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FATIGUE, RESIDUAL STRENGTH AND NON-DESTRUCTIVE TESTS OF AN AGING AIRCRAFT'S WING DETAIL

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Abstract. *This study concentrates on fatigue, residual strength and non-destructive tests of an aging aircraft's wing detail of the Finnish Air Force's Hawk Mk.51 jet trainer. The studied detail was the integral stiffener with a drain hole near the wing root. Fatigue tests were deemed necessary to verify experimentally the analytically observed short fatigue life, significant crack growth rates and eventually to re-assess the detail's inspection period for the fleet jets. The results of the study have been utilized e.g. at the complementary type approval accomplished for the Finnish Air Force's Hawk.*

1 INTRODUCTION

The Finnish Air Force (FIAF) has been operating their Hawk Mk.51/Mk.51A jet trainer fleet since the 1980s/1993s. As the fleet gets older, the fatigue related issues have become more acute. This study concentrates on fatigue, residual strength and nondestructive tests of an aging aircraft's wing detail. The studied detail was the integral stiffener with a drain hole near the wing root of the Hawk Mk.51. The Mk.51A has the modified wing with removed stress raisers (e.g. drain holes), strengthened structural details and changed materials. The results of the analytical life estimation calculations are introduced, too. The results of the study have been utilized e.g. at the complementary type approval accomplished for the FIAF´s Hawk.

Fatigue tests were deemed necessary to verify experimentally the analytically observed significant crack growth rate, high stress levels, short total life cycle and eventually to re-assess the detail's inspection period for the fleet jets. Parallel to the above studies, the service inspections of the fleet aircraft resulted in a crack indication within the Mk.51 drain hole region. The crack indication was made before the certified safe life of the wing while no known repair method existed. According to the CSI (company service instruction issued by the OEM) the wing with crack(s) in this location is not airworthy. Thus, the wing was removed from service.

2 GOALS

For fatigue test purposes described above, eight small-scale and six component test specimens were manufactured from a retired wing of a Hawk Mk.51 aircraft.

Small-scale test specimens were primarily used for the purpose of material properties' re-evaluation. To achieve an adequately comprehensive view of the desired material properties, both constant and variable amplitude tests were performed for small-scale specimens.

Only variable amplitude tests were performed for the component specimens. Component tests were desired to bring light for critical crack size, crack growth rate and eventually nondestructive inspection (NDI) interval. According to the CSI the NDI kick-off point, interval and method are presented in Table 1 (FI = Fatigue Index, FH = Flight Hours). The calculation of the fatigue index is based on the average aircraft mass and the G level exceedances. All Hawk jet trainers have a G-counter which registers the exceedings of the G levels of the flights. 68 FI is the certified safe life for wing of the Hawk Mk.51.

Kick-off [FI]	Interval [FH]	Method	NB
41	250	Eddy Current	both rotating and surface probe

Table 1 : Nondestructive inpection program of the studied detail[9]

Nondestructive tests were also carried out to evaluate the initial, critical and smallest detected crack size.

3 SPECIMENS

The test specimens were cut from two retired wings. Eight small-scale test specimens were cut from the right wing which had suffered 21.3 FI. One of the component specimens was cut out from the same wing as the small-scale specimens and six component specimens were cut out from another wing which had suffered 60.0 FI.

3.1 Small-scale test specimens

The small-scale specimens were cut out from stringers 1-4 between ribs 1-3 (Figures 1 and 2). The stringers, ribs and bottom surface of the wing form the integrated structure. The bottom surface side of the specimens was machined to constant dimensions. Finally the machined surfaces were polished. The holes were drilled for the specimens without any finishing treatment, thus corresponding better to real drain holes removed from sealing compound. The width, thickness, length and hole diameter of the specimens were about 20 mm, 6.7 mm, 280 mm and 5.1 mm, respectively.

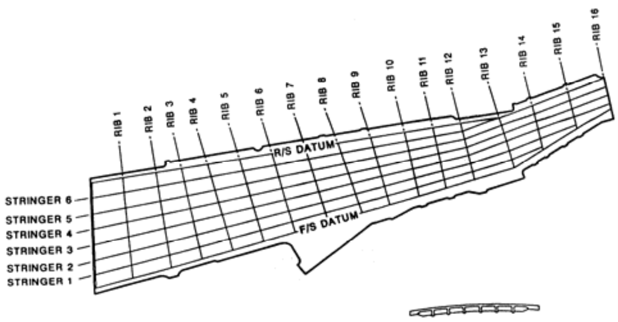


Figure 1: Numbering of the wing's ribs and stringer[5]

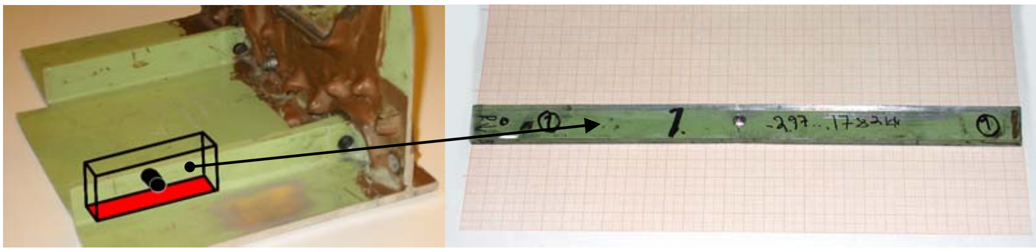


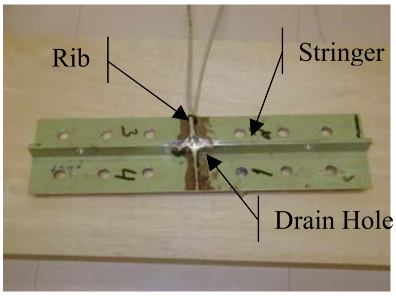
Figure 2: A small-scale test specimen[3]

3.1 Component test specimens

The component specimens represent drain holes where the rib 2 crossed the stingers 3-5 (Figure 1). The nominal main dimensions of the component specimens are presented in Table 2. Figure 3 reveals the meaning of the different indexes of Table 2. The indexes x_c and y_c specify the location of the center of gravity.

Index	Component specimens (ID)					
	STR3_r	STR3_l	STR4_r	STR4_l	STR5_l	STR5_aux
B_1	99.0	100.0	98.6	98.0	96.3	96.0
B_2	45.5	45.0	45.2	44.9	46.0	45.5
H_1	25.0	25.0	25.0	25.0	25.0	25.0
H_2	9.8	7.0	9.0	10.0	11.8	10.0
D	8.0	8.0	8.0	8.0	8.0	8.0
T_1	10.0	10.0	8.3	8.3	7.0	7.0
T_2	5.0	5.0	5.0	5.0	3.0	2.7
T_3	5.0	5.0	3.0	3.0	3.0	2.9
A	615.0	620.0	502.2	499.8	386.9	368.4
x_C	49.3	50.0	44.5	44.2	47.8	48.4
y_C	5.4	5.0	5.1	5.3	5.4	5.2

Table 2 : Dimensions of components at drain hole locations [mm][3]

a) 

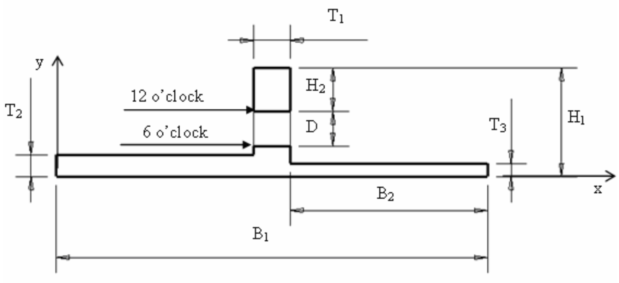
b) 

Figure 3: The STR4_r component specimen a) and the indexes of Table 2 b)[3]

4 TEST PROGRAM AND TEST FACILITY

4.1 Specimen material

The material of the specimens is aluminium in accordance with the British standard BS 2L 93:1971[2]. The material approximates the aluminium 2014-T6. In a long-transverse direction the material tensile properties are approximately as follows: yield stress 448 MPa, ultimate strength 510 MPa, modulus of elasticity 72400 MPa and Poisson's ratio 0.33.

4.2 Spectrum

Two FIAF's Hawk jet trainers have the permanent Operational Measurement Loads (OLM) set-up since 2000[e.g. 8]. One OLM set-up is in the Hawk Mk.51 and another in the Hawk Mk.51A. The test spectrum was created using six flights of the Hawk Mk.51. The used signal represented the wing root strain on the wing bottom surface. For the selected signals the peak-valley reduction was performed and the cut-off level of low signals was placed into 11.5 MPa based on the material SN-curve. The FIAF's average usage was aspired in the creation of the test spectrum. The prepared test spectrum has 1104 flights corresponding to 1000 Flight Hours (FH) and 13.2 FI as presented in Table 2. The order of the flights in the spectrum was random. With the proper transfer function (from the OLM strain gauge location to the fatigue critical location of interest) including offset the strains/stresses of the test spectrum corresponded to a load level of the real drain hole.

Flight	Number of Flights in "sub-spectrums"										Number of Flights	FH	FI
	1	2	3	4	5	6	7	8	9	10			
A	27	26	30	27	26	34	33	31	29	21	284	204	7.1
B	57	62	53	54	58	58	47	57	56	34	536	563	0.0
C	11	4	13	11	8	7	6	5	11	5	81	51	1.8
D	10	17	10	15	17	8	21	15	11	6	130	126	1.5
E	6	2	3	2	1	2	0	1	4	0	21	16	1.2
F	4	4	6	6	5	6	8	6	4	3	52	41	1.7
∑	115	115	115	115	115	115	115	115	115	69	1104	1000	13.2

Table 3 : Number of flights in the spectrum and "sub-spectrums"[3]

4.3 Small-scale specimen test program

Six constant amplitude tests were performed using three load levels and two specimens were tested using spectrum load. The load level or the mean stress – stress range combination were selected for the constant amplitude tests so that the consumption of a lifetime was severe. In the spectrum test the stress level of the real drain hole was aspired to the vicinity of the specimen hole. So the prepared spectrum was scaled by a proper linear function and the peak-valley values were connected with each others using half sine wave. Naturally the flexibility of the specimens and the performance of the test machine were considered.

The test facility was general purpose testing machine for the strength of materials – 100 kN INSTRON 8502.

4.4 Component specimen test program

The component tests were performed using an individual 250 kN hydraulic cylinder under load (force) control with INSTRON 8500 digital controller (Figure 4). The level of the cylinder force was controlled by strain gauges at the drain hole location: one on the stringer (R2) and the another on the bottom surface (R1) (Figure 5). The R1 strain gauge was used to control the cylinder and the secondary bending was specified by the R2 strain gauge.

The component specimens were asymmetric (Table 2) and the bottom surfaces were slightly convex. For these reasons, special attention had to be paid to the load transfer to the specimens. To obtain the typical secondary bending level e.g. the fastening had to be adjusted from specimen to specimen (Figure 4). In Figure 4b the extra fastening arrangement was created in order to achieve sufficient friction to the stringer and thus a proper secondary bending. Naturally the adjustment of the secondary bending caused different prestressing conditions. Because of this the stress level at the bottom of the drain hole (6 o'clock) (Figure 3b) was aimed to be kept near the desired value and the possible errors were allowed to accrue at the top of the drain hole (12 o'clock).

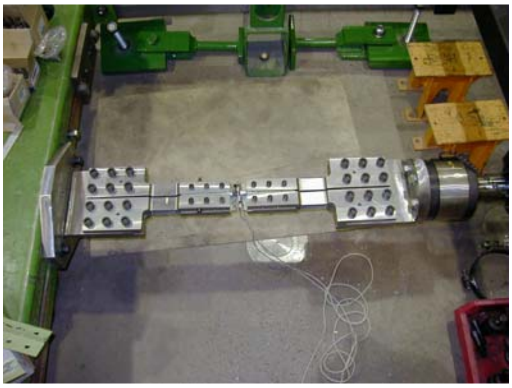
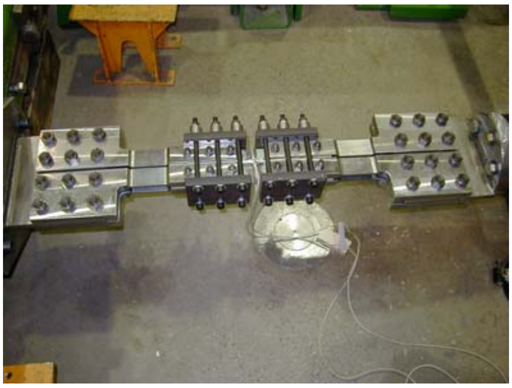
a)  b) 

Figure 4: Fastening arrangements for the specimens STR4_r a) and STR3_r b) [3]

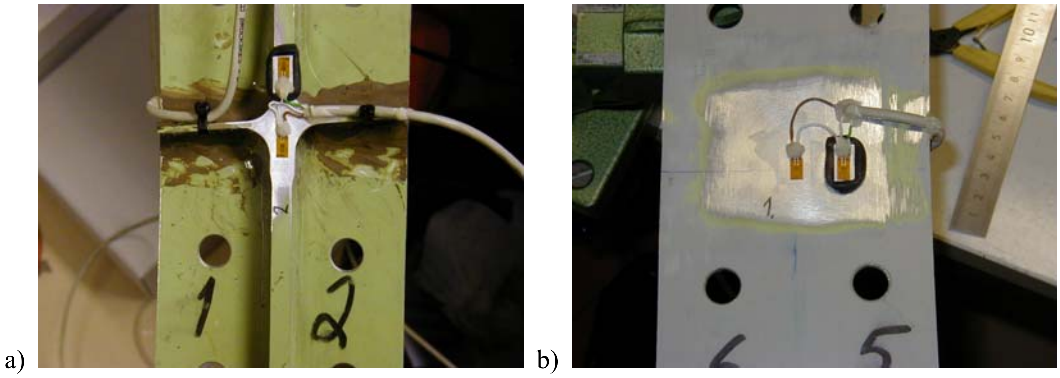


Figure 5: Strain gauges at the drain hole location: on the stringer (R2) a) and on the bottom surface (R1) b)[3]

4.5 Nondestructive testing

A visual and eddy current (ET) nondestructive testing methods were applied. Two constant amplitude small-scale specimens and six component specimens were tested. In the ET testing the rotating probe was applied. The visual inspection was performed using a magnifying glass and a microscope. Testing was carried out between the fatigue testing periods. In the beginning, the test intervals were long and when the flaws were appearing the intervals were shortened. The crack size on the side surface of the drain hole was tried to be detected using the separate ET pen probe during the component test with the specimen STR5_aux.

5 RESULTS

5.1 Small-scale specimens

The results of the constant amplitude tests are summarized in Table 4. Table 4 shows the specimen's ID, minimum and maximum cylinder forces and its range as well as the hole peak stress range and the number of cycles to failure. The results of Tables 4 and 5 reveal that the scatter in the life data is insignificant.

ID	F_{min} [kN]	F_{max} [kN]	ΔF [kN]	$\Delta\sigma$ [MPa]	N_f [cycles]	NB
SSS#1	-2.97	17.82	20.77	512.2	32 500	Eddy Current
SSS#2	-2.95	17.69	20.74	512.2	33 800	Eddy Current
SSS#3	0	17.57	17.57	365.9	69 600	
SSS#4	0	14.76	14.76	365.9	75 400	
SSS#5	2.08	11.76	9.69	256.1	148 400	
SSS#6	2.08	11.76	9.72	256.1	144 500	

Table 4 : Results of the constant-amplitude small-scale specimen tests[3]

The results of two variable amplitude tests are presented in Table 5. The total number of flights of the run spectrum is shown in columns corresponding to the presentation of Table 3.

ID	Number of Flights							NB
	A	B	C	D	E	F	Yht.	
SSS#7	1269	2411	368	585	98	233	4964	Fracture: 4.5 x spectrum
SSS#8	1311	2492	380	602	100	242	5127	Fracture: 4.7 x spectrum

Table 5 : Results of the variable amplitude small-scale specimen tests[3]

Based on the test results the adjusted material data for the strain-life method are presented in Figure 6. The data for the aluminium 2014-T6 from the database of the MSC.Fatigue-program[7] are also presented in Figure 6.

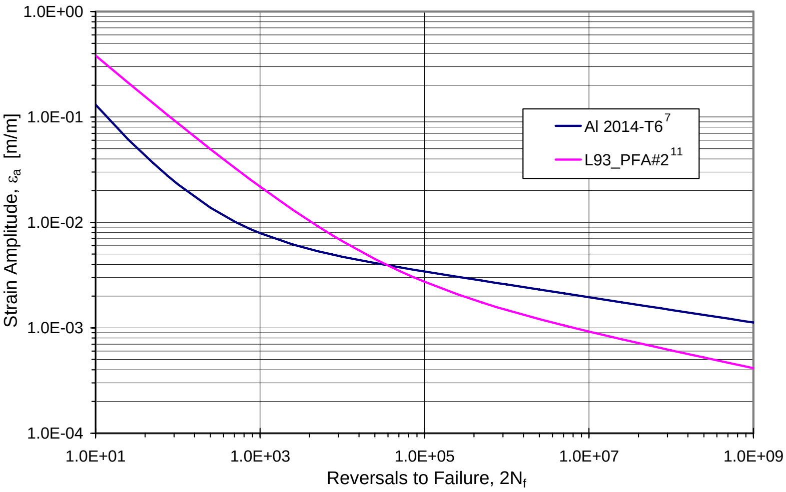


Figure 6: Strain-life curve based on the results of the small-scale specimens (L93_PFA#2)[11]

In Figure 7 the results of the NDI testing are presented. According to the indications of Figure 7 the average crack growth takes approximately 6000-6500 cycles when the corner crack grows from the size of 1 mm (a) up to 6.5 mm through the thickness. The previously performed crack growth calculation[1 6 9 10 11] showed that the growth took about 5500 cycles which correspond to the indications quite good. The test results also indicated the fracture toughness of 24.4 MPa√m[11] which is in line with the fracture toughness of 29.7 MPa√m[4] (long-transverse) and 19.8 MPa√m[4] (transverse-long) for aluminium 2014-T6.

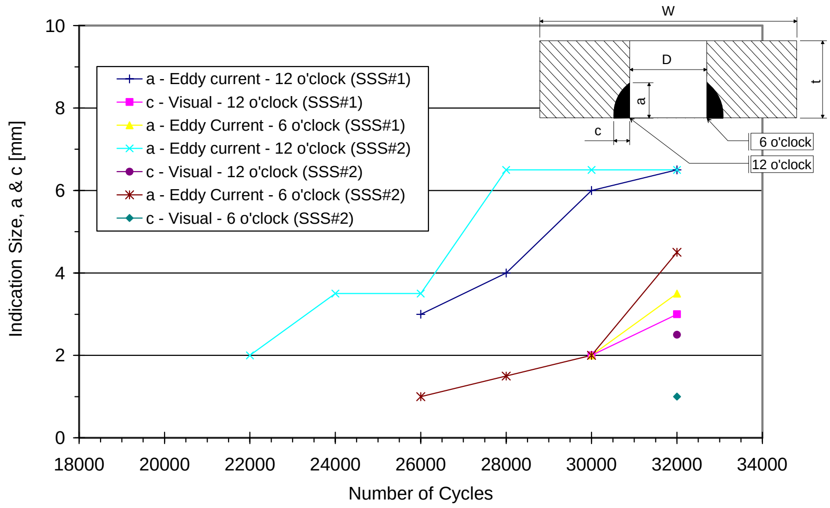


Figure 7: Indications of the Eddy Current and visual testing for two small scale-specimens[3]

5.2 Component specimens

In five of the six component specimens cracks were detected but only one specimen fractured from the desired drain hole. On the other hand it must be noted that the residual strength was evaluated by two specimens when the appeared cracks achieved an appropriate size. Table 6 concludes the test results. In this context a crack initiation (a_{init}) means the moment and the crack size when a crack or cracks were detected first time using the ET method.

ID	Crack Indication [FI]	a_{init} [mm]		Final Failure
		6 o'clock	12 o'clock	
STR3_r				No failure, no cracks. Secondary bending adjustment failed.
STR3_l	101	2		Fuel tank fastener hole fracture (127 FI)
STR4_r	101	2.5	2	Residual strength test when a = 2.8 mm (129 FI)
STR4_l	101	1.5	2.5	Drain hole fracture (146 FI)
STR5_l	87	3.5		Clamping jaw hole fracture (104 FI)
STR5_aux	90	3.5	3	Residual strength test when a = 5 mm (95 FI)

Table 6 : Results of the variable amplitude component specimen tests[11]

In Figure 8 the fatigue test results of the variable amplitude tests are summarized. The data of Figure 8 include the life cycles prior to fatigue tests (i.e. the flight-induced FI consumption). The results of the small-scale specimens do not include the life cycle prior to tests. The performed fatigue calculations showed that the stringer locations from which the small-scale specimens were cut of can stand exceedingly over 100 000 FI[11]. So the flight-induced FI consumption experienced prior to fatigue test cycling is negligible. In Figure 8 the initiation means life cycle consumption to the first flaw indication (ET) and the growth from that indication to failure. Now the margin against the crack initiation is about 1.4 and against the total life cycle is about 1.8. The crack growth takes on average 25.5 FI which means about 1900 FH based on the information of Table 3. On the other hand the crack growth scatter is significant from 7.6 FI to 46.1 FI. It is important to note that the test results are actual observations and they are not adjusted by any safety or scatter factors.

Based on the probability analysis crack initiation probability should be 6.6 %[11] at the point of 68 FI.

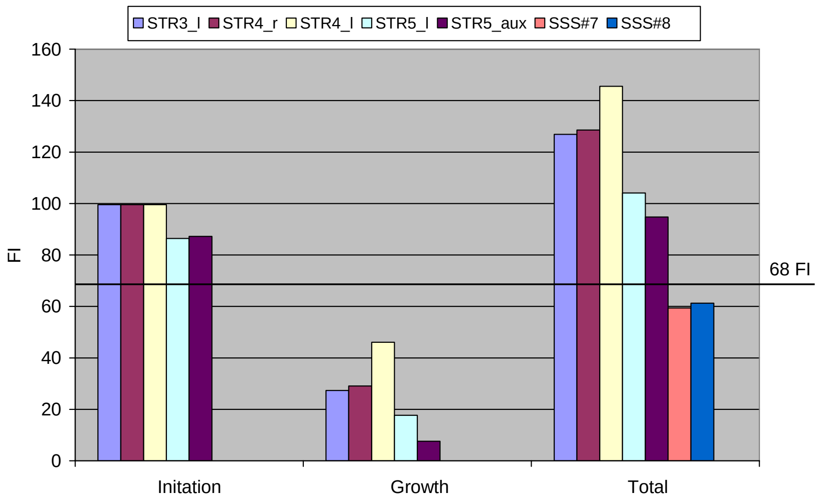


Figure 8: Results of the variable amplitude component specimen tests converted into FI[3]

The results of the NDI tests also indicated quite wide scatter in crack growth[3 11]. According to inspection tests the cracks tend to develop to the corners of the drain hole at 6 and 12 o'clock positions. Generally, a multiple crack pattern was observed. The NDI tests also showed that the appropriate initial crack size would be the corner crack of 2 mm x 1 mm (a x c, see Figure 7) which is caught with quite a high probability using the present NDI methods.

The residual strength tests of the specimens STR4_r and STR5_aux indicated the margin of 1.2 and 1.1[11] against the demanded residual strength, respectively. So crack sizes of the specimen STR5_aux have been very near their critical size. According to the residual strength tests with the NDI and fracture surface inspections the estimation for critical crack size was 4 mm[11] at 6 'clock position and 1.5 mm[11] at 12 o'clock position. The crack size means the c-dimensions (see Figure 3b) in which case the a-dimension is through the thickness.

6 CONCLUSIONS

The fatigue experiments of FIAF's Hawk Mk5.1 jet trainer wing drain holes and their results are presented in this paper. The drain hole investigations comprised the small-scale and component fatigue tests. The scatter of the results of the small-scale test specimens was narrow. On the other hand the scatter of the results of the component tests was quite wide thus complicating the conclusions from the test results.

The certified safe life for wing is 68 FI. The test results indicated that the probability of the crack initiation was 6.6 % at the point of 68 FI. On the basis of the fatigue test results, quite comprehensive material properties knowledge was achieved. The tests strengthened the validity of the previous used fracture properties: crack growth rate, fracture toughness and critical crack size. Above all, the results indicate that previously performed fatigue calculation would be conservative.

A reasonable estimate of the initial corner crack size was obtained based the test results, too. Overall the test results indicate that the NDI program presented in Table 1 could be followed in future although the scatter of the results of the component tests was quite wide.

REFERENCES

[1] AFGROW - Fracture Mechanics and Fatigue Crack Growth Analysis Software. http://www.siresearch.info/projects/afgrow/.

[2] BS 2L 93:1971. *Aerospace series specification for plate of aluminium-copper-magnesium-silicon-manganese alloy (solution treated, controlled stretched and precipitation treated) (Cu 4.4, Mg 0.5, Si 0.7, Mn 0.8)*. BSI British standards, (1971).

[3] M. Bäckström, S. Liukkonen, S. Merinen, A. Siljander, J. Juntunen, M. Sarkimo, K. Lahdenperä, *HW-small-scale specimen and component specimens tests*, Technical Research Centre of Finland, Research Report TUO33-032272 (in Finnish, restricted), (2003).

[4] Esacrack user's manual, European Space Agency, ESA PSS-03-209 Issue 2, (1995).

[5] Hawk for Finland, Aerodynamics & Structures Course, Structures, Volume 2, (1992).

[6] K. Koski, M. Bäckström, *Non-reference stress intensity factor solutions*, 2001 USAF Aircraft Structural Integrity Program (ASIP) Conference Proceedings, (2001).

[7] MSC.Fatigue. MSC.Sofware Corporation. http://www.mscsoftware.com/.

[8] A. Siljander (ed.), *A review of aeronautical fatigue investigations in Finland during the period February 2001 to March 2003*, the 28th International Committee on Aeronautical Fatigue (ICAF) Conference, (http://www.vtt.fi/inf/julkaisut/muut/2003/icaf_2003_finland_review.pdf),(2001)

[9] J. Tikka. *The extension of the fatigue research of the wing of the Hawk Mk 51*, Patria Aviation Oy, Design Report HW-L-0061 (in Finnish, restricted), (2001).

[10] J. Tikka, *Fatigue life evaluation of critical locations in aircraft structures using virtual fatigue test*, Proceedings of the 23rd International Congress of Aeronautical Sciences (ICAS), (2002).

[11] J. Tikka, *The fatigue tests of the Hawk wing fuel drain hole specimens*, Patria Aviation Oy, Design Report HW-L-0069 (in Finnish, restricted), (2003).