FATIGUE LIFE IMPROVEMENT OF HIGH-STRENGTH MATERIALS BY SHOT PEENING

W. Schütz

Industrieanlagen-Betriebsgesellschaft mbH
Einsteinstraße 20, 8012 Ottobrunn, Federal Republic of Germany

INTRODUCTION

In the interest of weight saving, the fatigue strength of components must often be increased. This can be done either by increasing the allowable strain in the critical section or by decreasing the strain due to the external loads in the critical section, see Fig. 1.

The allowable strain can be increased by using a higher strength material. This, however has several disadvantages, among them an increased notch and mean stress sensitivity, faster crack propagation, lower residual static strength in the cracked condition, higher stress corrosion sensitivity, worse corrosion resistance etc. / 1, 2 /. Also, for manufacturing reasons, it is not easily possible to change over from a relatively low strength material to a high strength material.

So the second possibility, decreasing the strain in the critical section, is the better choice: This can be done by improved detail design, the most elegant method, which results in decreased maximum strain and strain amplitude. However, in many engineering structures practically all the potential improvements have already been exhausted, for example in the automobile and aircraft industries. Also, high stress concentration factors cannot be avoided altogether, because they are necessary for functional reasons, i.e. bolt-threads or cylindrical holes.

A decrease of the maximum strain in the critical section can also be achieved by generating residual compressive stresses, using thermal or mechanical methods. The former usually also produce a surface zone of high hardness, useful for improved wear resistance and for an increased fatigue limit. Both effects are welcome in, for example, crankshafts. However, if occasional overloads must be reckoned with or if crack-like defects are present, the hard surface zone may be the cause for sudden brittle failure. The mechanical methods of generating residual compressive stresses avoid this danger. They are therefore to be preferred for components which are loaded by the variable amplitude sequences so typical for most engineering structures, see Fig. 2.

It is generally assumed that residual compressive stresses affect fatigue strength like compressive mean stresses as long as they are not relaxed by the service stresses. The residual compressive stresses are relaxed when the material yields in the most highly loaded zone, e.g. in the notch. Consequently, the methods for generating residual compressive stresses will result in a large increase of fatigue life especially when the yield strength of the material is high and, at the same time, the stresses due to service loads are low. That is, considerable gains in fatigue life may be expected from residual compressive stresses especially in the high-cycle region of the S-N-curve of high strength materials; lower gains are to be expected from low-strength materials and in the finite-life region of the S-N curve, but especially under random service stresses.

A high yield strength of the material is essential for building up high residual compressive mean stresses. On the other hand, compressive residual stresses increase the fatigue life of high-strength materials much more than that of low-strength materials, as the results of several investigations have shown. So there are at least two reasons for applying residual compressive stresses especially to high-strength materials.

The magnitude and sequence of stresses due to external loads influence the relaxation of residual stresses. For this reason, realistic load sequences must be simulated if we want to determine quantitatively the gain in fatigue life by experiment.

Most components of aircraft and vehicles are subject to stochastically or deterministically variable loads in service. This means that only flight-by-flight or random tests will yield the results desired.

Many earlier investigations on the influence of residual stresses on fatigue strength were strictly of an ad-hoc nature; this means that the parameters for the improvement method considered suitable were subjectively chosen and not optimized with regard to maximum fatigue life improvement.

TEST PROGRAM

In the present work, shot peening for the fatigue strength improvement of aircraft components by residual compressive stresses was investigated. The objectives of the test program were threefold:

- To optimize shot peening for three high-strength materials under constant amplitude loads.
- Studying possible fatigue life increases by using the optimized methods in realistic load sequences and
- Determining the changes in residual stress during realistic load sequences.

As explained above, high-strength materials are especially suitable for methods generating residual compressive stresses. Therefore, in the present work, one especially high strength material each was selected from Al- , Ti and Fe-alloys, namely:

- AZ 74.61 (AlZn Mg Cu Ag, German aircraft material designation 3.4354.7, a silverbearing Al-alloy
- Ti 6 Al 6 V 2 Sn (3.7174.1) and
- X 2 NiCoMo 18/9/5 (1.6354.9) a maraging steel.

The tensile properties of the three materials can be seen in Fig. 3.

The four point bending specimen used is shown in Fig. 4. It is supposed to simulate unnotched structure loaded in bending. One may well question whether it is necessary to shotpeen unnotched surfaces at all; however in high strength materials even small surface imperfections may considerably decrease the fatigue strength. Therefore such components are usually shotpeened all over; for example, all steel forgings made of materials with a tensile strength above 1400 N/mm^2 must be shotpeened / 4 /.

Such specimens, shotpeened with several shotpeening parameters (i.e. different Almen intensities) were then tested at two different stress levels, one in the finite life region of the S-N curve, one near the fatigue limit, with a minimum of 4 specimens per parameter. The Almen intensity resulting in the highest fatigue life improvement and the lowest scatter was considered the optimum one.

About 10 specimens per material were then shotpeened with this optimum treatment and tested under a realistic flight-by-flight sequence simulating the stresses in the lower wing surface of a tactical aircraft. Fig. 5 shows a brief section from this sequence. One repeat period consisted of 200 flights with 17 different flights and an average of 71 cycles per flight. More details are given in / 3 /. The variable amplitude tests were performed at two different stress levels, with three or more specimens each.

The stress ratio, that is the ratio of the largest stress excursion into compression to that into tension was $\bar{R} = \frac{\bar{\sigma}_{min}}{\bar{\sigma}_{max}} = -0.25$.

The constant amplitude tests were also carried out at this stress ratio.

For comparison, constant and variable amplitude tests also had to be performed on untreated specimens.

The third objective of the investigation was measuring the changes of residual stress occurring in optimized specimens during a realistic test. We used destructive techniques because it is doubtful that at the moment nondestructive techniques are reliable or exact enough. Electrical discharge machining was employed, that is a variant to the boring-out method developed by Sachs in the thirties. However, destructive techniques are not suited to track the relaxation of residual stresses *in one specimen. For this reason, the residual stresses in one specimen had to be* measured before the beginning of the fatigue test, a second one after a short and a third one, after a longer life time. Since the absolute residual stress values differ among the specimens used, only qualitative conclusions can be drawn from the results.

SHOT PEENING TREATMENT

The specimens were shotpeened at MBB-UH in Hamburg. The shot size used was 0,4 mm, the shot hardness 48 HRC; the Almen intensities were 0,1, 0,2 and 0,3 A for the Al-specimens and 0,2 and 0,3 A für the Ti- and NiCoMo specimens. (The relevant MIL-Specification MIL-S-13165 B gives Almen values between 0,15 and 0,25 A for steel and titanium and a maximum of 0,25 A for aluminium at the material thickness of the specimen.) Coverage was $>$ 100 % in all cases. The complete surface of the specimen was peened. After the steel shot peening procedure all specimens were wet glass bead peened with about 1/3 of the peen intensities of the steel shot peening procedure to reduce surface roughness and to remove any broken steel shot fragments that might be imbedded in the surface.

RESULTS AND DISCUSSION

In general terms it can be stated that

- The greatest fatigue life improvement was obtained in the high cycle region of the S-N curve near the fatigue limit.
- Under a realistic variable amplitude load sequence life improvement was less in all three materials. This is a very important result. It means that one cannot quantitatively assess fatigue life improvement by constant amplitude tests if the component if stressed by variable loads in service.
- Shot peening did not increase the scatter of fatigue lives. This is important, since for industrial applications low scatter is a must. A method increasing mean fatigue life, but also increasing scatter, may actually lead to more failures in service, especially in mass-produced components.
- The favourable effect of the residual compressive stresses induced by shot peening cannot explain the improved fatigue life of the Aluminium specimens.

Now for some quantitative data:

The optimum peening intensity for all three materials was 0,3 A, as seen in the following table:

Peening intensity	Al-alloy		Ti-alloy		Maraging Steel	
Shot peening plus glass bead peening	$\pm \sigma_a \left[\frac{N}{mm^2}\right]$	Fatigue life improvement factor	$\pm \sigma_a \left[\frac{N}{mm^2}\right]$	Fatigue life improvement factor	$\pm \sigma_a \left[\frac{N}{mm^2}\right]$	Fatigue life improvement factor
0,1 A + 0,1 N	275 196	1,8 >13				
0,2 A + 0,2 N	275 196	2,3 17	464 380	43 >60	814 608	3,6 >20
0,3 A + 0,3 N	275 196	2,1 >21	464 380	37 49	814 608	10 >32

The fatigue life improvement factors above are valid for a probability of survival of 50 %. As such, for the Ti-alloy a peening intensity of 0,2 A was slightly superior to 0,3 A, see above table. However, the scatter of fatigue lifes was also greater, therefore 0,3 A was selected as optimum also for the Ti-alloy.

The realistic variable amplitud tests were therefore carried out with specimens shot peened to 0,3 A (plus glass bead peening to 0,3 N) in all three materials. The results of these tests are shown in the following table:

Material	Maximum bending stress if the spectrum $\bar{\sigma}_{max}$ [N/mm²]	Fatigue life improvement factor	Ratio of maximum bending stress $\bar{\sigma}_{max}$ to tensile strength F_{tu}
Al-alloy	490	2,7	0,92
Ti-alloy	1125	10,0	0,92
	1000	19,4	0,82
Maraging steel	2100	12,3	≈ 1,00
	1900	>11,2	0,91

As can be seen from this table, the fatigue life improvement factor was very large (10) for the Ti-alloy and the maraging steel, but only 2,7 for the Al-alloy, although the maximum stress of the load spectrum at $_{max}$ 0,9 was equal for the three materials. This can be explained by the decay of the residual compressive stresses in the Al-alloy, see chapter 5.

The above table also shows that the maximum bending stresses applied $_{max}$ were extremely high. This was necessary to obtain failures in a reasonable number of flights to failure.

Residual stress measurements were carried out across the thickness of shot-peened plane-bending specimens of aluminum, titanium and NiCoM
shown for aluminum. It can be seen that, for the unloaded specimens, the maximum residual compressive stress is located below the surface at less than 0.1 mm distance from the surface. This maximum has an absolute value within the range of tys /1.3 and tys /1.5. There is a high -stress gradient in the transition zone from compression to tension; this limits the compression zone to a thickness of about 0.1 mm. The residual compressive stress in aluminum specimens is eliminated after 800 flights (that is already after about 2 % of the fatigue life). On the other hand the residual compressive stress in titanium specimens is not relaxed even after 10.000 flights (that is, approximately 30% of the fatigue life) (see Fig. 7). The same goes for the maraging steel where the residual stresses practically did not change (Fig. 8).

So, the marked increase of the fatigue life of shot-peened titanium and NiCoMo specimens (in comparison to unpeened specimens) is most probably caused by the large and very stable residual compressive stresses; on the other hand, the rapidly relaxing residual compressive stresses in Aluminium cannot be the cause for its fatigue life improvement. A slightly positive effect on the fatigue life of Aluminium specimens may be caused by the increased material strength resulting from shot-peening: measurements showed a 10 per cent increase of hardness.

REFERENCES

/ 1 / Schütz, W.: Über eine Beziehung zwischen der Lebensdauer bei konstanter und bei veränderlicher Beanspruchungsamplitude und ihre Anwendbarkeit auf die Bemessung von Flugzeugbauteilen. Zeitschrift für Flugwissenschaften, 15 (1967), H. 11, S. 407 - 419

/ 2 / Schütz, W.: Werkstoffoptimierung für schwingbeanspruchte Bauteile. Zeitschrift für Werkstofftechnik 2 (1971), H. 4, S. 189 - 197

/ 3 / Weißgerber, S.: Einzelflugprogramme. Bericht Panavia 200-Nr. M-Fe 2170-1233

/ 4 / N.N.: Materials and Processes Requirements for Air Force Systems. MIL-STD 1587 (USAF) July 1976

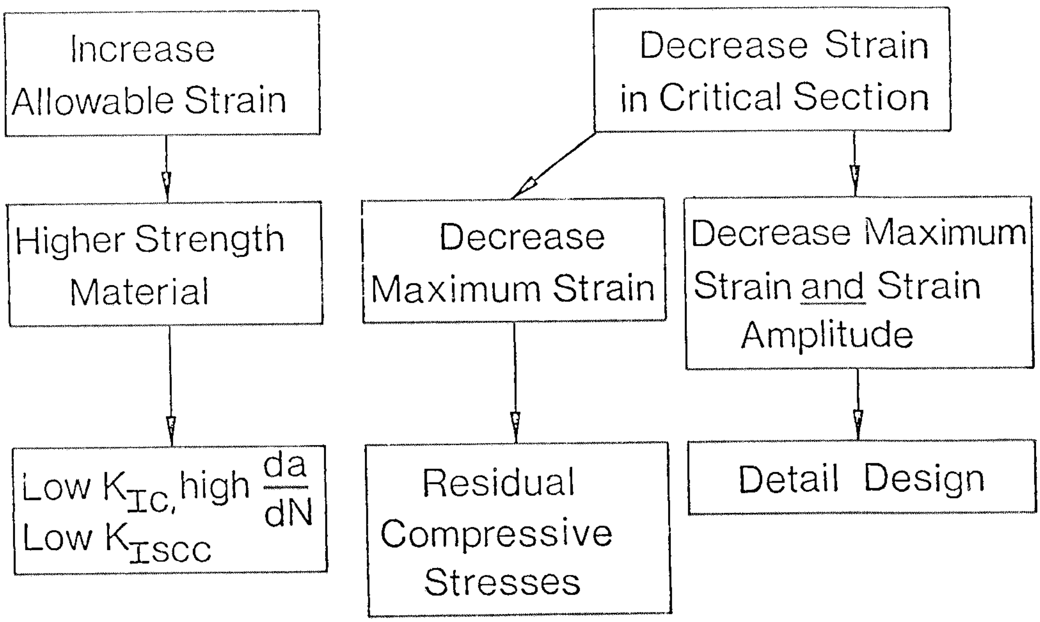


FIGURE 1: FATIGUE LIFE IMPROVEMENT

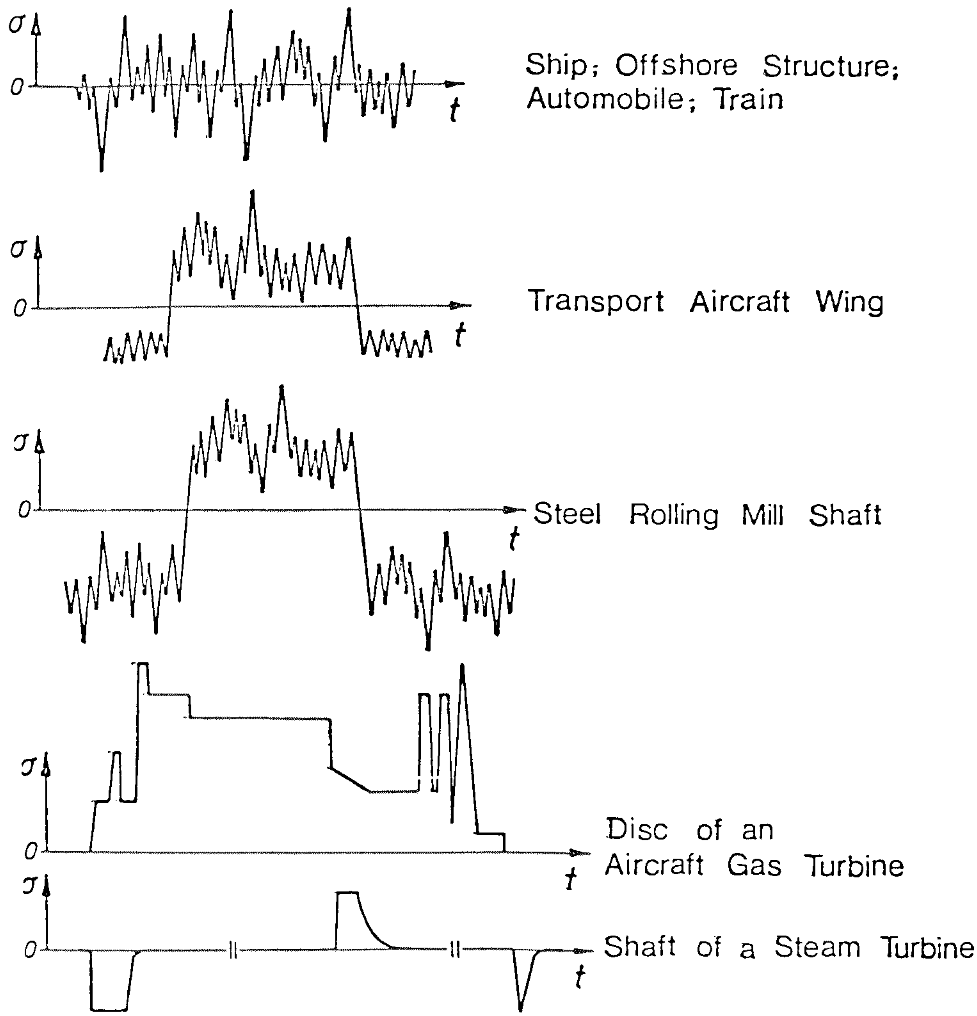
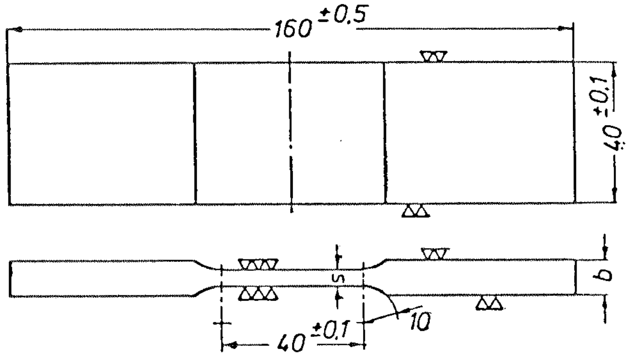


FIGURE 2: TYPICAL LOAD SEQUENCES (SCHEMATICAL)



Shot Peening

	Al	Ti	NiCoMo
s =	5 mm	4 mm	5 mm
b =	9 mm	9 mm	10 mm

FIGURE 3: SPECIMEN

AlZnMgCuAg *uts* = 531 N/mm^2; *tys* = 487 N/mm^2

Ti 6Al 6V 2Sn *uts* = 1250 N/mm^2; *tys* = 1151 N/mm^2

18/9/5 NiCoMo *uts* = 2125 N/mm^2; *tys* = 2076 N/mm^2

FIGURE 4: MATERIALS TESTED

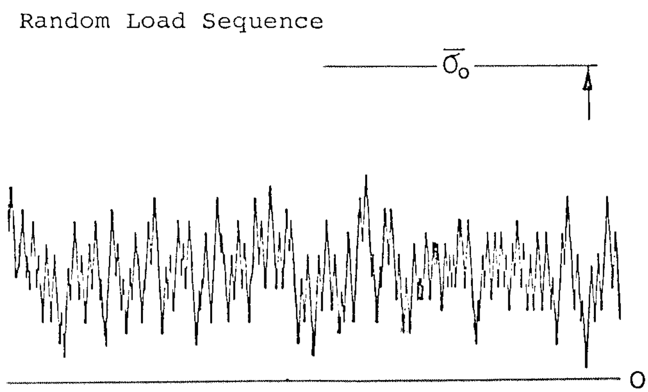


FIGURE 5: LOAD SEQUENCE

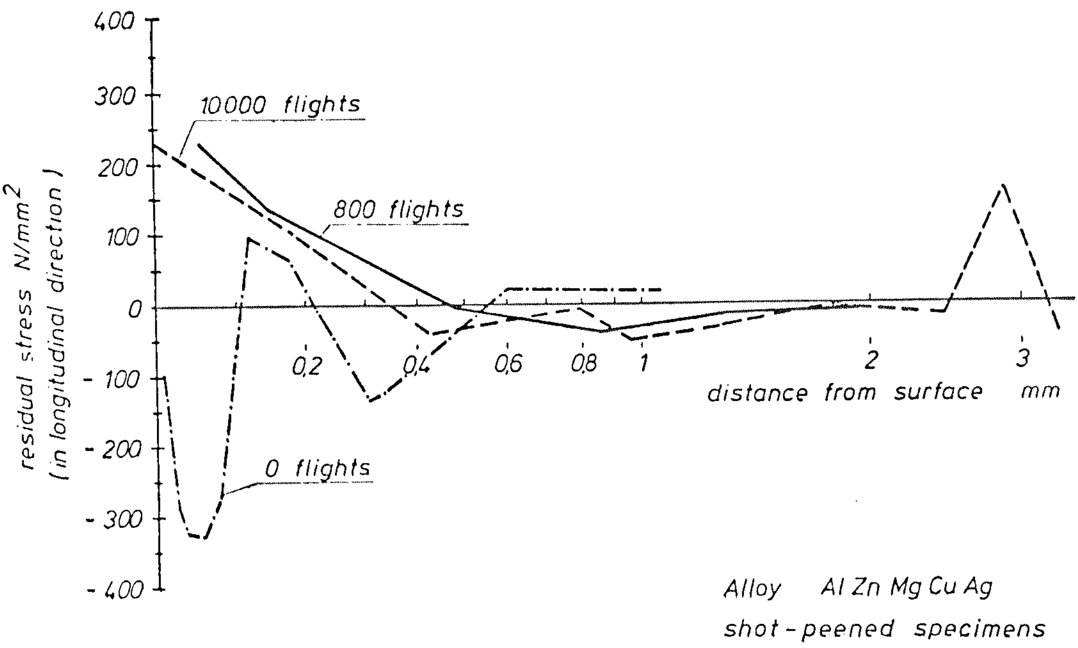


FIGURE 6: CHANGE OF RESIDUAL STRESSES
DURING FLIGHT-BY-FLIGHT TESTS

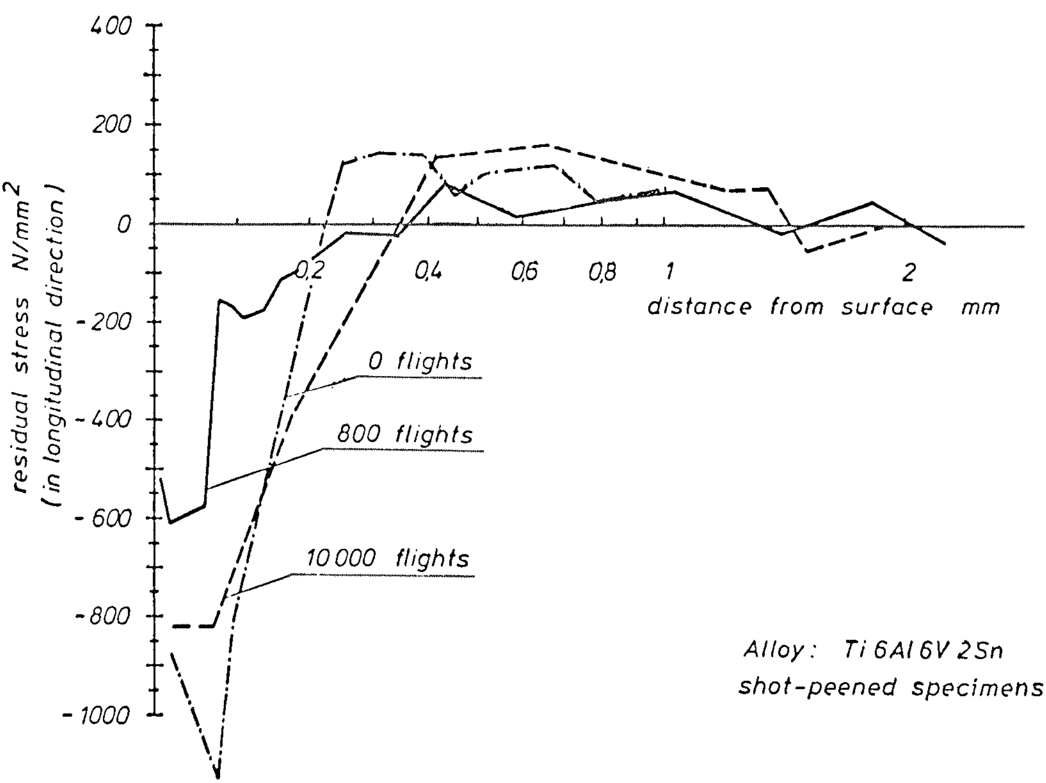


FIGURE 7 : CHANGE OF RESIDUAL STRESSES DURING FLIGHT-BY-FLIGHT TESTS

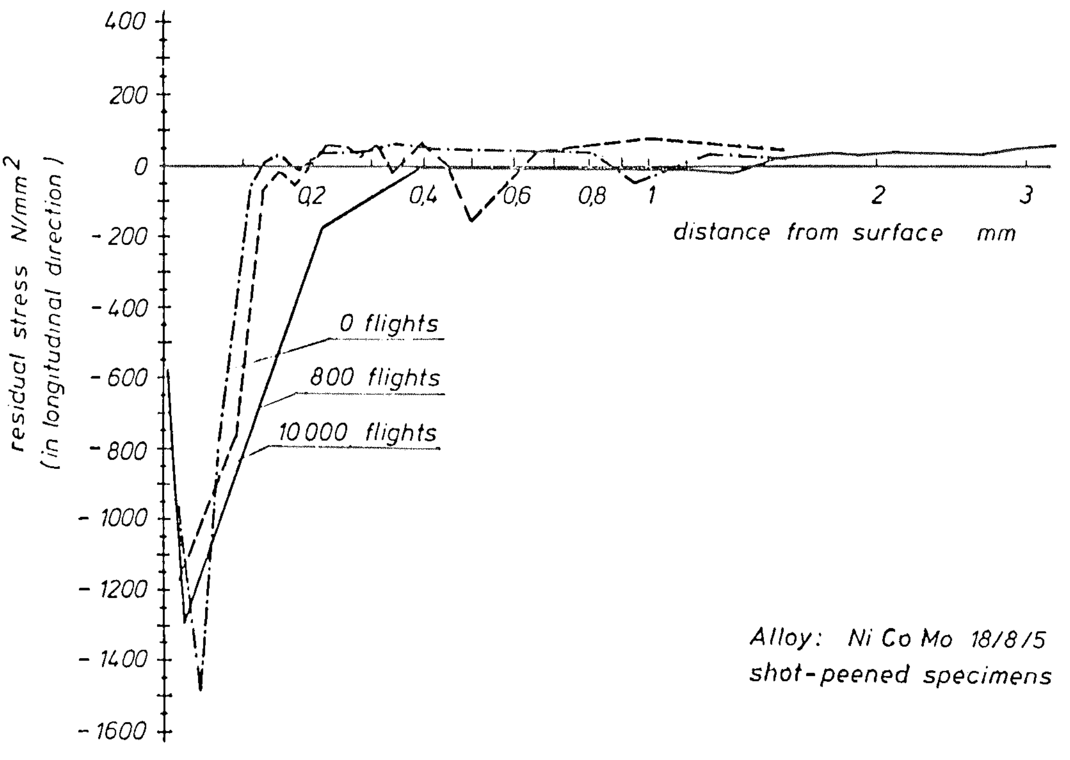


FIGURE 8 : CHANGE OF RESIDUAL STRESSES DURING FLIGHT-BY-FLIGHT TESTS