FATIGUE LIFE ANALYSIS OF MAIN LANDING GEAR PULL-ROD OF THE FIGHTER JET AIRCRAFT

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Abstract

Aircraft life extension carries the inherent risk of adverse structural effects occurring as a result of fatigue. These effects may lead to structural damage or complete destruction of the aircraft. This paper presents illustrates the possibilities of combining numerical analyses, nondestructive testing and component fatigue tests for the purposes of the main landing gear pull-rod damage investigation. The laboratory tests have been performed on samples taken from the decommissioned pull-rods.
Results of the present research will be utilized for revising maintenance bulletins and diagnostic methods needed to ensure the safe operation of the pull-rod elements until they are replaced with new ones.

1 Introduction

Aircraft landing gears are a critical group of subsystems. Any damage to almost any of their components results in a dangerous incident which in turn may lead to a crash. Providing are adequate level of safety during flights as well as during ground operations requires that the strictest quality criteria are fulfilled throughout the manufacturing process and subsequent operation.

Most in service aircrafts of the Polish Air Force (PLAF) were manufactured over twenty years ago in the former Soviet Union or in Poland. Currently these aircrafts have reached or have exceeded the planned period of operation. Cost of buying new aircrafts and the fact that the currently operated aircrafts have significant reserves of hourly service life invite the operator to make attempts of extending the operational lifetimes of particular aircraft. The process of extending the lifetime involves taking a series of measures designed to assess the usability of these aircrafts.

This paper describes the causes of damage in the form of rupture of a structural component of the main landing gear. There have been two incidents noted which both occurred during aircraft hangar standby. It should be highlighted that two mentioned cases concern a particular aircraft currently in operation, and that these incidents occurred a few days after the last flight. After the first event all the components under examination have been inspected using the non-destructive testing, which in turn led to decommissioning of the damaged elements. The diagnostic process is also carried out during airframe repairs in specialized workshops.

This article describes the process of investigation needed to determine the causes of failure in the test item. In order to clarify these causes, examination of the fracture surface's micro-structure has been conducted, as well as numerical research such as the analysis of the system's kinematics and the determination of loading forces, as well as the study of fatigue life of the system. The numerical analyses have been reinforced with results of the in-flight testing carried out as a part of the operational life extension program. The sequence of loads to which the landing-gear component may be subjected has been determined on the basis of the in-flight test results.

The current state of knowledge concerning the studied damage enables the authors to point out to corrosion-attributable fatigue initiated during manufacturing as the main cause of the damage.

2 Su-22 FITTER general information

Sukhoi Su-22 FITTER is a single engine supersonic fighter–bomber jet airplane. The aircraft was manufactured in the former Soviet Union in several variants. It has been operated by the Warsaw Pact countries as well as by some Middle Eastern and African states. In the Polish Air Force it has been in service since 1974, when the Su-20 variant was introduced as a successor to the Su-7. From the year 1984 many Su-22 fighters have been operated, in two variants: Su-22M4 - an export combat version and Su-22UM3K - a two-seat training version.

The airframe structure is a semi-monocoque with a variable sweep wing which improves the flight characteristics in the whole range of flight velocities. Top maximum velocity, depending on the variant is 1.7 ÷ 2.1 Ma, while the landing speed is 280 ÷ 290 km/h.

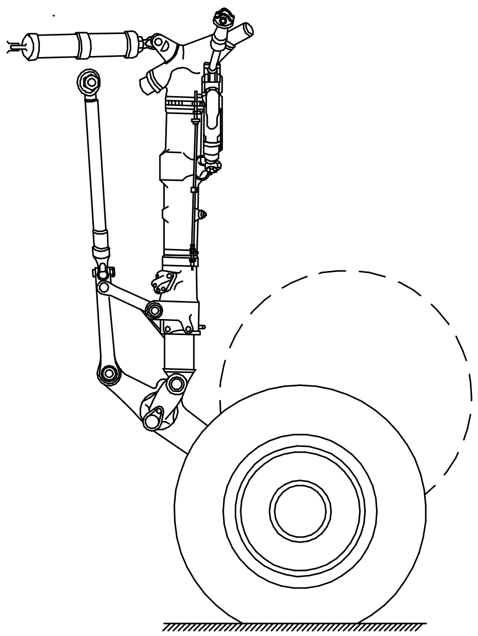


Fig. 1. Main landing gear schematics.

The Su-22 aircraft is equipped with a three-strut landing gear. Main loading forces acting on the aircraft during landing, runway maneuvers and standby are transmitted by the main (rear) landing gear struts (Fig.1), attached to the structural elements in the wing's landing gear recesses. Chief structural elements of the main landing gear are, among others, the strut, shock absorber cylinder, trailer arm, torsion links, and the pull-rod.

The pull-rod is the member responsible for transmitting the load to the wing's strength members via the torsion links. It consists of two structural elements, the lower eye and the pull-rod tube. These two elements are joined by welding. The tube is fastened to the other eye with a threaded connection which permits regulation of the pull-rod's length.

The component set-up (Fig. 2) for the welding process utilizes a steel alignment ring. The ring is attached to the pull-rod cylinder of the eye component by three spot welds that are evenly placed on the tube's circumference.

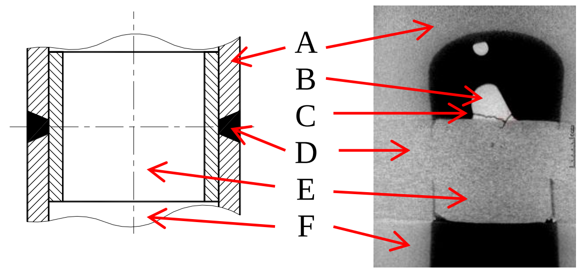


Fig. 2. Welded wiring diagram
A – pull-rod lower eye; B – spot welded seam; C – crack; D – welding seam; E – alignment ring; F – pull-rod tube

3 Problem description

The problem described in the present paper concerns the pull-rod of the Su-22's main landing gear – a critical structural element [3]. The problem is related to two incidents, of which first (in 2005) was surprise for the operator. During a morning inspection of hangar buildings, the aircraft was observed lying in an untypical position (Fig. 3, 4). Preliminary visual inspection revealed a rupture of the pull-rod of the left main landing gear. After the inspection, the aircraft has been elevated and restored to a proper position and the damaged component was replaced. Subsequently, a detailed inspection of the landing gear has been carried out, as well as any needed repairs. After the repairs, necessary tests have been performed to verify landing gear's performance – such as retraction and extension tests, automatic braking systems tests, cockpit signalization systems test and others. Next step in the landing gear's inspection was the inspection of the torsion links' structural health with the color defectoscopy method.

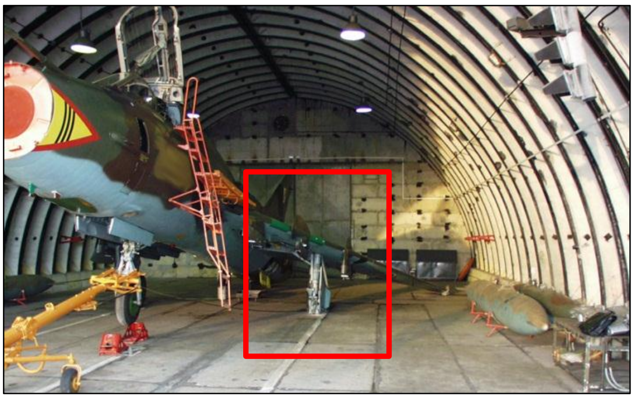


Fig. 3. Failure of the main landing gear [3]

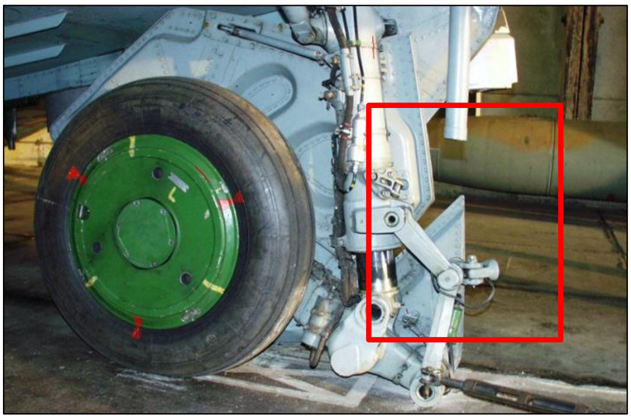


Fig. 4. Failured pull-rod [3]

The called Incident Board ordered an analysis of the flight parameter records from the last 5 months of service. Parameters were taken from the onboard flight data recorder "TESTER" and last 5 months of service were analyzed. During this period the aircraft in question has made 13 flights. This was the period between the last maintenance test flight and the day the gear damage occurred. In the course of investigation the following flight parameters have been taken into account:

- aircraft velocity - at gear retraction and extension,
- landing velocities and accelerations,
- fuel residue.

Upon the Board's motion, inspection of the fracture surface was performed with the use of a magnifying glass (x5). The analysis of the fracture face's character enabled the Board to evaluate the usefulness of the color penetrant defectoscopy method as a means of preliminary selection of pull-rods that might present a flight safety hazard for the Su-22 fleet. During the tests and investigation of the rupture cause macroscopic analyses using magnifying glass as well as microscopic analyses with the use of microscopes were performed. Several defectoscopy methods have also been utilized i.e. the visual, eddy-current, magnetic particle and the ultrasonic method.

It should be particularly noted that the described incident took place during hangar standby, a few days after the aircraft's last flight before the incident. This indicates that the critical crack length had almost been reached, yet without the crack being noticed. Prior to the emergence of the critical crack on the pull-rod, the aircraft has been in service for 19 years during which there have been about 1700 take-offs and landings. The in-flight time for the described aircraft amounted to almost 1300 hours, with the designed service life being 2000 hours.

According to the instruction of the Investigation Board, tests have been performed on the rest of the fleet. The tests consisted of color penetrant defectoscopy (Fig. 5) analysis of the critical welded joint. In some cases doubts have been raised as to a particular pull-rod's structural integrity.

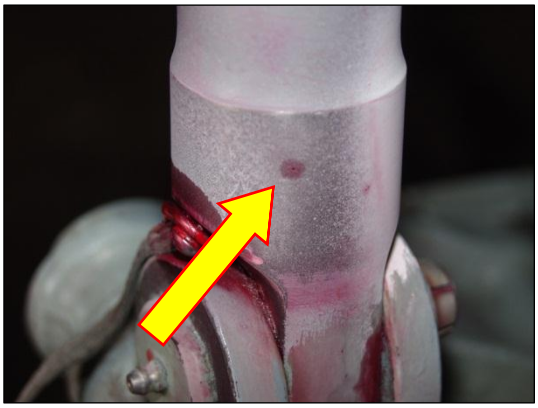


Fig. 5. An example of a defect in the welded joint area [3]

Although diagnostic tests were being performed since the incident, the problem re-emerged in 2010. Similarly to the earlier incident, a rupture of the pull-rod of an in-service aircraft occurred. As before, a few days after the last flight a rupture of the pull-rod elements took place. This time, the incident happened on an aircraft that has been in service for 34 years/~1300 flight hours. In this service period the aircraft has performed about 1700 landings.

Fig. 6. Fatigue crack of the landing gear pull-rod

Both rupture cases seem to have many common characteristics, and the damage occurred in the same area. The occurrence of another pull-rod rupture in an in-service aircraft suggests that the countermeasures applied since the first incident were insufficient. In consequence, further actions have been carried out to determine the causes of damage. Also, NDI methods hoped to enable the detection of damage before the pull-rod ruptures have been revised and verified.

4 Damage analysis

4.1 Operation profile and flight tests

Taking into account the 2010 service profile of the Su-22 in the PLAF, an attempt has been made to work out the assumptions that would enable the extension of the polish Su-22s' service life. The necessary research was carried out by the Air Force Institute of Technology in Poland, which is the R&D support institution for the PLAF. Su-22s operated by the PLAF are aircraft which have already reached their designed calendar service life. According to the initial assumptions, the fleet should be decommissioned. However, taking into account the enormous cost of replacing the aged aircraft and the fact that the aircraft have a significant reserve of hourly service life, they still remain in service. The designed service life was 2000 flight hours, yet its current realization in the PLAF is ca. 1500 FH per single airplane.

As a part of preparations for the flight tests and the maintenance system modification program, an average service profile for the PLAF's Su-22 aircraft has been determined. Service profile is the specification of percentage contributions to component fatigue damage from each of the flight phases distinguishable. In the course of the actions described, the landing gear's average performance profile has also been determined. The profile devised includes such elements as: take off, landing, taxiing, as well as any other operations that cause the loads on the landing gear components to vary in time. The elements - phases specified in the average service profile have been executed during the flight test program.

Most of the landing gear's fatigue damage can be attributed to the Ground Air Ground (GAG) cycle. The dynamics as well as the range of the loads in this particular cycle inflict the most fatigue damage to the landing gear's elements. Because of that, the manufacturers often express the landing gear durability in the number of permitted landings/take-offs.

Based on the service load profile a flight test program was devised. This program was carried out in 2004. For the tests, one Su-22M4 (single-seat combat version) has been employed. Prior to the tests, during an overhaul at the Military Aviation Depot no. 2 the aircraft has been prepared for the flight tests and fitted with the ACRA KAM-500 flight data recorder. The aircraft's preparation consisted of instrumenting the airframe with strain gauge sensors. Chosen structural elements and their corresponding strain gauges (including the landing gear pull-rod) have been calibrated with known loads. This operation enabled to measure and record the load signal values in units of force. In all, 12 flight parameters and 62 strain gauge signals were recorded during the flights. Throughout the program the KAM-500 recorder was synchronized with the on-board recorder TESTER. In the program, 11 flights have been performed - including the accompanying maneuvers - runway taxiing, aircraft hauling and engine tests. Flight tests have been prepared so that all the flight element blocks could be executed. One of the signals recorded was the force acting on the landing gear pull-rod, the subject of the present paper.

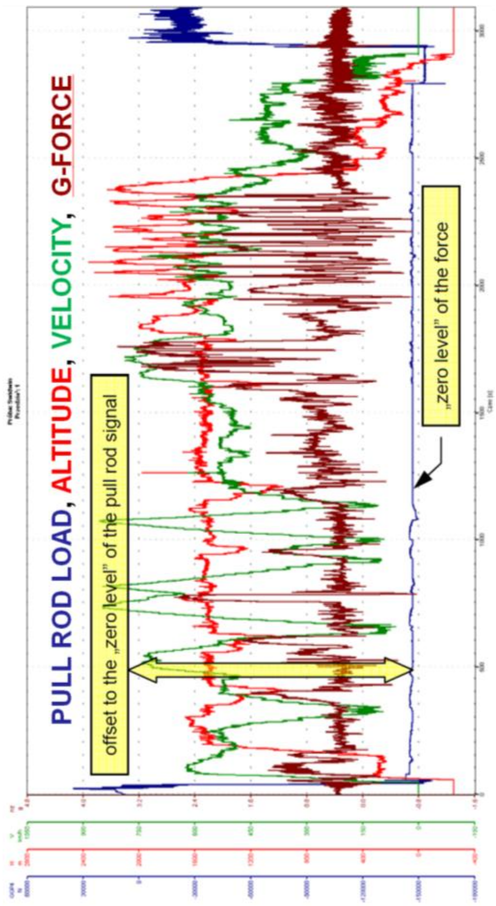
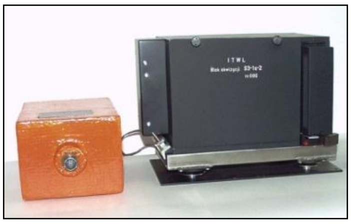


Fig. 7. Selected record of the test flight –
H – altitude, V – velocity, nz – vertical acceleration,
GGP4 – load in pull-rod signal



a)

b)

Fig. 8. Recording systems
a) TESTER U3L – b) ACRA KAM-500

4.2 Model of the landing gear

As a part of the service life extension program a computer model of the main landing gear was created. Landing gear dynamics/motion analyses were performed using the MSC.ADAMS Software [7]. Analyses have been performed for purposes of AFIT. The CAD model was created in the NX software with the help of reverse engineering methods. The MSC.ADAMS - Aircraft software is used for computer simulations of landing gear performance in different load conditions. The main advantage of such model is its diversity. The model (Fig.8) may be used for the determining of the load dynamics in each of the members as well as in analysing the performance of the dynamic components.

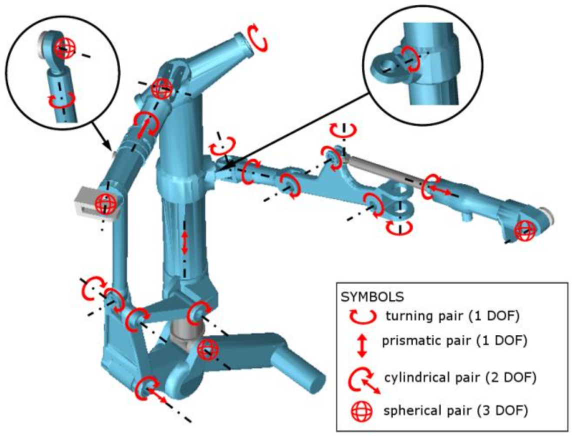


Fig. 9. View of the main landing gear ADAMS model

Fig. 10. View of the main landing gear

The computational model has been verified in comparison to the load signals recorded during the flight tests. The analysis results show a good agreement with the experimental data. Any differences can be attributed to the simplifications made in the dynamics model. The model can be used to analyse the load dynamics in each component as well the stresses in the parts - using the Finite Element Method.

4.3 Crack growth analysis

Determination of the critical crack length, as well as of the number of cycles necessary for the onset of critical damage is one of the most important tasks that the durability engineers face. The rupture incidents described in the paper consequenced in performing fatigue calculations of the main landing gear pull-rod. These crack growth analyses have been performed with the NASGRO 5.2 software. For such analyses a database containing the investigated part's load and performance history and material data is required, as well as making an assumption regarding the suspected damage mode.

The pull-rod's material properties were determined in the AFIT's Strength of Materials Testing Laboratory. It was concluded that the pull-rod is manufactured from a high-strength 30HGSNA steel (UTS = 1620 MPa, YS = 1370 MPa). The material test has been performed with the use of the MTS 810.23 Material Testing System.

The load sequence, to which the analysed element is submitted during typical operation, has been determined based on the flight test results. During the tests the time variation of load was recorded. Based on this record the load profile needed for the computational analyses has been devised. The load profile input in the simulation has been composed from the following blocks: take off (1) landing (2) runway taxiing (3) runway towing (4). These blocks have been set together in the sequence 4-4-4-1-2-1-2-1-2-4-4-4-3. This sequence is an equivalent of 3 GAG cycles along with the approximation of accompanying runway maneuvers. Hard and asymmetric landings were not considered because the AFIT doesn't keep records of such flight elements, and because their realisation during tests poses a significant risk and a safety hazard. The load sequence to which the element has been submitted in the simulation is shown in Fig. 10. The load sequence has been adequately modified. The modifications consisted of filtering of the load sequence with the cycle truncation in the range above <20 MPa and of counting the cycles with the Range-Pair method. Load value truncation reduces the number of fatigue cycles which helps to reduce the calculation time, with minimal error in calculating critical crack length and growth time. Cycle count reveals 3 GAG cycles in the load sequence. Not performing the cycle count for the load sequence assumed significantly influences the end result.

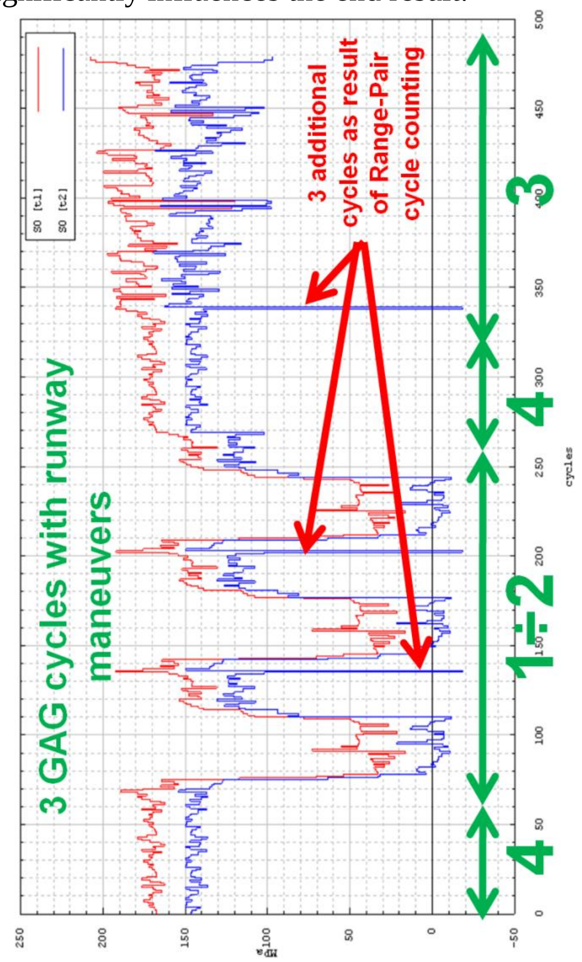


Fig. 11. Load sequence used in analysis

In the analysis performed, the resulting computed fatigue life is expressed in the number of take-off – landing cycles – with the assumption that a pre-existing crack of the length a_0 was present in the initial state. Results

shown in the Table 1 don't take into account the safety factors which adjust the number of cycles until critical crack length is reached. The calculated number of propagation cycles (NASGRO 5.2 software) in the analysed critical element significantly exceeds the number of GAG cycles at which the discussed failure occurred.
Because it has been noted that the load sequence recorded may not represent the excess of loads faithfully, additional analyses with the use of a multiplier have been performed. The sequence prepared from the records of the take-off – landing sequence and the runway maneuvers has been multiplied by the factor of 1.2. The critical crack length determined then in the computational simulation came close to the length of the actual crack observed on the fracture surface. Henceforth, it can be assumed that in the cracking process an additional factor must have been present that accelerated the crack growth. Apparently this factor was not taken into account in the present simulation.

Table 1. Sample of the analysis results

	Crack model	Max. load [MPa]	Initial crack size [mm]		Critical crack size [mm]		Number of flights
			a	c	a_{max}	2c	
1	SC05	207.4	1.905	1.905	6.20	41.86	76 838
2	TC08	207.4	-----	2.540	-----	41.34	16 982
3	SC05	248.9 *	1.905	1.905	6.20	35.28	38 483
4	TC08	248.9 *	-----	2.540	-----	34.74	8 807

*) multiplier – 120% of the load

5 Pull-rods inspections

5.1 Overhaul and inspections

During its service the pull-rod undergoes overhaul [3]. The overhaul consists of removing of the paint layer and the anti-corrosion protection. Afterwards the pull-rod is subjected to a NDI analysis (magnetic method) on magnetic flaw detector MD 1400. Elements on which cracks are detected is subjected to decommission. Scratches, abrasions, corrosion pits on the outside surfaces having the depth up to 0.3 mm are rubbed off using abrasive methods. Elements in which flaws which exceed 0.3 mm are detected also is subjected to decommission.

Throughout the period in which the aircraft is in service at the military unit it is subjected to periodic checks. The Su-22 aircraft, due to the nature of their service, are subjected to varied maintenance actions and checks. The checks pertain also to the landing gear elements and amount to, among others, the crack detection.

5.2 Technical condition verification

As a consequence of the 2004 incident actions have been undertaken with the aim of verifying of the currently serviced pull-rods. These actions were preceded with a research of the methods of inspecting the critical area shown in the illustration below [13]. Following NDI methods have been reviewed: the visual method, eddy-current, magnetic particle and the ultrasonic method. The review took into account any possible variants of failure modes, as well as the time and location of investigation. Based on this, conditions for conducting NDI have been specified.

In the case of performing the investigation *in situ* at the military base on an element installed on the airframe, the ultrasonic method has been specified as the most reliable. Utilizing the method necessitates however, that suitable crack test items are produced and the inspection probes are adapted accordingly. If a possibility of de-installing the pull-rod off the aircraft exists, utilization of the magnetic method inspection (performed on a stationary flaw detector) has been determined as the most suitable. This test should be reinforced with a visual inspection of the pull-rod. The magnetic particle method has also been mentioned if the flaws are searched for on the surface of the pull-rod. The magnetic particle method should also reinforce the inspection of the pull-rod tube's interior.

As a result of the research reports made in relation to the rupture incident, inspections of the in-service pull-rods have been carried out with the following methods: visual, magnetic particle and ultrasonic. In all, 50 pull-rods from 25 Su-22 aircraft have been inspected. Three pull-rods had been classified as damaged and were ordered for decommissioning.

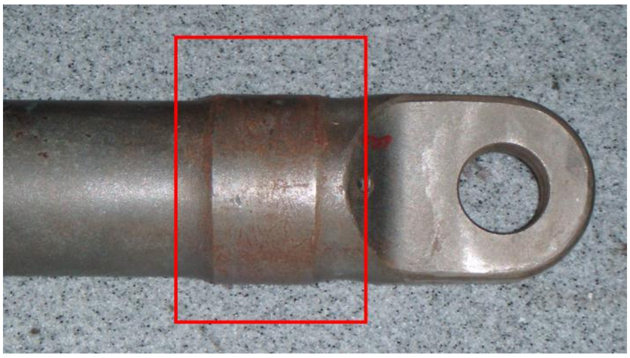


Fig. 12. Pull-rod critical area

6 Additional pull-rod material tests

In the process of durability assessment an additional test was carried out, that was supposed to determine the fatigue strength of the component in the area away from the weld. The middle part of the pull-rod was selected for this research. This middle part is a thick-walled tube with external diameter of 55 mm and wall thickness of 6,2 mm. The tube has been threaded at the ends to create the fittings for the test machine. Two distinct specimens have been elaborated in this way. These specimens have been weakened by removing a significant portion of the cross-section. For the strength tests only two load-paths were left that constituted together a quarter of the initial section area. Initially it has been assumed that the crack will develop in the form of a surface crack with initiated on the inner wall of the specimen. Therefore the specimen has been incised in that location and a of depth a=0.5 mm and width 2c = 9 mm was created. Shape of the specimen has is shown on the illustration below. Based on these assumptions, models for the FE calculations have been elaborated.

The tests have been performed with the use of the MTS 810.23 Material Testing System at AFIT's Strength of Materials Testing Laboratory. The specimens have been mounted at the test stand and subjected to variable loads. The first specimen has been subjected to a increasing value of load. The initial load value was 14 kN and it increased continuously to 63 kN. Overload cycles with 120% of nominal load value were introduced. Minimal load value has been set to 1 kN. This measure has been taken to assure that the specimen is in constant tension. After the signs of cracking have been observed at 80 000 cycles, the load value has been lowered to 52 kN. In the course of the test value of 101 000 cycles to failure has been measured.

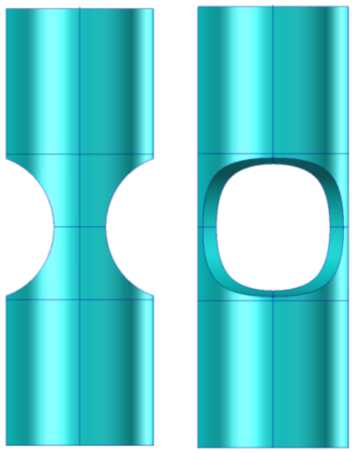


Fig. 13. Numerical model of the pull rod fatigue test specimen

The second specimen has been subjected to loads varying from 1 kN to 52 kN throughout the test. Every 1000 cycles, an overload cycle was introduced with 120% value of load (62,4 kN). The specimen (subjected to loads as described) withstood 192 500 cycles to failure. The observed cracks propagated from the inner edges of the specimens. The crack propagation was not uniform between the load paths.

The crack initiated at the inner edge, despite of the presence of an incision at the wall surface. Because of the geometry, the specimens were locally subjected not only to tension but also significant bending. The FE model has been subjected to the load of 52 kN which was equal to the load value used in the tests.

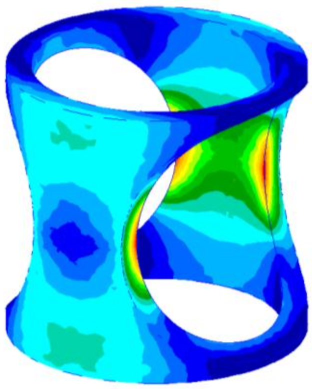


Fig. 14. Tension analysis (fatigue test load level)
52 kN Load, 760 MPa – HMH stress

Finite element calculations established that the Von Mises equivalent stress at the specimen edge reached 760 MPa. Actual loading of the cross section has been found to be more than two times larger than the analytical predictions for tension loading in the critical area.

After the fatigue tests specimen no. 2 was subjected to a microscopic examination at the Material Testing Laboratory at the Military University of Technology in Warsaw [15]. A fractographic analysis was conducted along with a qualitative microanalysis of the chemical composition. The illustrations below show the microstructure of fracture faces at the selected fatigue locus.

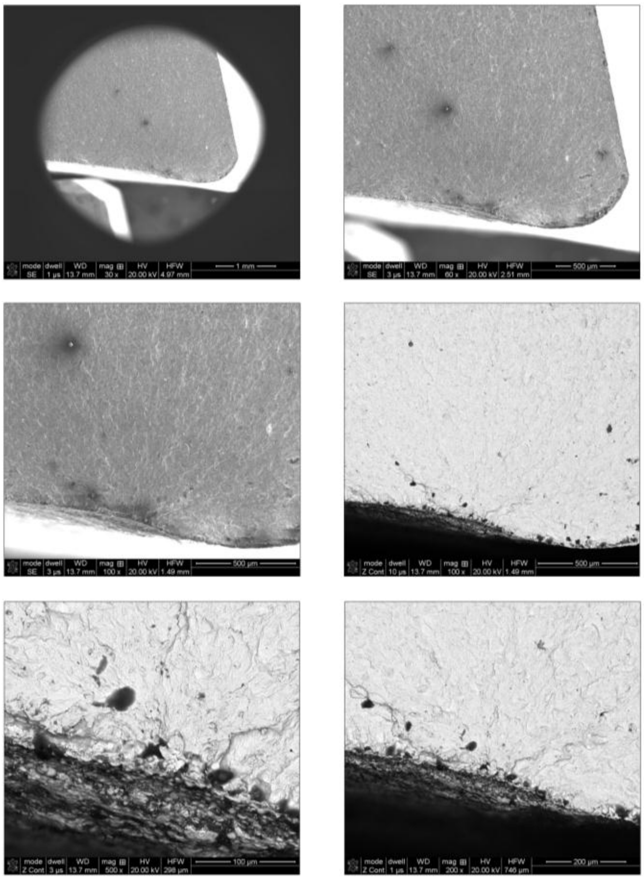


Fig. 15. Fatigue locus [15]

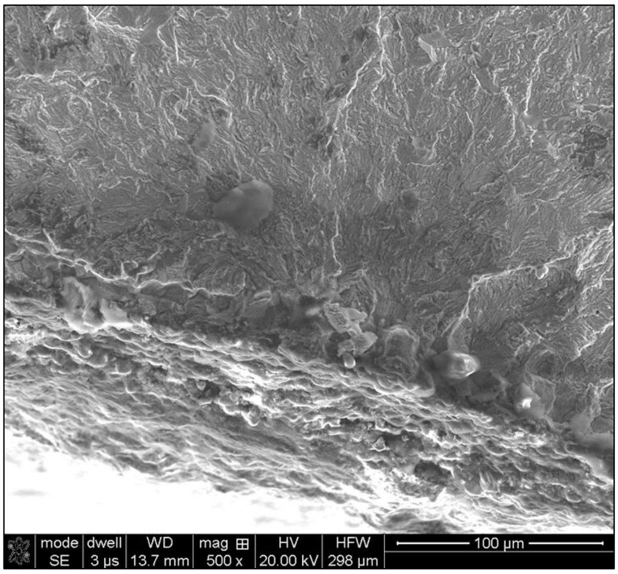


Fig. 16. Area of the crack [15]

Figures above show that fatigue which occurred during the material tests was caused by the load sequence and shape of the specimen. There were no material defects found.

7 Summary and conclusions

During the service of Polish Su-22 aircraft, a serious problem has been encountered that endangered the flight safety and the further service of the fleet. First case of damage has occurred in 2005 on an aircraft serviced according to the safe-life philosophy, even before the designed service life has been reached.

Material tests and other engineering analyses made then, pointed to fatigue fracture as the cause damage mode. Initiation of the cracking process originated in the area of a structural notch on the boundary of the pull-rod base material and one of the three spot welds. Calculated crack propagation time for the case was determined to be considerable. The corrosion of the inner surface of the cylinder was the factor that accelerated the crack growth.

Occurrence of yet another rupture showed that the problem is more severe than was initially assumed. The crack growth calculations suggest that there exists a factor that accelerates the crack growth. Stress-corrosion appears to be the factor in question. Contribution of stress-corrosion to the crack growth process is very probable because the discussed pull-rod significant tensile stresses occured during the aircraft hangar standby. The Su-22 aircraft are stored in non-air-conditioned hangar-shelters. The presence of stress-corrosion influences the determination of the NDI inspection interval. Exact determination of the corrosion's influence on the rate of crack propagation in the pull-rod requires additional research.

Further service of the Su-22 fleet will require an extensive rate of NDI inspections. A test program has been launched to validate the inspection methods used in maintenance. Recent inspections detected further occurrences of cracks in the pull-rods in service. These pull-rods have been disassembled from the aircraft and a destructive test will be performed to verify the NDI results. Detection of further cracks is going to be expected. Crack detection should result in immediate decommissioning and replacement of the damaged pull-rods. However, purchase of new pull-rod elements and equipping them on the fleet aircraft will permit abandoning of the cumbersome NDT inspection schedule and shall contribute to safety improvement. Until the elements are replaced an intensive inspection program will be in force, which, as the authors hope, shall eliminate the risk of further pull-rod ruptures occurring on the planes in service.

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