. Use the steps that follow to calculate the outer diameter of the part [1] bushing:

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(1) After you remove the damage, measure the diameter of the bore of the support fitting. Make a record of the minimum and maximum value of the diameter of the bore. Make sure that the maximum diameter of the bore (D1) is not larger than 1.620 in. (41.15 mm).

(2) Calculate the sum of the minimum diameter of bore and 0.0036 in. (0.091 mm). Use the result value as a maximum outer diameter of the part [1] bushing.

(3) Calculate the sum of the maximum diameter of bore and 0.0016 in. (0.041 mm). Use the result value as a minimum outer diameter of the part [1] bushing.

(4) Example: Assume that you ream the bore of the support fitting to a diameter of 1.6000 in. (40.640 mm) to 1.6008 in. (40.660 mm):

(a) The maximum outer diameter of the part [1] bushing equals the sum of 1.6000 in. (40.640 mm) and 0.0036 in. (0.091 mm) = 1.6036 in. (40.731 mm).

(b) The minimum outer diameter of the part [1] bushing equals the sum of 1.6008 in. (40.660 mm) and 0.0016 in. (0.041 mm) = 1.6024 in. (40.701 mm).

(c) The outer diameter (D3) of the part [1] bushing must be 1.6024 in. (40.701 mm) to 1.6036 in. (40.731 mm).

C. If you removed the damage from the outer surface of the support fitting around the bore:

(1) Make the part [1] bushing with a length that is shorter than the length of the initial bushing. Refer to Figure 203/REPAIR 2, Detail A, Section A-A to identify the length (L1) of the part [1] bushing. Use the steps that follow to calculate the necessary length (L1) of the part [1] bushing:

(a) Measure the depth of the material that you removed from the outer surface of the support fitting around the bore. Refer to Figure 202/REPAIR 2, Section A-A to identify the depth (X) of the material that you removed. Make sure that the maximum depth of removed material is not larger than 0.210 in. (5.3 mm). Make a record of the minimum and maximum values of the removed material depth.

(b) Calculate the sum of the minimum and maximum values of removed material depth.

(c) Divide the sum of the minimum and maximum values of removed material depth by two. Use the result value as the average depth of removed material.

(d) Subtract the value of the average depth of removed material from 1.155 in. (29.34 mm). Use the result value as the nominal length of the part [1] bushing.

(e) Add 0.005 in. (0.127 mm) to the nominal length of the part [1] bushing. Use the result value as a maximum length of the part [1] bushing.

(f) Subtract 0.005 in. (0.127 mm) from the nominal length of the part [1] bushing. Use the result value as a minimum length of the part [1] bushing.

(g) Example: Assume that you remove a depth of 0.100 in. (2.54 mm) to 0.102 in. (2.59 mm) of material from the outer surface of the support fitting:

1) The sum of the minimum and maximum values of removed material depth is equal to 0.100 in. (2.54 mm) plus 0.102 in. (2.59 mm) = 0.202 in. (5.13 mm).

2) The average depth of removed material is equal to 0.202 in. (5.13 mm) divided by 2 = 0.101 in. (2.57 mm).

3) The nominal length of the part [1] bushing is equal to 1.155 in. (29.34 mm) minus 0.101 in. (2.57 mm) = 1.054 in. (26.77 mm).

4) The maximum length of the part [1] bushing is equal to the sum of 1.054 in. (26.772 mm) and 0.005 in. (0.127 mm) = 1.059 in. (26.899 mm).

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5) The minimum length of the part [1] bushing is equal to 1.054 in. (26.772 mm) minus 0.005 in. (0.127 mm) = 1.049 in. (26.645 mm).

6) The length (L1) of the part [1] bushing must be 1.049 in. (26.645 mm) to 1.059 in. (26.899 mm).

(2) Make the part [2] washer with a thickness that is larger than the thickness of the initial special washer. Refer to Figure 203/REPAIR 2, Detail C and Figure 204/REPAIR 2 to identify the part [2] washer. Use the steps that follow to calculate the necessary thickness (T1) of the part [2] washer:

(a) Measure the depth of the material that you removed from the outer surface of the support fitting around the bore. Refer to Figure Figure 202/REPAIR 2, Section A-A to identify the depth (X) of the material that you removed. Make sure that the maximum depth of removed material is not larger than 0.210 in. (5.3 mm). Make a record of the minimum and maximum values of the removed material depth.

(b) Calculate the sum of the minimum and maximum values of removed material depth.

(c) Divide the sum of the minimum and maximum values of removed material depth by two. Use the result value as the average depth of removed material.

(d) Calculate the sum of the average depth of removed material and 0.070 in. (1.778 mm). Use the result value as a maximum thickness of the part [2] washer.

(e) Calculate the sum of the average depth of removed material and 0.060 in. (1.524 mm). Use the result value as a minimum thickness of the parts [2] washer.

(f) Example: Assume that the material you removed from the outer surface of the support fitting is a depth of 0.100 in. (2.540 mm) to 0.102 in. (2.591 mm).

1) The sum of the minimum and maximum values of removed material depth is equal to 0.100 in. (2.540 mm) plus 0.102 in. (2.591 mm) = 0.202 in. (5.131 mm)

2) The average depth of removed material is equal to 0.202 in. (5.131 mm) divided by 2 = 0.101 in. (2.565 mm).

3) The maximum thickness of the part [2] washer is equal to the sum of 0.101 in. (2.565 mm) and 0.070 in. (1.778 mm) = 0.171 in. (4.343 mm)

4) The minimum thickness of the part [2] washer is equal to the sum of 0.101 in. (2.565 mm) and 0.060 in. (1.524 mm) = 0.161 in. (4.089 mm)

5) The thickness (T1) of the part [2] washer is 0.161 in. (4.089 mm) to 0.171 in. (4.343 mm)

(3) Make the part [3] washer with a thickness that is larger than the thickness of the initial special washer. Refer to Figure 203/REPAIR 2, Detail D and Figure 204/REPAIR 2 to identify the part [3] washer. Use the steps shown in Paragraph 5.C.(2)(a)/REPAIR 2 through Paragraph 5.C.(2)(f)/REPAIR 2 to calculate the necessary thickness (T1) of the part [3] washer.

D. If there is no damage to be removed from the outer surface of the support fitting around the bore:

(1) Make the part [2] washer thickness the same as the initial special washer thickness. Refer to Figure 203/REPAIR 2, Detail C and Figure 204/REPAIR 2 to identify the part [2] washer.

(2) Make the part [3] washer thickness the same as the initial special washer thickness. Refer to Figure 203/REPAIR 2, Detail D and Figure 204/REPAIR 2 to identify the part [3] washer.

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- ZONE SHOWS THE TYPICAL LOCATIONS OF THE CORROSION DAMAGES. THE DAMAGED MATERIAL MUST BE REMOVED AS SHOWN IN SECTION A-A.

REMOVAL OF THE DAMAGE

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 1 of 4)

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(TYPICAL)

B

INSTALLATION OF THE PART [1] BUSHINGS

C

1317271 S0000239554_V2

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 2 of 4)

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B–B

1317272 S0000239555_V2

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 3 of 4)

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NOTES

1 HOLE DIAMETER TOLERANCE MUST NOT BE MORE THAN 0.0008 INCH (0.02 mm).

2 DO NOT REMOVE MATERIAL FROM THE GUSSETS.

3 IF THE BORE WAS REAMED TO AN OVERSIZE DIAMETER, MAKE A CHAMFER AT THIS EDGE. THE CHAMFER IS FROM 0.030 INCH (0.762 mm) TO 0.040 INCH (1.02 mm) AND 45 DEGREES.

4 KEEP THE CENTER OF THE HOLES IN THE INITIAL POSITION.

5 MAKE THE INDICATED AREA FLAT AFTER DAMAGE REMOVAL. IT IS NECESSARY FOR CORRECT INSTALLATION OF THE PARTS [2] AND [3] WASHERS.

6 REAM AND HONE THE BUSHING HOLE TO THE FINAL INNER DIAMETER AFTER THE INSTALLATION OF THE PART [1] BUSHING.

7 ADJUST THE DIAMETER TO PREVENT DAMAGE TO THE VERTICAL FLANGES.

8 MAKE SURE THAT THE BUSHING IS BELOW FLUSH TO SPECIFIED DIMENSIONS.

1317273 S0000239556_V2

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 4 of 4)

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PART [1] BUSHING

A

A-A

1317274 S0000239557_V2

Repair Parts
Figure 203 (Sheet 1 of 4)

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1317275 S0000239558_V2

Repair Parts
Figure 203 (Sheet 2 of 4)

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1562060 S0000289287_V2

Repair Parts
Figure 203 (Sheet 3 of 4)

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NOTES

- MAKE THE SURFACE FINISH OF ALL SURFACES (EXCEPT AS NOTED) EQUAL TO:
  - 63 MICROINCHES (1.6 MICROMETERS) Ra OR SMOOTHER FOR THE PART [1] BUSHING
  - 125 MICROINCHES (3.2 MICROMETERS) Ra OR SMOOTHER FOR THE PART [2] AND THE PART [3] WASHERS.
- REMOVE ALL SHARP EDGES. MAKE A RADIUS THAT IS 0.005 TO 0.015 INCH (0.127 TO 0.381 mm) AT THE EDGE.
- APPLY CADMIUM PLATE TYPE II, CLASS 2, TO THE PART [1] BUSHING, THE PART [2] AND THE PART [3] WASHERS.
- ALL DIMENSIONS ARE SHOWN AFTER THE PLATING PROCEDURE.

1 INNER DIAMETER OF THE BUSHING HOLE BEFORE INSTALLATION IS SHOWN. REAM AND HONE THE BUSHING HOLE TO THE FINAL INNER DIAMETER AFTER THE INSTALLATION OF THE PART [1] BUSHING. REFER TO FIGURE 202, REPAIR 2.

2 IT IS PERMITTED TO MAKE THE SURFACE FINISH OF THE INNER DIAMETER OF THE BUSHING HOLE TO 125 MICROINCHES (3.175 MICROMETERS) Ra OR SMOOTHER. CADMIUM PLATING IS OPTIONAL ON THE INSIDE DIAMETER OF THE BUSHING HOLE.

3 REFER TO PARAGRAPH 5/REPAIR 2 TO CALCULATE THE NECESSARY VALUE.

4 TOTAL RUNOUT TOLERANCE TO THE DATUM -A- MUST NOT BE MORE THAN 0.0010 INCH (0.0254 mm).

5 RUNOUT TOLERANCE TO THE DATUM -A- MUST NOT BE MORE THAN 0.0100 INCH (0.254 mm).

6 THE INDICATED SURFACE MUST BE PERPENDICULAR TO THE DATUM -A- WITH A MAXIMUM TOLERANCE OF 0.002 INCH (0.05 mm).

1317276 S0000239559_V2

Repair Parts
Figure 203 (Sheet 4 of 4)

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TYPICAL INSTALLATION OF THE PARTS [2] AND [3] WASHERS

NOTES

1 REFER TO AMM 32-11-83.

2 REMOVE AND DISCARD THE INITIAL SPECIAL WASHER THEN INSTALL THE PART [2] WASHER AT THE HEAD SIDE OF THE FUSE PIN.

3 REMOVE AND DISCARD THE INITIAL SPECIAL WASHER THEN INSTALL THE PART [3] WASHER AT THE NUT SIDE OF THE FUSE PIN.

1317285 S0000239560_V1

Installation of the Repair Washer
Figure 204

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REPAIR 3 - MAIN LANDING GEAR FORWARD TRUNNION SUPPORT FITTING CORROSION REPAIR

1. Applicability

A. This repair is applicable to corrosion damage to the support fitting of the main landing gear forward trunnion. Refer to Figure 201/REPAIR 3 for the support fitting locations.

B. This repair is applicable to corrosion damage on the surfaces of the support fitting shown in Figure 201/REPAIR 3.

C. This repair is not applicable to crack damage to the support fitting and corrosion in the bushing holes. If you find corrosion in the bushing holes use REPAIR 2. If you find a crack at a support fitting lug, replace the support fitting.

2. General

A. Repair 3 gives instructions for a permanent repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. This repair uses the interference fit procedure to install the replacement bushings for a support fitting that is shown in Figure 201/REPAIR 3.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

1700249 S0000309330_V1

Main Landing Gear Forward Trunnion Support Fitting Location
Figure 201

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-20-13	SURFACE ROUGHNESS FINISH REQUIREMENTS
AMM 32-11-83 P/B 401	MAIN LANDING GEAR FORWARD TRUNNION BEARING ASSEMBLY - REMOVAL/INSTALLATION
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-30-03	General Cleaning Procedures
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating
SOPM 20-50-03	Bearing and Bushing Replacement
737 NDT Part 6, 51-00-00, Procedure 1	Fastener Holes in Aluminum Parts (Meter Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instruction

A. Get access to the damaged area.

B. Remove the main landing gear forward trunnion bearing assembly as given in AMM PAGEBLOCK 32-11-83/401.

CAUTION: USE CARE WHEN YOU REMOVE THE BUSHINGS. THESE BUSHINGS WERE INSTALLED WITH AN INTERFERENCE FIT. IF YOU DO NOT, DAMAGE TO THE FITTING CAN OCCUR.

C. Remove the bushings from the support fitting where the corrosion damage occurs. Keep the removed bushings, if they are in satisfactory condition.

D. Do a detailed visual inspection of the damaged area. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02 for the inspection procedure.

E. If there is corrosion damage on the inner surface of the bore(s) of the support fitting, stop and use REPAIR 2 procedure.

CAUTION: USE CARE WHEN YOU WORK ON THE SURFACE OF THE FITTING HOLE. SOME TOOLS AND REPAIR PROCEDURES CAN CAUSE DAMAGE TO THE FITTING HOLE. THE RESULT CAN BE A HOLE THAT IS LARGER THAN THE MAXIMUM DIMENSION THAT IS PERMITTED.

F. If there is corrosion damage on the outer surface of the support fitting around the bore, remove the damaged material to a maximum depth of 0.210 in. (5.3 mm). Make the surface flat and parallel to the initial surface. Refer to Figure 202/REPAIR 3, Detail A, Detail B, and INSPECTION AND REMOVAL OF DAMAGE, 51-10-02

G. Inspect the area for cracks after corrosion removal. Refer to 737 NDT Part 6, 51-00-00, Procedure 4.

H. Spotface and/or chamfer the fitting with the requirements below:

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(1) Chamfer the hole 0.030 in. (0.762 mm) to 0.040 in. (1.016 mm).

(2) Transition fillet radius of the spotface to be a minimum of 0.250 in. (6.350 mm).

(3) Break all sharp edges from the transition radius and hand blend into the fitting radius.

(4) Spotface diameter to be such that the washer does not ride the radius of the spotface. Spotface diameter not to exceed 3.0 inches.

I. Remove the sharp edges and burrs from the cut surfaces.

WARNING: KEEP THE SOLVENTS AWAY FROM SOURCES OF HEAT, FIRE, OR SPARKS. HEAT OR SPARKS CAN CAUSE AN EXPLOSION. DO NOT LET THE SOLVENTS GET INTO YOUR EYES OR ON YOUR SKIN OR CLOTHING. USE EYE PROTECTION. MAKE SURE THAT YOU USE AIRFLOW EQUIPMENT WHEN YOU DO WORK IN A CLOSED SPACE. IF YOU DO NOT OBEY, YOU CAN CAUSE INJURY TO PERSONS.

J. Clean the reworked areas of the support fitting with solvent. Refer to SOPM 20-30-03 for the applicable solvents and procedures.

K. Do a penetrant inspection of the reworked areas to make sure that there is no other damage. Refer to SOPM 20-20-02. If you find more damage, do steps E through J of Paragraph 4./REPAIR 3 again.

NOTE: As an alternative, you can do a High Frequency Eddy Current (HFEC) inspection of the reworked area. Refer to 737 NDT Part 6, 51-00-00, Procedure 1 and 737 NDT Part 6, 51-00-00, Procedure 4.

L. Shot peen the outer surfaces of the support fitting where the material was removed, but do not apply the shot peen procedure to the bores of the support fitting.

(1) Keep 0.010A shot peening intensity and 2.0 coverage (manual procedure).

(2) Refer to SHOT PEENING, 51-20-06 for the shot number.

(3) Refer to SOPM 20-10-03 for shot peen procedure.

M. Make the surface finish of the shot peened surfaces of the support fitting equal to 125 microinches Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

N. Apply a chemical conversion coating to the bare surfaces of the support fitting. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

O. Apply a layer of BMS 10-11, Type I primer to the bare surfaces of the support fitting, but do not apply primer to the surfaces of the bores. Refer to SOPM 20-41-02.

P. Make the repair parts. Refer to Table 201/REPAIR 3 and Paragraph 5./REPAIR 3.

(1) Make or get the part [1] bushing with one of the two alternatives that follow:

(a) Use the initial undamaged bushing. Make sure the bushing is in good condition. Refer to Paragraph 5./REPAIR 3 for the necessary outer diameter dimension. Make sure that the diameter of the support fitting bore is 1.5602 in. (39.629 mm) to 1.5610 in. (39.649 mm).

(b) Get a new bushing. Order part number 115A5230-2 bushing from The Boeing Company or manufacture the part [1] bushing as given in Drawing Number 115A5230-2 and Figure 203 (Sheet 1).

(2) Make the part [2] and the part [3] washers to replace the initial washers. Make the part [2] and the part [3] washers as shown in Figure 203/REPAIR 3, Detail C. Refer to Paragraph 5./REPAIR 3 to calculate the necessary washer thickness for the part [2] and the part [3] washers.

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Table 201: Repair Material

ITEM	PART	QUANTITY	MATERIAL
[1]	Bushing	4	Use copper-beryllium (Cu – Be) tubing as given in AMS 4535. As an alternative, you can use copper-beryllium bar as given in AMS 4533. Refer to Figure 203/ REPAIR 3, Detail A for the necessary dimensions.
[2]	Washer for the Fuse Pin Head Side	1	Use 15-5PH CRES with a heat treat of 150 to 170 KSI as given in AMS 5659. Refer to Figure 203/REPAIR 3, Detail C for the necessary dimensions.
[3]	Washer for the Fuse Pin Nut Side	1	Use 15-5PH CRES with a heat treat of 150 to 170 KSI as given in AMS 5659. Refer to Figure 203/REPAIR 3, Detail C for the necessary dimensions.

Q. Apply the cadmium plating (Type II, Class 2) to the manufactured repair parts. It is optional to apply cadmium plating to the surface of the inner diameter of the part [1] bushing. Make the thickness of cadmium plating from 0.0003 in. (0.0076 mm) to 0.0005 in. (0.0127 mm). Refer to the SOPM 20-42-05.

WARNING: USE PROTECTION WHEN YOU DO WORK WITH LIQUID NITROGEN. MAKE SURE THAT THE AREA HAS A GOOD FLOW OF CLEAN AIR. THE TEMPERATURE FOR LIQUID NITROGEN IS APPROXIMATELY MINUS 320°F (MINUS 196°C). IF YOU DO NOT OBEY, INJURY TO PERSONS CAN BE THE RESULT.

R. Use the shrink fit procedure to install the part [1] bushing with BMS 5-95 sealant as shown in Figure 202 (Sheet 2), Detail C. Use liquid nitrogen at -320°F (-196°C). Refer to SOPM 20-50-03.

(1) Make sure that the part [1] bushing is installed in the correct location.

(2) Hold the assembly in position until the assembly is at room temperature.

S. Get or make the tools that follow to line ream the bushing through the pair of bushing holes to keep the bushing holes coaxial. Refer to tool drawing MIT115A5002.

(1) Use a long piloted reamer to keep the bushing holes coaxial. The pilot part of the reamer must be non-cutting to prevent damage to the bushing.

(2) Use a guide bushing to slip into the adjacent bushing to support the reamer and to protect the bushing holes.

T. Line ream and hone the holes of the part [1] bushing to the final inner diameter of 1.2905 in. (32.78 mm) to 1.2915 in. (32.80 mm). Refer to tool drawing MIT115A5002 for reaming and honing holes.

(1) Make sure that the pairs of bushing holes are coaxial as given in Figure 202 (Sheet 3), Detail C, Section B-B. If necessary, hone the pair of the new replacement bushings at the same time to keep them coaxial.

U. If you install new bushings to the support fitting then:

(1) Machine the flanges of the bushings to the dimension shown in Figure 202 (Sheet 3), Detail C, section B-B. It is not necessary to re-cad plate the bushing after you machine the bushing flange.

(2) Machine the chamfer of the bushing flange as shown in Figure 203 (Sheet 1) and Figure 203 (Sheet 2), Detail B.

V. Make sure the surface finish of the holes is equal to 32 microinches Ra or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

W. Clean the bushing holes to make sure that all the honing debris and machining oil are removed. Refer to SOPM 20-30-03.

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X. Install the main landing gear forward trunnion bearing assembly as given in AMM PAGEBLOCK 32-11-83/401, and as follow:

(1) Install the part [2] washer with BMS 3-38 corrosion inhibiting compound at the head side of the fuse pin. The part [2] washer replaces the initial special washer. Discard the initial special washer. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01 and Figure 204/REPAIR 3. Make sure that the washer is concentric with the support fitting bore when you install the fuse pin.

(2) Install the part [3] washer with BMS 3-38 corrosion inhibiting compound at the nut side of the fuse pin. The part [3] washer replaces the initial special washer. Discard the initial special washer. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01 and Figure 204/REPAIR 3. Make sure that the washer is concentric with the support fitting bore when you install the fuse pin.

(3) Apply a fillet seal with BMS 5-95 sealant to the edges of the part [2] and the part [3] washers. Refer to REPAIR SEALING, 51-20-05.

Y. Install the components that you removed for access.

5. Repair Parts Dimensions

A. If you use the initial bushings, make sure that the diameter of the bore is between 1.5602 in. (39.629 mm) and 1.5610 in. (39.649 mm). Make sure that the outer diameter of the part [1] bushing is between 1.5626 in. (39.690 mm) and 1.5638 in. (39.721 mm) for an interference fit.

NOTE: As an alternative, you can order a Part Number 115A5230-2 bushing from The Boeing Company or manufacture the part [1] bushing as given in Drawing 115A5230–2 and as shown in Figure 203 (Sheet 1) .

B. Manufacture the part [2] washer with an inner diameter to be larger than the part [1] bushing outer diameter by 0.010 in. (0.25 mm) to 0.020 in. (0.51 mm). Refer to Figure 203/REPAIR 3, Detail C and Figure 204/REPAIR 3 to identify the part [2] washer. Make the thickness of the part [2] washer to be larger than the thickness of the initial special washer. Use the steps that follow to calculate the necessary thickness of the part [2] washer (T1):

(1) Measure the depth of the material that you removed from the outer surface of the support fitting around the bore. Refer to Figure Figure 202 (Sheet 1), Section A-A to identify the depth (X) of the material that you removed. Make sure that the maximum depth of removed material is not larger than 0.210 in. (5.334 mm). Make a record of the minimum and maximum values of the removed material depth.

(2) Calculate the sum of the minimum and maximum values of removed material depth.

(3) Divide the sum of the minimum and maximum values of removed material depth by two. Use the result value as the average depth of removed material.

(4) Sum the average depth of removed material and 0.070 in. (1.778 mm). Use the result value as a maximum thickness of the part [2] washer.

(5) Sum the average depth of removed material and 0.060 in. (1.524 mm). Use the result value as a minimum thickness of the parts [2] washer.

(6) Example: Assume that the material you removed from the outer surface of the support fitting is a depth of 0.100 in. (2.540 mm) to 0.102 in. (2.591 mm).

(a) The sum of the minimum and maximum values of removed material depth is equal to 0.100 in. (2.540 mm) plus 0.102 in. (2.591 mm) = 0.202 in. (5.131 mm)

(b) The average depth of removed material is equal to 0.202 in. (5.131 mm) divided by 2 = 0.101 in. (2.565 mm).

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(c) The maximum thickness of the part [2] washer is equal to the sum of 0.101 in. (2.565 mm) and 0.070 in. (1.778 mm) = 0.171 in. (4.343 mm)

(d) The minimum thickness of the part [2] washer is equal to the sum of 0.101 in. (2.565 mm) and 0.060 in. (1.524 mm) = 0.161 in. (4.089 mm)

(e) The thickness of the part [2] washer is 0.161 in. (4.089 mm) to 0.171 in. (4.343 mm)

C. Manufacture the part [3] washer with an inner diameter to be larger than the part [1] bushing outer diameter by 0.010 in. (0.25 mm) to 0.020 in. (0.51 mm). Refer to Figure 203/REPAIR 3, Detail C and Figure 204/REPAIR 3 to identify the part [3] washer. Make the thickness of the part [3] washer to be larger than the thickness of the initial special washer. Use the steps shown in Paragraph 5.B.(1)/REPAIR 3 through Paragraph 5.B.(6)/REPAIR 3 to calculate the necessary thickness of the part [3] washer (T1).

D. If there is no damage to be removed from the outer surface of the support fitting around the bore:

(1) Make the part [2] washer thickness the same as the initial special washer thickness. Refer to Figure 203/REPAIR 3, Detail C and Figure 204/REPAIR 3 to identify the part [2] washer.

(2) Make the part [3] washer thickness the same as the initial special washer thickness. Refer to Figure 203/REPAIR 3, Detail C and Figure 204/REPAIR 3 to identify the part [3] washer.

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- ZONE SHOWS THE TYPICAL LOCATIONS OF THE CORROSION DAMAGES. THE DAMAGED MATERIAL MUST BE REMOVED AS SHOWN IN SECTION A-A.

REMOVAL OF THE DAMAGE

A-A

1700252 S0000309348_V2

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 1 of 4)

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INSTALLATION OF THE PART [1] BUSHINGS

C

1700253 S0000309834_V1

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 2 of 4)

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B–B

1700254 S0000309835_V1

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 3 of 4)

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NOTES

1 ADJUST THE DIAMETER TO PREVENT DAMAGE TO THE VERTICAL FLANGES.

2 DO NOT REMOVE MATERIAL FROM THE GUSSETS.

3 KEEP THE CENTER OF THE HOLES IN THE INITIAL POSITION.

4 MAKE THE INDICATED AREA FLAT AFTER DAMAGE REMOVAL. IT IS NECESSARY FOR CORRECT INSTALLATION OF THE PARTS [2] AND [3] WASHERS.

5 REAM AND HONE THE BUSHING HOLE TO THE FINAL INNER DIAMETER AFTER THE INSTALLATION OF THE PART [1] BUSHING.

6 MAKE SURE THAT THE BUSHING IS 0.040 INCH MINIMUM ABOVE FLUSH.

1700257 S0000309836_V1

Main Landing Gear Forward Trunnion Support Fitting Repair
Figure 202 (Sheet 4 of 4)

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Repair Parts
Figure 203 (Sheet 1 of 3)

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B

PART [2] AND PART [3] WASHERS

C

B–B

1700260 S0000309837_V1

Repair Parts
Figure 203 (Sheet 2 of 3)

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NOTES

- MAKE THE SURFACE FINISH OF ALL SURFACES (EXCEPT AS NOTED) EQUAL TO:
  - 63 MICROINCHES (1.6 MICROMETERS) Ra OR SMOOTHER FOR THE PART [1] BUSHING
  - 125 MICROINCHES (3.2 MICROMETERS) Ra OR SMOOTHER FOR THE PART [2] AND THE PART [3] WASHERS.
- REMOVE ALL SHARP EDGES. MAKE A RADIUS THAT IS 0.005 TO 0.015 INCH (0.127 TO 0.381 mm) AT THE EDGE.
- APPLY CADMIUM PLATE TYPE II, CLASS 2, TO THE PART [1] BUSHING, THE PART [2] AND THE PART [3] WASHERS.
- ALL DIMENSIONS ARE SHOWN AFTER THE PLATING PROCEDURE.

1 INNER DIAMETER OF THE BUSHING HOLE BEFORE INSTALLATION IS SHOWN. REAM AND HONE THE BUSHING HOLE TO THE FINAL INNER DIAMETER AFTER THE INSTALLATION OF THE PART [1] BUSHING. REFER TO FIGURE 202, REPAIR 3.

2 IT IS PERMITTED TO MAKE THE SURFACE FINISH OF THE INNER DIAMETER OF THE BUSHING HOLE TO 125 MICROINCHES (3.175 MICROMETERS) Ra OR SMOOTHER. CADMIUM PLATING IS OPTIONAL ON THE INSIDE DIAMETER OF THE BUSHING HOLE.

3 REFER TO PARAGRAPH 5/REPAIR 3 TO CALCULATE THE NECESSARY VALUE.

4 TOTAL RUNOUT TOLERANCE TO THE DATUM -A- MUST NOT BE MORE THAN 0.0010 INCH (0.0254 mm).

5 RUNOUT TOLERANCE TO THE DATUM -A- MUST NOT BE MORE THAN 0.0100 INCH (0.254 mm).

6 THE INDICATED SURFACE MUST BE PERPENDICULAR TO THE DATUM -A- WITH A MAXIMUM TOLERANCE OF 0.002 INCH (0.05 mm).

1700264 S0000309839_V1

Repair Parts
Figure 203 (Sheet 3 of 3)

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FUSE PIN HEAD SIDE

FUSE PIN NUT SIDE

NOTES

1 REFER TO AMM 32-11-83.

2 REMOVE AND DISCARD THE INITIAL SPECIAL WASHER THEN INSTALL THE PART [2] WASHER AT THE HEAD SIDE OF THE FUSE PIN.

3 REMOVE AND DISCARD THE INITIAL SPECIAL WASHER THEN INSTALL THE PART [3] WASHER AT THE NUT SIDE OF THE FUSE PIN.

4 INNER DIAMETER OF REPAIR WASHER SHOULD BE 0.010 INCH (0.25 mm) TO 0.020 INCH (0.52 mm) LARGER THAN BUSHING OUTER DIAMETER.

TYPICAL INSTALLATION OF THE PARTS [2] AND [3] WASHERS

1700306 S0000309350_V1

Installation of the Repair Washer
Figure 204

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IDENTIFICATION 1 - WING TIP SKIN

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F68289 S0006594384_V1

Wing Tip Skin Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
119A0100	Wing Tip Installation
119A1200	Wing Tip Fairing Assembly
119A1500	Wing Tip Fairing and Lighting Assembly

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

F69439 S0006594386_V1

Wing Tip Skin Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Tip Light Assembly		Supplied by: Grimes Aerospace Co. 550 Route 55 Urbana, Ohio 43078	
[2]	Wing Tip Lower Skin	0.063 (1.60)	7075-T62 bare sheet as given in QQ-A-250/12	
[3]	Wing Tip Upper Skin	0.063 (1.60)	7075-T62 bare sheet as given in QQ-A-250/12	
[4]	Torque Tube		A357.0-T6 as given in MIL-A-21180, Class 10	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 2 - WINGLET SKIN

1. Applicability

A. Identification 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglets.

B. Refer to Table 1 for the reference drawings and Table 2 for the List of Materials.

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
119A0105	Winglet Installation*[1]
737-0002	Winglet Installation*[2]
737-0010	Winglet Final Assembly
737-0310	Winglet Upper Skin Installation
737-0330	Winglet Lower Skin Installation

*[1] For winglet installed in production.

*[2] For winglet installed as a retrofit.

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Skin - Inboard	0.063 (1.60)	7075-T62 clad sheet as given in QQ-A-250/13	
[2]	Leading Edge Skin - Mid	0.063 (1.60)	7075-T62 clad sheet as given in QQ-A-250/13	
[3]	Leading Edge Skin - Outboard	0.063 (1.60)	7075-T62 clad sheet as given in QQ-A-250/13	
[4]	Bonded Upper Skin Panel		Fiberglass/Graphite/Epoxy Honeycomb Sandwich	
			Fiberglass/epoxy Fabric as given in BMS 8-139, Style 120, Class 1	
			Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190	
			Graphite/Epoxy Fabric as given in 8-256, Type IV, Class 2, Style 3K-70-PW	
	Skin		Refer to APB drawing 737-0311	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class 4, Type V and VI, Grade 3.0	
[5]	Bonded Lower Skin Panel		Fiberglass/Graphite/Epoxy Honeycomb Sandwich	
			Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1	
			Graphite/epoxy tape as given in BMS 8-256, Type II, Class 1, grade 190	
			Graphite/Epoxy Fabric as given in 8-256, Type IV, Class 2, Style 3K-70-PW	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Skin Core		Refer to APB drawing 737-0331 Non-metallic honeycomb as given in BMS 8-124, Class 4, Type V and VI, Grade 3.0	
[6]	Tip Fairing	0.063 (1.60)	6061-T62 sheet as given in QQ-A-250/11	
[7]	Inboard Trailing Edge Wedge		Epoxy sheet molding as given in BMS 8-327 A, Type 1 2024-T351 as given in QQ-A-250/4	Line Numbers 1 - 1979 Line Numbers 1980 and on.
[8]	Outboard Trailing Edge Wedge		2024-T3511 extrusion as given in QQ-A-200/3	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

L46869 S0006594391_V1

Winglet Skin Location
Figure 1

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

1 DUAL LIGHT CONFIGURATION.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

L46871 S0006594392_V3

Winglet Skin Identification
Figure 2

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ALLOWABLE DAMAGE 1 - WING TIP SKIN

1. Applicability

A. This subject gives the allowable damage limits for the wing tip skins shown in Wing Tip Skin Location, Figure 101/ALLOWABLE DAMAGE 1.

2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to reworked areas. Refer to 51-20-01.

(2) Apply one layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F70236 S0006594395_V1

Wing Tip Skin Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: ALL THE PARTS ARE MADE OF ALUMINUM.

F70296 S0006594396_V1

Wing Tip Skin
Figure 102

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage limits for the Wing Tip Skin

A. Cracks:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C .

B. Nicks, Gouges, Scratches, and Corrosion:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

C. Dents:

(1) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail F .

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(2) If the damage depth is more than the 0.125 inch and less than 0.30 inch, do the steps that follow:

(a) Fill or rework the damage as given in 51-70-01.

(b) Apply aluminum foil tape (speed) tape.

(c) Record the damage location.

(d) Do an inspection of the damage location at normal maintenance intervals.

(e) Repair the damage before 24 months have occurred.

D. Holes and Punctures:

(1) The damage is permitted if it is:

(a) A maximum of 0.25 inch in diameter.

(b) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

(c) Filled with a rivet without sealant.

1) For the wing tip skins, use a 2117-T3 or 2117-T4 protruding head rivet.

2) For the torque tube, use a 5056-H32 countersink head rivet.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

F68268 S0006594397_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 3)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

D

A–A

F68281 S0006594398_V2

Allowable Damage Limits
Figure 103 (Sheet 2 of 3)

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F68273 S0006594399_V2

Allowable Damage Limits
Figure 103 (Sheet 3 of 3)

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ALLOWABLE DAMAGE 2 - WINGLET SKIN

1. Applicability

A. ALLOWABLE DAMAGE 2 is applicable to airplanes that have had winglets installed as part of production.

B. ALLOWABLE DAMAGE 2 is applicable to airplanes that have Aviation Partners Boeing (APB) winglets installed per Supplemental Type Certificate (STC) ST00830SE, except for airplane line numbers 779, 787, or 836.

C. This subject gives the allowable damage limits for the winglet skins shown in Winglet Skin Location, Figure 101/ALLOWABLE DAMAGE 2.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

L46972 S0006594401_V1

Winglet Skin Location
Figure 101

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UPPER LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

L46987 S0006594402_V3

Winglet Skin Allowable Damage
Figure 102

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2. General

A. For damage to the leading edge or tip fairing:

(1) Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) Remove the damage as necessary.

(a) Refer to 51-10-02 for the investigation and removal of damage.

(b) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) After you remove the damage, do the steps that follow.

NOTE: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

(a) Apply a chemical conversion coating to reworked areas. Refer to 51-20-01.

(b) Apply one layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplane. Refer to AERODYNAMIC SMOOTHNESS, 51-10-01.

B. For damage to the winglet upper or lower skin:

(1) Refer to Paragraph 5./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) Do an inspection of the damaged areas to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

(a) Remove all contaminates and water from the structure. Use a vacuum and heat to a maximum temperature of 125 degrees F (52 degrees C).

(b) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C for the definitions of the length, width, and depth of damage.

(c) Refer to Definitions of the Facesheets, Figure 104/ALLOWABLE DAMAGE 2 for the definitions of the facesheets of a honeycomb core.

(3) Seal all permitted damage that is not more than one ply deep. Seal the damage with one of the two procedures that follow:

(a) Make a temporary seal with aluminum foil tape (speed tape).

1) Keep a record of the location

2) Inspect aluminum foil tape (speed tape) at each scheduled maintenance.

3) Replace the tape if more damage occurs.

4) Do a permanent repair no later than 24 months from the time the seal was made.

(b) Make a permanent seal.

1) Apply BMS 8-201, BMS 8-207, or BMS 8-301 epoxy resin to the area.

2) Apply one layer of BMS 10-79, Type III primer. Refer to SOPM 20-44-04.

3) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

NOTE: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

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(4) Seal all of the damage areas that are more than one ply in depth. Refer to the allowable damage limits as given in Paragraph 4. Seal the damage as follows:

(a) Use a vacuum and heat to remove the moisture from the solid laminate or honeycomb cells. Refer to REPAIR PROCEDURES FOR WET LAYUP MATERIALS, PAGEBLOCK 51-70-04, REPAIR.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Inspect aluminum foil tape (speed tape) at each scheduled maintenance.

(e) Do a permanent repair no later than 24 months from the time the seal was made.

(5) Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplane. Refer to 51-10-01.

C. For lightning strike damage to the winglet aluminum or graphite trailing edge:

(1) Refer to Paragraph 8./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) Remove the damage as necessary. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

(3) Do a detailed visual inspection with a minimum of 10X magnification of the reworked area to make sure that all of the damage is removed.

(4) Do a tap test on the composite part near the reworked area to make sure there is no damage. Refer to TAP TEST INSPECTION OF HONEYCOMB STRUCTURE, 51-10-06

(5) If the damage on the trailing edge meets the requirements of Paragraph 8./ALLOWABLE DAMAGE 2, do as follows:

(a) Apply a chemical conversion coating. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

(b) Apply one layer of BMS 10–79 Type III primer to the reworked areas. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10–60 Type II enamel to the reworked areas. Refer to SOPM 20-41-02

D. For lightning strike damage to fasteners:

(1) Refer to Paragraph 9./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) If the damage at the fasteners meets the requirements of Paragraph 9./ALLOWABLE DAMAGE 2, do as follows:

(a) Make a temporary seal with aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Inspect aluminum foil tape (speed tape) at each scheduled maintenance.

(d) Replace the damaged fastener with a permanent repair no later than 90 days from the time the seal was made.

E. For lightning strike damage to the upper or lower skin do as follows:

(1) Refer to Paragraph 7./ALLOWABLE DAMAGE 2 for the allowable damage limits.

(2) Do an inspection of the damaged areas to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

(a) Remove all contaminates and water from the structure. Use a vacuum and heat to a maximum temperature of 125°F (52°C).

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(b) Refer to Figure 103/ALLOWABLE DAMAGE 2 for the definitions of the length, width, and depth of the damage.

(c) Refer to Figure 104/ALLOWABLE DAMAGE 2 for the definitions of the facesheets of a honeycomb core.

(3) Seal all damaged areas permitted by the allowable damage limits in Paragraph 7./ ALLOWABLE DAMAGE 2

(a) Use a vacuum and heat to remove the moisture from the solid laminate or honeycomb cells. Refer to REPAIR PROCEDURES FOR WET LAYUP MATERIALS, 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(4) Keep a record of the location.

(5) Inspect aluminum foil tape (speed tape) at each scheduled maintenance.

(6) Do a permanent repair no later than 24 months from the time the seal was made.

(7) Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplanes. Refer to AERODYNAMIC SMOOTHNESS, 51-10-01.

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DEFINITIONS FOR
NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR
DENT DAMAGE

B

DEFINITIONS FOR
EDGE DAMAGE

C

L47002 S0006594403_V1

Damage Definitions
Figure 103

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L47003 S0006594404_V1

Definitions of the Facesheets
Figure 104

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-10-06	TAP TEST INSPECTION OF HONEYCOMB STRUCTURE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-04, REPAIR P/B REPAIR	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits for the Leading Edge and Tip Fairing

A. Cracks:

(1) Edge cracks are permitted as given in Figure 105, Details A, B, and C.

B. Nicks, Gouges, Scratches, and Corrosion:

(1) Remove the damage as shown in Figure 105/ALLOWABLE DAMAGE 2, Detail D .

C. Dents:

(1) The damage is permitted as shown in Figure 105/ALLOWABLE DAMAGE 2, Detail F .

D. Holes and Punctures for the Leading Edge Skin and Tip Fairing:

(1) The damage is permitted if it is:

(a) A maximum of 0.25 inch in diameter.

(b) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

(c) Filled with an aluminum rivet installed wet with BMS 5-95 sealant. Refer to 51-20-05.

1) Use a 2117-T3 or 2117-T4 protruding head rivet.

E. Holes and Punctures for the Trailing Edge Wedge is not permitted.

5. Allowable Damage Limits for the Winglet Graphite Upper and Lower Skins

A. Cracks:

(1) Edge cracks are permitted as given in Figure 105, Details A, B, and C.

B. Nicks, Gouges, and Scratches:

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers but not the carbon fibers are permitted.

C. Dents are not permitted.

D. Holes and Punctures are permitted if the damage is:

(1) A maximum of 1 damage location for each winglet installation

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(2) A maximum diameter of 2.0 inches

(3) A minimum of 6.0 inches from a part edge, fastener hole, or other damage

(4) Sealed and repaired as given in Paragraph 2.B./ALLOWABLE DAMAGE 2.

E. Delamination is permitted if the damage is:

(1) A maximum of 1 damage location for each winglet installation

(2) A maximum diameter of 2.0 inches

(3) A minimum of 6.0 inches from a part edge, fastener hole, or other damage

(4) Sealed and repaired as given in Paragraph 2.B./ALLOWABLE DAMAGE 2.

6. Allowable Damage Limits for the Winglet Lens

A. Damage is not permitted.

7. Allowable Damage Limits for Lightning Strike Damage to the Upper and Lower Skins

A. Lightning strike damage on the upper and lower skins are permitted if:

(1) The maximum diameter of the damage is less than or equal to 2.0 in. (50.8 mm).

(2) The damage is sealed as given in Paragraph 2.E./ALLOWABLE DAMAGE 2.

(3) The damage locations are greater than 10.0 in. (254.0 mm) apart.

8. Allowable Damage Limits for Lightning Strike Damage at the Aluminum or Graphite Trailing Edge

A. Lightning strike damage on the aluminum or graphite trailing edge is permitted if:

(1) The damage is permitted as given in Figure 106/ALLOWABLE DAMAGE 2.

9. Allowable Damage Limits for the Lightning Strike Damage to Fasteners

A. Lighting strike damage to fasteners are permitted if:

(1) There are a minimum of 10 fasteners between any two damaged or missing fasteners in any fastener row.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

L47027 S0000128242_V1

Allowable Damage Limits
Figure 105 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

E

L47030 S0000128210_V1

Allowable Damage Limits
Figure 105 (Sheet 2 of 3)

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DENT THAT IS PERMITTED

F

L47032 S0006594405_V1

Allowable Damage Limits
Figure 105 (Sheet 3 of 3)

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(LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE)

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH (2.54 mm)
FOR THE INBOARD TRAILING EDGE
= A MAXIMUM OF 0.50 INCH (12.70 mm)
FOR THE OUTBOARD TRAILING EDGE

B

NOTES

1 DO NOT BLEND MORE THAN 25% OF THE SPAN OF THE OUTBOARD TRAILING EDGE.

2 DO NOT BLEND MORE THAN 25% OF THE SPAN OF THE INBOARD TRAILING EDGE.

1984385 S0000384027_V1

Lightning Strike Damage Limits
Figure 106 (Sheet 1 of 2)

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2135046 S0000463084_V1

Lightning Strike Damage Limits
Figure 106 (Sheet 2 of 2)

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REPAIR 1 - EXTERNAL REPAIR OF THE WING TIP SKIN

1. Applicability

A. Repair 1 is applicable to damage to the upper and lower skins of the wing tip as shown in Wing Tip Skin Location, Figure 201/REPAIR 1.

2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

G29779 S0006594408_V1

Wing Tip Skin Location
Figure 201

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
AMM 57-21-11/401	Wing Tip Removal/Installation
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Remove the wing tip if necessary. Refer to AMM 57-21-11/401.

B. Remove the damage.

(1) If there is a crack, do one of the steps that follow:

(a) Drill a stop hole at the ends of a crack.

1) Drill a 0.25 inch diameter stop hole at the ends of the crack that do not end at a fastener hole.

2) Make sure you have removed the ends of the crack. Refer to 51-10-02.

(b) Cut and remove the damage. Refer to 51-10-02.

1) Refer to Wing Tip Skin - External Repair, Figure 202/REPAIR 1 for the shape of the cut out.

2) Make the corner radii of the cut a minimum of 0.50 inch.

3) Make sure there is a minimum of two rows of repair fasteners around the edges of the cut out.

C. Make the part [1] doubler as shown in Wing Tip Skin - External Repair, Figure 202/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair material.

(1) Make the contour of the part [1] doubler the same as the initial contour of the skin.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Doubler	1	Use bare or clad 7075-T6 that is 0.071 inch thick

D. Assemble the part [1] doubler and drill the fastener holes.

(1) Do not countersink the fastener holes more than 80 percent of the initial skin thickness.

(a) This will prevent a knife-edge condition of the part [1] doubler.

E. Remove the part [1] doubler.

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F. Remove all nicks, scratches, burrs, and sharp edges from the part [1] doubler.

G. Apply a chemical conversion coating to the part [1] doubler and to the bare surfaces of the skin.

H. Apply one layer of BMS 10-79, Type II primer to the part [1] doubler and to the bare edges of the initial skin. Refer to 51-20-01.

I. Install the part [1] doubler with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

J. Install the fasteners wet with BMS 5-95 sealant.

K. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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A–A

FASTENER SYMBOLS

- REPAIR FASTENER LOCATION. INSTALL A BACR15FP5E BLIND RIVET.
- REPAIR FASTENER LOCATION. INSTALL A BACR15FP5E BLIND RIVET. THIS FASTENER LOCATION IS NOT USED IF YOU MAKE THE OPTIONAL DAMAGE CUTOUT.

F68358 S0006594410_V1

Wing Tip Skin - External Repair
Figure 202

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REPAIR 2 - WING TIP SKIN FLUSH REPAIR

1. Applicability

A. Repair 2 is applicable to damage to the wing tip skin shown in Wing Tip Skin Location, Figure 201/ REPAIR 2.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

F80880 S0006594412_V1

Wing Tip Skin Location
Figure 201

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2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-07	MACHINING AND DRILLING OF COMPOSITE STRUCTURES
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
AMM 57-21-11/401	Wing Tip Removal/Installation

4. Repair Instructions

A. Remove the wing tip if necessary as given in AMM 57-21-11/401.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make a minimum corner radius of 0.50 inch.

(2) Make sure the cut out hole removes all of the damaged skin.

(a) Refer to Wing Tip Skin - Flush Repair, Figure 202/REPAIR 2 for the shape of the cutout.

(3) Make sure there is a minimum of two rows of repair fasteners around the edge of the cut out.

C. Make the repair parts as shown in Wing Tip Skin - Flush Repair, Figure 202/REPAIR 2. Refer to Table 201 .

(1) Make the contour of the part [1] filler and the part [2] doubler the same as the initial contour of the skin.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Filler	1	Use bare or clad 7075-T6 that is 0.063 inch thick
[2]	Doubler	1	Use bare or clad 7075-T6 that is 0.071 inch thick

D. Assemble the repair parts and drill the fastener holes.

(1) Do not countersink the fastener holes more than 80 percent of the initial skin thickness.

(a) This will prevent a knife-edge condition of the filler and the skin.

E. Remove the part [1] filler and the part [2] doubler.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and to the bare surfaces of the initial skin.

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G. Apply one layer of BMS 10-79, Type II primer to the repair parts and to the edges of the skin. Refer to 51-20-01.

H. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-07.

I. Install the fasteners without sealant.

J. Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.

K. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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A–A

FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET

F68393 S0006594414_V1

Wing Tip Skin - Flush Repair
Figure 202

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REPAIR 3 - WINGLET LEADING EDGE SKIN FLUSH REPAIR

1. Applicability

A. Repair 3 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

NOTE: Repair 3 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. Repair 3 is applicable to damage to the winglet leading edge skin shown in Figure 201.

NOTE: A maximum of (2) winglet leading edge skin repairs is permitted from rib 27 to midspan of the outboard leading edge panel.

C. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

2. General

A. Repair 3 gives instructions for a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

L46885 S0000128216_V1

Winglet Skin Location
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
737 NDT Part 6, 51-00-00	Structures - General

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge skin. Refer to 51-10-02.

(1) Make a minimum corner radius of 0.50 inch (12.70 mm).

B. Put the skin around the cut back to the initial contour.

C. Do a high frequency eddy current (HFEC) inspection of the damaged area to make sure that all the damage is removed. Refer to 737 NDT Part 6, 51-00-00, Figure 4 for the HFEC inspection procedures.

(1) The dye penetrant inspection is permitted as an alternative to the HFEC inspection. Refer to SOPM 20-20-02.

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D. Make the repair parts and make the shape the same as the initial contour of the skin. Refer to Table 201/REPAIR 3 and Figure 202.

Table 201:

REPAIR MATERIAL			
PART NUMBER	PART	QUANTITY	MATERIAL
[1]	Repair Plate	1	Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming.
[2]	Doubler	1	Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming.
[3]	Doubler	1	Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming.
[4]	Strap	1	Use 7075-0 clad sheet that is 0.063 inch (1.60 mm) thick. Heat treat to T62 or T6 after forming.

E. Put the repair parts in place as shown in Figure 202/REPAIR 3.

F. Drill and countersink the necessary fastener holes

NOTE: Do not countersink more than 80 percent of the skin thickness to prevent knife-edging of the skin.

G. Remove the repair parts.

H. Remove all the nicks, scratches, burrs, gouges, and sharp edges from the repair parts and the bare surfaces of the skin.

I. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial skin. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01

J. Apply one layer of BMS 10-11, Type I primer to the initial surfaces and to the repair parts. Refer to SOPM 20-41-02.

K. Install the repair parts [2], [3], and [4]. Refer to Figure 202/REPAIR 3.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin and to the repair parts. Refer to 51-20-05.

L. Install the aluminum fasteners without sealant.

M. Install repair part [1]. Refer to Figure 202/REPAIR 3.

(1) Apply BMS 5-95 sealant to the mating surfaces of the repair parts. Refer to 51-20-05.

N. Install the blind fasteners with BMS 5-95 sealant.

O. Fill the gap between the skin and the part [1] repair plate with BMS 5-95 sealant. Refer to 51-20-05.

P. Apply the finish to the repair area as given in AMM 51-21-00/701.

NOTE: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

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FASTENER SYMBOLS

┼ REPAIR FASTENER LOCATION. INSTALL A BACR15CE5AD RIVET

⊕ REPAIR FASTENER LOCATION. INSTALL A NAS1739E5 BLIND RIVET

L46887 S0000128218_V2

Leading Edge Skin Flush Repair
Figure 202 (Sheet 1 of 2)

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Leading Edge Skin Flush Repair
Figure 202 (Sheet 2 of 2)

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REPAIR 4 - WINGLET SKIN REPAIR

1. Applicability

A. Repair 4 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

NOTE: Repair 4 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. This subject gives repair limits for the winglet skins shown in Figure 201 .

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

L46878 S0000128224_V1

Winglet Skin Location
Figure 201

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

N82823 S0006594418_V2

Winglet Skin Repair
Figure 202

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2. General

A. A maximum of (6) 5.0 inches diameter wet layup repairs and (1) 12.0 inches diameter pre-preg repair is permitted. Contact The Boeing Company for repairs more than these limits.

B. Repair 4 gives instructions for Category A repairs to all Zone locations. Refer to 51-00-06 to find the definitions of the different categories of repairs.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Nondestructive Inspection (NDI) procedure. Refer to NDT, Part 1, 51-01-02 for inspection procedures.

(1) Step sanding is not permitted.

(2) Refer to Figure 203 for the definitions of the diameter and depth of damage.

(3) Refer to Figure 204 for the definitions of the facesheets of a honeycomb core area.

D. Do the repair as given in Paragraph 4./REPAIR 4 for all Zone areas.

E. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

F. Refer to 51-20-01 for interior and exterior finishes.

NOTE: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

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DEFINITIONS FOR
NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR
DENT DAMAGE

B

DEFINITIONS FOR
EDGE DAMAGE

C

L46880 S0000128225_V1

Damage Definitions
Figure 203

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M97136 S0000128226_V1

Definitions of the Facesheets
Figure 204

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS

4. Repair Instructions

A. For cracks, holes, delamination, nicks and gouges, refer to Table 201/REPAIR 4 for the repair limits and procedures.

B. For repairs made with wet layup materials, do as follows:

(1) For each facesheet that is damaged, use one structural repair fabric ply for each initial fabric ply that was damaged. Use two structural repair fabric plies for each initial tape ply that was damaged.

(2) Then add two more structural repair plies. Put one structural ply at 0 or 90 degrees and put the next ply at +/- 45 degrees to the core ribbon direction.

NOTE: A maximum of (6) 5.0 inches diameter repairs are permitted.

C. For repairs made with preimpregnated layup materials, do as follows:

(1) For each facesheet that is damaged, use one structural repair ply for each initial ply that was damaged. The type, class, grade, and style shall be the same as the initial plies.

(2) Then add two more structural repair plies. Put one structural ply at 0 or 90 degrees and put the next ply at +/- 45 degrees to the core ribbon direction.

NOTE: A maximum of (1) 12.0 inches diameter repair is permitted.

Table 201:

REPAIR DATA FOR THE 350°F (177°C) CURE, WINGLET FIBERGLASS/EPOXY SKIN PANELS			
REPAIR TYPE	**CATEGORY A WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**350°F (177°C)**
REPAIR SIZE	- This temperature repair is not permitted.	Damage that is a maximum of: - 5.0 inches in diameter A maximum of (6) repair locations is permitted. Contact The Boeing Company for repairs more than these limits 6.0 inches minimum clearance from other damage	Damage that is a maximum of: - 12.0 inches in diameter A maximum of (1) repair location is permitted. Contact The Boeing Company for repairs more than these limits.
REPAIR INSTRUCTIONS	------	SRM 51-70-04 and Paragraph 4.B	SRM 51-70-05 and Paragraph 4.C

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IDENTIFICATION 1 - WING TIP STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F67531 S0006594423_V1

Wing Tip Structure Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
119A1200	Wing Tip Installation
119A1500	Wing Tip Fairing and Light Assembly
119A0100	Wing Tip Assembly

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

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F68163 S0006594425_V1

Wing Tip Structure Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Intercostal	0.040 (1.02)	7075-T6 clad sheet	
[2]	Aft Rib Former		7050-T7451 plate as given in AMS 4050	
[3]	Mid Rib Former		7050-T7451 plate as given in AMS 4050	
[4]	Forward Rib Former		7050-T7451 plate as given in AMS 4050	
[5]	Trailing Edge Wedge		A357.0-T6 as given in MIL-A-21180, Class 10	
[6]	Bulkhead Rib		7050-T7451 plate as given in AMS 4050	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 2 - WINGLET STRUCTURE

1. Applicability

A. Identification 2 is applicable to all airplanes which includes those airplanes retrofitted by the Supplemental Type Certificate (STC) ST00830SE, except airplane line numbers 779, 787 and 836. For airplane line numbers 779, 787 and 836 contact Aviation Partners Boeing (APB).

NOTE: The Supplemental Type Certificate (STC) ST00830SE changes some wing structures for retrofit installation of the Aviation Partners Boeing (APB) blended winglets.

B. Refer to Table 1 for the reference drawings and Table 2 for the List of Materials.

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
119A0105	Winglet Installation*[1]
737-0002	Winglet Installation*[2]
737-0010	Winglet Final Assembly
737-0011	Winglet Assembly Substructure
737–0310	Winglet Upper Skin Installation
737–0330	Winglet Lower Skin Installation
737–0350	Leading Edge Skin Installation

*[1] For winglet installed in production.

*[2] For winglet installed as a retrofit.

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Trailing Edge Rib Sta 0		7050-T7451 aluminum plate as given in AMS 4050	
[2]	Inspar Rib Sta 0		7050-T7451 aluminum plate as given in BMS 7-323, Type III, Optional; Type 1.	
[3]	Inspar Rib Sta 1		7050-T7451 aluminum plate as given in AMS 4050	
[4]	Inspar Rib Sta 4		7050-T7451 aluminum plate as given in AMS 4050	
[5]	Leading Edge Rib Sta 0		7050-T7451 aluminum plate as given in AMS 4050	
[6]	Leading Edge Rib Sta 1		7050-T7451 aluminum plate as given in AMS 4050	
[7]	Leading Edge Rib Sta 4		7050-T7451 aluminum plate as given in AMS 4050	
[8]	Bonded Rear Spar		Fiberglass/Graphite/Epoxy Solid Laminate Graphite/Epoxy Fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190 Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1 Refer to APB drawing 737-0241	
[9]	Bonded Center Spar		Fiberglass/Graphite/Epoxy Solid Laminate	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
			Graphite/Epoxy Fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190 Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1 Refer to APB drawing 737-0231	
[10]	Bonded Front Spar		Fiberglass/Graphite/Epoxy Solid Laminate Graphite/Epoxy Fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW Graphite/Epoxy tape as given in BMS 8-256, Type II, Class 1, Grade 190 Fiberglass/Epoxy Fabric as given in BMS 8-139, Style 120, Class 1 Refer to APB drawing 737-0221	
[11]	Lower Outboard Stiffener		7075-T73 aluminum bare sheet as given in QQ-A-250/12	
[12]	Lower Inboard Stiffener		7075-T73 aluminum bare sheet as given in QQ-A-250/12	
[13]	Lower Splice Plate		Ti-6AL-4V as given in MIL-T–9046, AB-1 condition	
[14]	Upper Splice Plate		Ti-6AL-4V as given in MIL-T-9046, AB-1 condition	
[15]	Attach Clip, TE Sta 0		2024-T3 aluminum clad sheet as given in QQ-A-250/5	
[16]	Attach Clip, LE Sta 1		2024-T3 aluminum clad sheet as given in QQ-A-250/5	
[17]	Upper Outboard Stiffener		7075–T73 aluminum bare sheet as given in QQ-A-250/12	
[18]	Upper Inboard Stiffener		7075-T73 aluminum bare sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

L46592 S0006594430_V1

Winglet Structure Location
Figure 1

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

L46594 S0006594431_V2

Winglet Structure Identification
Figure 2

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ALLOWABLE DAMAGE 1 - WING TIP STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing tip structure shown in Wing Tip Structure Location, Figure 101/ALLOWABLE DAMAGE 1.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F70304 S0006594434_V1

Wing Tip Structure Location
Figure 101

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Wing Tip Structure
Figure 102

2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the ribs.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(c) Do not flap peen or shot peen the trailing edge wedge casting or the intercostal.

(2) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(3) Apply one layer of BMS 10-11, Type I primer to the reworked areas of the intercostal and ribs. Refer to SOPM 20-41-02.

(4) Apply one layer of BMS 10-79, Type III primer to the reworked areas of the trailing edge wedge. Refer to SOPM 20-44-04.

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Cracks:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and F .

B. Nicks, Gouges, Scratches and Corrosion:

(1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , E , and F .

C. Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail D .

D. Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

F67730 S0006594436_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

C

A–A

DENT THAT IS PERMITTED

D

F68462 S0006594437_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 3)

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REMOVAL OF CORROSION AROUND THE FASTENERS

E

B–B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

F

F67530 S0006594438_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 3)

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ALLOWABLE DAMAGE 2 - WINGLET STRUCTURE

1. Applicability

A. Allowable Damage 2 is applicable to airplanes that have had winglets installed as part of the production of the airplane.

NOTE: Allowable damage 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(1) For airplanes with cum line numbers 1 thru 1302:

(a) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(2) For airplanes with cum line numbers 1303 and on:

(a) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

B. This subject gives the allowable damage limits for the winglet structure shown in Figure 101.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

L46573 S0000128230_V1

Winglet Structure
Figure 101

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LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

L46571 S0000128231_V1

Winglet Structure
Figure 102

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2. General

A. Refer to Paragraph 4 through Paragraph 5 for the allowable damage limits.

WARNING: SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STREAM EXPLOSION. IF YOU DO NO OBEY, AN INJURY CAN OCCUR.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas of the aluminum parts. Refer to 51-20-01.

(2) Apply one layer of BMS 10-11, Type I primer to the reworked areas of the aluminum and titanium parts. Refer to SOPM 20-41-02.

(3) Apply one layer of BMS 10-60, Type II primer to the reworked areas of the aluminum parts. Refer to AMM 51-21-00/701.

NOTE: Before you apply new paint or primer to a reworked area of the winglet, make sure you have removed the initial paint and primer from that area.

D. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in economic performance of the airplane. Refer to 51-10-01.

3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits for the Metallic Winglet Structure

A. Edge Cracks:

(1) Remove the damage as shown in Figure 103 , Details A , B , and C .

B. Nicks, Gouges, Scratches and Corrosion:

(1) Remove the damage as shown in Figure 103 , Details A , B , C , D , and E .

C. Dents are permitted as shown in Figure 103 , Detail F .

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D. Holes and Punctures are not permitted.

5. Allowable Damage Limits for the Winglet Rear, Center, and Front Spar

A. Damage is not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

L46587 S0000128232_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

E

L46588 S0000128233_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 3)

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DENT THAT IS PERMITTED

F

L46589 S0000128234_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 3)

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REPAIR 1 - WING TIP STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

F67532 S0006594442_V2

Wing Tip Structure Repair
Figure 201

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REPAIR 2 - WINGLET STRUCTURE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

M33247 S0006594444_V2

Winglet Structure Repair
Figure 201

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IDENTIFICATION 1 - WING INBOARD FIXED LEADING EDGE UPPER SKIN PANELS

H00510 S0006594449_V1

Wing Inboard Fixed Leading Edge Upper Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
116A0010	Fixed Leading Edge Collector - Final Assembly
116A0015	Functional Product Collector - Front Spar Leading Edge Join, Left Wing
116A1200	Upper Removable Access Panel Installation, Inboard Leading Edge
116A4008	Skin Installation - Upper, Inboard Leading Edge, Wing

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H02195 S0006594451_V1

Wing Inboard Fixed Leading Edge Upper Skin Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin, Closeout	0.063 (1.6)	2024-T42 clad sheet as given in QQ-A-250/5	
[2]	Upper Skin, Strakelet to Seal Rib	0.063 (1.6)	2024-T42 clad sheet as given in QQ-A-250/5	
[3]	Bond Assembly, Upper Removable Access Panel Skin Assembly Core		 Refer to Figure 3 Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[4]	Stall Strip	0.50 (12.7)	7050-T7451 machined aluminum plate as given in AMS 4050	CUM LINE 21 AND ON

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H02613 S0006594453_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Upper Removable Access Panel, Figure 2, Item [3]
Figure 3

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Table 3:

PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [3]		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A)
P2 to P4, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P5, P6, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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IDENTIFICATION 2 - WING INBOARD FIXED LEADING EDGE LOWER SKIN PANELS

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H02625 S0006594456_V1

Wing Inboard Fixed Leading Edge Lower Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
116A0010	Fixed Leading Edge Collector - Final Assembly
116A0011	Fixed Leading Edge Collector - Post Paint - Left Wing
116A0013	Collector - Leading Edge, Wing Build-Up, Left Wing
116A0015	Functional Product Collector - Front Spar Leading Edge Join, Left Wing
116A1300	Lower Panel Installation - Removable, Inboard Fixed Leading Edge
116A2100	Lower Panel Installation, Inboard Fixed Leading Edge, Wing

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
116A2132	Lower Panel Bond Assy - Inboard Fixed Leading Edge, Wing

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LEFT SIDE IS SHOWN, RIGHT SIDE OPPOSITE
ENGINE HAS BEEN REMOVED FROM VIEW FOR CLARITY

H03112 S0006594458_V3

Wing Inboard Leading Edge Lower Skin Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin, Closeout	0.063 (1.6)	2024-T3 clad sheet as given in QQ-A-250/5	
[2]	Panel Bond Assembly Lower Removable			
	Skin Assembly		Refer to Figure 3	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[3]	Panel Bond Assembly Lower, Inboard Fixed Leading Edge, Wing			
	Skin Assembly		Refer to Figure 4	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H03242 S0006594460_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Upper Removable Access Panel, Figure 2, Item [2]
Figure 3

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Table 3:

PLY MATERIAL AND DIRECTION FOR FIGURE 3, ITEM [2]		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A)
P2 to P5, and P9 to P12	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6 to P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P13	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP DIRECTION AND CORE RIBBON DIRECTION

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

2109289 S0000449744_V1

Panel Bond Assembly Lower, Inboard Fixed Leading Edge, Wing
Figure 4

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Table 4:

PLY MATERIAL AND DIRECTION FOR FIGURE 4, ITEM [3]		
PLY	DIRECTION	MATERIAL
P1	OPTIONAL	Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A)
P2 to P5, and P9 to P12	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6 to P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P13	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B

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IDENTIFICATION 3 - WING OUTBOARD FIXED LEADING EDGE UPPER SKIN PANELS

H03275 S0006594463_V2

Wing Outboard Fixed Leading Edge Upper Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
116A0013	Collector - Leading Edge, Wing Build-Up, Left Wing
116A0015	Functional Product Collector - Front Spar Leading Edge Join, Left Wing
116A5410	Installation - Upper Panel, Slat 1, Outboard Fixed Leading Edge
116A5420	Installation - Upper Panel, Slat 2, Outboard Fixed Leading Edge
116A5430	Installation - Upper Panel, Slat 3, Outboard Fixed Leading Edge
116A5440	Installation - Upper Panel, Slat 4, Outboard Fixed Leading Edge

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
116A6100	Leading Edge Installation, SS 519.22 to WBL 658.17, Outboard Fixed Leading Edge
116A6200	Gap Cover Installation - Outboard Fixed Leading Edge
116A6201	Upper Panel Assembly - Gap Cover, Outboard Fixed Leading Edge

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

H03670 S0006594465_V1

Wing Outboard Fixed Leading Edge Upper Skin Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Upper Panel SS 519.22 to WBL 658.17	0.063 (1.6)	2024-T42 clad sheet as given in QQ-A-250/5	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

H03708 S0006594467_V2

Wing Outboard Fixed Leading Edge Gap Cover Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Upper Panel, Gap Cover	0.080 in. (2.032 mm)	2024-T42 clad sheet as given in QQ-A-250/5	
[2]	Strakelet Box Assembly Leading Edge			
	Rib Assembly			
	Canted Rib		7050-T7451 plate as given in AMS4050	
	Striker Plate	0.02 in. (0.508 mm)	304 sheet as given in AMS5513	
	Blowout Door			
	Panel	0.1 in. (2.54 mm)	2024-T3 clad sheet as given in QQ-A-250/5	
	Strakelet		5083 sheet as given in QQ-A-250/6	
	Seal Rib Assembly			
	Seal Rib			
	Striker Plate	0.02 in. (0.508 mm)	304 sheet as given in AMS5513	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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Wing Outboard Fixed Leading Edge Slat No.1 Upper Panel Identification
Figure 4

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Table 4:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bond Panel Assembly Slat 1			
	Skin Assembly		See Figure 8	
	Core (7)		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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Wing Outboard Fixed Leading Edge Slat No.2 Upper Panel Identification
Figure 5

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Table 5:

LIST OF MATERIALS FOR FIGURE 5				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bond Panel Assembly Slat 2			
	Skin Assembly		See Figure 9	
	Core (7)		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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Wing Outboard Fixed Leading Edge Slat No.3 Upper Panel Identification
Figure 6

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Table 6:

LIST OF MATERIALS FOR FIGURE 6				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bond Panel Assembly Slat 3			
	Skin Assembly		See Figure 10	
	Core (7)		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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Wing Outboard Fixed Leading Edge Slat No.4 Upper Panel Identification
Figure 7

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Table 7:

LIST OF MATERIALS FOR FIGURE 7				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bond Panel Assembly Slat 4			
	Skin Assembly		See Figure 11	
	Core (7)		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H04772 S0006594477_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.1 Panel, Figure 4, Item [1]
Figure 8 (Sheet 1 of 2)

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Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.1 Panel, Figure 4, Item [1]
Figure 8 (Sheet 2 of 2)

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 8		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A)
P2 to P5, and P11 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, P9, and P10	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H05676 S0006594480_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.2 Panel, Figure 5, Item [1]
Figure 9 (Sheet 1 of 2)

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A-A

B

H05698 S0006594481_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.2 Panel, Figure 5, Item [1]
Figure 9 (Sheet 2 of 2)

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Table 9:

PLY MATERIAL AND DIRECTION FOR FIGURE 9		
PLY	**DIRECTION**	**MATERIAL**
P1 to P4, and P10 to P13	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P5, P6, P8, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P7	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P14	----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H05774 S0006594483_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.3 Panel, Figure 6, Item [1]
Figure 10 (Sheet 1 of 2)

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A–A

H05885 S0006594484_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.3 Panel, Figure 6, Item [1]
Figure 10 (Sheet 2 of 2)

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Table 10:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1 to P4, and P10 to P13	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P5, P6, P8, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P7	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P14	----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 11 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H06047 S0006594486_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.4 Panel, Figure 7, Item [1]
Figure 11 (Sheet 1 of 2)

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Ply Direction, Core Ribbon Direction, and Ply Sequence for the Slat No.4 Panel, Figure 7, Item [1]
Figure 11 (Sheet 2 of 2)

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Table 11:

PLY MATERIAL AND DIRECTION FOR FIGURE 11		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Aluminum coated, epoxy impregnated, glass fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A)
P2 to P5, and P11 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, P9, and P10	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form I, Grade B. Refer to the production drawing for optional material

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IDENTIFICATION 4 - WING OUTBOARD FIXED LEADING EDGE LOWER SKIN PANELS

H06122 S0006594490_V1

Wing Outboard Fixed Leading Edge Lower Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
001A1001	Wing Left - Product Collector
116A0010	Fixed Leading Edge Collector - Final Assembly
116A0011	Fixed Leading Edge Collector - Post Paint - Left Wing
116A8500	Access Door Installation - Outboard Fixed Leading Edge
116A8510	Door Assembly - Engine Fuel Shutoff/Defuel Valves, Outboard Fixed Leading Edge
116A9100	Removable Lower Panel Installation - Outboard Fixed Leading Edge

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
116A9110	Panel Assembly - Lower, Type I, Outboard Fixed Leading Edge
116A9200	Seal Door Installation - Actuator Body, Outboard Fixed Leading Edge
116A9202	Assembly - Actuator Door, Outboard Fixed Leading Edge

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H06744 S0006594492_V3

Fixed Leading Edge Lower Outboard Removable Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2 SS 384.87 TO SS 519.22				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Panel Bond Assembly Lower Removable (2) Skin Assembly Core		 Refer to Figure 6 Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[2]	Panel Bond Assembly Lower Removable Skin Assembly Core		 Refer to Figure 7 Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[3]	Panel Bond Assembly Lower Removable (3) Skin Assembly Core		 Refer to Figure 8 Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[4]	Door	0.100 (2.54)	7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 9 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

H06751 S0006594494_V1

Fixed Leading Edge Lower Outboard Removable Panel Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3 SS 260.61 TO SS 384.87				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Panel Bond Assembly Lower Removable (2)			
	Skin Assembly		Refer to Figure 10	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[2]	Panel Bond Assembly Lower Removable			
	Skin Assembly		Refer to Figure 11	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[3]	Panel Bond Assembly Lower Removable (3)			
	Skin Assembly		Refer to Figure 12	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[4]	Door	0.100 (2.54)	7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 13 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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H06819 S0006594496_V1

Fixed Leading Edge Lower Outboard Removable Panel Identification
Figure 4

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Table 4:

LIST OF MATERIALS FOR FIGURE 4 SS 144.35 TO SS 260.61				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Panel Bond Assembly Lower Removable (2)			
	Skin Assembly		Refer to Figure 14	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[2]	Panel Bond Assembly Lower Removable			
	Skin Assembly		Refer to Figure 15	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[3]	Panel Bond Assembly Lower Removable (3)			
	Skin Assembly		Refer to Figure 16	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[4]	Door	0.100 (2.54)	7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 17 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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H06911 S0006594498_V3

Fixed Leading Edge Lower Outboard Removable Panel Identification
Figure 5

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Table 5:

LIST OF MATERIALS FOR FIGURE 5 SS 26.94 TO SS 144.35				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Panel Bond Assembly Lower Removable			
	Skin Assembly		Refer to Figure 18	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[2]	Panel Bond Assembly Lower Removable			
	Skin Assembly		Refer to Figure 19	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[3]	Panel Bond Assembly Lower Removable			
	Skin Assembly		Refer to Figure 20	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[4]	Panel Bond Assembly Lower Removable (2)			
	Skin Assembly		Refer to Figure 21	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[5]	Door	0.100 (2.54)	7075-T73 sheet as given in QQ-A-250/12. Refer to Figure 22 for the chem-mill thicknesses (Optional: 7075-T62 clad sheet as given in QQ-A-250/13)	
[6]	Access Door, Bonded Assembly			
	Skin Assembly		Refer to Figure 23	
	Core		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

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A-A

NOTES

- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO PRODUCTION DRAWINGS FOR SIMILAR PANELS IN THIS AREA.

H07260 S0006594500_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 2, Item [1]
Figure 6

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 6		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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PLY LAYUP AND CORE RIBBON DIRECTION

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A-A

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H07419 S0006594502_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 2, Item [2]
Figure 7

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 7		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Expanded aluminum Foil (EAF) as given in BMS 8-336, Type I, Class 1, Grade 016, Form A (Optional: Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350)
P2 to P4, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P5, P6, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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PLY LAYUP AND CORE RIBBON DIRECTION

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A-A

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

U50987 S0000201809_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 2, Item [3]
Figure 8

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 8		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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H07436 S0006594504_V2

Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 2, Item [4]
Figure 9

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PLY LAYUP AND CORE RIBBON DIRECTION

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A-A

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

H07439 S0006594505_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 3, Item [1]
Figure 10

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Table 9:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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A-A

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H07559 S0006594507_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 3, Item [2]
Figure 11

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Table 10:

PLY MATERIAL AND DIRECTION FOR FIGURE 11		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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A-A

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- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 11 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

H07629 S0006594509_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 3, Item [3]
Figure 12

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Table 11:

PLY MATERIAL AND DIRECTION FOR FIGURE 12		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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H07662 S0006594511_V1

Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 3, Item [4]
Figure 13

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A-A

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 12 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

H07726 S0006594512_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 4, Item [1]
Figure 14

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Table 12:

PLY MATERIAL AND DIRECTION FOR FIGURE 14		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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PLY LAYUP AND CORE RIBBON DIRECTION

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- THE PLY DIRECTION IS THE SAME AS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 13 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H07755 S0006594514_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 4, Item [2]
Figure 15

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Table 13:

PLY MATERIAL AND DIRECTION FOR FIGURE 15		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 14 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWING FOR SIMILAR PANELS IN THIS AREA.

H07816 S0006594516_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 4, Item [3]
Figure 16

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Table 14:

PLY MATERIAL AND DIRECTION FOR FIGURE 16		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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H07785 S0006594518_V1

Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 4, Item [4]
Figure 17

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PLY LAYUP AND CORE RIBBON DIRECTION

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A-A

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- THE PLY DIRECTION IS DIFFERENT FROM THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 15 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H07804 S0006594519_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [1]
Figure 18

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Table 15:

PLY MATERIAL AND DIRECTION FOR FIGURE 18		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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PLY LAYUP AND CORE RIBBON DIRECTION

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A-A

NOTES

- THE PLY DIRECTION IS THE SAME AS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 16 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H07827 S0006594521_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [2]
Figure 19

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Table 16:

PLY MATERIAL AND DIRECTION FOR FIGURE 19		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I Class 1, Grade 016, Form A)
P2 to P4, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P5, P6, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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A-A

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- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 17 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

U51018 S0000201827_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [3]
Figure 20

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Table 17:

PLY MATERIAL AND DIRECTION FOR FIGURE 20		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I Class 1, Grade 016, Form A)
P2 to P4, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P5, P6, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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A-A

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- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 18 FOR THE DIRECTION AND MATERIAL OF EACH PLY.
- REFER TO THE PRODUCTION DRAWINGS FOR SIMILAR PANELS IN THIS AREA.

H08269 S0006594523_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [4]
Figure 21

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Table 18:

PLY MATERIAL AND DIRECTION FOR FIGURE 21		
PLY	**DIRECTION**	**MATERIAL**
P1 to P3, and P12 to P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P4, P5, P10, and P11	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6, P7, and P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P8	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P15	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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H08289 S0006594525_V2

Chem-Mill Thicknesses for Actuator Door, Outboard Fixed Leading Edge, Figure 5, Item [5]
Figure 22

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PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

B-B

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- THE PLY DIRECTION IS DIFFERENT THAN THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 19 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H08428 S0006594526_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Lower Panel Outboard Fixed Leading Edge, Figure 5, Item [6]
Figure 23

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Table 19:

PLY MATERIAL AND DIRECTION FOR FIGURE 23		
PLY	**DIRECTION**	**MATERIAL**
P1	0 or 90 degrees	Epoxy impregnated aluminum coated glass woven fabric as given in BMS 8-278, Type I, Class 350 (Optional: Expanded aluminum foil (EAF) as given in BMS 8-336, Type I Class 1, Grade 016, Form A)
P2 to P5, P7, P8, and P10 to P17	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P6	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581
P9	±45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 120
P18	-----	Co-cured aluminum foil as given in BMS 8-289, Type 0/350/2/1100/002, Form 1, Grade B

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ALLOWABLE DAMAGE 1 - WING FIXED LEADING EDGE PANELS

1. Applicability

A. Allowable Damage 1 is applicable to damage on the fixed leading edge panels shown in Wing Fixed Leading Edge Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Wing Fixed Leading Edge Skin Panel Location
Figure 101

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limits that is applicable to each type of structure.

C. Do the steps that follow for the parts made of Glass Fiber Reinforced Plastic (GFRP).

(1) Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator, can be used.

(a) Refer to Definitions of the Damage Size, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(b) Refer to Definitions of the Facesheets, Figure 103/ALLOWABLE DAMAGE 1 for the definitions of the facesheets of a honeycomb core area.

(c) Refer to Wing Fixed Leading Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the GFRP parts.

(2) Remove all the contamination and water from the structure.

(a) Refer to 51-70-04 for the damage removal procedures.

(b) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF STRUCTURE	**PART NAME**	**PARAGRAPH**
ZONE 1 - GFRP/HONEYCOMB SANDWICH - EDGEBAND AND SOLID LAMINATE AREAS BETWEEN HONEY- COMB CORE AREAS - UPPER AND LOWER SKIN PANELS	UPPER AND LOWER PANELS	4.A
	REMOVABLE PANEL	4.A
	BEAM ASSEMBLY	4.A
	ACCESS DOOR	4.A
ZONE 2 - GFRP/HONEYCOMB SANDWICH - HONEYCOMB CORE AREAS OF THE UPPER SKIN PANELS	UPPER SKIN PANELS	4.B
ZONE 3 - GFRP/HONEYCOMB SANDWICH - SOLID LAMINATE AREAS OF THE UPPER SKIN PANELS (DOES NOT INCLUDE SOLID LAMINATE AREAS IN ZONES 1)	UPPER PANELS	4.C
ZONE 4 - GFRP/HONEYCOMB SANDWICH - HONEYCOMB CORE AREAS OF THE LOWER SKIN PANELS	LOWER PANELS	4.D
	REMOVABLE PANEL	4.D
	BEAM ASSEMBLY	4.D
	ACCESS DOOR	4.D
ALUMINUM	STRAKELETS	4.E
	PANEL	4.E
	OVERWING FAIRING	4.E
	CLOSURE PANEL	4.E

(3) Some GFRP leading edge panels have a thin layer of aluminum on the outer surface.

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(a) Refer to Wing Fixed Leading Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the GFRP parts with aluminum layers.

(b) The aluminum layer can be one of the three materials that follow:

1) BMS 8-278 aluminum coated glass fabric

2) BMS 8-289 bonded aluminum foil

3) BMS 8-336 expanded aluminum foil mesh.

(c) Refer to 51-70-14 for the allowable damage limits for the aluminum layer.

(4) Seal the damaged areas of the GFRP leading edge skin panels as follows:

(a) Seal all damaged areas that are not more than one ply deep (and that meet the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1 and 51-70-14) with one of the two procedures that follow:

1) Make a temporary seal.

a) Apply aluminum foil tape (speed tape).

b) Keep a record of the location.

c) Make sure the tape is in satisfactory condition at normal maintenance intervals.

2) Make a permanent seal.

a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

b) Apply one layer of BMS 10-103, Type I or BMS 10-79, Type III primer. Refer to SOPM 20-44-04 for the application of BMS 10-79, Type III primer.

c) Apply one layer of BMS 10-60 enamel. Refer to AMM PAGEBLOCK 51-21-99/701.

(b) Seal all damaged areas that are more than one ply deep (and that meet the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1 and 51-70-14) as follows:

1) Use a vacuum and heat to remove moisture from the solid laminate and/or honeycomb cells. Refer to 51-70-04.

2) Make a temporary seal with aluminum foil tape (speed tape).

3) Keep a record of the location.

4) Repair the damage permanently no later than 24 months from the time the seal was made.

D. Do the steps that follow for the aluminum parts.

(1) Refer to Wing Fixed Leading Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the aluminum parts.

(2) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of the abrasive and other materials you need to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

(d) Make the surface texture roughness for all cut surfaces 125 microinches Ra or smoother.

(3) After you remove the damage on the internal surfaces, do the steps that follow:

(a) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

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(b) Apply one layer of BMS 10-11, Type I primer to the conversion coated, reworked areas. Refer to SOPM 20-41-02.

(4) After you remove the damage on the external surfaces, do the step that follows:

(a) Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.

Definitions of the Damage Size
Figure 102

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Definitions of the Facesheets
Figure 103

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Wing Fixed Leading Edge Skin Allowable Damage
Figure 104 (Sheet 1 of 5)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS ALMOST THE SAME

LOWER SKIN PANELS

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Wing Fixed Leading Edge Skin Allowable Damage
Figure 104 (Sheet 2 of 5)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS ALMOST THE SAME

LOWER SKIN PANELS

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Wing Fixed Leading Edge Skin Allowable Damage
Figure 104 (Sheet 3 of 5)

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ALLOWABLE DAMAGE ZONES
(TYPICAL FOR ALL UPPER PANELS)

A

A–A
(TYPICAL)

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Wing Fixed Leading Edge Skin Allowable Damage
Figure 104 (Sheet 4 of 5)

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ALLOWABLE DAMAGE ZONES
(TYPICAL FOR GFRP LOWER SKIN PANELS)

B

NOTES

- ALL GFRP/HONEYCOMB SANDWICH PANELS HAVE BMS 8-289 BONDED ALUMINUM FOIL ON THE LOWER SURFACE.
- ZONE 1: EDGEBAND AND SOLID LAMINATE AREAS BETWEEN HONEYCOMB CORE AREAS
- ZONE 2: HONEYCOMB AREAS OF THE UPPER SKIN PANELS
- ZONE 3: SOLID LAMINATE AREAS (DOES NOT INCLUDE THE SOLID LAMINATE AREA OF ZONE 1)
- ZONE 4: HONEYCOMB CORE AREAS OF THE LOWER SKIN PANELS

1 BMS 8-336 EXPANDED ALUMINUM FOIL MESH OR BMS 8-278 ALUMINUM COATED GLASS FABRIC ON THE UPPER SURFACE.

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Wing Fixed Leading Edge Skin Allowable Damage
Figure 104 (Sheet 5 of 5)

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-05	FASTENER HOLE SIZES
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
51-70-14	STRUCTURES WITH ALUMINUM COATINGS AND FOILS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits

A. Zone 1: GFRP/honeycomb Sandwich - Edgebands and Solid Laminate Area Between Honeycomb Core Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 1.0 inch in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of a fastener hole

(e) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

Table 102:

MINIMUM SPACING DIMENSIONS		
ZONE	**PART NAME**	**MINIMUM DAMAGE SPACING "A" (INCHES)**
Zone 1	Upper Panels, Lower Panels, Beam Assembly, Access Door, Removable Panels	1.5x(D+d)
Zone 2	Upper Panel, Slat 1	1.5x(D+d)

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Table 102: (Continued)

MINIMUM SPACING DIMENSIONS		
ZONE	PART NAME	MINIMUM DAMAGE SPACING "A" (INCHES)
	Upper Panel, Slat 2, Upper Panel, Slat 3, Upper Panel, Slat 4	1.5x(D+d)
Zone 3	Upper Panels	1.5x(D+d)
Zone 4	Lower Panels, Beam Assembly, Access Door, Removable Panels	1.5x(D+d)

(3) Dents that do not cause damage to the glass fibers are permitted if they are:

(a) A maximum of 0.625 inch in diameter

(b) A maximum of two plies in depth

NOTE: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 2 plies.

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.625 inch in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch from the edge of a fastener hole.

(5) Delaminations are permitted if they are:

(a) A maximum of 0.625 inch in diameter

(b) A minimum of 0.50 inch away from the edge of a fastener hole

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(6) Edge Erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

B. Zone 2: GFRP/Honeycomb Sandwich - Honeycomb Core Areas of the Upper Skin Panels

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

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(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 5.0 inches in length

(c) A minimum of 0.5 inch from the edge of a fastener hole

(d) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A maximum of 2 plies in depth

NOTE: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 2 plies.

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 0.5 inch away from a fastener hole

(b) A maximum of 2.0 inches in diameter

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch away from a fastener hole.

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C. Zone 3: GFRP/Honeycomb Sandwich - Solid Laminate Areas (does not include the edgebands and solid laminate areas in Zone 1)

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 1.0 inch in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of a fastener hole

(e) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 1.0 inch in diameter

(b) A maximum of 2 plies in depth

NOTE: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 2 plies.

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 1.0 inch in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch away from a fastener hole.

(5) Delaminations are permitted if they are:

(a) A maximum of 1.0 inch in diameter

(b) A maximum depth of 0.10 inch from an edge

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(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(6) Edge Erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

D. Zone 4: GFRP/Honeycomb Sandwich - Honeycomb Core Areas of the Lower Skin Panels

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 5.0 inches in length

(c) A minimum of 0.5 inch from the edge of a fastener hole

(d) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 2 plies in depth

NOTE: Use the limits for holes and punctures if there is fiber damage or the dent depth is more than 0.10 inch.

(b) A maximum of 2.0 inches in diameter

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(d) A minimum of 0.5 inch away from a fastener hole.

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 0.5 inch away from a fastener hole

(b) A maximum of 2.0 inches in diameter

(c) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

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2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if they are:

(a) A maximum of 2.0 inches in diameter

(b) A minimum distance away from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and given in Table 102/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(c) A minimum of 0.5 inch away from a fastener hole.

E. Aluminum - Strakelet, Panel, Overwing Fairing, and Closure Panel

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail F .

(4) Holes and Punctures are permitted if they are:

(a) A minimum of 1.0 inch away from the edge of a hole, other damage, or material edge

(b) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

1) Use a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

2) Drill the holes to the applicable size as given in 51-40-05.

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NOTE: THE DIMENSION OF A DAMAGE AREA IS EITHER THE DIMENSION OF THE VISUAL DAMAGE OR THE DIMENSION OF THE DELAMINATION. USE THE DIMENSION OF THE LARGER DAMAGE.

D IS THE LARGER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

d IS THE SMALLER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

A IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

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Damage that is Permitted to Honeycomb Core Areas
Figure 105

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Cleanup and Sealing of Edge Erosion
Figure 106

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

G00430 S0006594542_V1

Allowable Damage Limits
Figure 107 (Sheet 1 of 3)

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REMOVAL OF CORROSION AROUND THE FASTENERS

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

G00431 S0006594543_V1

Allowable Damage Limits
Figure 107 (Sheet 2 of 3)

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DENT THAT IS PERMITTED

F

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Allowable Damage Limits
Figure 107 (Sheet 3 of 3)

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REPAIR 1 - WING FIXED LEADING EDGE SKIN PANELS MADE OF COMPOSITE MATERIALS

1. Applicability

A. Repair 1 is applicable to damage to the fixed leading edge skin panels made of Glass Fiber Reinforced Plastic (GFRP) as shown in Wing Fixed Leading Edge Skin Panel Location, Figure 201/ REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Wing Fixed Leading Edge Skin Panel Location
Figure 201 (Sheet 1 of 3)

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G04086 S0006594548_V1

Wing Fixed Leading Edge Skin Panel Location
Figure 201 (Sheet 2 of 3)

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G04088 S0006594549_V1

Wing Fixed Leading Edge Skin Panel Location
Figure 201 (Sheet 3 of 3)

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2. General

A. Repair 1 gives repair instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the panels as necessary to get access to the inner surface of the skin.

(1) Remove the necessary fasteners. Refer to 51-40-02 for information on fastener removal.

(2) If a fastener hole is damaged, refer to 51-70-04 or 51-70-05, as applicable.

C. Refer to Definitions of the Damage Size, Figure 202/REPAIR 1 for the definitions of the length, width, and depth of damage.

D. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

E. Some GFRP leading edge panels have a thin layer of aluminum on the outer surface.

(1) The aluminum layer can be one of the materials that follow:

(a) BMS 8-278 aluminum coated glass fabric

(b) BMS 8-289 bonded aluminum foil

(c) BMS 8-336 expanded aluminum foil mesh

(2) Refer to 51-70-14 before you repair a skin panel that has an aluminum layer.

(3) Refer to 51-70-14 after you repair a skin panel that has an aluminum layer.

F. Do the repair as given in Paragraph 4./REPAIR 1

G. Install the wing fixed leading edge panels that were removed.

H. Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.

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SIZE DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

SIZE DEFINITIONS FOR
DENT DAMAGE

B

SIZE DEFINITIONS FOR
EDGE DAMAGE

C

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Definitions of the Damage Size
Figure 202

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G02034 S0006594551_V1

Definitions of the Facesheets
Figure 203

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-05	REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS
51-70-14	STRUCTURES WITH ALUMINUM COATINGS AND FOILS

4. Repair Instructions for the 350°F (177°C) Cure Panels

A. For dents that are a maximum of 2 inches in diameter and have no fiber damage or delamination, do the steps that follow:

(1) Fill the dent with BMS 5-28, Type 7 potting compound

(2) Apply a fiberglass patch over the potted area as given in 51-70-04.

B. For dents that are not permitted by Paragraph 4.A./REPAIR 1 and for other damage that is not permitted by Allowable Damage 1, refer to:

(1) Table 201/REPAIR 1 for panel areas other than the edgebands

(2) Table 202/REPAIR 1 for the edgebands.

Table 201:

REPAIR DATA FOR HONEYCOMB CORE AREAS OF THE 350°F (177°C) CURE WING FIXED LEADING EDGE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Contact the Boeing Company	Damage that is a maximum of: - 6.0 inches in diameter One repair for each 144 square inches	Damage that is a maximum of: - 6.0 inches in diameter One repair for each 144 square inches	There are no limits on the dimensions of the repair

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Table 201: (Continued)

REPAIR DATA FOR HONEYCOMB CORE AREAS OF THE 350°F (177°C) CURE WING FIXED LEADING EDGE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
		2.0 inches minimum clearance from: - other repairs (includes repairs on the opposite panel surface) - fastener holes - damage permitted in Allowable Damage 1 - panel edges	2.0 inches minimum clearance from: - other repairs (includes repairs on the opposite panel surface) - fastener holes - damage permitted in Allowable Damage 1 - panel edges	
REPAIR INSTRUCTIONS	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.D	SRM 51-70-05 and Paragraph 4.E	SRM 51-70-05 and Paragraph 4.F

Table 202:

REPAIR DATA FOR THE EDGEBANDS OF 350°F (177°C) CURE LEADING EDGE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP REPAIR AS GIVEN IN SRM 51-70-04**	**CATEGORY A WET LAYUP REPAIR AS GIVEN IN SRM 51-70-04**	**CATEGORY A PREIMPREGNATED LAYUP REPAIR AS GIVEN IN SRM 51-70-05**	**CATEGORY A PREIMPREGNATED LAYUP REPAIR AS GIVEN IN SRM 51-70-05**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE	Contact the Boeing Company	Damage that is a maximum of: - 3.0 inches across the largest dimension of the damage - 10 percent of the edgeband length on the side of the damage	There are no limits on the dimensions of the repair	There are no limits on the dimensions of the repair
REPAIR INSTRUCTIONS	Refer to Paragraph 4.C	Refer to Paragraph 4.D	Refer to Paragraph 4.E	Refer to Paragraph 4.F

C. Use the instructions that follow to do a Category B repair with wet layup materials at 150°F (66°C) cure.

(1) Repair the damage as given in 51-70-04, but for each facesheet or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that were removed.

(b) Do an inspection of the repair at each 800 flight hour interval or more frequently.

1) Replace the repair with a Category A repair if you find deterioration.

D. Use the instructions that follow to do a Category A repair with wet layup materials at 200°F (93°C) cure.

(1) Repair the damage as given in 51-70-04, but for each facesheet or solid laminate area that is damaged, do the steps that follow:

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(a) Use the same number of repair plies as the number of initial plies that were removed.

(b) Add one structural ply of BMS 9-3, Type H-2, or Type H-3 glass fabric that is ±45 degrees.

(c) Add a second structural ply of BMS 9-3, Type H-2 or Type H-3 glass fabric that is 0 or 90 degrees.

E. Use the instructions that follow to do a Category A repair with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05, but for each facesheet or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that were removed.

(b) Add one structural ply of BMS 8-79 glass fabric that is ±45 degrees.

(c) Add a second structural ply of BMS 8-79 glass fabric that is 0 or 90 degrees.

F. Use the instructions that follow to do a Category A repair with preimpregnated layup materials at 350°F (177°C) cure.

(1) Repair the damage as given in 51-70-05.

(2) Use the same number of repair plies as the number of initial plies that were removed.

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REPAIR 2 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE BETWEEN RIBS AND STRINGERS

1. Applicability

A. Repair 2 is applicable to damage:

(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 2

(2) That is between ribs and stringers.

2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

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Wing Fixed Leading Edge Skin Location
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION

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(Continued)

Reference	Title
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts.

(1) Make the part [1] doubler and part [2] filler. Refer to Table 201/REPAIR 2.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 2.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Repair Plate	1	Use clad or bare 2024-T3 that is 0.071 inch thick
[2]	Filler	1	Use clad 2024-T3 that is 0.063 inch thick

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [2] filler and the initial skin with BMS 5-95 sealant.

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A-A

FASTENER SYMBOLS

✦ REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW5K HEX DRIVE BOLT AND A BACC30M COLLAR.

G16142 S0006594557_V2

Layout of the Repair Parts
Figure 202

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REPAIR 3 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE AT THE FRONT SPAR

1. Applicability

A. Repair 3 is applicable to damage:

(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 3

(2) That is at the front spar.

2. General

A. Repair 3 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

G05907 S0006594559_V1

Wing Fixed Leading Edge Skin Location
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION

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(Continued)

Reference	Title
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts. Refer to Table 201/REPAIR 3.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 3.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Filler	1	Use clad 2024-T3 that is 0.063 inch thick
[2]	Tapered Filler	1	Use clad or bare 2024-T3. Use the same thickness as the attach strap
[3]	Doubler	1	Use clad or bare 2024-T3 that is 0.071 inch thick

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.

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FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A FASTENER THAT IS THE SAME TYPE AND DIAMETER AS THE INITIAL FASTENER. YOU CAN USE A FASTENER THAT IS UP TO 1/32 INCH DIAMETER OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW(5)K HEX DRIVE BOLT AND A BACC30M COLLAR. G06240 S0006594561_V2

Layout of the Repair Parts
Figure 202

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REPAIR 4 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE AT A RIB

1. Applicability

A. Repair 4 is applicable to damage:

(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 4

(2) That is at a rib.

2. General

A. Repair 4 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

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Wing Fixed Leading Edge Skin Location
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES

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(Continued)

Reference	Title
51-40-08	COUNTERSINKING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts. Refer to Table 201/REPAIR 4.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 4.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Filler	1	Use clad 2024-T3 that is 0.063 inch thick
[2]	Doubler	2	Use clad or bare 2024-T3 that is 0.071 inch thick
[3]	Filler	2	Use clad or bare 2024-T3 that is 0.080 inch thick

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.

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FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A FASTENER THAT IS THE SAME TYPE AND DIAMETER AS THE INITIAL FASTENER. YOU CAN USE A FASTENER THAT IS UP TO 1/32 INCH DIAMETER OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW5K HEX DRIVE BOLT AND A BACC30M COLLAR.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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A-A

G06277 S0006594566_V1

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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REPAIR 5 - WING INBOARD FIXED LEADING EDGE SKIN - FLUSH REPAIR FOR DAMAGE AT A STRINGER

1. Applicability

A. Repair 5 is applicable to damage:

(1) To the wing fixed leading edge skin shown in Wing Fixed Leading Edge Skin Location, Figure 201/REPAIR 5

(2) That is at a stringer.

2. General

A. Repair 5 is a Category A repair. Refer to 51-00-06 for the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

G06542 S0006594568_V1

Wing Fixed Leading Edge Skin Location
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION

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(Continued)

Reference	Title
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Remove the necessary fasteners in the area of the damaged skin. Refer to 51-40-02.

B. Cut and remove the damaged part of the skin. Refer to 51-10-02.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to a stringer or rib.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

C. Put the skin around the cut back to the initial contour.

D. Make the repair parts Refer to Table 201/REPAIR 5.

E. Assemble the repair parts as shown in Layout of the Repair Parts, Figure 202/REPAIR 5.

F. Drill the necessary fastener holes. Refer to 51-40-05 for the fastener hole dimensions.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Filler	1	Use clad 2024-T3 that is 0.063 inch thick
[2]	Doubler	1	Use clad or bare 2024-T3 that is 0.071 inch thick
[3]	Filler	1	Use clad or bare 2024-T3 that is 0.070 inch thick.
[4]	Angle	1	Use clad or bare 2024 in the annealed condition that is 0.071 inch thick. Heat treat to the T42 condition after you bend the part

G. Disassemble the repair parts.

H. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01

J. Apply two layers of BMS 10-11, Type I primer to: (Refer to SOPM 20-44-04)

(1) The internal and mating surfaces of the repair parts

(2) The internal surfaces of the internal surface of the skin.

K. Install the repair parts.

(1) Apply BMS 5-95 sealant to the mating surfaces. Refer to 51-20-05.

(2) Install the rivets without sealant.

(3) Install the hex-drive bolts, if they are used, wet with BMS 5-95 sealant.

(4) Fill the space between the part [1] filler and the initial skin with BMS 5-95 sealant.

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FASTENER SYMBOLS

- INITIAL FASTENER LOCATION. INSTALL A FASTENER THAT IS THE SAME TYPE AND DIAMETER AS THE INITIAL FASTENER. YOU CAN USE A FASTENER THAT IS UP TO 1/32 INCH DIAMETER OVERSIZE.
- REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30NW5K HEX DRIVE BOLT AND A BACC30M COLLAR.
- REPAIR FASTENER LOCATION. INSTALL A BACR15BB6D RIVET. AS AN ALTERNATIVE, YOU CAN INSTALL A BACB30MY5K HEX DRIVE BOLT AND A BACC30M COLLAR.

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Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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A-A

B-B

G14578 S0006594571_V2

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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REPAIR 6 - WING LEADING EDGE SKIN - EROSION/ EDGE STRIP REPAIR

1. Applicability

A. This procedure is applicable to airplanes that have not had winglets installed and airplanes that had winglets installed as part of the production of the airplane.

B. This procedure is not applicable to airplanes that have had winglets installed after production of the airplane. For airplanes that have had winglets installed after production, refer to the applicable Aviation Partners Boeing (APB) document.

C. This repair is applicable to the wing fixed leading edge lower panels that are made of composite, nonmetallic material and have aluminum erosion / edge strips.

Do not use this repair if the skin is delaminated.

2. General

A. All dimensions are in inches, unless shown differently.

3. References

Reference	Title
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES

4. Repair Instruction

A. Get access to the fixed leading edge panels.

B. Carefully remove the delaminated strip from edge of the panel.

C. Make the repair part. Use initial strip as a template to cut and trim the repair part. You may use Drawing No: 116A9102 to make the new part. Refer to Table 201.

Table 201: Repair Material

ITEM	PART	QUANTITY	MATERIAL
1	Erosion Strip	As Necessary	0.04 in. (1.02 mm) Thick 6061–T6 Bare Sheet as given in QQ-A-250/11.

D. Prepare the surface of the strip for Phosphoric Acid Non-Tank Anodize method. Refer to ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES, 51-70-10.

E. Apply a layer of BMS 5–89 Corrosion Inhibitive Adhesive Primer. Refer to ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES, 51-70-10.

F. Prepare the surface of the panels. Make the mating surface rough with 180–Grit sandpaper, or finer.

G. Clean the panel surface with oil-free air, or vacuum to remove loose particles.

H. Wipe the surface of the panel with a clean cloth moistened with Methyl Isobutyl Ketone (MIBK), or acetone. Dry the surface with a clean cloth before the solvent dries on the panel surface.

I. Prepare the adhesive. Use BMS 5–109, Type II, Class 2. Refer to Table 202 and mix the material thoroughly.

Table 202: Mix Ratio of Adhesive Materials

MIX RATIO, PART BY WEIGHT		
MATERIAL	PART A	PART B
EA 934NA	100 ± 3	33 ± 1

J. Apply a continuous layer of the prepared adhesive to the panel surface and aluminum strip.

K. Assemble the parts and put a constant pressure on the parts to make a close bond.

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L. Remove unwanted adhesive before it has cured. Use a clean cloth moistened with MIBK or acetone. Do not permit the solvent to flow into the bonded area.

M. Apply and keep the pressure on the parts until the adhesive is cured. Refer to Table 203 for the cure time.

Table 203: Adhesive Cure Time

CURE TEMPERATURE	MINIMUM CURE TIME
60°F (16°C) — 100°F (38°C)	7 DAYS
120°F (49°C) ± 10°F (-12°C)	25 HOURS
160°F (71°C) ± 10°F (-12°C)	9 HOURS
200°F (93°C) ± 10°F (-12°C)	1 HOUR

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL 116A9121 IS SHOWN,
LOWER SKIN PANEL 116A9111 IS SIMILAR
A-A

2333751 S0000531464_V1

Wing Fixed Leading Edge - Erosion/Edge Strip
Figure 201

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IDENTIFICATION 1 - WING INBOARD FIXED LEADING EDGE STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

K31171 S0006594575_V1

Wing Inboard Fixed Leading Edge Structure Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
116A0010	Fixed Leading Edge Collector - Final Assembly
116A0015	Functional Product Collector - Front Spar Leading Edge Join, Left Wing
116A3001	Strakelet Installation - Inboard Fixed Leading Edge Wing
116A3010	Strakelet Assembly - Inboard Fixed Leading Edge Wing
116A4008	Skin Installation - Upper, Inboard leading Edge, Wing

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LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

K31260 S0006594577_V2

Wing Inboard Fixed Leading Edge Structure
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Hinge Rib (6)		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part	
[2]	Stringer (11)	0.071 (1.8)	7075-T62 sheet as given in in QQ-A-250/12	
[3]	Stringer (9)	0.080 (2.03)	7075-T62 sheet as given in QQ-A-250/12	
[4]	Stringer (4)	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
[5]	Stringer (3)		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part	
[6]	Seal Rib		7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part	
[7]	Nose Beam	0.063 (1.6)	7075-T62 clad sheet as given in BMS 7-302	
[8]	Strut		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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K31302 S0006594579_V2

Wing Inboard Leading Edge Straklet Structure Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Seal Rib Assembly			
	Seal Rib		7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part	
	Web	0.063 (1.6)	2024-T3 clad sheet as given in QQ-A-250/5	
[2]	Web	0.063 (1.6)	Ti-6Al-4V titanium sheet as given in MIL-T-9046, Code AB-1, Condition A	
[3]	Bulkhead Assembly			
	Web	0.032 (0.81)	2024-T3 clad sheet as given in QQ-A-250/5	
	Chord		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part	
	Stiffener (2)	0.050 (1.27)	7075-T62 clad sheet as given in QQ-A-250/13	
	Stiffener	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
	Angle	0.050 (1.27)	7075-T62 clad sheet as given in QQ-A-250/13	
[4]	Stiffener	0.071 (1.8)	7075-T62 sheet as given in QQ-A-250/12	
[5]	Web Assembly			
	Web	0.032 (0.81)	2024-T42 clad sheet as given in QQ-A-250/5	
	Angle	0.040 (1.01)	2024-T42 clad sheet as given in QQ-A-250/5	
[6]	Rib Assembly			
	Internal Rib	0.050 (1.27)	7075-T62 clad sheet as given in QQ-A-250/13	
	Angle	0.050 (1.27)	7075-T62 clad sheet as given in QQ-A-250/13	
[7]	Stiffener	0.050 (1.27)	7075-T62 sheet as given in QQ-A-250/12	
[8]	Frame	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
[9]	Splice Strap	0.071 (1.8)	7075-T6 clad sheet as given in QQ-A-250/13	
[10]	Lower Splice Strap	0.071 (1.8)	7075-T62 clad sheet as given in QQ-A-250/13	
[11]	End Cap	0.063 (1.6)	6061-T62 sheet as given in QQ-A-250/11	
[12]	Top Cover Assembly			
	Stiffener	0.050 (1.27)	7075-T62 clad sheet as given in QQ-A-250/13	
	Top Cover	0.040 (1.01)	7075-T6 clad sheet as given in QQ-A-250/13	
[13]	End Rib Assembly			
	End Rib		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part	
	Attach Angle	0.050 (1.27)	7075-T62 clad sheet as given in QQ-A-250/13	
	Frame	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
	Upper Splice Strap	0.071 (1.80)	7075-T62 clad sheet as given in QQ-A-250/13	
[14]	Nose Beam Splice	0.090 (2.29)	7075-T62 clad sheet as given in BMS 7-302	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 2 - WING OUTBOARD FIXED LEADING EDGE STRUCTURE

1. Locator

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE (EXCEPT AS NOTED)

K27837 S0006594582_V1

Wing Outboard Fixed Leading Edge Structure Locations
Figure 1

2. Reference Drawings

A. Reference Drawings

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
116A0010	Fixed Leading Edge Collector - Final Assembly
116A0011	Fixed Leading Edge Collector - Post Paint - Left Wing
116A0013	Collector - Leading Edge, Wing Build-Up, Left Wing
116A0015	Functional Product Collector - Front Spar Leading Edge Join, Left Wing
116A5600	Nose Beam Installation - Outboard Fixed Leading Edge
116A7000	Rib Installation - Outboard Fixed Leading Edge

3. Applicability

A. Identification 2 is applicable to:

(1) Airplanes that have not had winglets installed.

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(2) Airplanes that have had winglets installed as part of the production of the airplane.

NOTE: Identification 2 is not applicable to airplanes that have had winglets installed after production of the airplane. Refer to Aviation Partners Boeing (APB) documents AP37.8-0602 and AP37.8-0615 for the winglet data.

(a) For airplanes with cum line numbers 1 thru 1302:

1) Refer to drawing 119A0001 for a list of airplanes that were delivered with winglets installed.

(b) For airplanes with cum line numbers 1303 and on:

1) Refer to drawing 119A0105 for a list of airplanes that were delivered with winglets installed.

4. Wing Outboard Fixed Leading Edge Structure Identification

A. Lists of materials:

Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[2]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[3]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[4]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[5]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[6]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[7]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[8]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[9]	Splice Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[10]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[11]	Nose Beam (4)	0.050 (12.7)	7075-T62 clad sheet as given in BMS 7-302	
[12]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Splice Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	

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Table 3: (Continued)

LIST OF MATERIALS FOR FIGURE 3				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[2]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[3]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[4]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[5]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[6]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[7]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[8]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[9]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[10]	Nose Beam (4)	0.050 (12.7)	7075-T62 clad sheet as given in BMS 7-302	
[11]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[12]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[13]	Nose Beam	0.063 (1.6)	7075-T62 clad sheet as given in BMS 7-302	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

Table 4:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[1]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[2]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[3]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[4]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[5]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[6]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[7]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[8]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[9]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	

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Table 4: (Continued)

LIST OF MATERIALS FOR FIGURE 4				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[10]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[11]	Splice Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[12]	Nose Beam (4)	0.050 (12.7)	7075-T62 clad sheet as given in BMS 7-302	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

Table 5:

LIST OF MATERIALS FOR FIGURE 5				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[1]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[2]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[3]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[4]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[5]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[6]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[7]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[8]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[9]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[10]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[11]	Splice Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[12]	Nose Beam (4)	0.050 (12.7)	7075-T62 clad sheet as given in BMS 7-302	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

Table 6:

LIST OF MATERIALS FOR FIGURE 6				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[1]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[2]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[3]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	

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Table 6: (Continued)

LIST OF MATERIALS FOR FIGURE 6				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[4]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[5]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[6]	Stop Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[7]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[8]	Auxiliary Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[9]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[10]	Main Track Rib		7050-T7451 machined plate as given in AMS 4050 (Grain direction controlled part)	
[11]	Seal Rib		7050-T7451 machined plate as given in BMS 7-323 (Grain direction controlled part)	
[12]	Nose Beam (4)	0.050 (12.7)	7075-T62 clad sheet as given in BMS 7-302	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

SLAT NUMBER 4

K28408 S0006594589_V1

Wing Outboard Fixed Leading Edge Slat Number 4 Structure Identification
Figure 2

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

SLAT NUMBER 5

K29071 S0006594590_V2

Wing Outboard Fixed Leading Edge Slat Number 5 Structure Identification
Figure 3

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

SLAT NUMBER 3 IS SHOWN, SLAT NUMBER 6 IS OPPOSITE

K28194 S0006594591_V1

Wing Outboard Fixed Leading Edge Slat Number 3 Structure Identification
Figure 4

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NOTE: REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

SLAT NUMBER 2 IS SHOWN, SLAT NUMBER 7 IS OPPOSITE

K29200 S0006594592_V1

Wing Outboard Fixed Leading Edge Slat Number 2 Structure Identification
Figure 5

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NOTES

- REFER TO TABLE 6 FOR THE LIST OF MATERIALS.

1 FOR CUM LINE NUMBER 789 AND ON WITH WINGLET FULL PROVISION INSTALLATION.

SLAT NUMBER 1 IS SHOWN, SLAT NUMBER 8 IS OPPOSITE

L60593 S0006594593_V1

Wing Outboard Fixed Leading Edge Slat Number 1 Structure Identification
Figure 6

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ALLOWABLE DAMAGE 1 - WING INBOARD FIXED LEADING EDGE STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing inboard fixed leading edge structure shown in Wing Inboard Fixed Leading Edge Structure Location, Figure 101/ALLOWABLE DAMAGE 1.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G28442 S0006594596_V1

Wing Inboard Fixed Leading Edge Structure Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G28421 S0006594597_V1

Wing Inboard Fixed Leading Edge Structure
Figure 102 (Sheet 1 of 3)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BEAM

A

G35504 S0006594598_V1

Wing Inboard Fixed Leading Edge Structure
Figure 102 (Sheet 2 of 3)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

TYPICAL HINGE RIB

B

G28400 S0006594599_V1

Wing Inboard Fixed Leading Edge Structure
Figure 102 (Sheet 3 of 3)

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

(1) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Ribs and Struts

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , D , E , F , H , and I .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Stringers

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(2) Nicks, Gouges, Scratches, and Corrosion:

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(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and H.

(3) Dents are not permitted.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter

(b) A minimum of 4.0 inches away from other damage

(c) A minimum of 1.0 inch away from a fastener hole

(d) A minimum of 0.50 inch away from a material edge

(e) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

1) Install the rivet without sealant.

C. Beam

(1) Webs

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , and J .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , E , F , H , and J .

(c) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail G .

(d) Holes and Punctures are permitted if they are:

1) A maximum of 0.25 inch in diameter

2) A minimum of 4.0 inches away from other damage

3) A minimum of 1.0 inch away from a fastener hole

4) A minimum of 0.50 inch away from a material edge

5) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

a) Install the rivet without sealant.

(2) Stiffeners

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , and C .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 1, Details A , B , C , E , F , and I .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

D. Nose Beam

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and J .

(2) Nicks, Gouges, Scratches, and Corrosion:

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(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , F , H , and J .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail G .

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter

(b) A minimum of 4.0 inches away from other damage

(c) A minimum of 1.0 inch away from a fastener hole

(d) A minimum of 0.50 inch away from a material edge

(e) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

1) Install the rivet without sealant.

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Allowable Damage Limits
Figure 103 (Sheet 1 of 8)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

G28449 S0006594601_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 8)

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REMOVAL OF CORROSION AROUND THE FASTENERS

F

B–B

G28501 S0006594602_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 8)

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DENT THAT IS PERMITTED

G

REMOVAL OF DAMAGED MATERIAL FROM A WEB
(A TEE SECTION IS SHOWN)

H

G28502 S0006594603_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 8)

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D_2 = DIAMETER OF THE HOLE AT LOCATION 2

H = HEIGHT OF THE WEB

T = THICKNESS OF THE WEB

A_i = INITIAL AREA OF THE WEB
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $HT - D_2T$

A_1 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

A_3 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 3

$\left(\frac{A_1 + A_3}{A_i}\right) \times 100$ = PERCENT OF CROSS-SECTIONAL AREA REMOVED
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE WEB.

(ROTATED 90° CLOCKWISE)
D-D

G35296 S0006594604_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 8)

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REMOVAL OF DAMAGE MATERIAL ON A FLANGED SURFACE
(A TEE SECTION IS SHOWN)

I

G35583 S0006594605_V1

Allowable Damage Limits
Figure 103 (Sheet 6 of 8)

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D_3 = DIAMETER OF THE HOLE AT LOCATION 3

T = THICKNESS OF THE FLANGE

A_i = INITIAL AREA OF THE FLANGE
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $4T - D_3T$

A_1 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

A_2 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 2

A_4 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 4

$$\left(\frac{A_1 + A_2 + A_4}{A_i}\right) \times 100 = \text{PERCENT OF CROSS-SECTIONAL AREA REMOVED}$$
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE FLANGE SURFACE.

E-E

G35606 S0006594606_V1

Allowable Damage Limits
Figure 103 (Sheet 7 of 8)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL WEB

J

G36901 S0006594607_V1

Allowable Damage Limits
Figure 103 (Sheet 8 of 8)

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ALLOWABLE DAMAGE 2 - WING OUTBOARD FIXED LEADING EDGE STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing outboard fixed leading edge structure shown in Wing Outboard Fixed Leading Edge Structure Locations, Figure 101/ALLOWABLE DAMAGE 2.

G28525 S0006594609_V1

Wing Outboard Fixed Leading Edge Structure Locations
Figure 101

2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

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(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the procedures that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the main track ribs.

(a) Refer to 51-20-06 for shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(2) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.

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SLAT NUMBER 3 IS SHOWN, SLAT NUMBERS 2 AND 4 ARE SIMILAR
SLAT NUMBERS 5, 6, AND 7 ARE OPPOSITE

G29036 S0006594610_V2

Wing Outboard Fixed Leading Edge Slat Number 3 Structure
Figure 102

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NOTES

1 FOR CUM LINE NUMBER 789 AND ON WITH FULL WINGLET PROVISION INSTALLATION.

SLAT NUMBER 1 IS SHOWN, SLAT NUMBER 8 IS OPPOSITE

L60592 S0006594611_V1

Wing Outboard Fixed Leading Edge Slat Number 1 Structure
Figure 103

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Ribs and Upstop Fitting

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , D , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , D , E , F , H , I , and J .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Nose Beam

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , C , E , and F .

(3) Dents are permitted as given in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail G .

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter.

2) A minimum of 4.0 inches away from other damage.

3) A minimum of 1.0 inch away from a fastener hole.

4) A minimum of 0.50 inch away from a material edge.

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5) Filled with a 2117-T3 or 2117-T4 protruding head rivet.
   a) Install the rivet without sealant.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

G28540 S0006594613_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 7)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

D

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

G28543 S0006594614_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 7)

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REMOVAL OF CORROSION AROUND THE FASTENERS

F

DENT THAT IS PERMITTED

G

G28549 S0006594615_V1

Allowable Damage Limits
Figure 104 (Sheet 3 of 7)

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REMOVAL OF DAMAGED MATERIAL ON AN EDGE

H

REMOVAL OF DAMAGED MATERIAL FROM A WEB OF A TEE SECTION

I

G35305 S0006594616_V1

Allowable Damage Limits
Figure 104 (Sheet 4 of 7)

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D_2 = DIAMETER OF THE HOLE AT LOCATION 2

H = HEIGHT OF THE WEB

T = THICKNESS OF THE WEB

A_i = INITIAL AREA OF THE WEB
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $HT - D_2T$

A_1 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

A_3 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 3

$\left(\frac{A_1 + A_3}{A_i}\right) \times 100$ = PERCENT OF CROSS-SECTIONAL AREA REMOVED
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE WEB.

(ROTATED 90° CLOCKWISE)
D-D

G35306 S0006594617_V1

Allowable Damage Limits
Figure 104 (Sheet 5 of 7)

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REMOVAL OF DAMAGE MATERIAL ON A FLANGED SURFACE
(A TEE SECTION IS SHOWN)

J

G35609 S0006594618_V1

Allowable Damage Limits
Figure 104 (Sheet 6 of 7)

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D_3 = DIAMETER OF THE HOLE AT LOCATION 3

T = THICKNESS OF THE FLANGE

A_i = INITIAL AREA OF THE FLANGE
= THE TOTAL CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLES (AS MANUFACTURED BY BOEING)
= $4T - D_3T$

A_1 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 1

A_2 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 2

A_4 = CROSS-SECTIONAL AREA OF THE DAMAGE THAT IS REMOVED AT LOCATION 4

$$\left(\frac{A_1 + A_2 + A_4}{A_i}\right) \times 100 = \text{PERCENT OF CROSS-SECTIONAL AREA REMOVED}$$
= A MAXIMUM OF 10 PERCENT

THE TOTAL CROSS-SECTIONAL AREA REMOVED IN ALL ZONES A-B (2.00 INCHES ON EACH SIDE OF A LINE A-B) MUST NOT BE MORE THAN 10 PERCENT OF THE INITIAL AREA OF THE FLANGE SURFACE.

E-E

G35611 S0006594619_V1

Allowable Damage Limits
Figure 104 (Sheet 7 of 7)

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REPAIR 1 - WING INBOARD FIXED LEADING EDGE STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G34325 S0006594622_V3

Wing Inboard Fixed Leading Edge Structure Location
Figure 201

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REPAIR 2 - WING OUTBOARD FIXED LEADING EDGE STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G34315 S0006594624_V3

Wing Outboard Fixed Leading Edge Structure Locations
Figure 201

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IDENTIFICATION 1 - WING LEADING EDGE SLAT SKIN

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H46980 S0006594629_V1

Wing Leading Edge Slat Skin Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
114A0011	Left Wing, Slat Functional Product Collector
114A0012	Right Wing, Slat Functional Product Collector
114A4002	Slat Installation, Outboard Leading Edge, Wing
114A5010	Slat Assembly - No. 1 and 8
114A5020	Slat Assembly - No. 2 and 7
114A5030	Slat Assembly - No. 3 and 6
114A5040	Slat Assembly - No. 4 and 5

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

H47545 S0006594631_V1

Wing Leading Edge Slat Number 4 and Number 5 Skin Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Nose Skin	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
[2]	Wedge Assembly			LINE NUMBER 1 TO 19
	Upper Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Upper Doubler	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core (2)		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 6	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-10, Class NPA, Grade I. Refer to Figure 6	
[2]	Wedge Assembly			LINE NUMBER 20 TO 3569
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7	
[2]	Wedge Assembly			LINE NUMBER 3570 AND ON
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler	0.200 (5.06)	2024-T351 plate as given in QQ-A-250/4 (Optional: 7075–T651 plate as given in QQ-A-250/12)	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 8	
[3]	Backup Plate (2)	0.032 (0.81)	6013-T4 sheet as given in AMS 4347	
[4]	Cove Skin (4)	0.050 (1.27)	7075-T62 sheet as given in QQ-A-250/12	
[5]	Cove Skin (2)	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
[6]	Cove Skin Assembly			
	Cove Skin	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
	Doubler	0.160 (4.06)	7075-T62 sheet as given in QQ-A-250/12	
[7]	Cover Plate	0.050 (1.27)	7075-T6 sheet as given in QQ-A-250/12	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[8]	Flex Skin (3)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9	
[9]	Flex Tab (2)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 10	LINE NUMBER 1 TO 19
[9]	Flex Tab (2)		Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 11	LINE NUMBER 20 AND ON

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

H47869 S0006594633_V1

Wing Leading Edge Slat Number 3 and Number 6 Skin Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin Tab (2)	0.100 (2.54)	2024-T3 sheet as given in QQ-A-250/4	
[2]	Wedge Assembly			LINE NUMBER 1 TO 19
	Upper Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Upper Doubler	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core (2)		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 6	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-10, Class NPA, Grade I. Refer to Figure 6	
[2]	Wedge Assembly			LINE NUMBER 20 TO 3569
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7	
[2]	Wedge Assembly			LINE NUMBER 3570 AND ON
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler	0.200 (5.06)	2024-T351 plate as given in QQ-A-250/4 (Optional: 7075–T651 plate as given in QQ-A-250/12)	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 8	
[3]	Nose Skin	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
[4]	Backup Plate (2)	0.032 (0.81)	6013-T4 sheet as given in AMS 4347	
[5]	Cove Skin (4)	0.050 (1.27)	7075-T62 sheet as given in QQ-A-250/12	
[6]	Cove Skin (2)	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
[7]	Cove Skin Assembly			
	Cove Skin	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
	Doubler	0.160 (4.06)	7075-T62 sheet as given in QQ-A-250/12	
[8]	Cover Plate	0.063 (1.6)	7075-T6 sheet as given in QQ-A-250/12	

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Table 3: (Continued)

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[9]	Flex Skin (3)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9	
[10]	Flex Tab (2)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 10	LINE NUMBER 1 TO 19
[10]	Flex Tab (2)		Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 11	LINE NUMBER 20 AND ON

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

H47955 S0006594635_V2

Wing Leading Edge Slat Number 2 and Number 7 Skin Identification
Figure 4

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Table 4:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin Tab (2)	0.100 (2.54)	2024-T3 sheet as given in QQ-A-250/4	
[2]	Vortilon (2)		7050-T7451 plate as given in AMS 4050 (Optional: A356.0-T6 casting as given in AMS 4218)	
[3]	Wedge Assembly			LINE NUMBER 1 TO 19
	Upper Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Upper Doubler	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core (2)		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 6	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-10, Class NPA, Grade I. Refer to Figure 6	
[3]	Wedge Assembly			LINE NUMBER 20 TO 3569
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7	
[3]	Wedge Assembly			LINE NUMBER 3570 AND ON
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler	0.200 (5.06)	2024-T351 plate as given in QQ-A-250/4 (Optional: 7075–T651 plate as given in QQ-A-250/12)	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 8	
[4]	Nose Skin	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
[5]	Backup Plate (2)	0.032 (0.81)	6013-T4 sheet as given in AMS 4347	
[6]	Cove Skin (4)	0.050 (1.27)	7075-T62 sheet as given in QQ-A-250/12	
[7]	Cove Skin (2)	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
[8]	Cove Skin Assembly			
	Cove Skin	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
	Doubler	0.160 (4.06)	7075-T62 sheet as given in QQ-A-250/12	

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Table 4: (Continued)

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[9]	Cover Plate	0.063 (1.6)	7075-T6 sheet as given in QQ-A-250/12	
[10]	Flex Skin (3)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9	
[11]	Flex Tab (2)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 10	LINE NUMBER 1 TO 19
[11]	Flex Tab (2)		Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 11	LINE NUMBER 20 AND ON

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

H48319 S0006594637_V1

Wing Leading Edge Slat Number 1 and Number 8 Skin Identification
Figure 5

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Table 5:

LIST OF MATERIALS FOR FIGURE 5				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin Tab (2)	0.100 (2.54)	2024-T3 sheet as given in QQ-A-250/4	
[2]	Vortilon		7050-T7451 plate as given in AMS 4050 (Optional: A356.0-T6 casting as given in AMS 4218)	
[3]	Wedge Assembly			LINE NUMBER 1 TO 19
	Upper Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Upper Doubler	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core (2)		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 6	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-10, Class NPA, Grade I. Refer to Figure 6	
[3]	Wedge Assembly			LINE NUMBER 20 TO 3569
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 7	
[3]	Wedge Assembly			LINE NUMBER 3570 AND ON
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler	0.200 (5.06)	2024-T351 plate as given in QQ-A-250/4 (Optional: 7075–T651 plate as given in QQ-A-250/12)	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I. Refer to Figure 8	
[4]	Nose Skin	0.063 (1.6)	7075-T62 clad sheet as given in QQ-A-250/13	
[5]	Cove Skin (4)	0.050 (1.27)	7075-T62 sheet as given in QQ-A-250/12	
[6]	Cove Skin (4)	0.063 (1.6)	7075-T62 sheet as given in QQ-A-250/12	
[7]	Flex Skin (3)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 9	
[8]	Flex Tab (2)		Epoxy impregnated glass Woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781. Refer to Figure 10	LINE NUMBER 1 TO 19

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Table 5: (Continued)

LIST OF MATERIALS FOR FIGURE 5				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[8]	Flex Tab (2)		Woven fiberglass reinforced polyetherimide as given in BMS 8-353, Type 1, Class 1, Grade 11A. Refer to Figure 11	LINE NUMBER 20 AND ON

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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H83889 S0006594654_V2

Core Ribbon Direction for the Leading Edge Slat Wedge Assembly
Figure 6

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CORE RIBBON DIRECTION

A

ROTATED 90° CLOCKWISE
A-A

U52283 S0000203118_V1

Core Ribbon Direction for the Leading Edge Slat Wedge Assembly
Figure 7

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Core Ribbon Direction for the Leading Edge Slat Wedge Assembly
Figure 8

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION

A

ROTATED 90 ° COUNTER CLOCKWISE
PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION.
- REFER TO SECTION A-A FOR THE PLY DIRECTION AT THAT LOCATION.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

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Ply Direction and Ply Sequence for the Leading Edge Slat Flex Skin
Figure 9

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 8		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781

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NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H83833 S0006594657_V2

Ply Direction and Ply Sequence for the Leading Edge Slat Flex Tab
Figure 10

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 9		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P14	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE

PLY LAYUP DIRECTION

A

ROTATED 90 ° COUNTER CLOCKWISE

PLY LAYUP SEQUENCE

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

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Ply Direction and Ply Sequence for the Leading Edge Slat Flex Tab
Figure 11

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P11	0 or 90 degrees	Woven fiberglass reinforced polyetherimide as given in BMS 8–353, Type 1, Class 1, Grade 11A.

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ALLOWABLE DAMAGE 1 - WING LEADING EDGE SLAT SKIN

1. Applicability

A. This subject gives the allowable damage limits for the wing leading edge slat skin shown in Wing Leading Edge Slat Skin Locations, Figure 101/ALLOWABLE DAMAGE 1.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Wing Leading Edge Slat Skin Locations
Figure 101

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NOTE: ALL PARTS ARE MADE OF ALUMINUM SHEET.

SLAT NUMBER 2 IS SHOWN, SLAT NUMBERS 1, 3, AND 4 ARE SIMILAR, SLAT NUMBERS 5, 6, 7, AND 8 ARE OPPOSITE

A TYPICAL SECTION IS SHOWN
A-A

NOTES

1 ALLUMINUM ARROWHEAD TIP CONFIGURATION FOR LINE NUMBERS 3464, 3570 AND ON, OR LINE NUMBERS 1-3463 OR 3465-3569 WITH THE ALUMINUM ARROWHEAD TIP INSTALLED AFTER PRODUCTION. BOTH CONFIGURATIONS ARE INTERCHANGEABLE AND CAN BE INTERMIXED.

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Wing Leading Edge Slat Skin
Figure 102

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Do the steps that follow for the aluminum parts.

(1) Remove the damage as necessary.

(a) Refer to 51-10-02 for the inspection and removal of damage.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do the procedures that follow:

(a) Apply a chemical conversion coating to the surfaces of the reworked area. Refer to 51-20-01.

(b) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.

NOTE: Make sure the aerodynamic smoothness on the external surfaces is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

C. Do the steps that follow for the flex skin made of Glass Fabric Reinforced Plastic (GFRP) and to the trailing edge wedge.

(1) Do an inspection of the damaged area to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other equivalent inspection methods that have been examined and found to be satisfactory by the operator, can be used.

(a) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(2) Remove all the contamination and water from the structure.

(a) Refer to 51-70-04 for the damage removal procedures.

(b) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Use a vacuum and heat as necessary to remove moisture from the solid laminate. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Make sure the tape is in satisfactory condition at normal maintenance intervals.

(e) Seal the damage permanently.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(4) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

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(a) Use a vacuum and heat as necessary to remove moisture from the solid laminate. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Seal the damage permanently no later than 24 months from the time the seal was made.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM 51-21-00/701.

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DEFINITIONS FOR
NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR
DENT DAMAGE

B

DEFINITIONS FOR
EDGE DAMAGE

C

G39728 S0006594663_V1

Damage Definitions
Figure 103

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-08	EROSION PROTECTION
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
57-42-01, IDENTIFICATION 1	Wing Leading Edge Slat Skin
57-42-01, REPAIR 6	Leading Edge Slat Trailing Edge Wedge Lower Skin Wear Repair
AMM 51-21-81-100-801	Prepare the Surface to be Finished (P/B 701)
AMM 51-21-81-370-801	Apply The Abrasion-Resistant Teflon Finish, BMS 10-86 Type I (P/B 701)
AMM 51-21-81-370-802	Apply The Abrasion-Resistant Teflon Finish, BMS 10-86 Type II (P/B 701)
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits

A. Nose Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail F .

1) If the damage agrees with one or more of the conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than:

- 0.10 in. (2.54 mm) and located aft of the nose beam, or
- 0.060 in. (1.524 mm) and located forward of the nose beam.

b) The distance between two dents or between a dent and other damage is less than 0.50D (D = the larger dimension of the two damaged areas).

c) There are more than 10 damage locations on a slat.

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2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval, or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 1.00 inch away from a fastener hole, material edge, or other damage

3) Filled with a 2117-T3 or 2117-T4 countersunk rivet.

a) Install the rivet without sealant.

(5) Erosion:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 105/ ALLOWABLE DAMAGE 1, Detail I if you do as follows:

1) Blend the damage that is in the skin with a powdered household cleanser or 400 grit aluminum oxide paper.

WARNING: FLIGHT OPERATION IS NOT PERMITTED IF THE SLAT LEADING EDGE ROUGHNESS IS EQUIVALENT TO OR MORE THAN THAT WHICH IS MADE WITH 240 GRIT SANDPAPER.

2) You apply an erosion coating or speed tape as given in 51-20-08.

B. Trailing Edge Wedge (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Clean up cracks on the edges as shown in Allowable Damage Limits, Figure 105/ ALLOWABLE DAMAGE 1, Detail A .

(b) Cracks are permitted in the honeycomb area if:

1) The maximum length of the damage is 1.50 inches.

2) You stop drill the ends of the crack with a 0.19 inch diameter hole. Refer to 51-10-02.

3) A minimum of 1.0 inch from a panel edge.

4) You remove water and all contaminants.

5) You make an inspection of the damaged area every 250 flight cycles, or more frequently.

6) You make a permanent repair in less than 24 months.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents:

(a) Refer to Figure 104 (Sheet 1) to determine which Zone the dent is located for the Upper Surface. Refer to Figure 104 (Sheet 2) to determine which Zone the dent is located for the Lower Surface.

I (b) Refer to Figure 105 (Sheet 3), Detail G and Table 101/ALLOWABLE DAMAGE 1 to determine the allowable dent limit.

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Table 101: Dent Allowable Limits

		Maximum Depth of Dent	Maximum Diameter of Dent	Minimum Distance Between Dents
Upper Surface	Zone 1	Dents are not permitted	Dents are not permitted	Dents are not permitted
	Zone 2	0.04 inch (1.02 mm)	2.0 inch (50.8 mm)	1.0 D
	Zone 3	0.12 inch (3.05 mm)	2.0 inch (50.8 mm)	1.0 D
	Zone 4	0.04 inch (1.02 mm)	0.70 inch (17.8 mm)	4.0 D
Lower Surface	Zone 1	Dents are not permitted	Dents are not permitted	Dents are not permitted
	Zone 2	0.04 inch (1.02 mm)	2.0 inch (50.8 mm)	1.0 D
	Zone 3	0.12 inch (3.05 mm)	2.0 inch (50.8 mm)	1.0 D

(c) If the damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51–70–01.

1) The dent is located within Zone 3 on either the upper or lower surface, the depth of the dent is more than 0.12 in. (3.05 mm) and less than 0.20 in. (5.08 mm), the diameter of the dent is less than or equal to 2.0 in. (50.8 mm), and the dent is within the limits of Figure 105 (Sheet 3), Detail G.

2) The distance between two dents or between a dent and other damage is less than 1.0 D (D = the larger dimension of the two damage areas).

(d) If you fill or rework the dent, do the steps that follow:

1) Do an inspection of the damage area every 250 flight cycles, or more frequently. Repair the damage in less than 24 months.

2) Repair the dent if the damage becomes larger.

3) Remove water and all contaminants.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) Cleaned up to a maximum of 0.25 inch in diameter.

2) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

(b) Remove water and all contaminants.

(c) Do an inspection of the damage every 250 flight cycles.

(d) Repair the damage in less than 24 months.

(5) Delaminations are permitted away from the edges if they are:

(a) A maximum of 2.25 inches in diameter.

(b) A minimum of 4D (D= the larger dimension of the two damaged areas) from the edge of the other damage.

(c) A minimum of 1.0 inch away from the panel edge.

(d) Inspected every 250 flight cycles, or more frequently. Repair the damage in less than 24 months.

(6) Delaminations are permitted at panel edges if they are:

(a) A maximum of 0.10 inch along the panel edge.

(b) Sealed as shown in Paragraph 2.

(c) Inspected every 250 flight cycles, or more frequently. Repair the damage in less than 24 months.

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(7) Slat Wedge Wear Damage (Fiberglass Tip Configuration only):

(a) Applicability and location of the fiberglass tip configuration can be found in 57-42-01, IDENTIFICATION 1 and Figure 102/ALLOWABLE DAMAGE 1.

(b) Wear damage on the lower skin is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail J if it is:

1) A maximum of 0.015 in. (0.38 mm) deep with a maximum width of 0.625 in. (15.88 mm) as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail J. Apply a chemical conversion coating to the bare surfaces of the skin. Refer to 51-20-01, GENERAL. Apply a layer of BMS 10–79, Type III primer. Refer to SOPM 20-44-04.

2) If the depth of wear is more than 0.015 in. (0.38 mm) into the lower skin and less than or equal to 0.019 in. (0.48 mm) into the fiberglass filler, repair the damage per 57-42-01, REPAIR 6.

3) If the depth of the wear into the fiberglass filler is more than 0.019 in. (0.48 mm), repair the damage per 51–70–10, Repair 4, Figure 202, Detail H or Figure 203, Detail A. Use BMS 5–101 film adhesive for the repairs.

(8) Slat Wedge Wear Damage (Aluminum Arrowhead Tip Configuration only):

(a) Applicability and location of the aluminum arrowhead tip configuration can be found in 57-42-01, IDENTIFICATION 1 and Figure 102/ALLOWABLE DAMAGE 1.

(b) Wear damage on the machined aluminum trailing edge is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Details K and L if it is:

1) A maximum of 0.10 in. (2.5 mm) trim over a maximum length of 17 in. (431.8 mm) is permitted, as shown in Figure 105/ALLOWABLE DAMAGE 1, Details K and L.

2) The minimum trailing edge tip thickness is 0.02 in. (0.5 mm), as shown in Figure 105/ALLOWABLE DAMAGE 1 Detail L.

3) It is permitted to trim the lower surface to a maximum distance of 0.50 in. (12.7 mm) away from the initial tip as shown in Figure 105 (Sheet 6).

4) Apply a chemical conversion cotating to the bare surfaces of the skin. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01. Apply a layer of BMS 10-79, Type III primer. Refer to SOPM 20-44-04.

5) Apply an abrasion-resistant teflon finish to the damaged area. Prepare the surface as given in AMM TASK 51-21-81-100-801. Either spray the coating as per AMM TASK 51-21-81-370-801 or apply with brush as per AMM TASK 51-21-81-370-802.

C. Cove Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E.

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail F .

(4) Holes and Punctures:

(a) The damage is permitted if it is:

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1) A maximum of 0.25 inch in diameter
2) A minimum of 1.00 inch away from a fastener hole, material edge, or other damage
3) Filled with a 2117-T3 or 2117-T4 countersunk rivet.
   a) Install the rivet without sealant.

D. Flex Skin (GFRP Solid Laminate)

NOTE: Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Nicks, gouges, and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges, and scratches that cause damage to the glass fibers are permitted if they are:
   (a) A maximum of one ply in depth

   NOTE: Use the limits for holes and punctures if the damage is more than one ply.

   (b) A maximum of 2.0 inches in length
   (c) A maximum of 0.25 inch in width
   (d) A minimum of 0.50 inch away from the edge of a fastener hole
   (e) A minimum distance away from the edge of other damage, a fastener hole or part edge as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 106/ ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:
      1) Do not cause damage to the glass fiber plies
      2) Are sealed as given in Paragraph 2.C./ALLOWABLE DAMAGE 1

(3) Dents are permitted if they are:
   (a) A maximum of 2.0 inches in diameter
   (b) A maximum of 2 plies in depth
   (c) A minimum distance of 0.50 inch away from fasteners, a part edge, or other damage, a fastener hole as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 106/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:
      1) Do not cause damage to the glass fiber plies
      2) Are sealed as given in Paragraph 2.C./ALLOWABLE DAMAGE 1

(4) Holes and punctures are permitted if they are:
   (a) A maximum of 2.0 inches in diameter
   (b) A minimum distance from fasteners, a part edge, or other damage as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 106/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:
      1) Do not cause damage to the glass fiber plies
      2) Are sealed as given in Paragraph 2.C./ALLOWABLE DAMAGE 1

(5) Delaminations are permitted if they are:
   (a) A maximum of 2.0 inches in diameter
   (b) A minimum distance from fasteners, a part edge, or other damage as shown in Minimum Damage Spacing in Solid Laminate Parts, Figure 106/ALLOWABLE DAMAGE 1. Other damage does not include nicks, gouges, and scratches that:
      1) Do not cause damage to the glass fiber plies

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2) Are sealed as given in Paragraph 2.C./ALLOWABLE DAMAGE 1

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1 FOR ALUMINUM ARROWHEAD TRAILING EDGE, X = 1.60 INCH (40.64 mm)
FOR FIBERGLASS FILLER TRAILING EDGE, X = 1.30 INCH (33.02 mm)

UPPER SURFACE

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Trailing Edge Wedge Dent Allowable Zones
Figure 104 (Sheet 1 of 2)

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Trailing Edge Wedge Dent Allowable Zones
Figure 104 (Sheet 2 of 2)

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Allowable Damage Limits
Figure 105 (Sheet 1 of 6)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

REMOVAL OF CORROSION AROUND THE FASTENERS

E

G37994 S0006594665_V1

Allowable Damage Limits
Figure 105 (Sheet 2 of 6)

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DENT THAT IS PERMITTED IN A METAL SKIN

F

DENT THAT IS PERMITTED IN METAL HONEYCOMB STRUCTURE

G

H81585 S0006594666_V2

Allowable Damage Limits
Figure 105 (Sheet 3 of 6)

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DENT THAT IS PERMITTED

H

I

A TYPICAL SLAT LEADING EDGE SECTION IS SHOWN

A–A

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Allowable Damage Limits
Figure 105 (Sheet 4 of 6)

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TRAILING EDGE WEDGE (ALUMINUM HONEYCOMB SANDWICH)

J

REMOVAL OF DAMAGE TO THE ALUMINUM ARROWHEAD TIP

K

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Allowable Damage Limits
Figure 105 (Sheet 5 of 6)

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X = TOTAL TRIM ALLOWANCE
= A MAXIMUM OF 0.1 INCH (2.5 mm)

Y = LOWER SURFACE TRIM ALLOWANCE
= A MAXIMUM OF 0.5 INCH (12.7 mm)

Z = TIP THICKNESS
= A MINIMUM OF 0.02 INCH (0.51 mm)

REMOVAL OF DAMAGE TO THE ALUMINUM ARROWHEAD TIP

L

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Allowable Damage Limits
Figure 105 (Sheet 6 of 6)

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NOTE: THE DIMENSION OF A DAMAGE AREA IS EITHER THE DIMENSION OF THE VISUAL DAMAGE OR THE DIMENSION OF THE DELAMINATION. USE THE DIMENSION OF THE LARGER DAMAGE.

D IS THE LARGER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

d IS THE SMALLER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

A IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS WHICH IS EQUIVALENT TO 1.5 X (D+2).

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Minimum Damage Spacing in Solid Laminate Parts
Figure 106

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REPAIR 1 - LEADING EDGE SLAT SKIN - FLUSH REPAIR BETWEEN THE RIBS FORWARD OF THE NOSE BEAM

1. Applicability

A. Repair 1 is applicable to damage that:

(1) Is on slat numbers 1 through 4 on the left wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 1

(2) Is on slat numbers 5 through 8 on the right wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 1

(3) Is on the skin forward of the nose beam and between the ribs

(4) Is not at a nose beam.

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Wing Leading Edge Slat Skins Location
Figure 201

2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS

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(Continued)

Reference	Title
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
737 NDT Part 6, 51-00-00	Structures - General
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

CAUTION: BE CAREFUL THAT YOU DO NOT CAUSE DAMAGE TO THE THERMAL ANTI-ICE DUCTS OF THE LEADING EDGE SLATS NUMBER 1, 2, 3, 5, 6, AND 7 WHEN YOU DO THIS REPAIR.

CAUTION: MAKE SURE THAT YOU DO NOT CAUSE DAMAGE TO THE STRUCTURE ADJACENT TO THE LEADING EDGE SLATS.

A. Cut and remove the damaged part of the nose skin as shown in Layout of the Repair Parts, Figure 202/REPAIR 1. Refer to 51-10-02 for the removal of damage.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to the nose beam.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

B. Put the skin around the cut back to the initial contour.

C. Do a high frequency eddy current (HFEC) inspection of the damaged area to make sure that all the damage is removed. Refer to 737 NDT Part 6, 51-00-00, Figure 4 for the HFEC inspection procedures.

(1) The penetrant inspection is permitted as an alternative to the HFEC inspection. Refer to SOPM 20-20-02.

D. Repeat steps 4.A through 4.C until all the damage is removed. If all the damage is removed, continue with step 4.E.

E. Make the repair parts and make the shape the same as the initial contour of the skin. Refer to Table 201/REPAIR 1 and Figure 202.

F. Put the repair parts in place as shown in Layout of the Repair Parts, Figure 202/REPAIR 1.

G. Drill and countersink the necessary fastener holes.

NOTE: Do not countersink more than 75 percent of the skin thickness to prevent knife-edging of the skin.

H. Remove the repair parts.

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Table 201:

REPAIR MATERIAL			
PART NUMBER	PART	QUANTITY	MATERIAL
[1]	Repair Plate	1	Use 7075-0 clad sheet that is 0.063 inch (1.600 mm) thick. Heat treat to T62 after forming
[2]	Doubler	1	Use 7075-0 clad sheet that is 0.071 inch (1.803 mm) thick. Heat treat to T62 after forming
[3]	Doubler	1	Use 7075-0 clad sheet that is 0.071 inch (1.803 mm) thick. Heat treat to T62 after forming
[4]	Strap	2	Use 7075-T62 clad sheet that is 0.071 inch (1.803 mm) thick

I. Remove all the nicks, scratches, burrs, gouges, and sharp edges from the repair parts and the bare surfaces of the skin.

J. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial skin. Refer to 51-20-01.

K. Apply one layer of BMS 10-11, Type I primer (Refer to SOPM 20-41-02) to:

(1) The internal and mating surfaces of the repair parts

(2) The bare surfaces of the internal surface of the skin.

L. Install the parts [2] and [3] doublers and the part [4] straps.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin, the doublers, and the straps. Refer to 51-20-05.

(2) Install the doublers and the straps through the rectangular hole.

(3) Install the BACR15CE5 fasteners without sealant. Install the NAS1739 fasteners with BMS 5-95 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

M. Install the part [1] repair plate.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin, the parts [2] and [3] doublers, and the repair plate. Refer to 51-20-05.

(2) Install the repair plate.

(3) Install the NAS1739 fasteners with BMS 5-95 sealant. Refer to Layout of the Repair Parts, Figure 202/REPAIR 1.

N. Fill the space between the part [1] repair plate and the initial skin with BMS 5-95 sealant, if necessary. Refer to 51-20-05.

O. Apply the finish to the repair area as given in AMM 51-21-00/701.

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G47226 S0006594673_V1

Layout of the Repair Parts
Figure 202 (Sheet 1 of 3)

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H81586 S0006594674_V1

Layout of the Repair Parts
Figure 202 (Sheet 2 of 3)

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A-A

FASTENER SYMBOLS

┼ REPAIR FASTENER LOCATION. INSTALL A BACB30NW5K BOLT WITH A BACC30M COLLAR. AS AN ALTERNATIVE, INSTALL A BACR15CE5D RIVET, A NAS1739MW5, OR A NAS1739CW5 BLIND RIVET.

⊕ REPAIR FASTENER LOCATION. INSTALL A NAS1739MW5 OR NAS1739CW5 BLIND RIVET.

H82020 S0006594675_V2

Layout of the Repair Parts
Figure 202 (Sheet 3 of 3)

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REPAIR 2 - LEADING EDGE SLAT SKIN - FLUSH REPAIR AFT OF THE NOSE BEAM

1. Applicability

A. Repair 2 is applicable to damage that:

(1) Is on slat numbers 1 through 4 on the left wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 2

(2) Is on slat numbers 5 through 8 on the right wing as shown in Wing Leading Edge Slat Skins Location, Figure 201/REPAIR 2

(3) Is on the skin aft of the nose beam and between the ribs

(4) Is forward of the trailing edge wedge assembly

(5) Is not at a nose beam, a cove chord, a flex skin, a main track rib, an up stop rib, an auxiliary arm rib, or an actuator rib.

G48253 S0006594677_V1

Wing Leading Edge Slat Skins Location
Figure 201

2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE

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(Continued)

Reference	Title
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-00	INTERIOR AND EXTERIOR FINISHES
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
737 NDT Part 6, 51-00-00	Structures - General
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

CAUTION: MAKE SURE THAT YOU DO NOT CAUSE DAMAGE TO THE STRUCTURE ADJACENT TO THE LEADING EDGE SLATS.

A. Cut and remove the damaged part of the nose skin as shown in Layout of the Repair Parts, Figure 202/REPAIR 2. Refer to 51-10-02 for the removal of damage.

(1) Make the cut in the shape of a rectangle with the sides parallel or perpendicular to the nose beam.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

B. Put the skin around the cut back to the initial contour.

C. Do a high frequency eddy current (HFEC) inspection of the damaged area to make sure that all the damage is removed. Refer to 737 NDT Part 6, 51-00-00, Figure 4 for the HFEC inspection procedures.

(1) The penetrant inspection is permitted as an alternative to the HFEC inspection. Refer to SOPM 20-20-02.

D. Repeat steps 4.A through 4.C until all the damage is removed. If all the damage is removed, continue with step 4.E.

E. Make the repair parts and make the shape the same as the initial contour of the skin. Refer to Table 201/REPAIR 2 and Figure 202.

Table 201:

REPAIR MATERIAL			
PART NUMBER	PART	QUANTITY	MATERIAL
[1]	Repair Plate	1	Use 7075-T62 clad sheet that is 0.063 inch (1.600 mm) thick
[2]	Doubler	1	Use 7075-T62 clad sheet that is 0.071 inch (1.803 mm) thick

F. Put the repair parts in place as shown in Layout of the Repair Parts, Figure 202/REPAIR 2.

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G. Drill and countersink the necessary fastener holes.

NOTE: Do not countersink more than 70 percent of the skin thickness to prevent a knife-edge condition in the skin.

H. Remove the repair parts.

I. Remove all the nicks, scratches, burrs, gouges, and sharp edges from the repair parts and the bare surfaces of the skin.

J. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial skin. Refer to 51-20-01.

K. Apply one layer of BMS 10-11, Type I primer (Refer to SOPM 20-41-02) to:

(1) The internal and mating surfaces of the repair parts

(2) The bare surfaces of the internal surface of the skin.

L. Install the part [2] doubler.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin and the doubler. Refer to 51-20-05.

(2) Install the doubler.

(3) Install the repair fasteners without sealant. Refer to Layout of the Repair Parts, Figure 202/ REPAIR 2.

M. Install the part [1] repair plate.

(1) Apply BMS 5-95 sealant to the mating surfaces of the initial skin, the repair plate, and the part [2] doubler. Refer to 51-20-05.

(2) Install the repair plate.

(3) Install the repair fasteners without sealant. Refer to Layout of the Repair Parts, Figure 202/ REPAIR 2.

N. Fill the space between the part [1] repair plate and the initial skin with BMS 5-95 sealant, if necessary. Refer to 51-20-05.

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FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX DRIVE BOLT AND A BACC30M COLLAR.

NOTES

1 YOU MAY INSTALL A MINIMUM OF ONE ROW OR A MAXIMUM NUMBER OF ROWS, AS REQUIRED, FOR THE REPAIR PLATE.

G48442 S0006594679_V1

Layout of the Repair Parts
Figure 202 (Sheet 1 of 2)

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G49138 S0006594680_V1

Layout of the Repair Parts
Figure 202 (Sheet 2 of 2)

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REPAIR 3 - LEADING EDGE SLAT COVE SKIN

NOTE: THERE ARE NO REPAIRS FOR THE COVE SKINS OF THE WING LEADING EDGE SLATS. IF THE DAMAGE TO A COVE SKIN IS MORE THAN THE LIMITS GIVEN IN SRM 57-42-01, ALLOWABLE DAMAGE 1, REPLACE THE DAMAGED COVE SKIN.

LEFT WING IS SHOWN, RIGHT WING IS OPPOSITE

G47423 S0006594682_V2

Wing Leading Edge Slat Skins Location
Figure 201

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REPAIR 4 - OUTBOARD LEADING EDGE SLAT SKIN TRAILING EDGE WEDGE

L89363 S0006594684_V2

Wing Outboard Leading Edge Slat - Trailing Edge Wedge Repairs
Figure 201 (Sheet 1 of 2)

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A TYPICAL LEADING EDGE SLAT SECTION IS SHOWN

A

NOTE: REFER TO SRM 51-70-10 FOR TYPICAL METAL REPAIRS FOR THE OUTBOARD LEADING EDGE SLAT TRAILING EDGE WEDGE. MAKE SURE THERE IS SUFFICIENT CLEARANCE FROM ADJACENT STRUCTURE TO INSTALL THE REPAIR PARTS.

1 ONLY ADHESIVES EA-9628 (DEXTER CORPORATION) OR AF-163 (3M AEROSPACE MATERIALS) ARE PERMITTED ON THE SLAT TRAILING EDGE WEDGES EXCEPT AS NOTED.

L89366 S0006594685_V2

Wing Outboard Leading Edge Slat - Trailing Edge Wedge Repairs
Figure 201 (Sheet 2 of 2)

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REPAIR 5 - LEADING EDGE SLAT SKIN SPLICE REPAIR - SLATS 1 AND 8

1. Applicability

A. Repair 5 is applicable to damage to the wing leading edge slat skins 1 & 8 as shown in Figure 201.

LEFT WING IS SHOWN, RIGHT WING IS OPPOSITE

2056996 S0000421306_V1

Wing Leading Edge Slat Skin Locations - Slats 1 & 8
Figure 201

2. General

A. Repair 5 is a permanent repair. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01, GENERAL.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions

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(Continued)

Reference	Title
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge slat skin. Refer to 51-10-02, GENERAL for the removal of damage procedures.

(1) Make sure there is a minimum corner radius of 0.50 inch (12.70 mm) at all locations.

B. Make the repair parts as shown in Figure 202. Refer to Table 201 for the repair parts. Make sure that you form the repair parts to the contour of the initial skin.

NOTE: You will need to remove the initial cove skin shown in Figure 202 to permit installation of this repair.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Replacement Skin	1	Use clad 7075-T62 or -T6 as given in QQ-A-250/13 or as given in AMS 4049. Use the same thickness as the initial skin.
[2]	Tapered Filler	As required*[1]	Use clad or bare 7075-T6. Use the same thickness as the initial skin. Make sure there is a 50:1 taper.
[3]	Internal Splice Doubler	2	Use 17-7PH CRES as given in AMS 5528 that is 0.04 inches thick. Heat treat to 180-200 KSI.

*[1] Quantity will depend on use of the optional trailing edge configuration shown in Figure 202 (Sheet 1).

C. Assemble the repair parts and drill the fastener holes as shown in Figure 202.

D. Disassemble the repair parts.

E. Remove all nicks, scratches, burrs, and sharp edges from the initial skin and the repair parts.

F. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial skin. Refer to 51-20-01, GENERAL.

G. Apply one layer of BMS 10-79, Type III primer to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-44-04.

H. Apply a cadmium plating to the Splice Doubler in accordance with SOPM 20-42-05.

I. Install the repair parts and fasteners wet with BMS 5-95 sealant and make sure to fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Install the initial cove skin to match the initial configuration

J. Apply a layer of BMS 10-60 Type II enamel to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-41-02.

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SLAT NUMBER 1 IS SHOWN, SLAT NUMBER 8 IS OPPOSITE

TYPICAL SLAT CROSS SECTION
A–A

NOTES

1 OPTIONAL CONFIGURATION OF INTERNAL DOUBLER NOSE SKIN AND TRAILING EDGE WEDGE SPAR

2045489 S0000413877_V1

Leading Edge Slat Skin Splice Repair
Figure 202 (Sheet 1 of 2)

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(SKIN SHOWN TO APPEAR FLAT)
B-B

NOTES

1 USE TWO SPLICES IF THE DAMAGE IS NOT NEAR THE END OF THE SLAT. USE ONE SPLICE IF THE DAMAGE IS NEAR THE END OF THE SLAT.

2 TAPERED FILLER ONLY NECESSARY FOR OPTIONAL CONFIGURATION IN SHEET 1.

FASTENER SYMBOLS

- INITIAL FASTENER LOCATION. INSTALL INITIAL FASTENER TYPE.
- INITIAL FASTENER LOCATION. INSTALL A BACB30NZ6KY HEX-DRIVE BOLT WITH A BACC30M COLLAR
- REPAIR FASTENER LOCATION. INSTALL A PLT1058L6 BLIND RIVET AS AN ALTERNATIVE, INSTALL A PLT5470L6 OR NAS1739MW BLIND RIVET.
- REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.
- REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.

2045494 S0000413884_V1

Leading Edge Slat Skin Splice Repair
Figure 202 (Sheet 2 of 2)

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REPAIR 6 - LEADING EDGE SLAT TRAILING EDGE WEDGE LOWER SKIN WEAR REPAIR

1. Applicability

A. Repair 6 is applicable to wear damage to the wing leading edge slat trailing edge wedge with the fiberglass tip filler, lower skin aft edge that:

(1) Is on slat numbers 1 through 4 on the left wing as shown in Figure 201/REPAIR 6.

(2) Is on slat numbers 5 through 8 on the right wing as shown in Figure 201/REPAIR 6.

(3) Has wear depth that is more than 0.015 in. (0.38 mm) into the lower skin and less than or equal to 0.019 in. (0.48 mm) into the fiberglass filler.

NOTE: Damage is not permitted in the upper skin of the wedge. If you have wear damage in the upper skin of the wedge, contact The Boeing Company for a repair.

(4) Is within the 0.625 in. (15.88 mm) width area as shown in Figure 202 (Sheet 2).

B. Repair 6 is not applicable to wear damage to the wing leading edge slat trailing edge wedge with the aluminum tip filler.

1551415 S0000284947_V1

Wing Leading Edge Slat Locations
Figure 201

2. References

Reference	Title
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
AMM 27-81-21-000-801	Leading Edge Slat Removal (P/B 401)
AMM 51-21-00-100-801	Airplane Surface Preparation for Application of Finish (P/B 701)
SOPM 20-44-04	Application of Urethane Compatible Primer

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3. **Repair Instructions**

A. If necessary, remove the damaged leading edge slat from the airplane to gain access to the damaged area on the trailing edge wedge lower skin. Refer to AMM TASK 27-81-21-000-801.

B. Trim out the wear damage from the trailing edge wedge lower skin. Use a maximum trim width of 0.625 in. (15.88 mm), as measured from the aft edge of the trailing edge wedge.

(1) The trimmed area should be in the shape of a rectangle with the sides parallel to the lower skin aft edge as shown in Figure 202 (Sheet 1), Detail A.

(2) Keep the corner radii to 0.5 in. (12.70 mm).

C. Make the repair part as shown in Figure 202 (Sheet 1). Refer to Table 201/REPAIR 6. The thickness of the part [1] filler is equal to the total wear depth of the skin and the initial fiberglass filler. Make sure that the part [1] filler fits into the trimmed area on the skin.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Filler	1	Use 7075–T6 bare sheet per AMS 4045, AMS-QQ-A-250/12 or equivalent.

D. Remove all the nicks, scratches, burrs and sharp edges from the repair part and the lower skin.

E. Apply a chemical conversion coating to the repair part and to the bare surfaces of the lower skin. Refer to 51-20-01, GENERAL.

F. Bond the part [1] filler to the trailing edge wedge lower skin with BMS 5–95 sealant. Refer to 51-20-05, GENERAL. Make sure the surface of the the part [1] filler is flush with the surface of the initial skin.

G. Apply one layer of BMS 10–79, Type III primer to the repair area. Refer to SOPM 20-44-04.

H. Apply the finish to the repair area as given in AMM TASK 51-21-00-100-801.

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1551516 S0000285041_V2

Leading Edge Slat Trailing Edge Wedge Lower Skin Repair
Figure 202 (Sheet 1 of 2)

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TYPICAL REPAIR

A

1551444 S0000285040_V2

Leading Edge Slat Trailing Edge Wedge Lower Skin Repair
Figure 202 (Sheet 2 of 2)

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REPAIR 7 - LEADING EDGE SLAT SKIN SPLICE REPAIR - SLATS 2 THRU 7

1. Applicability

A. Repair 7 is applicable to damage to the wing leading edge slat skins 2 thru 7 as shown in Figure 201 (Sheet 1).

2059986 S0000423355_V1

Wing Leading Edge Slat Skins Location
Figure 201

2. General

A. Repair 7 is a permanent repair. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01, GENERAL.

3. References

Reference	Title

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(Continued)

51-00-06, GENERAL	Structural Repair Definitions
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge slat skin. Refer to 51-10-02, GENERAL for the removal of damage procedures.

(1) Make sure there is a minimum corner radius of 0.50 inch (12.70 mm) at all locations.

B. Make the repair parts as shown in Figure 202/REPAIR 7. Refer to Table 201/REPAIR 7 for the repair parts. Make sure that you form the repair parts to the contour of the initial skin.

NOTE: You will need to remove the initial cove skin shown in Figure 202/REPAIR 7 to permit installation of this repair.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Replacement Skin	1	Use clad 7075-T62 or -T6 as given in QQ-A-250/13 or as given in AMS 4049. Use the same thickness as the initial skin.
[2]	Tapered Filler	As required*[1]	Use clad or bare 7075-T6. Use the same thickness as the initial skin. Make sure there is a 50:1 taper.
[3]	Splice Doubler	2	Use 17-7PH CRES as given in AMS 5528 that is 0.04 inch thick. Heat treat to 180-200 KSI.

*[1] Quantity will depend on use of the optional trailing edge configuration shown in Figure 202 (Sheet 1)

C. Assemble the repair parts and drill the fastener holes as shown in Figure 202.

D. Disassemble the repair parts.

E. When cutting the skin, locate the skin away from the thermal anti-ice (TAI) vent holes.

NOTE: If the part [1] replacement skin includes an initial TAI vent hole location, then make sure to back drill the TAI vent holes through the part [1] replacement skin to match the intitial configuration.

F. It is permitted to shave the upper nose beam spacers to install the part [3] splice doubler. It is permitted to remove a maximum of 0.04 in. (1.02 mm) from the upper nose beam spacers.

G. Remove all nicks, scratches, burrs, and sharp edges from the initial skin and the repair parts.

H. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial skin. Refer to 51-20-01, GENERAL.

I. Apply one layer of BMS 10-79, Type III primer to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-44-04.

J. Apply a cadmium plating to the Splice Doubler in accordance with SOPM 20-42-05.

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K. Install the repair parts and fasteners wet with BMS 5-95 sealant and make sure to fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Install the initial cove skin to match the initial configuration

NOTE: Make sure there is no BMS 5-95 sealant to any of the Thermal Anti-Ice (TAI) air passages.

L. Apply a layer of BMS 10-60 Type II enamel to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-41-02.

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SLAT NUMBER 2 IS SHOWN, SLAT NUMBERS 3, AND 4 ARE SIMILAR,
SLAT NUMBERS 5, 6, AND 7 ARE OPPOSITE

TYPICAL SLAT CROSS SECTION
A–A

NOTES

1 OPTIONAL CONFIGURATION OF INTERNAL DOUBLER NOSE SKIN AND TRAILING EDGE WEDGE SPAR.

2 IT IS PERMITTED TO SHAVE THE NOSE BEAM SPACER TO INSTALL THE INTERNAL DOUBLER. A MAXIMUM OF 0.04 INCH (1.02 mm) MAY BE REMOVED FROM THE NOSE BEAM SPACER.

2045506 S0000413938_V1

Leading Edge Slat Skin Splice Repair
Figure 202 (Sheet 1 of 2)

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NOTES

1 USE TWO SPLICES IF THE DAMAGE IS NOT NEAR THE END OF THE SLAT. USE ONE SPLICE IF THE DAMAGE IS NEAR THE END OF THE SLAT.

2 TAPERED FILLER ONLY NECESSARY FOR OPTIONAL CONFIGURATION IN SHEET 1.

3 INITIAL NOSE BEAM SPACER BACR15CF286-5.

FASTENER SYMBOLS

┼ INITIAL FASTENER LOCATION. INSTALL INITIAL FASTENER TYPE.

✦ INITIAL FASTENER LOCATION. INSTALL A BACB30NZ6KY HEX-DRIVE BOLT WITH A BACC30M COLLAR

⌖ REPAIR FASTENER LOCATION. INSTALL A PLT1058L6 BLIND RIVET AS AN ALTERNATIVE, INSTALL A PLT5470L6 OR NAS1739MW BLIND RIVET.

⌖ REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.

⌖ REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.

2059987 S0000424047_V2

Leading Edge Slat Skin Splice Repair
Figure 202 (Sheet 2 of 2)

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REPAIR 8 - LEADING EDGE SLAT NOSE SKIN EXTERNAL DOUBLER REPAIR AT THE ACTUATOR RIB

1. Applicability

A. This repair is applicable to crack on the lower surface of the wing leading edge slat nose skin at the actuator rib at Slat Stations 85.19, 202.45, 322.71, and 451.97. Refer to Figure 201/REPAIR 8.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

TYPICAL LEADING EDGE SLAT NOSE SKIN

A

2047539 S0000415075_V1

Wing Leading Edge Slat Nose Skins Location
Figure 201

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2. General

A. REPAIR 8 is a permanent repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for the definitions of the different categories of repairs.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-08	COUNTERSINKING
AMM 51-21-00	INTERIOR AND EXTERIOR FINISHES
SOPM 20-10-06	Repair of Conductive Coatings
SOPM 20-44-03	Application of Polyurethane Rain Erosion Resistant Coating
SOPM 20-44-04	Application of Urethane Compatible Primer
SWPM 20-20-00	Electrical Bonding Processes

4. Repair Instructions

A. Cut and remove the damaged part of the slat nose skin.

(1) Make the corner radii of the skin cutout a minimum of 0.5 in. (12.7 mm).

(2) Make the surface finish on the edge of the cutout a minimum of 63 microinches Ra.

(3) Make sure the cutline for the cutout is midway between fasteners.

B. Do the inspection of the damaged area to make sure there are no more damages. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

C. Make the repair parts as shown in Figure 202/REPAIR 8. Refer to Table 201/REPAIR 8 for the repair material.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Doubler	1	Use clad 7075–T62 or 7075–T6 sheet as given in AMS QQ-A-250/13, 0.071 in. (1.803 mm) thick
[2]	Filler	1	Use clad 7075–T62 or 7075–T6 sheet as given in AMS QQ-A-250/13, same thickness as the skin of 0.063 in. (1.600 mm)

D. Assemble the repair parts as shown in Figure 202/REPAIR 8.

E. Drill the fastener holes. Keep a minimum of 2D edge margin and 4D to 6D fastener spacing.

F. Disassemble the repair parts.

G. Remove the nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the nose skin.

H. Apply a chemical conversion coating to the repair parts and the bare surfaces of the nose skin. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

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I. Apply one layer of BMS 10-79, Type III primer to the repair parts and to the bare surfaces of the initial skin. Refer to SOPM 20-44-04.

J. Install the repair parts.

K. Install the fasteners. Refer to FASTENER INSTALLATION AND REMOVAL, 51-40-02.

NOTE: Refer to the special instructions given in flagnote 2 for the installation of one non-designate bond, double flush rivet. Refer Figure 202/REPAIR 8 for the location of this rivet.

L. Apply a fillet seal. Refer to REPAIR SEALING, 51-20-05.

M. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to AERODYNAMIC SMOOTHNESS, 51-10-01.

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(VIEW IN THE UP DIRECTION)

SLAT 1 IS SHOWN, SLAT 8 IS OPPOSITE

2092884 S0000441675_V1

Wing Leading Edge Slat Lower Surface Skin Repair at the Actuator Rib
Figure 202 (Sheet 1 of 6)

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(VIEW IN THE UP DIRECTION)

SLAT 2 IS SHOWN, SLAT 7 IS OPPOSITE

2094712 S0000441676_V1

Wing Leading Edge Slat Lower Surface Skin Repair at the Actuator Rib
Figure 202 (Sheet 2 of 6)

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(VIEW IN THE UP DIRECTION)

SLAT 3 IS SHOWN, SLAT 6 IS OPPOSITE

2094700 S0000441677_V1

Wing Leading Edge Slat Lower Surface Skin Repair at the Actuator Rib
Figure 202 (Sheet 3 of 6)

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(VIEW IN THE UP DIRECTION)

SLAT 4 IS SHOWN, SLAT 5 IS OPPOSITE

2047544 S0000415927_V1

Wing Leading Edge Slat Lower Surface Skin Repair at the Actuator Rib
Figure 202 (Sheet 4 of 6)

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(VIEW ROTATED 90° CLOCKWISE)
B-B

2093052 S0000441678_V1

Wing Leading Edge Slat Lower Surface Skin Repair at the Actuator Rib
Figure 202 (Sheet 5 of 6)

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NOTES

- ALL DIMENSIONS ARE IN INCHES (mm).
- D = DIAMETER OF THE FASTENER.
- INSTALL COUNTERSINK WASHERS IN THE INITIAL COUNTERSINK NOSE SKIN FASTENER HOLES WET WITH BMS 5-95 SEALANT. REFER TO SRM 51-40-08.
- INSTALL BACB30VU6K OR BACB30NW6K HEX DRIVE BOLTS COMMON TO THE FLEX SKIN (AFT ROW) WITH NAS1149D0363J WASHER AND BACN10YR3CD NUT.

1 MAKE SURE THE CUTLINE OF THE CUTOUT IS MIDWAY BETWEEN THE INITIAL FASTENERS.

2 THIS IS A NON-DESIGNATED BOND. INSTALL RIVET ON BMS 10-21, TYPE III (ANTI-STATIC COATING) COATED SURFACE AS GIVEN BELOW. REFER TO SWPM 20-20-00.

- PREPARE THE FLEX SKIN SURFACE USING SOLVENT CLEANING METHOD CM3.
- IF THERE IS NO CONDUCTIVE COATING, REAPPLY BMS 10-21, TYPE III (ANTI-STATIC COATING) TO CONTACT AREA ON THE FLEX SKIN BEFORE INSTALLATION AS GIVEN IN SOPM 20-10-06.
- CLEAN RIVET HEAD BY SOLVENT CLEANING METHOD CM3.
- MEASURE THE ELECTRICAL RESISTANCE FROM THE FLEX SKIN ANTI-STATIC COATING TO THE DOUBLE FLUSH RIVET. THE MAXIMUM RESISTANCE IS 300,000 OHMS.

FINISH THE FLEX SKIN WITH BMS 10-103, TYPE I PRIMER AS GIVEN IN SOPM 20-44-03 AND BMS 10-60, TYPE II, BAC707 COLOR, GLOSS ENAMEL AS GIVEN IN AMM 51-21-00.

3 DO NOT CHAMFER ON THE AFT EDGE OF THE PART 1 DOUBLER AS SHOWN.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A BACB30VU6K OR BACB30NW6K HEX DRIVE BOLT. IT IS PERMITTED TO INSTALL A 1/64 INCH OR 1/32 INCH DIAMETER OVERSIZE BACB30NW6K()X OR BACB30NW6K()Y HEX DRIVE BOLT, IF NECESSARY. INSTALL WET WITH BMS 5-95 SEALANT.

INITIAL FASTENER LOCATION. INSTALL A 1/64 INCH OR 1/32 INCH DIAMETER OVERSIZE BACB30NW6K()X OR BACB30NW6K()Y HEX DRIVE BOLT. INSTALL WET WITH BMS 5-95 SEALANT.

INITIAL FASTENER LOCATION. INSTALL A BACR15CE6D OR BACR15GF6D RIVET (UP TO 1/32 INCH DIAMETER OVERSIZE).

REPAIR FASTENER LOCATION. INSTALL A BACR15CE6D OR BACR15GF6D RIVET.

REPAIR FASTENER LOCATION. INSTALL A NAS1739MW6 OR NAS1739CW6 BLIND RIVET.

2047756 S0000416110_V1

Wing Leading Edge Slat Lower Surface Skin Repair at the Actuator Rib
Figure 202 (Sheet 6 of 6)

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IDENTIFICATION 1 - OUTBOARD WING LEADING EDGE SLAT STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H20422 S0006594689_V1

Wing Outboard Leading Edge Slat Structure Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A1001	Wing Left - Product Collector
001A1002	Wing Right - Product Collector
114A0011	Left Wing, Slat Functional Product Collector
114A0012	Right Wing, Slat Functional Product Collector
114A4002	Slat Installation - Outboard Leading Edge, Wing
114A5010	Slat Assembly - Number 1 and 8
114A5020	Slat Assembly - Number 2 and 7

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
114A5030	Slat Assembly - Number 3 and 6
114A5040	Slat Assembly - Number 4 and 5
114A8220	Nose Beam Assembly - Slat 2 and 7
114A8230	Nose Beam Assembly - Slat 3 and 6
114A8240	Nose Beam Assembly - Slat 4 and 5

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SLAT NUMBER 1 IS SHOWN,
SLAT NUMBER 8 IS OPPOSITE

A–A

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

H20879 S0006594691_V2

Wing Outboard Leading Edge Slat Structure Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Wedge Assembly			LINE NUMBER 1 TO 19
	Upper Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Upper Doubler	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core		Aluminum honeycomb as given in BMS 4–4, Type 3–15 and Type 3–10, Class NPA, Grade I	
[1]	Wedge Assembly			LINE NUMBER 20 AND ON
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I	
[2]	Main Track Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[3]	Auxiliary Arm Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part (Optional: 7050–T7451 plate as given in BMS 7-323, Type 1)	
[4]	Actuator Rib		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[5]	Up Stop Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[6]	End Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[7]	Main Track (2)		4340M forging as given in BMS 7-26. Normalized and subcritical annealed to RC 33 maximum. Heat treat to 275- 300 KSI as given in BAC 5617	
[8]	Nose Beam	0.071 (1.8)	7075-T62 sheet as given in QQ-A-250/12	
[9]	Cove Chord	0.080 (2.03)	7075-T62 sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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SLAT NUMBER 2 IS SHOWN,
SLAT NUMBER 7 IS OPPOSITE,
SLATS NUMBER 3 THRU 6 ARE EQUIVALENT

A–A

NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

H20990 S0006594693_V2

Wing Outboard Leading Edge Slat Structure Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Wedge Assembly			LINE NUMBER 1 TO 19
	Upper Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Upper Doubler	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core		Aluminum honeycomb as given in BMS 4–4, Type 3–15 and Type 3–10, Class NPA, Grade I	
[1]	Wedge Assembly			LINE NUMBER 20 AND ON
	Upper Skin	0.032 (0.81)	7075-T6 sheet as given in QQ-A-250/12. Refer to Boeing production drawing for the chem-mill thicknesses	
	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
	Tip Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class 1, Style 1581 or 120	
	Core		Aluminum honeycomb as given in BMS 4-4, Type 3-15, Class NPA, Grade I	
[2]	Main Track Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[3]	Auxiliary Arm Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part (Optional: 7050–T7451 plate as given in BMS 7-323, Type 1)	
[4]	Actuator Rib		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[5]	Up Stop Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[6]	End Rib (2)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	
[7]	Main Track (2)		4340M forging as given in BMS 7-26. Normalized and subcritical annealed to RC 33 maximum. Heat treat to 275- 300 KSI as given in BAC 5617	
[8]	Nose Beam Assembly			
	Nose Beam	0.071 (1.8)	7075-T62 sheet as given in QQ-A-250/12	
	Doubler (2)	0.090 (2.29)	7075-T62 sheet as given in QQ-A-250/12	
[9]	Cove Chord	0.080 (2.03)	7075-T62 sheet as given in QQ-A-250/12	
[10]	Cove Chord Doubler	0.080 (2.03)	7075-T6 sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING LEADING EDGE SLAT STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing leading edge slat structure shown in Wing Outboard Leading Edge Slat Structure Locations, Figure 101/ALLOWABLE DAMAGE 1.

G38924 S0006594697_V1

Wing Outboard Leading Edge Slat Structure Locations
Figure 101

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SLAT NUMBER 2 IS SHOWN, SLATS NUMBER 1, 3, AND 4 ARE SIMILAR
SLAT NUMBER 8 IS OPPOSITE

G38979 S0006594698_V2

Outboard Leading Edge Slat Structure
Figure 102 (Sheet 1 of 3)

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A–A

B–B

G39059 S0006594699_V2

Outboard Leading Edge Slat Structure
Figure 102 (Sheet 2 of 3)

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C–C

D–D

E–E

G39140 S0006594700_V2

Outboard Leading Edge Slat Structure
Figure 102 (Sheet 3 of 3)

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2. **General**

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Do the steps that follow if you have damage to the aluminum, steel, or Corrosion Resistant Steel (CRES) parts:

(1) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures for the aluminum parts.

(b) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do as follows:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(a) Flap peen or shot peen all parts except the nose beams, cove chords, end ribs, end stop fittings, and slat track wear surfaces.

NOTE: You must prevent damage to the protective layer around the reworked area before you shot-peen.

1) Refer to 51-20-06 for the shot peen intensity and shot number.

2) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(b) Apply the finishes for the ribs.

1) Apply a chemical conversion coating to the bare surfaces. Refer to SRM 51-21-01 .

2) Apply a layer of BMS 10-79, Type III primer to the reworked surfaces. Refer to SOPM 20-44-04.

(c) Apply the finishes for the nose beams and cove chords.

1) Apply a chemical conversion coating to the bare surfaces. Refer to 51-20-01.

2) Apply a layer of BMS 10-11, Type I primer to the reworked surfaces. Refer to SOPM 20-41-02.

(d) Apply the finishes for main tracks except to the bores or roller surfaces.

1) Apply titanium-cadmium plating to the reworked surfaces. Refer to SOPM 20-42-01.

2) Apply Tungsten Carbide with Super Detonation Gun to the roller surfaces. Refer to SOPM 20-41-01.

3) Apply a layer of BMS 10-79, Type III primer to the reworked surfaces except to the bore surfaces. Refer to SOPM 20-44-04.

4) Apply a layer of BMS 10-60, Type II enamel to the reworked surfaces except to the bore surfaces. Refer to AMM PAGEBLOCK 51-21-99/701.

(e) It is optional to apply a finish to the reworked areas of the upper and lower main track roller surfaces. You can apply a flame spray coating of BMS 10-67 Type I, tungsten carbide powder, to improve the wear resistance of the track. Refer to:

1) CMM 57–43–01, Repair 7–2, for Slat Track Numbers 1 and 8.

2) CMM 57–43–02, Repair 7–2, for Slat Track Numbers 2 and 7.

3) CMM 57–43–03, Repair 7–2, for Slat Track Numbers 3 and 6.

4) CMM 57–43–04, Repair 7–2, for Slat Track Numbers 4 and 5.

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(f) Apply the finishes for the fittings.

1) Apply cadmium plating to the reworked surfaces. Refer to SOPM 20-42-05.

2) Apply a layer of BMS 10-79, Type III primer to the reworked surfaces. Refer to SOPM 20-44-04.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-01	Decoding Table For Boeing Finish Codes
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-01	Low Hydrogen Embrittlement Cadmium Plating
SOPM 20-42-05	Bright Cadmium Plating
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage 1 - Wing Leading Edge Slat Structure

A. Ribs (Machined Aluminum)

NOTE: No damage is permitted in the surfaces of the rib lug bore. You are permitted to drill the bore:

- To a maximum oversize diameter of 0.06 inch more than the initial bore design limits, and
- If there is no damage on the edge of the lug.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , F , and G .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Cove Chords and Nose Beams (Formed Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and H .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , G , and H.

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(3) Dents are not permitted.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet

a) Install the rivet without sealant.

C. Main Tracks (Forged Steel)

NOTE: No damage is permitted in the surface of the lug bore. You are permitted to drill the bore:

- To a maximum oversize diameter of 0.06 inch more than the initial bore design limits as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail I , and
- If there is no damage on the edge of the lug.

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion are not permitted.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

(5) Abrasion:

(a) Remove the damage as shown in Figure 103 (Sheet 8) Detail K.

NOTE: Removal of damage is permitted on only the upper and lower roller surfaces.

D. Fittings (CRES)

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , and J .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

E. Webs (Forged Steel)

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion are not permitted.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

(5) Abrasion:

(a) Remove the damage as shown in Figure 103 (Sheet 8) Detail K.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

G39141 S0006594701_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 9)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

G39144 S0006594702_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 9)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

F

NOTES

1 THESE ALLOWABLE DAMAGE LIMITS ARE APPLICABLE ONLY IF THE BORE OF THE LUG HAS NOT BEEN OVERSIZED. IF THE BORE IS OVERSIZED, THEN ONLY SURFACE DAMAGE IS PERMITTED AS SHOWN.

G39145 S0006594703_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 9)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

G

C–C

G40756 S0006594704_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 9)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

H

G40758 S0006594705_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 9)

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REMOVAL OF SURFACE AND EDGE DAMAGE

I

G39153 S0006594706_V2

Allowable Damage Limits
Figure 103 (Sheet 6 of 9)

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REMOVAL OF DAMAGED MATERIAL ON A LUG WITHOUT BUSHINGS

J

G43255 S0006594707_V1

Allowable Damage Limits
Figure 103 (Sheet 7 of 9)

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REMOVAL OF DAMAGED MATERIAL ON ROLLER SURFACE
(UPPER ROLLER SURFACE IS SHOWN,
LOWER ROLLER SURFACE IS OPPOSITE)

K

X = DEPTH OF BASE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.02 INCH (0.51 mm)

F–F

2166585 S0000476735_V1

Allowable Damage Limits
Figure 103 (Sheet 8 of 9)

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G–G

X = MINIMUM THICKNESS THAT REMAINS

= A MINIMUM OF 0.165 INCH

H–H

2171366 S0000478174_V1

Allowable Damage Limits
Figure 103 (Sheet 9 of 9)

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REPAIR 1 - WING LEADING EDGE SLAT STRUCTURE AT A NOSE BEAM

1. Applicability

A. Repair 1 is applicable to damage:

(1) To the structure of the wing leading edge slat shown in Wing Leading Edge Slat Skin Locations, Figure 201/REPAIR 1.

(2) That is at the nose beam.

G40016 S0006594710_V1

Wing Leading Edge Slat Skin Locations
Figure 201

2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Blind rivets that are loose, damaged, or that have fallen out must be replaced.

C. Repair 1 can only be used if you install the blind rivets flush against the internal structure.

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge slat nose beam. Refer to 51-10-02 for the damage removal procedures.

(1) Make the cut in the shape of a rectangle.

(a) Refer to Wing Leading Edge Slat Structure Repair at a Nose Beam, Figure 202/REPAIR 1 for a typical cut out.

B. Make the repair parts as shown in Wing Leading Edge Slat Structure Repair at a Nose Beam, Figure 202/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair parts.

Table 201:

REPAIR MATERIALS			
ITEM	**DESCRIPTION**	**QUANTITY**	**MATERIAL**
[1]	Doubler	1	Use 7075-T6 clad sheet that is 0.080 inch thick
[2]	Angle Filler (For the upper flange only)	1	Use 7075-T6 clad sheet that is 0.071 inch thick
[3]	Angle Filler (For the lower flange only)	1	Use 7075-T6 clad sheet that is 0.071 inch thick
[4]	Spacer Filler (For the upper flange only)	6	Use 7075-T6 clad sheet that is 0.125 inch thick
[5]	Repair Angle (For the upper flange only)	1	Use 7075-T6 clad sheet that is 0.080 inch thick

C. Assemble the repair parts.

D. Drill the necessary fastener holes.

E. Disassemble the repair parts.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and to the bare surfaces of the initial structure.

G. Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

H. Apply one layer of BMS 10-11, Type I primer to:

(1) The surfaces of the initial structure

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(2) The mating surfaces of the repair parts.

I. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

J. Install the fasteners.

(1) Install the aluminum fastener without sealant.

(2) Install the fasteners that are not made of aluminum with BMS 5-95 sealant. Refer to 51-20-05.

K. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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A TYPICAL SECTION IS SHOWN

1 INSTALL THE FASTENERS ALONG THE SAME SPAN LOCATION AS THE INITIAL UPPER SKIN FASTENERS. DO NOT INSTALL THE FASTENERS INTO THE REPAIR ANGLE BEND RADIUS.

G40019 S0006594712_V1

Wing Leading Edge Slat Structure Repair at a Nose Beam
Figure 202 (Sheet 1 of 4)

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UPPER FLANGE DAMAGE

A-A

FASTENER SYMBOLS

- INITIAL FASTENER LOCATION. INSTALL A BACB30NW6K HEX-DRIVE BOLT WITH A BACC30M COLLAR.
- REPAIR FASTENER LOCATION. INSTALL A NAS1738E6 BLIND RIVET.
- REPAIR FASTENER LICATION. INSTALL A BACR15BB6D RIVET.
- REFERENCE FASTENER

G40135 S0006594713_V1

Wing Leading Edge Slat Structure Repair at a Nose Beam
Figure 202 (Sheet 2 of 4)

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A TYPICAL SECTION IS SHOWN

G74019 S0006594714_V1

Wing Leading Edge Slat Structure Repair at a Nose Beam
Figure 202 (Sheet 3 of 4)

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LOWER FLANGE DAMAGE

B–B

G74020 S0006594715_V1

Wing Leading Edge Slat Structure Repair at a Nose Beam
Figure 202 (Sheet 4 of 4)

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IDENTIFICATION 1 - INBOARD AND OUTBOARD WING LEADING EDGE - KRUEGER FLAP STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G72833 S0006594720_V1

Inboard Wing Leading Edge Krueger Flap Structure Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
114A0010	Section 14 Final Assembly
114A1001	Krueger Flap Installation - Inboard Wing Leading Edge
114A1110	Krueger Flap and Bullnose Assembly - Number 1 and Number 4
114A1220	Krueger Flap and Bullnose Assembly - Number 2 and Number 3
114A1221	Flap Assembly - Inboard Krueger Flap
114A1320	Bullnose Assembly - Middle, Krueger Flap Number 2

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
114A1321	Bullnose Assembly Inboard Krueger Flap Number 2
114A1401	Rod Assembly - Krueger Flap, Inboard Wing Leading Edge
114A1410	Link Assembly - Outboard Bullnose, Krueger Flap, L. E.
114A1411	Link Assembly Bullnose, Outboard and Middle
114A1413	Bell Crank Assembly - Bullnose
114A1420	Link Assembly - Middle Bullnose Krueger Flap L. E., Wing
114A1430	Link Assembly - Inboard Bullnose Krueger Flap L. E., Wing
114A1510	Krueger Seal Assembly - Inboard Fixed Leading Edge
114A4001	Slat Rigging Installation - Outboard Leading Edge Wing

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

NUMBER 1 KRUEGER FLAP IS SHOWN,
NUMBER 4 KRUEGER FLAP IS OPPOSITE

G72835 S0006594722_V1

Outboard Krueger Flap Structure Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Krueger Flaps - Number 1 and Number 4		A357-T6 aluminum casting, as given in MIL-A-21180	
[2]	Bullnose, Outboard Flap		A357-T6 aluminum casting, as given in MIL-A-21180	
[3]	Krueger - Seal		A357-T6 aluminum casting, as given in MIL-A-21180	

*[1] Note: T = Pre-manufactured thicknesses in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

NUMBER 2 KRUEGER FLAP IS SHOWN,
NUMBER 3 KRUEGER FLAP IS OPPOSITE

G72836 S0006594724_V1

Inboard Krueger Flap Structure Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Krueger Flaps - Number 2 and Number 3		A357-T6 aluminum casting, as given in MIL-A-21180	
[2]	Bullnose, Inboard Flap - Number 2		A357-T6 aluminum casting, as given in MIL-A-21180	
[3]	Bullnose, Middle Flap - Number 2		A357-T6 aluminum casting, as given in MIL-A-21180	

*[1] Note: T = Pre-manufactured thicknesses in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING LEADING EDGE KRUEGER FLAP STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing leading edge Krueger flap structure shown in Inboard Wing Leading Edge Krueger Flap Structure Locations, Figure 101/ALLOWABLE DAMAGE 1.

F99803 S0006594728_V1

Inboard Wing Leading Edge Krueger Flap Structure Locations
Figure 101

2. General

A. Do the steps that follow if you have damage to the aluminum parts:

(1) Do a penetrant inspection of the damaged area. Refer to SOPM 20-20-02.

(2) Remove the damage as necessary.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) After you remove the damage, do the procedures that follow:

(a) Apply a chemical conversion coating to the bare surfaces of the aluminum parts. Refer to 51-20-01.

(b) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area. Refer to SOPM 20-41-02.

(c) Make sure the aerodynamic smoothness is satisfactory or there will be a loss in economic performance. Refer to 51-10-01.

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THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

F99814 S0006594729_V1

Outboard Krueger Flap Structure Location
Figure 102

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THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

F99818 S0006594730_V1

Inboard Krueger Flap Structure Location
Figure 103

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Seals, Flaps, and Bullnoses

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , and D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and F.

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Detail G .

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

G77074 S0006594731_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

D

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

E

G77073 S0006594732_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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G77072 S0006594733_V2

Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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REPAIR 1 - INBOARD WING LEADING EDGE KRUEGER FLAP STRUCTURE - EXTERNAL REPAIR BETWEEN THE RIBS

1. Applicability

A. Repair 1 is applicable to damage:

(1) To the structure of the wing leading edge Krueger flap

(2) That is between the ribs as shown in Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1

(3) Where it is not possible to make a satisfactory weld repair to the damaged area.

2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the economic performance of the airplane. Refer to 51-10-01.

G29942 S0006594736_V2

Wing Leading Edge Krueger Flap Structure Locations
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL

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(Continued)

Reference	Title
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-50-11	Application of Aerodynamic Smoothing Sealant

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge flap structure. Refer to 51-10-02 for the damage removal procedures.

(1) Make the cut in the shape of a rectangle with the sides perpendicular to the rib.

(a) Refer to Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1 for a typical cutout.

(2) Make the corner radii of the cut a minimum of 0.50 inch.

(a) Refer to Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1 for the shape of the cutout.

B. Make the repair parts as shown in Krueger Flap Structure Wing Leading Edge Repair Between Ribs, Figure 202/REPAIR 1. Refer to Table 201/REPAIR 1 for the materials.

(1) Make sure there is a minimum of two rows of repair fasteners around the edge of the cutout.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Internal Doubler	1	Use clad 2024-T3 sheet that is 0.050 inch thick
[2]	External Doubler	1	Use clad 2024-T3 sheet that is 0.071 inch thick
[3]	Filler	1	Use clad 2024-T3 sheet that is the same thickness as the initial leading edge flap structure

C. Assemble the repair parts and drill the fastener holes.

(1) Do not countersink the fastener holes more than 70 percent of the material thickness.

(a) This will prevent a knife-edge condition of the external structure.

D. Disassemble the repair parts.

E. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial structure.

F. Apply one layer of BMS 10-79, Type III primer to:

(1) The surfaces of the initial structure

(2) The mating surfaces of the repair parts.

G. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

(1) Fill the space between the part [3] filler and the initial structure with BMS 5-95 sealant.

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H. Install the fasteners by the squeeze rivet method and without sealant.

NOTE: It is optional to install the fasteners by the hand drive method. Make sure to visually inspect for cracks in the initial structure.

I. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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TYPICAL REPAIR AREA

A–A

FASTENER SYMBOLS

✦ REPAIR FASTENER LOCATION. INSTALL A BACR15BA6AD RIVET.

G30046 S0006594738_V2

Krueger Flap Structure Wing Leading Edge Repair Between Ribs
Figure 202

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REPAIR 2 - WING LEADING EDGE KRUEGER FLAP STRUCTURE - EXTERNAL REPAIR AT RIBS

1. Applicability

A. Repair 2 is applicable to damage:

(1) To the structure of the wing leading edge Krueger flap

(2) That is at the ribs as shown in Krueger Flap Structure Wing Leading Edge Repair at a Rib, Figure 202/REPAIR 2.

2. General

A. Repair 2 is a Category A repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

G31174 S0006594740_V2

Wing Leading Edge Krueger Flap Structure Locations
Figure 201

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING

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(Continued)

Reference	Title
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-50-11	Application of Aerodynamic Smoothing Sealant

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge flap structure. Refer to 51-10-02 for the removal of damage procedures.

(1) If possible, remove the tapered part of the rib.

(a) Install a tapered filler if the tapered rib can not be removed.

(2) Make the cut in the shape of a rectangle.

(a) Refer to Krueger Flap Structure Wing Leading Edge Repair at a Rib, Figure 202/REPAIR 2 for a typical cutout.

B. Make the repair parts as shown in Krueger Flap Structure Wing Leading Edge Repair at a Rib, Figure 202/REPAIR 2. Refer to Table 201/REPAIR 2 for the repair parts.

(1) Make sure you can put a minimum of two rows of repair fasteners around the edge of the cutout.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	2	Use clad 2024-T4 sheet that is 0.063 inch thick
[2]	Filler	As necessary	Use clad 2024-T3 clad. Use a maximum thickness of 0.03 inch to fill the space between the installed part [1] angle and the initial rib. Taper the filler as necessary
[3]	Filler	1	Use 2024-T3 clad sheet that is the same thickness as the initial rib
[4]	Filler	1	Use 2024-T3 clad sheet that is the same thickness as the initial rib
[5]	Doubler	1	Use 2024-T3 clad sheet that is 0.071 inch thick

C. Assemble the repair parts and drill the fastener holes.

(1) Do not countersink the fastener holes more than 70 percent of the material thickness.

(a) This will prevent a knife-edge condition of the initial structure.

D. Disassemble the repair parts.

E. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and to the bare surfaces of the initial structure.

F. Apply one layer of BMS 10-79, Type III primer to:

(1) The surfaces of the initial structure

(2) The mating surfaces of the repair parts.

G. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

NOTE: As necessary, fill the space between the part [2] filler and the mating surfaces with shims.

(1) Fill the spaces between the part [2] and part [3] fillers and the initial structure with BMS 5-95 sealant.

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H. Install the fasteners by the squeeze rivet method and without sealant. Refer to 51-40-02.

NOTE: It is optional to install the fasteners by the hand drive method. Make sure to visually inspect for any cracks to the initial structure.

I. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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TYPICAL REPAIR AREA

FASTENER SYMBOLS

- REPAIR FASTENER LOCATION. INSTALL A BACR15CE6D RIVET.
- REPAIR FASTENER LOCATION. INSTALL A BACR15BB6D RIVET.

G32513 S0006594742_V2

Krueger Flap Structure Wing Leading Edge Repair at a Rib
Figure 202 (Sheet 1 of 2)

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Krueger Flap Structure Wing Leading Edge Repair at a Rib
Figure 202 (Sheet 2 of 2)

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IDENTIFICATION 1 - INBOARD WING LEADING EDGE (KRUEGER) FLAP FITTINGS

NOTE: REFER TO TABLE 1 FOR THE LIST OF REFERENCE DRAWINGS.

THE LEFT SIDE WING LEADING EDGE FLAPS ARE SHOWN,
THE RIGHT SIDE WING LEADING EDGE FLAPS ARE OPPOSITE

F76601 S0006594747_V1

Inboard Wing Leading Edge (Krueger) Flap Fitting Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
114A1110	Krueger Flap and Bullnose Assembly - Numbers 1 and 4
114A1220	Krueger Flap and Bullnose Assembly - Numbers 2 and 3

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

F76854 S0006594749_V1

Outboard Krueger Flap Fitting Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Hinge Fitting Number 1	0.60 (15.2)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	
[2]	Hinge Fitting Numbers 2 and 3	0.60 (15.2)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	
[3]	Link - Bullnose	0.75 (19.0)	7050-T7451 plate as given in AMS 4050.	
[4]	Hinge Fitting Number 4	0.60 (15.2)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	
[5]	Hinge Fitting - Krueger Seal Hinge		17-4 PH CRES casting as given in AMS 5344, heat treated to 180 KSI minimum	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

F76855 S0006594751_V1

Inboard Krueger Flap Fitting Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Hinge Fitting Number 5	0.60 (15.2)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	
[2]	Link - Bullnose	0.75 (19.0)	7050-T7451 plate as given in AMS 4050	
[3]	Hinge Fitting Number 6	0.80 (20.3)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	
[4]	Hinge Fitting Number 7	0.80 (20.3)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	
[5]	Link - Bullnose	1.0 (25.4)	7050-T7451 plate as given in AMS 4050	
[6]	Hinge Fitting Number 8	0.70 (17.8)	Ti-6Al-4V titanium plate as given in MIL-T-9046, Code AB-1 in the annealed condition	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - INBOARD WING LEADING EDGE (KRUEGER) FLAP FITTINGS

1. Applicability

A. This subject gives the allowable damage limits for the inboard wing leading edge (Krueger) flap fittings shown in Inboard Wing Leading Edge (Krueger) Flap Fittings, Figure 101/ALLOWABLE DAMAGE 1.

2. General

WARNING: SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH ALL FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND, OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON, OR CARBON DIOXIDE. IF WATER TOUCHES TITANIUM THAT IS ON FIRE, A STEAM EXPLOSION CAN OCCUR.

A. Refer to SOPM 20-10-07 when you work with titanium.

B. Remove the damaged material as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of the abrasive materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the tools and equipment you can use to remove the damage.

C. After the damage is removed, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the bullnose links, but not the inner surfaces of the link holes.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the reworked aluminum parts. Refer to 51-20-01.

(3) Apply a layer of BMS 10-79, Type III primer to the surfaces of the reworked aluminum parts. Refer to SOPM 20-41-02.

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NOTE: REFER TO PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

THE NUMBER 1 KRUEGER FLAP IS SHOWN,
THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

OUTBOARD KRUEGER FLAP

A

F77282 S0006594755_V1

Inboard Wing Leading Edge (Krueger) Flap Fittings
Figure 101 (Sheet 1 of 2)

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NOTE: REFER TO PARAGRAPH 4 FOR THE ALLOWABLE DAMAGE LIMITS.

THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

INBOARD KRUEGER FLAP

B

F77278 S0006594756_V1

Inboard Wing Leading Edge (Krueger) Flap Fittings
Figure 101 (Sheet 2 of 2)

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-10-03	SHOT PEENING
SOPM 20-10-07	Machining of Titanium
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Cracks:

(1) Remove the damage as shown in Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , C , and E .

B. Nicks, Gouges, Scratches, and Corrosion:

(1) Remove the damage as shown in Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage, Figure 102/ALLOWABLE DAMAGE 1, Details A through F .

C. Dents are not permitted.

D. Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

F77284 S0006594757_V1

Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage
Figure 102 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

F77289 S0006594758_V1

Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage
Figure 102 (Sheet 2 of 3)

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REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BEARING

F

B–B

C–C

F77286 S0006594759_V1

Inboard Wing Leading Edge (Krueger) Flap Fittings Allowable Damage
Figure 102 (Sheet 3 of 3)

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REPAIR 1 - INBOARD WING LEADING EDGE (KRUEGER) FLAP FITTINGS

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

THE NUMBER 1 KRUEGER FLAP IS SHOWN, THE NUMBER 4 KRUEGER FLAP IS OPPOSITE

OUTBOARD KRUEGER FLAP

A

F77329 S0006594762_V2

Inboard Wing Leading Edge (Krueger) Flap Fitting Repair
Figure 201 (Sheet 1 of 2)

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NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

THE NUMBER 2 KRUEGER FLAP IS SHOWN,
THE NUMBER 3 KRUEGER FLAP IS OPPOSITE

INBOARD KRUEGER FLAP

B

F77342 S0006594763_V2

Inboard Wing Leading Edge (Krueger) Flap Fitting Repair
Figure 201 (Sheet 2 of 2)

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IDENTIFICATION 1 - WING FIXED TRAILING EDGE SKIN PANELS

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

G78848 S0006594768_V1

Wing Fixed Trailing Edge Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
115A0001	Functional Product Collector, Fixed Trailing Edge - Left Wing
115A0002	Functional Product Collector, Fixed Trailing Edge - Right Wing
115A0005	Functional Product Collector, Fixed Trailing Edge to Final Assembly
115A2000	Upper Installation - Inboard Fixed Trailing Edge
115A2001	Lower Installation - Inboard Fixed Trailing Edge
115A2200	Boltcover Installation - Inboard Fixed Trailing Edge
115A2500	Panel Installation - Inboard Fixed Trailing Edge

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
115A2510	Panel Assembly - Upper Inboard Fixed Trailing Edge
115A2511	Panel Assembly - Upper, Inboard Fixed Trailing Edge
115A2512	Panel Assembly - Upper Inboard Fixed Trailing Edge
115A2513	Panel Assembly - Upper Inboard Fixed Trailing Edge
115A2514	Panel Assembly - Upper Inboard Fixed Trailing Edge
115A2515	Panel Assembly - Upper Inboard Fixed Trailing Edge
115A2525	Panel Assembly - Bonded Part, Upper Inboard Fixed Trailing Edge
115A2710	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A2711	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A2712	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A2713	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A2714	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A2715	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A2716	Panel Assembly - Lower Inboard Fixed Trailing Edge
115A3500	Panel Installation - Mid Span Fixed Trailing Edge, Upper
115A3510	Panel Assembly - Mid Span Fixed Trailing Edge, Upper
115A3511	Panel Assembly - Mid Span Fixed Trailing Edge, Upper
115A3512	Panel Assembly - Mid Span Fixed Trailing Edge, Upper
115A3513	Panel Assembly - Mid Span Fixed Trailing Edge, Upper
115A3700	Panel Installation - Mid Span Fixed Trailing Edge, Lower
115A3710	Panel Assembly - Mid Span Fixed Trailing Edge, Lower
115A3711	Panel Assembly - Mid Span Fixed Trailing Edge, Lower
115A3712	Panel Assembly - Mid Span Fixed Trailing Edge, Lower
115A4500	Panel Installation - Outboard Foxed Trailing Edge, Upper
115A4510	Panel Assembly - Outboard Fixed Trailing Edge, Upper
115A4511	Panel Assembly - Outboard Fixed Trailing Edge, Upper
115A4512	Panel Assembly - Outboard Fixed Trailing Edge, Upper
115A4700	Panel Installation - Outboard Foxed Trailing Edge, Lower
115A4710	Panel Assembly - Outboard Fixed Trailing Edge, Lower
115A4711	Panel Assembly - Outboard Fixed Trailing Edge, Lower
115A4712	Panel Assembly - Outboard Fixed Trailing Edge, Lower
115A4714	Panel Assembly - Outboard Fixed Trailing Edge, Lower
115A4900	Structure Installation - Panels and Supports, Outboard Fixed Trailing Edge
115A4901	Structure Assembly - Panels and Supports, Outboard Fixed Trailing Edge
115A4910	Panel Assembly - Outboard Fixed Trailing Edge, Upper
115A4911	Panel Assembly - Outboard Fixed Trailing Edge, Lower
115A4912	Panel Assembly - Outboard Fixed Trailing Edge, Lower

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EXPANDED ALUMINUM FOIL MESH (BMS 8-336)

NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

UPPER SURFACE

G78854 S0006594770_V2

Wing Upper Fixed Trailing Edge Skin Panels
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Panel Assembly - Bonded Part		Glass fiber reinforced plastic (GFRP) honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	1.00 (25.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[2]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[3]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336 Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[4]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[5]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	0.30 (7.6)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[6]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[7]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[8]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	1.00 (25.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[9]	Panel Assembly	0.10 (2.5)	7075-T62 sheet as given in QQ-A-250/12	
[10]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
[11]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 778 *[2]	
[12]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[13]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	1.00 (25.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

*[2] Refer to the production drawings for the ply lay-up and orientation.

*[3] Refer to the production drawings for the core ribbon direction.

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EXPANDED ALUMINUM FOIL MESH (BMS 8-336)

NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LOWER SURFACE

G78867 S0006594772_V1

Wing Lower Fixed Trailing Edge Panels
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Panel Assembly - Bonded Part		Glass fiber reinforced plastic (GFRP) honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[2]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[3]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[4]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	

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Table 3: (Continued)

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[5]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 7781 or 1581 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[6]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.80 (20.3)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[7]	Panel Assembly		A356-T6 Casting as given in AMS 2418	
[8]	Panel Assembly	0.10 (2.5)	2024-T42 sheet	

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Table 3: (Continued)

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[9]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	For cum line numbers 475 and on and airplanes that include Service Bulletin 737-57-1255
	Skin Ply (3)		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies (3)		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120, 220, 1581, or 7781 *[2]	
	Core (5)	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[9]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	For all other airplanes
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or Style 1581 (Optional: 7781) *[2]	
	Core	0.30 (7.6) 0.40 (10.2) 0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[10]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.30 (7.6)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[11]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	

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Table 3: (Continued)

LIST OF MATERIALS FOR FIGURE 3				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
[12]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 or 220 *[2]	
			Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.50 (12.7) 0.80 (20.3)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[13]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 7781 or 1581 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.80 (20.3)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[14]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	

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Table 3: (Continued)

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
	Core	0.50 (12.7) 0.80 (20.3)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 Type V, Grade 3.0 *[3]	
[15]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 7781 or 1581 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	1.00 (25.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[16]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	0.90 (22.9) 1.00 (25.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	
[17]	Panel Assembly - Bonded Part		GFRP honeycomb sandwich	
	Lightning Protection Ply		Expanded aluminum foil as given in BMS 8-336, Type I, Class I, Grade 016	
	Skin Ply		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Doubler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 or 7781 *[2]	
	Filler Plies		Epoxy impregnated fiberglass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: 220) or 1581 (Optional: 7781) *[2]	
	Core	1.00 (25.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0 *[3]	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

*[2] Refer to the production drawings for the ply lay-up and orientation.

*[3] Refer to the production drawings for the core ribbon direction.

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ALLOWABLE DAMAGE 1 - WING FIXED TRAILING EDGE SKIN PANELS

1. Applicability

A. Allowable Damage 1 is applicable to damage on the wing fixed trailing edge skin panels as shown in Wing Fixed Trailing Edge Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.

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UPPER FIXED TRAILING EDGE PANELS
(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

LOWER FIXED TRAILING EDGE PANELS
(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

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Wing Fixed Trailing Edge Skin Panel Location
Figure 101 (Sheet 1 of 2)

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2438818 S0000565743_V1

Wing Fixed Trailing Edge Skin Panel Location
Figure 101 (Sheet 2 of 2)

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2. General

A. Do the steps that follow for the skin panels made of Glass Fabric Reinforced Plastic (GFRP).

(1) Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator, can be used.

(a) For the honeycomb core areas, the tap test is an alternative procedure to an instrumented NDT.

(2) Remove all the contamination and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and the cleanup procedures.

(a) Refer to 51-70-04 for the damage removal procedures.

(b) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(3) Refer to Definitions of the Damage Size, Figure 102/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(4) Refer to Definitions of the Facesheets, Figure 103/ALLOWABLE DAMAGE 1 for the definitions of the facesheets of a honeycomb core area.

(5) Refer to Wing Fixed Trailing Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the GFRP panels.

(6) Some GFRP panels have BMS 8-336 expanded aluminum foil mesh as shown in Wing Fixed Trailing Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1. If damage occurs to the expanded aluminum foil mesh, do the steps that follow:

(a) Refer to 51-70-14 for the allowable damage limits for the expanded aluminum foil mesh.

(b) Seal the damaged area as given in 51-70-14.

(7) Seal all damaged areas that do not have BMS 8-336 expanded aluminum foil mesh with the steps that follow.

(a) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

1) Make a temporary seal.

a) Apply aluminum foil tape (speed tape).

b) Keep a record of the location.

c) Make sure the tape is in satisfactory condition at each 400 flight hour interval or more frequently.

d) Seal the damage permanently no later than 5000 flight hours from the time the seal was made.

2) Make a permanent seal.

a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given 51-70-08.

b) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to SOPM 20-44-04.

c) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(b) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

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1) Use a vacuum and heat to remove moisture from the solid laminate or the honeycomb cells. Refer to 51-70-04.
2) Make a temporary seal with aluminum foil tape (speed tape).
3) Keep a record of the location.
4) Repair the damage no later than 5000 flight hours from the time the seal was made.

B. Do the steps that follow for the skin panels made of formed aluminum sheet.

(1) Refer to Wing Fixed Trailing Edge Skin Allowable Damage, Figure 104/ALLOWABLE DAMAGE 1 for the location of the aluminum panels.

(2) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of the abrasive and other materials you need to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

(3) After you remove the damage, do the steps that follow:

(a) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01.

(b) Apply one layer of BMS 10-11, Type I primer to the conversion coated, reworked areas. Refer to SOPM 20-41-02.

(c) Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the economic performance of the airplane. Refer to 51-10-01.

C. Do the steps that follow for the conductive strip

(1) Remove damage from the aluminum material as necessary.

(a) Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02 for the inspection and removal of damage.

(b) Refer to NON-METALLIC MATERIALS, 51-30-03 for possible sources of the abrasive and other materials you can use to remove the damage.

(c) Refer to EQUIPMENT AND TOOLS FOR REPAIRS, 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) Make sure there is a 125 micro inches Ra or smoother surface finish. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

(3) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

(4) Apply two layers of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

(5) Apply BMS 5-95 sealant at the damage location to make sure all gaps are filled. Refer to REPAIR SEALING, 51-20-05.

(6) Apply the decorative finish to the reworked areas as given in AMM 51-21-99/701.

(7) Make sure the aerodynamic smoothness is satisfactory and not more than the limits given in AERODYNAMIC SMOOTHNESS, 51-10-01. If the aerodynamic smoothness is not satisfactory, there will be a decrease in the performance of the aircraft.

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(8) Before 30 days, make an initial visual inspection of the damaged Conductive Strip for signs of new damage or for loose or missing fasteners. Repeat the inspections at each 30 day interval. If any new damage is found to the Conductive Strip then replace the part.

NOTE: Once the damaged conductive strip is replaced with a new part, this 30 day inspection requirement ends.

(9) Replace the damaged Conductive Strip with a new part in 24 months or less.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	PARAGRAPH
GFRP/HONEYCOMB SANDWICH PANELS - HONEYCOMB CORE AREAS	4.A
GFRP/HONEYCOMB SANDWICH PANELS - SOLID LAMINATE AREAS	4.B
GFRP SOLID LAMINATE PANELS	4.C
FORMED ALUMINUM SHEET PANELS	4.D
ALUMINUM CASTING PANELS	4.E

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SIZE DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

SIZE DEFINITIONS FOR
DENT DAMAGE

B

SIZE DEFINITIONS FOR
EDGE DAMAGE

C

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Definitions of the Damage Size
Figure 102

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F59847 S0006594779_V1

Definitions of the Facesheets
Figure 103

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Wing Fixed Trailing Edge Skin Allowable Damage
Figure 104 (Sheet 1 of 2)

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Wing Fixed Trailing Edge Skin Allowable Damage
Figure 104 (Sheet 2 of 2)

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-13	SURFACE ROUGHNESS FINISH REQUIREMENTS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
51-70-14	STRUCTURES WITH ALUMINUM COATINGS AND FOILS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits

A. GFRP/Honeycomb Sandwich Panels - Honeycomb Core Areas

(1) Nicks, gouges and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges and scratches that cause damage to the glass fibers are permitted if:

(a) The depth of the damage is less than the limits given in Table 102/ALLOWABLE DAMAGE 1 and Table 103/ALLOWABLE DAMAGE 1.

NOTE: Use the limits for holes and punctures if the depth of the damage is more than the limits given in Table 102/ALLOWABLE DAMAGE 1 and Table 103/ALLOWABLE DAMAGE 1.

(b) They are a maximum of 2.0 inches in length

(c) They are a maximum of 0.25 inch in width

(d) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

Table 102:

MAXIMUM DEPTH OF NICK, GOUGE, OR SCRATCH DAMAGE FOR GFRP/HONEYCOMB SANDWICH CORE AREAS ON THE WING UPPER SURFACE		
PANEL DRAWING NUMBER	MAXIMUM NUMBER OF PLIES IN DEPTH	
	UPPER SURFACE (TOOL SIDE FACESHEET)	LOWER SURFACE (BAG SIDE FACESHEET)
115A2511	2	2
115A2520	1	1

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Table 102: (Continued)

MAXIMUM DEPTH OF NICK, GOUGE, OR SCRATCH DAMAGE FOR GFRP/HONEYCOMB SANDWICH CORE AREAS ON THE WING UPPER SURFACE		
PANEL DRAWING NUMBER	MAXIMUM NUMBER OF PLIES IN DEPTH	
	UPPER SURFACE (TOOL SIDE FACESHEET)	LOWER SURFACE (BAG SIDE FACESHEET)
115A2521	1	1
115A2523	1	Fiber damage is not permitted
115A2524	1	1
115A2526	2	Fiber damage is not permitted
115A3522	2	1
115A3523	2	1
115A4520	2	Fiber damage is not permitted
115A4521	2	Fiber damage is not permitted
115A4920	2	Fiber damage is not permitted

Table 103:

MAXIMUM DEPTH OF NICK, GOUGE, OR SCRATCH DAMAGE FOR GFRP/HONEYCOMB SANDWICH CORE AREAS ON THE WING LOWER SURFACE		
PANEL DRAWING NUMBER	MAXIMUM NUMBER OF PLIES IN DEPTH	
	UPPER SURFACE (BAG SIDE FACESHEET)	LOWER SURFACE (TOOL SIDE FACESHEET)
115A2720	1	2
115A2721	1	2
115A2722	1	2
115A2724	1	2
115A2726	1	2
115A3720	1	1
115A3721	1	1
115A3722	1	1
115A4720	Fiber damage is not permitted	Fiber damage is not permitted
115A4721	Fiber damage is not permitted	Fiber damage is not permitted
115A4722	Fiber damage is not permitted	1
115A4921	1	1
115A4922	1	1
115A4923	1	Fiber damage is not permitted

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Table 104:

MAXIMUM DAMAGE DIMENSIONS AND MINIMUM SPACING DIMENSIONS FOR DENT, HOLE AND PUNCTURE, AND DELAMINATION DAMAGE ON GFRP PANEL HONEYCOMB CORE AREAS ON THE WING UPPER SURFACE				
PANEL DRAWING NUMBER	UPPER SURFACE (TOOL SIDE FACESHEET)		LOWER SURFACE (BAG SIDE FACESHEET)	
	MAXIMUM DAMAGE DIMENSION "D" (INCHES)	MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES)	MAXIMUM DAMAGE DIMENSION "D" (INCHES)	MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES)
115A2511	1.5	(1.5)x(D+d)	1.5	(1.5)x(D+d)
115A2520	2.0	(3.0)x(D+d)	2.0	(2.5)x(D+d)
115A2521	1.5	(3.5)x(D+d)	2.0	(2.0)x(D+d)
115A2523	1.0	(4.5)x(D+d)	1.0	(5.0)x(D+d)
115A2524	1.5	(2.5)x(D+d)	2.0	(1.5)x(D+d)
115A2526	2.0	(1.5)x(D+d)	1.0	(2.5)x(D+d)
115A3522	1.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A3523	1.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A4520	1.5	(1.5)x(D+d)	1.0	(1.5)x(D+d)
115A4521	1.5	(1.5)x(D+d)	0.75	(4.0)x(D+d)
115A4920	2.0	(1.5)x(D+d)	1.5	(2.5)x(D+d)

Table 105:

MAXIMUM DAMAGE DIMENSIONS AND MINIMUM SPACING DIMENSIONS FOR DENT, HOLE AND PUNCTURE, AND DELAMINATION DAMAGE ON GFRP PANEL HONEYCOMB CORE AREAS ON THE WING LOWER SURFACE				
PANEL DRAWING NUMBER	UPPER SURFACE (BAG SIDE FACESHEET)		LOWER SURFACE (TOOL SIDE FACESHEET)	
	MAXIMUM DAMAGE DIMENSION "D" (INCHES)	MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES)	MAXIMUM DAMAGE DIMENSION "D" (INCHES)	MINIMUM DAMAGE SPACING DIMENSION "A" (INCHES)
115A2720	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A2721	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A2722	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A2724	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A2726	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A3720	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A3721	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A3722	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A4720	1.0	(2.0)x(D+d)	1.5	(1.5)x(D+d)
115A4721	1.0	(2.0)x(D+d)	1.5	(1.5)x(D+d)
115A4722	1.0	(2.0)x(D+d)	1.5	(1.5)x(D+d)
115A4921	2.0	(1.5)x(D+d)	2.0	(1.5)x(D+d)
115A4922	2.0	(1.5)x(D+d)	1.5	(1.5)x(D+d)
115A4923	2.0	(1.5)x(D+d)	1.5	(2.5)x(D+d)

(3) Dents are permitted if:

(a) The dimension of the damage is less than the limits given in Table 104 and Table 105.

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(b) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if:

(a) They are a maximum of one facesheet and the core in depth

(b) The dimension of the damage is less than the limits given in Table 104 and and Table 105.

(c) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if:

(a) The dimension of the damage is less than the limits given in Table 104 and Table 105.

(b) They are a minimum distance from the edge of other damage as shown in Damage that is Permitted to Honeycomb Core Areas, Figure 105/ALLOWABLE DAMAGE 1 and Tables 104 and 105. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Pararaph 2.

B. GFRP/Honeycomb Sandwich Panels - Solid Laminate Areas

(1) Nicks, gouges and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges and scratches that cause damage to the glass fibers are not permitted.

(3) Dents are permitted if they are:

NOTE: Use the limits for holes and punctures if there is glass fiber damage.

(a) A maximum of 0.50 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter

(b) A minimum of 2.5D (D = the diameter of the damage) from other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(5) Delaminations are permitted if:

(a) They are a maximum of 0.25 inch in diameter

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(b) They are a minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, the edge of a hole, or the edge of the material. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.50 inch in width

(c) A minimum of 2.5D (D = the diameter of the damage) from the edge of other damage or the edge of a hole. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(7) Edge erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ ALLOWABLE DAMAGE 1.

C. GFRP Solid Laminate Panels (does not include the solid laminate areas in honeycomb sandwich panels)

(1) Nicks, gouges and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges and scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of 1 ply in depth

NOTE: Use the limits for holes and punctures if the depth of the damage is more than 1 ply in depth.

(b) A maximum of 3.0 inches in length

(c) A maximum of 0.25 inch in width

(d) A minimum of 0.50 inch away from the edge of other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(3) Dents are permitted if they are:

(a) A maximum of 0.5 inch in diameter.

(b) A minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch in diameter

(b) A minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, fastener holes or material edges. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies and

2) Are sealed as given in Paragraph 2.

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(5) Delaminations are permitted if:

(a) They are a maximum of 0.5 inch in diameter

(b) They are a minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, the edge of a hole, or the edge of the material. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.50 inch in width

(c) They are a minimum of 2.5D (D = the diameter of the damage) from the edge of other damage, the edge of a hole, or the edge of the material. Other damage does not include nicks, gouges, and scratches that:

1) Do not cause damage to the glass fiber plies

2) Are sealed as given in Paragraph 2.

(7) Edge erosion is permitted as shown in Cleanup and Sealing of Edge Erosion, Figure 106/ALLOWABLE DAMAGE 1.

D. Panels Made of Formed Aluminum Sheet

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , and C.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E.

(3) Dents:

(a) Dents are permitted if:

1) They agree with the conditions shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail F

2) There is 1.5 square inches or less of dent area for each 144 square inches of panel area.

(b) If the depth of a dent is larger than 0.050 inch and the dent agrees with the conditions shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail F , do the steps that follow:

1) Fill or rework the dent as given in 51-70-01

2) If you fill the dent, then do the steps that follow:

a) Seal the damage with aluminum foil tape (speed tape)

b) Make an inspection of the dent at each 400 flight hour interval or more frequently.

- Make sure the tape is in satisfactory condition
- Repair the dent if the damage becomes larger.

c) Repair the damage at or before 1000 flight hours have occurred.

(4) Holes and Punctures are permitted if:

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(a) They are 0.25 inch in diameter or less

(b) The edge of the damage is a minimum of 1.0 inch away from a fastener hole, an edge, or other damage

(c) They are filled with a 2017-T3 or 2117-T4 aluminum flush head rivet. Install the rivet without sealant.

E. Aluminum Casting Panel

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , and C.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

F. Conductive Strip

NOTE: This allowable damage section is approved only for conductive strip damage where none of the initial fasteners are loose or damaged. If there are initial fasteners loose or damaged, refer to 57-51-01, Repair 2 - Conductive Strip.

(1) Remove the damage. Refer to Figure 108 for the permitted damage limits. Determine the area removed from the initial cross-sectional area. If the area removed is less than or equal to 20% of the initial cross-sectional area, continue with Paragraph 4.F.(2). If the area removed is more than 20% of the initial cross-sectional area, refer to 57-51-01, Repair 2 - Conductive Strip.

(2) Before 30 days, make an initial visual inspection of the damaged Conductive Strip for signs of new damage and for loose or missing fasteners. Repeat the inspections at each 30 day interval. If any new damage is found to the Conductive Strip then replace the part.

NOTE: Once the damaged Conductive Strip is replaced with a new part, this 30 day inspection requirement ends.

(3) Replace the damaged Conductive Strip with a new part in 24 months or less.

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NOTE: THE DIMENSION OF A DAMAGE AREA IS EITHER THE DIMENSION OF THE VISUAL DAMAGE OR THE DIMENSION OF THE DELAMINATION. USE THE DIMENSION OF THE LARGER DAMAGE.

D IS THE LARGER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

d IS THE SMALLER DIMENSION OF TWO ADJACENT DAMAGE AREAS.

A IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

F65724 S0006594786_V1

Damage that is Permitted to Honeycomb Core Areas
Figure 105

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F65725 S0006594787_V1

Cleanup and Sealing of Edge Erosion
Figure 106

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

C

F65751 S0006594788_V1

Allowable Damage Limits
Figure 107 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

D

A–A

REMOVAL OF DAMAGE AROUND THE
FASTENERS ON AN EDGE
OR A SURFACE

E

B–B

F65754 S0006594789_V1

Allowable Damage Limits
Figure 107 (Sheet 2 of 3)

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DENT THAT IS PERMITTED

F

F65757 S0006594790_V2

Allowable Damage Limits
Figure 107 (Sheet 3 of 3)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20% OF STRIP CROSS SECTIONAL AREA

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20% OF STRIP CROSS SECTIONAL AREA

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 20 PERCENT OF THE STRIP CROSS SECTIONAL AREA

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

2435257 S0000564938_V1

Conductive Strip Allowable Damage
Figure 108 (Sheet 1 of 2)

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CROSS-SECTIONAL AREA OF THE CONDUCTIVE STRIP AT A FASTENER HOLE

A-A

D_1 = DIAMETERS OF THE INITIAL FASTENER HOLES

t = MATERIAL THICKNESS

W = STRIP WIDTH

A_i = INITIAL AREA OF THE STRIP

= THE CROSS-SECTIONAL AREA MINUS THE CROSS-SECTIONAL AREA OF THE INITIAL HOLE (AS MANUFACTURED BY BOEING)

$= Wt - D_1 t$

A_1 = CROSS-SECTIONAL AREA OF THE DAMAGE REMOVED AT LOCATION 1

$\left(\frac{A_1}{A_i}\right) \times 100$ = PERCENT OF CROSS-SECTIONAL AREA REMOVED

= A MAXIMUM OF 20 PERCENT

D

2435260 S0000564940_V1

Conductive Strip Allowable Damage
Figure 108 (Sheet 2 of 2)

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REPAIR 1 - WING FIXED TRAILING EDGE SKIN PANELS MADE OF COMPOSITE MATERIALS

1. Applicability

A. Repair 1 is applicable to damage to skin panels made of Glass Fiber Reinforced Plastic (GFRP) as shown in Wing Fixed Trailing Edge Skin Panel Location, Figure 201/REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

SEE B FOR THE LOWER FIXED
TRAILING EDGE SKIN PANELS

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F67443 S0006594793_V1

Wing Fixed Trailing Edge Skin Panel Location
Figure 201 (Sheet 1 of 3)

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NOTE: REFER TO PARAGRAPH 4 FOR THE REPAIR DATA.

UPPER SURFACE

A

F67452 S0006594794_V1

Wing Fixed Trailing Edge Skin Panel Location
Figure 201 (Sheet 2 of 3)

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NOTE: REFER TO PARAGRAPH 4 FOR THE REPAIR DATA.

LOWER SURFACE

B

F67454 S0006594795_V1

Wing Fixed Trailing Edge Skin Panel Location
Figure 201 (Sheet 3 of 3)

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2. General

A. Repair 1 gives repair instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the panels as necessary to get access to the inner surface of the skin.

(1) Remove the necessary fasteners. Refer to 51-40-02 for information on fastener removal.

(2) If a fastener hole is damaged, refer to 51-70-04 or 51-70-05, as applicable.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-01 for the inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Definitions of the Damage Size, Figure 202/REPAIR 1 for the definitions of the length, width, and depth of damage.

(2) Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

D. Some trailing edge panels have expanded aluminum foil mesh (BMS 8-336). If damage occurs:

(1) Refer to 51-70-14 before you repair a facesheet that has expanded aluminum foil mesh (BMS 8-336).

(2) Refer to 51-70-14 after you repair a facesheet that has expanded aluminum foil mesh (BMS 8-336).

E. Do the repair as given in Paragraph 4./REPAIR 1

F. Install the wing fixed trailing edge panels that were removed.

(1) Install the fasteners after all repairs are complete. Refer to 51-40-02 for information on fastener installation.

G. Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.

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SIZE DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

SIZE DEFINITIONS FOR
DENT DAMAGE

B

SIZE DEFINITIONS FOR
EDGE DAMAGE

C

F67444 S0006594796_V1

Definitions of the Damage Size
Figure 202

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F67445 S0006594797_V1

Definitions of the Facesheets
Figure 203

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-05	REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS
51-70-06, REPAIR GENERAL	Room Temperature Cure Repairs With Wet Layup Materials For Glass Fabric Reinforced Plastic Solid Laminates and Honeycomb Core Panels
51-70-13	TYPICAL WEB REPAIRS
51-70-14	STRUCTURES WITH ALUMINUM COATINGS AND FOILS
737 NDT Part 1, 51-01-01	Inspection of Repairs to Composite Structure

4. Repair Instructions for the 250 Degree F (121 Degree C) Cure Panels

A. For dents that are a maximum of 2 inches in diameter and have no fiber damage or delamination, do the steps that follow:

(1) Fill the dent with BMS 5-28, Type 7 potting compound

(2) Apply a fiberglass patch over the potted area as given in 51-70-04.

B. For dents that are not permitted by Paragraph 4.A./REPAIR 1 and for other damage that is not permitted by Allowable Damage 1, refer to:

(1) Table 201/REPAIR 1 for panel areas other than the edgebands

(2) Table 202/REPAIR 1 for the edgebands.

C. For repairs with wet layup materials, do as follows:

(1) Use the same number of repair plies as the number of initial plies that were removed.

(2) Add two structural plies for each facesheet that is repaired. Put one ply at ±45 and the other ply at 0 or 90 degrees.

(3) Inspect Category B repairs after each 800 flight hour interval or more frequently.

(a) Refer to 737 NDT Part 1, 51-01-01 for the inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

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(b) Replace the repair with a Category A repair if you find deterioration.

D. Use the instructions that follow to do a Category A repair with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05.

(2) Use the same number of repair plies as the number of initial plies that were removed.

Table 201:

REPAIR DATA FOR THE 250°F (121°C) CURE TRAILING EDGE SKIN PANELS FOR AREAS OTHER THAN THE EDGEBANDS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**Room Temperature**	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**
REPAIR SIZE	Damage that is a maximum of: - 2.0 inches in diameter - 30 percent of the smallest dimension across the panel at the damage location - One facesheet and the honeycomb core in depth One repair for each 144 square inches 6.0 inches minimum clearance from: - other repairs (includes repairs on the opposite panel surface) - fastener holes - damage permitted in Allowable Damage 1 - panel edges	Damage that is a maximum of: - 4.0 inches in diameter - 50 percent of the smallest dimension across the panel at the damage location One repair for each 144 square inches 4.0 inches minimum clearance from: - other repairs (includes repairs on the opposite panel surface) - fastener holes - damage permitted in Allowable Damage 1 - panel edges	Damage that is a maximum of: - 6.0 inches in diameter - 50 percent of the smallest dimension across the panel at the damage location One repair for each 144 square inches 4.0 inches minimum clearance from: - other repairs (includes repairs on the opposite panel surface) - fastener holes - damage permitted in Allowable Damage 1 - panel edges	There are no limits on the dimensions of the repair
REPAIR INSTRUCTIONS	SRM 51-70-06 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-05 and Paragraph 4.D

Table 202:

REPAIR DATA FOR THE EDGEBANDS OF 250°F (121°C) CURE TRAILING EDGE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**Room Temperature**	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**
REPAIR SIZE	Damage that is a maximum of: - 15 percent of the cross-sectional area of the edgeband at the damage location - 10 percent of the length of the edgeband on the side of the damage	Damage that is a maximum of: - 15 percent of the cross-sectional area of the edgeband at the damage location - 10 percent of the length of the edgeband on the side of the damage	There are no limits on the dimensions of the repair	There are no limits on the dimensions of the repair

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Table 202: (Continued)

REPAIR DATA FOR THE EDGEBANDS OF 250°F (121°C) CURE TRAILING EDGE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**Room Temperature**	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**
REPAIR INSTRUCTIONS	SRM 51-70-06 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-05 and Paragraph 4.D

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REPAIR 2 - WING FIXED TRAILING EDGE PANEL CONDUCTIVE STRIP

1. Applicability

A. Repair 2 is applicable to damage on the wing fixed trailing edge panel conductive strip.

B. Repair 2 is applicable to damage to the conductive strip that is more than the limits permitted in SRM 57-51-01-1A-1, Allowable Damage 1.

2. General

A. Repair 2 gives instructions for a time-limited repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for definitions of the different categories of repairs.

B. Gain access to the damaged area.

(1) If necessary, remove the trailing edge panels.

(2) Refer to FASTENER INSTALLATION AND REMOVAL, 51-40-02 for information on fastener removal.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-20-13	SURFACE ROUGHNESS FINISH REQUIREMENTS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-05	FASTENER HOLE SIZES
51-40-08	COUNTERSINKING
51-70-14	STRUCTURES WITH ALUMINUM COATINGS AND FOILS
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Repair Instructions

A. Remove fasteners as necessary to the conductive strip at the damaged location. Refer to the part [1] doubler given in Figure 201.

B. Blend or completely remove the damaged area as shown in Figure 201. Maintain a minimum 125 micro inches Ra surface smoothness or smoother. Refer to SURFACE ROUGHNESS FINISH REQUIREMENTS, 51-20-13.

C. Make the repair parts as shown in Figure 201/REPAIR 2. Refer to Table 201/REPAIR 2 for the repair material. Make the part [1] doubler to follow the skin panel contour. Make sure the part [1] doubler extends two fasteners minimum beyond the damage in either direction. Make sure there is 1.5D minimum edge margin on all the fasteners. make sure the corner radii are 0.5 inch minimum.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Doubler	1	0.025 inch (0.635 mm) thick, 6061-T4 sheet. It is optional to use 2024-T3 sheet
[2]	Filler (if necessary)	1	Use 6061-T4 sheet with a thickness to limit all gaps to a maximum 0.01 inch (0.25 mm). It is optional to use 2024-T3 sheet.

D. Remove the finish from the initial conductive strip surface that is common to the part [1] doubler.

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E. Make the countersink washers for necessary initial fastener locations. Refer to COUNTERSINKING, 51-40-08.

F. At necessary initial fastener locations, install countersink repair washers with BMS 5-95 sealant. Refer to COUNTERSINKING, 51-40-08. Perform a resistance check between the washers and the conductive strip as given in BAC 5117-6. The maximum resistance is to be 0.01 ohms.

G. Assemble the repair parts as shown in Figure 201/REPAIR 2.

H. Drill the fastener holes. Refer to FASTENER HOLE SIZES, 51-40-05.

I. Disassemble the repair parts.

J. Remove all nicks, scratches, gouges, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

K. Install the repair parts dry.

L. Install the fasteners without sealant. Do a resistance check between the fastener head and the initial structure as given in BAC 5117.

M. Apply a chemical conversion coating to the repair parts and bare surfaces. Refer to PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS, 51-20-01.

N. Apply two layers of BMS 10-11, Type II primer to the bare surfaces except at areas common to the static wicks. Refer to SOPM 20-41-02.

O. Fillet seal the repair parts and all gaps with BMS 5-95 sealant. Refer to REPAIR SEALING, 51-20-05.

P. Do a resistance test between the part [1] doubler and conductive strip. Refer to STRUCTURES WITH ALUMINUM COATINGS AND FOILS, 51-70-14 for test procedures.

Q. Apply all initial production drawing finishes as necessary.

R. After 30 days or less, make an initial visual inspection or the repaired conductive strip for signs of new damage or for loose or missing fasteners. Repeat the inspections at each 30 day interval. If there is new damage found to the conductive strip then replace the conductive strip. Replacement of the damaged part with a new conductive strip stops the necessary inspection.

S. Replace the damaged conductive strip with a new conductive strip in 24 months or less.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
(UPPER SURFACE IS SHOWN)

A-A

FASTENER SYMBOLS

- REFERENCE FASTENER LOCATION
- INITIAL FASTENER LOCATION. INSTALL A BACR15FP5ER() RIVET (UP TO 1/32-INCH DIAMETER OVERSIZE)
- INITIAL FASTENER LOCATION. A BACR15BA6AC() RIVET USING SQUEEZE METHOD (UP TO 1/32-INCH DIAMETER OVERSIZE)

1 THE PART [1] REPAIR DOUBLER MUST EXTEND 2 FASTENERS BEYOND DAMAGE IN EITHER DIRECTIONS.

2 FILL VOID WITH BMS 5-95 SEALANT.

3 IF THIS FASTENER ROW IS INCLUDED IN THE REPAIR, EXTEND THE DOUBLER AS SHOWN. EXAMPLE DAMAGE LOCATION SHOWN, OTHER DAMAGE LOCATIONS SIMILAR.

4 MAKE SURE THERE IS A MINIMUM OF 1.5D EDGE MARGIN.

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Conductive Strip Repair
Figure 201

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IDENTIFICATION 1 - WING FIXED TRAILING EDGE STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

G79005 S0006594803_V2

Wing Fixed Trailing Edge Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
115A0001	Functional Product Collector, Fixed Trailing Edge - Left Wing
115A0002	Functional Product Collector, Fixed Trailing Edge - Right Wing
115A0003	Functional Product Collector, Fixed Trailing Edge to Rear Spar - Left Wing
115A0004	Functional Product Collector, Fixed Trailing Edge to Rear Spar - Right Wing
115A2000	Upper Installation - Inboard Fixed Trailing Edge
115A2001	Lower Installation - Inboard Fixed Trailing Edge
115A2500	Panel Installation - Inboard Fixed Trailing Edge

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
115A3700	Panel Installation - Midspan Fixed Trailing Edge, Lower
115A4900	Structure Installation - Panels and Supports, Outboard Fixed Trailing Edge
115A6200	Rib Installation - Deflection Control, Midspan Fixed Trailing Edge
115A8100	Rib Installation - WBL 413.18 and 419.75, Outboard Fixed Trailing Edge
115A8300	Support Installation - Aileron Hinge, Outboard Fixed Trailing Edge

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1300376 S0000213362_V1

Inboard Fixed Trailing Edge Structure
Figure 2

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U77623 S0000213471_V1

Upper Inboard Fixed Trailing Edge Structure
Figure 3

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LIST OF MATERIALS FOR FIGURE 3			
ITEM	DESCRIPTION	T*[1]	MATERIAL
[1]	Panel Support Bracket (2)		7050-7451 machined plate as given in AMS 4050. Grain direction controlled part.
[2]	Panel Support J Beam (2)		BAC1505-100769 extrusion as given in QQ-A-200/11.
[3]	Panel Support Beam		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[4]	Panel Support Angle	0.050 (1.27)	2024-T3 bare sheet as given in QQ-A-250/4.
[5]	Panel Support Fitting		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[6]	Panel Stiffener (2)		AND10134-1004 7075-T62 extrusion as given in QQ-A-200/11.
[7]	Panel Support Bracket		7075-T7351 machined plate as given in QQ-A-250/12. Grain direction controlled part.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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U77702 S0000213506_V1

Inboard Fixed Trailing Edge Structure
Figure 4

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LIST OF MATERIALS FOR FIGURE 4			
ITEM	DESCRIPTION	T*[1]	MATERIAL
[1]	Support Beam		Ti-6Al-4V machined plate as given in Mil-T-9046, Code AB-1, in the annealed condition.
[2]	Systems Tray		BAC1505-101134 7075-T6511 extrusion as given in QQ-A-200/11.
[3]	Systems Tray		BAC1503-100093 7075-T6511 extrusion as given in QQ-A-200/11.
[4]	Deflection Control Rib		7050-T7451 machined plate as given in AMS 4050.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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U78639 S0000213482_V1

Lower Inboard Fixed Trailing Edge Structure
Figure 5

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LIST OF MATERIALS FOR FIGURE 5				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[1]	Panel Support Beam		BAC1505-101301 7075-T73511 extrusion as given in QQ-A-200/11.	
[2]	Panel Support Beam (3)		BAC1510-1335 7075-T73511 extrusion as given in QQ-A-200/11.	
[3]	Panel Support Beam		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.	
[4]	Panel Support		BAC1514–918 7075-T73511 extrusion as given in QQ-A-200/11.	
[5]	Panel Support Beam		BAC1506-3876 7075-T73511 extrusion as given in QQ-A-200/11. Optional: 7050-T7451 machined plate as given in BMS 7-323.	LINE NUMBER 1 TO 1980
[5]	Panel Support Beam		7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part.	LINE NUMBER 1981 AND ON
[6]	Panel Support		7075-T7351 machined plate as given in QQ-A-250/12. Grain direction controlled part.	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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Midspan Fixed Trailing Edge Structure
Figure 6

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LIST OF MATERIALS FOR FIGURE 6			
ITEM	DESCRIPTION	T*[1]	MATERIAL
[1]	Seal Bulkhead		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[2]	Lower Panel Support Beam (2)		BAC1506–1489 7075-T73511 extrusion as given in QQ-A-200/11.
[3]	Lower Panel Support Beam		BAC1509–100590 7075-T73511 extrusion as given in QQ-A-200/11.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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U78618 S0000213509_V1

Outboard Fixed Trailing Edge Structure
Figure 7

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LIST OF MATERIALS FOR FIGURE 7			
ITEM	DESCRIPTION	T*[1]	MATERIAL
[1]	Rib Assembly		
	Rib (2)		7050-T7451 machined plate as given in AMS 4050.
	Angle	0.040 (1.0)	2024-T3 clad sheet as given in QQ-A-250/5.
	Intercostal	0.040 (1.0)	2024-T3 clad sheet as given in QQ-A-250/5.
[2]	Lower Panel Support Beam		7050-T7451 machined plate as given in AMS 4050.
[3]	Panel Support Angle		7050-T7451 machined plate as given in AMS 4050.
[4]	Aileron Hinge Support Assembly		
	Rib		7050-T7451 machined plate as given in BMS 7-323. Grain direction controlled part.
	Bearing Link		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[5]	Aileron Hinge Support Assembly		
	Rib		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
	Bearing Link		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[6]	Aileron Tab Control Rod Support Assembly		
	Rib		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
	Clevis		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[7]	Panel Support Rib		
	Upper Chord		7075-T7451 machined plate as given in QQ-A-250/12. Grain direction controlled part.
	Lower Chord		7075-T7451 machined plate as given in QQ-A-250/12. Grain direction controlled part.
	Stiffener (2)	0.040 (1.0)	2024-T3 clad sheet as given in QQ-A-250/5.
	Web	0.040 (1.0)	2024-T3 clad sheet as given in QQ-A-250/5.
[8]	Panel Support Rib		
	Upper Chord		7075-T7451 machined plate as given in QQ-A-250/12. Grain direction controlled part.
	Lower Chord		AND10139-2405 7075-T73511 extrusion as given in QQ-A-200/11.
	Stiffener (2)	0.040 (1.0)	2024-T3 clad sheet as given in QQ-A-250/5.
	Web	0.040 (1.0)	2024-T3 clad sheet as given in QQ-A-250/5.
[9]	Panel Support Beam		BAC1505-29542 7075-T73 extrusion as given in QQ-A-200/11.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING FIXED TRAILING EDGE STRUCTURE

1. Applicability

A. Allowable Damage 1 is applicable to damage on the wing fixed trailing edge structure shown in Wing Fixed Trailing Edge Location, Figure 101/ALLOWABLE DAMAGE 1.

F55046 S0006594809_V2

Wing Fixed Trailing Edge Location
Figure 101

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1303108 S0000225324_V1

Inboard Fixed Trailing Edge Structure
Figure 102

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1300518 S0000221590_V1

Upper Inboard Fixed Trailing Edge Structure
Figure 103

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1300659 S0000221594_V1

Inboard Fixed Trailing Edge Structure
Figure 104

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1300693 S0000221596_V1

Lower Inboard Fixed Trailing Edge Structure
Figure 105

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Midspan Wing Fixed Trailing Edge Structure
Figure 106

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1300715 S0000221601_V1

Outboard Wing Fixed Trailing Edge Structure
Figure 107

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limit references that are applicable to each type of structure.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	PARAGRAPH
PANEL SUPPORT RIBS	4.A
PANEL SUPPORT STRUCTURE (DOES NOT INCLUDE PANEL SUPPORT RIBS)	4.B
AILERON SUPPORT RIBS	4.C
DEFLECTION CONTROL RIBS	4.D

C. Remove the damage.

(1) Refer to 51-10-02, GENERAL for the investigation and cleanup procedures.

(2) Refer to 51-30-03, GENERAL for possible sources of the abrasive and other materials you can use to remove the damage.

(3) Refer to 51-30-05, GENERAL for possible sources of the equipment and tools you can use to remove the damage.

D. After you remove the damage, do as follows:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the parts given in Table 102/ALLOWABLE DAMAGE 1.

(a) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 102:

PARTS THAT YOU MUST SHOT PEEN OR FLAP PEEN AFTER YOU REMOVE THE DAMAGE	
PART DRAWING NUMBER	PART NAME
115A6250	Rib Assembly - Deflection Control - Midspan
115A6252	Clevis Fitting Assembly - Deflection Control - Midspan

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01, GENERAL.

(3) Apply two layers of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-44-04.

3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-06, GENERAL	Shot Peening
51-30-03, GENERAL	Sources for Non-Metallic Repair Materials

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(Continued)

Reference	Title
51-30-05, GENERAL	Equipment and Tools For Repairs
51-40-05, GENERAL	Fastener Hole Sizes
57-51-02, REPAIR 1	Wing Fixed Trailing Edge Support Beam
SOPM 20-10-03	SHOT PEENING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Panel Support Ribs

(1) Rib Chords

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Rib Webs

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A and E .

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A , C , D , and E.

2) Damage that does not go though the clad surface is permitted.

(c) Dents are not permitted.

(d) Holes and Punctures are permitted if they are:

1) A minimum of 1.0 inch away from the edge of a fastener, a hole, other damage, or material edge.

2) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

a) Use a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

b) Drill the holes to the applicable size as given in 51-40-05, GENERAL.

(3) Rib Stiffeners

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A , B , C , and D.

2) Damage that does not go through the clad surface is permitted.

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(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

B. Panel Support Structure (Does not include Panel Support Ribs)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A and B .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Aileron Support Ribs

(1) Rib Chord Areas

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Rib Web Areas

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Detail C.

(c) Dents are permitted as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail F.

(d) Holes and Punctures are permitted if they are:

1) A minimum of 1.0 inch away from the edge of a fastener, a hole, other damage, or material edge

2) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

a) Use a 2117-T3 or 2117-T4 rivet. Install the rivet without sealant.

b) Drill the holes to the applicable size as given in 51-40-05, GENERAL.

D. Deflection Control Ribs

(1) Rib Chord Areas

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A and B.

(b) Nicks, Gouges, Scratches, and Corrosion:

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1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Rib Web Areas

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 108/ ALLOWABLE DAMAGE 1, Detail C.

(c) Dents are permitted as shown in Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail F.

(d) Holes and Punctures are permitted if they are:

1) A minimum of 1.0 inch away from the edge of a fastener hole, other damage, or material edge

2) Not more than 0.25 inch in diameter before they are drilled and filled with a 1/4 inch diameter or smaller rivet.

a) Use an aluminum rivet installed without sealant.

b) Drill the holes to the applicable size as given in 51-40-05, GENERAL.

(3) Rib Upper Flange

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

NOTE: If you have done 57-51-02, REPAIR 1, then Paragraph 4.D.(3)(b)1)/ALLOWABLE DAMAGE 1 does not apply. Use Paragraph 4.D.(3)(b)2)/ALLOWABLE DAMAGE 1

1) The damage is permitted as given in Figure 108/ALLOWABLE DAMAGE 1, Detail G if:

a) You remove the initial BACB30NM3K8 bolt, BACB30LE3DK bolt or BACB30US3K10D bolt, as applicable.

b) You do not blend more than 0.04 inch (1.02 mm) thickness from the upper vertical flange.

c) You install either a BACW10P440CG or BACW10P461CG washer as shown in Figure 108 (Sheet 4).

d) You install a new BACB30LE3DK8 bolt as given in drawing 115A6200 for airplane line numbers 1-1980 that have not completed Service Bulletin 737-57-1284. You install a new BACB30US3K10D bolt as given in Service Bulletin 737-57-1284 for airplane line numbers 1-1980 that have completed Service Bulletin 737-57-1284 or as given in drawing 115A2000 for airplane line numbers 1981 and on.

2) If you have done 57-51-02, REPAIR 1, the damage is permitted as given in Figure 108/ALLOWABLE DAMAGE 1, Detail G if:

a) You remove the initial BACB30US4K( )DM bolts.

b) You do not blend more than 0.04 inch (1.02 mm) thickness from the upper vertical flange.

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c) You re-install the BACW10BP4CD or BACW10P453CG washers below the bolt heads and the BACW10BP4AP or BACW10BP4DP washers below the nuts as shown in 57-51-02, REPAIR 1.

d) You re-install the BACB30US4K( )DM bolts as shown in 57-51-02, REPAIR 1.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

A

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

B

F54966 S0006594815_V1

Allowable Damage Limits
Figure 108 (Sheet 1 of 4)

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Allowable Damage Limits
Figure 108 (Sheet 2 of 4)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL
ON AN EDGE OF A WEB

E

DENT THAT IS PERMITTED

F

F54972 S0006594817_V1

Allowable Damage Limits
Figure 108 (Sheet 3 of 4)

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UPPER FLANGE DAMAGE THAT IS PERMITTED

G

A–A

NOTES

1 INSTALL A BACB30LE3DK8 BOLT FOR AIRPLANE LINE NUMBERS 1-1980 WITHOUT SB 737-57-1284 INCORPORATED. INSTALL A BACB30US3K10D BOLT FOR AIRPLANE LINE NUMBERS 1-1980 WITH SB 737-57-1284 INCORPORATED AND AIRPLANES WITH LINE NUMBERS 1981 AND ON. THE DIRECTION OF THE HEAD OF THE BOLT IS OPTIONAL.

433509 S0000141073_V2

Allowable Damage Limits
Figure 108 (Sheet 4 of 4)

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REPAIR 1 - WING FIXED TRAILING EDGE SUPPORT BEAM

1. Applicability

A. Repair 1 is applicable to damage to the wing fixed trailing edge support beam as shown in Figure 201.

464234 S0000142386_V2

Wing Fixed Trailing Edge Location
Figure 201

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NOTES

- NOT ALL OF THE FIXED TRAILING EDGE STRUCTURE IS SHOWN.

464966 S0000142387_V1

Wing Inboard Trailing Edge Structure
Figure 202

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2. General

A. Repair 1 is a permanent repair. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-40-02, GENERAL	Fastener Installation and Removal
51-40-03, GENERAL	Fastener Substitution
57-51-02, ALLOWABLE DAMAGE 1	Wing Fixed Trailing Edge Structure
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating

4. Repair Instructions

A. Trim out and remove all damage in the trailing edge upper support beam as shown in Figure 203 to a maximum slot of 1.20 inch (30.48 mm) by 0.56 inch (14.22 mm). Make sure that you use a minimum trim out radius of 0.11 inch (2.79 mm).

(1) Make sure that you can install the 0.25 inch (6.35 mm) diameter BACB30US4K( )DM bolt through all slots and holes as given in Paragraph 4.J.(1)/REPAIR 1. If necessary, increase the dimension of the slots or holes from a minimum of 0.266 inch (6.76 mm) to a maximum of 0.277 inch (7.04 mm).

B. Make sure that all of the damage is removed. Refer to 51-10-02, GENERAL for the removal of damage procedures, and to 57-51-02, ALLOWABLE DAMAGE 1 for damage to the deflection control rib.

C. Make the part [1] repair doubler as shown in Figure 203. Refer to Table 201 for the repair material.

(1) For airplanes with line numbers 1 through 1980 that have not incorporated SB 737-57-1284, make the thickness of the part [1] repair doubler a minimum of 0.30 in. (7.62 mm) thick.

(2) For airplanes with line numbers 1 through 1980 that have incorporated SB 737-57-1284 and airplane line numbers 1981 and on, make the thickness of the part [1] repair doubler a minimum of 0.15 in. (3.81 mm) thick.

Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Doubler	1	Use 15-5PH CRES as given in BMS 7-240 or AMS 5862 with a heat treat of 180 to 200KSI.

D. Assemble the part [1] repair doubler and drill the fastener holes as shown in Figure 203.

E. Disassemble the part [1] repair doubler.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair part and to the bare surfaces of the initial structure.

G. Restore the initial structure finish as given in 51-20-01, GENERAL.

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H. Finish the part [1] repair doubler with:

(1) CAD plate as given in SOPM 20-42-05.

(2) One layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel. Refer to SOPM 20-41-02.

I. For airplanes with line numbers 1 through 1980 that have not incorporated SB 737-57-1284, do the following:

(1) Apply a chemical conversion coating to the solid aluminum shim as given in Figure 203, Sheet 3. Refer to 51-20-01, GENERAL.

(2) Finish the solid aluminum shim as given in Figure 203, Sheet 3 with one layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel. Refer to SOPM 20-41-02.

J. Fay surface seal between the part [1] repair doubler and the mating surfaces with BMS 5-95 sealant. Install the part [1] repair doubler and fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL. Make sure there is a 0.005 inch (0.13 mm) or less gap prior to fastener installation. Provide an edge seal around the repair slot in the support beam with BMS 5-95 sealant. 51-20-05, GENERAL.

(1) Make sure you install the 0.25 inch (6.35 mm) diameter BACB30US4K()DM or equivalent as shown in Figure 203.

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465573 S0000142388_V1

Inboard Fixed Trailing Edge Support Beam Repair
Figure 203 (Sheet 1 of 6)

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A-A

NOTES

1. FOR AIRPLANES WITH LINE NUMBERS 1-1980 WITHOUT SB 737-57-1284 INCORPORATED.
2. FOR AIRPLANES WITH LINE NUMBERS 1-1980 WITH SB 737-57-1284 INCORPORATED AND AIRPLANES WITH LINE NUMBERS 1981 AND ON.
3. THE DIRECTION OF THE HEAD OF THE BOLT IS OPTIONAL. MAKE SURE THAT YOU INSTALL THE WASHERS BELOW THE BOLT HEAD AND NUT AS SHOWN.
4. WHEN YOU INSTALL THE PART [1] DOUBLER, MAKE SURE THAT YOU ALIGN THE PART [1] DOUBLER SLOT WITH HOLE LOCATIONS IN THE SHIM, THE DEFLECTION CONTROL RIB AND THE SERRATED PLATES.
5. YOU CAN COUNTERSINK INTO THE SERRATION, BUT MAKE SURE THAT YOU REMOVE BURRS AND SHARP EDGES. MAKE SURE THAT YOU ALSO RESTORE THE FINISH AND THAT THE SERRATIONS ATTACH ACCURATELY WITH THE MATING PART.

D57608 S0000159770_V4

Inboard Fixed Trailing Edge Support Beam Repair
Figure 203 (Sheet 2 of 6)

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2396753 S0000553516_V1

Inboard Fixed Trailing Edge Support Beam Repair
Figure 203 (Sheet 3 of 6)

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VIEW LOOKING INBOARD

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

C-C 2

2396883 S0000553655_V1

Inboard Fixed Trailing Edge Support Beam Repair
Figure 203 (Sheet 4 of 6)

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PART [1] DOUBLER 1

B

D–D 1

466896 S0000142389_V7

Inboard Fixed Trailing Edge Support Beam Repair
Figure 203 (Sheet 5 of 6)

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Inboard Fixed Trailing Edge Support Beam Repair
Figure 203 (Sheet 6 of 6)

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REPAIR 2 - WING INBOARD FIXED TRAILING EDGE DEFLECTION CONTROL RIB TURNBUCKLE LUG REPAIR

1. Applicability

A. This repair is applicable to damage on the turnbuckle lug of the wing inboard fixed trailing edge deflection control rib as shown in Figure 201/REPAIR 2.

B. This repair is not applicable to damage on the turnbuckle lug that extend into the primary chords of the wing inboard fixed trailing edge deflection control rib.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

NOTE

- NOT ALL OF THE FIXED TRAILING EDGE STRUCTURE IS SHOWN.

1959561 S0000374166_V1

Wing Inboard Trailing Edge Structure
Figure 201

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2. General

A. Repair 2 is a permanent repair. Refer to 51-00-06, GENERAL for the definitions of different categories of repairs.

B. D = fastener diameter

C. All dimensions are in inches, unless they are shown differently.

D. Make sure the cut edges of the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

E. Make sure the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-20-06, GENERAL	Shot Peening
51-20-13, GENERAL	Surface Roughness Finish Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
AMM 57-50	WING TRAILING EDGE
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-50-03	Bearing and Bushing Replacement
737 NDT Part 6, 51-00-00, Procedure 23	Aluminum Part Surface Inspection (Impedance Plane Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

A. Get access to the damaged area. Remove the turnbuckle and other structure as necessary to do the repair. Refer to AMM SECTION 57-50.

B. Remove the necessary fasteners. Refer to 51-40-02, GENERAL.

C. Trim out and remove the damaged turnbuckle lug completely from the deflection control rib as shown in Figure 202/REPAIR 2. Refer to 51-10-02, GENERAL.

(1) Make sure to keep the initial thickness of the deflection control rib flanges.

(2) Make sure that the area where turnbuckle lug was removed is flush with the deflection control rib flanges within +0.003/-0.000 inch (+0.076/-0.000 mm).

D. Do a dye penetrant inspection of the reworked areas to make sure that there is no other damage. Refer to SOPM 20-20-02.

NOTE: As an alternative, you can do a High Frequency Eddy Current (HFEC) inspection along the trimmed-out area to make sure that there is no more damage. Refer to 737 NDT Part 6, 51-00-00, Procedure 4 or 737 NDT Part 6, 51-00-00, Procedure 23.

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WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

E. Flap peen or shot peen the reworked areas of the deflection control rib.

(1) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(2) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(3) Use coverage 2.0.

F. Make the repair parts as shown in Figure 202/REPAIR 2. Refer to Table 201/REPAIR 2 for the repair material.

Table 201: Repair Material

ITEM	PART	QUANTITY	MATERIAL
[1]	Fitting	1	1.5 in. (38.1 mm) thick maximum 7075-T7351 plate as given in AMS 4078 or AMS-QQ-A-250/12 (Optional: 7075-T651 plate as given in AMS 4050 or AMS-QQ-A-250/12 and heat treat to T7351 condition after you form the part)
[2]	Radius Filler	6	0.125 in. (3.175 mm) thick 7075-T6 sheet as given in AMS 4050 or AMS-QQ-A-250/12

G. Do a dye penetrant inspection of the part [1] fitting to make sure that there is no damage. Refer to SOPM 20-20-02.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

H. Flap peen or shot peen the part [1] fitting.

(1) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(2) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(3) Use coverage 2.0.

I. Assemble the repair parts as shown in Figure 202/REPAIR 2.

(1) Clamp the part [1] fitting and the part [2] radius filler on one flange of the deflection control rib. Use feeler gages to check to see if there is any gap on the opposite side.

(2) Make sure the gap between the deflection control rib and the part [1] fitting is less than or equal to 0.005 in. (0.127 mm). If the gap is more than 0.005 in. (0.127 mm), remove the first repair part, and make the new part [1] fitting.

(3) Clamp both flanges of the part [1] fitting and part [2] radius fillers.

J. Drill the fastener holes as shown in Figure 202/REPAIR 2. Keep a minimum of 0.40 in. (10.16 mm) edge margin on the part [1] fitting . Fastener holes to be transition fit types. Refer to 51-40-05, GENERAL.

K. Unclamp and disassemble the repair parts.

L. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial part. Refer to 51-10-02, GENERAL.

M. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial part. Refer to 51-20-01, GENERAL.

N. Apply one layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel to the repair parts and the bare surfaces of the initial parts. Refer to SOPM 20-41-02. Make sure there are no primer and enamel in the fastener holes.

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O. Install a BACN28X4C012 bushing wet in the part [1] fitting with BMS 5-95 sealant. Make sure that the final installation has nominal interference fit of 0.2%. Refer to SOPM 20-50-03.

P. Install the repair parts with BMS 5-95 sealant between the mating surfaces as shown in Figure 202/REPAIR 2. Refer to 51-20-05, GENERAL.

Q. Install the fasteners wet with BMS 5-95 sealant. Refer to 51-40-02, GENERAL and 51-20-05, GENERAL.

R. Apply a fillet seal and fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

S. Apply a decorative finish if necessary. Refer to AMM SECTION 51-21.

T. Install the turnbuckle assembly and other structures that were removed back to serviceable condition. Refer to Figure 202/REPAIR 2.

NOTE: If the initial lug location is change, adjust the turnbuckle length. Refer to AMM SECTION 57-50.

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1959679 S0000374168_V1

Wing Inboard Trailing Edge Deflection Control Rib Turnbuckle Lug Repair
Figure 202 (Sheet 1 of 4)

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1960916 S0000374176_V1

Wing Inboard Trailing Edge Deflection Control Rib Turnbuckle Lug Repair
Figure 202 (Sheet 2 of 4)

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A–A

B–B

1961146 S0000374177_V1

Wing Inboard Trailing Edge Deflection Control Rib Turnbuckle Lug Repair
Figure 202 (Sheet 3 of 4)

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C-C

1 MAKE SURE THAT THE DIAMETER OF THE HOLE IN THE PART [1] FITTING IS 0.3747 TO 0.3754 INCH AND THE HOLE HAS A SURFACE SMOOTHNESS OF 63 MICROINCHES RA OR SMOOTHER. REFER TO SRM 51-20-13.

2 INSTALL BOLT WITH BMS 3-27.

FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A BACB30NX6K() HEX-DRIVE BOLT WITH A BACC30BH6 COLLAR IN TRANSITION FIT HOLES. PUT FASTENER HEADS ON THE PART [1] FITTING SIDE.

REPAIR FASTENER LOCATION. INSTALL A BACB30NX5K() HEX-DRIVE BOLT, NAS1149DN832J WASHER AND BACN10YR5CD NUT IN TRANSITION FIT HOLES. PUT FASTENER HEADS ON THE PART [1] FITTING SIDE.

1961046 S0000374178_V1

Wing Inboard Trailing Edge Deflection Control Rib Turnbuckle Lug Repair
Figure 202 (Sheet 4 of 4)

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REPAIR 3 - WING INBOARD FIXED TRAILING EDGE DEFLECTION CONTROL RIB UPPER TAB CRACK DAMAGE REPAIR

1. Applicability

A. This repair is applicable to crack damage on the upper tab of the wing inboard fixed trailing edge deflection control rib as shown in Figure 201/REPAIR 3.

B. This repair is not applicable to damage that extend into the primary chords of the wing inboard fixed trailing edge deflection control rib.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

NOTE

- NOT ALL OF THE FIXED TRAILING EDGE STRUCTURE IS SHOWN.

2392018 S0000549147_V1

Wing Fixed Trailing Edge Location
Figure 201

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2. General

A. This repair is a permanent repair. Refer to 51-00-06, GENERAL for the definitions of different categories of repairs.

B. All dimensions are in inches, unless they are shown differently.

C. D = fastener diameter.

D. Make sure the cut edges of the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

E. Make sure the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-20-06, GENERAL	Shot Peening
51-20-13, GENERAL	Surface Roughness Finish Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
AMM 57-50	WING TRAILING EDGE
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
737 NDT Part 6, 51-00-00, Procedure 23	Aluminum Part Surface Inspection (Impedance Plane Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

A. Get access to the damaged area. Remove other structure as necessary to do the repair. Refer to AMM SECTION 57-50.

B. Remove the bolts that attach the deflection control rib upper tab to the upper support beam through the serrated plates. Refer to 51-40-02, GENERAL.

C. For airplanes with line numbers 1-1980 that have not incorporated SB 737-57-1284, install the repair doubler as given in SRM 57-51-02, REPAIR 1 if the repair has not been done. Remove and discard the shim between the inboard serrated plate and upper support beam.

D. For airplanes with line numbers 1-1980 that have incorporated SB 737-57-1284 and airplanes with line numbers 1981 and on, install the repair doubler as given in SRM 57-51-02, REPAIR 1 if the repair has not been done. Check the gap between the outboard serrated plate and inboard serrated plate.

(1) If the gap between the inboard and outboard serrated plates is less than 0.050 in. (1.270 mm) and there is a shim between the inboard serrated plate and the upper support beam, remove and discard the shim. Reinstall the inboard serrated plate to the upper support beam.

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E. Trim out the damaged upper tab completely from the deflection control rib as shown in Figure 202/REPAIR 3. Refer to 51-10-02, GENERAL.

(1) Make sure to keep the thickness of the initial deflection control rib flanges.

(2) Make sure the area where the upper tab was removed is flush with the deflection control rib flanges within +0.002/-0.000 inch.

F. Discard the initial outboard serrated plate.

G. Do a dye penetrant inspection of the reworked areas to make sure that there is no damage. Refer to SOPM 20-20-02.

NOTE: As an alternative, you can do a High Frequency Eddy Current (HFEC) inspection along the trimmed-out area to make sure that there is no damage. Refer to 737 NDT Part 6, 51-00-00, Procedure 4 or 737 NDT Part 6, 51-00-00, Procedure 23.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

H. Flap peen or shot peen the reworked areas of the deflection control rib.

(1) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(2) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(3) Use coverage 2.0.

I. Make the repair parts as shown in Figure 202/REPAIR 3 and Table 201/REPAIR 3.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Fitting	1	2 in. (51 mm) thick maximum, 7050-T7451 plate as given in AMS 4050.
[2]	Radius Filler	As Necessary	0.125 in. (3.175 mm) thick, 7075-T6 sheet as given in AMS 4045 or AMS-QQ-A-250/12.
[3]	Radius Filler	As Necessary	0.125 in. (3.175 mm) thick, 7075-T6 sheet as given in AMS 4045 or AMS-QQ-A-250/12.
[4]	Shim	As Necessary	0.04 in. (1.02 mm) thick maximum, 7075-T651 sheet as given in AMS 4045 or AMS-QQ-A-250/12. Make the part [4] shim the same shape as the initial outboard serrated plate.
[5]	Outboard Serrated Plate	1	Procure the Serrated Plate with Part Number 115A2183-1 from the Boeing Company.

J. Do a dye penetrant inspection of the part [1] fitting to make sure that there is no damage. Refer to SOPM 20-20-02.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

K. Flap peen or shot peen the part [1] fitting.

(1) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(2) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(3) Use coverage 2.0. Make sure that shot peening is fully effective at the fillet radii.

L. Assemble the initial and the repair parts. Refer to Figure 202/REPAIR 3.

(1) Clamp the part [1] fitting on one flange of the deflection control rib. Use the feeler gages to check if there is any gap on the opposite side.

(2) Make sure that the gap between the deflection control rib and the part [1] fitting is less than or equal to 0.002 in. (0.051 mm).

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(3) Make sure that there is sufficient space between the part [1] fitting and the initial inboard serrated plate to install the part [5] outboard serrated plate.

(4) Make sure that the centerline of the part [1] fitting is located no more than 0.08 in. (2.03 mm) inboard or outboard direction from the centerline of the deflection control rib to remove any gap or preload between the serrated plates.

(a) Make sure there is fastener head clearance to install the BACB30VT5K() hex-drive bolts.

(b) If necessary, it is acceptable to make the additional part [2] radius filler and part [3] radius filler to fit under the BACB30VT5K() hex-drive bolt heads.

(5) If there is still a gap between the inboard and part [5] outboard serrated plates, make the part [4] shim with the thickness equal to the gap measurement within 0.000/-0.005 inch. The part [4] shim is located between the part [1] fitting and part [5] outboard serrated plate.

(6) Clamp both flanges of the part [1] fitting, part [2] radius fillers, and part [3] radius fillers on the deflection control rib.

M. Drill the fastener holes as shown in Figure 202/REPAIR 3. Refer to 51-40-05, GENERAL.

(1) Do not slot the two BACB30US4K() bolt holes in the vertical flange of the part [1] fitting, part [4] shim (if used), and part [5] outboard serrated plate. Drill the two BACB30US4K() bolt holes to 0.250 in. (6.350 mm)-0.277 in. (7.036 mm) diameter and make sure the bolt holes align with the slot in the inboard serrated plate and upper support beam.

(2) Drill the two BACB30NW6K() transition fit holes to 0.187 in. (4.750 mm)-0.190 in. (4.826 mm) diameter in the vertical flange of the part [1] fitting, part [4] shim, and part [5] outboard serrated plate. Countersink the two BACB30NW6K() fastener holes of the part [5] outboard serrated plate. Refer to 51-40-08, GENERAL.

(3) Drill the transition fit holes to 0.161 in. (4.089 mm)-0.164 in. (4.166 mm) diameter in the horizontal flange of the part [1] fitting, part [2] radius fillers, part [3] radius fillers, and the deflection control rib.

(4) Keep a minimum of 0.25 in. (6.35 mm) fastener edge margin for the aft pair of fasteners in the deflection control rib.

(5) Keep a minimum of 0.35 in. (8.89 mm) fastener edge margin on all other fastener hole locations on the deflection control rib and part [1] fitting.

(6) Spotface the part [2] and part [3] radius fillers so the nuts and washers will seat correctly. Spotface should be 0.03 in. (0.76 mm) deep maximum, with a minimum of 0.06 in. (1.52 mm) fillet radius.

N. Unclamp and disassemble the repair parts.

O. Fastener holes maybe opened up on the part [2] radius fillers and part [3] radius fillers only to class 1 clearance fit to facilitate the fastener installation later. Refer to 51-40-05, GENERAL.

P. Deburr the fastener holes. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts. Refer to 51-10-02, GENERAL.

Q. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the initial parts. Refer to 51-20-01, GENERAL.

R. Mask the fastener holes to make sure there are no primer and enamel in the fastener holes. Apply one layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel to the repair parts and the bare surfaces of the initial parts. Refer to SOPM 20-41-02.

S. Install the repair parts with BMS 5-95 sealant between the mating surfaces as shown in Figure 202/REPAIR 3. Refer to 51-20-05, GENERAL.

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T. Install the fasteners wet with BMS 5-95 sealant. Refer to 51-40-02, GENERAL and 51-20-05, GENERAL.

(1) Install the part [5] outboard serrated plate, part [4] shim (if used), and part [1] fitting. Refer to Figure 202/REPAIR 3, Section B-B.

(2) Install the remaining repair parts. Refer to Figure 202/REPAIR 3.

U. Reattach the deflection control rib to the upper support beam with REPAIR 1 concurrently.

V. Apply a fillet seal and fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

W. Apply a decorative finish as necessary. Refer to AMM SECTION 51-21.

X. Install the structure that were removed back to a serviceable condition. Refer to AMM SECTION 57-50.

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2392109 S0000550845_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 1 of 8)

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NOTES

1. FOR AIRPLANES WITH LINE NUMBERS 1-1980 WITHOUT SB 737-57-1284 INCORPORATED.
2. FOR AIRPLANES WITH LINE NUMBERS 1-1980 WITH SB 737-57-1284 INCORPORATED AND AIRPLANES WITH LINE NUMBERS 1981 AND ON.
3. THE HEAD OF THE BACB30US4K() BOLT IS ON EITHER THE DOUBLER OR THE PART [1] FITTING.
4. INSTALL THE BOLT WITH BMS 3-38.

FASTENER SYMBOLS

REPAIR FASTENER LOCATION. COUNTERSINK THE PART [5] OUTBOARD SERRATED PLATE FOR A BACB30NW6K() HEX-DRIVE BOLT. INSTALL A BACB30NW6K() HEX-DRIVE BOLT THROUGH THE PART [5] OUTBOARD SERRATED PLATE, PART [4] SHIM (IF USED), AND PART [1] FITTING WITH A BACC30M6 COLLAR IN TRANSITION FIT HOLE. PUT THE FASTENER HEAD ON THE PART [5] OUTBOARD SERRATED PLATE SIDE.

REPAIR FASTENER LOCATION. INSTALL A BACB30VT5K() HEX-DRIVE BOLT WITH A NAS1149DN832K WASHER, AND A BACN10YR08CD NUT IN TRANSITION FIT HOLE. PUT THE FASTENER HEAD ON THE PART [1] FITTING SIDE.

INITIAL FASTENER LOCATION. INSTALL A BACB30VT6K() HEX-DRIVE BOLT THROUGH THE UPPER SUPPORT BEAM, THE INITIAL SHIM (IF IT IS NOT REMOVED), AND THE INBOARD SERRATED PLATE WITH A BACC30BL6 COLLAR. PUT THE FASTENER HEAD ON THE UPPER SUPPORT BEAM SIDE.

2392115 S0000550846_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 2 of 8)

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B

2392331 S0000550847_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 3 of 8)

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 4 of 8)

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(THE UPPER SUPPORT BEAM IS NOT SHOWN FOR CLARITY)

(THE INBOARD SERRATED PLATE, THE SHIM (IF IT IS NOT REMOVED), AND THE UPPER SUPPORT BEAM ARE NOT SHOWN FOR CLARITY)

2392713 S0000550848_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 5 of 8)

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2402181 S0000555415_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 6 of 8)

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 7 of 8)

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Crack Repair
Figure 202 (Sheet 8 of 8)

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REPAIR 4 - WING INBOARD FIXED TRAILING EDGE DEFLECTION CONTROL RIB UPPER TAB WEAR DAMAGE REPAIR

1. Applicability

A. This repair is applicable to wear damage on the wing fixed trailing edge deflection control rib as shown in Figure 201/REPAIR 4.

B. This repair is not applicable to wear damage that extend beyond the repair cutout in the upper tab as shown in Figure 202/REPAIR 4.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

NOTE

- NOT ALL OF THE FIXED TRAILING EDGE STRUCTURE IS SHOWN.

2397250 S0000553713_V1

Wing Fixed Trailing Edge Location
Figure 201

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2. General

A. This repair is a permanent repair. Refer to 51-00-06, GENERAL for the definitions of different categories of repairs.

B. Do the Repair 4 at the same time with the SRM 57-51-02 Repair 1.

C. All dimensions are in inches, unless they are shown differently.

D. D = fastener diameter.

E. Make sure the cut edges of the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

F. Make sure the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-20-13, GENERAL	Surface Roughness Finish Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
AMM 57-50	WING TRAILING EDGE
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating
737 NDT Part 6, 51-00-00, Procedure 16	Aluminum Part Fastener Hole Inspection (Rotary Scanner)
737 NDT Part 6, 51-00-00, Procedure 23	Aluminum Part Surface Inspection (Impedance Plane Display)
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

A. Get access to the damaged area. Remove other structure as necessary to do the repair. Refer to AMM SECTION 57-50.

B. Remove the bolts that attach the deflection control rib upper tab to the upper support beam through the serrated plates. Refer to 51-40-02, GENERAL.

C. For airplanes with line numbers 1-1980 that have not incorporated SB 737-57-1284, do the repair as given in SRM 57-51-02 Repair 1, Figure 203, Flagnote 1 at the same time with this repair 4 if the repair 1 has not been done.

D. For airplanes with line numbers 1-1980 that have incorporated SB 737-57-1284 and airplanes with line numbers 1981 and on, do the repair as given in SRM 57-51-02 Repair 1, Figure 203, Flagnote 2 at the same time with this repair 4 if the repair 1 has not been done.

E. Cut out and remove the damaged part of the deflection control rib upper tab as shown in Figure 202/ REPAIR 4. Refer to 51-10-02, GENERAL.

F. Do a High Frequency Eddy Current (HFEC) inspection along the cut edges to make sure that there is no damage. Refer to 737 NDT Part 6, 51-00-00, Procedure 4 or 737 NDT Part 6, 51-00-00, Procedure 23.

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G. Do an open hole HFEC inspection of the initial fastener holes in the repair area to make sure that there are no cracks. Refer to 737 NDT Part 6, 51-00-00, Procedure 16.

H. Do a detailed visual inspection on the serrations of the initial outboard and/or inboard serrated plates. If there is damage on the serration of the initial outboard and/or inboard serrated plate, discard the damaged serrated plate and make the part [3] outboard serrated plate and/or part [4] inboard serrated plate. Refer to 51-10-02, GENERAL and Table 201/REPAIR 4.

I. Make the repair parts as shown in Figure 202/REPAIR 4 and Table 201/REPAIR 4.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Outboard Doubler*[1]	1	0.10 in. (2.54 mm) thick, 15-5PH CRES as given in BMS 7-240 or AMS 5862. Heat treat to 180 to 200 KSI.
[2]	Filler*[2]	1	0.10 in. (2.54 mm) thick, 7075-T6 clad sheet as given in AMS-QQ-A-250/13 or equivalent.
[3]	Outboard Serrated Plate	1	Procure the outboard serrated plate with part number 115A2183-1 from the Boeing Company.
[4]	Inboard Serrated plate	1	For airplanes with line numbers 1-1980 without SB 737-57-1284 incorporated, procure the inboard serrated plate with part number 115A2183-2 from the Boeing Company. For airplanes with line numbers 1-1980 with SB 737-57-1284 incorporated and line numbers 1981 and on, procure the inboard serrated plate with part number 115A2183-3/-4 from the Boeing Company.

*[1] Make sure the spacing between the two outboard fastener holes match with the spacing of the two outside holes of the deflection control rib upper tab and the outboard serrated plate.

*[2] Make sure the contour of the part [2] filler match the contour of the cutout on the deflection control rib upper tab with a maximum gap of 0.03 in. (0.76 mm) around the perimeter.

J. Assemble the initial and repair parts. Refer to Figure 202/REPAIR 4.

K. Drill the fastener holes as shown in Figure 202/REPAIR 4. Refer to 51-40-05, GENERAL.

(1) If the initial outboard serrated plate is not replaced, do the steps below:

(a) Do not slot the two BACB30US4K() bolt holes common to the part [1] outboard doubler and part [2] filler. Drill the two BACB30US4K() bolt holes of the part [1] doubler and part [2] filler to 0.259 in. (6.579 mm)-0.277 in. (7.036 mm) diameter. Make sure that the holes align with the slots of the outboard and inboard serrated plates, and upper support beam.

(b) Oversize the two initial BACB30NW6K() fastener holes of the initial deflection control rib upper tab and the initial outboard serrated plate to a maximum of 1/32 inch over nominal as necessary. Fastener holes to be close ream types.

(c) Drill the two BACB30NW6K()X or Y fastener holes of the part [1] outboard doubler to match with the two oversized fastener holes of the deflection control rib upper tab and initial outboard serrated plate as shown in Paragraph 4.K.1.(b).

(2) If the initial outboard serrated plate is replaced, do the steps below:

(a) Do not slot the two BACB30US4K() bolt holes common to the part [1] outboard doubler, part [2] filler and part [3] outboard serrated plate. Drill the bolt holes to 0.259 in. (6.579 mm)-0.277 in. (7.036 mm) diameter.

(b) Make sure that the two BACB30US4K() bolt holes on the part [1] outboard doubler, part [2] filler and part [3] outboard serrated plate align with the slots of the inboard serrated plate, shim and upper support beam.

(c) Make sure that the two BACB30US4K() bolt holes on the part [1] outboard doubler, part [2] filler and part [3] outboard serrated plate are located 0.01 in. (0.25 mm) maximum offset above or below the part [1] doubler centerline.

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(d) Oversize the two initial BACB30NW6K() fastener holes on the initial deflection control rib upper tab to a maximum of 1/32 inch over nominal as necessary. Fastener holes to be close ream types. Refer to 51-40-05, GENERAL.

(e) Drill the two BACB30NW6K()X or Y fastener holes of the part [1] outboard doubler and part [3] outboard serrated plate to match with the two oversized fastener holes of the deflection control rib upper tab as shown in Paragraph 4.K.2. (d).

(f) For airplanes with line numbers 1-1980 with SB 737-57-1284 incorporated and airplanes with line numbers 1981 and on, make sure that the serrations of the outboard serrated plate align perfectly with the serrations of the inboard serrated plate.

L. Disassemble the repair parts.

M. Remove all the nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts. Refer to 51-10-02, GENERAL.

N. Apply brush cadmium plating to the part [1] outboard doubler. Refer to SOPM 20-42-05.

O. Apply a chemical conversion coating to the repair parts and the bare surfaces of the initial parts except the part [1] outboard doubler. Refer to 51-20-01, GENERAL.

P. Mask the fastener holes to make sure that there are no primer and enamel in the fastener holes. Apply one layer of BMS 10-11, Type I primer and one layer of BMS 10-11, Type II enamel to the repair parts and the bare surfaces of the initial parts. Refer to SOPM 20-41-02.

Q. Install the repair and the initial parts with BMS 5-95 sealant between the mating surfaces as shown in Figure 202/REPAIR 4. Refer to 51-20-05, GENERAL.

R. Install the fasteners wet with BMS 5-95 sealant. Refer to 51-40-02, GENERAL and 51-20-05, GENERAL.

S. Apply a fillet seal and fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

T. Apply a decorative finish as necessary. Refer to AMM SECTION 51-21.

U. Install the structure that were removed back to a serviceable condition. Refer to AMM SECTION 57-50.

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair
Figure 202 (Sheet 1 of 6)

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NOTES

1. FOR AIRPLANES WITH LINE NUMBERS 1-1980 WITHOUT SB 737-57-1284 INCORPORATED.
2. FOR AIRPLANES WITH LINE NUMBERS 1-1980 WITH SB 737-57-1284 INCORPORATED AND AIRPLANES WITH LINE NUMBERS 1981 AND ON.
3. THE DIRECTION OF THE HEAD OF THE BOLT IS OPTIONAL. MAKE SURE THAT YOU INSTALL THE WASHER UNDER THE BOLT HEAD AND NUT AS SHOWN.
4. INSTALL THE BOLT WITH BMS 3-38.
5. THE SLOT IS FOR THE INITIAL OUTBOARD SERRATED PLATE ONLY.

FASTENER SYMBOLS

INITIAL FASTENER LOCATION. FOR THE 1/64 INCH OVERSIZE HOLE; INSTALL A BACB30NW6K()X OR A BACB30FQ6AK() HEX-DRIVE BOLT WITH A BACC30M6 COLLAR. FOR THE 1/32 INCH OVERSIZE HOLE; INSTALL A BACB30NW6K()Y OR A BACB30KF6AK() HEX-DRIVE BOLT WITH A BACC30R6 COLLAR. THE FASTENER TO BE INSTALLED IN CLOSE REAM HOLE AS GIVEN IN SRM 51-40-05, GENERAL. INSTALL THE HEX-DRIVE BOLT THRU THE PART [1] OUTBOARD DOUBLER, DEFLECTION CONTROL RIB UPPER TAB, INITIAL OUTBOARD SERRATED PLATE OR PART [3] OUTBOARD SERRATED PLATE. COUNTERSINK AND INSTALL THE FASTENER HEAD ON THE INITIAL OUTBOARD SERRATED PLATE OR PART [3] OUTBOARD SERRATED PLATE SIDE.

INITIAL FASTENER LOCATION. INSTALL A BACB30VT6K() HEX-DRIVE BOLT WITH A BACC30BL6 COLLAR. INSTALL THE HEX-DRIVE BOLT THRU THE UPPER SUPPORT BEAM, SHIM AND THE INITIAL INBOARD SERRATED PLATE OR THE PART [4] INBOARD SERRATED PLATE. PUT THE FASTENER HEAD IS ON THE UPPER SUPPORT BEAM SIDE.

2397298 S0000553715_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair
Figure 202 (Sheet 2 of 6)

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2397345 S0000553716_V1

Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair
Figure 202 (Sheet 3 of 6)

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair
Figure 202 (Sheet 4 of 6)

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair
Figure 202 (Sheet 5 of 6)

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Wing Inboard Fixed Trailing Edge Deflection Control Rib Upper Tab Wear Damage Repair
Figure 202 (Sheet 6 of 6)

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IDENTIFICATION 1 - MAIN LANDING GEAR BEAM AND STABILIZER STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

1945798 S0000369947_V1

Main Landing Gear Beam and Stabilizer
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
115A0001	Functional Product Collector, Fixed Trailing Edge - Left Wing
115A0002	Functional Product Collector, Fixed Trailing Edge - Right Wing
115A1100	Main Landing Gear Beam Assembly
115A1500	Hanger Link Assembly

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L06494 S0006594823_V2

Main Landing Gear Beam and Stabilizer Location
Figure 2

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

L06484 S0006594824_V3

Main Landing Gear Beam and Stabilizer Identification
Figure 3

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Table 2:

LIST OF MATERIALS FOR FIGURE 3			
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**
[1]	Main Landing Gear Beam		Ti-6Al-4V forged titanium as given in BMS 7-247, Condition A
[2]	Hinge Fitting - Inboard Spoiler Support - MLG Beam	2.000 (50.8)	7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)
[3]	Support Bracket	1.500 (38.1)	7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)
[4]	Support Bracket,	1.500 (38.1)	7050-T7451 plate as given in AMS 4050 (Grain direction controlled part)
[5]	Spoiler Support Fitting - MLG Beam	4.250 (107.95)	7050-T7451 plate as given in BMS 7-323, Type III
[6]	Spoiler Support Fitting - MLG Beam	5.000 (127)	7050-T7451 plate as given in BMS 7-323, Type III
[7]	Upper Stabilizer Link Support	5.500 (139.7)	Ti-6Al-4V forged titanium block as given in MIL-T-9047, annealed
[8]	Lower Stabilizer Support Fitting	5.250 (133.35)	7050-T7451 plate as given in BMS 7-323, Type I (Grain direction controlled part)
[9]	Upper Stabilizer Link Fitting	2.750 (69.85)	7050-T7451 plate as given in AMS 4050
[10]	Upper Stabilizer System Support Fitting	1.750 (44.45)	7050-T7451 plate as given n AMS 4050 (Grain direction controlled part)
[11]	Lower Stabilizer Link	4.300 (109.22)	7050-T7451 plate as given in BMS 7-323, Type III (Grain direction controlled part)
[12]	Hanger Link		7050–T7451 plate as given in BMS 7–323, Type III (Grain direction controlled part)

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - MAIN LANDING GEAR BEAM INSTALLATION

1. Applicability

A. This subject gives the allowable damage limits for the main landing gear beam, the beam stabilizers, and hanger link as shown in Main Landing Gear Beam Installation, Figure 101/ALLOWABLE DAMAGE 1.

F54137 S0006594828_V2

Main Landing Gear Beam Installation
Figure 101

2. General

A. Remove the parts as necessary to get access to the damaged area.

B. Refer to Main Landing Gear Beam Zones, Figure 102/ALLOWABLE DAMAGE 1 for the allowable damage zones of the beam. Refer to Main Landing Gear Beam Stabilizers, Figure 103/ALLOWABLE DAMAGE 1 for the beam stabilizers, and Figure 104/ALLOWABLE DAMAGE 1 for the hanger link.

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WARNING: SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE ON THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STEAM EXPLOSION.

C. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of abrasives and other materials you can use to remove damage.

(3) Refer to 51-30-05 for possible sources of equipment and tools you can use to remove damage.

(4) Make sure that all the damage has been removed within the limits. Refer to SOPM 20-20-02 for penetrant inspection procedures or 737 NDT Part 6, 51-00-00, Figure 4 for eddy current inspection procedures.

D. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the beam and beam stabilizers.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

1) Use an intensity 0.010A to 0.021A for the beam.

2) Use an intensity of 0.005A to 0.010A for the stabilizers.

3) Use coverage 2.0.

(2) Apply a chemical conversion coating to the bare surfaces of the reworked aluminum parts (stabilizers only). Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the surface of the reworked area. Refer to SOPM 20-41-02.

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MAIN LANDING GEAR BEAM (TITANIUM)

F54275 S0006594829_V1

Main Landing Gear Beam Zones
Figure 102

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UPPER STABILIZER LINK
(ALUMINUM)

LOWER STABILIZER LINK
(ALUMINUM)

F69296 S0006594830_V1

Main Landing Gear Beam Stabilizers
Figure 103

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Main Landing Gear Hanger Link
Figure 104

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
CMM 57-15-01,	Main Landing Gear Beam Installlation Components
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
737 NDT Part 6, 51-00-00	Structures - General
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Allowable Damage Limits

A. Main Landing Gear Beam

(1) Zone 1

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

(2) Zone 2 - Upper Chord

(a) Damage is not permitted.

(3) Zone 2 - Lower Chord, Web and Stiffeners

(a) Cracks are not permitted.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details B and C.

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

B. Stabilizers

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail D .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Details A , B and D.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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C. Hanger Link

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail E.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

(A)

A–A

F54140 S0006594831_V1

Allowable Damage Limits
Figure 105 (Sheet 1 of 4)

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REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

B

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE

C

B–B

F54144 S0006594832_V1

Allowable Damage Limits
Figure 105 (Sheet 2 of 4)

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X = THE WIDTH OF THE MATERIAL REMOVED
= 0.10W MAXIMUM

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

F54376 S0006594833_V1

Allowable Damage Limits
Figure 105 (Sheet 3 of 4)

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REMOVAL OF EDGE DAMAGE FROM THE HANGER LINK

E

C–C

1867291 S0000331868_V1

Allowable Damage Limits
Figure 105 (Sheet 4 of 4)

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IDENTIFICATION 1 - WING TRAILING EDGE FITTINGS

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

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Wing Trailing Edge Location
Figure 1

REFERENCE DRAWINGS	
DRAWING NUMBER	TITLE
115A0001	Functional Product Collector, Fixed Trailing Edge - Left Wing
115A0002	Functional Product Collector, Fixed Trailing Edge - Right Wing
115A6100	Hinge/Actuator Support Installation - Outboard Spoilers, Midspan
115A6200	Rib Installation - Deflection Control, Midspan Fixed Trailing Edge
115A6400	Rod Support Installation - Midspan Fixed Trailing Edge

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U87760 S0000214024_V2

Inboard Wing Trailing Edge Fittings
Figure 2

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LIST OF MATERIALS FOR FIGURE 2			
ITEM	DESCRIPTION	T*[1]	MATERIAL
[1]	Deflection Control Rib Attach Fitting		7050-T7451 machined plate as given in BMS 7-323.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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Midspan Wing Trailing Edge Fittings
Figure 3

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LIST OF MATERIALS FOR FIGURE 3			
ITEM	DESCRIPTION	T*[1]	MATERIAL
[1]	Spoiler Hinge/Actuator Support Fitting		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[2]	Spoiler Hinge Support Fitting		7050-T7451 machined plate as given in AMS 4050. Grain direction controlled part.
[3]	Rod Support Assembly		7050-T7451 machined plate as given in AMS 4050.

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING TRAILING EDGE FITTINGS

1. Applicability

A. Allowable Damage 1 is applicable to damage on the wing fixed trailing edge fittings shown in Figure 101/ALLOWABLE DAMAGE 1.

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Wing Trailing Edge Location
Figure 101

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Midspan Wing Trailing Edge Fittings
Figure 102

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

B. Refer to Table 101/ALLOWABLE DAMAGE 1 for the allowable damage limit references that are applicable to each type of structure.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	PARAGRAPH
ROD SUPPORT ASSEMBLY	4.A
SPOILER HINGE/ACTUATOR FITTING ASSEMBLY	4.B
SPOILER HINGE FITTING	4.B

C. Remove the damage.

(1) Refer to 51-10-02, GENERAL for the investigation and cleanup procedures.

(2) Refer to 51-30-03, GENERAL for possible sources of the abrasive and other materials you can use to remove the damage.

(3) Refer to 51-30-05, GENERAL for possible sources of the equipment and tools you can use to remove the damage.

D. After you remove the damage, do as follows:

WARNING: WEAR EYE PROTECTION WHEN YOU USE THE ROTARY PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the parts given in Table 102/ALLOWABLE DAMAGE 1.

(a) Refer to 51-20-06, GENERAL for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for flap peen and shot peen procedures.

Table 102:

PARTS THAT YOU MUST SHOT PEEN OR FLAP PEEN AFTER YOU REMOVE THE DAMAGE	
PART DRAWING NUMBER	PART NAME
115A6130	Hinge/Actuator Support Assembly - Spoiler No. 1
115A6132	Hinge/Actuator Support Assembly - Spoiler No. 2
115A6133	Hinge Fitting Assembly - Spoiler No. 2
115A6134	Hinge/Actuator Support Assembly - Spoiler No. 3
115A6136	Hinge/Actuator Support Assembly - Spoiler No. 4
115A6138	Hinge/Actuator Support Assembly - Spoiler No. 5
115A6410	Rod Support Assembly

(2) Apply a chemical conversion coating to the bare surfaces of the reworked areas. Refer to 51-20-01, GENERAL.

(3) Apply two layers of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-44-04.

3. References

Reference	Title
51-10-02, GENERAL	Inspection and Removal of Damage

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(Continued)

Reference	Title
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-06, GENERAL	Shot Peening
51-30-03, GENERAL	Sources for Non-Metallic Repair Materials
51-30-05, GENERAL	Equipment and Tools For Repairs
SOPM 20-10-03	SHOT PEENING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Rod Support Assemblies

NOTE: Damage on the edges of large holes and cutouts is not permitted.

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A and B.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Spoiler Hinge/Actuator Support Assemblies and Spoiler Hinge Fitting Assemblies

NOTE: Damage on the edges of large holes and cutouts is not permitted.

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A and B.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , and D.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

A

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

B

U87822 S0000214027_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 2)

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U87826 S0000214028_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 2)

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IDENTIFICATION 1 - WING TRAILING EDGE INBOARD MAIN FLAP SKIN

W85976 S0000130445_V1

Wing Trailing Edge - Inboard Main Flap Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable Trailing Edge Functional Product Collector
113A2001	Flap Installation - Inboard Trailing Edge Flap
113A2002	Flap Assembly - Inboard Trailing Edge Flap
113A2100	Main Flap Assembly - Inboard Trailing Edge Flap
113A2500	Wedge Assembly - Bonded Part, Inboard Trailing Edge Main Flap
113A2141	Skin Panel Assembly - Bonded Part, Upper Trailing Edge Cove

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A2142	Skin Panel - Upper Inboard, Inboard Trailing Edge Main Flap
113A2143	Skin Panel - Upper Outboard, Inboard Trailing Edge Main Flap
113A2144	Skin Panel - Upper Middle, Inboard Trailing Edge Main Flap
113A2145	Skin Panel - Lower Trailing Edge Cove, Inboard Trailing Edge Main Flap
113A2146	Skin Panel - Lower Inboard, Inboard Trailing Edge Main Flap
113A2147	Skin Panel - Lower Outboard, Aft, Inboard Trailing Edge Main Flap
113A2148	Skin Panel - Lower Outboard, Inspar, Inboard Trailing Edge Main Flap
113A2149	Skin Panel - Leading Edge, Inboard Trailing Edge Main Flap

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL

W85987 S0000130443_V2

Wing Trailing Edge - Inboard Main Flap Skin Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Inboard Main Flap, Upper Trailing Edge Cove - Bonded Part		Epoxy impregnated glass woven fabric per BMS 8-79 Style 1581, Class III, Grade B.	
	Skin		Refer to Figure 3	
	Core	0.500	Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction.	
	Phenolic Strip	0.025	MIL-I-247628/14.	
[2]	Inboard Main Flap, Upper Inboard Skin	0.080	2024-T3 bare sheet as given in QQ-A-250/4.	
[3]	Inboard Main Flap, Upper Outboard Skin - Bonded Part		Bond with structural adhesive as given in BMS 5-101, Type II, Class I, Grade 10, as given in BAC 5514-5101.	
	Outer Skin	0.100	7075-T62 bare sheet as given in QQ-A-250/12.	
	Inner Skin	0.016	7075-T6 bare sheet as given in QQ-A-250/12.	
	Core	0.500	Aluminum honeycomb as given in BMS 4-4, Type 3-10, Class NPA.	
[4]	Inboard Main Flap, Upper Middle Skin	0.125	2024-T3 bare sheet as given in QQ-A-250/4.	
[5]	Inboard Main Flap, Lower Trailing Edge Cove Skin	0.125	7075-T6 bare sheet as given in QQ-A-250/12.	
[6]	Inboard Main Flap, Lower Inboard Skin	0.125	2024-T3 bare sheet as given in QQ-A-250/4.	
[7]	Inboard Main Flap, Lower Outboard Aft Skin - Bonded Part		Bond with structural adhesive as given in BMS 5-137, Type II, Class I, Grade 10, as given in BAC 5514-5137.	
	Outer Skin	0.125	7075-T6 bare sheet as given in QQ-A-250/12.	
	Inner Skin	0.016	7075-T6 bare sheet as given in QQ-A-250/12.	
	Core	0.500	Aluminum honeycomb as given in BMS 4-4, Type 4-25, Class NPA.	
[8]	Inboard Main Flap, Inspar Lower Outboard Skin - Bonded Part		Bond with structural adhesive as given in BMS 5-101, Type II, Class I, Grade 10, as given in BAC 5514-5101.	
	Outer Skin	0.080	2024-T3 clad sheet as given in QQ-A-250/5.	
	Inner Skin	0.016	2024-T3 clad sheet as given in QQ-A-250/5.	
	Core	0.500	Aluminum honeycomb as given in BMS 4-4, Type 4-25, Class NPA.	
[9]	Inboard Main Flap, Leading Edge Skin	0.080	2024-T42 clad sheet as given in QQ-A-250/5.	

*[1] Note: T = Pre-manufactured thickness in inches.

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VIEW IS OF THE BAG SIDE (NON-AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTIONS

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, AND D-D FOR THE AREAS WHERE THE PLY SEQUENCE WILL BE IDENTIFIED.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

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Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]
Figure 3 (Sheet 1 of 2)

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W88680 S0000131202_V1

Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]
Figure 3 (Sheet 2 of 2)

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Table 3:

PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [1]		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P40	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Style 1581, Grade B.
P41	----	0.001 inch thick white bondable tedlar film

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VIEW IS ON THE TOOL SIDE (AERODYNAMIC) SURFACE
CORE RIBBON DIRECTIONS

A

NOTES

- REFER TO DETAIL A FOR THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, AND C-C FOR THE BONDED AREAS.

W86084 S0000131218_V1

Core Ribbon Direction for Figure 2, Item [3]
Figure 4 (Sheet 1 of 2)

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W86437 S0000131219_V1

Core Ribbon Direction for Figure 2, Item [3]
Figure 4 (Sheet 2 of 2)

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
CORE RIBBON DIRECTIONS

A

NOTES

- REFER TO DETAIL A FOR THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE BONDED AREAS.

W86380 S0000131220_V1

Core Ribbon Direction for Figure 2, Item [7]
Figure 5 (Sheet 1 of 2)

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BONDED PART
A-A

W86785 S0000131221_V1

Core Ribbon Direction for Figure 2, Item [7]
Figure 5 (Sheet 2 of 2)

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
CORE RIBBON DIRECTIONS

A

NOTES

- REFER TO DETAIL A FOR THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE BONDED AREAS.

W86417 S0000131222_V1

Core Ribbon Direction for Figure 2, Item [8]
Figure 6 (Sheet 1 of 2)

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BONDED PART
A–A

W86420 S0000131223_V1

Core Ribbon Direction for Figure 2, Item [8]
Figure 6 (Sheet 2 of 2)

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IDENTIFICATION 2 - WING TRAILING EDGE INBOARD AFT FLAP SKIN

490521 S0000140254_V1

Wing Trailing Edge - Inboard Aft Flap Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable Trailing Edge Functional Product Collector
113A2001	Flap Installation - Inboard Flap
113A2002	Flap Assembly - Inboard Flap
113A2100	Main Flap Final Assembly - Inboard Flap
113A2500	Wedge Assembly - Bonded Part, Inboard Main Flap
113A2700	Aft Flap Assembly - Inboard Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A2710	Bonded Assembly - Trailing Edge Wedge, Inboard Aft Flap
113A2731	Skin - Leading Edge, Inboard Aft Flap

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

490522 S0000142583_V2

Wing Trailing Edge - Inboard Aft Flap Skin Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Upper Skin	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
[2]	Trailing Edge Filler		Epoxy impregnated glass woven fiber as given in BMS 8-79, Class III, Grade B, Stye 1581 or Style 120 or Style 7781	
[3]	Lower Skin	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
			Glass fabric overlay as given in BMS 8-79, Class III, Style 7781 or BMS 9-3, Type H-2 as given in BAC 5337	For cum line numbers 1131 and on
			(Optional) Glass fabric overlay as given in BMS 8-79, Class III, Style 7781 or BMS 9-3, Type H-2 as given in BAC 5337	For cum line numbers 1 thru 1130
[4]	Honeycomb Core		Aluminum honeycomb core as given in BMS 4-4, Type 4-25, Class NPA, Form B, Grade I	
[5]	Leading Edge SKin	0.071 (1.80)	2024-T42 clad sheet as given in QQ-A-250/5	

*[1] Note: T = Pre-manufactured thickness in inches.

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IDENTIFICATION 3 - WING TRAILING EDGE OUTBOARD MAIN FLAP SKIN

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

490719 S0000142771_V1

Wing Trailing Edge - Outboard Main Flap Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable Trailing Edge Functional Product Collector
113A3001	Flap Installation - Outboard Trailing Edge Flap
113A3002	Flap Assembly - Outboard Trailing Edge Flap
113A3100	Main Flap Assembly - Outboard Trailing Edge Flap
113A3150	Upper Cove Panel - Outboard Main Trailing Edge Flap
113A3173	Lower Cove Panel - Outboard Main Trailing Edge Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A3411	Skin Panel - Upper Forward, Outboard Main Trailing Edge Flap
113A3421	Skin Panel - Lower Forward, Outboard Main Trailing Edge Flap
113A3432	Skin Panel - Leading Edge, Outboard Main Trailing Edge Flap
113A3511	Inboard Trailing Edge Wedge, Outboard Main Trailing Edge Flap
113A3514	Skin Panel - Inboard Lower, Aft, Outboard Main Trailing Edge Flap
113A3601	Skin Panel - Outboard Lower, Aft, Outboard Main Trailing Edge Flap

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD MAIN FLAP LOWER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD MAIN FLAP UPPER SKIN PANEL

490745 S0000142772_V1

Wing Trailing Edge - Outboard Main Flap Skin Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Upper Outboard Trailing Edge Cove Panel - Bonded Part		Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class 3, Grade B, Style 1581	
	Skin		Refer to Figure 3	
	Core (outboard)	0.500	Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction.	
	Core (inboard)	0.500	Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction.	
[2]	Upper Inboard Trailing Edge Cove Panel - Bonded Part		Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class 3, Grade B, Style 1581	
	Skin		Refer to Figure 4	
	Core (outboard)	0.500	Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction.	
	Core (inboard)	0.500	Nomex Core as given in BMS 8-124, Type V, Class IV, Grade 3.0. Refer to Figure 3 for the core ribbon direction.	
[3]	Upper Forward Skin	0.100	7075-T62 bare sheet as given in QQ-A-250/12.	
[4]	Leading Edge Skin	0.080	2024-T42 clad sheet as given in QQ-A-250/5.	
[5]	Lower Trailing Edge Cove Skin	0.080	2024-T3 bare sheet as given in QQ-A-250/4.	
[6]	Lower Forward Skin	0.080	2024-T3 bare sheet as given in QQ-A-250/4.	
[7]	Inboard Trailing Edge Wedge	2.500	7050-T7451 plate as given in AMS 4050.	
[8]	Lower Inboard Aft Skin	0.080	2024-T3 bare sheet as given in QQ-A-250/4.	
[9]	Lower Outboard Aft Skin	0.080	2024-T3 bare sheet as given in QQ-A-250/4.	

*[1] Note: T = Pre-manufactured thickness in inches.

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VIEW IS OF THE BAG SIDE (NON-AERODYNAMIC) SURFACE

PLY LAYUP AND CORE RIBBON DIRECTIONS

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, AND D-D FOR THE AREAS WHERE THE PLY SEQUENCE WILL BE IDENTIFIED.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

493217 S0000143958_V1

Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]
Figure 3 (Sheet 1 of 3)

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PLY SEQUENCE LAYUP
A-A

PLY SEQUENCE LAYUP
B-B

492155 S0000143959_V1

Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]
Figure 3 (Sheet 2 of 3)

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PLY SEQUENCE LAYUP
C-C

PLY SEQUENCE LAYUP
D-D

493216 S0000147703_V1

Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [1]
Figure 3 (Sheet 3 of 3)

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Table 3:

PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [1]		
PLY	**DIRECTION**	**MATERIAL**
P1, P5 thru P10, P21 thru P30, P50 thru P59, P62 thru P71, P91 thru P100, P111 thru P116, P120.	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581.
P2 thru P4, P11 thru P20, P31 thru P49, P60 thru P61, P72 thru P90, P101 thru P110, P117 thru P119.	+ or – 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581.

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VIEW IS OF THE BAG SIDE (NON-AERODYNAMIC) SURFACE

PLY LAYUP AND CORE RIBBON DIRECTIONS

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE AREAS WHERE THE PLY SEQUENCE WILL BE IDENTIFIED.
- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

493285 S0000144068_V1

Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [2]
Figure 4 (Sheet 1 of 2)

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PLY SEQUENCE LAYUP
A–A

PLY SEQUENCE LAYUP
B–B

492107 S0000144070_V1

Ply Direction, Ply Sequence, and Core Ribbon Direction for Figure 2, Item [2]
Figure 4 (Sheet 2 of 2)

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Table 4:

PLY MATERIAL AND DIRECTION FOR FIGURE 2, ITEM [2]		
PLY	**DIRECTION**	**MATERIAL**
P1, P4 thru P7, P10.	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581.
P2 thru P3, P8 thru P9.	+ or – 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581.

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IDENTIFICATION 4 - WING TRAILING EDGE OUTBOARD AFT FLAP SKIN

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

490523 S0000140199_V1

Wing Trailing Edge - Outboard Aft Flap Skin Panel Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable Trailing Edge Functional Product Collector
113A3001	Flap Installation - Outboard Trailing Edge Flap
113A3002	Flap Assembly - Outboard Trailing Edge Flap
113A3700	Aft Flap Assembly - Outboard Trailing Edge Flap
113A3710	Wedge Assembly - Bonded Part, Outboard Trailing Edge Aft Flap
113A3780	Skin Panel - Leading Edge, Outboard Trailing Edge Aft Flap

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

490524 S0000144415_V2

Wing Trailing Edge - Outboard Aft Flap Skin Panel Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin - Leading Edge	0.071 (1.80)	2024-T42 clad sheet as given in QQ-A-250/5	
[2]	Skin - Upper	0.016 (0.41)	7075-T6 bare sheet as given in QQ-A-250/12	
[3]	Fillet		Glass Fiber Reinforced Plastic (GFRP) Pre-preg as given in BMS 8-79, Style 1581 or 120	
[4]	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 4-25 NPA	
[5]	Skin - Lower	0.020 (0.51)	7075-T6 bare sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches.

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ALLOWABLE DAMAGE 1 - WING INBOARD TRAILING EDGE MAIN FLAP

1. Applicability

A. This subject gives the allowable damage limits for the wing inboard trailing edge main flap as shown in Wing Inboard Trailing Edge - Main Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.

F96585 S0006594846_V1

Wing Inboard Trailing Edge - Main Flap Skin Panel Location
Figure 101

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Wing Inboard Trailing Edge - Main Flap Skin Panel Allowable Damage
Figure 102

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2. General

A. Do the steps that follow if you have damage to the aluminum skins:

(1) Remove the damaged material. Refer to 51-10-02 for the procedures.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(4) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

B. Do the steps that follow if you find damage on the trailing edge cove made of Glass Fiber Reinforced Plastic (GFRP) on the upper skin panel.

(1) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Make a temporary seal.

1) Apply aluminum foil tape (speed tape).

2) Keep a record of the location.

3) Make sure the tape is in satisfactory condition at each 400 flight hour interval or more frequently.

4) Seal the damage permanently no later than 5000 flight hours.

(b) Make a permanent seal.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to AMM PAGEBLOCK 51-21-99/701.

(2) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 1

(a) Use a vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Repair the damage no later than 400 flight hours.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES

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(Continued)

Reference	Title
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING

4. Allowable Damage Limits

A. Leading Edge, Upper, Lower, and Bonded Skins (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 1, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.125 inch and less than or equal to 0.250 inch.

b) The distance between two dents or between a dent and other damage is less than 0.50D (D = the larger dimension of the two damage areas.)

2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

(4) Delamination outside of the Zone 1 and 2 locations is not permitted.

(5) Zone 1 - Upper and Lower Skin Delamination

(a) The damage is permitted if:

1) It is a maximum of 2.5 inches (63.5 mm) in diameter.

2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

3) It is a minimum of 1.5 inches (38.1 mm) away from the outboard and inboard edge of the facesheet.

4) You apply a temporary seal with aluminum foil tape (speed tape) and do the steps that follow:

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(6) Zone 2 - Upper and Lower Skin Delamination

(a) The damage is permitted if:

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1) It is a maximum of 1.0 inch (25.4 mm) in diameter.
2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.
3) It is a minimum of 1.0 inch (25.4 mm) aft of the spar web.
4) You apply a temporary seal with aluminum foil tape (speed tape) and do the steps that follow:
   a) Keep a record of the damage location.
   b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.
   c) Replace the tape as necessary.
   d) Repair the damage if the damage becomes larger.
   e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(7) Holes and Punctures outside of the Zone 1 and 2 locations are not permitted.

(8) Zones 1 and 2 - Upper and Lower Skin Holes and Punctures

(a) The damage is permitted if:
1) It is a maximum of 0.50 inch (12.7 mm) in diameter.
2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.
3) It is a minimum of 1.0 inch (25.4 mm) aft of the spar web.
4) You apply a temporary seal with aluminum foil tape (speed tape) and do the steps that follow:
   a) Keep a record of the damage location.
   b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.
   c) Replace the tape as necessary.
   d) Repair the damage if the damage becomes larger.
   e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

B. Trailing Edge Cove (Upper Skin)

NOTE: For edge damage only, refer to Damage Definitions, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Nicks, Gouges, and Scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth.

NOTE: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 3.0 inches in length.

(c) A maximum of 0.25 inch in width.

(d) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(3) Dents are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

NOTE: Use the limits for holes and punctures if there is glass fiber damage.

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(b) The damage is a minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

(b) A maximum of one facesheet and the core in depth.

(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A.

(5) Delaminations are permitted as shown in Figure 105 (Sheet 1), Detail A if it is:

(a) A maximum of 0.50 inch in length.

(b) A maximum of 0.50 inch in width.

(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail A .

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth.

(b) A maximum of 0.25 inch in width.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 105/ALLOWABLE DAMAGE 1, Detail B .

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

F96642 S0006594848_V2

Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage
Figure 103 (Sheet 1 of 3)

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REMOVAL OF CORROSION AROUND THE FASTENERS

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.15 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

E

F96645 S0006594849_V1

Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage
Figure 103 (Sheet 2 of 3)

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DENT THAT IS PERMITTED WITH OR WITHOUT ALUMINUM CORE

F

F96646 S0006594850_V1

Wing Inboard Trailing Edge - Main Flap Skin Allowable Damage
Figure 103 (Sheet 3 of 3)

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR DENT DAMAGE

B

DEFINITIONS FOR EDGE DAMAGE

C

G83224 S0006594851_V1

Damage Definitions
Figure 104

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 2d.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

(A)

CLEANUP AND SEALING OF EDGE EROSION

(B)

G83225 S0006594852_V1

Allowable Damage Limits for the Trailing Edge Cove (Upper Skin)
Figure 105

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ALLOWABLE DAMAGE 2 - WING INBOARD TRAILING EDGE AFT FLAP

1. Applicability

A. This subject gives the allowable damage limits for the wing inboard trailing edge aft flap shown in Wing Inboard Trailing Edge - Aft Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.

F96654 S0006594854_V3

Wing Inboard Trailing Edge - Aft Flap Skin Panel Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

ZONE 1 - DELAMINATION LIMITS

ZONE 2 - DELAMINATION LIMITS

F96694 S0006594855_V3

Wing Inboard Trailing Edge - Aft Flap Skin Panel Allowable Damage
Figure 102

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2. General

A. Do the steps that follow if you have damage:

(1) Remove the damaged material. Refer to 51-10-02 for the procedures.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you need to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

B. After you remove the damage, do as follows:

(1) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(2) Apply a layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E.

(3) Dents:

(a) The damage is permitted as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.125 inch.

b) The distance between two dents, or between a dent and other damage is less than 0.50D (D = the large dimension of the two damage areas).

2) If you fill or rework the dent, do the steps that follow:

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a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) It is a maximum of 0.25 inch in diameter and located in a non-critical area as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail G.

2) You do a close visual inspection of the damage area to verify that there are no cracks or sharp edges.

3) It is a minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

4) You make a temporary seal with aluminum foil tape (speed tape).

NOTE: Use of vacuum and heat to remove moisture, is recommended.

a) Keep a record of the location.

b) Make sure the tape is in satisfactory condition and inspect at each 150 flight hour interval.

5) You do a permanent repair before 5,000 flight hours have occurred, as specified in 51-70-10.

(b) Holes and Punctures are not permitted in critical areas as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail G .

(5) Zone 1 - Upper and Lower Skin Panel Delamination

(a) The damage is permitted if:

1) It is a maximum of 2.0 inches (50.8 mm) in diameter.

2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

3) It is a minimum of 1.0 inch (25.4 mm) aft of the spar web.

(6) Zone 2 - Upper and Lower Skin Panel Edge Delamination

(a) The damage is permitted if:

1) It is a maximum distance of 0.50 inch (12.7 mm) away from the aft edge.

2) It is a maximum of 0.50 inch (12.7 mm) in diameter.

3) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

4) You apply a temporary seal with aluminum foil tape (speed tape) for a 250 degree F cure initial part and do the steps that follow:

a) Keep a record of the damage location.

b) Do an inspection of the delamination every 250 flight cycles, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

B. Leading Edge Skins (Aluminum)

(1) Cracks:

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(a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage, Figure 103/ALLOWABLE DAMAGE 2, Detail F .

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.125 inch.

b) The distance between two dents, or between a dent and other damage is less than 0.50D (D = the large dimension of the two damage areas).

2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

2136592 S0000463705_V1

Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage
Figure 103 (Sheet 1 of 4)

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F96698 S0006594857_V1

Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage
Figure 103 (Sheet 2 of 4)

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DENT THAT IS PERMITTED

F

F96701 S0006594858_V1

Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage
Figure 103 (Sheet 3 of 4)

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TRAILING EDGE WEDGE SURFACES (ALUMINUM HONEYCOMB)

G

L35331 S0006594859_V1

Wing Inboard Trailing Edge - Aft Flap Skin Allowable Damage
Figure 103 (Sheet 4 of 4)

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ALLOWABLE DAMAGE 3 - WING OUTBOARD TRAILING EDGE MAIN FLAP SKIN

1. Applicability

A. This subject gives the allowable damage limits for the wing outboard trailing edge main flap skin shown in Wing Trailing Edge Main Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 3.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F96834 S0006594861_V1

Wing Trailing Edge Main Flap Skin Panel Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANEL

F96905 S0006594862_V2

Wing Outboard Trailing Edge Upper Main Flap Skin Zones
Figure 102

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL

F96904 S0006594863_V2

Wing Outboard Trailing Edge Lower Main Flap Skin Zones
Figure 103

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2. General

A. Do the steps that follow if you have damage to the aluminum skins:

(1) If you have to remove the damage, refer to 51-10-02 for the procedures.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

(1) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(2) Apply one layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

C. Do the steps that follow if you remove damage from the trailing edge cove made of Glass Fiber Reinforced Plastic (GFRP) on the upper skin panel.

(1) Seal the damage that is not more than one ply deep and that agrees with the allowable damage limits in Paragraph 4./ALLOWABLE DAMAGE 3.

(a) Make a temporary seal.

1) Apply aluminum foil tape (speed tape).

2) Keep a record of the location.

3) Make sure the tape is in satisfactory condition at each 400 flight hour interval or more frequently.

4) Seal the damage permanently no later than 5000 flight hours.

(b) Make a permanent seal.

1) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

2) Apply one layer of BMS 10-79, Type 3 or BMS 10-103, Type 1 primer. Refer to AMM 51-21-00/701.

3) Apply one layer of BMS 10-60 enamel to the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

(2) Seal the damaged areas that are more than one ply deep and that agree with the allowable damage limits given in Paragraph 4./ALLOWABLE DAMAGE 3.

(a) Use vacuum and heat to remove moisture from the solid laminate and the honeycomb core. Refer to 51-70-04.

(b) Make a temporary seal with aluminum foil tape (speed tape).

(c) Keep a record of the location.

(d) Repair the damage no later than 400 flight hours.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF PANEL	**ZONE LOCATION**	**PARAGRAPH**
Main Panels	1	4.A
	2	4.B
Leading Edge Skin and Trailing Edge Cove Lower Skin	--	4.C
Trailing Edge Wedge	--	4.D

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Table 101: (Continued)

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF PANEL	ZONE LOCATION	PARAGRAPH
Trailing Edge Cove Upper Skin	--	4.E

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
AMM 51-21-00/701	Interior And Exterior Finishes - Cleaning/Painting
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Main Panels (Upper and Lower Skins) - Zone 1

(1) Cracks:

(a) Remove the damage as shown in Wing Outboard Main Flap - Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 3, Details A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Wing Outboard Main Flap - Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:

(a) The damage is permitted as shown in Wing Outboard Main Flap - Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 3, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

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b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

B. Main Panels (Upper and Lower Skins) - Zone 2

(1) Cracks:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:

(a) The damage is permitted as shown in Figure 104/ALLOWABLE DAMAGE 3, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

C. Leading Edge Skin and Trailing Edge Cove (Lower Skin)

(1) Cracks:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:

(a) The damage is permitted as shown in Figure 104/ALLOWABLE DAMAGE 3, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

D. Trailing Edge Wedge (Upper and Lower Skins)

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(1) Cracks:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 104/ALLOWABLE DAMAGE 3, Details A, B, C, D, and E.

(3) Dents:

(a) The damage is permitted as shown in Figure 104/ALLOWABLE DAMAGE 3, Detail F.

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the dent as given in 51-70-01.

a) The depth of the dent is more than 0.10 inch

b) The distance between two dents or between a dent and other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the dent, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the dent if the damage becomes larger.

(4) Holes and Punctures are not permitted.

E. Trailing Edge Cove (Upper Skin)

NOTE: For edge damage only, refer to Damage Definitions, Figure 105/ALLOWABLE DAMAGE 3, Details A , B , and C for the definitions of the length, width, and depth of damage.

(1) Nicks, gouges, and scratches that do not cause damage to the glass fibers are permitted.

(2) Nicks, gouges, and scratches that cause damage to the glass fibers are permitted if they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply.

(b) A maximum of 3.0 inches in length

(c) A maximum of 0.25 inch in width

(d) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(3) Dents are permitted if they are:

(a) A maximum of 0.50 inch in diameter.

NOTE: Use the limits for holes and punctures if there is glass fiber damage.

(b) The damage is a minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch in diameter

(b) A maximum of one facesheet and the core in depth

(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

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(5) Delaminations are permitted as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A if they are:

(a) A maximum of 0.50 inch in length

(b) A maximum of 0.50 inch in width.

(c) A minimum distance away from other damage as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail A.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.25 inch in width.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits for the Trailing Edge Cove (Upper Skin), Figure 106/ALLOWABLE DAMAGE 3, Detail B.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

F96922 S0006594865_V1

Wing Outboard Main Flap - Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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REMOVAL OF CORROSION AROUND THE FASTENERS

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.15 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

F96935 S0006594866_V1

Wing Outboard Main Flap - Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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DENT THAT IS PERMITTED

F

F96939 S0006594867_V1

Wing Outboard Main Flap - Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR DENT DAMAGE

B

DEFINITIONS FOR EDGE DAMAGE

C

G00396 S0006594877_V1

Damage Definitions
Figure 105

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 2d.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

(A)

CLEANUP AND SEALING OF EDGE EROSION

(B)

G00403 S0006594878_V1

Allowable Damage Limits for the Trailing Edge Cove (Upper Skin)
Figure 106

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ALLOWABLE DAMAGE 4 - WING OUTBOARD TRAILING EDGE AFT FLAP

1. Applicability

A. This subject gives the allowable damage limits for the wing outboard trailing edge aft flap as shown in Wing Trailing Edge Aft Flap Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 4.

B. Allowable Damage 4 is only applicable to wing outboard trailing edge aft flaps which have initial 250 degree F (121 degree C) cure trailing edge wedge assemblies. Refer to drawing 113A3700-1 thru -22 for the 250 degree F (121 degree C) cure trailing edge aft flap assembly details.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G02666 S0006594880_V1

Wing Trailing Edge Aft Flap Skin Panel Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
LOWER SKIN PANEL

ZONE 1 - DELAMINATION LIMITS
ZONE 2 - DELAMINATION LIMITS
ZONE 3 - DELAMINATION LIMITS

F96839 S0006594881_V3

Wing Outboard Trailing Edge - Aft Flap Skin Panel Location
Figure 102 (Sheet 1 of 2)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL IS SHOWN, UPPER SKIN PANEL IS SIMILAR

A

W85732 S0000131653_V2

Wing Outboard Trailing Edge - Aft Flap Skin Panel Location
Figure 102 (Sheet 2 of 2)

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 4 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(2) Apply a layer of BMS 10-79, Type III primer to the reworked areas. Refer to SOPM 20-44-04.

(3) Apply a layer of BMS 10-60, Type II enamel to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Do the steps that follow if you have delamination to the upper or lower skin in the Zone 1 location as shown in Figure 102 for the trimout location.

(1) Trimout the skin, core, end rib, seal, and fiberglass aft filler to a maximum 5.5 inches (139.70 mm) radius. Refer to Figure 102 for the trimout location. Make sure there is a minimum 1.5D (where D = fastener diameter) edge margin to any adjacent fasteners.

(2) Extend the trimout of the core 0.20 inch (5.08 mm) beyond the trimout to the skin (trim the fiberglass aft filler and the inboard end rib flush with the skin trimout). Fill the trimout of the core flush to the skin trimout at all locations. Fill all gaps and apply a fillet seal to all skin-to-core bond locations with BMS 5-26 or BMS 5-45 sealant. Refer to 51-20-05, GENERAL.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05, GENERAL	Repair Sealing
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details, A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , C , D , and E .

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(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 4, Detail F .

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) It is a maximum of 0.25 inch in diameter and located in a non-critical area as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail G .

2) You do a close visual inspection of the damage to verify that there are no cracks or sharp edges.

3) It is a minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

4) You make a temporary seal with aluminum foil tape (speed tape).

NOTE: Use of vacuum and heat to remove moisture, is recommended.

a) Keep a record of the location.

b) Make sure the tape is in satisfactory condition and inspect at each 150 flight hour interval.

5) You do a permanent repair before 5,000 flight hours have occurred, as specified in 51-70-10.

(b) Holes and Punctures are not permitted in critical areas as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail G .

(5) Zone 1 - Upper and Lower Skin Delamination

(a) If you do not trim out the damage as given in Paragraph 2 and Figure 102 you may apply a temporary seal if you do one of the two steps that follow:

1) If the damage is located as shown in Figure 102, Sheet 2 and not more than 5.50 inches (139.70 mm) in radius from the inboard aft edge then seal the damage with aluminum foil tape (speed tape).

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

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2) If the damage is not located as shown in Figure 102, Sheet 2 but is located in Zone 1 and is not more than 2.00 inches (50.80 mm) in diameter and greater than 1.00 inch (25.4 mm) away from an edge, then seal the damage with aluminum foil tape (speed tape). Make sure that all adjacent damage in Zone 1 is as shown in Figure 103, Detail H.

a) Keep a record of the damage location.

b) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(6) Zone 2 - Upper and Lower Skin Delamination

(a) The damage is permitted as given in Figure 103, Detail H if you do the steps that follow:

1) If the damage is not more than 1.50 inches (38.10 mm) in diameter and greater than 1.00 inch (25.4 mm) away from an edge then seal the damage with aluminum foil tape (speed tape).

2) Keep a record of the damage location.

3) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

4) Replace the tape as necessary.

5) Repair the damage if the damage becomes larger.

6) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

(7) Zone 3 - Upper and Lower Skin Delamination

(a) The damage is permitted as given in Figure 103, Detail H if you do the steps that follow:

1) If the damage is not more than 0.50 inch (12.70 mm) in diameter and greater than 1.00 inch (25.4 mm) away from an edge then seal the damage with aluminum foil tape (speed tape).

2) Keep a record of the damage location.

3) Make an inspection of the delamination and tape at each 300 flight hour interval, or more frequently.

4) Replace the tape as necessary.

5) Repair the damage if the damage becomes larger.

6) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

B. Leading Edge Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Details A , B , C , D , and E .

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(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 4, Detail F .

1) If the damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

G02667 S0006594882_V2

Allowable Damage Limits
Figure 103 (Sheet 1 of 5)

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REMOVAL OF CORROSION AROUND THE FASTENERS

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.15 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

E

G02668 S0006594883_V2

Allowable Damage Limits
Figure 103 (Sheet 2 of 5)

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DENT THAT IS PERMITTED

F

G02669 S0006594884_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 5)

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TRAILING EDGE WEDGE SURFACES (ALUMINUM HONEYCOMB)

G

L35525 S0006594885_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 5)

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

D = THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

d = THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X = THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

DAMAGE THAT IS PERMITTED ON THE SURFACE OF AN ALUMINUM HONEYCOMB PANEL

H

W85733 S0000131655_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 5)

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ALLOWABLE DAMAGE 5 - WING OUTBOARD TRAILING EDGE AFT FLAP SKIN

1. Applicability

A. This subject gives the allowable damage limits for the outboard trailing edge aft flap as shown in Figure 101.

B. Allowable Damage 5 is applicable to wing outboard trailing edge aft flaps which have initial 350 degree F (177 degree C) cure trailing edge wedge assemblies. Refer to drawings 113A3700-23/-24, 113A3700-25/-26, or 113A3700-27/-28 for the 350 degree F (177 degree C) cure trailing edge aft flap assembly details.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

J74770 S0000178302_V1

Wing Trailing Edge Aft Flap Skin Panel Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL

ZONE 1 - DELAMINATION LIMITS

ZONE 2 - DELAMINATION LIMITS

J74778 S0000178303_V3

Wing Outboard Trailing Edge - Aft Flap Skin Panel Location
Figure 102

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2. General

A. Refer to Paragraph 4 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(2) Apply a layer of BMS 10-20, Type II primer to the reworked areas. Refer to AMM PAGEBLOCK 51-21-99/701.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-30-03	NON-METALLIC MATERIALS
51-30-03, GENERAL	Sources for Non-Metallic Repair Materials
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-30-05, GENERAL	Equipment and Tools For Repairs
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
51-70-10, REPAIR P/B REPAIR	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Upper and Lower Skins (Aluminum Honeycomb Sandwich)

(1) Cracks:

(a) Remove the damage as shown in Figure 103, Details A, B, and E.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103 Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Figure 103, Detail F .

1) If the permitted damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

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b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures:

(a) The damage is permitted if:

1) It is a maximum of 0.25 inch in diameter and located in a non-critical area as shown in Figure 103, Detail G .

2) You do a close visual inspection of the damage to verify that there are no cracks or sharp edges.

3) It is a minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage.

4) You make a temporary seal with aluminum foil tape (speed tape).

NOTE: Use of vacuum and heat to remove moisture, is recommended.

a) Keep a record of the location.

b) Make sure the tape is in satisfactory condition and inspect at each 150 flight hour interval.

5) You do a permanent repair before 5,000 flight hours have occurred, as specified in 51-70-10.

(b) Holes and Punctures are not permitted in critical areas as shown in Figure 103/ ALLOWABLE DAMAGE 5, Detail G .

(5) Zone 1 - Upper and Lower Skin Panel Delamination

(a) The damage is permitted if:

1) It is a maximum of 1.0 inch (25.4 mm) in diameter.

2) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

3) It is a minimum of 1.0 inch (25.4 mm) aft of the spar web.

(6) Zone 2 - Upper and Lower Skin Panel Edge Delamination

(a) The damage is permitted if:

1) It is a maximum distance of 0.5 inch (12.7 mm) away from the aft edge.

2) It is a maximum of 0.5 inch (12.7 mm) in diameter.

3) It is a minimum of 4.0 inches (101.6 mm) away from other damage.

4) You apply a temporary seal with aluminum foil tape (speed tape) for a 350 degree F cure initial part and do the steps that follow:

a) Keep a record of the damage location

b) Do an inspection of the delamination every 250 flight cycles, or more frequently.

c) Replace the tape as necessary

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 24 months or less from the time the initial seal was made if the damage has not become larger.

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B. Leading Edge Skin (Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Figure 103, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103, Details A , B , C , D , and E .

(3) Dents:

(a) The damage is permitted as shown in Figure 103, Detail F .

1) If the damage agrees with one of the two conditions that follow, then you must fill or rework the damage as given in 51-70-01.

a) The depth of the damage is more than 0.10 inch and less than or equal to 0.250 inch.

b) The distance away from other damage is less than 0.50D (D = the larger dimension of the two damage areas).

2) If you fill or rework the damage, do the steps that follow:

a) Do an inspection of the damage area each 400 flight hour interval or more frequently.

b) Repair the damage if it becomes larger.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

(A)

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

(B)

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

(C)

J74771 S0000178304_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 4)

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REMOVAL OF CORROSION AROUND THE FASTENERS

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.15 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

E

J74772 S0000178306_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 4)

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DENT THAT IS PERMITTED

F

J74773 S0000178307_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 4)

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TRAILING EDGE WEDGE SURFACES (ALUMINUM HONEYCOMB)

G

J74775 S0000178308_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 4)

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REPAIR 1 - WING TRAILING EDGE - MAIN FLAP LOWER SKIN

1. Applicability

A. Repair 1 is applicable to damage to the Inboard and Outboard Main Flap Lower Skins of the Wing Trailing Edge if they are:

(1) Between the front and rear spars

(2) A maximum dimension of 3.0 inches.

B. The wing trailing edge main flaps are shown in Wing Trailing Edge - Location of Main Flap Skin Panels Made of Aluminum , Figure 201/REPAIR 1.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F96860 S0006594888_V1

Wing Trailing Edge - Location of Main Flap Skin Panels Made of Aluminum
Figure 201

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD MAIN FLAP LOWER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

INBOARD MAIN FLAP LOWER SKIN PANEL

F96861 S0006594889_V1

Wing Trailing Edge - Main Flap Skin Panel
Figure 202

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2. General

A. Repair 1 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 1. Incorporation of these inspection requirements into the airplane maintenance program satisfies the damage tolerance assessment of the repair. Refer to 51-00-06 for repair categories and definitions.

B. Make sure the aerodynamic smoothness is satisfactory or there will be a decrease in the performance of the airplane. Refer to 51-10-01.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 27-51-11 P/B 401	INBOARD TRAILING EDGE FLAP - REMOVAL/INSTALLATION
AMM 27-51-21 P/B 401	OUTBOARD TRAILING EDGE FLAP - REMOVAL/INSTALLATION
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 6, 53-30-00, Procedure 5	Inspection of External Fuselage Repairs

4. Repair Instructions

A. Remove the main flap if necessary.

(1) Refer to AMM 27-51-11/401 for the inboard main flap.

(2) Refer to AMM 27-51-21/401 for the outboard main flap.

B. Remove the damage.

(1) If there is a crack, do the steps that follow:

(a) Cut and remove the damage. Refer to 51-10-02.

(b) Refer to Wing Trailing Edge Main Flap Lower Skin Repair, Figure 203/REPAIR 1 for the shape of the cut out.

(c) Make sure there is a minimum of two rows of repair fasteners around the edges of the cut out.

C. Make the repair parts as shown in Wing Trailing Edge Main Flap Lower Skin Repair, Figure 203/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair material.

(1) Refer to 51-40-08 for the countersink repair washers.

(2) Make the contour of the repair parts the same as the initial contour of the skin.

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Table 201:

REPAIR MATERIAL			
ITEM	PART	QUANTITY	MATERIAL
[1]	Doubler	1	Use bare or clad 2024-T3 that is 0.040 inch thick
[2]	Doubler	1	Use bare or clad 2024-T3 that is 0.040 inch thick
[3]	Doubler	1	Use bare or clad 2024-T3 that is 0.040 inch thick
[4]	Doubler	1	Use bare or clad 2024-T3 that is 0.040 inch thick
[5]	Filler	1	Use bare or clad 2024-T3 that is the same thickness as the initial skin

D. Assemble the repair parts and drill the fastener holes.

(1) Countersink the fastener holes in the repair parts to a depth that is 80 percent or less of the part thickness.

NOTE: If the depth of the countersink is more than 80 percent of the part thickness a knife edge condition will result.

(2) Use shims to make the clearance between the parts 0.005 inch or less before you install the fasteners.

E. Remove the repair parts.

F. Remove all nicks, scratches, burrs, and sharp edges from the repair parts.

G. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the skin.

H. Apply one layer of BMS 10-79, Type II primer to the repair parts and to the bare edges of the initial skin. Refer to 51-20-01.

I. Install countersink washers with BMS 5-95 sealant at the initial fastener locations as necessary. Refer to 51-40-08.

J. Install the repair parts.

(1) Apply BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

K. Install the fasteners.

(1) Install the hex drive bolts wet with BMS 5-95 sealant.

(2) Install the rivets without sealant.

L. Install the skin panel on the main flap.

M. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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NOTE: D=THE DIAMETER OF THE FASTENER

BOTTOM VIEW OF LOWER SKIN

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION

REPAIR FASTENER LOCATION. INSTALL A BACB30NW6K HEX DRIVE BOLT WITH A BACC30M COLLAR.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE6D RIVET.

F97027 S0006594891_V1

Wing Trailing Edge Main Flap Lower Skin Repair
Figure 203

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5. **Inspection Requirements**

A. Two sets of inspection instructions are given. The inspection given in Paragraph 5.B./REPAIR 1 define the inspections requirements after the skin repairs are installed on the inboard main flap lower skin panels. The inspection given in Paragraph 5.C./REPAIR 1 define the inspections requirements after the skin repairs installed are on the outboard main flap lower skin panels.

B. Do a Low Frequency Eddy Current (LFEC) inspection at 56,000 flight cycles after you install the repair on the inboard main flap lower skin panels.

(1) Do this inspection if the entire repair area or portion of the repair area is between WBL 68 and 148 of lower skin panel.

(2) No inspection is required on any part of the repair beyond WBL STA 148 of lower skin panel.

(3) Do the steps that follow to inspect the front and rear spar lower chords and the lower skin of the repair area from external.

(a) Inspect the entire repair area and extend the inspection into the regions one additional fastener beyond the perimeter of the repair area.

(b) Inspect the lower skin with LFEC skin crack-probing settings on the area defined in Paragraph 5.B.(3)(a)/REPAIR 1.

(c) Inspect the spar lower chords with LFEC chord crack-probing settings only on the chord areas attaching to the lower skin at the skin repair area.

(4) Refer to 737 NDT Part 6, 53-30-00, Procedure 5, Page 12.

(5) Repeat the inspection every 36,000 flight cycles interval after initial inspection.

C. Do a Low Frequency Eddy Current (LFEC) inspection at 56,000 flight cycles after you install the repair on the outboard main flap lower skin panels.

(1) Do this inspection if the entire repair area or portion of the repair area is between main flap WBL 254 and 358 of lower skin panel.

(2) No inspection is required on any part of the repair outside of the area defined in Paragraph 5.C.(1)/REPAIR 1 of the lower skin panel.

(3) Do the steps that follow to inspect the front and rear spar lower chords and the lower skin of the repair area from external.

(a) Inspect the entire repair area and extend the inspection into the regions one additional fastener beyond the perimeter of the repair area.

(b) Inspect the lower skin with LFEC skin crack-probing settings on the area defined in Paragraph 5.C.(3)(a)/REPAIR 1

(c) Inspect the spar lower chords with LFEC chord crack-probing settings only on the chord areas attaching to the lower skin at the skin repair area.

(4) Refer to 737 NDT Part 6, 53-30-00, Procedure 5, Page 12.

(5) Repeat the inspection every 5,000 flight cycles interval after initial inspection.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD MAIN FLAP LOWER SKIN PANEL

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

INBOARD MAIN FLAP LOWER SKIN PANEL

INSPECTION AREA AFTER THE IMPLEMENTATION OF LOWER SKIN REPAIR

1846593 S0000327571_V1

Wing Trailing Edge - Main Flap Skin Panel
Figure 204 (Sheet 1 of 2)

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1846545 S0000327568_V1

Wing Trailing Edge - Main Flap Skin Panel
Figure 204 (Sheet 2 of 2)

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REPAIR 2 - WING TRAILING EDGE AFT FLAP LEADING EDGE SKIN BLIND FASTENER REPAIR

1. Applicability

A. Repair 2 is applicable to damage to wing inboard and outboard trailing edges in the areas that satisfy each of these two conditions:

(1) The area is the aft flap leading edge shown in Wing Trailing Edge Aft Flap Skin Panel Locations, Figure 201/REPAIR 2.

(2) The area is between spar chords and ribs where the skin is of a constant thickness.

2. General

A. Repair 6 is an alternative to Repair 2. Use Repair 6 when the leading edge skin is repaired with solid fasteners. Use this repair when the leading edge skin is repaired with blind fasteners.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

G04335 S0006594893_V1

Wing Trailing Edge Aft Flap Skin Panel Locations
Figure 201

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LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

INBOARD TRAILING EDGE AFT FLAP SKIN PANELS

LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

OUTBOARD TRAILING EDGE AFT FLAP SKIN PANELS

G04362 S0006594894_V2

Wing Trailing Edge - Aft Flap Leading Edge Skin Repair
Figure 202

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 27-51-12 P/B 401	INBOARD AFT FLAP - REMOVAL/INSTALLATION
AMM 27-51-22 P/B 401	OUTBOARD AFT FLAP - REMOVAL/INSTALLATION
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Cut and remove the damaged part of the leading edge skin as shown in Wing Trailing Edge - Aft Flap Leading Edge Skin Repair, Figure 203/REPAIR 2. Refer to 51-10-02 for the procedures to remove the damage.

(1) Make the cut in the shape of a rectangle with the longer sides parallel to the leading edge ribs.

(2) Make sure there is a minimum of two rows of repair fasteners around the edges of the cutout.

B. Put the skin that is around the damage back to the initial contour.

C. Make the repair parts as shown in Wing Trailing Edge - Aft Flap Leading Edge Skin Repair, Figure 203/REPAIR 2. Refer to Table 201/REPAIR 2 for the repair material.

(1) Make the contour of the repair parts the same as the contour of the initial skin.

NOTE: Make sure that the repair doublers do not overlap the chem-milled radius of the skin.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Repair Plate	1	Use bare or clad 2024-T42 sheet that is 0.071 inch thick
[2]	Doubler	1	Use bare or clad 2024-T42 sheet that is 0.071 inch thick
[3]	Doubler	1	Use bare or clad 2024-T42 sheet that is 0.071 inch thick
[4]	Strap	1	Use bare or clad 2024-T4 sheet that is 0.071 inch thick

D. Assemble the parts as shown in Wing Trailing Edge - Aft Flap Leading Edge Skin Repair, Figure 203/REPAIR 2.

E. Drill and countersink the fastener holes.

F. Remove the nicks, scratches, gouges, and sharp edges from the repair parts and the bare surfaces of the initial skin.

G. Apply a chemical conversion coating to the repair parts and bare surfaces of the initial skin. Refer to 51-20-01 for the chemical conversion coating procedures.

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H. Apply one layer of BMS 10-11, Type I primer to the repair parts and the bare surfaces of the initial skin.

I. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05 for the procedures to apply the sealant.

J. Install the aluminum fasteners without sealant.

K. Fill the space between the [1] repair plate and the initial skin with BMS 5-95 sealant. Refer to 51-20-05.

L. Apply the decorative finish to the repair area as given in AMM PAGEBLOCK 51-21-99/701.

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FASTENER SYMBOLS

REPAIR FASTENER LOCATION. INSTALL A NAS1399D5 BLIND RIVET.

REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET.

G05439 S0006594896_V1

Wing Trailing Edge - Aft Flap Leading Edge Skin Repair
Figure 203 (Sheet 1 of 2)

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A–A

B–B

G05689 S0006594897_V1

Wing Trailing Edge - Aft Flap Leading Edge Skin Repair
Figure 203 (Sheet 2 of 2)

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REPAIR 3 - OUTBOARD TRAILING EDGE MAIN FLAP WEDGE

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

G17731 S0006594899_V2

Wing Outboard Main Flap Trailing Edge - Trailing Edge Wedge Location
Figure 201

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REPAIR 4 - INBOARD AND OUTBOARD AFT FLAP TRAILING EDGE WEDGE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTES

- REFER TO SRM 51-70-10 FOR TYPICAL METAL REPAIRS FOR THE INBOARD AND OUTBOARD AFT FLAP TRAILING EDGE WEDGE. MAKE SURE THERE IS SUFFICIENT CLEARANCE FROM ADJACENT STRUCTURE TO INSTALL THE REPAIR PARTS.
- THE SUM OF THE LENGTHS OF EXTERNAL REPAIRS TO THE INBOARD AFT FLAP TRAILING EDGE WEDGE AND THE OUTBOARD AFT FLAP TRAILING EDGE WEGE MUST BE LESS THAN OR EQUAL TO 24 INCHES. A SINGLE REPAIR THAT INCLUDES BOTH THE UPPER AND LOWER SURFACES IS COUNTED ONLY ONCE AGAINST THE SPANWISE TOTAL.
- REFER TO SRM 51-70-06 FOR TYPICAL REPAIRS WITH WET LAYUP MATERIALS FOR GLASS FABRIC REINFORCED HONEYCOMB CORE PANELS FOR THE INBOARD AFT FLAP TRAILING EDGE LOWER SKIN PROTECTIVE PLIES.

1 ONLY ADHESIVES EA 9696 OR EA-9628 (DEXTER CORPORATION) OR AF-163 (3M AEROSPACE MATERIALS) ARE PERMITTED ON THE OUTBOARD AFT FLAP TRAILING EDGE WEDGES.

G17913 S0006594901_V5

Wing Inboard and Outboard Aft Flap - Trailing Edge Wedge Locations
Figure 201

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REPAIR 6 - WING TRAILING EDGE AFT FLAP LEADING EDGE SKIN SOLID FASTENER REPAIR

1. Applicability

A. Repair 6 is applicable to damage to wing inboard and outboard trailing edges in the areas that satisfy each of these two conditions:

(1) The area is the aft flap leading edge shown in Figure 201/REPAIR 6.

(2) The area is between spar chords and ribs where the skin is of a constant thickness.

2. General

A. Repair 2 is an alternative to Repair 6. Use Repair 2 when when the leading edge skin is repaired with blind fasteners. Use this repair when the leading edge skin repaired with solid fasteners.

B. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the performance of the airplane. Refer to 51-10-01.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

2277679 S0000513872_V1

Wing Trailing Edge Aft Flap Skin Panel Locations
Figure 201

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LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

INBOARD TRAILING EDGE AFT FLAP SKIN PANELS

LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

OUTBOARD TRAILING EDGE AFT FLAP SKIN PANELS

2277680 S0000513873_V1

Wing Trailing Edge - Aft Flap Leading Edge Skin Repair
Figure 202

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3. References

Reference	Title
51-10-01	AERODYNAMIC SMOOTHNESS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
AMM 27-51-12 P/B 401	INBOARD AFT FLAP - REMOVAL/INSTALLATION
AMM 27-51-22 P/B 401	OUTBOARD AFT FLAP - REMOVAL/INSTALLATION
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions

A. Remove the damaged leading edge skin from the aft flap assembly.

B. Cut and remove the damaged part of the leading edge skin as shown in Figure 203/REPAIR 6. Refer to 51-10-02 for the procedures to remove the damage.

(1) Make the cut in the shape of a rectangle with the longer sides parallel to the leading edge ribs.

(2) Make sure there is a minimum of two rows of repair fasteners around the edges of the cutout.

C. Put the skin that is around the damage back to the initial contour.

D. Make the repair parts as shown in Figure 203/REPAIR 6. Refer to Table 201/REPAIR 6 for the repair material.

(1) Make the contour of the repair parts the same as the contour of the initial skin.

NOTE: Make sure that the repair doubler does not overlap the chem-milled radius of the skin.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Repair Plate	1	Use bare or clad 2024-T42 sheet that is 0.071 inch thick
[2]	Doubler	1	Use bare or clad 2024-T42 sheet that is 0.071 inch thick

E. Assemble the parts as shown in Figure 203/REPAIR 6.

F. Drill and countersink the fastener holes.

G. Remove the nicks, scratches, gouges, and sharp edges from the repair parts and the bare surfaces of the initial skin.

H. Apply a chemical conversion coating to the repair parts and bare surfaces of the initial skin. Refer to 51-20-01 for the chemical conversion coating procedures.

I. Apply one layer of BMS 10-11, Type I primer to the repair parts and the bare surfaces of the initial skin.

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J. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05 for the procedures to apply the sealant.

K. Install the aluminum fasteners without sealant.

L. Fill the space between the [1] repair plate and the initial skin with BMS 5-95 sealant. Refer to 51-20-05.

M. Apply the decorative finish to the repair area as given in AMM PAGEBLOCK 51-21-99/701.

N. Re-install the repaired leading edge skin to the aft flap assembly.

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FASTENER SYMBOLS

┼ REPAIR FASTENER LOCATION. INSTALL A BACR15CE5D RIVET.

2277686 S0000513874_V1

Wing Trailing Edge - Aft Flap Leading Edge Skin Repair
Figure 203 (Sheet 1 of 2)

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A–A

B–B

2277691 S0000513875_V1

Wing Trailing Edge - Aft Flap Leading Edge Skin Repair
Figure 203 (Sheet 2 of 2)

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REPAIR 7 - INBOARD MAIN FLAP TRAILING EDGE WEDGE

1. Applicability

A. REPAIR 7 is applicable to the inboard main flap trailing edge wedge 113A2500 as shown in Figure 201/REPAIR 7.

2. General

A. This repair is a permanent repair. Refer to 51-00-06, GENERAL for repair categories and definitions.

B. Do the repair as given in Paragraph 4./REPAIR 7.

C. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01, GENERAL.

D. Keep a minimum of 0.50 in. (12.70 mm) corner radius of all repair parts, unless they are shown differently.

E. Make sure the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

A

2335066 S0000531931_V1

Inboard Main Flap Trailing Edge Wedge Location
Figure 201

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-20-13, GENERAL	Surface Roughness Finish Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
51-70-10, REPAIR 3	Septumized Core Repairs
51-70-10, REPAIR 4	Repairs to Large Damage
51-70-10, REPAIR GENERAL	Aluminum Skin/Aluminum Honeycomb Panel Repairs
AMM 27-51-11-000-801	Inboard Trailing Edge Flap Removal (P/B 401)
AMM 27-51-11-400-801	Inboard Trailing Edge Flap Installation (P/B 401)
AMM 27-51-11-820-801	Inboard Trailing Edge Flap Adjustment (P/B 501)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
737 NDT Part 1, 51-04-00	Ultrasonic

4. Repair Instructions

A. Get access to the damaged area.

(1) If an autoclave repair is used, then:

(a) Remove the inboard trailing edge main flap. Refer to AMM TASK 27-51-11-000-801.

(b) Remove the inboard trailing edge main flap wedge.

(c) Remove all components from the inboard trailing edge main flap wedge.

(2) If a vacuum bag repair is used, then remove the inboard trailing edge main flap if necessary. Refer to AMM TASK 27-51-11-000-801.

B. Cut and remove the damaged skin and core, as necessary. Refer to 51-70-10, REPAIR GENERAL, Paragraph 7.

C. Make the repair parts. Refer to 51-70-10, REPAIR GENERAL, Paragraph 8.

(1) If the repair is done for part of the skin, then:

(a) Make the repair doubler and the repair skin from the same material and thickness as the initial skin.

(b) Make the repair doubler with 1.5 in. (38.1 mm) overlap to the initial skin.

(2) If the repair is done for the full skin, then make the repair skin from the same material and thickness as the initial skin.

(3) If the repair will cover the initial fasteners, then:

(a) Remove the initial fasteners in the repair area.

(b) Drill the fastener holes in the repair doubler and the repair skin.

D. Remove all nicks, scratches, burrs, and sharp edges from the repair doubler and the bare surfaces of the initial parts.

E. Replace the damaged core if necessary. Refer to 51-70-10, REPAIR GENERAL, Paragraph 9. The maximum cure temperature of the repair core is 260°F (127°C).

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F. Install countersink repair washers, if necessary, with BMS 5-95 sealant. Refer to 51-40-08, GENERAL.

G. Install the repair. Refer to Table 201/REPAIR 7 for the repair data and 51-70-10, REPAIR GENERAL, Paragraph 20, for the repair instructions. Make sure that you use an applicable tool to hold all of the parts in their positions during the repair procedure.

NOTE: In most (but not all) conditions, the tool must be the same contour as the part to be cured.

Table 201: Permanent Repair Data for the Inboard Main Flap Trailing Edge Wedge

REPAIR TYPE	FILM ADHESIVES			
	AUTOCLAVE		VACUUM BAG	
REPAIR CURE TEMPERATURE	250°F (121°C)	350°F (177°C) *[1]	250°F (121°C)	350°F (177°C)
DAMAGE SIZE *[2]	Permitted for skin damage less than 200 in^2 (1290 cm^2)	No limit.	Permitted for skin damage less than 64 in^2 (413 cm^2)	Repair is not permitted
SURFACE PREPARATION	BOEGEL (AC-30), or TANK PAA, or PACS, or PANTA	BOEGEL (AC-30), or TANK PAA, or PACS, or PANTA	BOEGEL (AC-30), or TANK PAA, or PACS, or PANTA	----
BMS 5-121 POSITIONING FABRIC*[3]	Optional	Optional	Optional	----
ADHESIVES	BMS 5-101, Type II (EA 9696, EA-9628 and AF-163)	BMS 5-137, Type II (Cytec FM-377S)	BMS 5-101, Type II (EA 9696, EA-9628 and AF-163)	----
REPAIRS PER LIFETIME	No limit	2 Repairs	No limit	----
REPAIR PROCEDURE	51-70-10, REPAIR 3 51-70-10, REPAIR 4	51-70-10, REPAIR 3 51-70-10, REPAIR 4	51-70-10, REPAIR 3 51-70-10, REPAIR 4	----

*[1] If there is a 250°F (121°C) initial repair in the repair area, you must replace the initial repair with this 350°F (177°C) autoclave repair.

*[2] The damage size is applicable to each of the upper and lower skin. You can have damage on both the upper and lower skin.

*[3] Put the positioning fabric between the film adhesive and the repair doubler or initial skin. Do not put the positioning fabric between the adhesive and the honeycomb core.

H. Inspect the fit of the repair parts. Refer to 51-70-10, REPAIR GENERAL, Paragraph 10.

I. Do an ultrasonic bond inspection of metal-to-metal bonding and honeycomb structure. Refer to 737 NDT Part 1, 51-04-00. Do not do a tap test inspection of the honeycomb panel.

J. Install the repair fasteners wet with BMS 5-95 in the initial fastener holes. Refer to 51-40-02, GENERAL, 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805.

K. Clean, seal and finish the repair. Refer to 51-70-10, REPAIR GENERAL, Paragraph 22.

L. Re-install the components of the inboard trailing edge main flap wedge if they were removed.

M. Re-install the inboard trailing edge main flap wedge if it was removed.

N. Re-install the inboard trailing edge main flap if it was removed. Refer to AMM TASK 27-51-11-400-801.

O. Adjust the inboard trailing edge main flap. Refer to AMM TASK 27-51-11-820-801.

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REPAIR 8 - INBOARD MAIN FLAP TRAILING EDGE WEDGE REPAIR OF DISBOND LESS THAN 75 SQUARE INCHES

1. Applicability

A. REPAIR 8 is applicable to the inboard main flap trailing edge wedge 113A2500 as shown in Figure 201/REPAIR 8.

B. REPAIR 8 is applicable to the upper and lower surfaces of the inboard main flap trailing edge wedge.

C. REPAIR 8 is applicable to damage that:

(1) Is located above honeycomb core areas

(2) Has no damage to the honeycomb core

(3) Is 3.0 in. (76.2 mm) or more away from the aft edge, and less than 8.0 in. (203.2 mm) away from the aft edge

(4) Is 4.5 in. (114.3 mm) or more away from the outboard edge

(5) Is 4.0 in. (101.6 mm) or more away from the inboard edge.

D. REPAIR 8 is applicable if the disbond area for each surface is 75 in^2 (484 cm^2) or less. If the disbonds are repaired on the upper and the lower skin, the doublers must not overlap. Refer to Figure 202/REPAIR 8. Keep all other limits as given in Paragraph 1.C./REPAIR 8.

E. Refer to REPAIR 9 and REPAIR 10 if the damage is larger than 75 in^2 (484 cm^2) or the damage extends to an edge.

2. General

A. This repair is a time-limited repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repair. Replace this time-limited repair with a permanent repair no later than 24 months from the time that the repair is installed.

B. D = fastener diameter.

C. All dimensions are in inches, unless they are shown differently.

D. Keep a minimum of 0.50 in. (12.70 mm) corner radius of all repair parts, unless they are shown differently.

E. Keep a 5D to 7D fastener spacing, unless it is shown differently.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
UPPER SKIN PANEL

A

2355463 S0000538277_V1

Inboard Main Flap Trailing Edge Wedge Location
Figure 201

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2361327 S0000539702_V1

Disbond Area
Figure 202

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
AMM 27-51-11-000-801	Inboard Trailing Edge Flap Removal (P/B 401)
AMM 27-51-11-400-801	Inboard Trailing Edge Flap Installation (P/B 401)
AMM 27-51-11-820-801	Inboard Trailing Edge Flap Adjustment (P/B 501)
AMM 51-21-73-370-803	BMS10-60 Topcoat Application (P/B 701)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-43-03	Chemical Conversion Coatings for Aluminum
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-05-01	Tap Test Inspection of Honeycomb Sandwich Structure

4. Repair Instructions

A. Get access to the damaged area. Remove the inboard trailing edge main flap if necessary. Refer to AMM TASK 27-51-11-000-801.

B. Do a tap test inspection of the honeycomb panel to find the dimensions of the disbond area. Refer to 737 NDT Part 1, 51-05-01.

C. Make the repair part. Refer to Table 201/REPAIR 8 and Figure 203/REPAIR 8. For repairs to upper and lower skins, make sure that doublers do not overlap.

(1) There must be two rows of fasteners on each side of the disbond.

(2) The forward fasteners must be 9.0 in. (228.6 mm) or less away from the aft edge.

(3) The aft fasteners must be 2.0 in. (50.8 mm) or more away from the aft edge.

(4) The outboard fasteners must be 3.5 in. (88.9 mm) or more away from the outboard edge.

(5) The inboard fasteners must be 3.0 in. (76.2 mm) or more away from the inboard edge.

(6) Keep a 0.75 in. (19.0 mm) minimum fastener edge margin.

(7) Chamfer the edges of the Part [1] Doubler with 10:1 taper ratio to a maximum thickness of 0.020 in. (0.508 mm).

Table 201: REPAIR MATERIALS

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.600 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12

D. Assemble the Part [1] Doubler.

E. Drill the fastener holes through the Part [1] Doubler and the initial skin. The drill depth into the core must have sufficient depth for the fastener length, but must not damage the opposite initial skin. Countersink the fastener holes in the Part [1] Doubler. Refer to 51-40-05, GENERAL and Figure 203/REPAIR 8 for the fastener locations, type and size.

F. Disassemble the Part [1] Doubler.

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G. Examine the damaged area for signs of water, oil, fuel, dirt or other contamination in the honeycomb. Remove all fluids, dirt and other contamination.

(1) Use a vacuum or oil-free compressed air to remove water.

(2) Dry the honeycomb core. You can heat the core or use hot air at the wet area for 1 hour (minimum) at 150°F (66°C). Make sure that the temperature does not increase more than 5°F (3°C) a minute. Dry again if necessary.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(3) If a honeycomb core has oil in it, then spray on MEK, MIBK, MPK, or acetone solvent to remove the contamination. Be careful to prevent damage to the adhesive on the opposite skin. Immediately remove the solvent from the core. Let the core become completely dry.

H. Remove all nicks, scratches, burrs, and sharp edges from the Part [1] Doubler and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the Part [1] Doubler and to the bare surfaces of the initial parts. Refer to SOPM 20-43-03, Class A.

J. Apply one layer of BMS 10-79 Type III primer to the Part [1] Doubler. Refer to SOPM 20-44-04.

K. Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05, GENERAL.

L. Install blind rivets wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Shave rivets flush. Refer to 51-40-02, GENERAL.

M. Apply a chemical conversion coating to the bare surfaces of the blind rivet heads. If necessary apply a chemical conversion coating to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-43-03, Class A.

N. If necessary apply BMS 10-79 Type III primer to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-44-04.

O. Apply one layer of BMS 10-60, Type II enamel to the repair area. Refer to AMM TASK 51-21-73-370-803.

P. Re-install the inboard trailing edge main flap if it was removed. Refer to AMM TASK 27-51-11-400-801.

Q. Adjust the inboard trailing edge main flap. Refer to AMM TASK 27-51-11-820-801.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

WING INBOARD MAIN FLAP TRAILING EDGE

INBOARD MAIN FLAP TRAILING EDGE WEDGE

A

2358082 S0000538798_V1

Inboard Main Flap Trailing Edge Wedge Repair of Disbond Less Than 75 Square Inches
Figure 203 (Sheet 1 of 2)

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A–A

TYPICAL REPAIR OF UPPER SKIN IS SHOWN
TYPICAL REPAIR OF LOWER SKIN IS SIMILAR
(ROTATED 90° COUNTERCLOCKWISE)

B–B

FASTENER SYMBOLS

+ REPAIR FASTENER LOCATION. INSTALL A BACR15FP4E() RIVET. AS AN ALTERNATIVE INSTALL A NAS1399D4 OR NAS1739E4 RIVET.

2358083 S0000539118_V1

Inboard Main Flap Trailing Edge Wedge Repair of Disbond Less Than 75 Square Inches
Figure 203 (Sheet 2 of 2)

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REPAIR 9 - INBOARD MAIN FLAP TRAILING EDGE WEDGE REPAIR OF DISBONDS THAT ARE 200 SQUARE INCHES OR LESS

1. Applicability

A. REPAIR 9 is applicable to the inboard main flap trailing edge wedge (113A2500) as shown in Figure 201/REPAIR 9.

B. REPAIR 9 is applicable to the upper or lower surface of the inboard main flap trailing edge wedge. This repair only applies to one surface. Tell Boeing if a repair of disbonds to the upper and lower surface is necessary.

C. REPAIR 9 is applicable to disbonds of a skin as shown in Figure 202/REPAIR 9 that:

(1) Are 200 in^2 (1290 cm^2) or less in area.

(2) Are 4.5 in. (114.3 mm) or less away from the aft edge.

(3) Can extend to the inboard, outboard and aft edges of the skin.

D. Refer to Figure 203/REPAIR 9 for the upper and lower skin disbond repair that extends the full length of the repair zone.

E. Refer to Figure 204/REPAIR 9 for the upper skin disbond repair of the outboard, center and inboard areas of the repair zone

F. Refer to Figure 205/REPAIR 9 for the lower skin disbond repair of the outboard, center and inboard areas of the repair zone

G. Refer to REPAIR 8 if the disbond is 75 in^2 (484 cm^2) or less and it does not extend to an edge. Refer to REPAIR 10 if the disbond is more than 200 in^2 (1290 cm^2).

2. General

A. This repair is a time-limited repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repair. Replace this time-limited repair with a permanent repair no later than 24 months from the time that the repair is installed.

B. D = fastener diameter.

C. All dimensions are in inches, unless they are shown differently.

D. Keep a minimum of 0.50 in. (12.70 mm) corner radius of all repair parts, unless it is shown differently.

E. Keep a 1.25 in. (31.75 mm) to 1.75 in. (44.45 mm) fastener spacing, unless it is shown differently.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

2362251 S0000538988_V1

Inboard Main Flap Trailing Edge Wedge Repair of Disbonds Less Than 200 Square Inches Location
Figure 201

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
UPPER WEDGE SKIN PANEL

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
LOWER WEDGE SKIN PANEL

ZONE 1 - DISBOND IS PERMITTED IN THIS AREA ONLY

ZONE 2 - DISBOND NOT PERMITTED IN THIS AREA

2362638 S0000538989_V1

Inboard Main Flap Trailing Edge Wedge Disbond Repair Zone 1
Figure 202

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
AMM 27-51-11-000-801	Inboard Trailing Edge Flap Removal (P/B 401)
AMM 27-51-11-400-801	Inboard Trailing Edge Flap Installation (P/B 401)
AMM 27-51-11-820-801	Inboard Trailing Edge Flap Adjustment (P/B 501)
AMM 51-21-73-370-803	BMS10-60 Topcoat Application (P/B 701)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-43-03	Chemical Conversion Coatings for Aluminum
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-05-01	Tap Test Inspection of Honeycomb Sandwich Structure

4. Repair Instructions

A. Get access to the damaged area. Remove the inboard trailing edge main flap if necessary. Refer to AMM TASK 27-51-11-000-801.

B. Do a tap test inspection of the honeycomb panel to find the dimensions of the disbond area. Refer to 737 NDT Part 1, 51-05-01.

C. Make the Part [1] Doubler and Part [2] Sleeves. Refer to Table 201/REPAIR 9, Figure 203/REPAIR 9, Figure 204/REPAIR 9 or Figure 205/REPAIR 9 as applicable.

(1) The end of the Part [2] Sleeve in the honeycomb core must be flat against the undamaged skin. Make sure that the opposite end is flush with the outside of the damaged skin.

NOTE: Sleeves are only necessary in areas that have a core.

(2) There must be two rows of fasteners on each side of the disbond that does not extend to an edge as shown in Figure 203/REPAIR 9, Figure 204/REPAIR 9 or Figure 205/REPAIR 9 .

(3) The forward fasteners must be 6.5 in. (165.1 mm) or less away from the aft edge.

(4) Fastener edge margin must be a minimum of 0.75 in. (19.1 mm) which includes the chamfer, except:

(a) If the Part [1] Doubler extends to the aft edge of the wedge, the edge margin is 0.5 in. (12.7 mm) from the aft edge.

(b) If the Part [1] Doubler extends to the inboard edge of the wedge, the edge margin is 1.0 in. (25.4 mm) from the inboard edge.

(c) If the Part [1] Doubler extends to the outboard edge of the wedge, the edge margin is 1.25 in. (31.8 mm) from the outboard edge.

(5) Chamfer all edges of the Part [1] Doubler with a 10:1 taper ratio to a maximum thickness of 0.020 in. (0.508 mm) as shown in Figure 203/REPAIR 9, Figure 204/REPAIR 9 or Figure 205/REPAIR 9, except:

(a) If the disbond extends to the aft edge of the wedge , the chamfer along the aft edge is 0.048 in. (1.2 mm) by 0.200 in. (5.1 mm) as shown in Figure 203/REPAIR 9, Figure 204/REPAIR 9 or Figure 205/REPAIR 9.

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(b) If the disbond extends to the inboard or outboard edge of the wedge , no chamfer is necessary along that edge.

Table 201: REPAIR MATERIALS

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.600 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12
2	Sleeve	As necessary	Any 2000 series or 7000 series aluminum extrusion. 0.25 in. (6.35 mm) outside diameter x0.16 in. (4.06 mm) inside diameter

D. Assemble the Part [1] Doubler. If the repair extends to the outboard or inboard edge then remove the initial fasteners as shown in Figure 203/REPAIR 9, Figure 204/REPAIR 9 or Figure 205/REPAIR 9, as applicable.

E. Drill the repair fastener holes through the Part [1] Doubler, the damaged skin and the undamaged skin. Countersink the repair fastener holes in the Part [1] Doubler. Refer to 51-40-05, GENERAL and Figure 203/REPAIR 9, Figure 204/REPAIR 9 or Figure 205/REPAIR 9 for the fastener locations, type and size.

(1) If the doubler extends to the inboard or outboard edge of the upper or lower skin, drill holes for the initial fastener hole locations shown as applicable.

(2) Holes must be perpendicular to the undamaged skin.

F. Disassemble the Part [1] Doubler.

G. Drill holes through the damaged skin and core for the sleeves. Do not drill into the undamaged skin.

H. Examine the damaged area for signs of water, oil, fuel, dirt or other contamination in the honeycomb. Remove all fluids, dirt and other contamination.

(1) Use a vacuum or oil-free compressed air to remove water.

(2) Dry the honeycomb core. You can heat the core or use hot air at the wet area for 1 hour (minimum) at 150°F (66°C). Make sure that the temperature does not increase more than 5°F (3°C) per minute. Dry again if necessary.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE OR CAUSE DAMAGE TO THE ENVIRONMENT. REFER TO THE MATERIAL SAFETY DATA SHEETS (MSDS) AND THE LOCAL SAFETY PRECAUTIONS.

(3) If a honeycomb core has oil in it, then spray on MEK, MIBK, MPK, or acetone solvent to remove the contamination. Be careful to prevent damage to the adhesive on the opposite skin. Immediately remove the solvent from the core. Let the core become completely dry.

I. Remove all nicks, scratches, burrs, and sharp edges from the Part [1] Doubler and the bare surfaces of the initial parts.

J. Apply a chemical conversion coating to the Part [1] Doubler, the Part [2] Sleeves, and to the bare surfaces of the initial parts. Refer to SOPM 20-43-03, Class A.

K. Apply one layer of BMS 10-79 Type III primer to the Part [1] Doubler and to the bare surfaces of the initial parts. Refer to SOPM 20-44-04. Do not apply primer to the core.

L. Install the Part [2] Sleeves in the damaged skin and core with BMS 5-95 sealant. Apply sealant to all surfaces of the sleeves and make sure that sealant is on each end of the sleeves. Refer to 51-20-05, GENERAL.

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M. Install countersink repair washers at initial fastener locations with BMS 5-95 sealant, if necessary. Refer to 51-40-08, GENERAL.

N. Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. If disbond extends to the edges, fill the space between the skin and filler or other structure with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

O. Install rivets wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Shave rivets flush with the Part [1] Doubler. Refer to 51-40-02, GENERAL.

NOTE: The protruding heads of the rivets must be on the undamaged skin.

P. Re-Install the initial fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL.

Q. Apply a chemical conversion coating to the bare surfaces of the fasteners. If necessary apply a chemical conversion coating to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-43-03, Class A.

R. If necessary apply BMS 10-79 Type III primer to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-44-04.

S. Install a fillet seal to the edges of the Part [1] Doubler and fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL

T. Apply one layer of BMS 10-60, Type II enamel to the repair area. Refer to AMM TASK 51-21-73-370-803

U. Re-install the inboard trailing edge main flap if it was removed. Refer to AMM TASK 27-51-11-400-801.

V. Adjust the inboard trailing edge main flap. Refer to AMM TASK 27-51-11-820-801.

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Inboard Main Flap Trailing Edge Wedge Disbond Repair That Extends The Full Length Of Zone 1
Figure 203 (Sheet 1 of 4)

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(ROTATED 90° COUNTERCLOCKWISE)
UPPER PANEL REPAIR
A-A

2363178 S0000540243_V1

Inboard Main Flap Trailing Edge Wedge Disbond Repair That Extends The Full Length Of Zone 1
Figure 203 (Sheet 2 of 4)

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2363380 S0000540241_V1

Inboard Main Flap Trailing Edge Wedge Disbond Repair That Extends The Full Length Of Zone 1
Figure 203 (Sheet 3 of 4)

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(ROTATED 90° COUNTERCLOCKWISE)
LOWER PANEL REPAIR
B-B

2363728 S0000541050_V1

Inboard Main Flap Trailing Edge Wedge Disbond Repair That Extends The Full Length Of Zone 1
Figure 203 (Sheet 4 of 4)

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
(VIEW IN THE DOWN DIRECTION)
UPPER PANEL

NOTES

- IF DISBOND EXTENDS TO THE INBOARD OR OUTBOARD EDGE, NO CHAMFER IS REQUIRED ALONG THAT EDGE OF THE DOUBLER.
- THERE MUST BE TWO ROWS OF FASTENERS ON EACH SIDE OF THE DISBOND THAT DOES NOT EXTEND TO AN EDGE.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER. UP TO 1/32 INCH OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15BB5D() OR A BACR15BB5DD() RIVET.

2361905 S0000539122_V1

Upper Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 204 (Sheet 1 of 5)

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OUTBOARD AREA
REPAIR OF ZONE 1

A

2362158 S0000540239_V1

Upper Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 204 (Sheet 2 of 5)

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INBOARD AREA
REPAIR OF ZONE 1

B

(ROTATED 90° COUNTERCLOCKWISE)
UPPER PANEL REPAIR
A–A

2362237 S0000540238_V1

Upper Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 204 (Sheet 3 of 5)

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CENTER AREA
REPAIR OF ZONE 1

C

(ROTATED 90° COUNTERCLOCKWISE)
UPPER PANEL REPAIR
B–B

2362241 S0000540240_V1

Upper Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 204 (Sheet 4 of 5)

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(TYPICAL)

D

(TYPICAL)

E

2363729 S0000541051_V1

Upper Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 204 (Sheet 5 of 5)

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
(VIEW IN THE UP DIRECTION)
LOWER PANEL

NOTES

- IF DISBOND EXTENDS TO THE INBOARD OR OUTBOARD EDGE, NO CHAMFER IS REQUIRED ALONG THAT EDGE OF THE DOUBLER.
- THERE MUST BE TWO ROWS OF FASTENERS ON EACH SIDE OF THE DISBOND THAT DOES NOT EXTEND TO AN EDGE.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER. UP TO 1/32 INCH OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15BB5D() OR A BACR15BB5DD() RIVET.

2363481 S0000541029_V1

Lower Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 205 (Sheet 1 of 5)

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OUTBOARD AREA
REPAIR OF ZONE 1

A

2363488 S0000541030_V1

Lower Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 205 (Sheet 2 of 5)

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INBOARD AREA
REPAIR OF ZONE 1

B

(ROTATED 90° COUNTERCLOCKWISE)
LOWER PANEL REPAIR
A-A

2363491 S0000541031_V1

Lower Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 205 (Sheet 3 of 5)

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CENTER AREA
REPAIR OF ZONE 1

C

(ROTATED 90° COUNTERCLOCKWISE)
LOWER PANEL REPAIR
B–B

2363496 S0000541032_V1

Lower Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 205 (Sheet 4 of 5)

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(TYPICAL)

D

(TYPICAL)

E

2363731 S0000541052_V1

Lower Panel Inboard Main Flap Trailing Edge Wedge Disbond Repair of the Outboard,Center and Inboard Areas of Zone 1
Figure 205 (Sheet 5 of 5)

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REPAIR 10 - INBOARD MAIN FLAP TRAILING EDGE WEDGE LARGE DISBOND REPAIR

1. Applicability

A. REPAIR 10 is applicable to the lower skin of the inboard main flap trailing edge wedge (113A2500) as shown in Figure 201/REPAIR 10.

B. REPAIR 10 is applicable to disbonds of the lower skin that:

(1) Are less than or equal to 23 in. (584 mm) when measured along the aft edge of the wedge lower skin. For more than one disbond location, the sum of the lengths must be less than 23 in. (584 mm).

(2) Are less than 300 in^2 (1935 cm^2).

C. Refer to REPAIR 7 if the damage is larger than 300 in^2 (1935 cm^2) for a full skin replacement.

D. Refer to REPAIR 9 if the damage is longer than 23 in. (584 mm) when you measure along the aft edge of the wedge lower skin. The damage must also be less than 200 in^2 (1290 cm^2) in area and less than 4.5 in. (114.3 mm) away from the aft edge.

2. General

A. This repair is a time-limited repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repair. Replace this time-limited repair with a permanent repair no later than 24 months from the time that the repair is installed. Do a detailed visual inspection every 500 flight cycles. If any damage is found, then you must replace this repair with a permanent repair.

B. D = fastener diameter.

C. Keep a 2D minimum fastener edge margin.

D. All dimensions are inpf inches, unless they are shown differently.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE
INBOARD MAIN FLAP LOWER SKIN PANEL

A

2359237 S0000539239_V1

Inboard Main Flap Trailing Edge Wedge Location
Figure 201

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-30-10	RESINS AND ADHESIVES
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
51-70-04, REPAIR GENERAL	Repair Procedures for Wet Layup Materials
AMM 27-51-11-000-801	Inboard Trailing Edge Flap Removal (P/B 401)
AMM 27-51-11-400-801	Inboard Trailing Edge Flap Installation (P/B 401)
AMM 27-51-11-820-801	Inboard Trailing Edge Flap Adjustment (P/B 501)
AMM 51-21-73-370-803	BMS10-60 Topcoat Application (P/B 701)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-43-03	Chemical Conversion Coatings for Aluminum
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-05-01	Tap Test Inspection of Honeycomb Sandwich Structure

4. Repair Instructions

A. Get access to the damaged area. Remove the inboard trailing edge main flap, if necessary. Refer to AMM TASK 27-51-11-000-801.

B. If necessary, remove other parts for access.

C. Do a tap test inspection of the honeycomb panel to find the dimensions of the disbond area. Refer to 737 NDT Part 1, 51-05-01.

D. Remove any disbonded lower skin from the inboard main flap trailing edge wedge. Make sure that the cut extends to the forward and aft edges of the wedge lower skin. Make sure that the cut angle is parallel to the inboard edge of the wedge lower skin. Make sure that when you remove the skin, you adjust the dimensions to keep the same fastener spacing as shown in Figure 202/REPAIR 10. Make sure that you do not cause damage to the honeycomb core structure.

E. Remove all the fasteners in the lower skin of the wedge assembly. Refer to FASTENER INSTALLATION AND REMOVAL, 51-40-02.

F. Make the repair Part [1] Doubler and Part [2] Filler. Refer to Table 201/REPAIR 10 and Figure 202/REPAIR 10 for the material, thickness and dimensions of the repair parts. Do not drill holes at this time.

(1) Make the contour of the repair Part [1] Doubler and Part [2] Filler the same as the contour of the initial lower skin.

(2) The Part [1] Doubler extends forward of the trailing edge wedge skin to the adjacent lower main flap skin.

(3) Make sure that the Part [2] Filler is the same dimension and shape as the initial lower skin in the damage area.

(4) Chamfer the aft and forward edges of the Part [1] Doubler as shown in Figure 202/REPAIR 10.

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Table 201: REPAIR MATERIALS

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.600 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12
2	Filler	1	0.020 in. (0.051 cm) thick 7075-T6 bare sheet as given in QQ-A-250/12

G. Assemble the Part [1] Doubler and Part [2] Filler on the lower surface of the wedge.

H. Drill the fastener holes in the Part [1] Doubler, Part [2] Filler and initial parts. Refer to 51-40-05, GENERAL and Figure 202/REPAIR 10 for the fastener location, type and size.

(1) Drill the repair fastener holes perpendicular to the upper skin if the fasteners will go through the upper skin. For all other repair fasteners, drill the repair fastener holes perpendicular to the lower skin.

(2) Countersink the fastener holes in the Part [1] Doubler. Make sure that the countersink is perpendicular to the Part [1] Doubler. Refer to 51-40-08, GENERAL.

(3) Disassemble the Part [1] Doubler and Part [2] Filler.

(4) Increase the diameter of the holes in the initial lower skin and honeycomb core for the NAS1832( ) inserts. Make sure that you only drill the hole for the insert to a depth in the honeycomb core that will hold the insert fully or you can cause damage to the opposite skin.

I. Examine the damaged area for signs of water, oil, fuel, dirt or other contamination in the honeycomb. Remove all fluids, dirt and other contamination.

(1) Use a vacuum or oil-free compressed air to remove water.

(2) Dry the honeycomb core. You can heat the core or use hot air at the wet area for 1 hour (minimum) at 150°F (66°C). Make sure that the temperature does not increase more than 5°F (3°C) a minute. Dry again if necessary.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(3) If a honeycomb core has oil in it, then spray on MEK, MIBK, MPK, or acetone solvent to remove the contamination. Be careful to prevent damage to the adhesive on the opposite skin. Immediately remove the solvent from the core. Let the core become completely dry.

J. Remove all nicks, scratches, burrs, and sharp edges from the Part [1] Doubler, Part [2] Filler and the bare surfaces of the initial parts.

K. Remove initial adhesive, as necessary, from the closure ribs, trailing edge filler and spar of the inboard main flap trailing edge wedge. Do not remove initial adhesive from the honeycomb core.

L. Apply a chemical conversion coating to the Part [1] Doubler, Part [2] Filler and to the bare surfaces of the initial parts. Make sure that you apply the chemical conversion coating to the bare surfaces of the areas where you removed the initial adhesive. Refer to SOPM 20-43-03, Class A.

M. Apply one layer of BMS 10-79, Type III primer to the Part [1] Doubler, Part [2] Filler and to the bare surfaces of the initial parts. Make sure that you apply the primer to the bare surfaces of the areas where you removed the initial adhesive. Refer to SOPM 20-44-04.

N. Bond the NAS1832( ) inserts into the honeycomb core with BMS 5-28 Type 18 or Type 19 compound. Refer to RESINS AND ADHESIVES, 51-30-10 and Figure 202/REPAIR 10.

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O. Install countersink repair washers in the fastener holes in the initial skin with BMS 5-95 sealant where necessary. Refer to 51-40-08, GENERAL.

P. Do the steps that follow. Make sure that you do all of the steps that follow while the BMS 5-95 sealant is wet.

(1) Install the Part [2] Filler with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05, GENERAL. Apply even pressure to the Part [2] Filler. Make sure that the pressure is sufficient to squeeze out the sealant along the edges.

(2) Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. If you did not remove the wedge assembly, make sure that you apply BMS 5-95 sealant between the mating surfaces of the Part [1] Doubler and lower main flap skin. Refer to 51-20-05, GENERAL, 51-70-04, REPAIR GENERAL and Figure 202/REPAIR 10.

(3) Install the repair fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL. Shave the rivets flush with the Part [1] Doubler. Make sure that the contour of the Part [1] Doubler is same as the contour of the initial lower skin. Refer to 51-40-02, GENERAL and Figure 202/REPAIR 10.

(4) Re-Install the initial fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL. If you removed the wedge assembly, do not install fasteners along the forward edge of the Part [1] Doubler at this time.

NOTE: You can use a vacuum bag procedure to make sure that the Part [1] Doubler has the same contour as the initial lower skin. Refer to 51-70-04, REPAIR GENERAL.

Q. If you removed the wedge assembly, re-install the wedge assembly with BMS 5-95 sealant between the mating surfaces. Re-install the initial fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL.

R. Re-install the parts that you removed for access as necessary.

S. Apply a chemical conversion coating to the bare surfaces of the fasteners. If necessary, apply a chemical conversion coating to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-43-03, Class A.

T. If necessary, apply BMS 10-79 Type III primer to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-44-04.

U. Apply a fillet seal to the leading edge of the Part [1] Doubler with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

V. Apply one layer of BMS 10-60, Type II enamel to the repair area. Refer to AMM TASK 51-21-73-370-803.

W. Re-install the inboard trailing edge main flap if it was removed. Refer to AMM TASK 27-51-11-400-801.

X. Adjust the inboard trailing edge main flap. Refer to AMM TASK 27-51-11-820-801.

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Inboard Main Flap Trailing Edge Wedge Large Disbond Repair
Figure 202 (Sheet 1 of 5)

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(VIEW IN THE OUTBOARD DIRECTION)

B-B

2360495 S0000539246_V1

Inboard Main Flap Trailing Edge Wedge Large Disbond Repair
Figure 202 (Sheet 2 of 5)

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A

B

2360762 S0000539247_V1

Inboard Main Flap Trailing Edge Wedge Large Disbond Repair
Figure 202 (Sheet 3 of 5)

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2371130 S0000542724_V1

Inboard Main Flap Trailing Edge Wedge Large Disbond Repair
Figure 202 (Sheet 4 of 5)

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NOTES

D = FASTENER DIAMETER

FASTENER SYMBOLS

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER (UP TO 1/32-INCH DIAMETER OVERSIZE). INSTALL A COUNTERSINK REPAIR WASHER, AS NECESSARY. REFER TO SRM 51-40-08.

REPAIR FASTENER LOCATION. INSTALL A BACR15FT6AD() RIVET. AS AN ALTERNATIVE, INSTALL A BACR15FT6D() RIVET, BACR15FT6DD() RIVET, OR BACR15FT6KE() RIVET.

REPAIR FASTENER LOCATION. INSTALL A BACS12ER3K() BOLT WITH A NAS1832-3-4 INSERT. AS AN ALTERNATIVE, INSTALL A BACS12ER3K() BOLT WITH A NAS1832A3-4 INSERT OR A BACS12ER3K() BOLT WITH A NAS1832C3-4 INSERT. BOND THE INSERT INTO THE CORE WITH BMS 5-28 TYPE 18 OR 19 POTTING COMPOUND. REFER TO SRM 51-30-10.

2360780 S0000539300_V1

Inboard Main Flap Trailing Edge Wedge Large Disbond Repair
Figure 202 (Sheet 5 of 5)

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REPAIR 11 - OUTBOARD AFT FLAP TRAILING EDGE WEDGE REPAIR OF DISBONDS LESS THAN 75 SQUARE INCHES

1. Applicability

A. REPAIR 11 is applicable to the lower surface of the outboard aft flap trailing edge wedge 113A3710 as shown in Figure 201/REPAIR 11.

B. REPAIR 11 is applicable for localized disbond that:

(1) Is 3.0 in. (76.2 mm) or more away from the aft edge

(2) Is 3.0 in. (76.2 mm) or more away from the forward edge

(3) Is 3.0 in. (76.2 mm) or more away from the inboard edge, and less than 29.0 in. (736.6 mm) away from the inboard edge.

(4) Is located above the honeycomb core areas

(5) Has no damage to the honeycomb core.

C. Refer to REPAIR 12 and REPAIR 13 if the damage is larger than 75.0 in^2 (483.9 cm^2) or if the damage extends to an edge.

2. General

A. This repair is a time-limited repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repair. Replace this time-limited repair with a permanent repair no later than 24 months from the time that the repair is installed.

B. D = fastener diameter.

C. All dimensions are in inches, unless they are shown differently.

D. Keep a minimum of 0.50 in. (12.70 mm) corner radius for the repair part, unless they are shown differently.

E. Keep a 5D to 7D fastener spacing.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL

A

2358572 S0000538870_V2

Outboard Aft Flap Trailing Edge Wedge Location
Figure 201 (Sheet 1 of 2)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER AFT FLAP SKIN PANEL

B

2358990 S0000538872_V2

Outboard Aft Flap Trailing Edge Wedge Location
Figure 201 (Sheet 2 of 2)

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
AMM 27-51-22-000-801	Outboard Aft Flap Removal (P/B 401)
AMM 27-51-22-400-801	Outboard Aft Flap Installation (P/B 401)
AMM 51-21-73-370-803	BMS10-60 Topcoat Application (P/B 701)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-43-03	Chemical Conversion Coatings for Aluminum
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-05-01	Tap Test Inspection of Honeycomb Sandwich Structure

4. Repair Instructions

A. Get access to the damaged area. Remove the outboard aft flap from the aircraft if necessary. Refer to AMM TASK 27-51-22-000-801.

B. Do a tap test inspection of the honeycomb panel to find the dimensions of the disbond area. Refer to 737 NDT Part 1, 51-05-01.

C. Make the repair part. Refer to Table 201/REPAIR 11 and Figure 202/REPAIR 11.

(1) There must be two rows of fasteners on each side of the disbond area.

(2) The forward fasteners must be 1.5 in. (38.1 mm) or more away from the forward edge.

(3) The aft fasteners must be 2.0 in. (50.8 mm) or more away from the aft edge.

(4) The outboard fasteners must be 30.0 in. (762.0 mm) or less away from the inboard edge.

(5) The inboard fasteners must be 2.0 in. (50.8 mm) or more away from the inboard edge.

(6) The fastener edge margin must be a minimum of 0.75 in. (19.0 mm).

(7) Chamfer the edges of the Part [1] Doubler with 10:1 taper ratio to a maximum thickness of 0.020 in. (0.508 mm).

Table 201: REPAIR MATERIAL

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.600 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12

D. Assemble the Part [1] Doubler.

E. Drill the fastener holes through the Part [1] Doubler and the initial skin. The drill depth into the core must have sufficient depth for the fastener length, but must not damage the opposite initial skin. Countersink the fastener holes in the Part [1] Doubler. Refer to 51-40-05, GENERAL and Figure 202/REPAIR 11 for the fastener locations, type and size.

F. Disassemble the Part [1] Doubler.

G. Examine the damaged area for signs of water, oil, fuel, dirt or other contamination in the honeycomb. Remove all fluids, dirt and other contamination.

(1) Use a vacuum or oil-free compressed air to remove water.

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(2) Dry the honeycomb core. You can heat the core or use hot air at the wet area for 1 hour (minimum) at 150°F (66°C). Make sure that the temperature does not increase more than 5°F (3°C) a minute. Dry again if necessary.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(3) If a honeycomb core has oil in it, then spray on MEK, MIBK, MPK, or acetone solvent to remove the contamination. Be careful to prevent damage to the adhesive on the opposite skin. Immediately remove the solvent from the core. Let the core become completely dry.

H. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Doubler and the bare surfaces of the initial parts.

I. Apply a chemical conversion coating to the Part [1] Doubler and to the bare surfaces of the initial parts. Refer to SOPM 20-43-03, Class A.

J. Apply one layer of BMS 10-79, Type III primer to the Part [1] Doubler. Refer to SOPM 20-44-04.

K. Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05, GENERAL.

L. Install blind rivets wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Shave rivets flush. Refer to 51-40-02, GENERAL.

M. Apply a chemical conversion coating to the bare surfaces of the blind rivet heads. If necessary apply a chemical conversion coating to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-43-03, Class A.

N. If necessary apply BMS 10-79, Type III primer to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-44-04.

O. Apply one layer of BMS 10-60, Type II enamel to the repair area. Refer to AMM TASK 51-21-73-370-803.

P. Reinstall the outboard aft flap if it was removed. Refer to AMM TASK 27-51-22-400-801.

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TOP VIEW
SOME STRUCTURES NOT SHOWN FOR CLARITY

TYPICAL REPAIR OF LOWER SKIN IS SHOWN
(ROTATED 90° COUNTERCLOCKWISE)
A-A

FASTENER SYMBOLS

+ REPAIR FASTENER LOCATION. INSTALL A BACR15FP4E(), BLIND RIVET. AS AN ALTERNATIVE, INSTALL A NAS1399D4 OR NAS1739E4 BLIND RIVET.

2359043 S0000539112_V2

Outboard Aft Flap Trailing Edge Wedge Small Disbond Repair
Figure 202

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REPAIR 12 - OUTBOARD AFT FLAP TRAILING EDGE WEDGE REPAIR OF DISBONDS THAT ARE 200 SQUARE INCHES OR LESS

1. Applicability

A. REPAIR 12 is applicable to the outboard aft flap trailing edge wedge (113A3710) as shown in Figure 201/REPAIR 12.

B. REPAIR 12 is applicable to the lower surface of the outboard aft flap trailing edge wedge. Contact The Boeing Company if the upper surface is damaged.

C. REPAIR 12 is applicable to damage that:

(1) Has disbond less than 200 in^2 (1290 cm^2) of area.

(2) Does not extend more than 5 in. (127 mm) from the aft edge (but can include the aft edge).

(3) Does not extend more than 30 in. (762 mm) from the inboard edge (but can include the inboard edge).

D. Refer to REPAIR 13 if the disbond is larger than 200 in^2 (1290 cm^2).

2. General

A. This repair is a time-limited repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repair. Replace this time-limited repair with a permanent repair no later than 24 months from the time that the repair is installed.

B. D = fastener diameter.

C. Keep a minimum of 0.50 in. (12.70 mm) corner radius of all repair parts, unless it is shown differently.

D. Keep a 1.25 in. (31.75 mm) to 1.75 in. (44.45 mm) fastener spacing, unless it is shown differently.

NOTE: For fasteners at the inboard edge, a spacing of 5D to 7D is necessary.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

LOWER SKIN PANEL

A

2359076 S0000539292_V1

Wing Outboard Aft Flap Trailing Edge - Trailing Edge Wedge Locations
Figure 201 (Sheet 1 of 2)

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DISBOND NOT PERMITTED IN THIS AREA

NOTE: ALL DIMENSIONS ARE INCHES (mm).

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

LOWER SKIN PANEL IS SHOWN

B

2359111 S0000539293_V1

Wing Outboard Aft Flap Trailing Edge - Trailing Edge Wedge Locations
Figure 201 (Sheet 2 of 2)

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
AMM 27-51-22-000-801	Outboard Aft Flap Removal (P/B 401)
AMM 27-51-22-400-801	Outboard Aft Flap Installation (P/B 401)
AMM 51-21-73-370-803	BMS10-60 Topcoat Application (P/B 701)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-43-03	Chemical Conversion Coatings for Aluminum
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-05-01	Tap Test Inspection of Honeycomb Sandwich Structure

4. Repair Instructions

A. Get access to the damaged area. Remove the outboard aft flap if necessary. Refer to AMM TASK 27-51-22-000-801.

B. Do a tap test inspection of the honeycomb panel to find the dimensions of the disbond area. Refer to 737 NDT Part 1, 51-05-01.

C. Make the Part [1] Doubler and Part [2] Sleeves. Refer to Table 201/REPAIR 12 and Figure 202/REPAIR 12.

(1) The end of the Part [2] Sleeve in the honeycomb core must be flat against the undamaged skin. Make sure that the opposite end is flush with the outside of the damaged skin.

NOTE: Sleeves are only necessary in areas that have a core.

(2) There must be two rows of fasteners on each side of the disbond.

NOTE: If the repair extends to the inboard or aft edge, the two fastener row requirement is not applicable at that edge.

(3) The forward fasteners must be no further than 7.5 in. (190.5 mm) from the aft edge.

(4) Fastener edge margin must be a minimum of 0.75 in. (19.05 mm) which includes the chamfer, except:

(a) If the Part [1] Doubler extends to the aft edge of the aft flap, the edge margin is 0.4 in. (10.2 mm) at the aft edge.

(b) If the Part [1] Doubler extends to the inboard edge of the aft flap, use the initial or reference fastener edge margin on the lower skin.

(5) Chamfer all edges of the Part [1] Doubler with a 10:1 taper ratio to a maximum thickness of 0.020 in. (0.508 mm) as shown in Figure 202 (Sheet 3), except:

(a) If the disbond extends to the aft edge of the aft flap, the chamfer along the aft edge is 0.048 in. (1.219 mm) x 0.200 in. (5.080 mm).

(b) If the disbond extends to the inboard edge of the aft flap or below the fairing, no chamfer is necessary along that edge.

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Table 201: REPAIR MATERIALS

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.600 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12
2	Sleeve	as necessary	Any 2000 series or 7000 series aluminum extrusion. 0.25 in. (6.35 mm) outside diameter x 0.16 in. (4.06 mm) inside diameter

D. Assemble the Part [1] Doubler. If the repair extends to the inboard edge, then remove the initial fasteners as shown in Figure 202/REPAIR 12, if necessary.

E. Drill holes through the Part [1] Doubler, the damaged skin and the undamaged skin for the rivets. Countersink the fastener holes in the Part [1] Doubler. Refer to 51-40-05, GENERAL and Figure 202/REPAIR 12 for the fastener location, type and size.

(1) If the doubler extends to the inboard edge, make sure to use the initial fastener hole locations.

(2) Holes must be perpendicular to the undamaged skin.

F. Disassemble the Part [1] Doubler.

G. Drill holes through the damaged skin and core for the sleeves. Do not drill into the undamaged skin.

H. Examine the damaged area for signs of water, oil, fuel, dirt or other contamination in the honeycomb. Remove all fluids, dirt and other contamination.

(1) Use a vacuum or oil-free compressed air to remove water.

(2) Dry the honeycomb core. You can heat the core or use hot air at the wet area for 1 hour (minimum) at 150°F (77°C). Make sure that the temperature does not increase more than 5°F (3°C) per minute. Dry again if necessary.

WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE OR CAUSE DAMAGE TO THE ENVIRONMENT. REFER TO THE MATERIAL SAFETY DATA SHEETS (MSDS) AND THE LOCAL SAFETY PRECAUTIONS.

(3) If a honeycomb core has oil in it, then spray on MEK, MIBK, MPK, or acetone solvent to remove the contamination. Be careful to prevent damage to the adhesive on the opposite skin. Immediately remove the solvent from the core. Let the core become completely dry.

I. Remove all nicks, scratches, burrs, and sharp edges from the repair parts and the bare surfaces of the initial parts.

J. Apply a chemical conversion coating to the Part [1] Doubler, the Part [2] Sleeves, and to the bare surfaces of the initial parts. Refer to SOPM 20-43-03, Class A.

K. Apply one layer of BMS 10-79, Type III primer to the Part [1] Doubler and to the bare surfaces of the initial parts. Refer to SOPM 20-44-04. Do not apply primer to the core.

L. Install the Part [2] Sleeves in the damaged skin and core with BMS5-95 sealant. Apply sealant to all surfaces of the sleeves and make sure that sealant is on each end of the sleeves. Refer to 51-20-05, GENERAL.

M. Install countersink repair washers at initial fastener locations with BMS 5-95 sealant, if necessary. Refer to 51-40-08, GENERAL.

N. Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. If disbond extends to the edges, fill the space between the skin and filler or inboard rib with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

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O. Install the rivets wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Shave rivets flush with the Part [1] Doubler. Refer to 51-40-02, GENERAL

NOTE: The protruding heads of the rivets must be on the undamaged skin.

P. Re-Install the initial fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL.

Q. Apply a chemical conversion coating to the bare surfaces of the rivet heads. If necessary apply a chemical conversion coating to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-43-03, Class A.

R. If necessary apply BMS 10-79, Type III primer to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-44-04.

S. Apply a fillet seal to the edges of the Part [1] Doubler and fill all gaps with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

T. Apply one layer of BMS 10-60, Type II enamel to the repair area. Refer to AMM TASK 51-21-73-370-803.

U. Re-install the outboard aft flap trailing edge wedge if it was removed. Adjust the outboard aft flap trailing edge wedge. Refer to AMM TASK 27-51-22-400-801.

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KEEP A 5D TO 7D REPAIR FASTENER SPACING AT THE INBOARD EDGE.

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

(LOWER PANEL IS SHOWN)

(VIEW IN THE UP DIRECTION)

2359207 S0000539302_V1

Outboard Aft Flap Trailing Edge Wedge Repair of Disbond Less Than 200 Square Inches
Figure 202 (Sheet 1 of 3)

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(ROTATED 90° COUNTERCLOCKWISE)

LOWER PANEL REPAIR
A-A

2360095 S0000539303_V1

Outboard Aft Flap Trailing Edge Wedge Repair of Disbond Less Than 200 Square Inches
Figure 202 (Sheet 2 of 3)

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(TYPICAL)

A

(TYPICAL)

B

NOTES

- ALL DIMENSIONS ARE IN INCHES (mm).

1 IF THE PART [1] DOUBLER EXTENDS TO THE AFT EDGE, THE FASTENER EDGE MARGIN IS 0.40 (10.16) FROM THE AFT EDGE. IF THE PART [1] DOUBLER EXTENDS TO THE INBOARD EDGE, USE THE INITIAL OR REFERENCE FASTENER EDGE MARGIN ON THE LOWER SKIN.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER, UP TO 1/32 INCH OVERSIZE.

REPAIR FASTENER LOCATION. INSTALL A BACR15BB5D() OR A BACR15BB5DD() RIVET.

2370832 S0000542705_V1

Outboard Aft Flap Trailing Edge Wedge Repair of Disbond Less Than 200 Square Inches
Figure 202 (Sheet 3 of 3)

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REPAIR 13 - OUTBOARD AFT FLAP TRAILING EDGE WEDGE LARGE DISBOND REPAIR

1. Applicability

A. REPAIR 13 is applicable to the lower skin of the outboard aft flap trailing edge wedge (113A3710) as shown in Figure 201/REPAIR 13.

B. REPAIR 13 is not applicable:

(1) If the disbond is more than 23.50 in. (596.90 mm) from the inboard edge when you measure along the forward edge of the outboard aft flap wedge lower skin. Refer to Figure 202/REPAIR 13.

(2) If the disbond is more than 27.75 in. (704.85 mm) from the inboard edge when you measure along the aft edge of the outboard aft flap wedge lower skin. Refer to Figure 202/REPAIR 13.

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

2365085 S0000541359_V1

Outboard Aft Flap Trailing Edge Wedge Location
Figure 201

2. General

A. This repair is a time-limited repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repair. Replace this time-limited repair with a permanent repair no later than 24 months from the time that the repair is installed. Do a detailed visual inspection every 500 flight cycles. If any damage is found, then you must replace this repair with a permanent repair.

B. D = fastener diameter.

C. Keep a 2D minimum fastener edge margin, unless it is shown differently.

D. All dimensions are in inches, unless they are shown differently.

3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions

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(Continued)

Reference	Title
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-30-10	RESINS AND ADHESIVES
51-40-02, GENERAL	Fastener Installation and Removal
51-40-05, GENERAL	Fastener Hole Sizes
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
51-70-04, REPAIR GENERAL	Repair Procedures for Wet Layup Materials
AMM 27-51-22-000-801	Outboard Aft Flap Removal (P/B 401)
AMM 27-51-22-400-801	Outboard Aft Flap Installation (P/B 401)
AMM 51-21-73-370-803	BMS10-60 Topcoat Application (P/B 701)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-43-03	Chemical Conversion Coatings for Aluminum
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-05-01	Tap Test Inspection of Honeycomb Sandwich Structure

4. Repair Instructions

A. Get access to the damaged area. Remove the outboard aft flap from the aircraft, if necessary. Refer to AMM TASK 27-51-22-000-801.

B. If necessary, remove other parts for access.

C. Do a tap test inspection of the honeycomb panel to find the dimensions of the disbond area. Refer to 737 NDT Part 1, 51-05-01.

D. Remove any disbonded lower skin from the outboard aft flap trailing edge wedge. Refer to Figure 202/REPAIR 13 for the maximum dimension of lower skin you can remove. You can remove less skin than the maximum dimension, but make sure that the outboard edge angle you cut is equal to the angle as shown in Figure 203/REPAIR 13. Make sure that you do not cause damage to the honeycomb core structure.

E. Remove all the fasteners in the lower skin of the outboard aft flap wedge within 30 in. (76.2 cm) of the inboard edge. If the damage is less than 30 in. (76.2 cm), remove all the necessary fasteners in the damaged area.

F. Make the repair Part [1] Doubler and Part [2] Filler. Refer to Table 201/REPAIR 13 and Figure 203/REPAIR 13 for the material and thickness of the repair parts. Do not drill holes at this time.

(1) If the damage is less than the maximum dimensions, you can make a Part [1] Doubler that is smaller than the maximum dimensions shown in Figure 203/REPAIR 13. Adjust the dimensions of the Part [1] Doubler to keep the same fastener spacing as shown in Figure 203/REPAIR 13.

(2) Make the contour of the repair Part [1] Doubler and Part [2] Filler the same as the contour of the initial lower skin.

(3) The Part [1] Doubler extends forward of the aft flap trailing edge wedge skin to the adjacent lower outboard leading edge skin.

(4) The angle cut on the outboard edge of the Part [1] Doubler must be as shown in Figure 203 (Sheet 1) to align with the flap fairing.

(5) Make sure that the Part [1] Doubler is trimmed around the drain holes in the leading edge skin.

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(6) Make sure that the Part [2] Filler is the same dimension and shape as the initial lower skin in the damage area.

(7) Chamfer the aft edge, forward edge and inboard cutout of the Part [1] Doubler as shown in Figure 203/REPAIR 13.

Table 201: REPAIR MATERIAL

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.600 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12
2	Filler	1	0.020 in. (0.508 mm) thick 7075-T6 bare sheet as given in QQ-A-250/12

G. Assemble the Part [1] Doubler and Part [2] Filler on the area where you remove the initial lower skin.

H. Drill the fastener holes in the Part [1] Doubler and Part [2] Filler. Refer to 51-40-05, GENERAL and Figure 203/REPAIR 13 for the fastener location, type and size.

(1) If you use a Part [1] Doubler that is less than the maximum dimensions, make sure to keep the same fastener spacing shown. You must install all the outboard edge fasteners as shown. Refer to Figure 203/REPAIR 13.

(2) Make sure that you drill holes for all initial and repair fastener locations in the Part [1] Doubler and Part [2] Filler, where necessary.

(3) Drill the repair fastener holes perpendicular to the upper skin if the fasteners will go through the upper skin. For all other repair fasteners, drill the repair fastener holes perpendicular to the lower skin.

(4) For the NAS1832( ) inserts, the drill depth into the core must have sufficient depth for the insert length, but must not damage the opposite initial skin.

(5) For the blind rivets, the drill depth into the core must have sufficient depth for the fastener length, but must not damage the opposite initial skin.

(6) Countersink the fastener holes on the Part [1] Doubler. Make sure that the countersink is perpendicular to the Part [1] Doubler. Refer to 51-40-08, GENERAL.

(7) Disassemble the Part [1] Doubler and Part [2] Filler.

(8) Increase the diameter of the holes in the honeycomb core and the Part [2] Filler for the NAS1832( ) inserts. Make sure that you only drill the hole for the insert to a depth that will hold the insert fully or you can cause damage to the opposite skin.

I. Examine the damaged area for signs of water, oil, fuel, dirt or other contamination in the honeycomb. Remove all fluids, dirt and other contamination.

(1) Use a vacuum or oil-free compressed air to remove water.

(2) Dry the honeycomb core. You can heat the core or use hot air at the wet area for 1 hour (minimum) at 150°F (66°C). Make sure that the temperature does not increase more than 5°F (3°C) a minute. Dry again if necessary.

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WARNING: DO NOT GET SOLVENTS IN YOUR MOUTH, YOUR EYES, OR ON YOUR SKIN. DO NOT BREATHE THE FUMES FROM SOLVENTS. SOLVENTS ARE DANGEROUS MATERIALS. SOLVENTS CAN BE FLAMMABLE. OBEY THE MATERIAL SAFETY DATA SHEETS (MSDS) FOR SOLVENTS. OBEY LOCAL REGULATIONS FOR THE CORRECT PROCEDURES TO USE OR DISCARD SOLVENTS. SOLVENTS CAN CAUSE INJURIES TO PERSONNEL AND DAMAGE TO EQUIPMENT.

(3) If a honeycomb core has oil in it, then spray on MEK, MIBK, MPK, or acetone solvent to remove the contamination. Be careful to prevent damage to the adhesive on the opposite skin. Immediately remove the solvent from the core. Let the core become completely dry.

J. Remove all nicks, scratches, burrs, and sharp edges from the Part [1] Doubler, Part [2] Filler and the bare surfaces of the initial parts.

K. Remove initial adhesive, as necessary, from the inboard closure rib, trailing edge filler and spar of the outboard aft flap trailing edge wedge. Do not remove initial adhesive from the honeycomb core.

L. Apply a chemical conversion coating to the Part [1] Doubler, Part [2] Filler and to the bare surfaces of the initial parts. Make sure that you apply the chemical conversion coating to the bare surfaces of the areas where you removed the initial adhesive. Refer to SOPM 20-43-03, Class A.

M. Apply one layer of BMS 10-79, Type III primer to the Part [1] Doubler, Part [2] Filler and to the bare surfaces of the initial parts. Make sure that you apply the primer to the bare surfaces of the areas where you removed the initial adhesive. Refer to SOPM 20-44-04.

N. Bond the NAS1832( ) inserts into the honeycomb core with BMS 5-28 Type 18 or Type 19 compound. Refer to RESINS AND ADHESIVES, 51-30-10 and Figure 203/REPAIR 13.

O. Re-install the leading edge skin, if you removed it.

P. Install countersink repair washers in the fastener holes in the initial skin with BMS 5-95 sealant where necessary. Refer to 51-40-08, GENERAL.

Q. Do the steps that follow. Make sure that you do all of the steps that follow while the BMS 5-95 sealant is wet.

(1) Install the Part [2] Filler with BMS 5-95 sealant between the mating surfaces (it is not necessary to apply BMS 5-95 sealant on the core). Refer to 51-20-05, GENERAL. Apply even pressure to the edges of the disbonded trailing edge wedge. Make sure that the pressure is sufficient to squeeze out the sealant along the edge.

(2) Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. Make sure that you apply BMS 5-95 sealant between the mating surfaces of the Part [1] Doubler and lower leading edge skin. Refer to 51-20-05, GENERAL, 51-70-04, REPAIR GENERAL, and Figure 203/REPAIR 13.

(3) Install the repair fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL. Shave the rivets flush with the Part [1] Doubler. Make sure that the contour of the Part [1] Doubler is same as the contour of the initial lower skin. Refer to 51-40-02, GENERAL and Figure 203/REPAIR 13.

(4) Re-Install the initial fasteners wet with BMS 5-95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL.

NOTE: You can use a vacuum bag procedure to make sure that the Part [1] Doubler has the same contour as the initial lower skin. Refer to 51-70-04, REPAIR GENERAL.

R. Re-install the other parts that you removed for access as necessary.

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S. Apply a chemical conversion coating to the bare surfaces of the fasteners. If necessary apply a chemical conversion coating to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-43-03, Class A.

T. If necessary, apply BMS 10-79 Type III primer to any bare surfaces of the Part [1] Doubler and the initial parts. Refer to SOPM 20-44-04.

U. Apply a fillet seal to the leading edge and the outboard edge of the Part [1] Doubler with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

V. Apply one layer of BMS 10-60, Type II enamel to the repair area. Refer to AMM TASK 51-21-73-370-803.

W. Re-install the outboard aft flap if it was removed. Refer to AMM TASK 27-51-22-400-801.

X. Adjust the outboard aft flap as necessary. Refer to AMM TASK 27-51-22-400-801.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
LOWER SKIN PANEL

DISBOND PERMITTED IN THIS AREA

NOTE: ALL DIMENSIONS ARE INCHES (mm).

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

(LOWER SKIN PANEL IS SHOWN)

A

2365233 S0000541361_V1

Outboard Aft Flap Trailing Edge Wedge Repair Area
Figure 202

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Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair
Figure 203 (Sheet 1 of 6)

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Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair
Figure 203 (Sheet 2 of 6)

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2366869 S0000541368_V1

Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair
Figure 203 (Sheet 3 of 6)

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2371129 S0000542715_V2

Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair
Figure 203 (Sheet 4 of 6)

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NOTES

- ALL DIMENSIONS ARE IN INCHES (mm).
- D = FASTENER DIAMETER
- TOLERANCE TO BE +/- 0.010 (.254) ON ALL DIMENSIONS UNLESS IT IS SHOWN DIFFERENTLY.

1 SPACING IS APPROXIMATE AND MAY VARY SLIGHTLY TO ACCOMMODATE VARIATIONS IN DOUBLER SIZE. USE AN APPROPRIATE NUMBER OF INSERTS SO THAT WHEN EQUALLY SPACED THE DIMENSION IS AS CLOSE TO 2.50 INCHES AS POSSIBLE.

2 DIMENSION MUST BE MEASURED ALONG THE UPPER SURFACE.

3 KEEP A MINIMUM OVERLAP OF 1.50 INCHES BETWEEN THE DOUBLER AND THE BONDED SKIN.

4 SPACING IS APPROXIMATE AND MAY VARY SLIGHTLY TO ACCOMMODATE VARIATIONS IN DOUBLER SIZE. USE AN APPROPRIATE NUMBER OF FASTENERS SO THAT WHEN EQUALLY SPACED THE DIMENSION IS AS CLOSE TO 1.50 INCH AS POSSIBLE.

5 GAP BETWEEN WEDGE SKIN AND PART [2] FILLER TO BE 0.000 - 0.050 (0 - 1.27).

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER (UP TO 1/32-INCH DIAMETER OVERSIZE). INSTALL A COUNTERSINK REPAIR WASHER, AS NECESSARY. REFER TO SRM 51-40-08.

REPAIR FASTENER LOCATION. INSTALL A BACR15FT6AD() RIVET. AS AN ALTERNATIVE, INSTALL A BACR15FT6D() RIVET, BACR15FT6DD() RIVET, OR BACR15FT6KE() RIVET.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER (UP TO 1/32-INCH DIAMETER OVERSIZE). INSTALL A COUNTERSINK REPAIR WASHER, AS NECESSARY. REFER TO SRM 51-40-08.

INITIAL FASTENER LOCATION. INSTALL THE SAME TYPE AND DIAMETER FASTENER AS THE INITIAL FASTENER (UP TO 1/32-INCH DIAMETER OVERSIZE). INSTALL A COUNTERSINK REPAIR WASHER, AS NECESSARY. REFER TO SRM 51-40-08.

REPAIR FASTENER LOCATION. INSTALL A BACS12ER3K() BOLT WITH A NAS1832-3-4 INSERT. AS AN ALTERNATIVE, INSTALL A BACS12ER3K() BOLT WITH A NAS1832A3-4 INSERT. AS AN ALTERNATIVE, INSTALL A BACS12ER3K() BOLT WITH A NAS1832C3-4 INSERT. BOND THE INSERT INTO THE CORE WITH BMS 5-28 TYPE 18 OR 19 POTTING COMPOUND. REFER TO SRM 51-30-10.

REPAIR FASTENER LOCATION. INSTALL A BACR15FP4E() BLIND RIVET. AS AN ALTERNATIVE, INSTALL A NAS1399D4 BLIND RIVET, OR NAS1739E4 BLIND RIVET.

2366870 S0000541378_V2

Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair
Figure 203 (Sheet 5 of 6)

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FASTENER SYMBOLS (CONTINUED)

INITIAL FASTENER LOCATION. INSTALL A NAS1399D6() BLIND RIVET. AS AN ALTERNATIVE, INSTALL A NAS1739B6() BLIND RIVET, NAS1739E6() BLIND RIVET, NAS1739A6() BLIND RIVET, BACR15FP6B() BLIND RIVET OR BACR15FP6E() BLIND RIVET. INSTALL A COUNTERSINK REPAIR WASHER, AS NECESSARY. REFER TO SRM 51-40-08.

REPAIR FASTENER LOCATION. INSTALL A BACR15FT6AD() RIVET. AS AN ALTERNATIVE, INSTALL A BACR15FT6D() RIVET, BACR15FT6DD() RIVET, BACR15FT6KE() RIVET, OR BACB30YP6K() HEX-DRIVE BOLT WITH BACC30BL6 MATING COLLAR.

2418652 S0000558993_V1

Outboard Aft Flap Trailing Edge Wedge Large Disbond Repair
Figure 203 (Sheet 6 of 6)

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REPAIR 14 - OUTBOARD AFT FLAP TRAILING EDGE WEDGE

1. Applicability

A. REPAIR 14 is applicable to the outboard aft flap trailing edge wedge bonded assembly 113A3710-11, -12 as shown in Figure 201/REPAIR 14.

B. This repair is applicable to disbond of a skin that is 30.0 in. (762.0 mm) or less away from the inboard edge.

2. General

A. This repair is a permanent repair. Refer to 51-00-06, GENERAL for repair categories and definitions.

B. Do the repair as given in Paragraph 4./REPAIR 14.

C. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01, GENERAL.

D. Keep a minimum of 0.50 in. (12.70 mm) corner radius for the repair part, unless they are shown differently.

E. Make sure the repair and initial parts have a surface smoothness of 125 microinches Ra or smoother. Refer to 51-20-13, GENERAL

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

2367019 S0000542093_V1

Outboard Aft Flap Trailing Edge Wedge Location
Figure 201 (Sheet 1 of 3)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANEL

A

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL

B

2367020 S0000542094_V1

Outboard Aft Flap Trailing Edge Wedge Location
Figure 201 (Sheet 2 of 3)

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANEL IS SHOWN, UPPER SKIN PANEL IS SIMILAR

C

2367022 S0000542096_V1

Outboard Aft Flap Trailing Edge Wedge Location
Figure 201 (Sheet 3 of 3)

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-20-13, GENERAL	Surface Roughness Finish Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
51-70-10, REPAIR 3	Septumized Core Repairs
51-70-10, REPAIR 4	Repairs to Large Damage
51-70-10, REPAIR GENERAL	Aluminum Skin/Aluminum Honeycomb Panel Repairs
AMM 27-51-22-000-801	Outboard Aft Flap Removal (P/B 401)
AMM 27-51-22-400-801	Outboard Aft Flap Installation (P/B 401)
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
737 NDT Part 1, 51-04-00	Ultrasonic

4. Repair Instructions

A. Get access to the damaged area.

(1) If an autoclave repair is used, then:

(a) Remove the outboard aft flap from the aircraft. Refer to AMM TASK 27-51-22-000-801.

(b) Remove all components from the outboard aft flap trailing edge wedge bonded assembly.

(2) If a vacuum bag repair is used, then remove the outboard aft flap from the aircraft if necessary. Refer to AMM TASK 27-51-22-000-801.

B. Cut and remove the damaged skin and core, as necessary. Refer to 51-70-10, REPAIR GENERAL, Paragraph 7.

C. Make the repair parts. Refer to 51-70-10, REPAIR GENERAL, Paragraph 8.

(1) Make the repair doubler and the repair skin from the same material and thickness as the initial skin.

(2) Make the repair doubler with 1.5 in. (38.1 mm) overlap to the initial skin.

(3) If the repair will cover the initial fasteners, then:

(a) Remove the initial fasteners in the repair area.

(b) Drill the fastener holes in the repair doubler and the repair skin.

D. Remove all nicks, scratches, burrs, and sharp edges from the repair doubler and the bare surfaces of the initial parts.

E. Replace the damaged core if necessary. Refer to 51-70-10, REPAIR GENERAL, Paragraph 9. The maximum cure temperature of the repair core is 260°F (127°C).

F. Install countersink repair washers, if necessary, with BMS 5-95 sealant. Refer to 51-40-08, GENERAL.

G. Install the repair. Refer to Table 201/REPAIR 14 for the repair data and 51-70-10, REPAIR GENERAL, Paragraph 20, for the repair instructions. Make sure that you use and applicable tool to hold all of the parts in their positions during the repair procedure.

NOTE: In most (but not all) conditions, the tool must be the same contour as the part to be cured.

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Table 201: Permanent Repair Data for the Outboard Aft Flap Trailing Edge Wedge Bonded Assembly

REPAIR TYPE	FILM ADHESIVES			
	AUTOCLAVE		VACUUM BAG	
REPAIR CURE TEMPERATURE	250°F (121°C)	350°F (177°C) *[1]	250°F (121°C)	350°F (177°C)
DAMAGE SIZE *[2]	Permitted for skin damage less than 200 in^2 (1290 cm^2)	No limit	Permitted for skin damage less than 64 in^2 (413 cm^2)	Repair is not permitted
SURFACE PREPARATION	BOEGEL (AC-130), or TANK PAA, or PACS, or PANTA	BOEGEL (AC-130), or TANK PAA, or PACS, or PANTA	BOEGEL (AC-130), or TANK PAA, or PACS, or PANTA	----
BMS 5-121 POSITIONING FABRIC*[3]	Optional	Optional	Optional	----
ADHESIVES	BMS 5-101, Type II (EA 9696, EA-9628 and AF-163)	BMS 5-137, Type II (Cytec FM-377S)	BMS 5-101, Type II (EA 9696, EA-9628 and AF-163)	----
REPAIRS PER LIFETIME	No limit	1 Repair	No limit	----
REPAIR PROCEDURE	51-70-10, REPAIR 3 51-70-10, REPAIR 4	51-70-10, REPAIR 3 51-70-10, REPAIR 4	51-70-10, REPAIR 3 51-70-10, REPAIR 4	----

*[1] If there is a 250°F (121°C) initial repair in the repair area, you must replaced the initial repair with this 350°F (177°C) autoclave repair.

*[2] The damage size is applicable to each of the upper and lower skin. You can have damage on both the upper and lower skin.

*[3] Put the positioning fabric between the film adhesive and the repair doubler or initial skin. Do not put the positioning fabric between the adhesive and the honeycomb core.

H. Inspect the fit of the repair parts. Refer to 51-70-10, REPAIR GENERAL, Paragraph 10.

I. Do an ultrasonic bond inspection of metal-to-metal bonding and honeycomb structure. Refer to 737 NDT Part 1, 51-04-00. Do not do a tap test inspection of the honeycomb panel.

J. Install the repair fasteners wet with BMS 5-95 in the initial fastener holes. Refer to 51-40-02, GENERAL, 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805.

K. Clean, seal and finish the repair. Refer to 51-70-10, REPAIR GENERAL, Paragraph 22.

L. Re-install the components of the outboard aft flap if they were removed.

M. Re-install the outboard aft flap if it was removed. Refer to AMM TASK 27-51-22-400-801.

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IDENTIFICATION 1 - WING TRAILING EDGE INBOARD MAIN FLAP STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

H41875 S0006594905_V1

Wing Trailing Edge Inboard Main Flap Structure Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable T.E. Functional Product Collector
113A2001	Flap Installation - Inboard Trailing Edge Flap
113A2002	Flap Assembly - Inboard Trailing Edge Flap
113A2100	Main Flap Assembly - Inboard Trailing Edge Flap
113A2107	Fitting Assembly - Fairing Attach, Inboard Main Flap
113A2110	Front Spar Assembly - Inboard Main Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A2130	Rear Spar Assembly - Inboard Main Flap
113A2215	Support Fitting Assembly - Crankshaft, Aft Flap Drive, Inboard Main Flap
113A2251	Fitting Assembly - Inboard Bellcrank Support, Inboard Main Flap
113A2252	Support Assembly - Outboard, Bellcrank Aft Flap Drive, Inboard Main Flap
113A2280	Support Assembly - Inboard, Aft Flap Track Support
113A2285	Support Assembly - Outboard, Aft Flap Track Support
113A2500	Wedge Assembly - Bonded Part, Inboard Main Flap

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

H41876 S0006594907_V3

Wing Trailing Edge Inboard Main Flap Structure Identification
Figure 2 (Sheet 1 of 5)

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TORQUE TUBE SUPPORT SRUCTURE IDENTIFICATION

A

H41877 S0006594908_V1

Wing Trailing Edge Inboard Main Flap Structure Identification
Figure 2 (Sheet 2 of 5)

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REAR SPAR ASSEMBLY IDENTIFICATION

B

FRONT SPAR ASSEMBLY IDENTIFICATION

C

H41879 S0006594909_V1

Wing Trailing Edge Inboard Main Flap Structure Identification
Figure 2 (Sheet 3 of 5)

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TRAILING EDGE WEDGE ASSEMBLY IDENTIFICATION

D

H41878 S0006594910_V1

Wing Trailing Edge Inboard Main Flap Structure Identification
Figure 2 (Sheet 4 of 5)

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LOWER FLAP SURFACE IDENTIFICATION

E

H41880 S0006594911_V1

Wing Trailing Edge Inboard Main Flap Structure Identification
Figure 2 (Sheet 5 of 5)

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[1]	Rib - Wedge		7050-T7451 plate as given in AMS 4050	
[2]	Rib - Inspar		7050-T7451 plate as given in AMS 4050	
[3]	Fitting		7050-T7451 plate as given in AMS 4050	
[4]	Tee - Cove Support		BAC1505-100967 2024-T8511 extrusion as given in QQ-A-200/3	
[5]	Rib - Inspar		7050-T7451 plate as given in AMS 4050	
[6]	Stiffener		7050-T7451 plate as given in AMS 4050	
[7]	Rib - LE		7050-T7451 plate as given in AMS 4050	
[8]	Rib - LE		7050-T7451 plate as given in AMS 4050	
[9]	Rib - LE	0.90 (22.86)	7050-T7451 plate as given in AMS 4050	
[11]	Rib - Closure		7050-T7451 plate as given in AMS 4050	
[12]	Rib - LE		7050-T7451 plate as given in AMS 4050	
[13]	Support Fitting		7050-T7451 plate as given in AMS 4050	
[14]	Rib - LE		7050-T7451 plate as given in AMS 4050	
[15]	Fitting - Torque Tube		Ti-6Al-4V plate as given in Mil-T-9046, Code AB-1, in the annealed condition	
[16]	Fitting - Torque Tube		Ti-6Al-4V plate as given in Mil-T-9046, Code AB-1, in the annealed condition	
[17]	Chord - Upper		7050-T7451 plate as given in AMS 4050	
[18]	Chord - Lower		7050-T7451 plate as given in AMS 4050	
[19]	Web	0.100 (2.54)	2024-T3 sheet as given in QQ-A-250/4	
[20]	Web	0.160 (0.41)	2024-T3 sheet as given in QQ-A-250/4	
[21]	Web	0.200 (0.51)	2024-T3 sheet as given in QQ-A-250/4	
[22]	Stiffener		BAC1503-100180 7075-T73511 extrusion as given in QQ-A-200/11	
[23]	Stiffener	0.90 (22.86)	7075–T7351 plate as given in QQ-A-250/12	
[24]	Chord - Upper		7050-T7451 plate as given in AMS 4050	
[25]	Chord - Lower		7050-T7451 plate as given in BMS 7-323, Type III	
[26]	Web	0.063 (1.60)	2024-T3 sheet as given in QQ-A-250/4	
[27]	Skin - Upper	0.016 (0.41)	7075-T6 bare sheet as given in QQ-A-250/12	
[28]	Filler		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581	
[29]	Rib - Closure		2024-T351 plate as given in QQ-A-250/4	
[30]	Skin - Lower	0.020 (0.51)	7075-T6 bare sheet as given in QQ-A-250/12	
[31]	Honeycomb Core		Aluminum honeycomb core as given in BMS 4-4, Type 4-25, Class NPA, Form B, Grade I	

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Table 2: (Continued)

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[32]	Spar - T.E. Wedge		2024-T351 plate as given in QQ-A-250/4	
[33]	Rib - Closure		2024-T351 plate as given in QQ-A-250/4	
[34]	Tube - Torque		Ti-6Al-4V plate as given in Mil-T-9047	
[35]	Rib - Wedge		7050-T7451 plate as given in AMS 4050	
[36]	Fitting		7050-T7451 plate as given in AMS 4050	
[37]	Stiffener		7050-T7451 plate as given in AMS 4050	
[38]	Stiffener (3)		7050-T7451 plate as given in AMS 4050	Line 1-18
[38]	Stiffener		BAC1503-100144 7075-T73511 extrusion as given in QQ-A-200/11	Line 19 and on
[39]	Rib - Cove Support	0.063 (1.60)	2024-T351 plate as given in QQ-A-250/4	
[40]	Bolt Pad	0.25 (6.35)	7050-T7351 plate as given in QQ-A-250/12	
[41]	Track - Flap Programming		Ti-6Al-4V plate, in the annealed condition (Optional: 15–5PH stainless steel bar)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 2 - WING TRAILING EDGE INBOARD AFT FLAP STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

H97469 S0006594914_V1

Wing Trailing Edge Movable Flap Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable Trailing Edge Functional Product Collector
113A3001	Flap Installation - Outboard Trailing Edge Flap
113A3002	Flap Assembly - Outboard Trailing Edge Flap
113A2100	Main Flap Final Assembly - Inboard Flap
113A2700	Aft Flap Assembly - Inboard Flap
113A2710	Bonded Assembly - Trailing Edge Wedge, Inboard Aft Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A2722	Fitting Assembly - Inboard Pushrod, Inboard Aft Flap
113A2725	Fitting Assembly - Inboard Support Track, Inboard Aft Flap
113A2726	Fitting Assembly - Outboard Support Track, Inboard Aft Flap

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

H97470 S0006594916_V1

Wing Trailing Edge Inboard Aft Flap Structure
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Skin - Upper	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
[2]	Trailing Edge Filler		Epoxy impregnated glass woven fiber as given in BMS 8-79, Class III, Grade B, Style 1581 120, or 7781	
[3]	Rib - Closure		7050-T7451 plate as given in AMS 4050	
[4]	Rib	0.90 (22.86)	7050-T7451 plate as given in AMS 4050	
[5]	Skin - Lower	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	
			Glass fabric overlay as given in BMS 8-79, class III, style 7781 or BMS 9-3, type H-2 as given in BAC 5337	For cum line numbers 1131 and on
			(Optional) Glass fabric overlay as given in BMS 8-79, class III, style 7781 or BMS 9-3, type H-2 as given in BAC 5337	For cum line numbers 1 through 1130
[6]	Honeycomb Core		Aluminum honeycomb core as given in BMS 4-4, Type 4-25, Class NPA, Form B, Grade I	
[7]	Spar		7075-T6 sheet as given in QQ-A-250/12	
[8]	Fitting		7075-T6 sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 3 - WING TRAILING EDGE OUTBOARD MAIN FLAP STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

H41883 S0006594919_V1

Wing Trailing Edge Movable Flap Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable Trailing Edge Functional Product Collector
113A3001	Flap Installation Outboard Trailing Edge Flap
113A3002	Flap Assembly - Outboard Trailing Edge Flap
113A3100	Main Flap Assembly - Outboard Trailing Edge Flap
113A3135	Failsafe Fitting Assembly, Outboard Trailing Edge Flap
113A3310	Rib Assembly - Aft Track #1 Support Outboard Trailing Edge Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A3320	Rib Assembly - Aft Track #2 Support Outboard Trailing Edge Flap
113A3330	Rib Assembly - Aft Track #3 Support Outboard Trailing Edge Flap
113A3340	Rib Assembly - Aft Track #4 Support Outboard Trailing Edge Flap

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

H41884 S0006594921_V2

Wing Trailing Edge Outboard Main Flap Structure Identification
Figure 2 (Sheet 1 of 3)

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FRONT SPAR IDENTIFICATION

A

H41886 S0006594922_V2

Wing Trailing Edge Outboard Main Flap Structure Identification
Figure 2 (Sheet 2 of 3)

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REAR SPAR

B

B-B

H41885 S0006594923_V2

Wing Trailing Edge Outboard Main Flap Structure Identification
Figure 2 (Sheet 3 of 3)

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	DESCRIPTION	T*[1]	MATERIAL	EFFECTIVITY
[1]	Rib - Closure (2)		7050-T7451 plate as given in BMS 7-323	
[2]	Rib - Deflection Control		15-5PH bar as given in AMS 5659, heat treat to 180-200 KSI	
[3]	Rib - Aft Track Support (8)		7050-T7451 plate as given in AMS 4050	
[4]	Rib - Airload (13)		7050-T7451 plate as given in AMS 4050, 7050-T7451 plate as given in BMS 7-323	
[5]	Rib - Nose (5)		7050-T7451 plate as given in AMS 4050	
[6]	Rib - Deflection Control		Ti-6Al-4V bar as given in AMS 4928	
[7]	Programming Track (2)		Ti-6Al-4V bar as given in AMS 4928	
[8]	Rib - Programming (2)		7050-T7451 plate as given in AMS 4050	
[9]	Fitting - Aft Track Support (4)		7050-T7451 plate as given in AMS 4050 7050-T7451 plate as given in BMS 7-323	
[10]	Fitting - Cove Support (12)		7050-T7451 plate as given in AMS 4050	
[11]	Fitting - Carriage Failsafe (2)		Ti-6Al-4V bar as given in AMS 4928	
[12]	Fitting - Carriage Support (4)		15-5PH bar as given in AMS 5659, heat treat to 180-200 KSI	
[13]	Rib - Carriage Support (2)		7050-T7451 plate as given in BMS 7-323	
[14]	Stiffener - Lower (2)		BAC1503-100413 2024–T3511 extrusion as given in QQ-A-200/3	
[15]	Stiffener - Upper		BAC1503-100892 7075-T73511 extrusion as given in QQ-A-200/11	
[16]	Stiffener - Upper		BAC1503-100413 7075-T73511 extrusion as given in QQ-A-200/11	
[17]	Chord - Upper		BAC1505-101692 7075-T73511 extrusion as given in QQ-A-200/11	
[18]	Chord - Lower		BAC1505-101691 2024-T3511 extrusion as given in QQ-A-200/3	
[19]	Web	0.050 (1.27)	2024-T3 sheet as given in QQ-A-250/4	
[20]	Doubler	0.080 (2.03)	2024-T3 sheet as given in QQ-A-250/4	
[21]	Doubler	0.025 (0.64)	2024-T3 sheet as given in QQ-A-250/4	
[22]	Chord - Upper Tee		BAC1505-101690 7075-T73511 extrusion as given in QQ-A-200/11	
[23]	Chord - Lower		BAC1505-101689 2024-T3511 extrusion as given in QQ-A-200/3	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 4 - WING TRAILING EDGE OUTBOARD AFT FLAP STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

H41887 S0006594926_V1

Wing Trailing Edge Movable Flap Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable T.E. Functional Product Collector
113A3001	Flap Installation - Outboard Trailing Edge Flap
113A3002	Flap Assembly - Outboard Trailing Edge Flap
113A3700	Aft Flap Assembly - Outboard Trailing Edge Flap
113A3725	Drive Fitting Assembly - Inboard, Outboard Aft Flap
113A3726	Drive Fitting Assembly - Outboard, Outboard Aft Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A3732	Fitting Assembly - Track #1 Support, Outboard Aft Flap
113A3742	Fitting Assembly - Track #2 Support, Outboard Aft Track
113A3752	Fitting Assembly - Track #3 Support, Outboard Aft Flap
113A3762	Fitting Assembly - Track #4 Support, Outboard Aft Flap

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

H41889 S0006594928_V1

Wing Trailing Edge Outboard Aft Flap Structure
Figure 2 (Sheet 1 of 2)

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TRAILING EDGE WEDGE BONDED ASSEMBLY IDENTIFICATION

A

H41890 S0006594929_V1

Wing Trailing Edge Outboard Aft Flap Structure
Figure 2 (Sheet 2 of 2)

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Rib		7050-T7451 plate as given in AMS 4050	
[2]	Fitting		7050-T7451 plate as given in AMS 4050	
[3]	Rib		7050-T7451 plate as given in AMS 4050	
[4]	Fitting		7050-T7451 plate as given in AMS 4050	
[5]	Fitting		7050-T7451 plate as given in AMS 4050	
[6]	Fitting		7050-T7451 plate as given in AMS 4050	
[7]	Rib - Closure		7050-T7451 plate as given in AMS 4050	
[8]	Skin - Upper	0.016 (0.41)	7075-T6 sheet as given in QQ-A-250/12	
[9]	Fillet		Glass Fiber Reinforced Plastic (GFRP) Pre-preg as given in BMS 8-79, Style 1581	
[10]	Rib - Closure		7050-T7451 plate as given in AMS 4050	
[11]	Honeycomb Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 4-25 NPA	
[12]	Skin - Lower	0.020 (0.51)	7075-T6 sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING INBOARD TRAILING EDGE MAIN FLAP STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the Main Flap Structure of the Wing Inboard Trailing Edge shown in Wing Inboard Trailing Edge Main Flap Structure Location, Figure 101/ ALLOWABLE DAMAGE 1.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G00140 S0006594933_V1

Wing Inboard Trailing Edge Main Flap Structure Location
Figure 101

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G04443 S0006594934_V4

Wing Inboard Trailing Edge Main Flap Structure
Figure 102

BOEING PROPRIETARY - Copyright © Unpublished Work - See title page for details

2. General

WARNING: SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON, OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STEAM EXPLOSION. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

A. Do the steps that follow if you have damage to the aluminum, stainless steel or titanium parts:

(1) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the rear spar chords, support ribs, aft flap fittings, aft flap tracks, and closure ribs. Flap peen or shot peen the reworked areas of the aft flap programming track, if it is stainless steel.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum parts. Refer to 51-20-01.

(3) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked area of aluminum parts. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection

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(Continued)

Reference	Title
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. **Allowable Damage Limits**

A. Front Spar Web and Rear Spar Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , E , G , and H .

(3) Dents:

(a) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail I .

NOTE: Fill the space between the flange and skin with BMS 5-95 sealant if the space is more than 0.005 inch.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch maximum in diameter

2) A minimum of 2.0 inches away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

a) Install the rivet without sealant.

B. Front and Rear Spar Chords, Ribs, Rib Wedge, Stiffeners, and Cove Support Tees.

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , C , D , F , G , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

C. Crank Shaft Support Fitting and Aft Flap Fitting

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , F , G , H , and J .

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(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Aft Flap Track

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Details K, and L.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

(5) Wear:

(a) Horizontal flange thickness wear:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail L

a) This is an interim repair if the tungsten carbide coating is worn away and the base metal is exposed. Inspect the repair as given in Paragraph 5./ALLOWABLE DAMAGE 1

b) This is a permanent repair if the tungsten carbide coating is not worn away.

(b) Horizontal flange edge wear:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail K

E. Aft Flap Programming Track

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail M.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail M.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

(5) Wear:

(a) Horizontal flange width wear (either flange or both flanges):

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 1, Detail M.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

G00374 S0006594935_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 7)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

G00375 S0006594936_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 7)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

F

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

G

G00381 S0006594937_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 7)

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G00395 S0006594938_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 7)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

J

G00397 S0006594939_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 7)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.050 INCH (1.270mm).

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

K

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

L

A–A

1543831 S0000282147_V1

Allowable Damage Limits
Figure 103 (Sheet 6 of 7)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.025 INCH, IF T ≥ 0.222 INCH
= A MAXIMUM OF 0.045 INCH, IF T ≥ 0.240 INCH
WHERE T = THICKNESS OF THE MATING ROLLER RUB BLOCK

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

M

1936207 S0000367203_V2

Allowable Damage Limits
Figure 103 (Sheet 7 of 7)

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5. Inspection Requirements

A. Do the inspection as given in Table 101/ALLOWABLE DAMAGE 1

Table 101: Inspection Requirements

INSPECTION THRESHOLD	REPEAT INSPECTIONS	
	METHOD	INTERVAL
When you remove the damage	As given in 51-10-02, GENERAL	6 Months or 1000 Flight Cycles whichever comes sooner
NOTES: Do a detailed inspection of the Aft Flap Track to ensure the horizontal flange thickness does not wear beyond a maximum of 0.020 in. (0.508 mm) into the base metal. The 0.020 in. (0.508 mm) limit applies to the base metal thickness reduction and does not include the tungsten carbide coating.		

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ALLOWABLE DAMAGE 2 - WING INBOARD TRAILING EDGE AFT FLAP STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing inboard trailing edge aft flap structure shown in Wing Inboard Trailing Edge Aft Flap Location, Figure 101/ALLOWABLE DAMAGE 2.

2. General

A. Do the steps that follow if you have damage to the parts:

(1) Remove the damage. Refer to 51-10-02 for the investigation and cleanup procedures.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

B. After you remove the damage, do as follows:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Do not flap peen or shot peen the spar, trailing edge wedge, or the closure ribs.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply one layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-44-04.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G00405 S0006594941_V1

Wing Inboard Trailing Edge Aft Flap Location
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

INBOARD TRAILING EDGE AFT FLAP

G04159 S0006594942_V2

Wing Inboard Trailing Edge - Aft Flap Structure
Figure 102

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
57-53-01	TRAILING EDGE FLAP SKIN
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Closure Ribs, Stiffener Ribs, Segment Chord, and Spar Web

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , and F .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , E , F , G , and H .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail I .

(a) If there is a gap between the flap skin and the structure, that is more than 0.005 inch, then fill the gap with a shim.

1) Refer to 57-53-01 for the dent limits for the flap skin.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet.

a) Install the rivet without sealant.

B. Inboard Rib, Outboard Rib, Spar Chords, and Fittings

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , and C .

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(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

G01787 S0006594943_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 5)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

G01789 S0006594944_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 5)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

F

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

G

G01791 S0006594945_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 5)

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REMOVAL OF CORROSION AROUND THE FASTENERS

H

B–B

G01795 S0006594946_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 5)

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DENT THAT IS PERMITTED

I

G01796 S0006594947_V2

Allowable Damage Limits
Figure 103 (Sheet 5 of 5)

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

G11208 S0006594948_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

G11223 S0006594949_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

E

G11226 S0006594950_V1

Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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ALLOWABLE DAMAGE 3 - WING OUTBOARD TRAILING EDGE MAIN FLAP STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the Main Flap Structure of the Wing Outboard Trailing Edge shown in Figure 101/ALLOWABLE DAMAGE 3.

G04471 S0006594952_V1

Wing Outboard Trailing Edge Main Flap Structure Location
Figure 101

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LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

G04602 S0006594953_V3

Wing Outboard Trailing Edge Main Flap Structure Allowable Damage
Figure 102 (Sheet 1 of 5)

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G18360 S0006594954_V1

Wing Outboard Trailing Edge Main Flap Structure Allowable Damage
Figure 102 (Sheet 2 of 5)

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Wing Outboard Trailing Edge Main Flap Structure Allowable Damage
Figure 102 (Sheet 3 of 5)

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OUTBOARD DEFLECTION CONTROL RIB

C

C–C

1814437 S0000320730_V2

Wing Outboard Trailing Edge Main Flap Structure Allowable Damage
Figure 102 (Sheet 4 of 5)

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INBOARD PROGRAMMING RIB AND TRACK

D

OUTBOARD PROGRAMMING RIB AND TRACK

E

2457415 S0000569683_V1

Wing Outboard Trailing Edge Main Flap Structure Allowable Damage
Figure 102 (Sheet 5 of 5)

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 3 for the allowable damage limits.

WARNING: SMALL PARTICLES AND THIN CUTS OF TITANIUM ARE FLAMMABLE. IN A SUFFICIENT CONCENTRATION, AN EXPLOSION CAN OCCUR. EXTINGUISH FIRES OF TITANIUM WITH FULLY DRY TALC, CALCIUM CARBONATE, SAND OR GRAPHITE. APPLY THE POWDER TO A DEPTH OF 1/2 INCH OR MORE TO THE AREA THAT IS ON FIRE. DO NOT USE FOAM, WATER, CARBON TETRACHLORIDE, HALON OR CARBON DIOXIDE. WATER IN CONTACT WITH MOLTEN TITANIUM CAN CAUSE A STEAM EXPLOSION. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

B. Do the steps that follow if you have damage to the aluminum, titanium, or Corrosion Resistant Steel (CRES) parts:

(1) Remove the damage.

(a) Refer to 51-10-02 for the investigation and cleanup procedures for the aluminum and titanium parts.

(b) Refer to SOPM 20-20-01 for the investigation and refer to SOPM 20-10-01 for cleanup procedures for the CRES parts.

(c) Refer to 51-30-03 for possible sources of non-metallic materials you need to remove the damage.

(d) Refer to 51-30-05 for possible sources of the equipment and tools you need to remove the damage.

(2) After you remove the damage, do as follows:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(a) Flap peen or shot peen all parts but not the parts that follow:

1) Airload Ribs

2) Inboard and Outboard Programming Ribs

3) Cove Panel Support Ribs

4) Stiffeners

(b) Refer to 51-20-06 for shot peen intensity and shot number.

(c) Refer to SOPM 20-10-03 for flap peen or shot peen procedures and coverage.

(d) Apply the solvent resistant chemicals.

1) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

2) Apply cadmium plating to the bare surfaces of the CRES parts. Refer to 51-20-01.

(e) Apply a layer of BMS 10-11, Type I primer to the surfaces of the reworked CRES parts. Refer to SOPM 20-41-02.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	**PARAGRAPH**
STIFFENER AND SPAR CHORDS	4.A
FRONT SPAR WEB	4.B

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Table 101: (Continued)

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS	
TYPE OF STRUCTURE	PARAGRAPH
RIBS, RIB SUPPORTS AND FITTINGS	4.C
DEFLECTION CONTROL RIBS	4.D

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-10-01	Repair and Refinish of High Strength Steel Parts
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-01	Magnetic Particle Inspection
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Front and Rear Spar Chords, and Stiffener

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , and F.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , F , G , and H .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

B. Front Spar Web

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details A , B , and E .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details A , B , and E.

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(3) Dents are not permitted.

(4) Holes and Punctures:

(a) The damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 2.0 inches away from a fastener hole, the edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet installed without sealant.

C. Ribs, Rib Supports, and Fittings

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , and F.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details A , B , C , D , F , G , and H.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

D. Deflection Control Rib (CRES)

(1) Cracks:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details I and J.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Figure 103/ALLOWABLE DAMAGE 3, Details I and J.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

C

G04619 S0006594957_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 7)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

G04637 S0006594958_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 7)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

F

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

G

G19494 S0006594959_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 7)

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REMOVAL OF CORROSION AROUND THE FASTENERS

H

B–B

G19492 S0006594960_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 7)

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REMOVAL OF DAMAGED MATERIAL
ON A SURFACE FOR THE DEFLECTION CONTROL RIB

I

C–C

1814417 S0000320731_V1

Allowable Damage Limits
Figure 103 (Sheet 5 of 7)

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X = DEPTH OF THE MATERIAL
THAT IS REMOVED
= A MAXIMUM OF 0.050 INCH

D–D

1989694 S0000386792_V1

Allowable Damage Limits
Figure 103 (Sheet 6 of 7)

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REMOVAL OF DAMAGED MATERIAL
AT AN EDGE FOR THE
DEFLECTION CONTROL RIB

J

E-E

1982595 S0000382969_V1

Allowable Damage Limits
Figure 103 (Sheet 7 of 7)

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ALLOWABLE DAMAGE 4 - WING OUTBOARD TRAILING EDGE AFT FLAP STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the Aft Flap Structure of the Wing Outboard Trailing Edge shown in Wing Outboard Trailing Edge Aft Flap Location, Figure 101/ALLOWABLE DAMAGE 4.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G05591 S0006594962_V1

Wing Outboard Trailing Edge Aft Flap Location
Figure 101

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NOTE: ALL PARTS IDENTIFIED ARE MADE OF ALUMINUM.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

OUTBOARD TRAILING EDGE AFT FLAP STRUCTURE

G05612 S0006594963_V2

Wing Outboard Trailing Edge - Aft Flap Structure Location
Figure 102

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 4 for the allowable damage limits.

B. Do the steps that follow if you have damage.

(1) Remove the damage. Refer to 51-10-02 for the investigation and cleanup procedures.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(4) Make the edges of the blendouts and holes smooth to a finish of 125 microinches Ra or better.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(5) Flap peen or shot peen the reworked areas of the track support fittings, but not the inner surfaces of the lug bores.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(6) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(7) Apply a layer of BMS 10-11, Type I primer to the reworked surfaces. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Track Ribs, Drive Ribs, and Spar

(1) Rib and Spar Webs

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 4, Detail E .

(b) Nicks, Gouges, Scratches, and Corrosion:

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1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 4, Details E , G , and H .

(c) Dents:

1) The damage is permitted as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 4, Detail I .

NOTE: Fill the space between the flange and skin with BMS 5-95 sealant if the space is more than 0.005 inch.

(d) Holes and Punctures:

1) The damage is permitted if it is:

a) A maximum of 0.25 inch in diameter

b) A minimum of 1.0 inch away from a fastener hole, the edge of the part, or other damage

c) Filled with a 2117-T3 or 2117-T4 protruding head rivet installed without sealant.

(2) Rib and Spar Chords

(a) Cracks:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 4, Details A , B , C , D , and F.

(b) Nicks, Gouges, Scratches, and Corrosion:

1) Remove the damage as shown in Allowable Damage Limits, Figure 103/ ALLOWABLE DAMAGE 4, Details A , B , C , D , F , G , and H .

(c) Dents are not permitted.

(d) Holes and Punctures are not permitted.

B. Inboard Closure Rib, Outboard Closure Rib, and Fittings

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 4, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 4, Details A , B , C , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE OF A FORMED PART

C

G05619 S0006594964_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 4)

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X = THE WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF A METAL SKIN OR WEB

E

G05620 S0006594965_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 4)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE OF AN EXTRUDED PART

F

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

G

G05622 S0006594966_V1

Allowable Damage Limits
Figure 103 (Sheet 3 of 4)

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G05623 S0006594967_V1

Allowable Damage Limits
Figure 103 (Sheet 4 of 4)

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

G11434 S0006594968_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

G11436 S0006594969_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

E

G11438 S0006594970_V1

Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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REPAIR 1 - WING INBOARD TRAILING EDGE MAIN FLAP STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G14610 S0006594973_V2

Wing Inboard Trailing Edge Main Flap Location
Figure 201

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REPAIR 2 - WING INBOARD TRAILING EDGE AFT FLAP STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G15031 S0006594975_V2

Wing Inboard Trailing Edge Aft Flap Location
Figure 201

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REPAIR 3 - WING OUTBOARD TRAILING EDGE MAIN FLAP STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G16789 S0006594977_V2

Wing Outboard Trailing Edge Main Flap Location
Figure 201

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REPAIR 4 - WING OUTBOARD TRAILING EDGE AFT FLAP STRUCTURE

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G16773 S0006594979_V2

Wing Outboard Trailing Edge Aft Flap Location
Figure 201

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REPAIR 5 - INBOARD FLAP AFT FLAP TRACK OVERSIZED BEARING BORE REPAIR

1. Applicability

A. This repair is applicable for oversized bearing bores in the titanium inboard flap aft flap track with part numbers 113A2672–3 and 113A2672–7.

B. Do not use this repair if the wear in the lug bore is more than 1.000 in. (25.40 mm) in diameter.

2. General

A. For the location of the repair area, see Figure 201/REPAIR 5.

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1856210 S0000331818_V1

Location of the Inboard Flap Aft Flap Track Bushing Bore
Figure 201 (Sheet 1 of 2)

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AFT FLAP TRACK BEARING BORE LOCATION

A

VIEW IN THE OUTBOARD DIRECTION

B

1857827 S0000331819_V1

Location of the Inboard Flap Aft Flap Track Bushing Bore
Figure 201 (Sheet 2 of 2)

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3. References

Reference	Title
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
BAC 5619	Heat Treat of Corrosion Resistant Steel
BAC 5625	SURFACE TREATMENTS FOR FERROUS ALLOYS
CMM 57-53-05	Inboard Flap - Main Flap Final Assembly
SOPM 20-50-03	Bearing and Bushing Replacement

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the initial bearing. Refer to SOPM 20-50-03.

C. Ream the lug bore to a maximum of 1.000 in. (25.40 mm) in diameter.

D. Chamfer the inboard and outboard edges of the lug bore. Refer to Figure 203/REPAIR 5

E. Make the Part [1] Repair Sleeve. See Table 201/REPAIR 5

(1) The outer diameter of the Part [1] Repair Sleeve must be between 0.0007 in. (0.018 mm) and 0.0020 in. (0.051 mm) larger than the reamed bore diameter.

(2) The wall thickness of the Part [1] Repair Sleeve must be a minimum of 0.040 in. (1.02 mm) thick after you do the inner diameter ream.

(3) Heat treat the Part [1] Repair Sleeve. Refer to BAC 5619.

(a) If you make the Part [1] Repair Sleeve out of 15–5PH, heat treat the sleeve to between 140 KSI and 160 KSI.

(b) If you make the Part [1] Repair Sleeve out of 17–4PH, heat treat the sleeve to between 150 KSI and 170 KSI.

(4) If you install this repair on a 113A2672–3 flap track, you must do the steps below. If you install this repair on a 113A2672–7 flap track, do not do these steps.

(a) Make a 0.090 in. (2.29 mm) wide and 0.015 in. (0.38 mm) deep grease groove in the outer diameter of the Part [1] Repair Sleeve. See Figure 203/REPAIR 5.

NOTE: The grease groove in the Part [1] Repair Sleeve must align with the grease groove in the Part [2] Bearing.

(b) Make four 0.090 in. (2.29 mm) diameter holes through the grease groove to make a grease path to the bearing. Make sure that there is an equal distance between all the holes. See Figure 203/REPAIR 5.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Repair Sleeve	1	Use 15–5PH steel per AMS 5659, or use 17–4PH steel per AMS 5643
[2]	Bearing	1	Same size and type as the initial bearing.

F. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Repair Sleeve and the lug bore repair area.

G. Make the surfaces of the Part [1] Repair Sleeve and the lug bore repair area smooth to a finish of 63 microinches RA or smoother.

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H. Passivate the Part [1] Repair Sleeve. Refer to Method II of BAC 5625.

WARNING: WHEN YOU DO WORK WITH DRY ICE, MAKE SURE YOU USE SAFETY GOGGLES AND INSULATED GLOVES. IF DRY ICE TOUCHES YOUR EYES OR SKIN, FROSTBITE CAN OCCUR.

WARNING: LIQUID NITROGEN IS AN AGENT THAT IS AN ASPHYXIANT. MAKE SURE ALL PERSONS OBEY ALL OF THE PRECAUTIONS WHEN LIQUID NITROGEN IS USED.

- USE IN AN AREA OPEN TO THE AIR.
- CLOSE THE CONTAINER WHEN NOT USED.
- DO NOT GET LIQUID NITROGEN IN THE EYES, ON THE SKIN, OR ON YOUR CLOTHES.
- DO NOT BREATHE THE GAS.

I. Install the Part [1] Repair Sleeve. Use a shrink fit method. Install the Part [1] Repair Sleeve with BMS 3-33. Refer to SOPM 20-50-03, Section 7, paragraph B.

J. Swage the Part [1] Repair Sleeve on both the inboard and outboard ends. See Figure 203/REPAIR 5.

K. Ream the inner diameter of the Part [1] Repair Sleeve to the necessary diameter for the Part [2] Bearing. Refer to CMM 57-53-05.

L. Chamfer the inboard and outboard edges of the Part [1] Repair Sleeve to the necessary dimensions for the Part [2] Bearing. See Figure 203/REPAIR 5.

M. Install the Part [2] Bearing. Refer to CMM 57-53-05.

N. Do a push-out load test on the bearing. Refer to SOPM 20-50-03.

O. Refinish the repair area around the lug bore if necessary. Refer to AMM SECTION 51-21.

P. Replace the items you removed for access.

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NOTES

- ALL DIMENSIONS ARE INCHES (mm)

1 THE NOMINAL DIAMETER OF THE BEARING RING IS 0.7818 INCH (19.858 mm).

2 ALIGN THE GREASE GROOVE OF THE [1] REPAIR SLEEVE WITH THE GREASE GROOVE OF THE [2] BEARING.

1856688 S0000331820_V1

Notes and Fastener Symbols
Figure 202

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ROTATED 90° CLOCKWISE
SECTION A-A

1856623 S0000331823_V2

Repair of the Inboard Flap Aft Flap Track Oversized Bearing Bore
Figure 203 (Sheet 1 of 2)

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[1] REPAIR SLEEVE
NOT SWAGED

A

B–B

GREASE GROOVE DETAIL

B

1856536 S0000331824_V1

Repair of the Inboard Flap Aft Flap Track Oversized Bearing Bore
Figure 203 (Sheet 2 of 2)

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REPAIR 6 - #1 OUTBOARD FLAP AFT FLAP TRACK OVERSIZED BEARING BORE REPAIR

1. Applicability

A. This repair is applicable for oversized bearing bores in the steel #1 aft flap track on the outboard flap with part number 113A3940–1.

B. Do not use this repair if the wear in the lug bore is more than 0.85 in. (21.6 mm) in diameter.

2. General

A. For the location of the repair area, see Figure 201/REPAIR 6.

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1885012 S0000344809_V1

Location of the Outboard Flap #1 Aft Flap Track Bushing Bore
Figure 201 (Sheet 1 of 2)

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NO. 1 OUTBOARD AFT FLAP TRACK BEARING BORE LOCATION

A

VIEW IN THE OUTBOARD DIRECTION

B

1888519 S0000344826_V1

Location of the Outboard Flap #1 Aft Flap Track Bushing Bore
Figure 201 (Sheet 2 of 2)

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3. References

Reference	Title
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
BAC 5619	Heat Treat of Corrosion Resistant Steel
BAC 5625	SURFACE TREATMENTS FOR FERROUS ALLOYS
CMM 57-53-06	Flap Assembly - Outboard Trailing Edge
SOPM 20-50-03	Bearing and Bushing Replacement

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the initial bearing. Refer to SOPM 20-50-03.

C. Ream the lug bore to a maximum of 0.85 in. (21.6 mm) in diameter.

D. Chamfer the inboard and outboard edges of the lug bore. Refer to Figure 203/REPAIR 6.

E. Make the Part [1] Repair Sleeve. See Table 201/REPAIR 6.

(1) The outer diameter of the Part [1] Repair Sleeve must be between 0.0007 in. (0.018 mm) and 0.0020 in. (0.051 mm) larger than the reamed bore diameter.

(2) The wall thickness of the Part [1] Repair Sleeve must be a minimum of 0.040 in. (1.02 mm) thick after you do the inner diameter ream.

(3) Heat treat the Part [1] Repair Sleeve. Refer to BAC 5619.

(a) If you make the Part [1] Repair Sleeve out of 15–5PH, heat treat the sleeve to between 140 KSI and 160 KSI.

(b) If you make the Part [1] Repair Sleeve out of 17–4PH, heat treat the sleeve to between 150 KSI and 170 KSI.

(4) Make a grease groove in the Part [1] Repair Sleeve:

(a) Make a 0.090 in. (2.29 mm) wide and 0.015 in. (0.38 mm) deep grease groove in the outer diameter of the Part [1] Repair Sleeve. See Figure 203/REPAIR 6.

NOTE: The grease groove in the Part [1] Repair Sleeve must align with the grease groove in the Part [2] Bearing.

(b) Make four 0.090 in. (2.29 mm) diameter holes through the grease groove to make a grease path to the bearing. Make sure that there is an equal distance between all the holes. See Figure 203/REPAIR 6.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Repair Sleeve	1	Use 15–5PH steel per AMS 5659, or use 17–4PH steel per AMS 5643
[2]	Bearing	1	Same size and type as the initial bearing.

F. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Repair Sleeve and the lug bore repair area.

G. Make the surfaces of the Part [1] Repair Sleeve and the lug bore repair area smooth to a finish of 63 microinches RA or smoother.

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H. Passivate the Part [1] Repair Sleeve and the reworked surfaces of the flap track. Refer to Method II of BAC 5625.

WARNING: WHEN YOU DO WORK WITH DRY ICE, MAKE SURE YOU USE SAFETY GOGGLES AND INSULATED GLOVES. IF DRY ICE TOUCHES YOUR EYES OR SKIN, FROSTBITE CAN OCCUR.

WARNING: LIQUID NITROGEN IS AN AGENT THAT IS AN ASPHYXIANT. MAKE SURE ALL PERSONS OBEY ALL OF THE PRECAUTIONS WHEN LIQUID NITROGEN IS USED.

- USE IN AN AREA OPEN TO THE AIR.
- CLOSE THE CONTAINER WHEN NOT USED.
- DO NOT GET LIQUID NITROGEN IN THE EYES, ON THE SKIN, OR ON YOUR CLOTHES.
- DO NOT BREATHE THE GAS.

Install the Part [1] Repair Sleeve with a layer of grease. Use a shrink fit method. Install the Part [1] Repair Sleeve with BMS 3-33. Refer to SOPM 20-50-03, Section 7, paragraph B.

J. Swage the Part [1] Repair Sleeve on both the inboard and outboard ends. See Figure 203/REPAIR 6.

K. Ream the inner diameter of the Part [1] Repair Sleeve to the necessary diameter for the Part [2] Bearing. Refer to CMM 57-53-06.

L. Chamfer the inboard and outboard edges of the Part [1] Repair Sleeve to the necessary dimensions for the Part [2] Bearing. See Figure 203/REPAIR 6.

M. Install the Part [2] Bearing. Refer to CMM 57-53-06.

N. Do a push-out load test on the bearing. Refer to SOPM 20-50-03.

O. Refinish the repair area around the lug bore if necessary. Refer to AMM SECTION 51-21.

P. Replace the items you removed for access.

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NOTES

- ALL DIMENSIONS ARE INCHES (mm)

1 THE NOMINAL DIAMETER OF THE BEARING RING IS 0.6875 INCH (17.463 mm).

2 ALIGN THE GREASE GROOVE OF THE [1] REPAIR SLEEVE WITH THE GREASE GROOVE OF THE [2] BEARING.

1885707 S0000344811_V1

Notes and Fastener Symbols
Figure 202

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ROTATED 90° CLOCKWISE
SECTION A-A

1885712 S0000344812_V1

Repair of the Outboard Flap #1 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 1 of 2)

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[1] REPAIR SLEEVE
NOT SWAGED

A

B–B

GREASE GROOVE DETAIL

B

1885716 S0000344813_V1

Repair of the Outboard Flap #1 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 2 of 2)

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REPAIR 7 - #2 OUTBOARD FLAP AFT FLAP TRACK OVERSIZED BEARING BORE REPAIR

1. Applicability

A. This repair is applicable for oversized bearing bores in the steel #2 aft flap track on the outboard flap with part number 113A3941–1.

B. Do not use this repair if the wear in the lug bore is more than 0.875 in. (22.23 mm) in diameter.

2. General

A. For the location of the repair area, see Figure 201/REPAIR 7.

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FWD OUTBD

1885018 S0000344815_V1

Location of the Outboard Flap #2 Aft Flap Track Bushing Bore
Figure 201 (Sheet 1 of 2)

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NO. 2 OUTBOARD AFT FLAP TRACK BEARING BORE LOCATION

A

VIEW IN THE OUTBOARD DIRECTION

B

1887509 S0000344816_V1

Location of the Outboard Flap #2 Aft Flap Track Bushing Bore
Figure 201 (Sheet 2 of 2)

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3. References

Reference	Title
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
BAC 5619	Heat Treat of Corrosion Resistant Steel
BAC 5625	SURFACE TREATMENTS FOR FERROUS ALLOYS
CMM 57-53-06	Flap Assembly - Outboard Trailing Edge
SOPM 20-42-02	Low Hydrogen Embitterment Cadmium Plating - Titanium Alloy Plating
SOPM 20-50-03	Bearing and Bushing Replacement

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the initial bearing. Refer to SOPM 20-50-03.

C. Ream the lug bore to a maximum of 0.875 in. (22.23 mm) in diameter.

D. Chamfer the inboard and outboard edges of the lug bore. Refer to Figure 203/REPAIR 7.

E. Cadmium-titanium alloy plate the inner surface of the bore with chromate treatment. Refer to SOPM 20-42-02.

F. Make the Part [1] Repair Sleeve. See Table 201/REPAIR 7.

(1) The outer diameter of the Part [1] Repair Sleeve must be between 0.0007 in. (0.018 mm) and 0.0020 in. (0.051 mm) larger than the reamed bore diameter.

(2) The wall thickness of the Part [1] Repair Sleeve must be a minimum of 0.040 in. (1.02 mm) thick after you do the inner diameter ream.

(3) Heat treat the Part [1] Repair Sleeve. Refer to BAC 5619.

(a) If you make the Part [1] Repair Sleeve out of 15–5PH, heat treat the sleeve to between 140 KSI and 160 KSI.

(b) If you make the Part [1] Repair Sleeve out of 17–4PH, heat treat the sleeve to between 150 KSI and 170 KSI.

(4) Make a grease groove in the Part [1] Repair Sleeve:

(a) Make a 0.090 in. (2.29 mm) wide and 0.015 in. (0.38 mm) deep grease groove in the outer diameter of the Part [1] Repair Sleeve. See Figure 203/REPAIR 7.

NOTE: The grease groove in the Part [1] Repair Sleeve must align with the grease groove in the Part [2] Bearing.

(b) Make four 0.090 in. (2.29 mm) diameter holes through the grease groove to make a grease path to the bearing. Make sure that there is an equal distance between all the holes. See Figure 203/REPAIR 7.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Repair Sleeve	1	Use 15–5PH steel per AMS 5659, or use 17–4PH steel per AMS 5643
[2]	Bearing	1	Same size and type as the initial bearing.

G. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Repair Sleeve and the lug bore repair area.

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H. Make the surfaces of the Part [1] Repair Sleeve and the lug bore repair area smooth to a finish of 63 microinches RA or smoother.

I. Passivate the Part [1] Repair Sleeve and the reworked surfaces of the flap track. Refer to Method II of BAC 5625.

WARNING: WHEN YOU DO WORK WITH DRY ICE, MAKE SURE YOU USE SAFETY GOGGLES AND INSULATED GLOVES. IF DRY ICE TOUCHES YOUR EYES OR SKIN, FROSTBITE CAN OCCUR.

WARNING: LIQUID NITROGEN IS AN AGENT THAT IS AN ASPHYXIANT. MAKE SURE ALL PERSONS OBEY ALL OF THE PRECAUTIONS WHEN LIQUID NITROGEN IS USED.

- USE IN AN AREA OPEN TO THE AIR.
- CLOSE THE CONTAINER WHEN NOT USED.
- DO NOT GET LIQUID NITROGEN IN THE EYES, ON THE SKIN, OR ON YOUR CLOTHES.
- DO NOT BREATHE THE GAS.

Install the Part [1] Repair Sleeve with a layer of grease. Use a shrink fit method. Install the Part [1] Repair Sleeve with BMS 3-33. Refer to SOPM 20-50-03, Section 7, paragraph B.

K. Swage the Part [1] Repair Sleeve on both the inboard and outboard ends. See Figure 203/REPAIR 7.

L. Ream the inner diameter of the Part [1] Repair Sleeve to the necessary diameter for the Part [2] Bearing. Refer to CMM 57-53-06.

M. Chamfer the inboard and outboard edges of the Part [1] Repair Sleeve to the necessary dimensions for the Part [2] Bearing. See Figure 203/REPAIR 7.

N. Install the Part [2] Bearing. Refer to CMM 57-53-06.

O. Do a push-out load test on the bearing. Refer to SOPM 20-50-03.

P. Refinish the repair area around the lug bore if necessary. Refer to AMM SECTION 51-21.

Q. Replace the items you removed for access.

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NOTES

- ALL DIMENSIONS ARE INCHES (mm)

1 THE NOMINAL DIAMETER OF THE BEARING RING IS 0.7813 INCH (19.845 mm).

2 ALIGN THE GREASE GROOVE OF THE [1] REPAIR SLEEVE WITH THE GREASE GROOVE OF THE [2] BEARING.

1887510 S0000344817_V1

Notes and Fastener Symbols
Figure 202

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ROTATED 90° CLOCKWISE
SECTION A-A

1887512 S0000344818_V1

Repair of the Outboard Flap #2 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 1 of 2)

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[1] REPAIR SLEEVE
NOT SWAGED

A

B–B

GREASE GROOVE DETAIL

B

1887513 S0000344819_V1

Repair of the Outboard Flap #2 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 2 of 2)

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REPAIR 8 - #3 OUTBOARD FLAP AFT FLAP TRACK OVERSIZED BEARING BORE REPAIR

1. Applicability

A. This repair is applicable for oversized bearing bores in the steel outboard flap aft flap track with part number 113A3942–1.

B. Do not use this repair if the wear in the lug bore is more than 0.90 in. (22.9 mm) in diameter.

2. General

A. For the location of the repair area, see Figure 201/REPAIR 8.

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FWD OUTBD

1885066 S0000344825_V1

Location of the Outboard Flap #3 Aft Flap Track Bushing Bore
Figure 201 (Sheet 1 of 2)

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NO. 3 OUTBOARD AFT FLAP TRACK BEARING BORE LOCATION

A

VIEW IN THE OUTBOARD DIRECTION

B

1887691 S0000344821_V1

Location of the Outboard Flap #3 Aft Flap Track Bushing Bore
Figure 201 (Sheet 2 of 2)

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3. References

Reference	Title
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
BAC 5619	Heat Treat of Corrosion Resistant Steel
BAC 5625	SURFACE TREATMENTS FOR FERROUS ALLOYS
CMM 57-53-06	Flap Assembly - Outboard Trailing Edge
SOPM 20-50-03	Bearing and Bushing Replacement

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the initial bearing. Refer to SOPM 20-50-03.

C. Ream the lug bore to a maximum of 0.90 in. (22.9 mm) in diameter.

D. Chamfer the inboard and outboard edges of the lug bore. Refer to Figure 203/REPAIR 8.

E. Make the Part [1] Repair Sleeve. See Table 201/REPAIR 8

(1) The outer diameter of the Part [1] Repair Sleeve must be between 0.0007 in. (0.018 mm) and 0.0020 in. (0.051 mm) larger than the reamed bore diameter.

(2) The wall thickness of the Part [1] Repair Sleeve must be a minimum of 0.040 in. (1.02 mm) thick after you do the inner diameter ream.

(3) Heat treat the Part [1] Repair Sleeve. Refer to BAC 5619.

(a) If you make the Part [1] Repair Sleeve out of 15–5PH, heat treat the sleeve to between 140 KSI and 160 KSI.

(b) If you make the Part [1] Repair Sleeve out of 17–4PH, heat treat the sleeve to between 150 KSI and 170 KSI.

(4) Make a grease groove in the Part [1] Repair Sleeve:

(a) Make a 0.090 in. (2.29 mm) wide and 0.015 in. (0.38 mm) deep grease groove in the outer diameter of the Part [1] Repair Sleeve. See Figure 203/REPAIR 8.

NOTE: The grease groove in the Part [1] Repair Sleeve must align with the grease groove in the Part [2] Bearing.

(b) Make four 0.090 in. (2.29 mm) diameter holes through the grease groove to make a grease path to the bearing. Make sure that there is an equal distance between all the holes. See Figure 203/REPAIR 8.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Repair Sleeve	1	Use 15–5PH steel per AMS 5659, or use 17–4PH steel per AMS 5643
[2]	Bearing	1	Same size and type as the initial bearing.

F. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Repair Sleeve and the lug bore repair area.

G. Make the surfaces of the Part [1] Repair Sleeve and the lug bore repair area smooth to a finish of 63 microinches RA or smoother.

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H. Passivate the Part [1] Repair Sleeve and the reworked surfaces of the flap track. Refer to Method II of BAC 5625.

WARNING: WHEN YOU DO WORK WITH DRY ICE, MAKE SURE YOU USE SAFETY GOGGLES AND INSULATED GLOVES. IF DRY ICE TOUCHES YOUR EYES OR SKIN, FROSTBITE CAN OCCUR.

WARNING: LIQUID NITROGEN IS AN AGENT THAT IS AN ASPHYXIANT. MAKE SURE ALL PERSONS OBEY ALL OF THE PRECAUTIONS WHEN LIQUID NITROGEN IS USED.

- USE IN AN AREA OPEN TO THE AIR.
- CLOSE THE CONTAINER WHEN NOT USED.
- DO NOT GET LIQUID NITROGEN IN THE EYES, ON THE SKIN, OR ON YOUR CLOTHES.
- DO NOT BREATHE THE GAS.

Install the Part [1] Repair Sleeve with a layer of grease. Use a shrink fit method. Install the Part [1] Repair Sleeve with BMS 3-33. Refer to SOPM 20-50-03, Section 7, paragraph B.

J. Swage the Part [1] Repair Sleeve on both the inboard and outboard ends. See Figure 203/REPAIR 8.

K. Ream the inner diameter of the Part [1] Repair Sleeve to the necessary diameter for the Part [2] Bearing. Refer to CMM 57-53-06.

L. Chamfer the inboard and outboard edges of the Part [1] Repair Sleeve to the necessary dimensions for the Part [2] Bearing. See Figure 203/REPAIR 8.

M. Install the Part [2] Bearing. Refer to CMM 57-53-06.

N. Do a push-out load test on the bearing. Refer to SOPM 20-50-03.

O. Refinish the repair area around the lug bore if necessary. Refer to AMM SECTION 51-21.

P. Replace the items you removed for access.

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NOTES

- ALL DIMENSIONS ARE INCHES (mm)

1 THE NOMINAL DIAMETER OF THE BEARING RING IS 0.7813 INCH (19.845 mm).

2 ALIGN THE GREASE GROOVE OF THE [1] REPAIR SLEEVE WITH THE GREASE GROOVE OF THE [2] BEARING.

1887693 S0000344822_V1

Notes and Fastener Symbols
Figure 202

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ROTATED 90° CLOCKWISE
SECTION A-A

1887694 S0000344823_V1

Repair of the Outboard Flap #3 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 1 of 2)

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[1] REPAIR SLEEVE
NOT SWAGED

A

B–B

GREASE GROOVE DETAIL

B

1887709 S0000344824_V1

Repair of the Outboard Flap #3 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 2 of 2)

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REPAIR 9 - #4 OUTBOARD FLAP AFT FLAP TRACK OVERSIZED BEARING BORE REPAIR

1. Applicability

A. This repair is applicable for oversized bearing bores in the steel outboard flap aft flap track with part number 113A3943–1.

B. Do not use this repair if the wear in the lug bore is more than 0.95 in. (24.1 mm) in diameter.

2. General

A. For the location of the repair area, see Figure 201/REPAIR 9.

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1885020 S0000344820_V1

Location of the Outboard Flap #4 Aft Flap Track Bushing Bore
Figure 201 (Sheet 1 of 2)

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NO. 4 OUTBOARD AFT FLAP TRACK BEARING BORE LOCATION

A

VIEW IN THE OUTBOARD DIRECTION

B

1885704 S0000344810_V1

Location of the Outboard Flap #4 Aft Flap Track Bushing Bore
Figure 201 (Sheet 2 of 2)

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3. References

Reference	Title
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
BAC 5619	Heat Treat of Corrosion Resistant Steel
BAC 5625	SURFACE TREATMENTS FOR FERROUS ALLOYS
CMM 57-53-06	Flap Assembly - Outboard Trailing Edge
SOPM 20-50-03	Bearing and Bushing Replacement

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the initial bearing. Refer to SOPM 20-50-03.

C. Ream the lug bore to a maximum of 0.95 in. (24.1 mm) in diameter.

D. Chamfer the inboard and outboard edges of the lug bore. Refer to Figure 203/REPAIR 9.

E. Make the Part [1] Repair Sleeve. See Table 201/REPAIR 9.

(1) The outer diameter of the Part [1] Repair Sleeve must be between 0.0007 in. (0.018 mm) and 0.0020 in. (0.051 mm) larger than the reamed bore diameter.

(2) The wall thickness of the Part [1] Repair Sleeve must be a minimum of 0.040 in. (1.02 mm) thick after you do the inner diameter ream.

(3) Heat treat the Part [1] Repair Sleeve. Refer to BAC 5619.

(a) If you make the Part [1] Repair Sleeve out of 15–5PH, heat treat the sleeve to between 140 KSI and 160 KSI.

(b) If you make the Part [1] Repair Sleeve out of 17–4PH, heat treat the sleeve to between 150 KSI and 170 KSI.

(4) Make a grease groove in the Part [1] Repair Sleeve:

(a) Make a 0.090 in. (2.29 mm) wide and 0.015 in. (0.38 mm) deep grease groove in the outer diameter of the Part [1] Repair Sleeve. See Figure 203/REPAIR 9.

NOTE: The grease groove in the Part [1] Repair Sleeve must align with the grease groove in the Part [2] Bearing.

(b) Make four 0.090 in. (2.29 mm) diameter holes through the grease groove to make a grease path to the bearing. Make sure that there is an equal distance between all the holes. See Figure 203/REPAIR 9.

Table 201: Repair Materials

ITEM	PART	QUANTITY	MATERIAL
[1]	Repair Sleeve	1	Use 15–5PH steel per AMS 5659, or use 17–4PH steel per AMS 5643
[2]	Bearing	1	Same size and type as the initial bearing.

F. Remove all nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Repair Sleeve and the lug bore repair area.

G. Make the surfaces of the Part [1] Repair Sleeve and the lug bore repair area smooth to a finish of 63 microinches RA or smoother.

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H. Passivate the Part [1] Repair Sleeve and the reworked surfaces of the flap track. Refer to Method II of BAC 5625.

WARNING: WHEN YOU DO WORK WITH DRY ICE, MAKE SURE YOU USE SAFETY GOGGLES AND INSULATED GLOVES. IF DRY ICE TOUCHES YOUR EYES OR SKIN, FROSTBITE CAN OCCUR.

WARNING: LIQUID NITROGEN IS AN AGENT THAT IS AN ASPHYXIANT. MAKE SURE ALL PERSONS OBEY ALL OF THE PRECAUTIONS WHEN LIQUID NITROGEN IS USED.

- USE IN AN AREA OPEN TO THE AIR.
- CLOSE THE CONTAINER WHEN NOT USED.
- DO NOT GET LIQUID NITROGEN IN THE EYES, ON THE SKIN, OR ON YOUR CLOTHES.
- DO NOT BREATHE THE GAS.

Install the Part [1] Repair Sleeve with a layer of grease. Use a shrink fit method. Install the Part [1] Repair Sleeve with BMS 3-33. Refer to SOPM 20-50-03, Section 7, paragraph B.

J. Swage the Part [1] Repair Sleeve on both the inboard and outboard ends. See Figure 203/REPAIR 9.

K. Ream the inner diameter of the Part [1] Repair Sleeve to the necessary diameter for the Part [2] Bearing. Refer to CMM 57-53-06.

L. Chamfer the inboard and outboard edges of the Part [1] Repair Sleeve to the necessary dimensions for the Part [2] Bearing. See Figure 203/REPAIR 9.

M. Install the Part [2] Bearing. Refer to CMM 57-53-06.

N. Do a push-out load test on the bearing. Refer to SOPM 20-50-03.

O. Refinish the repair area around the lug bore if necessary. Refer to AMM SECTION 51-21.

P. Replace the items you removed for access.

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NOTES

- ALL DIMENSIONS ARE INCHES (mm)

1 THE NOMINAL DIAMETER OF THE BEARING RING IS 0.7813 INCH (19.845 mm).

2 ALIGN THE GREASE GROOVE OF THE [1] REPAIR SLEEVE WITH THE GREASE GROOVE OF THE [2] BEARING.

1888539 S0000344827_V1

Notes and Fastener Symbols
Figure 202

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ROTATED 90° CLOCKWISE
SECTION A-A

1888541 S0000344828_V1

Repair of the Outboard Flap #4 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 1 of 2)

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[1] REPAIR SLEEVE
NOT SWAGED

A

B–B

GREASE GROOVE DETAIL

B

1888542 S0000344829_V1

Repair of the Outboard Flap #4 Aft Flap Track Oversized Bushing Bore
Figure 203 (Sheet 2 of 2)

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REPAIR 10 - WING TRAILING EDGE MAIN FLAP TRACK NUMBERS 1, 2, 3, 6, 7, AND 8

1. Applicability

A. Repair 10 is applicable to damage on Main Flap Track Numbers 1, 2, 3, 6, 7, and 8 of the Wing Trailing Edge. Refer to Figure 201/REPAIR 10.

B. Repair 10 is applicable to damage that is not more than 0.020 in. (0.51 mm) in depth into the base metal of the flap track. If the depth of the damage is more than 0.020 in. (0.51 mm), contact The Boeing Company for repair instructions, or replace the flap track.

1917031 S0000347951_V1

Wing Trailing Edge Flap Track Location
Figure 201

2. General

A. All permitted damage depths are the depth into the base metal of the flap track. If there is a coating on the flap track, do not include the thickness of the coating when you calculate the damage depth.

B. If you do not remove more than 0.010 in. (0.25 mm) of material from a flange, then Repair 10 is a Category A repair.

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C. If you remove more than 0.010 in. (0.25 mm) of material from a flange, and no more than 0.020 in. (0.51 mm), then Repair 10 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 10. Incorporation of these inspection requirements into the airplanes maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for repair categories and definitions.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-06	SHOT PEENING
BAC 5625	SURFACE TREATMENTS FOR FERROUS ALLOYS
CMM 57-53-03	Inboard and Outboard Flap Track Assembly
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-01	Magnetic Particle Inspection
SOPM 20-20-02	Penetrant Methods of Inspection

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the damage. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02 for the investigation and clean up procedures.

(1) To remove the damage, use a 1.00 in. (25.4 mm) minimum blend radius and a minimum 20:1 taper ratio. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

C. Do a Class A critical magnetic particle inspection of the reworked area. Refer to SOPM 20-20-01.

NOTE: If you do the magnetic particle inspection, then the inspection must meet the Class A acceptance criteria.

NOTE: As an alternative, you are permitted to do a fluorescent penetrant inspection of the reworked area. Refer to SOPM 20-20-02.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

D. Flap peen or shot peen the reworked area of the track. Use an intensity of 0.006A to 0.011A with 200% coverage. Refer to SHOT PEENING, 51-20-06 and SOPM 20-10-03.

E. Do one of the steps that follow:

(1) Apply a flame spray coating of BMS10-67 Type I, tungsten carbide powder, to the reworked area. This will improve the wear resistance of the track. Refer to

- CMM 57-53-03, Repair 3-2 for Flap Track Numbers 3 and 6.
- CMM 57-53-03, Repair 4-2 for Flap Track Numbers 2 and 7.
- CMM 57-53-03, Repair 5-2 for Flap Track Numbers 1 and 8.

NOTE: If you apply a flame spray coating, then the coating thickness must not be more than 0.010 in. (0.25 mm). It is not necessary to keep the flange to flange dimension of the flap track cross section as shown in the applicable repair of CMM 57-53-03. Refer to Figure 202/REPAIR 10 for the flange to flange dimension.

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(2) Passivate the reworked area as given in BAC 5625, Method II.

NOTE: As an alternative, you are permitted to do local passivation as given in section 8.4.2 of BAC 5625.

F. Replace the items that you removed for access.

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LEFT SIDE FLAP TRACK NUMBER 1 IS SHOWN,
RIGHT SIDE FLAP TRACK NUMBER 8 IS OPPOSITE

LEFT SIDE FLAP TRACK NUMBER 2 IS SHOWN,
RIGHT SIDE FLAP TRACK NUMBER 7 IS OPPOSITE

LEFT SIDE FLAP TRACK NUMBER 3 IS SHOWN,
RIGHT SIDE FLAP TRACK NUMBER 6 IS OPPOSITE

TRACK WEAR ZONE

1917068 S0000349775_V1

Damage Removal Limits
Figure 202 (Sheet 1 of 3)

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1917264 S0000355157_V1

Damage Removal Limits
Figure 202 (Sheet 2 of 3)

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B–B

NOTES

- IF THERE IS A COATING ON THE FLAP TRACK, THEN DO NOT INCLUDE THE THICKNESS OF THE COATING WHEN YOU MEASURE THE DISTANCES. ONLY MEASURE THE DISTANCES FROM THE BASE METAL.

1 REMOVE WEAR DAMAGE WITH A MINIMUM 1.00 INCH (25.4 mm) RADIUS.

2 NO WEAR DAMAGE OR REWORK PERMITTED IN THIS AREA.

3 YOU MUST MEASURE THE DAMAGE DEPTH ON THE UPPER FLANGE, THE DAMAGE DEPTH ON THE LOWER FLANGE, AND THE SEPARATION BETWEEN THE FLANGES AT THE DAMAGE LOCATIONS. THE DAMAGE THAT YOU REMOVE MUST NOT BE MORE THAN THE DIMENSIONS PERMITTED.

1917293 S0000194351_V1

Damage Removal Limits
Figure 202 (Sheet 3 of 3)

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5. **Inspection Requirements**

A. If the removed depth on a flange is more than 0.010 in. (0.25 mm) and not more than 0.020 in. (0.508 mm), then you must do inspections of the repaired area. See Table 201/REPAIR 10.

Table 201: Inspection Requirements

INSPECTION THRESHOLD	REPEAT INSPECTIONS		
	METHOD	INTERVAL	REFERENCE
24 Months or 5,500 Flight Cycles, whichever comes sooner	Detailed visual inspection *[1]	6 Months or 1,000 Flight Cycles, whichever comes sooner	INSPECTION AND REMOVAL OF DAMAGE, 51-10-02

*[1] Do a detailed visual inspection of the flap track. Make sure that the depth of the wear damage on the horizontal flange is not more than a maximum of 0.020 in. (0.51 mm) into the base metal. Make sure that there is no other damage to the flap track.

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REPAIR 11 - WING TRAILING EDGE MAIN FLAP TRACK NUMBERS 4 AND 5

1. Applicability

A. Repair 11 is applicable to damage on Main Flap Track Numbers 4 and 5 of the Wing Trailing Edge. Refer to Figure 201/REPAIR 11.

B. Repair 11 is applicable to damage that is not more than 0.016 in. (0.41 mm) in depth into the base metal of the flap track, unless noted differently. Refer to Figure 202/REPAIR 11 for permitted damage depths. If the depth of the damage is more than permitted, contact The Boeing Company for repair instructions, or replace the flap track.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

1917570 S0000349798_V1

Wing Trailing Edge Flap Track Location
Figure 201

2. **General**

A. All permitted damage depths are the depth into the base metal of the flap track. If there is a coating on the flap track, do not include the thickness of the coating when you calculate the damage depth.

B. If you do not remove more than 0.006 in. (0.15 mm) of material from a flange, then Repair 11 is a Category A repair.

NOTE: You are not permitted to remove more than 0.006 in. (0.15 mm) on the lower flange of the flap track between P1 and P2. Refer to Figure 202/REPAIR 11 for permitted damage depths.

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C. If you remove more than 0.006 in. (0.15 mm) of material from a flange, and no more than 0.016 in. (0.41 mm), then Repair 11 is a Category B repair. This Category B repair has FAA approval if you do the supplemental inspections given in Paragraph 5./REPAIR 11. Incorporation of these inspection requirements into the airplanes maintenance program satisfies the damage tolerance assessment of the repair. Refer to STRUCTURAL REPAIR DEFINITIONS, 51-00-06 for repair categories and definitions.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-06	SHOT PEENING
CMM 57-53-03	Inboard and Outboard Flap Track Assembly
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection

4. Repair Instructions

A. Get access to the damaged area.

B. Remove the damage. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02 for the investigation and cleanup procedures.

(1) To remove the damage, use a 1.00 in. (25.4 mm) minimum blend radius and a minimum 20:1 taper ratio. Refer to INSPECTION AND REMOVAL OF DAMAGE, 51-10-02.

C. Do a fluorescent penetrant inspection of the reworked area. Refer to SOPM 20-20-02.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT, AN INJURY CAN BE THE RESULT.

D. Flap peen or shot peen the reworked areas of the track. Use an intensity of 0.004A to 0.010A with 200% coverage. Refer to SHOT PEENING, 51-20-06 and SOPM 20-10-03.

E. Apply a flame spray coating of BMS10-67 Type I, tungsten carbide powder, to the reworked area. Refer to CMM 57-53-03, Repair 2-2.

NOTE: If you apply a flame spray coating, then the coating thickness must not be more than 0.010 in. (0.25 mm). It is not necessary to keep the flange to flange dimension of the flap track cross section as shown in CMM 57-53-03, Repair 2-2. Refer to Figure 202/REPAIR 11 for the flange to flange dimension.

F. Replace the items that you removed for access.

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LEFT SIDE FLAP TRACK NUMBER 4 IS SHOWN,
RIGHT SIDE FLAP TRACK NUMBER 5 IS OPPOSITE

A

1917697 S0000356688_V1

Damage Removal Limits
Figure 202 (Sheet 1 of 3)

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1921213 S0000194352_V1

Damage Removal Limits
Figure 202 (Sheet 2 of 3)

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NOTES

- IF THERE IS A COATING ON THE FLAP TRACK, THEN DO NOT INCLUDE THE THICKNESS OF THE COATING WHEN YOU MEASURE THE DISTANCES. ONLY MEASURE THE DISTANCES FROM THE BASE METAL.

1 REMOVE WEAR DAMAGE WITH A MINIMUM 1.00 INCH (25.4 mm) RADIUS.

2 NO WEAR DAMAGE OR REWORK PERMITTED IN THIS AREA.

3 YOU MUST MEASURE THE DAMAGE DEPTH ON THE UPPER FLANGE, THE DAMAGE DEPTH ON THE LOWER FLANGE, AND THE SEPARATION BETWEEN THE FLANGES AT THE DAMAGE LOCATIONS. THE DAMAGE THAT YOU REMOVE MUST NOT BE MORE THAN THE DIMENSIONS PERMITTED.

4 0.006 INCH (0.15 mm) IS THE MAXIMUM PERMITTED DAMAGE DEPTH ON THE LOWER FLANGE BETWEEN P1 AND P2. SEE DETAIL A.

1918503 S0000356689_V1

Damage Removal Limits
Figure 202 (Sheet 3 of 3)

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5. Inspection Requirements

A. If the removed depth on a flange is more than 0.006 in. (0.15 mm) and not more than 0.016 in. (0.41 mm), then you must do inspections of the repaired area. See Table 201/REPAIR 11.

Table 201: Inspection Requirements

INSPECTION THRESHOLD	REPEAT INSPECTIONS		
	METHOD	INTERVAL	REFERENCE
24 Months or 5,500 Flight Cycles, whichever comes sooner	Detailed visual inspection*[1]	6 Months or 1,000 Flight Cycles, whichever comes sooner	INSPECTION AND REMOVAL OF DAMAGE, 51-10-02

*[1] Do a detailed visual inspection of the flap track. Make sure that the depth of the wear damage on the horizontal flange is not more than a maximum of 0.016 in. (0.41 mm) into the base metal. Make sure that the wear damage is not more than 0.006 in. (0.15 mm) into the base metal on the lower flange of the flap track between P1 and P2. Refer to Figure 202/REPAIR 11 for permitted damage depths. Make sure that there is no other damage to the flap track.

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REPAIR 14 - OUTBOARD AFT FLAP INBOARD CLOSURE RIB CRACK REPAIR

1. Applicability

A. This repair is applicable to crack damage on the inboard closure rib (113A3712-3/-4) of the outboard aft flap as shown in Figure 201/REPAIR 14.

2. General

A. This repair is a permanent repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repairs.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

(113A3326-5/-6 CARTRIDGE AND 113A3171-83/-84
SEAL RETAINER NOT SHOWN FOR CLARITY)
INBOARD CLOSURE RIB

B

2353570 S0000535177_V1

Outboard Aft Flap Inboard Closure Rib Location
Figure 201

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-02, GENERAL	Inspection and Removal of Damage
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-20-05, REPAIR 1	Wet Installation of Fasteners
51-30-01, GENERAL	Sheet Metal Materials
51-40-02, GENERAL	Fastener Installation and Removal
51-40-08, GENERAL	Countersink Data and Procedures for Metal Structures
AMM 27-51-22-000-801	Outboard Aft Flap Removal (P/B 401)
AMM 27-51-22-400-801	Outboard Aft Flap Installation (P/B 401)
AMM 51-21	INTERIOR AND EXTERIOR FINISHES
AMM 51-21-91	CORROSION INHIBITING COMPOUND
AMM 51-31-00-390-805	Fastener Seal Application (P/B 201)
SOPM 20-50-19	General Sealing

4. Repair Instructions

A. Get access to the damaged area. Remove the outboard aft flap from the aircraft. Refer to AMM TASK 27-51-22-000-801.

B. Measure and record dimension “XX” shown in Figure 206 (Sheet 2) View A. Use this dimension when you install the new cartridge assembly.

C. Make or get the new cartridge assembly. Make the cartridge assembly as given in drawing number 113A3326-11/-12 or order part number 113A3326-11/-12 from The Boeing Company.

D. Remove the initial cartridge assembly(113A3326-5/-6) and the initial seal retainer (113A3171-83/-84) from the inboard closure rib. Use the initial cartridge assembly (113A3326-5/-6) to identify the fastener holes on the new cartridge assembly (113A3326-11/-12).

E. You can keep and use the roller components of the initial cartridge assembly. It is recommended that you discard the initial cartridge assembly after you drill the fastener holes on the new cartridge assembly.

F. Do a 0.25 in. (6.35 mm) diameter stop drill at the end of the crack as given in 51-10-02, GENERAL with the exceptions as follow:

(1) The crack or the stop hole cannot extend into the fillet radius at the web of the inboard closure rib.

NOTE: If this condition cannot be met, then you cannot do this repair and the inboard closure rib must be replaced.

(2) You do not have to install plug rivets.

G. Remove the vertical stiffener from the inboard closure rib as shown in Figure 203/REPAIR 14.

(1) Make sure the vertical stiffener is completely removed on the upper flange of the inboard closure rib with a tolerance of +0/-0.010 in. (-0.254 mm).

(2) Make sure that the thickness of the inboard closure rib lower flange does not decrease. Make sure that the vertical stiffener on the lower rib flange protrudes no more than 0.250 in. (6.350 mm) and no less than 0.030 in. (0.762 mm) after removal.

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(3) Make sure the vertical stiffener on the vertical flange does not protrude more than 0.030 in. (0.762 mm) after removal.

H. Trim the 113A3171-83/-84 seal retainer as shown in Figure 204/REPAIR 14.

I. Remove the three forward nutplates on the modified seal retainer.

J. Make the Part [1] Doubler. Refer to Table 201/REPAIR 14 and Figure 205/REPAIR 14. Make the contour of the Part [1] Doubler the same as the contour of the inboard closure rib.

Table 201: Repair Material

PART		QUANTITY	MATERIAL
1	Doubler	1	0.063 in. (1.60 mm) thick, 7075-T6 bare sheet as given in AMS 4045.

K. Assemble the Part [1] Doubler, the modified seal retainer, and the cartridge. Refer to Figure 206/REPAIR 14.

(1) Align the fastener holes of the cartridge with the inboard closure rib.

(2) The vertical face of the cartridge must be:

(a) Positioned dimension “XX” (as measured in Paragraph 4.B./REPAIR 14) +/- 0.005 in. (0.127 mm) away from the edge of the inboard closure rib upper flange at the forward end.

(b) 0.35 in. (8.89 mm) away from the edge of the inboard closure rib upper flange at the aft end.

(3) Use a BACF3T01A08-43 tapered filler at the three most forward nutplate locations of the modified seal retainer. Refer to Figure 206 (Sheet 4).

(4) Use a BACS40R009C047F shim between the forward lower flange of the inboard closure rib and the cartridge. Use a BACS40R009C040F shim between the aft lower flange of the inboard closure rib and the cartridge. Taper the shims as necessary. The pull-up must be 0.005 in. (0.127 mm) or less. Refer to 51-30-01, GENERAL.

(5) Align the forward and outboard edges of the Part [1] Doubler to the inboard closure rib.

L. Drill the fastener holes. Refer to Figure 206/REPAIR 14. Countersink the fastener holes on the upper surface of the Part [1] Doubler and the lower surface of the cartridge.

M. Disassemble the Part [1] Doubler, the modified seal retainer, and the cartridge.

N. Remove the nicks, scratches, gouges, burrs, and sharp edges from the Part [1] Doubler, the modified seal retainer, the cartridge (if necessary) and the inboard closure rib.

O. Apply a chemical conversion coating to the Part [1] Doubler and to the bare surfaces of the modified seal retainer, the cartridge, and the inboard closure rib. Refer to 51-20-01, GENERAL.

P. Apply one layer of BMS 10-79 primer to the Part [1] Doubler and to the bare surfaces of the modified seal retainer, the cartridge, and the inboard closure rib.

Q. Install the countersink repair washers with BMS 5-95 sealant at the necessary fastener locations. Refer to 51-40-08, GENERAL.

R. Assemble the BACF3T01A08-43 tapered filler and the modified seal retainer with a layer of BMS 5-95 sealant between the mating surfaces.

S. Install the seven BACN10KE3B3CD nutplates to the lower surface of the upper flange of the cartridge with BACR15BA3AD3C fasteners. Install the fasteners wet with BMS 5-95 sealant.

T. Install the Part [1] Doubler, the modified seal retainer, and the cartridge.

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(1) Install the three BACN10KE3B3CD nutplates to the BACF3T01A08-43 tapered filler and the modified seal retainer with BACR15BA3AD( )C fasteners. Install the fasteners wet with BMS 5-95.

(2) Install the modified seal retainer and tapered filler assembly to the upper surface of the lower flange of the inboard closure rib with a layer of BMS 5-95 sealant between the mating surfaces.

(3) Install the cartridge. Make a removable fay surface seal for the cartridge. Apply a parting agent to the cartridge before installation. Apply BMS 5-95 sealant to the mating surface on the inboard closure rib and shims as a removable fay surface seal. Make sure that you also apply a fay surface seal on the other side of the shims with BMS 5-95 sealant. Refer to 51-20-05, GENERAL and SOPM 20-50-19.

(4) Install the Part [1] Doubler with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05, GENERAL.

U. Install the fasteners. Fasteners must be installed wet with BMS 5–95 sealant. Refer to 51-20-05, REPAIR 1 and AMM TASK 51-31-00-390-805. Refer to 51-40-02, GENERAL.

V. Apply a chemical conversion coating to the bare surfaces of the fasteners. If necessary, apply a chemical conversion coating to the Part [1] Doubler and to the bare surfaces of the modified seal retainer, the cartridge, and the inboard closure rib. Refer to 51-20-01, GENERAL.

W. If necessary, apply one layer of BMS 10-79 primer to the Part [1] Doubler and to the bare surfaces of the modified seal retainer, the cartridge, and the inboard closure rib.

X. Apply a layer of BMS 10-60, Type II, BAC 707 color, gloss enamel to the external surfaces of the repair area. Make sure that you do not apply enamel to the bushing on the cartridge assembly. Refer to AMM SECTION 51-21.

Y. Apply BMS 3-23 corrosion inhibiting compound to the repair area. Make sure that you do not apply compound to the bushing on the cartridge assembly. Refer to 51-20-01, GENERAL and AMM SUBJECT 51-21-91.

Z. Install the roller components of the cartridge assembly. You are permitted to use the roller components of the initial cartridge assembly. Refer to AMM TASK 27-51-22-400-801.

AA. Reinstall the outboard aft flap on the aircraft. Refer to AMM TASK 27-51-22-400-801.

AB. Adjust the outboard aft flap on the aircraft. Refer to AMM TASK 27-51-22-400-801.

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NOTES

- ALL DIMENSIONS ARE IN INCHES (mm).

1 FORM THE SEAL RETAINER AS NECESSARY SO THAT THE VERTICAL FACE OF THE CARTRIDGE IS FLUSH WITH THE RADIUS OF THE SEAL RETAINER TO WITHIN +/- 0.010 INCH (0.254 mm) IN THE INBOARD/OUTBOARD DIRECTION.

2 MEASURED ALONG THE UPPER SURFACE OF THE OUTBOARD AFT FLAP.

3 MAKE A REMOVABLE FAY SURFACE SEAL. APPLY THE PARTING AGENT TO THE 113A3326 CARTRIDGE. APPLY BMS 5-95 SEALANT TO THE OTHER MATING SURFACE AS A REMOVABLE FAY SURFACE SEAL. REFER TO SRM 51-20-05 AND SOPM 20-50-19.

4 TAPER THE SHIM AS NECESSARY TO GET A PULL-UP OF 0.005 INCH (0.127 mm) OR LESS.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A BACB30VF3KF() BOLT.

REPAIR FASTENER LOCATION. INSTALL A BACB30VF3KF() BOLT.

2355916 S0000538243_V1

Notes and Fastener Symbols
Figure 202

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INBOARD CLOSURE RIB
(VERTICAL STIFFENER REMOVED)

A–A

B–B

2353702 S0000535272_V1

Inboard Closure Rib Modification
Figure 203

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INITIAL SEAL RETAINER
(BEFORE IT IS TRIMMED)

MODIFIED SEAL RETAINER
(AFTER IT IS TRIMMED)

MODIFIED SEAL RETAINER
(113A3171-83 IS SHOWN, 113A3171-84 IS OPPOSITE)
A-A

2353781 S0000535273_V1

Seal Retainer Modification
Figure 204

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PART [1] DOUBLER PROFILE

2353923 S0000537706_V1

Part [1] Doubler Profile
Figure 205

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OTHER STRUCTURES NOT SHOWN FOR CLARITY

2355543 S0000538244_V2

Outboard Aft Flap Inboard Closure Rib Crack Repair
Figure 206 (Sheet 1 of 4)

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TOP VIEW
OTHER STRUCTURES NOT SHOWN FOR CLARITY

A

BOTTOM VIEW
OTHER STRUCTURES NOT SHOWN FOR CLARITY

B

2354029 S0000537708_V2

Outboard Aft Flap Inboard Closure Rib Crack Repair
Figure 206 (Sheet 2 of 4)

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(APPLICABLE TO 4 FORWARD PAIRS OF FASTENERS)
A-A

2355940 S0000538245_V1

Outboard Aft Flap Inboard Closure Rib Crack Repair
Figure 206 (Sheet 3 of 4)

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(APPLICABLE TO 3 AFT PAIRS OF FASTENERS)
B-B

TOP VIEW
C

2365422 S0000541486_V2

Outboard Aft Flap Inboard Closure Rib Crack Repair
Figure 206 (Sheet 4 of 4)

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IDENTIFICATION 1 - FAIRING SKINS - WING OUTBOARD TRAILING EDGE FLAP SUPPORTS - NUMBERS 1 AND 8

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H21015 S0006594983_V1

Wing Trailing Edge Flap Support Fairing Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A0001	Movable Trailing Edge Functional Product Collector
113A9100	Fairing Installation - Flap Support Numbers 1 and 8, Outboard Trailing Edge Flap
113A9110	Forward Fairing Assembly - Flap Support Number 1, Outboard Trailing Edge Flap
113A9150	Aft Fairing Assembly - Flap Support Number 1, Outboard Trailing Edge Flap
113A9252	Tailcone Assembly - Flap Support Fairing Numbers 1,2,7,8

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NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H21034 S0006594985_V1

Outboard Trailing Edge Flap Support Fairing Skins
Figure 2

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H21108 S0006594986_V1

Outboard Trailing Edge Flap Support Forward Fairing Skin Identification
Figure 3

Table 2:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Skin Assembly		Glass Fiber Reinforced Plastic (GFRP) honeycomb sandwich	
	Assembly			
	Skin		Refer to Figure 6	
	Core (1)	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
	Core (1)	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H21283 S0006594988_V1

Outboard Trailing Edge Flap Support Aft Fairing Skin Identification
Figure 4

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Table 3:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Skin Assembly		GFRP and Carbon Fiber Reinforced Plastic (CFRP) honeycomb sandwich	
	Skin		Refer to Figure 7	
	Core	0.50 (12.7)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Bonded Access Door Assembly		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) Refer to Figure 8	
[3]	Bonded Aft Fairing Cap		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781, 120, or 220) Refer to Figure 9	
[4]	Bonded Access Cover		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

NUMBER 1 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 8 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H21575 S0006594990_V1

Outboard Trailing Edge Flap Support Tailcone Fairing Skin Identification
Figure 5

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Table 4:

LIST OF MATERIALS FOR FIGURE 5				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Lower Bonded Tailcone Assembly		Glass epoxy honeycomb sandwich	
	Skin		Refer to Figure 10	
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Upper Bonded Tailcone Assembly		Glass epoxy honeycomb sandwich	
	Skin		Refer to Figure 11	
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B, C-C, AND D-D FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H22298 S0006594992_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]
Figure 6 (Sheet 1 of 2)

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Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]
Figure 6 (Sheet 2 of 2)

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Table 5:

PLY MATERIAL AND DIRECTION OF THE FORWARD FAIRING BONDED SKIN FOR FIGURE 3 *[1]		
PLY	**DIRECTION**	**MATERIAL**
P1, P10, P11, and P20	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
P2 thru P4, P6, P8, P13, P15, and P17 thru P19	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P5, P7, P9, P12, P14, and P16	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220)

*[1] Note: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, D-D, E-E, F-F, AND G-G FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H22522 S0006594995_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 1 of 5)

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PLIES [P2], [P3], [P11], [P12], [P15], [P16], [P24], AND [P25] NOT SHOWN FOR CLARITY

A–A

PLIES [P2], [P3], [P11], [P12], [P15], [P16], [P24], AND [P25] NOT SHOWN FOR CLARITY

B–B

H22850 S0006594996_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 2 of 5)

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PLIES [P2], [P3], [P11], [P12], [P15], [P16], [P24], AND [P25] NOT SHOWN FOR CLARITY

B

C–C

H22924 S0006594997_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 3 of 5)

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Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 4 of 5)

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F–F

G–G

H23256 S0006594999_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 5 of 5)

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 7*[1]		
PLY	**DIRECTION**	**MATERIAL**
P1, P13, P14, and P26	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
P2, P11, P16, and P25	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P3, P12, P15, and P24	+ or - 45 degrees	CFRP as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW
P4, P9, P18, and P23	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P5 thru P8, P10, P17, and P19 thru P22	0 or 90 degrees	CFRP as given in BMS 8-168, Type I, Class 2, Style 3K-70-PW
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220)

*[1] Note: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

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Warp Direction of Material for Bonded Access Door, Figure 4, Item [2]
Figure 8

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H23356 S0006595002_V1

Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [3]
Figure 9

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H23469 S0006595003_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 10 (Sheet 1 of 2)

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A–A

B–B

H23485 S0006595004_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 10 (Sheet 2 of 2)

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P3, and P8 thru P10	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
P4 thru P7	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781)

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H23546 S0006595006_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]
Figure 11

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 11		
PLY	**DIRECTION**	**MATERIAL**
P11 thru P16	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781)

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IDENTIFICATION 2 - FAIRING SKINS - WING OUTBOARD TRAILING EDGE FLAP SUPPORTS - NUMBERS 2 AND 7

H23564 S0006595009_V4

Wing Trailing Edge Flap Support Fairing Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A0001	Movable Trailing Edge Functional Product Collector
113A9200	Fairing Installation - Flap Support Numbers 2 and 7, Outboard Trailing Edge Flap
113A9210	Forward Fairing Assembly - Flap Support Number 2, Outboard Trailing Edge Flap
113A9250	Aft Fairing Assembly - Flap Support Number 2, Outboard Trailing Edge Flap
113A9252	Tailcone Assembly - Flap Support Fairing Numbers 1,2,7,8

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NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

NUMBER 2 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H23578 S0006595011_V1

Outboard Trailing Edge Flap Support Fairing Skins
Figure 2

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

NUMBER 2 FLAP SUPPORT FORWARD FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT FORWARD FAIRING (RIGHT SIDE) IS OPPOSITE

H23603 S0006595012_V1

Outboard Trailing Edge Flap Support Forward Fairing Skin Identification
Figure 3

Table 2:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Skin Assembly		Glass Fiber Reinforced Plastic (GFRP) honeycomb sandwich	
	Skin		Refer to Figure 6	
	Core (2)		Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

NUMBER 2 FLAP SUPPORT AFT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT AFT FAIRING (RIGHT SIDE) IS OPPOSITE

H23784 S0006595014_V1

Outboard Trailing Edge Flap Support Aft Fairing Skin Identification
Figure 4

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Table 3:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Skin Assembly Skin Core (2)	 0.50 (12.7) 0.25 (6.4)	CFRP and GFRP honeycomb sandwich Refer to Figure 7 Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Bonded Access Door Assembly		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781). Refer to Figure 8	
[3]	Bonded Aft Fairing Cap		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781, 120, or 220). Refer to Figure 9	
[4]	Bonded Access Cover		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

NUMBER 2 FLAP SUPPORT TAILCONE FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 7 FLAP SUPPORT TAILCONE FAIRING (RIGHT SIDE) IS OPPOSITE

H23807 S0006595016_V1

Outboard Trailing Edge Flap Support Tailcone Fairing Skin Identification
Figure 5

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Table 4:

LIST OF MATERIALS FOR FIGURE 5				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Lower Bonded Tailcone Assembly		GFRP honeycomb sandwich	
	Skin		Refer to Figure 10	
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Upper Bonded Tailcone Assembly		GFRP honeycomb sandwich	
	Skin		Refer to Figure 11	
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO SECTION A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B AND C-C FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H24105 S0006595018_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]
Figure 6 (Sheet 1 of 2)

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B–B

C–C

H24253 S0006595019_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]
Figure 6 (Sheet 2 of 2)

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Table 5:

PLY MATERIAL AND DIRECTION FOR FIGURE 6		
PLY	**DIRECTION**	**MATERIAL**
P1, P10, P11, and P20	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120. (Optional: Style 220)
P2 thru P4, P6, P8, P13, P15, and P17 thru P19	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P5, P7, P9, P12, P14, and P16	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220)

NOTE: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A, B-B, C-C, AND D-D FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H24401 S0006595021_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 1 of 3)

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PLIES [P2], [P3], [P10], [P11], [P14], [P15], [P22], AND [P23] NOT SHOWN FOR CLARITY

A-A

PLIES [P2], [P3], [P10], [P11], [P14], [P15], [P22], AND [P23] NOT SHOWN FOR CLARITY

B-B

H24565 S0006595022_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 2 of 3)

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C–C

D–D

H24635 S0006595023_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 3 of 3)

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 7		
PLY	**DIRECTION**	**MATERIAL**
P1, P12, P13, and P24	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120. (Optional: Style 220)
P2, P10, P15, and P23	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P3, P11, P14, and P22	+ or - 45 degrees	Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW
P4, P9, P16, and P21	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P5 thru P8, and P17 thru P20	0 or 90 degrees	Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW
Filler Plies	+ or - 45 degrees	Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-168, Type II, Class 2, Style 3K-70-PW

NOTE: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

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Warp Direction of Material for Bonded Access Door, Figure 4, Item [2]
Figure 8

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H25113 S0006595026_V1

Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [3]
Figure 9

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H25182 S0006595027_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 10 (Sheet 1 of 2)

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A–A

B–B

H25192 S0006595028_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 10 (Sheet 2 of 2)

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P3, and P8 thru P10	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
P4 thru P7	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781)

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H25214 S0006595030_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]
Figure 11

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 11		
PLY	**DIRECTION**	**MATERIAL**
P11 thru P16	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120 (Optional: Style 220)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 120, 220, or 7781)

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IDENTIFICATION 3 - FAIRING SKINS - WING INBOARD TRAILING EDGE FLAP SUPPORTS - NUMBERS 3 AND 6

H25263 S0006595033_V1

Wing Trailing Edge Flap Support Fairing Locations
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A0001	Movable Trailing Edge Functional Product Collector
113A9300	Fairing Installation - Flap Support Numbers 3 and 6, Inboard Trailing Edge Flap
113A9310	Forward Fairing Assembly - Flap Support Number 3, Inboard Trailing Edge Flap
113A9350	Aft Fairing Assembly - Flap Support Number 3, Inboard Trailing Edge Flap
113A9352	Tailcone Assembly - Flap Support Fairing Numbers 3 and 6

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NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

NUMBER 3 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H25306 S0006595035_V1

Inboard Trailing Edge Flap Support Fairing Skin Numbers
Figure 2

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

NUMBER 3 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H25363 S0006595036_V1

Inboard Trailing Edge Flap Support Forward Fairing Skin Identification
Figure 3

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Table 2:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Skin Assembly		Glass Fiber Reinforced Plastic (GFRP) honeycomb sandwich	
	Skin		Refer to Figure 6	
	Core (2)	0.50 (12.7) 0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

NUMBER 3 FLAP SUPPORT FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT FAIRING (RIGHT SIDE) IS OPPOSITE

H25486 S0006595038_V1

Inboard Trailing Edge Flap Support Aft Fairing Skin Identification
Figure 4

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Table 3:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Skin Assembly Skin Core (2)	 0.75 (19.0)	GFRP honeycomb sandwich Refer to Figure 7 Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Bonded Aft Fairing Cap		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 (Optional: Style 7781)	Airplane line numbers 1 thru 2187 and does not have SB-737–57–1288 Incorporated Refer to Figure 8
[2]	Bonded Aft Fairing Cap		Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120	Airplane line numbers 2188 and on, or airplanes that have SB-737–57–1288 incorporated Refer to Figure 9
[3]	Bonded Access Cover (2)		Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

NUMBER 3 FLAP SUPPORT TAILCONE FAIRING (LEFT SIDE) IS SHOWN,
NUMBER 6 FLAP SUPPORT TAILCONE FAIRING (RIGHT SIDE) IS OPPOSITE

H25494 S0006595040_V1

Inboard Trailing Edge Flap Support Tailcone Fairing Skin Identification
Figure 5

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Table 4:

LIST OF MATERIALS FOR FIGURE 5				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Lower Bonded Tailcone Assembly		GFRP honeycomb sandwich	
	Skin		Refer to Figure 1	Airplane line numbers 1 thru 2187 and does not have SB-737–57–1288 Incorporated
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[1]	Lower Bonded Tailcone Assembly		GFRP honeycomb sandwich	
	Skin		Refer to Figure 11	Airplane line numbers 2188 and on, or airplanes that have SB-737–57–1288 incorporated
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Upper Bonded Tailcone Assembly		GFRP honeycomb sandwich	
	Skin		Refer to Figure 12	Airplane line numbers 1 thru 2187 and does not have SB-737–57–1288 Incorporated
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	
[2]	Upper Bonded Tailcone Assembly		GFRP honeycomb sandwich	
	Skin		Refer to Figure 13	Airplane line numbers 2188 and on, or airplanes that have SB-737–57–1288 incorporated
	Core	0.25 (6.4)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type VI, Grade 3.0	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO SECTION A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B AND C-C FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H26261 S0006595042_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]
Figure 6 (Sheet 1 of 2)

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B–B

C–C

H26280 S0006595043_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Forward Fairing Panel, Figure 3, Item [1]
Figure 6 (Sheet 2 of 2)

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Table 5:

PLY MATERIAL AND DIRECTION FOR FIGURE 6		
PLY	**DIRECTION**	**MATERIAL**
P1, P10, P11, and P20	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 120. (Optional: Style 220)
P2 thru P4, P6, P8, P13, P15, and P17 thru P19	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
P5, P7, P9, P12, P14, and P16	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: Style 7781) or Style 120 (Optional: Style 220)

NOTE: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO SECTION A-A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS B-B, C-C, AND D-D FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H26429 S0006595045_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 1 of 2)

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H26564 S0006595046_V1

Ply Direction, Core Ribbon Direction and Ply Sequence for the Flap Support Aft Fairing Panel, Figure 4, Item [1]
Figure 7 (Sheet 2 of 2)

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 7 ITEM [1]		
PLY	**DIRECTION**	**MATERIAL**
P1, P5, P6, and P10	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 120. (Optional: Style 220)
P2, P4, P7, and P9	+ or - 45 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 (Optional: Style 7781)
P3, and P8	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581. (Optional: Style 7781)
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 (Optional: Style 7781)

NOTE: The plies given above as ±45 degrees are all +45 degrees or all -45 degrees.

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A–A

H26627 S0006595048_V1

Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [2]
Figure 8

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U74405 S0000215201_V1

Warp Direction of Material for Bonded Fairing Cap, Figure 4, Item [2]
Figure 9

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 9		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P5, P6, P8, and P10	0 or 90 degrees	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
P2, P4, P7, and P9	+ or - 45 degrees	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
Filler Plies	Optional	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H26778 S0006595049_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 10 (Sheet 1 of 2)

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A-A

B-B

H26794 S0006595050_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 10 (Sheet 2 of 2)

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1 thru P3, and P8 thru P10	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 120
P4 thru P7	0 or 90 degrees	Glass fabric as given in BMS 8-139, Class I, Style 120
Filler Plies	0 or 90 degrees	Glass fabric as given in BMS 8-139, Class I, Style 120

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VIEW IS ON THE TOOL SIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

(A)

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTIONS A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

U74544 S0000215203_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 11 (Sheet 1 of 2)

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A-A

B-B

U74747 S0000215204_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Panel, Figure 5, Item [1]
Figure 11 (Sheet 2 of 2)

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Table 9:

PLY MATERIAL AND DIRECTION FOR FIGURE 11		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P4, P6, P8, P10, and P11	0 or 90 degrees	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
P13	+ or - 90 degress	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
P2, P5, P7, P9, and P12	+ or - 45 degrees	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
Filler Plies	Optional	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581

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VIEW IS ON THE TOOLSIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

H26860 S0006595052_V2

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]
Figure 12

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Table 10:

PLY MATERIAL AND DIRECTION FOR FIGURE 12		
PLY	**DIRECTION**	**MATERIAL**
P11 thru P16	0 or 90 degrees	Glass fabric as given in BMS 8-139, Class I, Style 120
Filler Plies	0 or 90 degrees	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class I, Style 1581 or Style 120 (Optional: Style 7781)

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VIEW IS ON THE TOOL SIDE (AERODYNAMIC) SURFACE
PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION AND THE CORE RIBBON DIRECTION.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION.
- REFER TO TABLE 11 FOR THE DIRECTION AND MATERIAL OF EACH PLY.

U74859 S0000215205_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Flap Support Tailcone Fairing Upper Panel, Figure 5, Item [2]
Figure 13

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Table 11:

PLY MATERIAL AND DIRECTION FOR FIGURE 13		
PLY	**DIRECTION**	**MATERIAL**
P14, P16, P17, P19, P21, P23, P24, and P26	0 or 90 degrees	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
P15, P18, P20, P22, and P25	+ or - 45 degrees	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581
Filler Plies	Optional	Fiberglass reinforced epoxy sheet as given in BMS 8-139, Class I, Style 120 or Style 1581

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ALLOWABLE DAMAGE 1 - FLAP SUPPORT FAIRING SKIN, WING TRAILING EDGE FLAP

1. Applicability

A. This subject gives the allowable damage limits for the flap support fairing skin of the wing trailing edge flap shown in Flap Support Fairing Location for the Wing Trailing Edge Flap, Figure 101/ ALLOWABLE DAMAGE 1.

G13310 S0006595056_V1

Flap Support Fairing Location for the Wing Trailing Edge Flap
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

W24285 S0006595057_V2

Flap Support Fairing Number 1 For the Outboard Trailing Edge Flap
Figure 102

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H01588 S0006595058_V2

Flap Support Fairing Number 2 For the Outboard Trailing Edge Flap
Figure 103

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G13311 S0006595059_V2

Flap Support Fairing Number 3 for the Inboard Trailing Edge Flap
Figure 104

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2. General

A. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other approved inspection methods can be used.

(1) Refer to Damage Definitions, Figure 105/ALLOWABLE DAMAGE 1 for the definitions of the length, width, depth, and diameter of damage.

(2) The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause fiberglass or graphite damage and are sealed.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two methods that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) If the tape is on an external surface, then do as follows:

1) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

(d) Repair the damage with a permanent seal at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Make a temporary seal with aluminum foil tape (speed tape).

(2) Keep a record of the location.

(3) Repair the damage at or before 400 flight hours from the time the seal was made.

3. References

Reference	Title
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

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4. **Allowable Damage Limits**

A. Zone 1 - Honeycomb Core Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the fiberglass or graphite fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the fiberglass or graphite fibers are permitted if they are:

(a) A maximum of one ply in depth.

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail A .

(3) Dents:

(a) Dents that do not cause damage to the carbon glass fibers are permitted if they are:

1) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail A .

NOTE: Use the limits for holes and punctures if there is carbon glass fiber damage or if the dent depth is more than 0.05 inch.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail A.

(5) Delamination is permitted if it is:

(a) A maximum of 2.0 inches in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail A.

B. Zone 1 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the fiberglass or graphite fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the fiberglass or graphite fibers are permitted if they are:

(a) A maximum of one ply in depth.

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail B.

(3) Dents:

(a) Dents that do not cause damage to the fiberglass or graphite fibers are permitted if they are:

1) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

NOTE: Use the limits for holes and punctures if there is carbon glass fiber damage or if the dent depth is more than 0.05 inch.

(4) Holes and Punctures are permitted if they are:

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(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail B.

(5) Delamination is permitted if it is:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail B.

C. Zone 1 - Edgeband Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the fiberglass or graphite fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the fiberglass or graphite fibers are permitted if they are:

(a) A maximum of one ply in depth.

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail B.

(3) Dents:

(a) Dents that do not cause damage to the carbon glass fibers are permitted if they are:

1) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail B.

NOTE: Use the limits for holes and punctures if there is carbon glass fiber damage or if the dent depth is more than 0.05 inch.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail B.

(5) Delamination is permitted if it is:

(a) A maximum of 0.25 inch in diameter as shown in Allowable Damage Limits, Figure 106/ ALLOWABLE DAMAGE 1, Detail B.

(6) Edge damage is permitted if it is:

(a) The maximum as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1 Details C and D if there is a fastener present.

(b) The maximum as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail E if there is not a fastener present.

(c) Sealed as given in Paragrqaph 2.

(7) Edge Erosion is permitted if it is:

(a) The maximum as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, Detail F.

D. Zone 2 - Honeycomb Core, Solid Laminate, and Edgeband Areas

(1) Damage is not permitted.

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR DENT DAMAGE

B

DEFINITIONS FOR EDGE DAMAGE

C

G13324 S0006595060_V1

Damage Definitions
Figure 105

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 2.0 INCHES.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3D SPACING.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS (HONEYCOMB SANDWICH)

A

G14920 S0006595061_V1

Allowable Damage Limits
Figure 106 (Sheet 1 of 4)

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.25 INCH.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 4D SPACING.

THE MINIMUM DISTANCE FROM AN EDGE IS 2.5D + 0.05 INCH.

THE MINIMUM DISTANCE FROM A DRAIN HOLE IS 4.5D.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS
(EDGEBANDS AND SOLID LAMINATES)

B

H01664 S0006595062_V1

Allowable Damage Limits
Figure 106 (Sheet 2 of 4)

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

C

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

D

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGEBAND

E

H01674 S0006595063_V1

Allowable Damage Limits
Figure 106 (Sheet 3 of 4)

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CLEANUP AND SEALING OF EDGE EROSION

F

G13327 S0006595064_V1

Allowable Damage Limits
Figure 106 (Sheet 4 of 4)

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REPAIR 1 - FLAP SUPPORT FAIRING SKIN, WING TRAILING EDGE FLAP

1. Applicability

A. Repair 1 is applicable to the skin panels of the flap support fairing shown in Flap Support Fairing Location for the Wing Trailing Edge Flap, Figure 201/REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

L91984 S0006595067_V3

Flap Support Fairing Location for the Wing Trailing Edge Flap
Figure 201

2. General

A. Repair 1 gives instructions for Permanent repairs for the number 1 and 2 (7 and 8 opposite) outboard flap fairing skins. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the skin as necessary to get access to the inner surface.

(1) Remove the fasteners. Refer to 51-40-02 for information on fastener removal.

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(2) If the fastener hole is damaged, refer to 51-70-04 or 51-70-05 for a repair.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Nondestructive Inspection (NDI) procedure. Refer to NDT, Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 203/REPAIR 1 for the definitions of the diameter and depth of damage.

(2) Refer to Definitions of the Facesheets, Figure 204/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

D. Do the repair as given in Paragraph 4./REPAIR 1.

E. Install the skin, as necessary.

F. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)

FLAP SUPPORT FAIRING NUMBER 1 IS SHOWN,
FLAP SUPPORT FAIRING NUMBER 2 IS SIMILAR

M95058 S0006595068_V2

Flap Support Fairing Numbers 1 and 2 For the Outboard Trailing Edge Flap
Figure 202 (Sheet 1 of 3)

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A–A

B–B

NOTE: REPAIRS PERMITTED IN ZONE 1 PROVIDED THAT:

REPAIRS PLIES DO NOT EXTEND INTO THE CORE RAMP AREA.
REPAIRS PLIES DO NOT EXTEND INTO CURVE PORTION OF THE CORNERS.

M54787 S0006595069_V1

Flap Support Fairing Numbers 1 and 2 For the Outboard Trailing Edge Flap
Figure 202 (Sheet 2 of 3)

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NOTE: REPAIRS PERMITTED IN ZONE 1 PROVIDED THAT: REPAIRS PLIES DO NOT EXTEND INTO THE CORE RAMP AREA.

C–C

2424005 S0000560740_V1

Flap Support Fairing Numbers 1 and 2 For the Outboard Trailing Edge Flap
Figure 202 (Sheet 3 of 3)

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

DEFINITIONS FOR EDGE DAMAGE

DEFINITIONS FOR DENT DAMAGE

DEFINITIONS FOR DENT DAMAGE

L92973 S0006595070_V1

Damage Definitions
Figure 203

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L92974 S0006595071_V1

Definitions of the Facesheets
Figure 204

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-05	REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS
AMM 51-21-00 P/B 701	INTERIOR AND EXTERIOR FINISHES - CLEANING/PAINTING
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Repair Instructions

A. Refer to:

(1) Table 201/REPAIR 1 for Zone 1 honeycomb core areas.

(2) Table 202/REPAIR 1 for the Zone 1 solid laminate areas.

B. Contact The Boeing Company for a Zone 2 repair or a repair made with:

(1) Room temperature wet layup materials.

(2) Wet layup materials at 150°F (66°C) cure.

C. Use the repair instructions that follow for a repair made with wet layup materials at 200°F (93°C) cure.

(1) Repair the damage as given in 51-70-04 and do the steps that follow:

(a) Use the same number of structural repair plies as the number of initial plies that are removed. Make sure that you do not remove more than (3) initial plies.

(b) Add two extra structural repair plies and one non-structural fiberglass repair ply. Put one structural ply at +/- 45 degrees to the core ribbon direction and the other at 0 or 90 degrees. Use an optional ply direction for the non-structural ply.

D. Use the repair instructions that follow for a repair with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05 and do the steps that follow:

(a) Use the same number of structural repair plies as the number of initial plies that are removed.

(b) Add one extra non-structural fiberglass repair ply with an optional ply direction.

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Table 201:

ZONE 1 - REPAIR DATA FOR THE FLAP SUPPORT FAIRING SKIN 250°F (121°C) CURE, HONEYCOMB CORE PANEL AREA				
REPAIR TYPE	**INTERIM WET LAYUP**	**PERMANENT WET LAYUP**	**PERMANENT WET LAYUP**	**PERMANENT PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**ROOM TEMPERATURE**	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**
REPAIR SIZE AND LIMITS	- Contact The Boeing Company for this temperature repair.	- Contact The Boeing Company for this temperature repair.	Damage that is a maximum of - 2.0 inches in diameter One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - panel edges - fastener holes	There are no limits on the size of the repair.
REPAIR PROCEDURES	----	---	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-05 and Paragraph 4.D

Table 202:

ZONE 1 - REPAIR DATA FOR THE FLAP SUPPORT FAIRING SKIN 250°F (121°C) CURE, SOLID LAMINATE PANEL AREAS				
REPAIR TYPE	**INTERIM WET LAYUP**	**PERMANENT WET LAYUP**	**PERMANENT WET LAYUP**	**PERMANENT PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**ROOM TEMPERATURE**	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**
REPAIR SIZE AND LIMITS	- Contact The Boeing Company for this temperature repair.	- Contact The Boeing Company for this temperature repair.	- Contact The Boeing Company for this temperature repair.	There are no limits on the size of the repair.
REPAIR PROCEDURES	----	-----	-----	SRM 51-70-05 and Paragraph 4.D

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REPAIR 2 - FLAP SUPPORT FAIRING - NUMBERS 3 AND 6 INBOARD TRAILING EDGE FLAP

1. Applicability

A. This repair is applicable to damage at the seal attachment fastener holes that are common to the 113A9380 rub strip on the flap support fairing as shown in Figure 201/REPAIR 2.

B. The damaged fastener hole diameter cannot be more than 0.50 in. (12.7 mm).

2. General

A. Repair categorization:

(1) This repair is an interim repair for Alternative 1. When you do Alternative 1 repair, do a detailed visual inspection of this interim repair every 500 flight cycles or more frequently. If damage is found, then you must replace this interim repair with a permanent repair. Refer to 51-00-06, GENERAL for the definitions of the different types of repairs.

(2) This repair is a permanent repair for Alternatives 2 and 3. Refer to 51-00-06, GENERAL for the definitions of the different types of repairs.

B. This repair has three alternatives:

(1) Alternative 1: Use this repair for flap support fairings that have a 113A9380-1 rub strip, and if the damaged fastener hole size is 0.325 in. (8.255 mm) or less. Refer to Paragraph 4./REPAIR 2 and Figure 203/REPAIR 2.

(2) Alternative 2: Use this repair for flap support fairings that have a 113A9380-1 rub strip, and if the damaged fastener hole size is 0.50 in. (12.7 mm) or less. Refer to Paragraph 5./REPAIR 2 and Figure 204/REPAIR 2.

NOTE: You can use Alternative 3 instead of Alternative 2 if you replace the initial 113A9380-1 rub strip with a 113A9380-2 rub strip.

(3) Alternative 3: Use this repair for flap support fairings that have a 113A9380-2 rub strip, and if the damaged fastener hole size is 0.50 in. (12.7 mm) or less. Refer to Paragraph 6./REPAIR 2 and Figure 205/REPAIR 2.

C. All dimensions are in inches (mm), unless they are shown differently.

D. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01, GENERAL.

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2315487 S0000525844_V1

Numbers 3 and 6 Inboard Trailing Edge Flap Support Fairing - Location of Damage
Figure 201

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-20-01, GENERAL	Protective Treatment of Metallic and Composite Materials
51-20-05, GENERAL	Repair Sealing
51-40-02, GENERAL	Fastener Installation and Removal
51-70-04, REPAIR 9	Repair of Damage to a Fastener Hole With Laminating Resin and Chopped Fiber
51-70-05, REPAIR 9	Repair of Damage That is 0.50 Inch (12.70 mm) or Less to Solid Laminates or a Fastener Hole
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Repair Instructions for Alternative 1

A. Get access to the damaged area.

B. Remove the fastener from the damaged fastener hole, as necessary. Make sure the damaged fastener hole diameter is 0.325 in. (8.255 mm) or less.

C. Install a BACB30NN3K( ) bolt with the head of the bolt on the inside of the flap fairing. As an alternative, you can use a BACB30XD3K( ) bolt. Refer to Figure 203/REPAIR 2.

D. Install a BACW10U-D( ) dimpled washer on the outside of the flap fairing over the damaged fastener hole. As an alternative, you can use a BACW10U-C( ) dimpled washer. Refer to Figure 203/REPAIR 2.

E. Install a NAS1149D0416J washer over the dimpled washer. Refer to Figure 203/REPAIR 2.

F. Install a BACN10JC3C nut. As an alternative, you can use a BACN10JC3CD nut.

5. Repair Instructions for Alternative 2

A. Get access to the damaged area.

B. Remove the five fasteners in the repair area. Refer to 51-40-02, GENERAL.

C. Remove the 113A9380-1 rub strip and the 113A9350-11 seal. Keep these parts to be installed in Paragraph 5.L./REPAIR 2.

NOTE: As an alternative to the 113A9380-1 rub strip, you can install the new 113A9380-2 rub strip. If you install the new 113A9380-2 rub strip, then you must do the repair as given in Paragraph 6./REPAIR 2.

D. Increase the fastener hole diameters on the 113A9380-1 rub strip to 0.210 in. (5.334 mm) - 0.215 in. (5.461 mm).

E. Do the repair on all of the five fairing holes as given in 51-70-04, REPAIR 9 or 51-70-05, REPAIR 9 with the exceptions as follow:

(1) Use a single repair ply (BMS 9-3 H2 or H3 glass fabric) on each side. The repair ply must extend 0.25 in. (6.35 mm) minimum past the Part 1 Strip in all directions. Trim the repair ply flush with the upper edge of the flap support fairing.

(2) Do not countersink the fastener holes.

F. Make the Part 1 Strip. Refer to Table 201/REPAIR 2 and Figure 204/REPAIR 2.

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Table 201: Repair Material for Alternative 2

PART		QUANTITY	MATERIAL
1	Strip	1	0.063 in. (1.60 mm) thick 2024-T3 clad sheet as given in QQ-A-250/5 or 2024-T3 bare sheet as given in QQ-A-250/4

G. Remove the nicks, scratches, gouges, burrs, and sharp edges from the Part 1 Strip.

H. Apply a chemical conversion coating to the Part 1 Strip. Refer to 51-20-01, GENERAL.

I. Apply one layer of BMS 10-11, Type I or BMS 10-79, Type II primer to the Part 1 Strip. Refer to SOPM 20-41-02 for BMS 10-11 primer and SOPM 20-44-04 for BMS 10-79 primer.

J. Install the Part 1 Strip.

(1) Bond the Part 1 Strip to the fairing with BMS 5-95 sealant. It is recommended that you use the vacuum bag method.

(2) Make sure the forward edge margin of the Part 1 Strip is 0.63 in. (16.00 mm).

K. Drill the fastener holes in the Part 1 Strip.

L. Reinstall the 113A9380-1 rub strip and the 113A9350-11 seal.

M. Install the fasteners.

N. Apply a fillet seal with BMS 5-95 sealant along the edges of the Part 1 Strip. Refer to 51-20-05, GENERAL.

O. Restore the finishes.

6. Repair Instructions for Alternative 3

A. Get access to the damaged area.

B. Remove the five fasteners in the repair area. Refer to 51-40-02, GENERAL.

C. Remove the 113A9380-2 rub strip and the 113A9350-11 seal.

D. Make sure the fastener hole diameters on the 113A9380-2 rub strip are 0.210 in. (5.334 mm) - 0.215 in. (5.461 mm). If the fastener hole diameters are smaller, increase them to 0.210 in. (5.334 mm) - 0.215 in. (5.461 mm).

E. Do the repair on all of the five fairing holes as given in 51-70-04, REPAIR 9 or 51-70-05, REPAIR 9 with the exceptions as follow:

(1) Use a single repair ply (BMS 9-3 H2 or H3 glass fabric) on each side. The repair ply must extend 0.25 in. (6.35 mm) minimum past the Part 1 Strip in all directions. Trim the repair ply flush with the upper edge of the flap support fairing.

(2) Do not countersink the fastener holes.

F. Make the Part 1 Strip. Refer to Table 202/REPAIR 2 and Figure 205/REPAIR 2.

Table 202: Repair Material for Alternative 3

PART		QUANTITY	MATERIAL
1	Strip	1	0.063 in. (1.60 mm) thick 2024-T3 clad sheet as given in QQ-A-250/5 or 2024-T3 bare sheet as given in QQ-A-250/4

G. Remove the nicks, scratches, gouges, burrs, and sharp edges from the Part 1 Strip.

H. Apply a chemical conversion coating to the Part 1 Strip. Refer to 51-20-01, GENERAL.

I. Apply one layer of BMS 10-11, Type I or BMS 10-79, Type II primer to the Part 1 Strip. Refer to SOPM 20-41-02 for BMS 10-11 primer and SOPM 20-44-04 for BMS 10-79 primer.

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J. Install the Part 1 Strip.

(1) Bond the Part 1 Strip to the fairing with BMS 5-95 sealant. It is recommended that you use the vacuum bag method.

(2) Make sure the forward edge margin of the Part 1 Strip is 0.63 in. (16.00 mm).

K. Drill the fastener holes in the Part 1 Strip.

L. Reinstall the 113A9380-2 rub strip and the 113A9350-11 seal.

M. Install the fasteners and washers.

N. Apply a fillet seal with BMS 5-95 sealant along the edges of the Part 1 Strip. Refer to 51-20-05, GENERAL.

O. Restore the finishes.

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FASTENER SYMBOLS

INITIAL FASTENER LOCATION. INSTALL A BACB30NN3K() BOLT WITH A BACN10JC3 NUT. AS AN ALTERNATIVE, INSTALL A BACB30XD3K() BOLT WITH A BACN10JC3CD NUT.

INITIAL FASTENER LOCATION. INSTALL A BACR15CE6D() RIVET.

INITIAL FASTENER LOCATION. INSTALL A BACR15FT6D()C RIVET AND A BACW10P216S WASHER.

2315492 S0000526435_V1

Fastener Symbols
Figure 202

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2315552 S0000526436_V1

Numbers 3 and 6 Inboard Trailing Edge Flap Support Fairing at the Seal Attachment Fastener Holes Repair - Alternative 1
Figure 203

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(SEAL AND RUB STRIP NOT SHOWN FOR CLARITY)
(VIEW IN THE INBOARD DIRECTION)
PART 1 STRIP PROFILE

A-A

NOTES

1 MAKE SURE THAT THE CENTERLINE OF THE PART 1 STRIP IS ON THE CENTERLINE OF THE ROW OF THE FIVE FASTENER HOLES AS SHOWN.

2 MAKE SURE THAT THE PART 1 STRIP IS A MINIMUM OF 0.10 INCH (2.54 mm) BELOW THE EDGE OF THE FLAP SUPPORT FAIRING.

2321634 S0000526437_V1

Numbers 3 and 6 Inboard Trailing Edge Flap Support Fairing at the Seal Attachment Fastener Holes Repair - Alternative 2
Figure 204

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(SEAL AND RUB STRIP NOT SHOWN FOR CLARITY)
(VIEW IN THE INBOARD DIRECTION)
PART 1 STRIP PROFILE

A-A

NOTES

1 MAKE SURE THAT THE CENTERLINE OF THE PART 1 STRIP IS ON THE CENTERLINE OF THE ROW OF THE FIVE FASTENER HOLES AS SHOWN.

2 MAKE SURE THAT THE PART 1 STRIP IS A MINIMUM OF 0.10 INCH (2.54 mm) BELOW THE EDGE OF THE FLAP SUPPORT FAIRING.

2321618 S0000526438_V1

Numbers 3 and 6 Inboard Trailing Edge Flap Support Fairing at the Seal Attachment Fastener Holes Repair - Alternative 3
Figure 205

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REPAIR 3 - FLAP SUPPORT FORWARD FAIRING SKIN, WING TRAILING EDGE FLAP

1. Applicability

A. Repair 3 is applicable to damage to the skin panels of the flap support forward fairing shown in Flap Support Forward Fairing Location for the Wing Trailing Edge Flap, Figure 201/REPAIR 3.

B. Repair 3 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

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2430467 S0000562025_V1

Flap Support Forward Fairing Location for the Wing Trailing Edge Flap
Figure 201

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2. General

A. Repair 3 gives instructions for Permanent repairs for the number 1 and 2 (7 and 8 opposite) outboard flap forward fairing skins and the number 3 (6 opposite) inboard flap forward fairing skins. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

B. Remove the skin as necessary to get access to the inner surface.

(1) Remove the fasteners. Refer to 51-40-02, GENERAL for information on fastener removal.

(2) If the fastener hole is damaged, refer to Paragraph 4.F./REPAIR 3 for a repair.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Nondestructive Inspection (NDI) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 203/REPAIR 3 for the definitions of the diameter and depth of the damage.

(2) Refer to Definitions of the Facesheets, Figure 204/REPAIR 3 for the definitions of the facesheets of a honeycomb core area.

D. Do the repair as given in Paragraph 4./REPAIR 3.

E. Install the skin, as necessary.

F. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01, GENERAL.

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(LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE)
FLAP FAIRING NUMBER 3 IS SHOWN,
FLAP SUPPORT FAIRING NUMBERS 1 AND 2 ARE SIMILAR

2430469 S0000562022_V1

Flap Support Forward Fairing Numbers 1 and 2 for the Outboard Trailing Edge Flap and Number 3 for the Inboard Trailing Edge Flap
Figure 202

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

DEFINITIONS FOR EDGE DAMAGE

DEFINITIONS FOR DENT DAMAGE

DEFINITIONS FOR DENT DAMAGE

2430471 S0000562023_V1

Damage Definitions
Figure 203

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2430472 S0000562024_V1

Definitions of the Facesheets
Figure 204

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-70-04, REPAIR 10	Repair of Damage That is More Than 0.50 Inch in Solid Laminates
51-70-04, REPAIR 14	Rework of Permitted Surface Dents with Potting Compound and a Fiberglass Patch
51-70-04, REPAIR 9	Repair of Damage to a Fastener Hole With Laminating Resin and Chopped Fiber
51-70-05, REPAIR 10	Repair of Damage That is More Than 0.50 Inch (12.70 mm) in Diameter to Solid Laminates
51-70-05, REPAIR 8	Repair of Damage to a Fastener Hole With Prepreg Materials
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Repair Instructions

A. Refer to:

(1) Table 201/REPAIR 3 for Zone 1 honeycomb core areas.

(2) Table 202/REPAIR 3 for Zone 1 solid laminate areas.

B. Contact The Boeing Company for:

(1) A solid laminate repair in Zone 2 not located at a fastener hole.

(2) A repair made with:

(a) Room temperature wet layup materials.

(b) Wet layup materials at 150 degrees F (66 degrees C) cure.

C. Use the repair instructions that follow for a repair made with wet layup materials at 200 degrees F (93 degrees C) cure.

(1) Repair the damage as given in 51-70-04 with the requirements that follow:

(a) For structural plies:

1) Use only BMS 9-3, Type H-2 or H-3 for all structural plies.

NOTE: Use of Type D or D-1 is not permitted for structural plies.

2) Use the same number of structural repair plies as the number of initial plies that are removed.

3) For each structural ply removed, orient the replacement ply in the original ply orientation.

4) Make sure that you do not remove more than (3) initial plies.

(b) For extra repair plies:

1) Add two extra structural repair plies. Put one structural ply at +/- 45 degrees to the core ribbon direction and the other at 0 or 90 degrees.

(c) For non-structural sanding plies:

1) If damage is on the external surface of the fairing, add one non-structural sanding ply (direction optional). May use BMS 9-3, Type D or D-1 for non-structural sanding ply. A sanding ply is necessary only on the external surface of the fairing.

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D. Use the repair instructions that follow for a repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by vacuum bag procedure (oven cure or heat blanket cure).

(1) Repair the damage as given in 51-70-05 with the requirements that follow:

(a) For structural plies:

1) Use the same number of structural repair plies as the number of initial structural plies that are removed.

2) For each structural ply removed, replace with the same ply material and orientation

3) No material substitution is permitted.

(b) For Filler plies:

1) Use either Style 120 or Style 1581 plies as necessary to achieve solid laminate thickness of 0.110-0.090 inch (2.79-2.29 mm).

2) Thickness calculation does not include the extra plies.

(c) For extra repair plies:

1) Add one extra Style 1581 (optional Style 7781) structural repair ply at 0 or 90 degrees to the core ribbon.

2) If the damage goes through the core or filler plies, add one extra Style 1581 (optional Style 7781) structural repair ply at 0 or 90 degrees to the core ribbon so that there is an extra ply on each side of the panel.

3) No material substitution is permitted.

(d) For non-structural sanding plies:

1) If an extra repair ply is on the external surface of the fairing, add one non-structural sanding ply (direction optional). A sanding ply is only necessary on the external surface of the fairing.

E. Use the repair instructions that follow for a repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by autoclave procedure.

(1) Repair the damage as given in 51-70-05, with the requirements that follow:

(a) For a repair in the honeycomb core panel area, use a 50 to 1 taper ratio only with the appropriate overlap for the repair plies.

(b) For structural plies:

1) Use the same number of structural repair plies as the number of initial structural plies that are removed.

2) For each structural ply removed, replace with the same ply material and orientation.

3) No material substitution is permitted.

(c) For filler plies:

1) Use either Style 120 or Style 1581 plies as necessary to achieve solid laminate thickness 0.1100-0.0900 inch (2.79-2.29 mm).

2) Thickness calculation excludes the extra plies.

(d) For non-structural sanding plies:

1) If damage is on the external surface of the fairing, add one non-structural ply (direction optional). A sanding ply is only permitted on the external surface of the fairing.

F. If the fastener hole is damaged,

(1) If a bulb seal fastener hole (located in Zone #1) is damaged,

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(a) For D less than or equal to 0.261 inch diameter: Determine if the damage is removed when drilling oversized hole including the countersink.

1) Remove BACB30XD3K() bolt, NAS1149E0332P washer, and BACN10YR3CD nut at the damaged fastener hole.

2) Drill 0.261-0.250 inch diameter hole through the fairing, removing the hole damage.

3) Enlarge the seal retainer and seal slotted holes as given in Figure 205, Detail A.

4) Install a BACB30VF4K() bolt, NAS1149E0432P washer (under nut), and BACN10YR4CD nut in the repaired fastener hole as given in 51-40-02- thru 51-40-04.

(b) For D less than or equal to 0.50 inch diameter: Perform a chopped fiber repair as given in 51-70-04, REPAIR 9 with a dimpled washer bonded with BMS 5-95 on the external side of the fairing.

1) If the repair diameter D is less than or equal to 0.262 inch diameter: Bond a BACW10U-C10P (optional BACW10U-C10) dimpled washer to external side of the fairing.

2) For a repair where D is greater than 0.262 inch diameter: Fabricate a dimpled washer as given in BACW10U-C10P (optional BACW10U-C10 or BACW10U-D10) but with a diameter (dimension C in specification BACW10U) of +1/16 inch oversized from the repair diameter. Bond dimpled washer on the external side of the fairing.

NOTE: If there are multiple fastener repairs in a row, it is acceptable to replace the dimpled washer with a strap with dimpled countersinks. Use the washer specification for the strap dimensions. Bond strap with BMS 5-95 (vacuum bagging is the preferred method). See Figure 205, Detail B for dimpled strap definition.

(c) For D greater than 0.50 inch diameter: Perform a scarf repair as given in 51-70-04, REPAIR 10 or 51-70-05, REPAIR 10.

1) For a scarf repair made with layup materials at 200 degrees F (93 degrees C) cure, use Paragraph 4.C.(1)(b)/REPAIR 3. for the number of extra repair plies.

2) For a scarf repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by vacuum bag procedure (oven cure or heat blanket cure), use Paragraph 4.D.(1)(c)/REPAIR 3 for the number of extra repair plies.

3) For a scarf repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by autoclave procedure, no extra repair plies are necessary.

(2) If a support frame electrically bonded fastener hole (located in Zone #1) is damaged:

(a) For D less than or equal to 0.253 inch diameter: Determine if the damage is removed when drilling oversized holes including the countersink.

1) Remove the BACB30LH3-4 bolt, BACJ40A20-6 bonding jumper, NAS1149E0332P washer, and BACN10YR3CD nut that aligns with the damaged hole.

2) Match drill 0.253-0.250 inch diameter hole through the fairing and support frame.

3) Install a BACB30ZE4-4 bolt, BACJ40A21-6 bonding jumper, NAS1149E0432P washer and BACN10YR4CD nut in the repaired fastener hole as given in 51-40-02 thru 51-40-04. On the external side of the fairing, install the fastener on BMS 10-21, Type III (anti static coating) surface as given in BAC 5117-13, except do not apply coating to the aluminum structure. Maximum resistance as given in BAC 5117-13. Refinish composite surface as given in BMS 10-103, Type I primer and as given in BAC 5639 for the external side of the fairing.

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4) On the aluminum frame side of the fairing, install 3/16 inch bolt, bonding jumper, washer, and nut in aluminum structure as given in BAC 5117-4 category 2 (fay sealed). Seal for non-fuel tank areas. Prepare structure surface with cleaning method CM1. Refinish metal surface with manually applied chemical conversion coating in accordance with BAC5719, Type II, Class A plus apply one coat of BMS10-79, Type III primer in accordance with BAC5882 plus apply BMS10-60, Type II, gloss enamel in accordance with BAC5845 (Boeing color BAC 707 gray). Do not apply sealant to the head or shank of the fastener. Install with BMS 5-95 sealant as given in REPAIR SEALING, 51-20-05 over the bonding jumper terminal, washer, and nut. Install red terminal on this fastener.

(b) For D less than or equal to 0.257 inch diameter: Perform hole repair as given in 51-70-04, REPAIR 9 except:

1) Use BMS 5-28, Type 6 potting in place of laminating resin and chopped fiber.

2) Do not taper sand the repair.

3) Install dimpled washer (BACW10U-C10P) on the external side of the fairing and reinstall the fastener. On the external side of the fairing, install the fastener on BMS 10-21, Type III (anti static coating) surface as given in BAC 5117-13, except do not apply coating to the aluminum structure. Maximum resistance as given in BAC 5117-13. Refinish composite surface as given in BMS 10-103, Type I primer and as given in BAC 5639 for the external side of the fairing.

4) On the aluminum frame side of the fairing, install 1/4 inch bolt, bonding jumper, washer, and nut in aluminum structure as given in BAC 5117-4 category 2 (fay sealed). Seal for non-fuel tank areas. Prepare structure surface with cleaning method CM1. Refinish metal surface with manually applied chemical conversion coating in accordance with BAC5719, Type II, Class A plus apply one coat of BMS10-79, Type III primer in accordance with BAC5882 plus apply BMS10-60, Type II, gloss enamel in accordance with BAC5845 (Boeing color BAC 707 gray). Do not apply sealant to the head or shank of the fastener. Install with BMS 5-95 sealant as given in REPAIR SEALING, 51-20-05 over the bonding jumper terminal, washer, and nut. Install red terminal on this fastener.

(c) For D greater than 0.257 inch diameter: Perform a scarf repair as given in 51-70-04, REPAIR 10 or 51-70-05, REPAIR 8. Maximum permitted repair diameter is 0.312 inch. Bond a dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS 5-95 on the external side.

1) For a scarf repair made with layup materials at 200 degrees F (93 degrees C), use Paragraph 4.C.(1)(b)/REPAIR 3 for the number of extra repair plies.

2) For a scarf repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by vacuum bag procedure (oven cure or heat blanket cure), use Paragraph 4.D.(1)(c)/REPAIR 3 for the number of extras repair plies.

3) For a scarf repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by autoclave procedure, no extra repair plies are necessary.

(d) For D greater than 0.312 inch diameter: Contact The Boeing Company.

(3) If a support frame rivet hole (located in Zone #1) is damaged:

(a) For D less than or equal to 0.265 inch diameter: Determine if the damage is removed when drilling oversized hole including the countersink

1) Match drill 0.265-0.255 inch diameter hole through the fairing and support frame.

2) Install a BACR15GF8D() or BACR15CE8AD() rivet in the repaired fastener hole as given in 51-40-02 thru 51-40-03.

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(b) For D less than or equal to 0.265 inch diameter: Perform the hole repair as given in 51-70-04, REPAIR 9 except:

1) Use BMS 5-28, Type 6 potting in place of laminating resin and chopped fiber.

2) Do not taper sand the repair.

3) Bond a dimpled washer (BACW10U-C10P or BACW10U-C10) on the external side with BMS 5-95.

(c) For D greater than 0.265 inch: Perform a scarf repair as given in 51-70-04, REPAIR 10 or 51-70-05, REPAIR 8. Maximum permitted repair diameter 0.312 inch. Bond a dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS 5-95 on the external side.

1) For a scarf repair made with layup materials at 200 degrees F (93 degrees C) cure, use Paragraph 4.C.(1)(b)/REPAIR 3 for the number of extra repair plies.

2) For a scarf repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by vacuum bag procedure (oven cure or heat blanket cure), use Paragraph 4.D.(1)(c)/REPAIR 3 for the number of extra repair plies.

3) For a scarf repair made with preimpregnated layup materials at 250 degrees F (121 degrees C) cure by autoclave procedure, no extra repair plies are necessary.

(d) For D greater than 0.312 inch in diameter: Contact The Boeing Company.

(4) If a forward fitting rivet hole (located in Zone #2) is damaged:

(a) For D less than or equal to 0.265 inch in diameter: Determine of the damage is removed when drilling oversized hole including the countersink.

1) Match drill 0.265-0.254 inch diameter hole through the fairing and mount block

2) Install a BACR15BB8AD()C rivet in the repaired fastener hole as given in 51-40-02 thru 51-40-03.

(b) For damage a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch (10.033 mm):

1) Make sure all damage is removed.

2) Fill and fair with BMS 5-28, Type 6 using the 51-70-04, REPAIR 14 repair procedures.

3) No overlay ply is necessary.

4) Bond dimpled washer (BACW10U-C10 or BACW10U-C10P) with BMS 5-95 on the external side.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

(c) For D less than or equal to 0.325 inch in diameter: Perform a hole repair as given in 51-70-04, REPAIR 9 except:

1) Use BMS 5-28, Type 6 potting in place of laminating resin and chopped fiber.

2) Do not taper sand the repair.

3) Fabricate a dimpled washer as given in BACW10U-C10P (optional BACW10U-C10) but with a diameter (dimension C in specification BACW10U) of +1/16 oversized from the repair diameter. Bond dimpled washer with BMS 5-95 on the external side of the fairing.

(d) For D greater than 0.325 inch in diameter: Contact The Boeing Company.

(5) If a forward fitting bolt hole (located in Zone #2) is damaged:

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(a) For D less than or equal to 0.316 inch diameter: Perform the hole repair as given in 51-70-04, REPAIR 9 except:

1) Use BMS 5-28, Type 6 potting in place of laminating resin and chopped fiber.

2) Do not taper sand the repair.

3) Bond dimpled washer (BACW10U-C416 or BACW10U-C416P) on the external side with BMS 5-95.

(b) For D less than or equal to 0.316 inch in diameter:

1) Remove the BACN11G4A3CD nutplate on the forward support fitting that aligns with the damaged hole. Refer to 51-40-02 for information on fastener removal.

2) Match drill through the fairing hole to the mount block, shim (if present) and forward support fitting to a diameter of 0.316-0.312 inch.

3) Install new BACN11G5A3CD nutplate on the forward support fitting as given in 51-40-02.

4) Install a BACB30NN5K() bolt in the repaired fastener hole as given in SRM 51-40-02 thru 51-40-04. Apply BMS 10-11, Type I primer as given in BAC 5736 to all areas of the hole, including any countersink, counterbore, or other recess and allow to dry. Apply MIL-C-11796 or BMS 3-38 grease to all areas of the hole and then install the fastener.

(c) For D greater than 0.316 inch diameter: Contact The Boeing Company.

(d) Forward Fairing #1/#8 or #2/#7:

1) For damage a maximum depth of 3 plies or less and maximum repair diameter of 0.517 inch:

a) Make sure all damage is removed.

b) Fill and fair with BMS 5-28, Type 6 using the 51-70-04, REPAIR 14 repair procedures.

c) No overlay ply is necessary.

d) Bond dimpled washer (BACW10U-C416 or BACW10U-C416P) with BMS 5-95 on the external side.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

(e) Forward Fairing #3/#6:

1) For damage a maximum depth of 2 plies or less and maximum repair diameter 0.517 inch:

a) Make sure all damage is removed.

b) Fill and fair with BMS 5-28, Type 6 using the 51-70-04, REPAIR 14 repair procedures.

c) No overlay ply is necessary.

d) Bond dimpled washer (BACW10U-C416 or BACW10U-C416P) with BMS 5-95 on the external side.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

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Table 201:

REPAIR DATA FOR THE SUPPORT FORWARD FAIRING SKIN 250°F (121°C) CURE, HONEYCOMB CORE PANEL AREA			
REPAIR TYPE	**PERMANENT WET LAYUP**	**PERMANENT PREIMPREGNATED LAYUP, VACUUM BAG**	**PERMANENT PREIMPREGNATED LAYUP AUTOCLAVE**
REPAIR CURE TEMPERATURE	**200°F (93°C)**	**250°F (121°C)**	**250°F (121°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 3.0 inches in diameter One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - panel edges - fastener holes	Damage that is a maximum of - 12 inches in diameter 3.0 inches minimum clearance from: - other repairs - panel edges - fastener holes	There are no limits on the size of the repair.
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.C.	SRM 51-70-05 and Paragraph 4.D.	SRM 51-70-05 and Paragraph 4.E.

Table 202:

REPAIR DATA FOR THE FLAP SUPPORT FORWARD FAIRING SKIN 250°F (121°C) CURE, LAMINATE PANEL AREA			
REPAIR TYPE	**PERMANENT WET LAYUP**	**PERMANENT PREIMPREGNATED LAYUP, VACUUM BAG**	**PERMANENT PREIMPREGNATED LAYUP, AUTOCLAVE**
REPAIR CURE TEMPERATURE	**200°F (93°C)**	**250°F (121°C)**	**250°F (121°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 3.0 inches in diameter One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs	Damage that is a maximum of: - 12.0 inches in diameter 3.0 inches minimum clearance from: - other repairs	There are no limits on the size of the repair.
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.C.	SRM 51-70-05 and Paragraph 4.D.	SRM 51-70-05 and Paragraph 4.E.

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BULB SEAL AND SEAL RETAINER SLOTTED HOLE MODIFICATION

A

NOTE: THE DIMENSIONS ARE DEFINED IN PROCESS SPECIFICATION BACW10U

DIMPLED STRAP

B

2430473 S0000562222_V1

Fastener Hole Repair
Figure 205

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REPAIR 4 - WING INBOARD TRAILING EDGE FLAP SUPPORT AFT FAIRING SKIN

1. Applicability

A. Repair 4 is applicable to the skin panel of the flap support aft fairing shown in Location for the Wing Inboard Trailing Edge Flap Support Fairing, Figure 201/REPAIR 4.

B. Repair 4 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

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2455989 S0000571194_V1

Location for the Wing Inboard Trailing Edge Flap Support Fairing
Figure 201

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2. General

A. Repair 4 gives instructions for Permanent repairs for the number 3 (6 opposite) inboard flap aft fairing skins. Refer to 51-00-06, GENERAL to find the definitions of the different categories of repairs.

B. Remove the skin as necessary to get access to the inner surface.

(1) Remove the fasteners. Refer to 51-40-02, GENERAL for information on fastener removal.

(2) If the fastener hole is damaged, refer to Paragraph 4.E./REPAIR 4 for a repair.

C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Nondestructive Inspection (NDI) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 203/REPAIR 4 for the definitions of the diameter and depth of the damage.

(2) Refer to Definitions of the Facesheets, Figure 204/REPAIR 4 for the definitions of the facesheets of a honeycomb core area.

D. Do the repair as given in Paragraph 4./REPAIR 4.

E. Install the skin, as necessary.

F. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01, GENERAL.

2456173 S0000571195_V1

Inboard Trailing Edge Flap Support Fairing Number 3
Figure 202

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2456390 S0000571196_V1

Tailcone Fairing Skin Location
Figure 203

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

DEFINITIONS FOR EDGE DAMAGE

DEFINITIONS FOR DENT DAMAGE (SOLID LAMINATE)

DEFINITIONS FOR DENT DAMAGE (CORE AREA)

2456185 S0000571197_V1

Damage Definitions
Figure 204

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2456189 S0000571198_V1

Definitions of the Facesheets
Figure 205

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-10-01, GENERAL	Aerodynamic Smoothness Requirements
51-40-02, GENERAL	Fastener Installation and Removal
51-40-03, GENERAL	Fastener Substitution
51-40-04, GENERAL	Torque Values
51-70-04, REPAIR 14	Rework of Permitted Surface Dents with Potting Compound and a Fiberglass Patch
51-70-04, REPAIR 9	Repair of Damage to a Fastener Hole With Laminating Resin and Chopped Fiber
51-70-04, REPAIR GENERAL	Repair Procedures for Wet Layup Materials
51-70-05, REPAIR GENERAL	Repair Procedures for Pre-impregnated Materials
BAC 5000	General Sealing
BAC 5117-13	ELECTRICAL BONDING OF FASTENERS TO CONDUCTIVE FINISHES ON COMPOSITES
BAC 5117-6	Electrical Faying Surface Bonds
BAC 5719	CHEMICAL CONVERSION COATINGS FOR ALUMINUM AND ALUMINUM ALLOYS
BAC 5845	Application of Polyurethane Enamel
SOPM 20-11-03	Repair of Electrical Terminations and Electrical Bonding Areas
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Repair Instructions

A. Refer to:

(1) Table 201/REPAIR 4 for honeycomb core areas.

(2) Table 202/REPAIR 4 for solid laminate areas.

B. Contact The Boeing Company for:

(1) A repair made with:

(a) Room temperature wet layup materials.

(b) Wet layup materials at 150 degrees F (66 degrees C) cure.

(c) Preimpregnated layup materials at 350 degrees F (177 degrees C) by vacuum bag procedure.

(d) Preimpregnated layup materials at 350 degrees F (177 degrees C) by autoclave cure procedure (no postcure).

C. Use the repair instructions that follow for a repair made with wet layup materials at 200°F (93°C) cure.

(1) Repair the damage as given in 51-70-04, REPAIR GENERAL, with the requirements that follow:

(a) For structural plies:

1) Use only BMS9-3 Type H-2 or H-3 for all structural plies.

NOTE: Use of Type D or D-1 is not acceptable for structural plies.

2) Use the same number of structural repair plies as the number of initial plies that are removed.

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3) For each structural ply removed, orient the replacement ply in the original ply orientation.

4) Make sure that no more than (3) initial plies are removed.

(b) For extra repair plies:

1) For the aft fairing and tailcone fairing solid laminate area: Add two extra structural repair plies at ±45 degrees to the material direction.

2) For the aft fairing and tailcone fairing honeycomb core area: Add one extra structural repair ply at ±45 degrees to the core ribbon direction.

3) For the aft fairing tailcap: Add three extra structural repair plies. Put two structural plies at ±45 degrees to the material direction and one at 0 or 90 degrees.

(c) For non-structural sanding plies:

1) If damage is on the external surface of fairing, add one non-structural sanding ply (direction optional). May use BMS9-3 Type D or D-1 for non-structural sanding ply. A sanding ply is required only on the external surface of fairing.

D. Use the repair instructions that follow for a repair made with preimpregnated layup materials at 350°F (177°C) cure by autoclave procedure with a 400°F (204°C) postcure.

NOTE: The post cure shall be done with the part under uniform restraint. The post cure may be done either in an autoclave or in an air circulating oven. Raise the part temperature from ambient temperature to 400°F±10°F at a rate not to exceed 7°F per minute and hold for 120 minutes to 180 minutes. Cool below 125°F (52°C) before releasing restraints. Maximum cool down rate is 5°F (3°C) per minute. The post cure may be performed after the completion of the 120 minute hold at 355°F±10°F (Reference 51-70-04 Figure 211) under continued autoclave pressure.

(1) If there is a 200°F (93°C) initial repair in the repair area, the initial repair must be replaced with this 350°F (177°C) autoclave repair.

(2) Repair the damage as given in 51-70-05, REPAIR GENERAL, with the requirements that follow:

(a) For repairs in the honeycomb core panel area, use a 50 to 1 taper ratio only with the appropriate overlap for the repair plies.

(b) For structural plies:

1) Use the same number of structural repair plies as the number of initial structural plies that are removed.

2) For each structural ply removed, replace with the same ply material and orientation.

3) No material substitution is permitted.

(c) For filler plies:

1) Use either Style 120 or Style 1581 plies as required to achieve solid laminate thickness as defined by the production drawing.

2) Thickness calculation excludes the extra plies

(d) For extra repair plies:

1) For the aft fairing:

a) Add one extra Style 120 (optional Style 220) structural repair ply at ±45 degrees to the material direction.

b) If the damage goes through the core or filler plies, add one extra Style 120 (optional Style 220) structural repair ply at ±45 degrees to the material direction so that there is an extra ply on each side of the panel.

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2) For the aft fairing tailcap:

a) Add one extra Style 120 (optional Style 220) structural repair ply at 0 or 90 degrees and one extra Style 1581 (optional Style 7781) structural repair ply at ±45 degrees to the material direction.

b) If the damage goes through the filler plies, add one extra Style 120 (optional Style 220) structural repair ply at 0 or 90 degrees and one extra Style 1581 (optional Style 7781) structural repair ply at ±45 degrees to the material direction so that there are two extra plies on each side of the panel.

3) For the tailcone fairing:

a) Add one extra Style 1581 (optional Style 7781) structural repair ply at 0 or 90 degrees to the material direction.

b) If the damage goes through the core or filler plies, add one extra Style 1581 (optional Style 7781) structural repair ply at 0 or 90 degrees to the material direction so that there is an extra ply on each side of the panel.

(e) For nonstructural sanding plies:

1) If damage is on the external surface of fairing, add one non-structural sanding ply (direction optional). A sanding ply is only required on external surface of fairing.

E. If the fastener hole is damaged:

(1) If a bulb seal fastener hole is damaged,

(a) If the fastener hole is common to the 113A9380 rub strip, refer to Repair 2.

(b) For D≤ 0.261 inch:

Determine if the damage is removed when drilling oversized hole including the countersink

1) Remove BACB30XD3K* bolt, NAS1149E0332P washer, and BACN10YR3CD nut at the damaged fastener hole.

2) Drill 0.261-0.250 inch diameter hole through the fairing, removing the hole damage

3) For the aft fairing bulb seals, enlarge the seal retainer and seal slotted holes per Figure 206, Detail A. For the tailcap bulb seal, enlarge the hole in the retainer and bulb seal to diameter of 0.341-0.316 inches.

4) Install a BACB30VF4K* bolt, NAS1149E0432P washer (under nut), and BACN10YR4CD nut in the repaired fastener hole as given in 51-40-02 through 51-40-04.

(c) For D ≤ 0.40 inch:

Perform a chopped fiber repair per 51-70-04 Repair 9 with a dimpled washer bonded with BMS5-95 on the external side of the fairing. The maximum repair diameter is limited to D = 0.40 inch.

1) If the repair diameter D≤ 0.262 inch, bond a BACW10U-C10P (optional BACW10U-C10) dimpled washer to external side of fairing.

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2) For a repair D>0.262 inch, fabricate a dimpled washer per BACW10U-C10P (optional BACW10U-C10 or BACW10U-D10) but with a diameter (dimension C in specification BACW10U) of +1/16 inch oversized from the repair diameter. Bond dimpled washer on the external side of fairing.

NOTE: If there are multiple fastener repairs in a row, it is acceptable to replace the dimpled washer with a strap with dimpled countersinks. Use the washer specification for the strap dimensions. Bond strap with BMS5-95 (vacuum bagging is the preferred method). See Figure 206, Detail B for dimpled strap definition.

3) Reinstall original fastener.

(d) D > 0.40 inch:

Perform a scarf repair as given in 51-70-04 Repair 10 or as given in 51-70-05 Repair 10.

1) For a scarf repair made with layup materials at 200°F (93°C) cure, use Paragraph 4.C.(1)(b)/REPAIR 4 for the number of extra repair plies. Apply extra repair plies on each side of panel.

2) For a scarf repair made with preimpregnated layup materials at 350°F (177°C) cure with 400°F (204°C) postcure by autoclave procedure, use Paragraph 4.D.(2)(d)/REPAIR 4 for the number of extra repair plies.

3) Reinstall original fastener.

(2) If a BACR15BA6AD*C rivet hole is damaged:

(a) D ≤ 0.265 inch:

Determine if the damage is removed when drilling oversized hole including the countersink.

1) Match drill 0.265-0.254 inch diameter hole through the fairing and mating structure.

2) Bond dimpled washer (BACW10U-C420P or BACW10U-C420) with BMS5-95 on the external side.

3) Install a BACR15GF8D* or BACR15CE8AD* rivet in the repaired fastener hole per 51-40-02 and 51-40-03.

(b) D ≤ 0.265 inch:

NOTE: The fastener repairs do not include the fasteners shown in Figure 207.

Perform hole repair per 51-70-04 Repair 9 except:

1) Use BMS 5-28 Type 6 potting in place of laminating resin and chopped fiber.

2) Do not taper sand the repair.

3) Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

4) Reinstall original fastener.

(c) For damage on the external surface a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch:

NOTE: The fastener repairs do not include the fasteners shown in Figure 207.

1) Make sure all damage is removed

2) Fill and fair with BMS 5-28 Type 6 using the 51-70-04 Repair 14 repair procedures.

3) No overlay ply is required

4) Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

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5) Reinstall original fastener.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent

(d) D > 0.265 inch: Contact The Boeing Company

(3) If a NAS1399MW6A*C blind rivet hole is damaged:

NOTE: For the electrically bonded rivets:

Install the fastener on BMS 10-21, Type III (anti static coating) surface as given in BAC 5117-13, except do not apply coating to the aluminum structure. Refer to SOPM 20-11-03. Fastener substitution not allowed at these locations. Dimpled washer must be cadmium plated.

(a) D ≤ 0.261 inch:

Determine if the damage is removed when drilling oversized hole including the countersink.

1) Match drill 0.261-0.256 inch diameter hole through the tailcone fairing and splice plate.
2) Bond dimpled washer (BACW10U-C420P) with BMS5-95 on the external side.
3) Install a CR4624CW-8-* or BACR15AN8-* rivet in the repaired fastener hole as given in 51-40-02, GENERAL and 51-40-03, GENERAL.

(b) D ≤ 0.259 inch:

Perform hole repair as given in 51-70-04, REPAIR 9 except:

1) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.
2) Do not taper sand the repair.
3) Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.
4) Reinstall original fastener.

(c) For damage on the external surface a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch:

1) Make sure all damage is removed.
2) Fill and fair with BMS5-28 Type 6 using the 51-70-04, REPAIR 14 procedures.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

3) Reinstall original fastener.

(d) D > 0.261 inch: Contact The Boeing Company

(4) If a BACB30LH3-* bolt hole is damaged:

NOTE: For the pivot frame electrically bonded bolts: On the external side of the fairing, install the fastener on BMS 10-21, Type III (anti static coating) surface per BAC 5117-13, except do not apply coating to the aluminum frame. Maximum resistance per BAC 5117-13. Refinish external composite surface with one coat of BMS 10-103, Type I primer per BAC5325 plus BMS10-60, Type II, gloss enamel per BAC5845. Do not apply primer or enamel over conductive coating in countersink.

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On the aluminum frame side of the fairing, install 3/16 inch bolt, bonding jumper, washer, and nut in aluminum frame as given in BAC 5117-6 category 6 (fillet specification sealed). Seal for non-fuel tank areas. Prepare aluminum fame surface with cleaning method CM1. Prepare steel bolt surface with cleaning method CM3. Maximum resistance per BAC 5117-6. Refinish aluminum frame surface with manually applied chemical conversion coating in accordance with BAC 5719, Type II, Class A plus apply one coat of BMS10-79, Type III primer in accordance with BAC5882 plus apply BMS10-60, Type I, gloss enamel in accordance with BAC 5845 (Boeing color BAC 707 gray). Install BMS 5-95 sealant over the bonding jumper terminal, washer, and nut per BAC 5000.

NOTE: For the electrically bonded bolts without bonding jumper: Install the fastener on BMS 10-21, Type III (anti static coating) surface as given in BAC 5117-13, except do not apply coating to the aluminum structure. Refer to SOPM 20-11-03. Fastener substitution not allowed at these locations. Dimpled washer must be cadmium plated.

(a) For electrically bonded pivot frame damaged hole in aft fairing:

1) D ≤ 0.254 inch:

Determine if the damage is removed when drilling oversized hole including the countersink.

a) Remove the BACB30LH3-* bolt, BACJ40A20-8 bonding jumper, NAS1149E0332P washer, and BACN10YR3CD nut that aligns with the damaged hole.

b) Match drill 0.254-0.250 inch diameter hole through the fairing and mating structure.

c) Bond dimpled washer (BACW10U-C420P) with BMS5-95 on the external side.

d) Install a BACB30ZE4-* bolt, BACJ40A21-8 bonding jumper, NAS1149E0432P washer and BACN10YR4CD nut in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

2) D ≤ 0.257 inch:

Perform hole repair as given in 51-70-04, REPAIR 9 except:

a) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.

b) Do not taper sand the repair.

c) Bond dimpled washer (BACW10U-C10P) with BMS5-95 on the external side.

d) Reinstall original fastener.

3) For damage on the external surface a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch:

a) Make sure all damage is removed.

b) Fill and fair with BMS5-28 Type 6 using the 51-70-04, REPAIR 14 repair procedures.

c) No overlay ply is required.

d) Bond dimpled washer (BACW10U-C10P) with BMS5-95 on the external side.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

e) Reinstall original fastener.

4) D > 0.257 inch: Contact The Boeing Company

(b) Damaged hole in tailcone fairing where bolt is mated to a nutplate:

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1) D ≤ 0.254 inch:

   Determine if the damage is removed when drilling oversized hole.

   a) Remove the BACN10KE3B3CD or BACN10KE3B4CD nutplate on the aft fairing that aligns with the damaged tailcone fairing hole.

   b) Match drill 0.254-0.250 inch diameter hole through the tailcone fairing and aft fairing. Do not countersink the holes.

   c) Install new BACN11G4A3CD or BACN11G4A4CD nutplate on the aft fairing.

   d) Install a fabricated countersink repair washer and a BACB30LK4K* or BACB30NT4K* bolt in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

      NOTE: At the electrically bonded fasteners, the repair washer should not be fabricated with aluminum. BMS10-21 coating is highly corrosive to aluminum.

      NOTE: It is optional to install a BACB30ZE4-* bolt without a countersink repair washer in the repaired fastener hole if a BACB30LK4K* or BACB30NT4K* bolt is unavailable. The hole in the tailcone must be countersunk. Bond dimpled washer (BACW10U-C420P or BACW10U-C420) with BMS5-95 on the external side.

2) D ≤ 0.257 inch:

   Perform hole repair as given in 51-70-04, REPAIR 9 except:

   a) Recommend to remove the BACN10KE3B3CD or BACN10KE3B4CD nutplate on the aft fairing that aligns with the damaged tailcone fairing hole. Install new BACN11G4A3CD or BACN11G4A4CD nutplate on the aft fairing.

   b) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.

   c) Do not taper sand the repair.

   d) Do not countersink the repaired fastener hole.

   e) Install a BACB31A3-* bolt in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

      NOTE: It is optional but not preferred to install a BACB30LH3-* bolt in the repaired fastener hole if a BACB31A3-* bolt is unavailable. The hole in the tailcone must be countersunk. Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

3) For damage a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch:

   a) Recommend to remove the BACN10KE3B3CD or BACN10KE3B4CD nutplate on the aft fairing that aligns with the damaged tailcone fairing hole. Install new BACN11G4A3CD or BACN11G4A4CD nutplate on the aft fairing.

   b) Make sure all damage is removed.

   c) Fill and fair with BMS5-28 Type 6 using the 51-70-04, REPAIR 14 repair procedures.

   d) No overlay ply is required.

   e) Do not countersink the repaired fastener hole.

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f) Install a BACB31A3-* bolt in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

NOTE: It is optional but not preferred to install a BACB30LH3-* bolt in the repaired fastener hole if a BACB31A3-* bolt is unavailable. The hole in the tailcone must be countersunk. Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

NOTE: It is optional to maintain the hole location by inserting a locating device

4) D > 0.257 inch: Contact The Boeing Company

(c) For damaged hole in aft fairing (including tailcap) where bolt is mated to a nutplate:

1) D ≤ 0.291 inch:

Determine if the damage is removed when drilling oversized hole.

a) Remove the BACN10KE3B3CD or BACN10KE3B4CD nutplate on the aft fairing that aligns with the damaged aft fairing hole.

b) Match drill 0.254-0.250 inch diameter hole through the tailcone fairing and aft fairing (or tailcap). Increase the hole size in the aft fairing (or tailcap) to 0.291-0.279 inch in diameter.

c) Install new BACN11G4A3CD or BACN11G4A4CD nutplate on the aft fairing.

d) Install a BACB30LK4K* or BACB30NT4K* bolt in the repaired fastener hole as given 51-40-02, GENERAL thru 51-40-04, GENERAL.

NOTE: It is optional but not preferred to install a BACB30ZE4-* bolt in the repaired fastener hole if a BACB30LK4K* or BACB30NT4K* bolt is unavailable

2) D ≤ 0.292 inch:

Perform hole repair as given in 51-70-04, REPAIR 9 except:

a) Remove the BACN10KE3B3CD or BACN10KE3B4CD nutplate on the aft fairing (or tailcap) that aligns with the damaged aft fairing hole.

b) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.

c) Do not taper sand the repair.

d) Install new BACN11G3A3CD or BACN11G434CD nutplate on the aft fairing (or tailcap).

e) Install a BACB31A3-* bolt in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

NOTE: It is optional but not preferred to install a BACB30LH3-* bolt in the repaired fastener hole if a BACB31A3-* bolt is unavailable. The hole in the tailcone must be countersunk. Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

3) D > 0.292 inch: Contact The Boeing Company

(d) For all other damaged holes:

1) D ≤ 0.254 inch:

Determine if the damage is removed when drilling oversized hole including the countersink.

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a) Remove the BACB30LH3-* bolt, NAS1149E0332P washer, and BACN10YR3CD nut that aligns with the damaged hole.

b) Match drill 0.254-0.250 inch diameter hole through the fairing and mating structure.

c) Bond dimpled washer (BACW10U-C420P or BACW10U-C420) with BMS5-95 on the external side.

d) Install a BACB30ZE4-* bolt, NAS1149E0432P washer and BACN10YR4CD nut in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

2) D ≤ 0.257 inch:

NOTE: The fastener repairs do not include the fasteners shown in Figure 207. Contact Boeing for a repair for these fasteners.

Perform hole repair as given in 51-70-04, REPAIR 9 except:

a) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.

b) Do not taper sand the repair.

c) Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

d) Reinstall original fastener.

3) For damage a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch:

NOTE: The fastener repairs do not include the fasteners shown in Figure 207. Contact Boeing for a repair for these fasteners.

a) Make sure all damage is removed.

b) Fill and fair with BMS5-28 Type 6 using the 51-70-04, REPAIR 14 repair procedures.

c) No overlay ply is required.

d) Bond dimpled washer (BACW10U-C10P or BACW10U-C10) with BMS5-95 on the external side.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

e) Reinstall original fastener.

4) D > 0.267 inch: Contact The Boeing Company

(5) If a BACB31A3-* bolt hole is damaged:

NOTE: For the electrically bonded bolts:

Install the fastener on BMS 10-21, Type III (anti static coating) surface as given in BAC 5117-13, except do not apply coating to the aluminum structure. Fastener substitution not allowed at these locations. Refer to SOPM 20-11-03.

(a) Damaged hole in tailcone fairing

1) D ≤ 0.254 inch:

Determine if the damage is removed when drilling oversized hole.

a) Remove the BACN11G3A3CD or BACN11G3A4CD nutplate on the aft fairing that aligns with the damaged tailcone fairing hole.

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b) Match drill 0.254-0.250 inch diameter hole through the tailcone fairing and aft fairing.

c) Install new BACN11G4A3CD or BACN11G4A4CD nutplate on the aft fairing.

d) Install a BACB30LK4K* or BACB30NT4K* bolt in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

2) D ≤ 0.257 inch:

Perform hole repair as given in 51-70-04, REPAIR 9 except:

a) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.

b) Do not taper sand the repair.

c) Reinstall original fastener.

3) For damage a maximum depth of 3 plies or less and maximum repair diameter 0.395 inch:

a) 1) Make sure all damage is removed.

b) Fill and fair with BMS5-28 Type 6 using the 51-70-04, REPAIR 14 repair procedures.

c) No overlay ply is required.

NOTE: It is optional to maintain the hole location by inserting a locating device coated with release agent.

d) Reinstall original fastener.

4) D > 0.257 inch: Contact The Boeing Company

(b) Damaged hole in aft fairing (including tailcap):

1) D ≤ 0.291 inch:

Determine if the damage is removed when drilling oversized hole.

a) Remove the BACN11G3A3CD or BACN11G3A4CD nutplate on the aft fairing that aligns with the damaged tailcone fairing hole.

b) Match drill 0.254-0.250 inch diameter hole through the aft fairing (or tailcap) and tailcone fairing. Increase the hole size in the aft fairing (or tailcap) to 0.291-0.279 inch diameter.

c) Install new BACN11G4A3CD or BACN11G4A4CD nutplate on the aft fairing (or tailcap).

d) Install a BACB31A4-* bolt in the repaired fastener hole as given in 51-40-02, GENERAL thru 51-40-04, GENERAL.

2) D ≤ 0.292 inch:

Perform hole repair as given in 51-70-04, REPAIR 9 except:

a) Use BMS5-28 Type 6 potting in place of laminating resin and chopped fiber.

b) Do not taper sand the repair.

c) Reinstall original fastener.

3) D > 0.292 inch: Contact The Boeing Company

(6) If a BACB30VF3K* bolt hole is damaged:

(a) D ≤ 0.254 inch:

Determine if the damage is removed when drilling oversized hole including the countersink.

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1) Match drill 0.254-0.250 inch diameter hole through the fairing and mating structure.
2) Bond dimpled washer (BACW10U-C420P or BACW10U-C420) with BMS5-95 on the external side.
3) Install a BACB30VF4K* bolt in the repaired fastener hole per 51-40-02 and 51-40-03.

(b) D> 0.254, contact The Boeing Company

5. Gap Check Instructions

A. Measure the gap between the tailcone and the aft flap track fairing assembly as given in Figure 208.

(1) If the gap is less than or equal to 0.015 inch around the entire edge band of the tailcone, then no fillers are required on the inner mold line of the tailcone.

B. If at any location around the edge band the gap is more than 0.015 inch, then bond solid fillers (002A0001-78) to the inner mold line of the tailcone, where required. Cut to fit. Bond using BMS 5-109 Type II, Class 2.

(1) Maximum permitted gap after installing filler is 0.015 inch. If the gap after filler installation is greater than 0.015 inch, contact The Boeing Company.

Table 201:

REPAIR DATA FOR THE INBD TE FLAP SUPPORT AFT AND TAILCONE FAIRING SKIN, HONEYCOMB CORE PANEL AREA				
REPAIR TYPE	**PERMANENT WET LAYUP**	**PERMANENT PREIMPREGNATED LAYUP, VACUUM BAG**	**PERMANENT PREIMPREGNATED LAYUP, AUTOCLAVE**	**PERMANENT PREIMPREGNATED LAYUP, AUTOCLAVE WITH POSTCURE**
REPAIR CURE TEMPERATURE	**200°F (93°C)**	**350°F (177°C)**	**350°F (177°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 3.0 inches in diameter One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - panel edges - fastener holes	- Contact The Boeing Company for this type repair	- Contact The Boeing Company for this type repair	There are no limits on the size of the repair.
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.C.	----	----	SRM 51-70-05 and Paragraph 4.D.

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Table 202:

REPAIR DATA FOR THE INBD TE FLAP SUPPORT AFT AND TAILCONE FAIRING SKIN, SOLID LAMINATE PANEL AREA				
REPAIR TYPE	**PERMANENT WET LAYUP**	**PERMANENT PREIMPREGNATED LAYUP, VACUUM BAG**		**PERMANENT PREIMPREGNATED LAYUP, AUTOCLAVE**
REPAIR CURE TEMPERATURE	**200°F (93°C)**	**350°F (177°C)**	**350°F (177°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 3.0 inches in diameter One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs	- Contact The Boeing Company for this type repair	- Contact The Boeing Company for this type repair	There are no limits on the size of the repair.
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.C.	.----	.----	SRM 51-70-05 and Paragraph 4.D.

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BULB AND SEAL RETAINER SLOTTED HOLE MODIFICATION

A

NOTE: THE DIMENSIONS ARE DEFINES IN PROCESS SPECIFICATION BACW10U

DIMPLED STRAP

B

2456195 S0000571489_V1

Bulb Seal Fastener Hole Repair
Figure 206

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NOTES

1 FOR FASTENER REPAIRS (EXCLUDING THE OVERSIZE REPAIR), CONTACT BOEING.

2456200 S0000571491_V1

Aft Fairing Aft Bulkhead Fasteners
Figure 207

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2456246 S0000571496_V1

Tailcone Gap Measurement
Figure 208

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IDENTIFICATION 1 - FLAP SUPPORT FAIRINGS STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G67892 S0006595077_V1

Flap Support Fairing Location for the Wing Trailing Edge Flap
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
001A0101	Final Assembly - Product Collector
113A0001	Movable T.E. Functional Product Collector
113A9100	Fairing Installation - Flap Support No. 1 and 8, Outbd T.E. Flap
113A9104	Attach Beam Assembly - Aft Fairing, Flap Support No. 1
113A9105	Adjust Link Assembly - Aft Fairing, Flap Support No. 1
113A9110	FWD Fairing Assembly - Flap Support No. 1, Outbd T.E. Flap
113A9150	Aft Fairing Assembly - Flap Support No. 1, Outbd T.E. Flap

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A9155	Pivot Frame Assembly - Aft Fairing, Flap Support No. 1
113A9160	Aft Bulkhead Assembly - Aft Fairing, Flap Support No. 1
113A9200	Fairing Installation - Flap Support No. 2 and 7, Outbd T.E. Flap
113A9204	Attach Beam Assembly - Aft Fairing, Flap Support No. 2
113A9216	Attach Link Assembly - Aft Fairing, Flap Support No. 2
113A9205	Adjust Link Assembly - Aft Fairing, Flap Support No. 2
113A9210	FWD Fairing Assembly - Flap Support No. 2, Outbd T.E. Flap
113A9250	Aft Fairing Assembly - Flap Support No. 2, Outbd T.E. Flap
113A9255	Pivot Frame Assembly - Aft Fairing, Flap Support No. 2
113A9260	Aft Bulkhead Assembly - Aft Fairing, Flap Support No. 2
113A9300	Fairing Installation - Flap Support No. 3 and 6, Outbd T.E. Flap
113A9304	Attach Beam Assembly - Aft Fairing, Flap Support No. 3
113A9307	Adjust Link Assembly - Aft Fairing, Flap Support No. 3
113A9310	FWD Fairing Assembly - Flap Support No. 3, Outbd T.E. Flap
113A9350	Aft Fairing Assembly - Flap Support No. 3, Outbd T.E. Flap
113A9355	Pivot Frame Assembly - Aft Fairing, Flap Support No. 3
113A9358	Mid Frame Assembly - Aft Fairing, Flap Support No. 3
113A9360	Aft Bulkhead Assembly - Aft Fairing, Flap Support No. 3

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G67901 S0006595079_V1

Flap Support Fairing Structure for the Number 1 Outboard Trailing Edge Flap
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Adjust Link	1.9 (48.3)	7050-T7451 plate as given in AMS 4050	
[2]	Bulkhead	2.25 (57.2)	7050-T7451 plate as given in AMS 4050	
[3]	Attach Beam	2.76 (70.1)	7050-T7451 plate as given in AMS 4050	
[4]	Pivot Frame	3.8 (96.5)	7050-T7451 plate as given in AMS 4050	
[5]	Frame	1.5 (38.1)	7050-T7451 plate as given in AMS 4050	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G67905 S0006595081_V1

Flap Support Fairing Structure for the Number 2 Outboard Trailing Edge Flap
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Adjust Link	1.9 (48.3)	7050-T7451 plate as given in AMS 4050	
[2]	Bulkhead	2.25 (57.2)	7050-T7451 plate as given in AMS 4050	
[3]	Attach Beam	2.86 (72.6)	7050-T7451 plate as given in AMS 4050	
[4]	Pivot Frame	3.8 (96.5)	7050-T7451 plate as given in AMS 4050	
[5]	Frame	1.5 (38.1)	7050-T7451 plate as given in AMS 4050	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G67915 S0006595083_V1

Flap Support Fairing Structure for the Number 3 Inboard Trailing Edge Flap
Figure 4

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Table 4:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Adjust Link	1.9 (48.3)	7050-T7451 plate as given in AMS 4050	
[2]	Bulkhead	2.25 (57.2)	7050-T7451 plate as given in AMS 4050	
[3]	Attach Beam	1.64 (41.7)	7050-T7451 plate as given in AMS 4050	
[4]	Mid Frame	1.8 (45.7)	7050-T7451 plate as given in AMS 4050	
[5]	Pivot Frame	3.8 (96.5)	7050-T7451 plate as given in AMS 4050	
[6]	Frame	1.5 (38.1)	7050-T7451 plate as given in AMS 4050	
[7]	Attach Fitting	1.96 (49.8)	7050-T7451 plate as given in AMS 4050	
[8]	Stub Beam	1.8 (45.7)	7050-T7451 plate as given in AMS 4050	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - WING TRAILING EDGE FLAP TRACK FAIRING SUPPORT STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the wing trailing edge flap track fairing support structure shown in Flap Support Fairing Location for the Wing Trailing Edge Flap, Figure 101/ ALLOWABLE DAMAGE 1.

G19149 S0006595087_V1

Flap Support Fairing Location for the Wing Trailing Edge Flap
Figure 101

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

FAIRING NUMBER 1 IS SHOWN, FAIRING NUMBER 2 IS ALMOST THE SAME

G19148 S0006595088_V1

Flap Support Fairing Structures Number 1 and Number 2 for the Outboard Trailing Edge Flap
Figure 102

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G19143 S0006595089_V1

Flap Support Fairing Structure Number 3 for the Inboard Trailing Edge Flap
Figure 103

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2. General

A. Do the steps that follow if you have damage to the aluminum parts:

(1) Remove the damage as necessary.

(a) Refer to 51-10-02 for the investigation and cleanup procedures.

(b) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

(2) After you remove the damage, do the procedures that follow:

(a) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(b) Apply a layer of BMS 10-11, Type I primer to the surfaces of the aluminum parts. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-40-08	COUNTERSINKING
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Frames, Bulkheads, Adjust Links, and Beams

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , C , D , E , and F.

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

G19152 S0006595090_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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Y = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL
ON A SURFACE AT AN EDGE

D

A–A

G19153 S0006595091_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

E

REMOVAL OF CORROSION AROUND THE FASTENERS

F

G19156 S0006595092_V1

Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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REPAIR 1 - WING TRAILING EDGE FLAP TRACK FAIRING SUPPORT STRUCTURE

1. Applicability

A. Repair 1 is applicable to damage to the flap support fairing adjust link as shown in Figure 201.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the size and type of damage that is permitted.

2. General

A. Repair 1 is a Category A repair. Refer to 51-00-06, GENERALto find the different categories of repairs.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G21607 S0006595095_V2

Flap Support Fairing Location for the Wing Trailing Edge Flap
Figure 201

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

FAIRING NUMBER 1 IS SHOWN, FAIRING NUMBER 2 IS ALMOST THE SAME

340003 S0000132252_V1

Adjust Link For the Flap Support Fairing Structure Numbers 1 and 2
Figure 202

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

340922 S0000132253_V1

Adjust Link For the Flap Support Fairing Structure Number 3
Figure 203

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3. References

Reference	Title
51-00-06, GENERAL	Structural Repair Definitions
51-20-05, GENERAL	Repair Sealing
AMM 27-51-18 P/B 401	INBOARD FLAP SUPPORT FAIRING - REMOVAL/INSTALLATION
AMM 27-51-18 P/B 501	INBOARD FLAP SUPPORT FAIRING - ADJUSTMENT/TEST
AMM 27-51-28 P/B 401	OUTBOARD FLAP SUPPORT FAIRINGS - REMOVAL/INSTALLATION
AMM 27-51-28 P/B 501	OUTBOARD FLAP SUPPORT FAIRINGS - ADJUSTMENT/TEST
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-42-05	Bright Cadmium Plating

4. Repair Instructions

A. Remove the adjust link. Refer to AMM 27-51-18 and 27-51-28. Remove all necessary bolts, retainer plates, and bearings from the lug.

B. Make sure that you machine the lower lug bore between 1.080 (27.4 mm) and 1.085 (27.56 mm) inch diameter. Refer to Figure 204.

(1) Make the surface finish to all cut surfaces 63 microinches Ra or smoother.

C. Remove material from the lug thickness on the side opposite the shimmed side. Make sure that the final lug thickness is between 0.47 inch (11.94 mm) and 0.48 inch (12.19 mm). Refer to Figure 204. Blend the machined lug surface into the adjacent flange.

(1) Make the surface finish to all cut surfaces 125 microinches Ra or smoother.

D. Dye penetrant inspect the lug bore as given in SOPM 20-20-02.

E. Apply one coat of BMS 10-79, Type III primer to the reworked locations other than the bore.

F. Apply one coat of BMS 10-60, Type I enamel to the reworked locations other than the bore.

G. Use bushing part number 113A9120-1 or fabricate a repair bushing as shown in Figure 204 from 17-4PH or 15-5PH CRES, condition A and do as follows:

NOTE: The four bolt holes shown in Figure 204 may be drilled after the bushing has been installed on the lug. It is permitted to use a 1/64 inch oversize fastener.

(1) If a repair bushing is fabricated, then heat treat it to 180 to 200 KSI and provide a CAD plate as given in SOPM 20-42-05, type II, class 2. Continue to complete all other part number 113A9120-1 requirements.

(2) Use the shrink fit method to install the bushing as given in BAC5435. Install the bushing with BMS 5-95 sealant in the lug bore. Refer to 51-20-05, GENERAL. Make sure you provide a fillet seal around the bushing edges.

(3) If necessary, obtain a new KSC278607BZ bearing as shown in Figure 204 and install wet with BMS 5-95 sealant. Refer to 51-20-05, GENERAL.

H. If the gap between the lug and the retainer plate is more than 0.004 inch (0.10 mm) as shown in Figure 204 then install a new part number 69B14136-3 shim to get a 0.001 (0.03 mm) to 0.004 inch (0.10 mm) gap prior to bolt installation and do as follows:

NOTE: As an alternative, fabricate the shim from BAC1534-15C or machine the shim from 301 CRES.

(1) Install the shim wet with BMS 5-95 sealant as given in 51-20-05, GENERAL.

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(2) Apply one layer of BMS 10-11, Type I primer to the shim as given in SOPM 20-41-02.

I. Install the retainer with (4) BACB30NM3HK7 bolts and the BACW10DS3S washers. It is permitted to use 1/64 inch oversize for the bolt diameter.

J. Lock wire the bolts together into two groups of two bolts. Use the double twist method as given in SOPM 20-50-02.

K. Reinstall the adjust link on the fairing as given in AMM PAGEBLOCK 27-51-18/401, AMM PAGEBLOCK 27-51-18/501, or AMM PAGEBLOCK 27-51-28/401, AMM PAGEBLOCK 27-51-28/501.

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Flap Support Fairing Adjust Link Repair
Figure 204

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GENERAL - AILERON DIAGRAM

WING PLAN VIEW

NOTES

- REFER TO 57-51-00 FOR TRAILING EDGE.
- REFER TO 57-53-00 FOR TRAILING EDGE FLAPS.
- REFER TO 57-70-00 FOR SPOILERS.

F91845 S0006595100_V1

Aileron Diagram
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
110A1502	Structures Diagram, Wing - Model 737-X
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly

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IDENTIFICATION 1 - AILERON SKIN

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT AILERON IS SHOWN, RIGHT AILERON IS OPPOSITE

F91887 S0006595105_V1

Aileron Skin Identification
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly
113A7100	Aileron Assembly
113A7200	Tab Assembly - Aileron
113A7300	Balance Panel Assembly - Aileron
113A7315	Panel Assembly - Balance Bay, Aileron
113A7400	Aileron Assembly

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UPPER SKIN OF LEFT AILERON IS SHOWN,
RIGHT AILERON IS OPPOSITE

A

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

F92326 S0006595107_V1

Aileron Skin Identification
Figure 2 (Sheet 1 of 2)

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LOWER SKIN OF LEFT AILERON IS SHOWN,
RIGHT AILERON IS OPPOSITE

B

F92430 S0006595108_V1

Aileron Skin Identification
Figure 2 (Sheet 2 of 2)

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Hinge Cover, Upper and Lower		Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: GFRP as given in BMS 8-139, Class I, Style 7781) Refer to the production drawing for the different hinge cover ply layups. Refer to Figure 3 for a typical hinge cover ply layup	
[2]	Bonded Upper Skin Panel		Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW	
	Skin		Refer to Figure 4	
	Core (Between the outboard closure rib and the tab cutout rib)	0.400 (10.16)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
	Core (Between the tab cutout rib and ASTA 65.05)	0.400 (10.16)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
	Core (ASTA 59.15 to ASTA 11.31)	0.400 (10.16)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[3]	Bonded Lower Skin Panel		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW	
	Skin		Refer to Figure 5	
	Core (Between the outboard closure rib and the tab cutout rib)	0.400 (10.16)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
	Core (Between the tab cutout rib and ASTA 65.05)	0.400 (10.16)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
	Core (ASTA 59.15 to ASTA 11.31)	0.400 (10.16)	Non-metallic honeycomb as given in BMS 8-124, Class IV, Type V, Grade 3.0	
[4]	Fixed Fairing - Actuator, Aileron		GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 6	
[5]	Removable Fairing- Actuator, Aileron		GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 7	
[6]	Removable Fairing - Control Rod, Aileron		GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 8	
[7]	Fixed Fairing - Control Rod, Aileron		GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 9	
[8]	Balance Panel (4)	0.071 (1.80)	7075-T73 sheet as given in QQ-A-250/12	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT AILERON HINGE COVER IS SHOWN,
RIGHT AILERON HINGE COVER IS OPPOSITE

TYPICAL UPPER AND LOWER HINGE COVERS

A–A

B–B

F92474 S0006595110_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for Figure 2, Item [1]
Figure 3

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Table 3:

PLY MATERIAL AND DIRECTION FOR FIGURE 3		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P5, P7, P9, P11	+ or - 45 degrees	GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: GFRP as given in BMS 8-139, Class I, Style 7781)
P2, P4, P6, P8, P10	0 or 90 degrees	GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581 (Optional: GFRP as given in BMS 8-139, Class I, Style 7781)

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LEFT AILERON SKIN IS SHOWN, RIGHT AILERON SKIN IS OPPOSITE

VIEW IS ON THE BAGSIDE (NON-AERODYNAMIC) SURFACE

PLY LAYUP AND CORE RIBBON DIRECTION

A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS. REFER TO THE ENGINEERING DRAWING FOR MORE DATA.
- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN A-A AND B-B.

F92629 S0006595112_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Upper Skin Panel, Figure 2, Item [2]
Figure 4 (Sheet 1 of 3)

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Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Upper Skin Panel, Figure 2, Item [2]
Figure 4 (Sheet 2 of 3)

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F92672 S0006595114_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Upper Skin Panel, Figure 2, Item [2]

Figure 4 (Sheet 3 of 3)

Table 4:

PLY MATERIAL AND DIRECTION FOR FIGURE 4		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P7, P9, P15, P17, P21, P23	+ or - 45 Degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P5, P6, P8, P10, P11, P12, P13, P14, P16, P18, P19, P20, P22	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P24	Optional	GFRP woven fabric as given in BMS 8-139, Class III, Style 108

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LEFT AILERON SKIN IS SHOWN, RIGHT AILERON SKIN IS OPPOSITE

VIEW IS ON THE BAGSIDE (NON-AERODYNAMIC) SURFACE

PLY LAYUP AND CORE RIBBON DIRECTION

A

A-A

B-B

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO A-A AND B-B FOR THE PLY SEQUENCE AT THOSE LOCATIONS. REFER TO THE ENGINEERING DRAWING FOR MORE DATA.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN A-A AND B-B.

F92758 S0006595116_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Lower Skin Panel, Figure 2, Item [3]
Figure 5 (Sheet 1 of 3)

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F92868 S0006595117_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Lower Skin Panel, Figure 2, Item [3]
Figure 5 (Sheet 2 of 3)

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F92887 S0006595118_V1

Ply Direction, Core Ribbon Direction, and Ply Sequence for the Bonded Lower Skin Panel, Figure 2, Item [3]

Figure 5 (Sheet 3 of 3)

Table 5:

PLY MATERIAL AND DIRECTION FOR FIGURE 5		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P5, P7, P9, P10, P15, P16, P18, P20, P22, P24	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P6, P8, P11, P12, P13, P14, P17, P19, P21, P23	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P25	Optional	GFRP woven fabric as given in BMS 8-139, Class III, Style 108

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LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

FIXED FAIRING - AILERON ACTUATOR

A-A

A

NOTES

- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

F92940 S0006595120_V1

Ply Direction and Ply Sequence for Figure 2, Item [4]
Figure 6

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 6		
PLY	**DIRECTION**	**MATERIAL**
P1 through P4	0 or 90 degrees	GFRP woven fabric as given in BMS 8-139, Class III, Grade B, Style 1581

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NOTES

- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

F93057 S0006595122_V1

Ply Direction and Ply Sequence for Figure 2, Item [5]
Figure 7

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 7		
PLY	**DIRECTION**	**MATERIAL**
P1 through P5	0 or 90 degrees	GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581

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LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

REMOVABLE FAIRING - AILERON CONTROL ROD

A–A

A

NOTES

- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

F93078 S0006595124_V1

Ply Direction and Ply Sequence for Figure 2, Item [6]
Figure 8

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 8		
PLY	**DIRECTION**	**MATERIAL**
P1, P4, P7, P10, P13	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P3, P5, P6, P8, P9, P11, P12	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

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LEFT AILERON FAIRING SHOWN, RIGHT AILERON FAIRING OPPOSITE

FIXED FAIRING - AILERON CONTROL ROD

NOTES

- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

F93203 S0006595126_V1

Ply Direction and Ply Sequence for Figure 2, Item [7]
Figure 9

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Table 9:

Y MATERIAL AND DIRECTION FOR FIGURE 9		
PLY	**DIRECTION**	**MATERIAL**
P1 through P7	0 or 90 degrees	GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581

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IDENTIFICATION 2 - AILERON TAB SKIN

+

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT AILERON TAB IS SHOWN,
RIGHT AILERON TAB IS OPPOSITE

F93219 S0006595129_V1

Aileron/Tab Skin Identification
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly
113A7200	Tab Assembly - Aileron
113A7300	Balance Panel Assembly - Aileron
113A7315	Panel Assembly - Balance Bay, Aileron

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

F93309 S0006595131_V1

Aileron Tab Skin Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Tab Part Skin Tab Spar		Carbon Fiber Reinforced Plastic (CFRP) as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW Refer to Figure 3 Refer to Figure 4	
[2]	Hinge Cover		Nylon injection molding resin as given in BMS 8-323, Type I, Class 3, Grade 30, Form B. Color same as BAC 707 Gray	
[3]	Removable Fairing - Aileron Tab Control Rod		Glass Fiber Reinforced Plastic (GFRP) as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 5	
[4]	Fixed Fairing - Aileron Tab Control Rod		GFRP as given in BMS 8-79, Class III, Grade B, Style 1581. Refer to Figure 6	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT AILERON TAB IS SHOWN, RIGHT AILERON TAB IS OPPOSITE

OUTBOARD VIEW OF BONDED AILERON TAB SKIN

A–A

B–B

NOTES

- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN B–B.

F93370 S0006595133_V1

Ply Direction and Ply Sequence for the Skin, Figure 2, Item [1]
Figure 3

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Table 3:

PLY MATERIAL AND DIRECTION FOR THE AILERON TAB SKIN, FIGURE 2, ITEM [1]		
PLY	**DIRECTION**	**MATERIAL**
P1, P4, P6, P9	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P3, P5, P7, P8	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

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REAR VIEW OF LEFT AILERON TAB SPAR IS SHOWN,
RIGHT AILERON TAB SPAR IS OPPOSITE

BONDED AILERON TAB SPAR

A

A-A

B-B

NOTES

- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN B-B.

F93378 S0006595135_V1

Ply Direction and Ply Sequence for the Tab Spar, Figure 2, Item [1]
Figure 4

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Table 4:

PLY MATERIAL AND DIRECTION FOR THE AILERON TAB SPAR, FIGURE 2, ITEM [1]		
PLY	**DIRECTION**	**MATERIAL**
P1	Optional	GFRP woven fabric as given in BMS 8-79, Class III, Style 108
P2 through P10, P13	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P11, P12	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

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LEFT TAB FAIRING IS SHOWN, RIGHT TAB FAIRING IS OPPOSITE

REMOVABLE FAIRING - AILERON TAB CONTROL ROD

A-A

A

NOTES

- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

F93383 S0006595137_V1

Ply Direction and Ply Sequence for the Removable Fairing, Figure 2, Item [3]
Figure 5

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Table 5:

PLY MATERIAL AND DIRECTION FOR THE REMOVABLE FAIRING, FIGURE 2, ITEM [3]		
PLY	**DIRECTION**	**MATERIAL**
P1 through P6	0 or 90 degrees	GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581

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LEFT CONTROL ROD FAIRING IS SHOWN,
RIGHT CONTROL ROD FAIRING IS OPPOSITE

FIXED FAIRING - AILERON TAB CONTROL ROD

A-A

A

NOTES

- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN DETAIL A.

F93387 S0006595139_V1

Ply Direction and Ply Sequence for Fixed Fairing, Figure 2, Item [4]
Figure 6

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Table 6:

PLY MATERIAL AND DIRECTION FOR THE FIXED FAIRING, FIGURE 2, ITEM [4]		
PLY	**DIRECTION**	**MATERIAL**
P1 through P6	0 or 90 degrees	GFRP woven fabric as given in BMS 8-79, Class III, Grade B, Style 1581

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ALLOWABLE DAMAGE 1 - AILERON SKIN

1. Applicability

A. This subject gives the allowable damage limits for the upper and lower aileron skins shown in Wing Aileron Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 1.

B. For the allowable damage zones of the aileron, refer to:

(1) Aileron Upper Skin Zones, Figure 102/ALLOWABLE DAMAGE 1 for the aileron upper skin

(2) Aileron Lower Skin Zones, Figure 103/ALLOWABLE DAMAGE 1 for the aileron lower skin.

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F86614 S0006595143_V1

Wing Aileron Skin Panel Location
Figure 101

2. General

A. Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) For the honeycomb core areas, the tap test is an alternative procedure to an instrumented NDT.

(2) Refer to Damage Definitions, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

(3) Refer to Definitions of the Facesheets, Figure 105/ALLOWABLE DAMAGE 1 for the definitions of the facesheets of a honeycomb area.

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B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two procedures that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Make sure the tape is in satisfactory condition after each 400 flight hours interval or more frequently.

(d) Repair the damage at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area. Refer to 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Use a vacuum and heat to remove moisture from the solid laminate and honeycomb cells. Refer to 51-70-04.

(2) Make a temporary seal with aluminum foil tape (speed tape).

(3) Keep a record of the location.

(4) Repair the damage at or before 400 flight hours from the time the seal was made.

E. The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause carbon fiber damage and are sealed.

F. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF STRUCTURE	**ZONE LOCATION**	**PARAGRAPH**
HONEYCOMB CORE AREAS	1	4.A
SOLID LAMINATE AREAS	2	4.B
	3	4.C
	4	4.D
	5	4.E
	6	4.F

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NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANELS

F86618 S0006595145_V1

Aileron Upper Skin Zones
Figure 102

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NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LOWER SKIN PANELS

F86619 S0006595146_V1

Aileron Lower Skin Zones
Figure 103

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F89816 S0006595147_V1

Damage Definitions
Figure 104

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G39273 S0006595148_V1

Definitions of the Facesheets
Figure 105

3. References

Reference	Title
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-60-01	AILERON BALANCING PROCEDURES
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits

A. Zone 1 - Honeycomb Core Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 2.0 inches (50.8 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a fitting.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage.

(4) Holes and Punctures are permitted if they are:

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(a) A maximum of 0.50 inch (12.7 mm) of one toolside facesheet and the core in depth

(b) A maximum of 0.50 inch (12.7 mm) in diameter

(c) A minimum of 3.0 inches (76.2 mm) away from other damage

(d) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 106/ALLOWABLE DAMAGE 1, if it is:

(a) A maximum of one toolside facesheet in depth

(b) A maximum of 0.50 inch (12.7 mm) in length

(c) A maximum of 0.50 inch (12.7 mm) in width.

B. Zone 2 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 2.0 inches (50.8 mm) in length

(b) A maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.45 inch (11.43 mm) in diameter

(b) A minimum of 3.0 inches (76.2 mm) away from other damage.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A, if it is:

(a) A maximum of 2.0 inches (50.8 mm) in length

(b) A maximum of 0.90 inch (22.86 mm) in width.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.06 inch (1.52 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A.

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(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail B.

C. Zone 3 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 2.0 inches (50.8 mm) in length

(b) A maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch 1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.60 inch (15.24 mm) in diameter

(b) A minimum of 3.0 inches (76.2 mm) away from other damage.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 1.50 inches (38.1 mm) in length

(b) A maximum of 0.90 inch (22.86 mm) in width.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.05 inch (1.27 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail B.

D. Zone 4 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 2.0 inches (50.8 mm) in length

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(b) A maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage

(e) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage

4) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.50 inch (12.7 mm) in diameter

(b) A minimum of 3.0 inches (76.2 mm) away from other damage

(c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.50 inch (12.7 mm) in length

(b) A maximum of 0.50 inch (12.7 mm) in width

(c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.08 inch (2.03 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A

(e) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail B.

E. Zone 5 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of 0.50 inch (12.7 mm) in length

(b) A maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

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(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.0 inches (76.2 mm) away from other damage

(e) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.50 inch (12.7 mm) in width

2) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

3) A minimum of 3.0 inches (76.2 mm) away from other damage

4) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(4) Holes and Punctures are permitted if they are:

(a) A maximum of 0.25 inch (6.35 mm) in diameter

(b) A minimum of 3.0 inches (76.2 mm) away from other damage

(c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.50 inch (12.7 mm) in length

(b) A maximum of 0.50 inch (12.7 mm) in width

(c) A minimum of 3.0 inches (76.2 mm) away from a fitting.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.09 inch (2.29 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A maximum of 0.50 inch (12.7 mm) away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail A.

(e) A minimum of 3.0 inches (76.2 mm) from a fitting.

(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 107/ALLOWABLE DAMAGE 1, Detail B.

F. Zone 6 - Solid Laminate Areas

(1) Damage is not permitted.

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.0 INCHES (76.2 mm).

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

F88622 S0006595149_V1

Allowable Damage Limits
Figure 106

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.0 INCHES (76.2 mm).

A MINIMUM OF 0.50 INCH (12.7 mm) FROM THE EDGE OF THE PART IS ALLOWED.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A

CLEANUP AND SEALING OF EDGE EROSION

B

F88623 S0006595150_V1

Allowable Damage Limits
Figure 107

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ALLOWABLE DAMAGE 2 - AILERON TAB SKIN

1. Applicability

A. This subject gives the allowable damage limits for the aileron tab skin as shown in Aileron Tab Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.

B. Refer to Aileron Tab Skin Zones, Figure 102/ALLOWABLE DAMAGE 2 for the aileron tab skin shapes.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F87613 S0006595154_V1

Aileron Tab Skin Panel Location
Figure 101

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NOTE: ALL DIMENSIONS ARE IN INCHES.

UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A-A

F87698 S0006595155_V1

Aileron Tab Skin Zones
Figure 102

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2. General

A. Do an inspection of the damaged area to find the length, width, and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C for the definitions of the length, width, and depth of damage.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two methods that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

(d) Repair the damage at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Use a vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04.

(2) Make a temporary seal with aluminum foil tape (speed tape).

(3) Keep a record of the location.

(4) Repair the damage at or before 400 flight hours from the time the seal was made.

E. The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause carbon fiber damage and are sealed.

F. Make sure that the aileron tab is balanced. Refer to 51-60-03 for the balance procedures.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF STRUCTURE	**ZONE LOCATION**	**PARAGRAPH**
SOLID LAMINATE AREAS	1	4.A
	2	4.B

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DEFINITIONS FOR
NICK, GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR
DENT DAMAGE

B

DEFINITIONS FOR
EDGE DAMAGE

C

F89787 S0006595157_V1

Damage Definitions
Figure 103

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3. References

Reference	Title
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-60-03	AILERON TAB BALANCE PROCEDURE
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits

A. Zone 1 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if:

(a) The damage is a maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) The damage is up to a maximum of 2.0 inches in length

(c) The damage is a maximum of 0.25 inch in width

(d) The damage is a minimum of 3.0 inches away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if:

1) The damage is a maximum of 0.50 inch in diameter

2) The damage is a maximum of 0.05 inch in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch.

3) The damage is a minimum of 3.0 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) The damage is a maximum of 0.50 inch in diameter

(b) The damage is a minimum of 3.0 inches away from other damage.

(5) Delamination is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail A if:

(a) The damage is a maximum of 0.50 inch in length

(b) The damage is a maximum of 0.50 inch in width.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.10 inch in depth

(b) A maximum of 0.25 inch in width

(c) A minimum of 0.50 inch away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail A or 4.0 inches.

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(7) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail B .

B. Zone 2 - Solid Laminate Areas

(1) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(2) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if:

(a) The damage is a maximum of one ply

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) The damage is up to a maximum of 1.50 inches in length

(c) The damage is a maximum of 0.25 inch in width

(d) The damage is a minimum of 3.0 inches away from other damage.

(3) Dents:

(a) Dents that do not cause damage to the carbon fibers is permitted if:

1) The damage is a maximum of 0.50 inch in diameter

2) The damage is a maximum of 0.05 inch in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch.

3) The damage is a minimum of 3.0 inches away from other damage.

(4) Holes and Punctures are permitted if:

(a) The damage is a maximum of 0.35 inch in diameter

(b) The damage is a minimum of 3.0 inches away from other damage.

(5) Delamination is permitted as shown in Figure 104/ALLOWABLE DAMAGE 2, Detail A if:

(a) The damage is a maximum of 0.40 inch in length

(b) The damage is a maximum of 0.40 inch in width.

(6) Edge damage is permitted if it is:

(a) A maximum of 0.08 inch in depth

(b) A maximum of 0.25 inch in width

(c) A minimum of 0.50 inch away from the edge of a fastener hole

(d) A minimum distance from the edge of other damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 2, Detail A .

(7) Edge Erosion is permitted as shown in Figure 104/ALLOWABLE DAMAGE 2, Detail B .

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 1.0 INCH.

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

0.50 IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS THE LARGER OF 0.75D OR 2d.

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A

F87726 S0006595158_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 2)

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SEALING OF EROSION DAMAGE AT AN EDGE OF COMPOSITE PARTS

B

F88233 S0006595159_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 2)

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REPAIR 1 - AILERON SKIN

1. Applicability

A. Repair 1 is applicable to the upper and lower skin panels of the aileron shown in Wing Aileron Skin Panel Locations, Figure 201/REPAIR 1.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F92554 S0006595164_V1

Wing Aileron Skin Panel Locations
Figure 201

2. General

A. Repair 1 gives instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the aileron, as necessary. Refer to AMM 27-11-11/401.

C. Remove the skin as necessary to get access to the inner surface. Refer to AMM 27-11-11/401.

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(1) If the fastener hole is damaged, refer to 51-70-04 or 51-70-05 for a repair.

D. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) For the honeycomb core areas that have damage on a facesheet with 3 or less initial plies, the tap test is an alternative procedure to an instrumented NDT.

(2) For honeycomb areas that have damage on a facesheet with 4 or more initial plies, the tap test is an alternative procedure to an instrumented NDT if:

(a) It can be shown that a defect that is less than or equal to the maximum allowable damage size can be found.

(3) Refer to Damage Definitions, Figure 202/REPAIR 1 for the definitions of the diameter and depth of damage.

(4) Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definitions of the facesheets of a honeycomb core area.

E. Do the repair as given in Paragraph 4./REPAIR 1

F. Install the skin you removed as necessary, to get access to the damage. Refer to AMM 27-11-11/401.

(1) If the fastener hole is damaged, refer to 51-70-04 or 51-70-05 for a repair.

G. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

H. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

I. Install the aileron, as applicable. Refer to AMM 27-11-11/401.

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DEFINITIONS FOR NICK, GOUGE, OR SCRATCH DAMAGE

DEFINITIONS FOR EDGE DAMAGE

DEFINITIONS FOR DENT DAMAGE

DEFINITIONS FOR DENT DAMAGE

G88102 S0006595165_V1

Damage Definitions
Figure 202

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F92559 S0006595166_V1

Definitions of the Facesheets
Figure 203

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NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

UPPER SKIN PANELS

F92569 S0006595167_V1

Aileron Upper Skin Repair Zones
Figure 204 (Sheet 1 of 3)

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NOTE: ALL DIMENSIONS ARE IN INCHES.
ALL DIMENSIONS IN PARENTHESES ARE IN MILLIMETERS.

RIGHT SIDE IS SHOWN, LEFT SIDE IS OPPOSITE

LOWER SKIN PANELS

F92573 S0006595168_V1

Aileron Upper Skin Repair Zones
Figure 204 (Sheet 2 of 3)

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DEFINITION OF THE SOLID LAMINATE
AND HONEYCOMB CORE AREAS

A

G88192 S0006595169_V1

Aileron Upper Skin Repair Zones
Figure 204 (Sheet 3 of 3)

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-60-01	AILERON BALANCING PROCEDURES
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-05	REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS
AMM 51-21-00	INTERIOR AND EXTERIOR FINISHES
AMM 27-11-11/401	Aileron - Removal/Installation
737 NDT Part 1, 51-01-01	Inspection of Repairs to Composite Structure
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Repair Instructions

A. If a dent is 2 inches in diameter or less and has no fiber damage or delamination, then fill the dent with potting compound and apply a fiberglass patch as given in Repair 14 of 51-70-04.

B. If Paragraph 4.A./REPAIR 1 is not applicable, then refer to Table 201/REPAIR 1 for the repair limits that are applicable to each zone.

Table 201:

REPAIR INDEX	
ZONES	REPAIR TABLES
1	202
2 and 3	203
4 and 5	204
6	205

C. If there is damage in more than one zone, then make sure:

(1) Each part of the repair that is in a given zone agrees with the repair size limits of that zone

(2) The type of repair and repair instructions agree with the minimum requirements for all the zones that the repair is in

(3) You refer to the example given in Paragraph 5./REPAIR 1 for damage in Zones 1 and 2.

D. For a repair made with wet layup materials, do as follows, as applicable:

(1) If the core is damaged, then fill with BMS 5-28, Type 7 potting compound.

(2) Use one repair ply of fabric for each initial ply that was damaged.

(3) Add two structural fiberglass plies of fabric for each facesheet that is repaired. Refer to 51-70-04. Put one structural ply at ±45 degrees to the core ribbon direction and the other at 0 or 90 degrees. Add a third nonstructural ply made of fiberglass where the ply direction is optional.

NOTE: Repair plies or added plies are not necessary in the repair of a delamination at an edge if the delamination is a minimum of 2D (D = fastener diameter) away from a fastener hole.

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(4) Inspect a Category B repair after each 600 flight hour interval or more frequently. Refer to 737 NDT Part 1, 51-01-01 and 51-70-04 for the inspection procedures. If deterioration is found, then the repair must be replaced with a Category A repair.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator, can be used.

E. For a repair made with preimpregnated layup materials, use the same number of repair plies as the number of initial plies that were damaged.

F. Use the repair instructions that follow for a repair in Zone 1 made with wet layup materials at 200°F (93°C) cure.

(1) Repair the damage as given in 51-70-04, but for each facesheet (over a honeycomb core or solid laminate) that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that are removed. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definition of a facesheet.

(b) Add one ply (structural) that is + or - 45 degrees.

(c) Add a second ply (structural) that is 0 or 90 degrees.

(d) Add a third ply (non-structural) made of fiberglass with an optional ply direction.

G. Use the repair instructions that follow for a repair in Zone 1 made with preimpregnated layup materials at 250°F (121°C) cure.

(1) Repair the damage as given in 51-70-05, but for each facesheet (over a honeycomb core or solid laminate) that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that are removed from a facesheet. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definition of a facesheet.

(b) Add one ply (structural) that is + or - 45 degrees.

(c) Add a second ply (structural) that is 0 or 90 degrees.

(d) Add a third ply (structural) that is + or - 45 degrees.

(e) Add a fourth ply (non-structural) made of fiberglass with an optional ply direction.

H. Use the repair instructions that follow for a repair in Zones 1, 2, and 3 made with preimpregnated layup materials at 350°F (177°C) cure.

(1) Repair the damage as given in 51-70-05, but for each facesheet (over a honeycomb core) or solid laminate area that is damaged, do the steps that follow:

(a) Use the same number of repair plies as the number of initial plies that are removed from a facesheet or solid laminate area. Refer to Definitions of the Facesheets, Figure 203/REPAIR 1 for the definition of a facesheet.

(b) Add one ply (nonstructural) made of fiberglass with an optional ply direction.

5. Example of Repair Options for Damage That Is In Zones 1 and 2.

A. Refer to Table 201/REPAIR 1 for the repairs that are applicable in Zone 1.

(1) 250°F preimpregnated layup repairs are permitted if:

(a) The damage is a maximum of 10.0 inches in length

(b) The damage is a maximum of 65 percent of the smallest dimension across the panel at the damage location

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(c) You add two more structural plies than the initial number of structural plies and one nonstructural fiberglass ply

(d) You then add a nonstructural ply made of fiberglass.

(2) 200°F wet layup repairs are permitted if:

(a) The damage is a maximum of 10.0 inches in length

(b) The damage is a maximum of 65 percent of the smallest dimension across the panel at the damage location

(c) You add two more structural plies than the initial number of structural plies

(d) You then add a nonstructural ply made of fiberglass.

(3) 350°F preimpregnated layup repairs are permitted with no size limits.

B. Refer to Table 201/REPAIR 1 for the repairs that are applicable in Zone 2.

(1) 200°F wet layup repairs are permitted if:

(a) The damage is a maximum of 10.0 inches in length

(b) The damage is a maximum of 65 percent of the smallest dimension across the panel at the damage location

(c) You add four more structural plies than the initial number of structural plies

(d) You then add a ply nonstructural made of fiberglass.

(2) 250°F preimpregnated layup repairs are permitted since they are listed in the table.

(3) 350°F preimpregnated layup repairs are permitted with no size limits.

C. Do a repair that agrees with the requirements of both Zones 1 and 2.

(1) Since a 250°F preimpregnated layup repair is not permitted in Zone 2, it can not be used for damage that is in both Zones 1 and 2.

(2) A 350°F preimpregnated layup repair is permitted for both Zones 1 and 2 with no size limits and the same repair instructions (Paragraph 4.F./REPAIR 1).

(3) A 200°F wet layup repair is permitted for both Zones 1 and 2 if the conditions that follow are true:

(a) The part of the damage that is in Zone 1 must agree with the size limits given in Paragraphs A.(1)(a) and A.(1)(b).

(b) The part of the damage that is in Zone 2 must agree with the size limits given in Paragraphs B.(1)(a) and B.(1)(b).

(c) Zone 1 adds two more structural plies than the initial number of structural plies and Zone 2 adds four more. The result is that you must use four more structural plies in both zones to agree with the minimum requirements of Zone 1. A nonstructural ply made of fiberglass is then added.

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Table 202:

REPAIR DATA FOR ZONE 1 OF THE 350°F (177°C) CURE, AILERON HONEYCOMB SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 2.00 inch in diameter - 30 percent of the smallest dimension across the panel at the damage location - one facesheet and the honeycomb core in depth One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 10.0 inches in diameter - 65 percent of the smallest dimension across the panel at the damage location One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 10.0 inches in diameter - 65 percent of the smallest dimension across the panel at the damage location One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	There are no limits on the dimension of the repair.
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.D	SRM 51-70-04 and Paragraph 4.F	SRM 51-70-05 and Paragraph 4.G	SRM 51-70-05 and Paragraph 4.H

Table 203:

REPAIR DATA FOR ZONE 2 AND 3 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED lAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS FOR ZONE 2	Damage that is a maximum of: - 2.00 inches in diameter - 30 percent of the smallest dimension across the panel at the damage location - one facesheet depth One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 10.0 inches in diameter - 65 percent of the smallest dimension across the panel at the damage location One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 10.0 inches in diameter - 65 percent of the smallest dimension across the panel at the damage location One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	There are no size limits on the repair.
REPAIR SIZE AND LIMITS FOR ZONE 3	- Contact The Boeing Company	- Contact The Boeing Company	- Contact The Boeing Company	There are no size limits on the repair.

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Table 203: (Continued)

REPAIR DATA FOR ZONE 2 AND 3 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED IAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.D	SRM 51-70-04 and Paragraph 4.F	SRM 51-70-05 and Paragraph 4.G	SRM 51-70-05 and Paragraph 4.H

Table 204:

REPAIR DATA FOR ZONE 4 AND 5 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED IAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS FOR ZONE 5	A repair is not permitted.	Damage that is a maximum of: - 2.0 inches in diameter - 10 percent of the edgeband length on the side of the damage, as applicable One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 2.0 inches in diameter - 10 percent of the edgeband length on the side of the damage, as applicable One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	There are no size limits on the repair.
REPAIR SIZE AND LIMITS FOR ZONE 4	- Contact The Boeing Company	- Contact The Boeing Company	- Contact The Boeing Company	There are no size limits on the repair.
REPAIR PROCEDURES	------	SRM 51-70-04 and Paragraph 4.F	SRM 51-70-05 and Paragraph 4.G	SRM 51-70-05 and Paragraph 4.H

Table 205:

REPAIR DATA FOR ZONE 6 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED IAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	A repair is not permitted.	Damage that is a maximum of: - 2.0 inches in diameter	Damage that is a maximum of: - 2.0 inches in diameter	There are no size limits on the repair.

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Table 205: (Continued)

REPAIR DATA FOR ZONE 6 OF THE 350°F (177°C) CURE, AILERON LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED IAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
		- 10 percent of the edgeband length on the side of the damage, as applicable One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	- 10 percent of the edgeband length on the side of the damage, as applicable One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	
REPAIR PROCEDURES	----	SRM 51-70-04 and Paragraph 4.F	SRM 51-70-05 and Paragraph 4.G	SRM 51-70-05 and Paragraph 4.H

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REPAIR 2 - AILERON TAB SKIN

1. Applicability

A. Repair 2 is applicable to the Aileron Tab Skin shown in Aileron Tab Skin Panel Location, Figure 201/ REPAIR 2.

B. Repair 2 is applicable to damage that is more than the limits permitted in Allowable Damage 2. Refer to Allowable Damage 2 for the type and size of damage that is permitted.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F93521 S0006595176_V1

Aileron Tab Skin Panel Location
Figure 201

2. General

A. Repair 2 gives instructions for Category A and B repairs. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the aileron, as necessary. Refer to AMM 27-11-11/401.

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C. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Definitions of Damage Size, Figure 202/REPAIR 2 for the definitions of the diameter and depth of damage.

D. Do the repair as given in Paragraph 4./REPAIR 2

E. Make sure the aerodynamic smoothness is satisfactory or there can be a decrease in the airplane performance. Refer to 51-10-01.

F. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

G. Install the aileron tab, as applicable. Refer to AMM 27-11-11/401.

F93522 S0006595177_V1

Definitions of Damage Size
Figure 202

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NOTE: ALL DIMENSIONS ARE IN INCHES.

UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

F93542 S0006595178_V1

Aileron Tab Skin Repair Zones
Figure 203

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-60-01	AILERON BALANCING PROCEDURES
51-60-03	AILERON TAB BALANCE PROCEDURE
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
AMM 51-21-00	INTERIOR AND EXTERIOR FINISHES
AMM 27-11-11/401	Aileron - Removal/Installation
737 NDT Part 1, 51-01-01	Inspection of Repairs to Composite Structure
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Repair Instructions

A. If a dent is 2 inches in diameter or less, and has no fiber damage or delamination, then fill the dent with potting compound and apply a fiberglass patch as given in Repair 14 of 51-70-04.

B. If Paragraph 4.A./REPAIR 2 is not applicable, then refer to:

(1) Table 201/REPAIR 2 for the repair data that is applicable to damage in Zone 1

(2) Table 202/REPAIR 2 for the repair data that is applicable to damage in Zone 2

(3) Table 203/REPAIR 2 for the repair data that is applicable to damage in Zone 3.

C. For repairs made with wet layup materials, do as follows, as applicable:

(1) Use one repair ply of fabric for each initial ply that was damaged.

(2) Add two structural plies of fabric for each facesheet that is repaired. Put one structural ply at ±45 degrees to the core ribbon direction and the other at 0 or 90 degrees.

NOTE: Repair plies or added plies are not necessary in the repair of delamination at an edge if the delamination is a minimum of 2D (D = fastener diameter) away from a fastener hole.

(3) Inspect Category B repairs after each 800 flight hour interval or more frequently. Refer to 737 NDT Part 1, 51-01-01 for inspection procedures. If deterioration is found, then they must be replaced with Category A repairs.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

D. For repairs made with preimpregnated layup materials, use the same number of repair plies as the number of initial plies that were damaged.

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Table 201:

REPAIR DATA FOR ZONE 1 OF THE 350°F (177°C) CURE, AILERON TAB LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 1.0 inch in diameter - 30 percent of the smallest dimension across the panel. - One facesheet in depth One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 2.0 inches in diameter - 50 percent of the smallest dimension across the panel. One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 2.50 inches in diameter - 50 percent of the smallest dimension across the panel.	There are no size limits on the repair.
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-05 and Paragraph 4.D	SRM 51-70-05 and Paragraph 4.D

Table 202:

REPAIR DATA FOR ZONE 2 OF THE 350°F (177°C) CURE, AILERON TAB LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 0.50 inch in diameter - 15 percent of the edgeband length on the side of the damage, as applicable One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 1.0 inch in diameter - 10 percent of the edgeband length on the side of the damage, as applicable One repair for each 144 square inches 3.0 inches minimum clearance from: - other repairs - fastener holes - panel edges	Damage that is a maximum of: - 1.5 inches in diameter - 10 percent of the edgeband length on the side of the damage, as applicable	There are no size limits on the repair
REPAIR PROCEDURES	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-04 and Paragraph 4.C	SRM 51-70-05 and Paragraph 4.D	SRM 51-70-05 and Paragraph 4.D

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Table 203:

REPAIR DATA FOR ZONE 3 OF THE 350°F (177°C) CURE, AILERON TAB LAMINATE SKIN PANELS				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	Damage that is a maximum of: - 1.0 inch in length - 30 percent of the length of the leading edge between cutouts	Damage that is a maximum of: - 2.0 inches in length - 50 percent of the length of the leading edge between cutouts	Damage that is a maximum of: - 2.5 inches length - 50 percent of the length of the leading edge between cutouts	There are no size limits on the repair.
REPAIR PROCEDURES	SRM 51-70-04	SRM 51-70-04	SRM 51-70-05	SRM 51-70-05

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IDENTIFICATION 1 - AILERON STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT AILERON IS SHOWN, RIGHT AILERON IS OPPOSITE

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Aileron Structure Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly
113A7100	Aileron Assembly
113A7101	Front Spar Assembly - Aileron
113A7400	Aileron Assembly

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LEFT AILERON IS SHOWN, RIGHT AILERON IS OPPOSITE

AILERON ASSEMBLY

NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

F96313 S0006595187_V1

Aileron Structure Identification
Figure 2 (Sheet 1 of 2)

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F96577 S0006595188_V1

Aileron Structure Identification
Figure 2 (Sheet 2 of 2)

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Bonded Front Spar Assembly			
	Bonded Spar		Carbon Fiber Reinforced Plastic (CFRP) woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 3	
	Bonded Rib Post (6)		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to the production drawing for each rib post ply layup. Refer to Figure 4 for a typical rib post ply layup	
	Nose Fitting (10)		15-5PH Investment Casting	
	Nose Rib (5)	0.250 (6.35)	7075-T7451 plate	
[2]	Bonded Rear Spar		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 5	
[3]	Bonded Outboard Closure Rib		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 6	
[4]	Bonded Tab Cutout Rib		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 7	
[5]	Bonded Inboard Closure Rib		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 8	
[6]	Bonded Actuator Backup Rib		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 9	
[7]	Bonded Support Rib		CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW. Refer to Figure 10	
[8]	Rib Post	0.050 (1.27)	Ti-6Al-4V titanium sheet as given in MIL-T-9046, Code AB-1 in the annealed condition	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT BONDED FRONT SPAR IS SHOWN,
RIGHT BONDED FRONT SPAR IS OPPOSITE

INBD

VIEW IS LOOKING FORWARD

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 3 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96150 S0006595190_V1

Ply Ribbon Direction and Ply Sequence for the Bonded Spar,Figure 2, Item [1]
Figure 3

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Table 3:

PLY MATERIAL AND DIRECTION FOR FIGURE 3		
PLY	**DIRECTION**	**MATERIAL**
P1	Optional	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class III, Style 108
P2, P4, P34, P5, P8, P9, P11 thru P13, P17, P18, P22 thru P24, P26, P27, P30, P36, P31, P33	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P3, P35, P6, P7, P10, P14 thru P16, P19 thru P21, P25, P28, P29, P37, P32	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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PLAN VIEW

BONDED RIB POST

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 4 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96575 S0006595192_V1

Ply Ribbon Direction and Ply Sequence for the Bonded Rib Post, Figure 2, Item [1]
Figure 4

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Table 4:

PLY MATERIAL AND DIRECTION FOR FIGURE 4		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P5, P8, P10, P12	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P6, P7, P9, P11	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

REAR VIEW OF BONDED REAR SPAR

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 5 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96626 S0006595194_V1

Ply Direction and Ply Sequence for the Bonded Rear Spar, Figure 2, Item [2]
Figure 5

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Table 5:

PLY MATERIAL AND DIRECTION FOR FIGURE 5		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P6, P7, P10, P12	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P5, P8, P9, P11	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P13	Optional	Epoxy impregnated glass woven fabric as given in BMS 8-139, Class III, Style 108

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED OUTBOARD CLOSURE RIB

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 6 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96584 S0006595196_V1

Ply Direction and Ply Sequence for the Bonded Outboard Closure Rib, Figure 2, Item [3]
Figure 6

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Table 6:

PLY MATERIAL AND DIRECTION FOR FIGURE 6		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P4, P6	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P5	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED TAB CUTOUT RIB

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 7 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96760 S0006595198_V1

Ply Direction and Ply Sequence for the Tab Cutout Rib, Figure 2, Item [4]
Figure 7

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Table 7:

PLY MATERIAL AND DIRECTION FOR FIGURE 7		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P6, P8	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P5, P7	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED INBOARD CLOSURE RIB

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 8 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96763 S0006595200_V1

Ply Direction and Ply Sequence for the Inboard Closure Rib, Figure 2, Item [5]
Figure 8

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Table 8:

PLY MATERIAL AND DIRECTION FOR FIGURE 8		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P4, P6	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P5	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

BONDED ACTUATOR BACKUP RIB

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 9 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96710 S0006595202_V1

Ply Direction and Ply Sequence for the Actuator Backup Rib, Figure 2, Item [6]
Figure 9

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Table 9:

PLY MATERIAL AND DIRECTION FOR FIGURE 9		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P5, P8, P10, P12	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P6, P7, P9, P11	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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BONDED SUPPORT RIB

PLY LAYUP DIRECTION

A

PLY LAYUP SEQUENCE
A-A

NOTES

- THE PLY DIRECTION IS THE WARP DIRECTION OF THE FABRIC. REFER TO DETAIL A FOR THE 0 DEGREE PLY DIRECTION OF THE BONDED PART.
- REFER TO SECTION A-A FOR THE PLY SEQUENCE AT THAT LOCATION. REFER TO THE ENGINEERING DRAWING FOR MORE INFORMATION.
- REFER TO TABLE 10 FOR THE DIRECTION AND MATERIAL OF THE PLIES SHOWN IN SECTION A-A.

F96608 S0006595204_V1

Ply Direction and Ply Sequence for the Bonded Support Rib, Figure 2, Item [7]
Figure 10

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Table 10:

PLY MATERIAL AND DIRECTION FOR FIGURE 10		
PLY	**DIRECTION**	**MATERIAL**
P1, P3, P6, P8	+ or - 45 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW
P2, P4, P5, P7	0 or 90 degrees	CFRP woven fabric as given in BMS 8-256, Type IV, Class 2, Style 3K-70-PW

NOTE: The plies given above as + or - 45 degrees are all +45 degrees or all -45 degrees.

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IDENTIFICATION 2 - AILERON TAB STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

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Aileron Tab Structure Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly
113A7200	Tab Assembly - Aileron

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

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AILERON TAB ASSEMBLY

F96033 S0006595209_V1

Aileron Tab Structure Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Outboard Closure Rib	0.625 (15.9)	Phenolic Sheet as given in ASTM D709 (Optional: CIBA-GEIGY Resin ARALDITE and HY956 Hardener)	
[2]	Inboard Closure Rib	0.625 (15.9)	Phenolic Sheet as given in ASTM D709 (Optional: CIBA-GEIGY Resin ARALDITE and HY956 Hardener)	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - AILERON STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the aileron structure shown in Wing Aileron Structure Location, Figure 101/ALLOWABLE DAMAGE 1.

B. The allowable damage limits are only applicable if they are sealed as given in Paragraph 2.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Wing Aileron Structure Location
Figure 101

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LEFT SIDE UPPER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

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Aileron Structure
Figure 102 (Sheet 1 of 3)

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Aileron Structure
Figure 102 (Sheet 2 of 3)

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Aileron Structure
Figure 102 (Sheet 3 of 3)

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2. General

A. Do an inspection of the damaged area to find the length, width and depth of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to 737 NDT Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 103/ALLOWABLE DAMAGE 1, Details A , B , and C for the definitions of the length, width, and depth of damage.

B. Remove all contaminates and water from the structure. Refer to 51-30-05 and 51-70-04 for the tools and cleanup procedures.

C. Seal all permitted damage areas that are not more than one ply deep. Refer to the allowable damage limits. Seal the damage with one of the two methods that follow:

(1) Make a temporary seal.

(a) Apply aluminum foil tape (speed tape).

(b) Keep a record of the location.

(c) Make sure the tape is in satisfactory condition after each 400 flight hour interval or more frequently.

(d) Repair the damage at or before 5000 flight hours from the time the seal is made.

(2) Make a permanent seal.

(a) Apply BMS 8-207 or BMS 8-301 epoxy resin to the area as given in 51-70-08.

(b) Apply one layer of BMS 10-79, Type III or BMS 10-103, Type I primer. Refer to SOPM 20-44-04.

(c) Apply one layer of BMS 10-60, Type II enamel to the exterior surfaces of the areas sealed with epoxy resin. Refer to AMM PAGEBLOCK 51-21-99/701.

D. Seal all permitted damage areas that are more than one ply deep. Refer to the allowable damage limits. Seal the damage as follows:

(1) Use a vacuum and heat to remove moisture from the solid laminate. Refer to 51-70-04.

(2) Make a temporary seal with aluminum foil tape (speed tape).

(3) Keep a record of the location.

(4) Repair the damage at or before 400 flight hours from the time the seal was made.

E. The definition of the words "other damage" as used in the allowable damage limits, does not include nicks, gouges, and scratches that do not cause carbon fiber damage and are sealed.

F. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

G. Refer to Table 101/ALLOWABLE DAMAGE 1 for the references for the allowable damage limits of the different parts.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF STRUCTURE	**ZONE LOCATION**	**PARAGRAPH**
Ribs	---	4.A
Rear Spar	1	4.B
	2	4.C

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Table 101: (Continued)

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF STRUCTURE	ZONE LOCATION	PARAGRAPH
Front Spar	3	4.D
	4	4.E
	5	4.F

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Damage Definitions
Figure 103

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3. References

Reference	Title
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-60-01	AILERON BALANCING PROCEDURES
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-08	RESIN SWEEP-FAIR PROCEDURES
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Allowable Damage Limits

A. Ribs

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the chord radius and are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.5 inch (12.7 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(4) Dents are permitted if they are not in the chord radius and are:

(a) A maximum of 0.05 inch (1.27 mm) in depth

(b) A maximum of 2.0 inch (50.8 mm) in diameter

(c) A minimum of 3.00 inches (76.2 mm) away from other damage

(d) A minimum of 3.00 inches (76.2 mm) away from a rib post.

(5) Holes and Punctures are permitted that are not in the chord radius if they are:

(a) A maximum of 0.25 inch (6.4 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(6) Delamination is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1 if it is:

(a) Not in the chord radius

(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1 if it is:

(a) A maximum of 0.5 inch (12.7 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

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B. Rear Spar - Zone 1

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the radius and they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.5 inch (12.7 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are not in the radius and are:

1) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

4) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(5) Holes and Punctures are permitted if they are not in the radius and are:

(a) A maximum of 0.25 inch (6.4 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(6) Delamination is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) Not in the radius

(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.5 inch (12.7 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width.

(8) Edge Erosion is permitted as shown in Allowable Damage Limits, Figure 105/ALLOWABLE DAMAGE 1, Detail B .

C. Rear Spar - Zone 2

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

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(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

4) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(5) Holes and Punctures are permitted if they are:

(a) A maximum of 0.1 inch (2.54 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(6) Delamination is permitted as shown in Figure 104/ALLOWABLE DAMAGE 1, Detail A .

D. Front Spar - Zone 3

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the radius and are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are not in the radius and are:

1) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

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4) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(5) Holes and Punctures are permitted if they are not in the radius if and are:

(a) A maximum of 0.1 inch (2.54 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(6) Delamination is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) Not in the radius

(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.25 inch (6.35 mm) in width

(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(8) Edge Erosion is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail B .

E. Front Spar - Zone 4

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are not in the radius and are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are not in the radius and are:

1) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

4) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(5) Holes and Punctures are permitted if they are not in the radius and are:

(a) A maximum of 0.1 inch (2.54 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a fitting.

(6) Delamination is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) Not in the radius

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(b) A minimum of 3.00 inches (76.2 mm) away from a rib post or a fitting.

(7) Edge damage is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail A if it is:

(a) A maximum of 0.25 inch (6.35 mm) in depth

(b) A maximum of 0.25 inch (6.35 mm) in width

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(8) Edge Erosion is permitted as shown in Figure 105/ALLOWABLE DAMAGE 1, Detail B .

F. Front Spar - Zone 5

(1) Nicks, Gouges, and Scratches that cause damage to the fiberglass ply are permitted.

(2) Nicks, Gouges, and Scratches that do not cause damage to the carbon fibers are permitted.

(3) Nicks, Gouges, and Scratches that cause damage to the carbon fibers are permitted if they are:

(a) A maximum of one ply in depth

NOTE: Use the limits for holes and punctures if the damage is more than one ply in depth.

(b) A maximum of 0.25 inch (6.35 mm) in length

(c) A maximum of 0.25 inch (6.35 mm) in width

(d) A minimum of 3.00 inches (76.2 mm) away from other damage

(e) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(4) Dents:

(a) Dents that do not cause damage to the carbon fibers are permitted if they are:

1) A maximum of 0.05 inch (1.27 mm) in depth

NOTE: Use the limits for holes and punctures if there is carbon fiber damage or if the dent depth is more than 0.05 inch (1.27 mm).

2) A maximum of 0.50 inch (12.7 mm) in diameter

3) A minimum of 3.00 inches (76.2 mm) away from other damage

4) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(5) Holes and Punctures are permitted if they are:

(a) A maximum of 0.1 inch (2.54 mm) in diameter

(b) A minimum of 3.00 inches (76.2 mm) away from other damage

(c) A minimum of 3.00 inches (76.2 mm) away from a rib post or fitting.

(6) Delamination is permitted as shown in Figure 104/ALLOWABLE DAMAGE 1, Detail A .

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.00 INCHES (76.2 mm).

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

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Allowable Damage Limits
Figure 104

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NOTE: TO FIND DELAMINATION, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

THE DIAMETER OF A DAMAGE AREA IS EITHER THE DIAMETER OF THE VISUAL DAMAGE OR THE DIAMETER OF THE DELAMINATION. USE THE DIAMETER OF THE LARGER DAMAGE.

D IS THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS AND CAN BE A MAXIMUM OF 0.50 INCH (12.7 mm).

d IS THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

IS THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

THE MINIMUM X THAT IS PERMITTED IS 3.00 INCHES (76.2 mm).

DAMAGE THAT IS PERMITTED TO COMPOSITE PANELS

A

CLEANUP AND SEALING OF EDGE EROSION

B

F91163 S0006595220_V1

Allowable Damage Limits
Figure 105

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ALLOWABLE DAMAGE 2 - AILERON TAB STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the aileron tab structure shown in Aileron Tab Structure Location, Figure 101/ALLOWABLE DAMAGE 2.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Aileron Tab Structure Location
Figure 101

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UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A-A

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Aileron Tab Structure
Figure 102

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2. General

A. Do the steps that follow.

(1) Removed the damaged material.

(a) Refer to 51-10-02 for the procedures.

(b) Refer to 51-30-03 for the sources of non-metallic materials you can use to remove the damage.

(c) Refer to 51-30-05 for the sources of equipment and tools you can use to remove the damage.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply one layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Nose Spar

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Details A , B , C , D , and E .

(3) Dents are permitted as shown in Allowable Damage Limits, Figure 103/ALLOWABLE DAMAGE 2, Detail F .

(4) Holes and Punctures:

(a) Damage is permitted if it is:

1) A maximum of 0.25 inch in diameter

2) A minimum of 4D (D = The diameter of the hole or puncture) from a hole, edge of the part, or other damage

3) Filled with a 2117-T3 or 2117-T4 protruding head rivet installed without sealant.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE

E

F87775 S0006595228_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

REMOVAL OF DAMAGE AROUND THE FASTENERS ON AN EDGE OR A SURFACE

E

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Allowable Damage Limits
Figure 103 (Sheet 2 of 3)

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Allowable Damage Limits
Figure 103 (Sheet 3 of 3)

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REPAIR 1 - AILERON STRUCTURE

1. Applicability

A. Repair 1 is applicable to the front and rear spars of the aileron shown in Wing Aileron Structure Repair, Figure 201/REPAIR 1.

(1) Boeing has not found it necessary to supply repairs to the ribs of the aileron in the Structural Repair Manual (SRM) at this time.

B. Repair 1 is applicable to damage that is more than the limits permitted in Allowable Damage 1. Refer to Allowable Damage 1 for the type and size of damage that is permitted.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

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Wing Aileron Structure Repair
Figure 201

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LEFT SIDE UPPER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

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Aileron Structure Repair
Figure 202

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2. General

A. REPAIR 1 gives instructions for Category A repairs. Category B repairs are not permitted for REPAIR 1. Refer to 51-00-06 to find the definitions of the different categories of repairs.

B. Remove the aileron as necessary. Refer to AMM 27-11-11/401.

C. Remove the skin as necessary to get access to the spar.

(1) If a fastener is damaged, refer to 51-70-04 or 51-70-05 for a repair.

D. Remove the fitting as necessary to get access to the damaged area. Refer to SOPM 20-10-08.

E. Do an inspection of the damaged area to find the dimensions of the damage. Boeing recommends that you use an instrumented Non-Destructive Test (NDT) procedure. Refer to NDT, Part 1, 51-01-02 for inspection procedures.

NOTE: Other inspection methods that have been examined and found to be satisfactory by the operator can be used.

(1) Refer to Damage Definitions, Figure 203/REPAIR 1 for the definitions of the diameter and depth of damage.

F. Do the repair as given in Paragraph 4./REPAIR 1

G. Install the fittings if they were removed.

(1) Apply BMS 5-95 sealant to:

(a) The reworked areas of the spar and fittings

(b) The edges around the fittings

(c) All parts that connect to the spar and fittings

(d) All gaps.

(2) Apply BMS 10-79, Type III primer to the edges around the fittings. Refer to SOPM 20-44-04.

(3) Apply BMS 10-60, Type II primer to the edges around the fittings. Refer to AMM PAGEBLOCK 51-21-99/701.

H. Make sure that the aileron is balanced. Refer to 51-60-01 for the balance procedures.

I. Install the aileron, as applicable. Refer to AMM 27-11-11/401.

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DEFINITIONS FOR NICK,
GOUGE, OR SCRATCH DAMAGE

A

DEFINITIONS FOR
DENT DAMAGE

B

DEFINITIONS FOR
EDGE DAMAGE

C

H28341 S0006595235_V1

Damage Definitions
Figure 203

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NOTE: THE EDGE OF THE HINGE AREAS ALIGN WITH THE EDGE OF THE HINGE FITTINGS.

REAR SPAR

H28342 S0006595236_V1

Aileron Structure Repair Zones - Rear Spar
Figure 204 (Sheet 1 of 3)

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H28343 S0006595237_V1

Aileron Structure Repair Zones - Rear Spar
Figure 204 (Sheet 2 of 3)

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H28346 S0006595238_V1

Aileron Structure Repair Zones - Rear Spar
Figure 204 (Sheet 3 of 3)

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H28347 S0006595239_V1

Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 1 of 8)

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H28349 S0006595240_V1

Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 2 of 8)

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Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 3 of 8)

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H28353 S0006595242_V1

Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 4 of 8)

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H28354 S0006595243_V1

Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 5 of 8)

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Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 6 of 8)

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Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 7 of 8)

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H28359 S0006595246_V1

Aileron Structure Repair Zones - Front Spar
Figure 205 (Sheet 8 of 8)

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3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-01	AERODYNAMIC SMOOTHNESS
51-20-05	REPAIR SEALING
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-60-01	AILERON BALANCING PROCEDURES
51-70-04	REPAIR PROCEDURES FOR WET LAYUP MATERIALS
51-70-05	REPAIR PROCEDURES FOR PREIMPREGNATED MATERIALS
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
AMM 27-11-11/401	Aileron - Removal/Installation
SOPM 20-10-08	Removal of Faying Surface Sealed Metal Fittings from Composite Structures
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 1, 51-01-02	NDT Examination of Composite Structure for Impact Damage

4. Repair Instructions

A. If a dent is 2.00 inches in diameter or less, and has no fiber damage or delamination, then do as follows:

(1) Fill the dent with potting compound.

(2) Apply a fiberglass patch as given in Repair 14 of 51-70-04.

B. Refer to the Tables that follow for all other damage:

(1) Table 201/REPAIR 1 for the repair data that is applicable to damage to Zone 1 of the Front and Rear Spars

(2) Table 202/REPAIR 1 for the repair data that is applicable to damage to Zone 2 of the Front and Rear Spars

(3) Table 203/REPAIR 1 for the repair data that is applicable to damage to Zone 3 of the Front and Rear Spars

(4) Table 204/REPAIR 1 for the repair data that is applicable to damage to Zone 4 of the Front and Rear Spars

(5) Table 205/REPAIR 1 for the repair data that is applicable to damage to Zone 5 of the Front and Rear Spars.

C. Do as follows when you make a repair:

(1) When you remove the damage, do not cut or make an abrasion into the radius of the structure.

(2) If the repair plies make an overlap of a hole or cutout, do the steps that follow:

(a) Cure the repair.

(b) Drill or cut the plies to the initial diameter of the hole or cutout.

(3) If you need clearance with adjacent structure, install the tapered shim on each side of the repair.

(4) It is permitted to put the repair plies around the full width of the structure.

(a) Do not make an overlap of the edges of the structure.

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D. For repairs made with wet layup materials that are cured at 200° (93°C), do as follows:

(1) Use one repair ply of fabric for each initial ply that was removed from the damaged area.

(2) Add three structural plies of fabric. Put one structural ply at ±45 degrees and the other at 0 or 90 degrees.

NOTE: Repair plies or added plies are not necessary for edge delamination that is more than 2D (D= fastener diameter) away from a fastener hole.

(3) Add a ply of fiberglass.

E. For repairs made with preimpregnated layup materials that are cured at 250°F (121°C), do as follows:

(1) Use one repair ply of fabric for each initial ply that was removed from the damaged area.

(2) Add three structural plies of fabric. Put one structural ply at ±45 degrees and the other at 0 or 90 degrees.

NOTE: Repair plies or added plies are not necessary for edge delamination that is more than 2D (D= fastener diameter) away from a fastener hole.

(3) Add a ply of fiberglass.

F. For repairs made with preimpregnated layup materials that are cured at 350° (177°C), do as follows:

(1) Use one repair ply of fabric for each initial ply that was removed from the damaged area.

(2) Add a ply of fiberglass.

(3) If the damage applies to more than one zone then:

(a) The repair must not be more than the limits for each zone

(b) Do the repair instructions that are applicable to all zones.

Table 201:

REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 1				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	A repair is not permitted	Damage that is a maximum of: - 4.0 inches in length - 65 percent of the smallest dimension across the panel	Damage that is a maximum of: - 4.0 inches in length - 65 percent of the smallest dimension across the panel	There are no limits on the dimensions of the repair
REPAIR PROCEDURES	-----	SRM 51-70-04 and Paragraphs 4.C and 4.D	SRM 51-70-05 and Paragraphs 4.C and 4.E	SRM 51-70-05 and Paragraphs 4.C and 4.F

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Table 202:

REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 2				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY B WET LAYUP**	**CATEGORY B PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	A repair is not permitted	A repair is not permitted	A repair is not permitted	There are no limits on the dimensions of the repair
REPAIR PROCEDURES	----	----	----	SRM 51-70-05 and Paragraph 4.F

Table 203:

REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 3				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY A WET LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	A repair is not permitted	Damage that is a maximum of: - 4.0 inches in length - 65 percent of the smallest dimension across the panel	A repair is not permitted	There are no limits on the dimensions of the repair
REPAIR PROCEDURES	----	SRM 51-70-04 and Paragraph 4.C	----	SRM 51-70-05 and Paragraph 4.F

Table 204:

REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 4				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY B WET LAYUP**	**CATEGORY B PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	A repair is not permitted	A repair is not permitted	A repair is not permitted	There are no limits on the dimensions of the repair
REPAIR PROCEDURES	----	----	----	SRM 51-70-05 and Paragraphs 4.C and 4.F

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Table 205:

REPAIR DATA FOR 350°F (177°C) CURE AILERON FRONT AND REAR SPARS IN ZONE 5				
REPAIR TYPE	**CATEGORY B WET LAYUP**	**CATEGORY B WET LAYUP**	**CATEGORY B PREIMPREGNATED LAYUP**	**CATEGORY A PREIMPREGNATED LAYUP**
REPAIR CURE TEMPERATURE	**150°F (66°C)**	**200°F (93°C)**	**250°F (121°C)**	**350°F (177°C)**
REPAIR SIZE AND LIMITS	A repair is not permitted	A repair is not permitted	A repair is not permitted	There are no limits on the dimensions of the repair
REPAIR PROCEDURES	----	----	----	SRM 51-70-05 and Paragraph 4.F

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REPAIR 2 - AILERON TAB STRUCTURE

NOTE: THERE ARE NO REPAIRS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F93825 S0006595253_V2

Wing Aileron Tab Structure Location
Figure 201

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ALLOWABLE DAMAGE 1 - WING AILERON FAIRING

NOTE: THERE IS NO ALLOWABLE DAMAGE FOR THIS PART IN THE STRUCTURAL REPAIR MANUALAT THIS TIME.

LEFT SIDE LOWER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A

G03327 S0006595263_V2

Wing Aileron Fairing
Figure 101

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REPAIR 1 - WING AILERON FAIRING

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE LOWER SURFACE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A

G03338 S0006595266_V2

Wing Aileron Fairing
Figure 201

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IDENTIFICATION 1 - AILERON FITTINGS

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H12631 S0006595270_V1

Aileron Fitting
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A0003	Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (LH)
113A0004	Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (RH)
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly
113A7100	Aileron Assembly
113A7101	Front Spar Assembly - Aileron
113A7400	Aileron Assembly

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H13153 S0006595272_V1

Aileron Fitting Locations
Figure 2

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H13161 S0006595273_V1

Aileron Front Spar Hinge Fittings Identification
Figure 3

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Table 2:

LIST OF MATERIALS FOR FIGURE 3				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Actuator Fitting		Ti-6Al-4V titanium investment casting as given in BMS 7-310, Grade B, annealed	
[2]	Hinge Half (4 pairs)		15-5PH CRES investment casting as given in AMS 5346, heat treated to 180 KSI	
[3]	Hinge Fitting (Number 1 thru Number 5)		7050-T7451 plate as given in AMS 4050. Grain direction controlled part	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H39357 S0006595275_V1

Aileron Rear Spar Hinge Fittings Identification
Figure 4

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Table 3:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Hinge Fitting (5)		A356.0-T6 aluminum casting as given in MIL-A-21180, Class 2	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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IDENTIFICATION 2 - AILERON TAB FITTINGS

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H39373 S0006595278_V1

Aileron Tab Fitting Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A0003	Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (LH)
113A0004	Flap Supports, Aileron, and Spoilers - Trailing Edge Collector (RH)
113A7001	Aileron Installation
113A7002	Aileron/Tab Assembly
113A7200	Tab Assembly - Aileron

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

H40085 S0006595280_V1

Aileron Tab Hinge Fitting Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Tab Hinge Fitting (3)		A356.0-T6 aluminum casting as given in MIL-A-21180, strength Class 2	
[2]	Tab Actuator Fitting		A356.0-T6 aluminum casting as given in MIL-A-21180, strength Class 2	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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ALLOWABLE DAMAGE 1 - AILERON HINGE FITTINGS

1. Applicability

A. This subject gives the allowable damage limits for the Wing Aileron Hinge Fittings shown in Aileron Fitting Locations, Figure 101/ALLOWABLE DAMAGE 1.

G08352 S0006595284_V1

Aileron Fitting Locations
Figure 101

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G08664 S0006595285_V1

Aileron Front Spar Fitting
Figure 102 (Sheet 1 of 2)

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A

B

C

G08706 S0006595286_V1

Aileron Front Spar Fitting
Figure 102 (Sheet 2 of 2)

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G08705 S0006595287_V1

Aileron Rear Spar Hinge Fitting
Figure 103 (Sheet 1 of 2)

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A

G09638 S0006595288_V1

Aileron Rear Spar Hinge Fitting
Figure 103 (Sheet 2 of 2)

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2. General

A. Do the steps that follow if you have damage.

(1) Remove the damage. Refer to 51-10-02 for the investigation and cleanup procedures.

(2) Refer to 51-30-03 for the sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for the sources of the equipment and tools you can use to remove the damage.

B. Do the steps that follow after you remove the damage.

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the parts, but not the inner surfaces of the lug bores.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
AMM 51-21-99	DECORATIVE EXTERIOR PAINT SYSTEM
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Hinge Fittings (Machined Aluminum)

(1) Cracks:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , and C .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits, Figure 104/ALLOWABLE DAMAGE 1, Details A , B , , D , and E .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

G09472 S0006595289_V1

Allowable Damage Limits
Figure 104 (Sheet 1 of 3)

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X = WIDTH OF THE MATERIAL REMOVED
= A MAXIMUM OF 10 PERCENT OF THE WIDTH OF THE FLANGE

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

G09473 S0006595290_V1

Allowable Damage Limits
Figure 104 (Sheet 2 of 3)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

E

G09474 S0006595291_V1

Allowable Damage Limits
Figure 104 (Sheet 3 of 3)

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ALLOWABLE DAMAGE 2 - AILERON TAB HINGE FITTINGS

1. Applicability

A. This subject gives the allowable damage limits for the aileron tab hinge fittings shown in Aileron Tab Skin Panel Location, Figure 101/ALLOWABLE DAMAGE 2.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

G05243 S0006595293_V1

Aileron Tab Skin Panel Location
Figure 101

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UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

A

G05428 S0006595294_V1

Aileron Tab Hinge Fitting
Figure 102

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2. General

A. Refer to Paragraph 4./ALLOWABLE DAMAGE 2 for the allowable damage limits.

B. Remove the damage as necessary.

(1) Refer to 51-10-02 for the investigation and removal of damage.

(2) Refer to 51-30-03 for possible sources of non-metallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tolls you can use to remove the damage.

C. After you remove the damage, do the steps that follow:

WARNING: MAKE SURE THAT YOU WEAR EYE PROTECTION WHEN YOU USE THE FLAP PEEN WHEEL. IF YOU DO NOT OBEY, AN INJURY CAN OCCUR.

(1) Flap peen or shot peen the reworked areas of the bullnose links, but not the inner surfaces of the link holes.

(a) Refer to 51-20-06 for the shot peen intensity and shot number.

(b) Refer to SOPM 20-10-03 for the flap peen and shot peen procedures.

(2) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(3) Apply a layer of BMS 10-79, Type III primer to the reworked areas. Refer to SOPM 20-44-04.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-06	SHOT PEENING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
SOPM 20-10-03	SHOT PEENING
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-44-04	Application of Urethane Compatible Primer

4. Allowable Damage Limits

A. Hinge Fittings

(1) Cracks:

(a) Remove the damage as shown in Aileron Tab Hinge Fitting, Figure 102/ALLOWABLE DAMAGE 2, Details A and .

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Aileron Tab Hinge Fitting, Figure 102/ALLOWABLE DAMAGE 2, Details A , B , C , and D .

(3) Dents are not permitted.

(4) Holes and Punctures are not permitted.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

REMOVAL OF DAMAGED MATERIAL ON A SURFACE

C

G05431 S0006595295_V1

Allowable Damage Limits
Figure 103 (Sheet 1 of 2)

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NOTE: DAMAGED SEALANT IS NOT PERMITTED. IF THE SEALANT IS DAMAGED, LOOK FOR MIGRATION OR ROTATION OF THE BUSHING. IF THERE IS NO MIGRATION, ROTATION, OR CORROSION, REMOVE THE DAMAGED SEALANT AND APPLY A NEW FILLET SEAL.

REMOVAL OF SURFACE AND EDGE DAMAGE FROM A LUG THAT HAS A BUSHING

D

G05432 S0006595296_V1

Allowable Damage Limits
Figure 103 (Sheet 2 of 2)

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REPAIR 1 - AILERON HINGE FITTINGS

NOTE: THERE ARE NO REPAIRS FOR THESE PARTS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

G10148 S0006595299_V2

Aileron Fitting Locations
Figure 201

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REPAIR 2 - AILERON TAB HINGE FITTINGS

UPPER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

NOTE: THERE ARE NO REPAIRS IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

G05437 S0006595301_V2

Aileron Tab Hinge Fitting Repair
Figure 201

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GENERAL - SPOILER DIAGRAM

NOTE: FOR TRAILING EDGE, SEE SRM 57-51-00

FOR TRAILING EDGE FLAPS, SEE SRM 57-53-00

FOR AILERONS, SEE SRM 57-60-00

F87597 S0006595306_V1

Spoiler Diagram
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
110A1502	Structures Diagram, Wing - Model 737X
113A4000	Drawing Index - Spoilers
113A4001	Spoiler Installation

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IDENTIFICATION 1 - INBOARD SPOILER SKIN

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F77939 S0006595311_V1

Inboard Spoiler Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A4001	Spoiler Installation
113A4600	Spoiler Assembly - 6 and 7, Inboard Ground

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NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F77948 S0006595313_V1

Inboard Spoiler Number 6 and 7 Skin Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Spoiler Assembly - Bonded Panel			
	Skin - Upper	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - lower	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - Upper	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - lower	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 3 for the core ribbon direction	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A

A–A

NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

F75117 S0006595315_V1

Core Ribbon Direction for Figure 2, Item [1]
Figure 3

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IDENTIFICATION 2 - OUTBOARD SPOILER SKINS

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F78025 S0006595317_V1

Outboard Spoiler Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A4001	Spoiler Installation
113A4100	Spoiler Assembly - 1 and 12, Outboard Ground
113A4110	Panel Bond Assembly - Spoiler 1 and 12
113A4200	Spoiler Assembly - 2 and 11, Flight
113A4210	Panel Bond Assembly - Spoiler 2 and 11
113A4300	Spoiler Assembly - 3 and 10, Flight
113A4310	Panel Bond Assembly - Spoiler 3 and 10

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A4400	Spoiler Assembly - 4 and 9, Flight
113A4410	Panel Bond Assembly - Spoiler 4 and 9
113A4500	Spoiler Assembly - 5 and 8, Flight
113A4510	Panel Bond Assembly - Spoiler 5 and 8

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NOTES

- REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F79115 S0006595319_V1

Outboard Spoiler No. 1 and 12 Skin Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Spoiler Assembly, Numbers 1 and 12 Bonded Panel			
	Doubler - Lower	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - Upper	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Lower	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Upper	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 3 for the core ribbon direction	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

F78727 S0006595321_V1

Core Ribbon Direction for Figure 2, Item [1]
Figure 3

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NOTES

- REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F78544 S0006595322_V1

Outboard Spoiler Number 2 and 11 Skin Identification
Figure 4

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Table 3:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Spoiler Assembly, Number 2 and 11 Bonded Panel			
	Doubler - lower	0.032 (0.812)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - Upper	0.032 (0.812)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - lower	0.032 (0.812)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Upper	0.032 (0.812)	2024-T3 sheet as given in QQ-A-250/4	
	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 5 for the core ribbon direction	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

F80268 S0006595324_V1

Core Ribbon Direction for Figure 4, Item [1]
Figure 5

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NOTES

- REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERNCE)

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F79724 S0006595325_V1

Outboard Spoiler Number 3 and 10 Skin Identification
Figure 6

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Table 4:

LIST OF MATERIALS FOR FIGURE 6				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Spoiler Assembly, Number 3 and 10 Bonded Panel			
	Doubler - Lower	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - Upper	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Lower	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Upper	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 7 for the core ribbon direction	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

F79803 S0006595327_V1

Core Ribbon Direction for Figure 6, Item [1]
Figure 7

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NOTES

- REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F79726 S0006595328_V1

Outboard Spoiler Number 4 and 9 Skin Identification
Figure 8

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Table 5:

LIST OF MATERIALS FOR FIGURE 8				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Spoiler Assembly, Number 4 and 9 Bonded Panel			
	Doubler - lower	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - Upper	0.020 (0.058)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - lower	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Upper	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 9 for the core ribbon direction	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

F79808 S0006595330_V1

Core Ribbon Direction for Figure 8, Item [1]
Figure 9

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NOTES

- REFER TO TABLE 6 FOR THE LIST OF MATERIALS.

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F79741 S0006595331_V1

Outboard Spoiler Number 5 and 8 Skin Identification
Figure 10

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Table 6:

LIST OF MATERIALS FOR FIGURE 10				
ITEM	**DESCRIPTION**	**T*[1]**	**MATERIAL**	**EFFECTIVITY**
[1]	Spoiler Assembly, Number 5 and 8 Bonded Panel			
	Doubler - lower	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Doubler - Upper	0.020 (0.508)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - lower	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Skin - Upper	0.016 (0.406)	2024-T3 sheet as given in QQ-A-250/4	
	Core		5052 aluminum honeycomb core as given in BMS 4-4, Type 3-10N. Refer to Figure 11 for the core ribbon direction.	

*[1] Note: T = Pre-manufactured thickness in inches (millimeters).

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NOTES

1 0.005 INCH DEPRESSION ON THE UPPER SURFACE (REFERENCE)

F79863 S0006595333_V1

Core Ribbon Direction for Figure 10, Item [1]
Figure 11

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ALLOWABLE DAMAGE 1 - INBOARD AND OUTBOARD SPOILER SKINS

1. Applicability

A. Allowable Damage 1 is applicable to damage on the inboard and outboard spoiler skins shown in Inboard and Outboard Spoiler Skin Allowable Damage, Figure 101/ALLOWABLE DAMAGE 1 and given in Table 101/ALLOWABLE DAMAGE 1.

Table 101:

PARAGRAPH REFERENCES FOR THE ALLOWABLE DAMAGE LIMITS		
TYPE OF STRUCTURE	ZONE LOCATION	PARAGRAPH
ALUMINUM SKIN AREAS	1	4.A
	2	4.B
	3	4.C

NOTE: REFER TO TABLE 101 FOR THE ALLOWABLE DAMAGE LIMITS PARAGRAPH THAT IS APPLICABLE TO EACH ZONE.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F79812 S0006595337_V1

Inboard and Outboard Spoiler Skin Allowable Damage
Figure 101

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LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 0.005 INCH DEPRESSION (REFERENCE)

F79951 S0006595338_V1

Inboard Spoiler No.6 and 7 Skin Allowable Damage
Figure 102

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LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

F80295 S0006595339_V1

Outboard Spoiler No.1 and 12 Skin Allowable Damage
Figure 103

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LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

F80439 S0006595340_V1

Outboard Spoiler No.2 and 11 Skin Allowable Damage
Figure 104

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LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

F80311 S0006595341_V1

Outboard Spoiler No.3 and 10 Skin Allowable Damage
Figure 105

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LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

F80377 S0006595342_V1

Outboard Spoiler No.4 and 9 Skin Allowable Damage
Figure 106

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LEFT SIDE LOWER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

LEFT SIDE UPPER SKIN PANEL IS SHOWN, RIGHT SIDE IS OPPOSITE

ZONE 1 ZONE 2 ZONE 3

NOTES

- ALL DIMENSIONS ARE IN INCHES

1 A DEPRESSION OF 0.005 INCH MAXIMUM IS PERMITTED ON THE UPPER SURFACE IN THIS AREA.

2 0.005 INCH DEPRESSION (REFERENCE)

F80400 S0006595343_V1

Outboard Spoiler No.5 and 8 Skin Allowable Damage
Figure 107

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2. General

A. Do the steps that follow if you have damage to the aluminum skins.

(1) Remove the FOD prevention tape prior to any damage removal or inspections if installed on the lower surface of the spoiler.

(2) Remove the damaged material.

(a) Refer to 51-10-02 for the procedures.

(b) Refer to 51-30-03 for the sources of nonmetallic materials you need to remove the damage.

(c) Refer to 51-30-05 for the sources of the equipment and tools you need to remove the damage.

(3) Apply a chemical conversion coating to the bare surfaces of the aluminum. Refer to 51-20-01.

(4) Apply one layer of BMS 10-11, Type I primer to the bare surfaces of the aluminum. Refer to SOPM 20-41-02.

B. Do the steps that follow if you remove damage from the skin and honeycomb core:

(1) Remove 0.25 inch of the honeycomb core at the edge of the cutout to make a recess.

(2) Fill the recess with BMS 5-28, Type 15 or 17 potting compound. Refer to 51-70-10, Repair General.

(3) As an alternative, fill the recess with BMS 5-95 sealant. Refer to 51-20-05.

3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
51-70-01	REPAIRS FOR MINOR DENTS IN METALLIC SHEET MATERIALS
51-70-10	ALUMINUM HONEYCOMB STRUCTURE REPAIR PROCEDURES
57-70-02, ALLOWABLE DAMAGE 1	Inboard and Outboard Spoiler Structure
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Zone 1 - Skin Panel Surface

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 1, Figure 108/ALLOWABLE DAMAGE 1, Detail A if the largest dimension of the damage is not more than 3.0 inches.

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(3) Dents:

(a) Shallow dents are permitted as given in Paragraph 4.D./ALLOWABLE DAMAGE 1.

(b) The damage is permitted as shown in Allowable Damage Limits for Zone 1, Figure 108/ALLOWABLE DAMAGE 1, Detail B if:

1) The depth of the damage is a maximum of 0.05 inch

2) The maximum diameter of the damage is 0.50 inch

3) The damage area is not at an attachment to the spoiler structure.

(c) If the damage depth is more than 0.05 inch and less than 0.25 inch, do the steps that follow:

1) Fill or rework the dent as given in 51-70-01.

2) If you fill the dent, do the steps that follow:

a) Seal the damage with aluminum foil tape (speed tape 3M-436 or the equivalent).

b) Make an inspection of the dent at each 400 flight hour interval, or more frequently.

- Make sure the tape is in satisfactory condition.
- Repair the dent if the damage becomes larger.
- If the damage has not become larger, repair the damage in 5000 flight hours or less.

(4) Holes and Punctures are not permitted.

(5) Disbond is not permitted.

B. Zone 2 - Skin Panel Surface

(1) Cracks are not permitted.

(2) Nicks, Gouges, Scratches, and Corrosion:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 2, Figure 109/ALLOWABLE DAMAGE 1, Detail A if the condition that follows is true:

1) The largest dimension of the damage is not more than 25 percent of the length of the panel chord.

(b) Scratches are permitted if the conditions that follow are true:

1) Two or more scratches do not go across each other

2) The damage is not less than 1.50 inches away from:

a) A material edge

b) A potted area

c) An attachment to the spoiler structure.

(3) Dents:

(a) Shallow dents are permitted as given in Paragraph 4.D./ALLOWABLE DAMAGE 1.

(b) The damage is permitted as shown in Allowable Damage Limits for Zone 2, Figure 109/ALLOWABLE DAMAGE 1, Detail B if:

1) The depth of damage is a maximum of 0.10 inch

2) The maximum diameter of the damage is 25 percent of the panel chord

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3) The distance between two dents, or between a dent and other damage is a minimum of 0.5D (D = the large dimension of the two damage areas)

4) The damage area is not at an attachment to the spoiler structure.

(c) If the damage depth is more than 0.10 inch and less than 0.25 inch, do the steps that follow:

1) Fill or rework the dent as given in 51-70-01.

2) If you fill the dent, do the steps that follow:

a) Seal the damage with aluminum foil tape (speed tape 3M-436 or the equivalent).

b) Make an inspection of the dent at each 400 flight hour interval, or more frequently.

- Make sure the tape is in satisfactory condition.
- Repair the dent if the damage becomes larger.
- Repair the damage in 5000 flight hours or less.

(4) Holes and Punctures:

(a) The damage is permitted up to a maximum length of 15 percent of the length of the panel chord if you do as follows:

1) Trim the skin around the damage to a smooth circular or oval shape.

2) Clean and remove the water from the damaged area.

3) Apply aluminum foil tape (speed tape 3M-436 or the equivalent).

a) Keep a record of the damage location.

b) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

c) Replace the tape as necessary.

d) Repair the damage if the damage becomes larger.

e) Repair the damage in 5000 flight hours or less.

(5) Disbond:

(a) The damage is permitted as shown in Allowable Damage Limits for Zone 2, Figure 109/ALLOWABLE DAMAGE 1, Detail C if you do as follows:

1) Make an inspection of the damage at each 400 flight hour interval, or more frequently.

a) Repair the damage in 5000 flight hours or less.

C. Zone 3 - Skin Panel Edges

(1) Cracks:

(a) For panel edges other than the trailing edge:

1) Remove the damage as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Details A and B.

2) Cracks are not permitted aft of the spar.

(b) For the trailing edge of the panel:

1) The length of the damage must not be more than 2.0 inches.

2) The damage must not be less than 5D (D = largest damage dimension) from other damage.

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3) Remove water from the damage area and seal as follows:
   a) Apply one layer of speed tape (3M-436 or the equivalent) that extends 2.0 inches all around the damage.
   b) Apply a second layer of speed tape that extends 1.0 inch all around the first layer of speed tape.
4) Make an inspection of and repair the damage.
   a) Keep a record of the damage location.
   b) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.
   c) Repair the damage in 5000 flight hours or less.

(2) Nicks, Gouges, Scratches, and Corrosion:
   (a) Remove the damage as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Details A and B.
      1) The damage must not be less than 1.50 inches from a fastener hole.

(3) Dents:

NOTE: If there is damage to the spoiler structure, you must refer to 57-70-02, ALLOWABLE DAMAGE 1.

NOTE: If disbond is found at a dent location, use the limits for disbond.

   (a) Shallow dents are permitted as given in Paragraph 4.D./ALLOWABLE DAMAGE 1.
   (b) For panel edges other than the trailing edges, the damage is permitted as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail C if:
      1) The largest dimension of damage is 15 percent of the length of the panel chord
      2) The largest dimension of damage is 0.125 inch in depth
      3) The distance between two dents, or a dent and other damage, is not less than 0.5D (D = The larger dimension of the two damage areas)
   (c) For the trailing edge of the panels, the damage is permitted as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail C if:
      1) The largest dimension of damage is 15 percent of the length of the panel span.
      2) The largest dimension of damage is 0.10 inch in depth.
      3) The distance between two dents, or a dent and other damage, is not less than 0.5D (D = The larger dimension of the two damage areas)

(4) Holes and Punctures:
   (a) The damage is permitted as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail D.
   (b) Apply aluminum foil tape (speed tape 3M-436 or the equivalent).
      1) Keep a record of the damage location.
      2) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.
      3) Repair the damage in 5 flight hours or less if the damage becomes larger.
      4) Replace the tape as necessary.
      5) Repair the damage in 5000 flight hours or less if the damage has not become larger.

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(5) Disbond:

(a) Remove the damage as shown in Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail E with the limits that follow:

1) The length of the damage must not be more than 2.0 inches.

2) The width of the damage must not be more than 1.50 inches.

3) The damage is not less than 5D (D = largest damage dimension) from other damage.

(b) Seal the damage as follows:

NOTE: Refer to Allowable Damage Limits for Zone 3, Figure 110/ALLOWABLE DAMAGE 1, Detail E for the definition of the damage dimensions.

1) Apply one layer of speed tape (3M-436 or the equivalent) that extends 2.0 inches all around the damage.

2) Apply a second layer of speed tape that extends 1.0 inch all around the first layer of speed tape.

(c) Make an inspection of and repair the damage.

1) Keep a record of the damage location.

2) Remove the tape and examine the damage at each 150 flight hour interval, or more frequently.

3) Repair the damage if the damage becomes larger.

4) Repair the damage in 5000 flight hours or less if the damage has not become larger.

D. Zone 1, 2, 3 - Shallow Dents

(1) The damage is permitted as shown in Allowable Damage Limits for Shallow Dents, Figure 111/ALLOWABLE DAMAGE 1 if:

(a) The depth of each dent is a maximum of 0.020 in. (0.508 mm).

(b) W/Y is greater than 30.

(c) The diameter of damage is a maximum of 3.00 in. (76.20 mm). Multiple shallow dents may be grouped together with no minimum spacing.

(d) The total dented area is less than 40% for each square area of 144 in^2 (929 cm^2).

NOTE: Include the dented area of all dents on the spoiler skin when you sum the area. Use the criteria from Paragraph 4.A.(3)/ALLOWABLE DAMAGE 1, Paragraph 4.B.(3)/ALLOWABLE DAMAGE 1, and Paragraph 4.C.(3)/ALLOWABLE DAMAGE 1 when you calculate the area of dents that do not meet the shallow dent criteria.

(e) The dent is smooth and has no creases, gouges, or cracks.

(f) There are no pulled or loose fasteners.

(g) There are no disbonds.

(h) There is at least 1D spacing to fasteners.

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

(A)

DENT THAT IS PERMITTED

(B)

F79956 S0006595344_V2

Allowable Damage Limits for Zone 1
Figure 108

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

(A)

DENT THAT IS PERMITTED

(B)

F83862 S0006595345_V2

Allowable Damage Limits for Zone 2
Figure 109 (Sheet 1 of 2)

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NOTE: TO FIND DISBOND, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

A = THE AREA OF THE DAMAGE.

D = THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

d = THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X = THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

DAMAGE THAT IS PERMITTED ON THE SURFACE OF
AN ALUMINUM HONEYCOMB PANEL

C

F83888 S0006595346_V2

Allowable Damage Limits for Zone 2
Figure 109 (Sheet 2 of 2)

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

F79960 S0006595347_V1

Allowable Damage Limits for Zone 3
Figure 110 (Sheet 1 of 5)

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$\frac{W}{Y}$ MUST BE 10 OR MORE

L = LARGEST DIMENSION OF THE DENT
= A MAXIMUM OF 15% OF THE PANEL SPAN FOR THE TRAILING EDGE
= A MAXIMUM OF 15% OF THE PANEL CHORD FOR THE PANEL EDGES EXCEPT THE TRAILING EDGE

Y = MAXIMUM DEPTH OF DENT
= A MAXIMUM OF 0.10 INCH FOR THE TRAILING EDGE OF THE PANEL
= A MAXIMUM OF 0.125 INCH FOR THE PANEL EDGES EXCEPT THE TRAILING EDGE

NOTE: DENT MUST BE INSIDE ZONE 3. IF 2 OR MORE ZONES, USE THE LOWER LIMIT.

DENT THAT IS PERMITTED

C

2209534 S0000492995_V1

Allowable Damage Limits for Zone 3
Figure 110 (Sheet 2 of 5)

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DAMAGE CLEANUP

D

F79954 S0006595348_V2

Allowable Damage Limits for Zone 3
Figure 110 (Sheet 3 of 5)

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X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.50 INCH

REMOVAL OF DAMAGED MATERIAL ON AN EDGE ONLY
(NOT APPLICABLE IF HONEYCOMB CORE MATERIAL IS REMOVED)

C–C

F85106 S0006595349_V1

Allowable Damage Limits for Zone 3
Figure 110 (Sheet 4 of 5)

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DEFINITIONS OF THE DAMAGE DIMENSIONS
FOR A DISBOND AT AN EDGE

E

F85093 S0006595350_V3

Allowable Damage Limits for Zone 3
Figure 110 (Sheet 5 of 5)

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DENT THAT IS PERMITTED

A

2457623 S0000572141_V1

Allowable Damage Limits for Shallow Dents
Figure 111 (Sheet 1 of 2)

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NOTE: TO FIND DISBOND, YOU CAN USE NONDESTRUCTIVE INSPECTION PROCEDURES. REFER TO NDT PART 1, 51-01-02.

A = THE AREA OF THE DAMAGE.

D = THE LARGER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

d = THE SMALLER DIAMETER OF TWO ADJACENT DAMAGE AREAS.

X = THE DISTANCE BETWEEN TWO ADJACENT DAMAGE AREAS.

DAMAGE THAT IS PERMITTED ON THE SURFACE OF
AN ALUMINUM HONEYCOMB PANEL

B

2457624 S0000572142_V1

Allowable Damage Limits for Shallow Dents
Figure 111 (Sheet 2 of 2)

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REPAIR 1 - INBOARD AND OUTBOARD SPOILER SKINS

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTES

- REFER TO SRM 51-70-10 FOR TYPICAL METAL REPAIRS FOR THE SPOILER SKIN. MAKE SURE THERE IS SUFFICIENT CLEARANCE FROM ADJACENT STRUCTURE TO INSTALL THE REPAIR PARTS.
- THERE ARE NO REPAIRS IN THE STRUCTURE REPAIR MANUAL AT THIS TIME FOR TWO AREAS:
  - EXTERNAL DOUBLERS
  - AREAS THAT ARE 3.0 INCHES OR LESS FROM A FITTING

F84896 S0006595353_V2

Inboard and Outboard Spoiler Skin Locations
Figure 201

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NOTE: REFER TO THE IDENTIFICATION SECTION FOR THE SHAPES OF THE SPOILER.

LOWER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE

F84897 S0006595354_V1

Inboard Spoiler No.6 Skin Location
Figure 202

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NOTE: REFER TO THE IDENTIFICATION SECTION FOR THE SHAPES OF THE SPOILER.

LOWER SURFACE OF THE LEFT SIDE IS SHOWN,
RIGHT SIDE IS OPPOSITE
A TYPICAL OUTBOARD SPOILER

F84901 S0006595355_V1

Outboard Spoiler Skin Location
Figure 203

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IDENTIFICATION 1 - INBOARD SPOILER STRUCTURE

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

F74898 S0006595359_V1

Inboard Spoiler Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A4001	Spoiler Installation
113A4600	Spoiler Assembly - 6 and 7, Inboard Ground
113A4610	Panel Bond Assembly - Spoiler 6 and 7

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82225 S0006595361_V1

Inboard Spoiler No. 6 and 7 Structure Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2			
ITEM	**DESCRIPTION**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Channel	BAC1510-1354 7075-T73511 extrusion as given in QQ-A-200/11	
[2]	End Rib (2)	7050-T7451 plate as given in AMS 4050	

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IDENTIFICATION 2 - OUTBOARD SPOILER STRUCTURE

NOTE: REFER TO TABLE 1 FOR THE REFERENCE DRAWINGS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F90315 S0006595364_V1

Outboard Spoiler Location
Figure 1

Table 1:

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A4001	Spoiler Installation
113A4100	Spoiler Assembly - 1 and 12, Outboard Ground
113A4110	Panel Bond Assembly - Spoiler 1 and 12
113A4200	Spoiler Assembly - 2 and 11, Flight
113A4210	Panel Bond Assembly - Spoiler 2 and 11
113A4300	Spoiler Assembly - 3 and 10, Flight
113A4310	Panel Bond Assembly - Spoiler 3 and 10

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Table 1: (Continued)

REFERENCE DRAWINGS	
DRAWING NUMBER	**TITLE**
113A4400	Spoiler Assembly - 4 and 9, Flight
113A4410	Panel Bond Assembly - Spoiler 4 and 9
113A4500	Spoiler Assembly - 5 and 8, Flight
113A4510	Panel Bond Assembly - Spoiler 5 and 8

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NOTE: REFER TO TABLE 2 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F91015 S0006595366_V1

Outboard Spoiler No. 1 and 12 Structure Identification
Figure 2

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Table 2:

LIST OF MATERIALS FOR FIGURE 2			
ITEM	**DESCRIPTION**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Channel (2)	BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11	
[2]	End Rib (2)	7050-T7451 plate as given in AMS 4050	

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NOTE: REFER TO TABLE 3 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F90963 S0006595368_V1

Outboard Spoiler No. 2 and 11 Structure Identification
Figure 3

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Table 3:

LIST OF MATERIALS FOR FIGURE 3			
ITEM	**DESCRIPTION**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Channel (2)	BAC1510-1345 7075-T73511 extrusion as given in QQ-A-200/11	
[2]	End Rib (2)	7050-T7451 plate as given in AMS 4050	

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NOTE: REFER TO TABLE 4 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F90903 S0006595370_V1

Outboard Spoiler No. 3 and 10 Structure Identification
Figure 4

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Table 4:

LIST OF MATERIALS FOR FIGURE 4				
ITEM	**DESCRIPTION**	**T**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Channel (2)		BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11	
[2]	End Rib (2)		7050-T7451 plate as given in AMS 4050	

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NOTE: REFER TO TABLE 5 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F90727 S0006595372_V1

Outboard Spoiler No. 4 and 9 Structure Identification
Figure 5

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Table 5:

LIST OF MATERIALS FOR FIGURE 5			
ITEM	**DESCRIPTION**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Channel (2)	BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11	
[2]	End Rib (2)	7050-T7451 plate as given in AMS 4050	

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NOTE: REFER TO TABLE 6 FOR THE LIST OF MATERIALS.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F90886 S0006595374_V1

Outboard Spoiler No. 5 and 8 Structure Identification
Figure 6

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Table 6:

LIST OF MATERIALS FOR FIGURE 6			
ITEM	**DESCRIPTION**	**MATERIAL**	**EFFECTIVITY**
[1]	Leading Edge Channel (2)	BAC1510-1324 7075-T73511 extrusion as given in QQ-A-200/11	
[2]	End Rib (2)	7050-T7451 plate as given in AMS 4050	

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ALLOWABLE DAMAGE 1 - INBOARD AND OUTBOARD SPOILER STRUCTURE

1. Applicability

A. This subject gives the allowable damage limits for the inboard and outboard spoiler structure shown in Inboard and Outboard Spoiler Structure Locations, Figure 101/ALLOWABLE DAMAGE 1.

F82556 S0006595378_V1

Inboard and Outboard Spoiler Structure Locations
Figure 101

2. General

A. For each spoiler, refer to the figures that follow.

(1) Inboard Spoiler No.6 Structure, Figure 102/ALLOWABLE DAMAGE 1, for Inboard Spoiler Number 6

(2) Outboard Spoiler No. 5 Structure, Figure 103/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 5

(3) Outboard Spoiler No.4 Structure, Figure 104/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 4

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(4) Outboard Spoiler No.3 Structure, Figure 105/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 3

(5) Outboard Spoiler No.2 Structure, Figure 106/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 2

(6) Outboard Spoiler No.1 Structure, Figure 107/ALLOWABLE DAMAGE 1, for Outboard Spoiler Number 1.

B. Refer to Paragraph 4./ALLOWABLE DAMAGE 1 for the allowable damage limits.

C. Remove the damage as necessary.

(1) Refer to 51-10-02 for the inspection and removal of damage.

(2) Refer to 51-30-03 for possible sources of nonmetallic materials you can use to remove the damage.

(3) Refer to 51-30-05 for possible sources of the equipment and tools you can use to remove the damage.

D. After you remove the damage, do the steps that follow:

(1) Apply a chemical conversion coating to the reworked areas. Refer to 51-20-01.

(2) Apply one layer of BMS 10-11, Type I primer to the reworked areas. Refer to SOPM 20-41-02.

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82595 S0006595379_V1

Inboard Spoiler No.6 Structure
Figure 102

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82700 S0006595380_V1

Outboard Spoiler No. 5 Structure
Figure 103

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82713 S0006595381_V1

Outboard Spoiler No.4 Structure
Figure 104

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82681 S0006595382_V1

Outboard Spoiler No.3 Structure
Figure 105

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82705 S0006595383_V1

Outboard Spoiler No.2 Structure
Figure 106

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LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

A-A

F82708 S0006595384_V1

Outboard Spoiler No.1 Structure
Figure 107

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3. References

Reference	Title
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-30-03	NON-METALLIC MATERIALS
51-30-05	EQUIPMENT AND TOOLS FOR REPAIRS
SOPM 20-41-02	Application of Chemical and Solvent Resistant Finishes

4. Allowable Damage Limits

A. Cracks:

(1) Remove edge damage as shown in Spoiler Structure Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , and C .

B. Nicks, Gouges, Scratches, and Corrosion:

(1) You are permitted to do one of the procedures that follows:

(a) Remove damage as shown in Spoiler Structure Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Details A , B , C , D , and E .

(b) Drill out the damage as given for Holes and Punctures.

NOTE: The damage cannot be drilled out as a hole if the damage has been blended out.

C. Dents:

(1) The damage is permitted as shown in Spoiler Structure Allowable Damage Limits, Figure 108/ALLOWABLE DAMAGE 1, Detail F .

D. Holes and Punctures are permitted if they are:

(1) A maximum of 0.25 inch in diameter

(2) A minimum of 4D (D = diameter of the damage) from other damage, fastener holes, or material edges

(3) Filled with an aluminum 2117-T3 or 2117-T4 protruding head rivet.

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REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS DO NOT HAVE AN OVERLAP

A

REMOVAL OF DAMAGED MATERIAL AT EDGES WHERE THE FASTENER EDGE MARGINS HAVE AN OVERLAP

B

X = WIDTH OF THE MATERIAL THAT IS REMOVED
= A MAXIMUM OF 0.10 INCH

REMOVAL OF DAMAGED MATERIAL AT AN EDGE

C

F82823 S0006595385_V1

Spoiler Structure Allowable Damage Limits
Figure 108 (Sheet 1 of 3)

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REMOVAL OF DAMAGED MATERIAL ON A SURFACE

D

REMOVAL OF CORROSION AROUND THE FASTENERS

E

F82826 S0006595386_V1

Spoiler Structure Allowable Damage Limits
Figure 108 (Sheet 2 of 3)

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DENT THAT IS PERMITTED

F

F82827 S0006595387_V1

Spoiler Structure Allowable Damage Limits
Figure 108 (Sheet 3 of 3)

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REPAIR 1 - SPOILER END RIB

1. Applicability

A. Repair 1 is applicable to damage to the spoiler end rib shown in Inboard and Outboard Spoiler Locations, Figure 201/REPAIR 1.

NOTE: BOEING HAS NOT FOUND IT NECESSARY TO SUPPLY REPAIRS FOR THE INBOARD SPOILER NO. 6 (LEFT AND RIGHT) IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE IS SHOWN, RIGHT SIDE IS OPPOSITE

F84539 S0006595390_V1

Inboard and Outboard Spoiler Locations
Figure 201

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THE LEFT SIDE SPOILER IS SHOWN,
THE RIGHT SIDE SPOILER IS OPPOSITE
(A TYPICAL OUTBOARD SPOILER IS SHOWN)

F84548 S0006595391_V1

Typical Outboard Spoiler
Figure 202

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2. General

A. For the shapes of the spoilers, refer to the identification section.

B. Repair 1 is a Category C repair. Refer to 51-00-06 to find the definitions of the different categories of repairs.

C. Do an inspection of the blind rivet repair at each 5000 flight cycle interval.

D. Blind rivets that are loose, damaged, or that have fallen out must be replaced.

E. Repair 1 can be used only if you install the blind rivets flush against the internal structure.

3. References

Reference	Title
51-00-06	STRUCTURAL REPAIR DEFINITIONS
51-10-02	INSPECTION AND REMOVAL OF DAMAGE
51-20-01	PROTECTIVE TREATMENT OF METALLIC AND COMPOSITE MATERIALS
51-20-05	REPAIR SEALING
51-40-02	FASTENER INSTALLATION AND REMOVAL
51-40-03	FASTENER SUBSTITUTION
51-40-05	FASTENER HOLE SIZES
51-40-06	FASTENER EDGE MARGINS
AMM 27-61-11	FLIGHT SPOILER
AMM 27-62-91	OUTBOARD GROUND SPOILER
AMM 51-21-99 P/B 701	DECORATIVE EXTERIOR PAINT SYSTEM - CLEANING/PAINTING
AMM 27-61-11	Flight Spoiler
AMM 27-62-12	Inboard Ground Spoiler Removal and Installation
SOPM 20-20-02	Penetrant Methods of Inspection
SOPM 20-44-04	Application of Urethane Compatible Primer
737 NDT Part 6, 51-00-00	Structures - General
737 NDT Part 6, 51-00-00, Procedure 4	Surface Inspection of Aluminum Parts (Meter Display)

4. Repair Instructions

A. Get access to the damaged area.

(1) If necessary, remove the spoiler from the wing. Refer to the sections that follow:

(a) AMM 27-62-12 for the Inboard Spoiler Number 6

(b) Refer to AMM 27-62-91/401 for the Outboard Spoiler Number 1

(c) AMM 27-61-11 for the Outboard Spoiler Numbers 2, 3, 4, and 5.

B. Remove the damage to a maximum of 50 percent of the initial part thickness. Refer to 51-10-02.

(1) If necessary, cut out the damaged area.

C. Do a 10X visual inspection of the area to make sure that all damaged material is removed.

(1) It is optional to do an eddy current inspection. Refer to 737 NDT Part 6, 51-00-00, Figure 4 .

(2) If corroded material was removed, it is optional to do a dye penetrant inspection. Refer to SOPM 20-20-02.

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D. Make the repair parts as shown in Spoiler End Rib Repair, Figure 203/REPAIR 1. Refer to Table 201/REPAIR 1 for the repair materials.

Table 201:

REPAIR MATERIAL			
ITEM	**PART**	**QUANTITY**	**MATERIAL**
[1]	Angle	1	Use 2024-T4 extrusion that is one gage more than the rib flange
[2]	Strap	1	Use bare or clad 2024-T3 that is one gage more than the rib flange
[3]	Filler	As necessary	Use bare or clad 2024-T3 that is the same gage as the material that was removed

E. Assemble the repair parts.

F. Drill the fastener holes.

G. Remove the repair parts.

H. Remove all nicks, scratches, burrs, and sharp edges from the repair parts.

I. Apply a chemical conversion coating to the repair parts and to the bare surfaces of the rib flange.

J. Apply one layer of BMS 10-79, Type II primer to the repair parts and to the bare edges of the rib flange. Refer to 51-20-01.

K. Install the repair parts with BMS 5-95 sealant between the mating surfaces. Refer to 51-20-05.

L. Install the fasteners without sealant.

M. Apply a decorative finish if necessary. Refer to AMM PAGEBLOCK 51-21-99/701.

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NOTES

- D = THE DIAMETER OF THE FASTENER.

FASTENER SYMBOLS

REFERENCE FASTENER LOCATION.

INITIAL FASTENER LOCATION. INSTALL A BACR15FT5 RIVET OR AN MS20470DD5 RIVET.

REPAIR FASTENER LOCATION. INSTALL A BACR15FT5 RIVET OR AN MS20470DD5 RIVET.

REPAIR FASTENER LOCATION. INSTALL AN NAS1398B BLIND RIVET.

F84597 S0006595393_V1

Spoiler End Rib Repair
Figure 203

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ALLOWABLE DAMAGE 1 - WING SPOILER FITTINGS

NOTE: FOR THE ALLOWABLE DAMAGE LIMITS TO THE SPOILER FITTINGS, REFER TO:

- CMM 57-56-63 FOR OUTBOARD SPOILER NUMBER 1
- CMM 57-56-53 FOR OUTBOARD SPOILER NUMBER 2
- CMM 57-56-54 FOR OUTBOARD SPOILER NUMBER 3
- CMM 57-56-55 FOR OUTBOARD SPOILER NUMBER 4
- CMM 57-56-56 FOR OUTBOARD SPOILER NUMBER 5
- CMM 57-56-64 FOR INBOARD SPOILER NUMBER 6

LEFT SIDE SPOILER IS SHOWN, RIGHT SIDE SPOILER IS OPPOSITE

A

G02857 S0006595397_V1

Typical Spoiler Fitting
Figure 101

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REPAIR 1 - WING SPOILER FITTINGS

NOTE: THERE ARE NO REPAIRS FOR THIS PART IN THE STRUCTURAL REPAIR MANUAL AT THIS TIME.

LEFT SIDE SPOILER IS SHOWN, RIGHT SIDE SPOILER IS OPPOSITE

A

G02839 S0006595400_V2

Typical Spoiler Fitting Repair
Figure 201

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