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THE EFFECT OF SECONDARY METALWORKING PROCESSES ON SUSCEPTIBILITY OF AIRCRAFT TO CATASTROPHIC FAILURES AND PREVENTION METHODS

ANALIZA WPŁYWU PROCESÓW WTÓRNEGO KSZTAŁTOWANIA MATERIAŁÓW NA SKŁONNOŚĆ SAMOLOTÓW DO USZKODZEŃ KATASTROFICZNYCH ORAZ NA DOBÓR METOD ZAPOBIEGANIA TYM USZKODZENIOM

The causes of plane crashes, stemming from the subcritical growth of fatigue cracks, are examined. It is found that the crashes occurred mainly because of the negligence of the defects arising in the course of secondary metalworking processes. It is shown that it is possible to prevent such damage, i.e. voids, wedge cracks, grain boundary cracks, adiabatic shear bands and flow localization, through the use of processing maps indicating the ranges in which the above defects arise and the ranges in which safe deformation mechanisms, such as deformation in dynamic recrystallization conditions, superplasticity, globularization and dynamic recovery, occur. Thanks to the use of such maps the processes can be optimized by selecting proper deformation rates and forming temperatures.

Keywords: aircraft, materials, cracking, prevention, workability optimization

Dokonano analizy przyczyn katastrof lotniczych wskutek podkrytycznego rozwoju pęknięć zmęczeniowych. Stwierdzono, że do katastrof tych dochodziło głównie z powodu nieuwzględniania uszkodzeń powstających podczas wtórnego kształtowania materiałów. Wykazano, że istnieje możliwość eliminowania takich uszkodzeń, to jest powstawania: pustek, pęknięć klinowych, pęknięć międzykrystalicznych oraz adiabatycznych pasm ścinania i lokalizacji odkształceń. Wymaga to jednak stosowania map procesów kształtowania, określających zakresy występowania w/w uszkodzeń oraz zakresy występowania bezpiecznych mechanizmów odkształceń, do jakich należy odkształcanie w warunkach rekrystalizacji dynamicznej, superplastyczności, globularyzacji i zdrowienia dynamicznego. Stosowanie takich map umożliwia optymalizację procesów poprzez dobór odpowiedniej prędkości odkształceń i temperatury kształtowania.

1. Introduction

Materials to be used in aircraft should be characterized by high strength (yield point R_{pl}), high rigidity (elastic modulus E), high resistance to catastrophic crack growth (K_{IC}) and low density ρ. Owing to the high yield point and low density of the material one can reduce the weight of the aircraft and increase its payload capacity. However, as the material's yield point increases, its resistance to catastrophic crack growth decreases whereby the aircraft becomes more susceptible to failure as a result of the rapid development of fatigue cracks or cracks which arose in the course of manufacturing the aircraft. There is little awareness of this danger despite the fact that as early as after World War II the need to adopt other than the conventional design criteria was highlighted. The conventional criteria took into account only material strength and a safety factor. The innovation consisted in taking into account fatigue damage and attaching key (for aircraft safety) importance to them.

It was soon found, however, that even the most thorough laboratory testing of the input materials did not guarantee aircraft safety. It turned out that one of the causes of premature plane crashes was the unexpected presence and accelerated propagation of cracks which had arisen in the course of processing the input materials into aircraft parts.

Hence the principal aim of this publication is to shed light on the danger stemming from neglecting the damage (cracks) which may arise during the postforming of the materials used in aircraft manufacture. There are many indications that the knowledge of the causes of such damage and their criticality and the ways of preventing it is not common. This observation mainly applies to aircraft parts manufacturers who may have difficulties in optimizing the plastic forming of atypical materials such as novel titanium alloys.

2. Material fatigue and catastrophic crack growth as the basis for evolution of aircraft design criteria

Until 1958 the U.S. Air Force (USAF) had no formal fatigue requirements [1]. Aircraft had been generally designed on the basis of only static strength considerations and the applied safety factor had been expected to account for deterioration from usage and quality problems for uncertainties about loading and material strength (Fig. 1).

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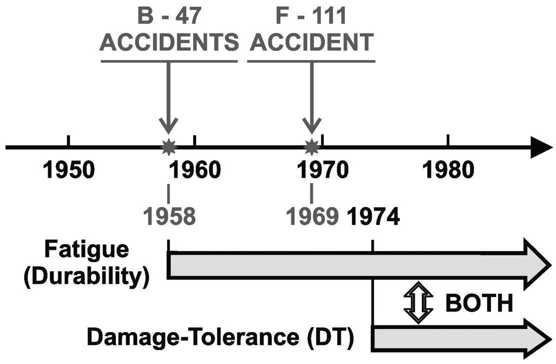

Fig. 1. Key events leading to the adoption of damage tolerance requirements by USAF: (F-111). A full-scale fatigue test of the wing box indicated 16 000 flight hours, a safe life of 6000 hours, whereas the total time in service at the time of the accident was 100 hours. The failure was attributed to a defect which had arisen during the manufacture of the forgings from which the plane was fabricated. The anomalous fatigue resulted in wing separation. Based on [1]

In the late 1960s and early 1970s a number of aircraft structural failures occurred both during testing and in service [2]. Some of those failures were attributed to flaws, defects or discrepancies that were either inherent or induced during the manufacturing and assembly of the structure. The presence of such flaws had not been accounted for in the design. The design had been based on a "safe life" fatigue analysis. Mean life predictions had been based on materials unflawed fatigue test data and on a conventional fatigue analysis. A scatter factor of four had been used to account for the initial quality, the environment, variation in material properties, and so on. This conventional fatigue (safe life) analysis had not adequately accounted for the presence and development of such flaws.

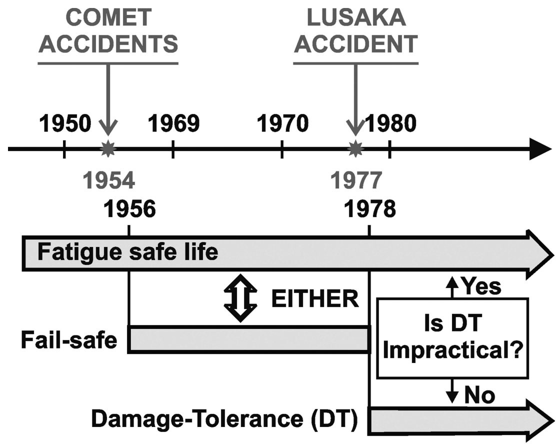

Fig. 2. Federal Aviation Administration (FAA) key events which led to adoption of damage tolerance requirements. In the case of the Lusaka crash, unexpected normal fatigue led to the separation of the horizontal stabilizer. Based on [1]

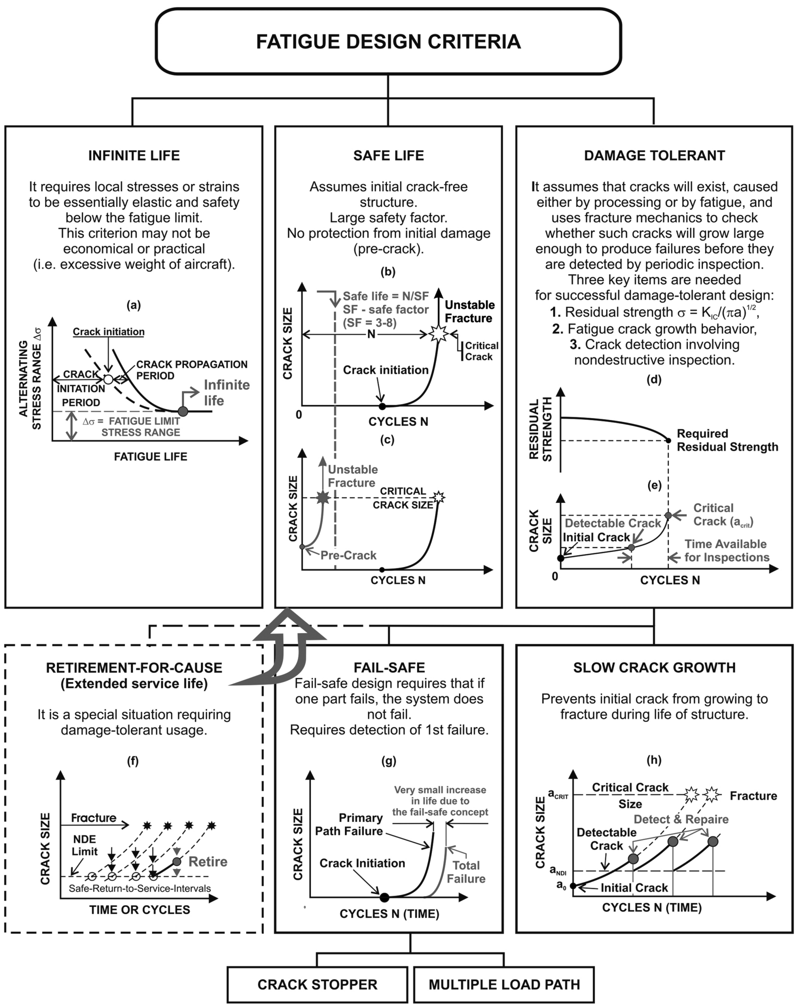

Fig. 3. Fatigue design criteria. Based on [3]

In order to ensure the safety of the aircraft structure, beginning from the mid 1970s the U.S. Air Force adopted a damage tolerance design approach, replacing the conventional fatigue design approach. The damage tolerance (or fracture control) approach assumes that flaws are initially present in the structure. The structure must be designed in such a way that these flaws do not grow to a critical size and cause a catastrophic failure of the structure within a specified period of time.

History indicates that, similarly to the USAF experience, fatigue was not a major issue in early civil aircraft [1]. As civil aircraft designs became more challenging (e.g. pressurized fuselages) fatigue events became more common (Fig. 2).

The principal differences between the particular aircraft design criteria when fatigue is taken into account are illustrated in Fig. 3.

Figure 3 shows that the damage-tolerant criterion is today the highest currently attainable guarantee of safety. In order to use this criterion one must:

1. know the resistance of the material to catastrophic crack growth (K_{IC}) and consequently, the diagram of residual strength $\sigma = K_{IC}/(\pi a)^{1/2}$;
2. know the rate of crack growth over time (the number of cycles): a = f(N) – Fig. 1e;
3. be able to detect and remove or reduce a crack ($a \geqslant a_{NDI}$) by means of nondestructive methods, before it reaches critical length a_{CRIT} (Fig. 3h).

The use of novel high strength materials characterized by relatively low crack resistance introduces additional problems, i.e. cracks grow fast while the remaining strength decays rapidly (Fig. 4).

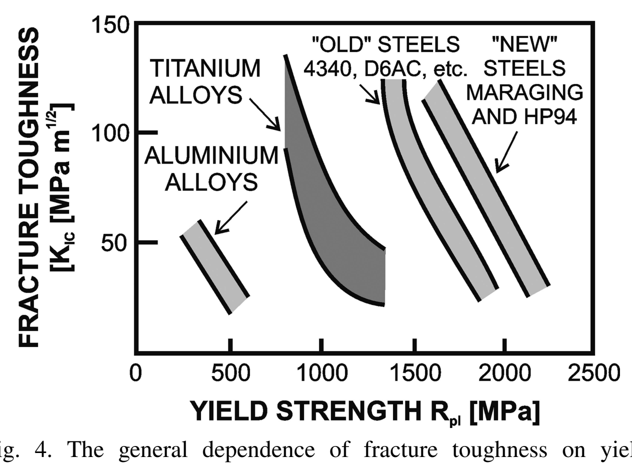

Fig. 4. The general dependence of fracture toughness on yield strength. Based on [4]

In certain structures the material may contain initial flaws serious enough to cause cracking immediately after the structure is put into service (Fig. 1 – an F-111 accident due to a forging fold in high strength steel D6ac). In this case, the flaw had passed undetected through inspection during manufacture and grew to a critical size after only 100 flying hours.

The problem of the premature and rapid growth of cracks in materials with flaws arising during the manufacture of aircraft parts can be solved provided that the aircraft parts subcontractors (often haphazardly selected) are made aware of the problem and understand its causes.

3. Consequence of damage arising during postforming of materials

In order to understand well the principles of applying the damage-tolerant (DT) criterion one needs to know what determines the time until failure, i.e. the time after which a crack reaches critical length a_{CRIT}. The problem is that this length is not an inherent characteristic of the material. This means that for the same crack growth resistance KIC one can obtain different critical crack length values since the latter significantly depend on the design stress level (Fig. 5).

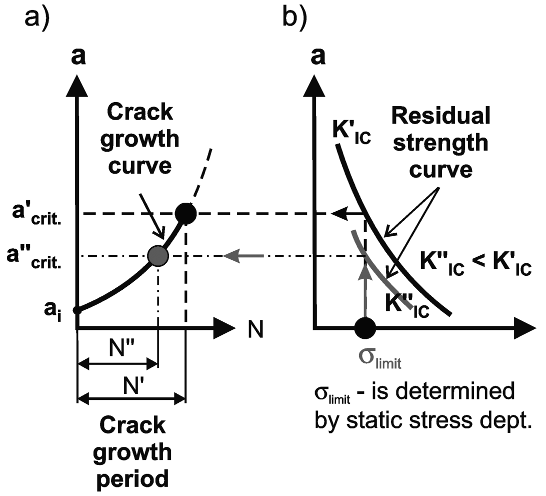

Fig. 5. The method of determining critical crack length a_{CRIT} and crack development time (in cycles N) depending on adopted service stress level σ_{LIMIT}

The above is further complicated by the fact that critical crack length decreases as strength (plastic limit) increases and an increase in strength means a decrease in crack resistance K_{IC} (Fig. 5 – K_{IC}**). All of this contributes to a significant reduction in crack propagation time and consequently, the time needed to detect a crack and take appropriate measures.

One of the ways of extending the controlled crack propagation time is to increase the sensitivity of the nondestructive testing methods by lowering the crack detectability threshold (reducing a_d – Fig. 6a) and/or reducing the rate of crack propagation (Fig. 6b).

As it appears from Fig. 6a, a twofold reduction in detectable crack length a_d results in a much greater extension of the crack propagation time than a twofold increase in critical crack length a_{CRIT}.

It should be emphasized, however, that the effort put into the improvement of the sensitivity of the nondestructive testing methods may bring about meagre results if the possible accelerated growth of cracks introduced by the manufacturing processes is neglected (Fig. 7).

It appears from Figure 7 that if the initial crack length is similar or larger than the length detectable by the nondestructive testing methods, a failure will occur anyway since the rate of crack growth is the higher, the longer the initial crack length. It should be noted, however, that this does not have to be so if any cracks arising in the manufacture of aircraft parts are eliminated or their length is considerably reduced, as indicated by the location of the curves corresponding to the ever shorter initial cracks (Fig. 7). It is particularly necessary to pay attention to the length of the initial cracks in the case of high-strength materials such as steels and titanium alloys (Fig. 8).

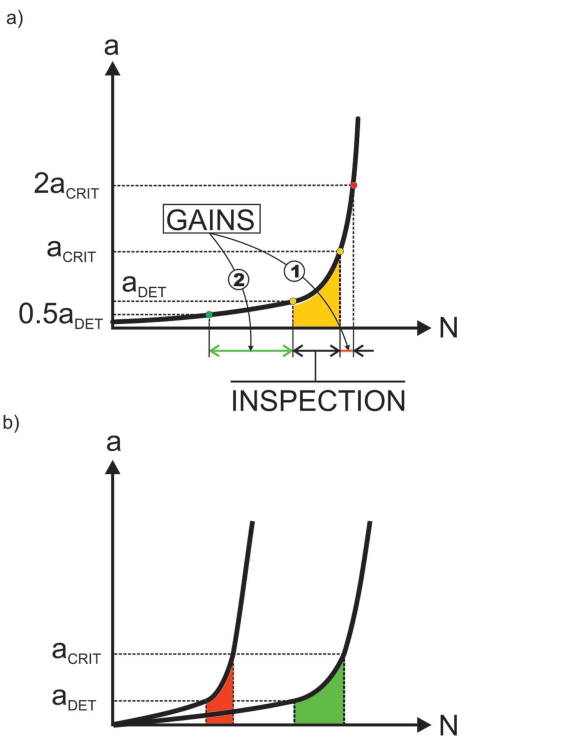

Fig. 6. The effect of various parameters on fracture safety: a) an increase in toughness and a decrease in the minimum detectable crack size, b) a reduction in the crack growth rate. Based on [5]

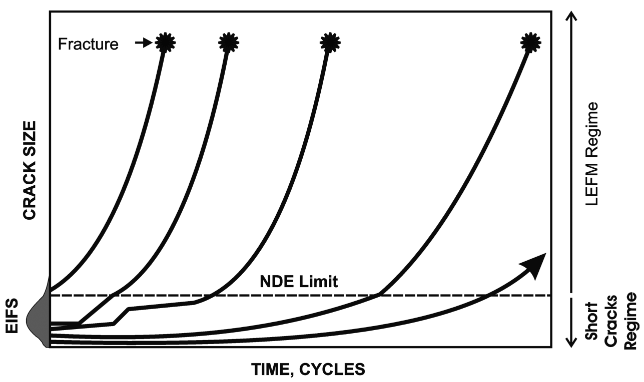

Fig. 7. Total fatigue life segmented into stages of crack growth. The variation in equivalent initial flaw size (EIFS) and small crack behaviour dominates total fatigue life. Based on [6]

According to Figure 8, the longer the initial crack in steel and in titanium alloysis, the fewer cycles the structural component containing them will withstand.

The problem is that not every aircraft parts manufacturer is aware that it is possible to prevent cracks by optimizing the manufacturing process parameters. In the case of advanced materials, to which titanium alloys belong, one needs to have maps of the processes of forming them [8]. On the basis of such maps one can optimize the processes and prevent damage having the form of various cracks or their initiators.

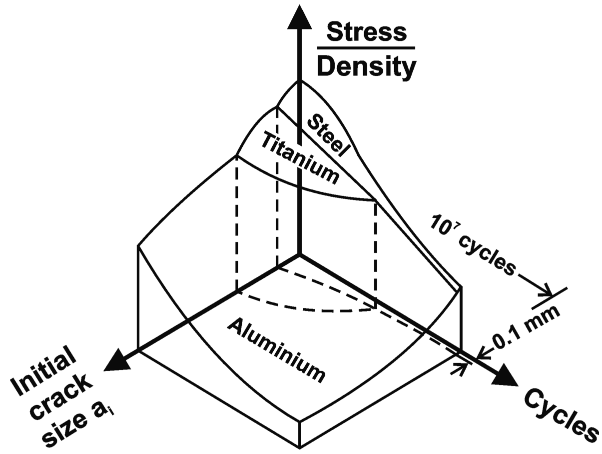

Fig. 8. A comparison of life expectancy. Based on [5]

4. Processing maps as way of preventing damage during postforming of materials

Even though titanium alloys have been increasingly used in aircraft construction for some time now, their plastic forming still poses difficulties. Processing maps are a great help in optimizing the processes of the plastic forming of titanium alloys [8]. The final form of such a map depends on the kind of the crystal lattice of a given alloy. At room temperature commercially pure titanium has a hexagonal close-packed (hpc) structure referred to as the alpha (α) phase. At a temperature of 880°C the alpha phase changes into a beta phase with a body-centred cubic (bcc) structure. A wide range of possibilities can be obtained by modifying the crystallographic forms through the addition of alloying agents, and thermomechanical treatment. The titanium alloys obtained this way can be divided into alpha (α), beta (β) and alpha plus beta ($\alpha + \beta$) alloys.

Alpha alloys are less forgeable than beta alloys. Because of their poorer forgeability they are more likely to develop defects in the course of forging. This problem can be minimized by the periodic soaking of the material.

Alfa + beta alloys are a mixture of phase α and phase β and at room temperature they may contain 10-50% of beta phase. The most widely known $\alpha + \beta$ alloy is the Ti-6Al-4V alloy.

Beta alloys are easily forgeable in a wide range of temperatures. Moreover, these alloys are more resistant to cracking, which is of critical importance from the damage-tolerant criterion point of view.

Figures 9-11 show processing maps for the most often used titanium alloy, i.e. Ti-6Al-4V.

The maps include areas representing the processes safe for the structure to be obtained. The recommended safe mechanisms here include: dynamic recrystallization, superplasticity, globularization and dynamic recovery. Conditions in which voids, wedge cracks, grain boundary cracks, adiabatic shear bands and flow localization arise should be avoided. On the basis of the processing maps it is possible not only to arrive at conditions for optimizing hot workability, but also to determine the conditions conducive to the onset of defect-generating mechanisms or to flow instabilities.

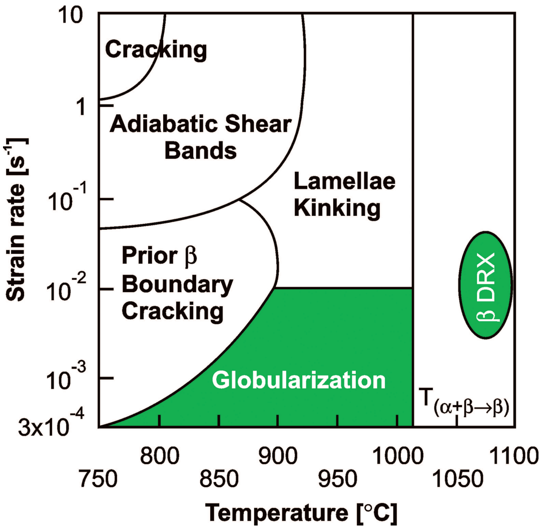

Fig. 9. A processing map for the hot working of commercial grade Ti=6Al-4V with a transformed β starting microstructure, illustrating microstructurally "safe" and damage mechanisms. Microstructures: bottom left – prior β boundary cracking, bottom right – globularization, top left – adiabatic shear band cracking, top right – lamellae kinking. Based on [7]

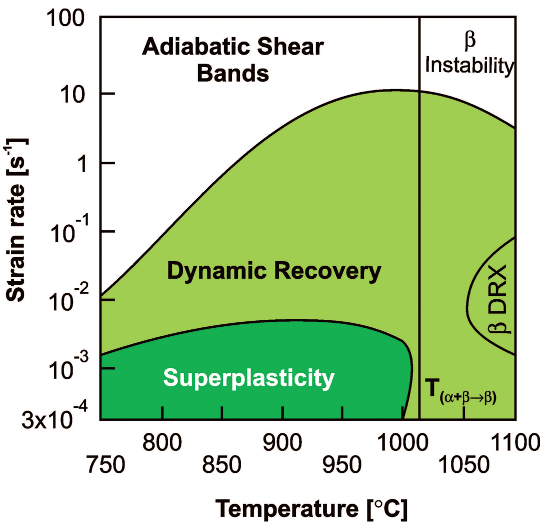

Fig. 10. A processing map for the hot working of commercial grade Ti-6Al-4V with an equiaxed $(\alpha + \beta)$ starting microstructure. Based on [8, 9]

It is worth noting that Ti-6Al-4V ELI has the highest crack resistance (K_{IC}) but it is the least workable alloy. The optimum processing parameters for this alloy are: the roughing temperature of 925°C and the strain rate of $0.001 s^{-1}$. If this temperature is reduced too much, this may result in intercrystalline cracking, whereas a slight increase in this temperature will cause the development of voids. One cannot allow the strain rate to increase above 10^{-2} s^{-1} (10^{-1} s^{-1}). In practice, this necessitates forging with a slow-speed hydraulic press (Fig. 12).

It appears from Figures 8, 11 and 12 that titanium alloys whose initial structure has the form of phase β should not be forged using machines faster than slow-speed hydraulic presses. Hammer forging is out of the question here, which for many titanium alloy forging manufacturers is not so obvious as it would seem.

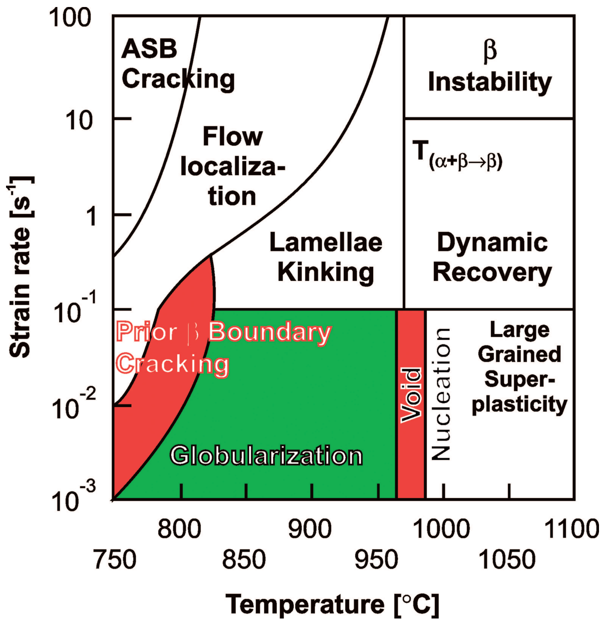

Fig. 11. The design and optimization of the cogging process for ELI grade Ti-6Al-4V. The processing map for the transformed β starting microstructure, showing the preferred window of globularization (green) and its limits of prior β boundary cracking at lower temperatures. The microstructure on the left shows prior beta boundary cracking at lower temperatures. The microstructure on the right shows void nucleation and growth at higher temperatures. Based on [8, 10]

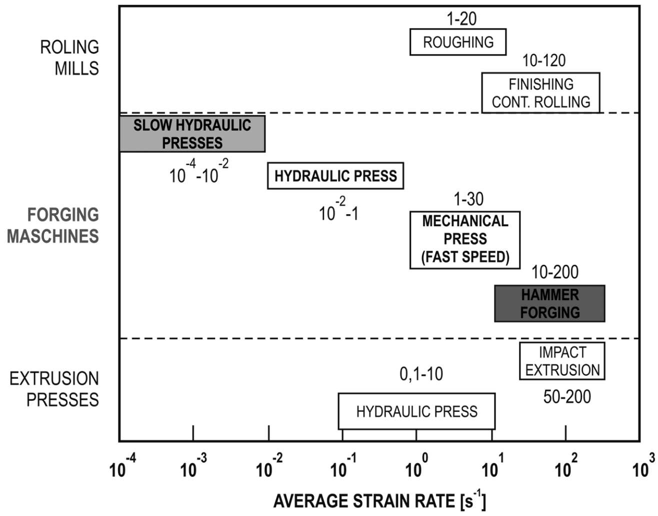

Fig. 12. Strain rates typical for the machines used for the plastic forming of metals, including forging machines. Based on [11]

5. Conclusions

This analysis of the aircraft design criteria and of the consequences of the defects arising in the course of the post-forming of materials has shown that:

1. Designing according to the damage-tolerant criterion offers the greatest chances of avoiding plane crashes caused by the subcritical growth of cracks, particularly cracks (not taken into account before) arising in the processes of manufacturing aircraft parts.
2. The use of such advanced materials as titanium alloys, characterized by high strength and low crack resistance,

necessitates the avoidance of any cracking, particularly in the course of the postforming of the materials.

3. Pre-cracks propagate faster than the cracks which nucleate when the aircraft is already in service. This accelerated propagation of pre-cracks significantly (as much as tens-fold) reduces aircraft safe service life.
4. The problem of cracks arising in the course of postforming can be solved through the use of processing maps.
5. As the presented processing maps show, titanium alloys characterized by greater crack resistance (K_{IC}) should be forged using slow-speed hydraulic presses, i.e. at strain rates of about $10^{-3}s^{-1}$. This particularly applies to alloys whose initial structure has the form of phase β.

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