AL-ZN-MG-CU ALLOYS AND PRODUCTS WITH HIGH MECHANICAL CHARACTERISTICS AND STRUCTURAL MEMBERS SUITABLE FOR AERONAUTICAL CONSTRUCTION MADE THEREOF

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(57) ABSTRACT

The present invention further relates to 7xxx alloys and products produced therewith that can be flat rolled, extruded or forged, as well as associated methods. Al—Zn—Mg—Cu alloys of the present invention preferably comprise (in mass percentage):

a) Zn 8.3-14.0 Cu 0.3-2.0 Mg 0.5-4.5 Zr 0.03-0.15 Fe+Si<0.25

b) at least one element selected from the group consisting of Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y and Yb, where the content of each of the elements, if selected, is between 0.02 and 0.7%,

c) remainder aluminum and inevitable impurities. The present invention further is directed to products wherein

Mg/(Cu+Mg)>2.4 and (7.9-0.4 Zn)>(Cu+Mg)>2.4 and (7.9-0.4 Zn). The disclosed products can be used for example, as structural members in aeronautical construction, especially as stiffeners capable for use in fuselages of civilian and other aircrafts as well as in related applications.
Figure 3
Figure 4
Figure 5
Figure 7
Figure 8
Figure 9

\[ a = 1.8 \text{mm} \]
\[ Lf = D + 3a \]
Figure 12
AL-ZN-MG-CU ALLOYS AND PRODUCTS WITH HIGH MECHANICAL CHARACTERISTICS AND STRUCTURAL MEMBERS SUITABLE FOR AERONAUTICAL CONSTRUCTION MADE THEREOF

CLAIM FOR PRIORITY


BACKGROUND OF THE INVENTION

[0002] 1. Field of the Invention

[0003] The present invention relates generally to Al—Zn—Mg—Cu type alloys with high mechanical characteristics, typically having a Zn content greater than 8.3%, as well as to products and structural members suitable for aeronautical products and/or constructions manufactured from such products.

[0004] 2. Description of the Related Art

[0005] Al—Zn—Mg—Cu alloys (belonging to the family of 7xxx alloys) are currently in use in aeronautical construction, and are particularly used in the construction of civilian aircraft wings. For the wing exterior, a skin of plate made in alloys such as 7150, 7055, 7449 is often used, and optionally includes stiffeners (also called stringers) made from profiles in 7150, 7055, or 7449 alloys. 7150, 7050 and 7349 alloys are also commonly used for the manufacture of fuselage stiffeners or stringers.

[0006] Some of these alloys have been known for decades, such as for example 7075 and 7175 alloys (zinc content between 5.1 and 6.1% by weight), 7050 (zinc content between 5.7 and 6.7%), 7150 (zinc content between 5.9 and 6.9%) and 7049 (zinc content between 7.2 and 8.2%). Such alloys have a high tensile yield strength, as well as good fracture toughness and good resistance to stress corrosion and to exfoliation corrosion. More recently, it has appeared that for certain applications the use of an alloy with a higher zinc content can have advantages, such that such an alloy allows the tensile yield strength to be increased further. 7349 and 7449 alloys contain between 7.5 and 8.7% zinc. Wrought alloys still higher in zinc have been described in the literature, but have not been deemed in aeronautical construction.

[0007] The article “Microstructure and Properties of a New Super-High-Strength Al—Zn—Mg—Cu alloy C912” by Y. L. Wu et al., published in Materials & Design, vol 18, p. 211-215 (1998) describes an alloy including Zn 8.7%, Mg 2.6%, Cu 2.5%, Si and Fe each <0.05%, which is for the manufacture of structural members for wings and fuselage.

[0008] U.S. Pat. No. 5,560,789 (Pechiney) discloses an alloy including Zn 10.7%, Mg 2.84% and Cu 0.92% which is transformed by extrusion. The alloying elements of this alloy are very high in zinc, magnesium and copper, and as such, are difficult to put into solid solution because the temperature of solution heat treatment is limited by the melting point of intermetallic phases, which have the lowest incipient melting point. As a consequence, products formed with such an alloy have a high mechanical strength, but a very low elongation at fracture due to the presence of coarse precipitates. Thus, such a product has a low formability.

[0009] U.S. Pat. No. 5,221,377 (Aluminum Company of America) discloses several Al—Zn—Mg—Cu alloys with a zinc content of up to 11.4% and with a rather high copper content. These alloys are difficult to cast and the alloying elements are difficult to put into solid solution, which favours the presence of coarse precipitates, which are not welcome.

[0010] Moreover, it has been proposed to utilise Al—Zn—Mg—Cu alloys with a high zinc content to manufacture hollow bodies intended to resist high pressures, such as for example cylinders for compressed gas. European Patent Application No. EP 020 282 AI (Société Métallurgique de Gerzat) discloses alloys with a zinc content of between 7.6% and 9.5%. European Patent Application No. EP 081 441 AI (Société Métallurgique de Gerzat) discloses a process for obtaining such cylinders. European Patent Application No. EP 257 167 A1 (Société Métallurgique de Gerzat) states that none of the known Al—Zn—Mg—Cu alloys can safely and reproducibly satisfy the strict technical demands imposed by this specific application. This document proposes moving towards a lower zinc content, namely between 6.25% and 8.0%.

[0011] The teaching of the above described documents is specific to the problem of cylinders for compressed gas, particularly concerning maximizing the bursting pressure of these cylinders, and thus, is not relevant to other wrought products because the required mechanical properties would be completely different, among other things.

[0012] In general, in Al—Zn—Mg—Cu alloys, a high zinc content, and also a high Mg and Cu content are typically required in order to obtain good static mechanical characteristics (ultimate tensile strength (Rm) or UTS) and tensile yield strength (Rpo.2 or TYS), but this is only possible if these elements can be put into solid solution. It is also well known (see for example U.S. Pat. No. 5,221,377) that discloses that when the zinc content of an alloy of the 7xxx family is increased beyond around 7% to 8%, then problems associated with insufficient resistance to exfoliation corrosion and stress corrosion will arise. More generally, it is known that the most charged Al—Zn—Mg—Cu alloys are likely to pose corrosion problems. These problems are generally resolved by using particular thermal or thermo-mechanical treatments, especially by pushing the aging treatment beyond the peak, for example during T7 type treatment. But such treatments can then cause a corresponding drop in the static mechanical characteristics. In other words, for a minimum level of resistance to corrosion to be obtained, when optimising an Al—Zn—Mg—Cu alloy one must find a compromise between static mechanical characteristics (TYS Rpo.2, UTS Rm, elongation at fracture A) and damage tolerance characteristics (fracture toughness, crack propagation rate etc.). According to the minimal level of resistance to corrosion to be envisaged, either (i) a temper close to peak strength is utilised (T6 temper), which generally offers (ii) acceptable toughness—TYS compromise favouring static mechanical characteristics, or (iii) annealing is pushed beyond peak strength (T7 tempers), by seeking a compromise favouring fracture toughness.

[0013] Whichever approach is used, the manufacture and use of such products poses at least two problems. On one
hand, alloys with a high zinc and magnesium content are difficult to cast and to transform, especially by extrusion, rolling or forging. For example, the maximum force that can be supplied by an extrusion press can be a limiting factor. In particular, among 7xxx series alloys, 7349 and 7449 alloys require very high extrusion forces. On the other hand, for certain applications, the formability of rolled and extruded products is critical. This is especially true with respect to fuselage stringers. Therefore, up until now, alloys that could be developed having a mechanical strength still higher than 7349 and 7449 alloys would likely be difficult to cast and to transform, and products made therewith would tend to have a low formability.

SUMMARY OF THE INVENTION

A problem which the present invention attempted to resolve was therefore to obtain a novel alloy and associated novel wrought Al—Zn—Mg—Cu type products with a high zinc content, (i.e. greater than 8.3%) and especially extruded products, as well as their associated methods. Products of the present invention preferably possess a very high ultimate tensile strength (UTS), a very high tensile yield strength (TYS) as well as adequate resistance to corrosion and a high formability, and also are capable of being manufactured industrially under conditions of highest reliability compatible with the severe requirements of the aeronautical industry.

The present inventors have found that these and other problems can be addressed inter alia by finely adjusting the concentration of Zn, Cu and/or Mg as well as controlling the content of certain impurities (particularly Fe and Si), and further by optionally adding other elements.

In accordance with these and other objects, one embodiment of the present invention is directed to an Al—Zn—Mg—Cu alloy that can be rolled, extruded and/or forged, characterised in that it comprises (in mass percentage):

a) Zn 8.3-14.0 Cu 0.3-2.0
b) at least one element selected from the group consisting of Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y, Yb, Cr and Mn, the content of each of said elements, if selected, being between 0.02 and 0.7%,
c) remainder aluminum and inevitable impurities,

and wherein

- Mg/Cu>2.4 and
- (7.9-0.4 Zn)>(Cu+Mg)>(6.4-0.4 Zn).

In yet further accordance with the present invention, there is provided another embodiment directed to a structural member for aeronautical construction incorporating at least one product, and particularly to a structural member suitable for the construction a fuselage of an aircraft, such as a fuselage stringer.

Additional objects, features and advantages of the invention will be set forth in the description which follows, and in part, will be obvious from the description, or may be learned by practice of the invention. The objects, features and advantages of the invention may be realized and obtained by means of the instrumentalities and combination particularly pointed out in the appended claims.

BRIEF DESCRIPTION OF FIGURES

FIG. 1 shows the section of profile T1 according to an embodiment of the present invention.
FIG. 2 shows the section of profile T2 according to an embodiment of the present invention.
FIG. 3 shows the section of profile T3 according to an embodiment of the present invention.
FIG. 4 shows the section of profile T4 according to an embodiment of the present invention.
FIG. 5 shows the section of profile T5 according to an embodiment of the present invention.
FIG. 6 diagrammatically illustrates the zone of a fuselage stringer which has been formed by jogging according to an embodiment of the present invention.
FIG. 7 diagrammatically illustrates the location of sampling on profile T1, where a test piece for the 3 point bending test is cut.
FIG. 8 diagrammatically illustrates the definition of the bending angle.
FIG. 9 diagrammatically illustrates the important parameters of the 3 point bending test.
FIG. 10 diagrammatically illustrates a two stringer bay crack.
FIG. 11a and 11b diagrammatically illustrate a buckling test.
FIG. 12 compares the predicted crippling stress for different Z stringers according to the invention (grey bars) and according to prior art (white bars) for the same geometry.
DETAILED DESCRIPTION OF THE INVENTION

[0048] In FIGS. 1, 2, 3, 4 and 5, the dimensions are approximate values expressed in millimetres. In FIGS. 1, 2, 3 and 4, letter (a) designates the foot section, and letter (b) the top section of the profile.

[0049] In FIG. 6, the reference letters are as follows

[0050] a Joggling depth
[0051] b Joggling width
[0052] c Upper foot: important plane deformation
[0053] d Lower foot: important plane deformation

[0054] Unless indicated otherwise, the chemical compositions are given as percentages by weight based on total weight of the article being described. Therefore, in a mathematical formula, "0.4 Zn" means "0.4 times the zinc content, expressed in percentage by weight". This also applies to other chemical elements as well as Zn. The alloy designations follow the rules of The Aluminum Association. The metallurgical tempers are as defined in the European Standard EN 515. Unless indicated otherwise, the static mechanical characteristics, i.e. the ultimate tensile strength $R_{un}$, the tensile yield strength $R_{y0.2}$, elongation at fracture $A_{e}$, are determined by a tensile test according to the standard EN 10002-1. The term "extruded product" includes so-called "drawn" products obtained by extrusion, followed by drawing.

[0055] During preparatory studies, the present inventors arrived at a conclusion that a novel material exhibiting a significantly improved compromise between mechanical strength and formability should preferably possess a sufficiently high zinc content, typically above 8.3%, and advantageously above 9.0%. According to the present invention, the inventors have found a very specific domain of composition which presents formation of wrought products, and especially extruded products, which at the same time have, high static mechanical properties, sufficient resistance to corrosion, and good formability.

[0056] In connection with the present invention, extruded products have been developed which can advantageously be used, for example, as fuselage stringers in aircraft, particularly civilian aircraft. For this use, damage tolerance is generally not a limiting factor. It is therefore feasible to optimise UTS and TYS properties while sacrificing some damage tolerance, so long as corrosion resistance is not affected to any measurable extent. However, pushing the UTS and TYS to maximum values usually leads to a decrease in formability. It is known to one skilled in the art that fuselage stringers can be often subjected to complex and very peculiar shaping operations, usually utilizing a technique called "joggling." When developing an alloy for fuselage stringers that has high mechanical strength, it is therefore advantageous that the formability be at least as good as, or preferably better than, the mechanical strengths of conventional alloys.

[0057] According to one embodiment of the present invention, this task can be solved inter alia by fine adjustment of the content of the elements of the alloys and certain impurities, as well as by optionally adding a controlled concentration of certain other elements to the alloy composition.

[0058] The present invention includes Al—Zn—Mg—Cu alloys comprising:

[0059] Zn 8.3-14.0 Cu 0.3-2.0 Mg 0.5-4.5

as well as certain other elements specified hereinbelow, and the rest being aluminum with its inevitable impurities.

[0060] Alloys according to some embodiments of the present invention should preferably include at least 0.5% magnesium, since it may be not possible to obtain satisfactory static mechanical characteristics with a lower magnesium content. In alloys with a zine content of less than 8.3%, the inventors have not found much improvement with respect to conventional alloys. Preferably, the zine content is higher than 9.0%, and still more preferably higher than 9.5%. However, specific arithmetic relationships between certain alloying elements should generally be respected in some embodiments, as will be explained below. In another advantageous embodiment of the invention, the zine content is comprised between 9.0 and 11.0%. In any case, the zine content should preferably not exceed a value of about 14%, because beyond about 14%, irrespective of the magnesium and copper content, the results may be less than satisfactory in some instances.

[0061] The preferable addition of at least 0.3% of copper serves to improve resistance to corrosion. A minimum copper content of about 0.6% is generally preferred. To ensure satisfactory solution heat treatment, the Cu content should preferably not exceed about 2%, and the Mg content should preferably not exceed about 4.5%. A maximum content of about 3.6% is preferred for magnesium. In a preferred embodiment, the copper content is between 0.6% and 1.2%, while the magnesium content is between 2.5% and 3.4%. In another preferred embodiment, the copper content is between 0.8% and 1.2%, while the magnesium content is between 2.2% and 3.0%. As will be explained below, the ratio between the magnesium and copper content should advantageously conform to certain criteria.

[0062] The present inventors have found that to address certain problems in the art regarding Al—Zn—Mg—Cu alloys, several additional technical features can be considered if desired.

[0063] First of all, the alloy should typically be sufficiently loaded with alloying elements likely to precipitate during maturation or during on annealing treatment, in order for the alloy to be capable of presenting advantageous static mechanical characteristics. As such, in addition to the minimum and maximum limits for the zinc, magnesium and copper contents indicated hereinabove, the content of these alloy additions should advantageously satisfy the condition Mg+Cu=6.4-0.4 Zn in some embodiments. This was a finding that was completely unexpected based on the teachings of the prior art.

[0064] To reinforce the effect achieved using the disclosed preferred alloy composition(s), disclosed above, a sufficient content of so-called anti-recrystallising elements can also advantageously be added. More precisely, for alloys with more than 9.5% zinc, at least one element selected from the group consisting of Zr, Sc, Hf, La, Ti, Y, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Yb, Cr and Mn can preferably be added. And each of these elements, if added, should preferably be present in a concentration of between 0.02 and 0.7%. It is
preferred that the total concentration of the elements of this group not exceed 1.5%, based on the total weight of the alloy.

[0065] These presence of one or more anti-recrystallising elements, in the form of fine precipitates formed during thermal or thermomechanical treatment, serve to block or at least minimise recrystallisation. However, it has unexpectedly been found that when the alloy is highly charged with zinc (Zn>9.5%), excessive precipitation should be avoided when a wrought product is being quenched. A compromise then was found for the anti-recrystallising elements content by the present inventors. Namely, according to one embodiment of the invention, for alloys with a zinc content of between 8.5% and 9.5%, zirconium between 0.05% and 0.15% should advantageously be added, preferably along with at least one element selected from the group consisting of Sc, Hf, La, Ti, Y, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er and Yb. Each element present in this group, is preferably present in a concentration of between 0.02 and 0.7%. In a preferred embodiment, Ti is present, alone or together with one or more other elements from the above group.

[0066] For anti-recrystallising elements, it is advantageous in some embodiments, irrespective of the zinc content, for such elements not to exceed the following maximum contents: Cr 0.40; Mn 0.60; Sc 0.50; Zr 0.15; Hf 0.60; Ti 0.15; Ce 0.35 and preferably 0.30; Nd 0.35 and preferably 0.50; Eu 0.35 and preferably 0.30; Gd 0.35; Tb 0.35; Ho 0.40; Dy 0.40; Er 0.40; Yb 0.40; Y 0.20; La 0.35 and preferably 0.30. It is further preferred that the total concentration of the elements of this group not exceed about 1.5%.

[0067] The inventors have unexpectedly found that in order to optimize the UTS and TYS, preferably a ratio Mg/Cu>2.4 should be adopted, more preferably if a ratio of at least 2.8, and still more preferably, 3.5 or even 4.0.

[0068] Another technical feature is associated with the need to be able to manufacture wrought products industrially under conditions of very high or even the highest reliability compatible with the severe requirements of the aeronautical industry, as well as under satisfactory economic conditions. So it is highly advantageous to choose a chemical composition which minimises the appearance of hot cracks or splits during solidification of the plates or billets. Hot cracks or splits are crippling defaults leading to the plates or billets that are discarded. It has been noted during numerous tests that the appearance of hot cracks or splits was unexpectedly much more probable when the 7xxx alloys finished solidifying below 470° C. To significantly reduce the probability of hot cracks or splits during casting to an acceptable industrial level, it was determined according to the present invention that a chemical composition such as Mg>1.95+0.5 (Cu-2.3)+0.16 (Zn-6)+1.9 (Si-0.04) may be advantageous in some instances.

[0069] Within the scope of the present invention, this empirical criterion is called the “castability criterion.” The alloys produced according to this variant of the invention typically complete their solidification at a temperature of about 473° C. and 478° C., and thus allow an industrial reliability of casting processes (that is, a constant and excellent quality of the cast ingots) to be reached that is generally compatible with some if not all of the severe requirements of the aeronautical industry.

[0070] Another technical feature of one embodiment of the invention is associated with a need to substantially minimise to the extent possible the quantity of insoluble precipitates following homogenisation and aging treatments. This is because the presence of such insoluble precipitates decreases the fracture toughness, the elongation at rupture, and especially the formability of certain products. Thus, it may be advantageous to employ, a Mg, Cu and Zn content such as Mg+Cu<7.9-0.4 Zn. The inventors have found that there likely are little or no disadvantages associated with selecting a composition close to the borderline represented by the relationship Mg+Cu<7.9-0.4 Zn. However, beyond this borderline, the excellent aptitude to deep formability by joggling (which is one of the main advantages of the present invention) may decrease rapidly.

[0071] And finally, the inventors have noted that incorporating a small quantity, of between 0.02 and 0.15% per element, of one or more elements chosen from the group consisting of Sn, Cd, Ag, Ge and In, improves the response of the alloy to an annealing treatment, and also has beneficial effects in terms of mechanical resistance and resistance to corrosion of products made from such alloys. A concentration between 0.05% and 0.10% is preferred. Among these elements, silver is advantageous in some embodiments.

[0072] For profiles, adding one or more anti-recrystallisation elements such as scandium may be particularly advantageous. Such an effect is also seen, for example, in alloys that may be used to prepare thick plates. An increase of mechanical strength in profiles is observed which is higher when the width or thickness of the section is low. This is known as “press effect” to one skilled in the art. The inventors have found that when scandium is employed as an anti-recrystallization element, a concentration between 0.02% and 0.50% is preferred.

[0073] The present invention is especially advantageous for use in extruded products. Extruded products can be used advantageously to produce structural members suitable for use in aeronautical construction. Preferred applications for products according to the present invention include their use as structural members for a fuselage of civilian or other aircrafts or related products. Such structural members, in particular stringers, are principally dimensioned for mechanical strength, whereas damage tolerance, (provided that is has an acceptable minimum value), is normally not a property used for dimensioning such structural members. Therefore, if needed and up to a certain point in terms of structural members, it is often desirable to optimise mechanical strength of such materials while accepting a certain loss of damage tolerance, without decreasing the usefulness of the final product. However, corrosion resistance should always be maintained at an acceptable minimum level. The increase of mechanical strength of fuselage stringers allows, at the discretion of the manufacturer operator, one to reduce the weight of a fuselage being manufactured while maintaining the same strength and/or permits the formation of a stronger fuselage structure at the same weight as compared to alloys now utilized by aircraft manufacturers. By increasing the distance between two adjacent stringers (within the limits of bending of the fuselage skin sheet), the number of fuselage stringers can be decreased. This leads to a corresponding decrease in the number of assembly points between a stringer and a fuselage skin. Such a decrease in number of assembly points can be very advantageous,
because the assembly points, such as bolts or rivets, are very significant contributors to the overall manufacturing cost of such a structure.

[0074] In preferred embodiments of the present invention, it is possible to obtain a fuselage skin and stringer assembly that is novel over those of the prior art in terms of the location and number of assembly points. Thus, a particularly advantageous use of a product according to the present invention is the use of an alloy described herein as a structural member in aeronautical constructions. In particular such alloys are suitable, inter alia, in the construction of aircrafts comprising a fuselage assembled from a plurality of stringers and a plurality of sheets, wherein at least part of the stringers comprise products according to the present invention. Such an aircraft will generally either (i) have a structure that is a reduced weight as compared with fuselages prepared from known materials, which is at least as strong, or (ii) will have a stronger structure, which will not be much heavier, if at all, than fuselages of aircrafts made according to the state of the art.

[0075] It is not only advantageous to minimize the number of assembly points between structural members of different type (such as fuselage stringers and fuselage skin), but it is also often desirable to minimize the number of assembly points between structural members of the same type, and especially between two stringers. In order to achieve this goal, it may be advantageous to utilize sheets or extruded profiles with as large a relevant size parameter as possible. In the case of extruded profiles, this relevant size parameter is essentially the length.

[0076] The manufacture of very long profiles in Al—Zn—Mg—Cu alloys with a high content of alloying elements may in some cases require fairly detailed process control during casting, extrusion and thermal treatment, and may require the adjustment of the chemical alloy composition. In particular, the present inventors have found that a product according to the present invention can be obtained by using a reduced extrusion force with respect to known products having a comparable zinc content. Thus products of the present invention are generally capable of extruding longer profiles.

[0077] Airworthiness authorities require that stringers be designed to resist limit load with large damage. It is generally recommended that a 2-stringer-bay crack is taken for evaluation of the required damage tolerance. This is a crack extending over two stringer bays, with the center stringer broken (see FIG. 10). It was recognized by the present inventors that the residual strength of fuselage shells working in tension could benefit from the high strength of stringers according to the present invention. The use of stringers according to the present invention as structural members in aircraft fuselage panels can therefore improve the residual strength of the structure, inter alia because they close the crack in the skin, thus preventing unstable fracture. This leads to a higher residual strength of the panel after damage. This effect can be used either to increase the safety margin of constructions by using stringers according to the present invention instead of stringers of the prior art, and/or the weight of the construction can be reduced by using reduced size and/or weight stringer sections and thinner skin panels, and/or by increasing the spacing between adjacent stringers.

[0078] Airworthiness authorities also generally require that such structures be designed to resist ultimate load for 3 seconds without excessive deformation. However, yielding is permitted. This usually leads to post-buckling designs for fuselage shells in stability critical locations. Although buckling of perfect columns (Euler theory) or real-life structure that is very slender is essentially an elastic phenomenon (governed by Young’s modulus), post-buckling designs display plastic deformation and can therefore benefit from an increase in yield strength. The buckling test is shown on FIG. 11.

[0079] It was recognized by the present inventors that the shear- and compression stability of fuselage shells working in compression and/or shear could benefit from the high strength of stringers according to the present invention. The use of stringers according to the present invention as structural members in aircraft fuselage panels can improve the shear- and compression stability of fuselage cells, because these stringers exhibit a higher local buckling stability. This effect can be used either to increase the safety margin of constructions in which stringers according to prior art are substituted by stringers according to the invention, and/or to decrease the weight of the construction, by using reduced stringer sections and thinner skin panels, and/or increased stringer spacing. Alternatively, increased rivet pitch can be obtained, leading to a lower assembly cost.

[0080] Another problem that arises when using extruded profiles in Al—Zn—Mg—Cu alloys as fuselage stringers is their formability. One technique which is commonly used during the industrial manufacture of fuselage stringers from profiles is joggling. Joggling involves the introduction of a step localised over a zone of a few millimetres (see FIG. 6). This can be achieved, for certain products according to the present invention, either at elevated temperature (preferably at about 130°C.), and/or at room temperature. When joggling is performed at room temperature, it is preferable to employ a solution heat treatment of the profile delivered in the instable T temper, followed by quenching and joggling. Joggling at room temperature generally does not allow forming to be as deep as joggling at elevated temperature, but when applicable, it is often more practical.

[0081] Joggling as an industrial process typically does not readily lend itself to the characterisation of materials under development. However it is known that defects appearing during joggling are related to the maximum plane deformation which can be supported by the material. Thus, it is possible to evaluate the aptitude of the material to joggling using a 3 point bending test such as DIN standard 50111. According to DIN standard 50111 (September 1987, see especially section 3.1), the specimen should have a width which is sufficient with respect to its thickness, in order to be in the centre of the test piece under conditions of plane deformation.

[0082] The formability of products of the present invention are evaluated at 130°C. (i.e. warm formability of the product in its temper of use), wherein a flat test piece is deformed in a furnace at 130°C. until a drop of the applied force is detected. This drop indicates that a crack has initiated. The temperature should be precisely controlled during this test. Since deformation takes place at elevated temperature, the rate of deformation is a parameter which influences the result. In the present case, this rate was fixed
at 50 mm/mn. The higher the bending angle as described in FIG. 8, the better the aptitude of the material to deformation by joggling. For mechanical reasons, it is highly recommended that all test pieces to be compared have the same thickness. Therefore when comparing two test pieces of different thicknesses, the face which will be under compressive stress should be machined down to the thickness of the thinnest test piece. In the case of a profile, the test piece is cut at a representative location, as shown, for example, in FIG. 7 for the profile T1.

[0084] A 3 point bending test at 130° C. can be applied to test pieces cut from products in T6x or T7x temper. It is also possible to characterize formability in an as-quenched condition W, if the period of time between quenching and bending testing is controlled. In the case of extruded products, the bending angle at 130° C. can be expressed as an average value computed from individual measurements on test pieces taken from different locations over the length of the profile.

[0085] A product according to the present invention which is particularly preferred is an extruded product that exhibits a bending angle in the T6511 temper, determined at 130° C. by a 3 point bending test according to DIN 50 111 (section 3.1) on a sample of 1.6 mm thickness cut from a plane area, of at least 34°, and a TYS of at least 720 MPa. Even more preferably extended products of the present invention possess a bending angle of at least 35° and a TYS of at least 750 MPa. For a thickness up to 60 mm, the static mechanical properties (UTS, TYS and A) do not depend much on the thickness of the section.

[0086] Extruded products of the present invention including the two particular ones identified above, can be used advantageously as fuselage stringers in many applications and are particularly useful for aircrafts including civilian aircrafts.

[0087] And mentioned above, the present inventors have surprisingly found that compared to prior art products, and including prior art products with a comparable zinc content, products according to certain embodiments of the present invention exhibit a high warm formability. On the other hand, cold formability in the unstable W temper after solution heat treatment and quenching may be slightly less. For the manufacture of structural members of aircrafts, such as fuselage stringers, warm forming process is preferable, particularly if deformation during joggling is deep.

[0088] Products according to the present invention can also be used, for example, as floorbeams for aircrafts. In the form of extruded profiles, the can also be used, for example, as seat tracks. In civilian aircrafts, seat tracks are profiles, generally of great length, which are normally parallel to the length of the cabin, and on which the seats are mounted.

According to the present invention, seat tracks in T76511 temper can be obtained which exhibit a UTS in the area where the seats are fixed (i.e. the top of a “I” shape) of 670 MPa or more, and even of 680 MPa or more, and a TYS of 640 MPa or more, and even of 660 MPa or more. Seat tracks of commercial aircrafts have to be resistant to corrosion by corrosive liquid foodstuff under high mechanical stress. Indeed, seat tracks according to the present invention exhibit a good stress corrosion resistance as determined according to ASTM G47.

[0089] The use of structural members according to the present invention for aircraft construction leads to significant weight savings, which allows to increase the load capacity of said aircraft, or to decrease fuel consumption.

[0090] The following examples illustrate different embodiments of the invention and demonstrate its advantages; they do not restrict this invention.

EXAMPLES

Example 1

[0091] Several Al—Zn—Mg—Cu alloys were prepared by semi-continuous casting of rolling ingots, and were then subjected to a range of conventional transformation techniques, comprising a homogenisation stage, the parameters of which have been determined according to U.S. Pat. No. 5,560,789, the content of which is incorporated herein by reference in its entirety. The homogenisation stage was followed by hot rolling, solution heat treatment which was followed by quenching and stress relieving operations, and finally an aging treatment was conducted in order to obtain a product in temper T651. This process resulted in the formation of plates in the T651 temper having a thickness of 20 mm.

[0092] Compositions of plates according to this example are specified in Table 1 below:

<table>
<thead>
<tr>
<th>Alloy</th>
<th>Zn</th>
<th>Mg</th>
<th>Cu</th>
<th>Fe</th>
<th>Si</th>
<th>Zr</th>
<th>Ti</th>
<th>Mn</th>
<th>Sc</th>
<th>Mg/Cu</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>8.40</td>
<td>2.11</td>
<td>1.83</td>
<td>0.09</td>
<td>0.06</td>
<td>0.11</td>
<td>0.017</td>
<td>0</td>
<td>0</td>
<td>1.15</td>
</tr>
<tr>
<td>B</td>
<td>10.27</td>
<td>3.2</td>
<td>0.71</td>
<td>0.08</td>
<td>0.03</td>
<td>0.11</td>
<td>0.017</td>
<td>0</td>
<td>0</td>
<td>4.57</td>
</tr>
<tr>
<td>C</td>
<td>10.08</td>
<td>2.69</td>
<td>0.95</td>
<td>0.08</td>
<td>0.03</td>
<td>0.11</td>
<td>0.014</td>
<td>0</td>
<td>0</td>
<td>2.83</td>
</tr>
<tr>
<td>D</td>
<td>9.97</td>
<td>2.14</td>
<td>1.32</td>
<td>0.09</td>
<td>0.03</td>
<td>0.11</td>
<td>0.017</td>
<td>0</td>
<td>0</td>
<td>1.62</td>
</tr>
</tbody>
</table>

[0093] The static mechanical characteristics were determined by a tensile test according to standard EN 10002-1. The fracture toughness KIC was determined according to standard ASTM E399.

[0094] The results are specified in Table 2 below:

<table>
<thead>
<tr>
<th>Alloy</th>
<th>Rm</th>
<th>Rp0.2</th>
<th>A</th>
<th>Rm</th>
<th>Rp0.2</th>
<th>A</th>
<th>KIc</th>
</tr>
</thead>
<tbody>
<tr>
<td>A</td>
<td>627</td>
<td>665</td>
<td>14.7</td>
<td>566</td>
<td>623</td>
<td>13.6</td>
<td>31.9</td>
</tr>
<tr>
<td>B</td>
<td>716</td>
<td>726.5</td>
<td>6.5</td>
<td>640</td>
<td>696</td>
<td>5.2</td>
<td>21.1</td>
</tr>
</tbody>
</table>
TABLE 2-continued

<table>
<thead>
<tr>
<th>Fracture toughness L-T</th>
<th>Tensile test</th>
<th>Tensile test</th>
</tr>
</thead>
<tbody>
<tr>
<td></td>
<td>TL direction</td>
<td>L direction</td>
</tr>
<tr>
<td>Alloy</td>
<td>R_p0.2 MPa</td>
<td>R_m MPa</td>
</tr>
<tr>
<td></td>
<td></td>
<td></td>
</tr>
<tr>
<td></td>
<td>%</td>
<td></td>
</tr>
<tr>
<td>C</td>
<td>700</td>
<td>717</td>
</tr>
<tr>
<td>D</td>
<td>665</td>
<td>685</td>
</tr>
</tbody>
</table>

It appears that plate C according to the invention presents a good compromise between mechanical strength and elongation. Compared to plate D, the mechanical strength is significantly improved. Compared to plate A in alloy 7449 according to prior art, plate C exhibits a mechanical strength that is very significantly improved. The fact that fracture toughness is less good in plate C than in plate B could potentially limit the possibilities of application of plate C to those applications for which fracture toughness is not taken into account when dimensioning the structural members, but which require both a high mechanical strength and a good formability. Compared to plate B, the elongation at fracture of plate C is significantly improved.

Moreover, in order to obtain the properties given in Table 2, plate B needs to be submitted to a rather long solution heat treatment, which does not lend itself to the requirements of industrial production. And yet, coarse phases have been found in the product which has an adverse effect on the homogeneity of mechanical properties, both within the same production lot and within the same individual product (plate or profile). Such presence of precipitates may preclude the use of product B as a structural member in aircrafts.

Example 2

Several rolling ingots whose chemical composition is specified in Table 3 were cast. The silicon content was approximately the same for all of them, about 0.04%.

Alloys G1, G2, G3 and G4, as well as alloys B and D, described in example 1 are used as comparisons with certain preferred embodiments. Alloy C is an alloy according to the invention described in example 1. During testing, all these alloys exhibited satisfactory castability, that is, no splits or cracks were observed during casting tests performed on an industrial scale.

Alloys G5, G6, G7, G8 were used as comparisons with certain preferred embodiments of the present invention, and alloy G9 is an alloy 7060 as per the prior art; these alloys exhibited cracks during casting tests.

Difficulties showing up during casting of these alloys do not necessarily render the wrought products from these plates unsuitable for use, but they are the cause of extra cost because their implementation (that is, the quantity of vendible metal relative to the quantity of charged metal, a parameter directly associated with the quantity of discarded ingots) will be greater than for the alloys corresponding to certain preferred domains of the invention. In addition, the propensity of these alloys to form splits during their solidification makes it very difficult to render the casting process reliable within the scope of a quality assurance program based on statistical process control.

It is noted that all the 7xxx alloys having a very pronounced propensity to form splits or cracks in casting generally have a magnesium content lower than desired magnesium content typically employed for such alloys. A desirable Mg value was obtained by calculating the Mg limit value defined by the empirical castability criterion.

### TABLE 3

<table>
<thead>
<tr>
<th>Alloy</th>
<th>Zn [%]</th>
<th>Mg [%]</th>
<th>Cu [%]</th>
<th>Critical Mg content</th>
<th>Mg &gt; Critical Mg</th>
</tr>
</thead>
<tbody>
<tr>
<td>G1</td>
<td>7.5</td>
<td>3</td>
<td>3</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>G2</td>
<td>8.5</td>
<td>3</td>
<td>2.3</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>G3</td>
<td>7.5</td>
<td>3</td>
<td>1.6</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>G4</td>
<td>6.5</td>
<td>3</td>
<td>2.3</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>B</td>
<td>10.27</td>
<td>3.2</td>
<td>0.71</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>C</td>
<td>10.08</td>
<td>2.69</td>
<td>0.95</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>D</td>
<td>9.97</td>
<td>2.14</td>
<td>1.32</td>
<td>low</td>
<td>yes</td>
</tr>
<tr>
<td>G5</td>
<td>8.5</td>
<td>2.3</td>
<td>3</td>
<td>high</td>
<td>no</td>
</tr>
<tr>
<td>G6</td>
<td>6.5</td>
<td>2.3</td>
<td>3</td>
<td>high</td>
<td>no</td>
</tr>
<tr>
<td>G7</td>
<td>8.5</td>
<td>1.6</td>
<td>2.3</td>
<td>high</td>
<td>no</td>
</tr>
<tr>
<td>G8</td>
<td>7.5</td>
<td>1.6</td>
<td>1.6</td>
<td>high</td>
<td>no</td>
</tr>
<tr>
<td>G9</td>
<td>7</td>
<td>1.65</td>
<td>2.1</td>
<td>high</td>
<td>yes</td>
</tr>
</tbody>
</table>

Example 3

Extrusion ingots have been cast from alloys whose composition is summarized in Table 4. Homogenization was carried out as follows:

Ingot Q1 and Q2: 4 h at 465° C, + 20 h at 476° C.
Ingot Q3 and Q4: 4 h at 465° C, + 20 h at 471° C.
Ingot P1 through P3: 20 h at 471° C.

M, T and S phases were completely dissolved by these homogenization treatments; this was checked by differential enthalpic analysis (according U.S. Pat. No. 5,560,789, incorporated herein by reference).

### TABLE 4

<table>
<thead>
<tr>
<th>Ingot</th>
<th>Zn</th>
<th>Mg</th>
<th>Cu</th>
<th>Cr</th>
<th>Mn</th>
<th>Si</th>
<th>Fe</th>
<th>Zr</th>
<th>Ti</th>
<th>Mg/Cu</th>
</tr>
</thead>
<tbody>
<tr>
<td>P1</td>
<td>8.10</td>
<td>2.48</td>
<td>1.65</td>
<td>0.14</td>
<td>0.17</td>
<td>0.01</td>
<td>0.08</td>
<td>0.15</td>
<td>0.03</td>
<td>1.80</td>
</tr>
<tr>
<td>P2</td>
<td>8.45</td>
<td>2.60</td>
<td>1.76</td>
<td>0.18</td>
<td>0.18</td>
<td>0.05</td>
<td>0.14</td>
<td>0.12</td>
<td>0.02</td>
<td>1.48</td>
</tr>
<tr>
<td>P3</td>
<td>8.39</td>
<td>2.55</td>
<td>1.71</td>
<td>0.18</td>
<td>0.16</td>
<td>0.04</td>
<td>0.15</td>
<td>0.11</td>
<td>0.02</td>
<td>1.49</td>
</tr>
<tr>
<td>Q1</td>
<td>10.20</td>
<td>3.10</td>
<td>0.68</td>
<td>0.17</td>
<td>0.17</td>
<td>0.07</td>
<td>0.08</td>
<td>0.13</td>
<td>0.04</td>
<td>4.56</td>
</tr>
<tr>
<td>Q2</td>
<td>10.20</td>
<td>2.84</td>
<td>0.95</td>
<td>0.18</td>
<td>0.17</td>
<td>0.06</td>
<td>0.11</td>
<td>0.13</td>
<td>0.03</td>
<td>2.99</td>
</tr>
<tr>
<td>Q3</td>
<td>9.98</td>
<td>2.10</td>
<td>1.24</td>
<td>0.18</td>
<td>0.16</td>
<td>0.06</td>
<td>0.14</td>
<td>0.12</td>
<td>0.03</td>
<td>1.69</td>
</tr>
</tbody>
</table>
From these homogenized and scalped ingots, five types of profiles T1, T2, T3, T4 and T5 were extruded. Their sections are represented on Figs. 1, 2, 3, 4 and 5. The temperature of the container and of the die was above 400° C., and the extrusion speed was below 0.50 m/min.

Maximum extrusion forces are summarized in Table 5. It can be seen that surprisingly the alloys according to the present invention do not require a higher extrusion force, and that the extrusion force surprisingly even decreases for certain types of profiles with increasing magnesium content.

Extrusion force

<table>
<thead>
<tr>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
<th></th>
</tr>
</thead>
<tbody>
<tr>
<td>T1</td>
<td>179</td>
<td>175</td>
<td>170</td>
<td>164</td>
<td>164</td>
<td>58</td>
</tr>
<tr>
<td>T2</td>
<td>151</td>
<td>145</td>
<td>142</td>
<td>137</td>
<td>139</td>
<td>24</td>
</tr>
<tr>
<td>T3</td>
<td>203</td>
<td>208</td>
<td>200</td>
<td>193</td>
<td>195</td>
<td>13</td>
</tr>
</tbody>
</table>

Profiles Q1 through Q4 were solution heat treated at 471° C., while profiles P1 through P3 were solution heat treated at 472° C. (profiles T1, T2 and T3). All profiles were water quenched and then stretched with a permanent set comprised between 1.5% and 2%. Products in tempers T6511 or T76511 were obtained. Their mechanical properties are summarized in Table 6 for specimens of three different thickness values in temper T6511, cut from a flat area of the profile. This temper has been obtained by artificial aging under the following conditions:

Alloys Q1 and Q2: 18 h at 120° C.

Alloys P1, P2, P3, Q3 and Q4: 36 h at 120° C.

Sampling:
T1 = foot of profile T1.
T2 = top of profile T2.
T3 = top of profile T3.

It can be seen that compared to alloy P1, alloys Q1 and Q2 have a mechanical strength significantly higher.
For R1 and R2, the following mechanical properties were found:

### TABLE 8

<table>
<thead>
<tr>
<th>Alloy, profile</th>
<th>T6511 $R_m$ [MPa]</th>
<th>T6511 $R_p0.2$ [MPa]</th>
<th>A [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>R1, T4(a)</td>
<td>753</td>
<td>738</td>
<td>8</td>
</tr>
<tr>
<td>R2, T4(b)</td>
<td>756</td>
<td>743</td>
<td>8</td>
</tr>
<tr>
<td>R2, T5</td>
<td>755</td>
<td>743</td>
<td>7</td>
</tr>
</tbody>
</table>

NOTE: profile T4(a) = samples cut from the top of the profile, see FIG. 4, feature (a).

### Example 4

Formability of profiles of shape T1 according to example 3 was studied by using the three point bending test according to DIN 50111 (September 1987, section 3.1). The location of sampling, a flat zone, is shown on FIG. 7. The important parameters of the three point bending test are shown on FIG. 9. The test was performed at 130°C.

Both tempers T6511 and T76511 were tested. The resulting values for the bending angle $\alpha$ (as defined on FIG. 8) are summarized in Table 9. These are mean values computed from half a dozen individual determinations using specimens cut at different locations distributed over the length of the profiles.

### Example 5

Cold forming of samples similar to those used in the example 4 (in the unstable W temper after solution heat treatment and quenching) was studied at room temperature by using the same three point bending test. The variation of the bending angle $\alpha$ (as defined on FIG. 8) over the length of the profiles is small. Table 10 refers to values obtained in the W temper.

### TABLE 9

<table>
<thead>
<tr>
<th>Bending angle</th>
<th>Sample</th>
</tr>
</thead>
<tbody>
<tr>
<td>Q1</td>
<td>33.6°</td>
</tr>
<tr>
<td>Q2</td>
<td>34.5°</td>
</tr>
</tbody>
</table>

Example 6

Rolling ingots were elaborated by a process similar to the one described in example 1. The chemical composition is given in Table 11. Hot rolled plates with a thickness of 25 mm were obtained by a process similar to the one described in example 1. The plates were solution heat treated at a temperature between 472 and 480°C for 2 hours, quenched and stretched with a permanent set comprised between 1.5% and 2%. Finally, the stretched plates were artificially aged at a temperature of 150°C.

### TABLE 10-continued

<table>
<thead>
<tr>
<th>Sample</th>
<th>Bending angle</th>
</tr>
</thead>
<tbody>
<tr>
<td>Q3</td>
<td>33.6°</td>
</tr>
<tr>
<td>P1</td>
<td>34.5°</td>
</tr>
</tbody>
</table>

Example 7

For several profiles elaborated according to example 3; the resistance to stress corrosion was evaluated. The results are summarized in Table 13.
TABLE 13

<table>
<thead>
<tr>
<th>Sample</th>
<th>Temper</th>
<th>Stress [MPa]</th>
<th>Duration of</th>
<th>the test</th>
</tr>
</thead>
<tbody>
<tr>
<td>Alloy Q1, profile T1, L direction</td>
<td>T76511</td>
<td>530</td>
<td>&gt;30 days</td>
<td></td>
</tr>
<tr>
<td>Alloy Q1, profile T1, L direction</td>
<td>T76511</td>
<td>350</td>
<td>&gt;30 days</td>
<td></td>
</tr>
<tr>
<td>Alloy P1, profile T4, L direction</td>
<td>T76511</td>
<td>430</td>
<td>&gt;30 days</td>
<td></td>
</tr>
<tr>
<td>Alloy P1, profile T4, LT direction</td>
<td>T76511</td>
<td>400</td>
<td>&gt;30 days</td>
<td></td>
</tr>
<tr>
<td>Alloy P1, profile T4, LT direction</td>
<td>T76511</td>
<td>280</td>
<td>&gt;30 days</td>
<td></td>
</tr>
<tr>
<td>Alloy R1, profile T4, LT direction</td>
<td>T76511</td>
<td>280</td>
<td>&gt;30 days</td>
<td></td>
</tr>
</tbody>
</table>

It can be seen that the products according to the invention show a satisfactory resistance to stress corrosion.

Example 8

Profiles in alloys 7349 or 7449 were produced with and without scandium, according to a process similar to the one described in example 3. Table 14 lists the chemical compositions, Table 15 the obtained mechanical properties.

TABLE 14

<table>
<thead>
<tr>
<th>Alloy</th>
<th>Zn</th>
<th>Mg</th>
<th>Cu</th>
<th>Fe</th>
<th>Si</th>
<th>Zr</th>
<th>Ti</th>
<th>Mn</th>
<th>Cr</th>
<th>Sc</th>
<th>Mg/Cu</th>
</tr>
</thead>
<tbody>
<tr>
<td>X1</td>
<td>8.1</td>
<td>2.5</td>
<td>1.7</td>
<td>0.08</td>
<td>0.01</td>
<td>0.15</td>
<td>0.03</td>
<td>0.17</td>
<td>0.14</td>
<td>0</td>
<td>1.47</td>
</tr>
<tr>
<td>X2</td>
<td>8.4</td>
<td>2.1</td>
<td>1.9</td>
<td>0.06</td>
<td>0.02</td>
<td>0.10</td>
<td>0.02</td>
<td>0</td>
<td>0</td>
<td>0</td>
<td>1.11</td>
</tr>
</tbody>
</table>

Stress corrosion testing according to ASTM G47 shows good resistance to stress corrosion.

Example 10

Numerical models for damage tolerance of fuselage shells employing high-strength stringers according to the invention were evaluated in order to determine the residual strength of a fuselage shells. Airworthiness authorities require that such structure be designed to resist limit load with large damage; it is recommended that a 2-stringer-bay crack is taken for evaluation of the required damage tolerance. This is a crack 12 extending over two stringer bays 14, 16, with the center stringer 18 broken (see FIG. 10). It was recognized by the present inventors that the residual strength of fuselage shells working in tension could benefit from the high strength of stringers according to the present invention. The use of stringers according to the present invention as structural members in aircraft fuselage panels can improve the residual strength of the structure, because they close the crack 12 in the skin 20, thus preventing unstable fracture. This leads to a higher residual strength of the panel after damage. This effect can be used either to increase the safety margin of constructions in which stringers according to prior art are substituted by stringers according to the invention, or to decrease the weight of the construction, by using reduced stringer sections and thinner skin panels, and/or increased stringer spacing.

A comparison with the results of example 3 shows that the products according to the invention have increased mechanical strength (R_m, R_p0.2) compared to products X1 and X2 according to prior art.

Example 9

Seat tracks for aircraft have been manufactured from extrusion billets of chemical composition R1 and Q1 according to the previous examples. These profiles are “I type” profiles including a foot section, a centre section, and a top section on which the seats are fixed. The thickness of the centre section was of the order of 2 mm, and the height of the profile was of the order of 65 mm.

Table 16 summarizes the static mechanical properties in temper T76511.

TABLE 16

<table>
<thead>
<tr>
<th>Alloy</th>
<th>Sampling</th>
<th>R_m [MPa]</th>
<th>R_p0.2 [MPa]</th>
<th>A [%]</th>
</tr>
</thead>
<tbody>
<tr>
<td>R1</td>
<td>Foot</td>
<td>688</td>
<td>669</td>
<td></td>
</tr>
<tr>
<td>R1</td>
<td>Top</td>
<td>686</td>
<td>667</td>
<td></td>
</tr>
<tr>
<td>Q1</td>
<td>Foot</td>
<td>672</td>
<td>643</td>
<td></td>
</tr>
<tr>
<td>Q1</td>
<td>Top</td>
<td>683</td>
<td>660</td>
<td></td>
</tr>
</tbody>
</table>

The fracture of fuselage skin is governed by the stress intensity factors (SIF) at the crack tips. For a typical fuselage crown structure with a stringer pitch of 200 mm and a stiffening ratio (section of stringer/total section) of 0.25, the SIF for a crack of 2-stringer-bay length in a panel with stringers made of the present invention shows a reduction of about 5% compared to a panel with stringers in the widely used 2024 T3 alloy. For longer cracks, 2024 stringers are more often solicited in the plastic domain, and the stress in the stringers will not reach the yield point. In the case of plastic domain 2024 stringers, the SIF of the present invention is reduced by about 15%. However, it should be noted that for stringers according to prior art in alloy 2024, there...
is also a risk of the stringers reaching their ultimate tensile stress and failing, whereas stringers according to the invention will not break under these conditions.

Example 11

[0133] Numerical models of fuselage shells working in compression and/or shear were evaluated in order to determine the shear- and compression stability. Airworthiness authorities require that such structure be designed to resist ultimate load for 3 seconds without excessive deformation. However, yielding is permitted. This usually leads to post-buckling designs for fuselage shells in stability critical locations. Although buckling of perfect columns (Faulier theory) or real-life structure that is very slender is essentially an elastic phenomenon (governed by Young’s modulus), post-buckling designs display plastic deformation and can therefore benefit from an increase in yield strength. The buckling test is shown in FIGS. 11a and 11b. FIG. 11b is taken along line A-A rotated 90°. A fuselage skin 20 is shown with two stringers 14, 16 attached thereto. Rivets 22 attach the stringers 16, 18 and the fuselage skin 20. The gap 24 between the stringers and skin is clearly shown in both FIG. 11a and rotated FIG. 11b.

[0134] It was recognized by the present inventors that the shear- and compression stability of fuselage shells working in compression and/or shear could benefit from the high strength of stringers according to the present invention. The use of stringers according to the present invention as structural members in aircraft fuselage panels can improve the shear- and compression stability of fuselage cells, because these stringers exhibit a higher local buckling stability. This effect can be used either to increase the safety margin of constructions in which stringers according to prior art are substituted by stringers according to the invention, or to decrease the weight of the construction, by using reduced stringer sections and thinner skin panels, and/or increased stringer spacing. Alternatively, increased rivet pitch can be obtained, leading to a lower assembly cost.

[0135] The gain in buckling stability can be obtained by applying a very general method given in Michael C. Y. Niu, Airframe Stress Analysis and Sizing, 2nd edition, chapter 10, incorporated herein by reference in its entirety. The present inventors have found that stringer stability of the stringer according to the invention (with compressive yield strength of 700 MPa and compressive elastic modulus of 73 GPa) compared to the widely used 7150 T77511 stringer (with a typical compressive yield strength of 538 MPa and an elastic modulus of 73 GPa) is increased on the order of about 15%, for typical fuselage Z-stringers based on the data in Table 17 and FIG. 12.

[0136] Table 17 shows the parameters of different stringer geometries analyzed. FIG. 12 compares crippling stress for these different stringer geometries Z1 to Z8 (from the left to the right).

**TABLE 17-continued**

<table>
<thead>
<tr>
<th>Small Z-stringer designs:</th>
<th>Z1</th>
<th>Z2</th>
<th>Z3</th>
<th>Z4</th>
<th>Z5</th>
<th>Z6</th>
<th>Z7</th>
<th>Z8</th>
</tr>
</thead>
<tbody>
<tr>
<td>Fastened flange width [mm]</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
</tr>
<tr>
<td>Height [mm]</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
</tr>
<tr>
<td>Free flange thickness [mm]</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Fastened flange thickness [mm]</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Web thickness [mm]</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
</tr>
<tr>
<td>Section [mm²]</td>
<td>76</td>
<td>83</td>
<td>95</td>
<td>102</td>
<td>95</td>
<td>102</td>
<td>102</td>
<td>114</td>
</tr>
<tr>
<td>Equivalent thickness [mm]</td>
<td>1.0</td>
<td>1.1</td>
<td>1.3</td>
<td>1.3</td>
<td>1.3</td>
<td>1.3</td>
<td>1.3</td>
<td>1.5</td>
</tr>
</tbody>
</table>

[0137] Additional advantages, features and modifications will readily occur to those skilled in the art. Therefore, the invention in its broader aspects is not limited to the specific details, and representative devices, shown and described herein. Accordingly, various modifications may be made without departing from the spirit or scope of the general inventive concept as defined by the appended claims and their equivalents.


[0139] As used herein and in the following claims, articles such as “the”, “a” and “an” can connote the singular or plural.

[0140] All documents referred to herein are specifically incorporated herein by reference in their entirety.

1. (canceled)
2. A fuselage structure according to claim 79, wherein Mg/Cu>2.8.
3. A fuselage structure according to claim 79, wherein Mg/Cu>3.5.
4. A fuselage structure according to claim 79, wherein Mg/Cu>4.0.
5. A fuselage structure according to claim 79, wherein the maximum content of the elements Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y and Yb does not exceed 1.5% in total.
6. A fuselage structure according to claim 79, comprising only Ti from said group consisting of Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y and Yb.
7. A fuselage structure comprising a stringer comprising an Al—Zn—Mg—Cu alloy, comprising (in mass percentage):
   a) Zn 9.5-14.0 Cu 0.3-2.0 Mg 0.5-4.5 Fe+Si<0.25
   b) at least one element selected from the group consisting of Zr Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y and Yb, the content of each of said elements, if selected, being between 0.02 and 0.7%,
   c) remainder aluminum and inevitable impurities, and
   wherein
   Mg/Cu>2.4 and
   (7.9-0.4 Zn)>(Cu+Mg)>(6.4-0.4 Zn).
8. A fuselage structure according to claim 7, wherein the maximum content of the elements Zr, Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y and Yb does not exceed 1.5% in total.

9. A fuselage structure according to claim 79, wherein Zn>9.0%.

10. A fuselage structure according to claim 9, wherein Zn>9.5%.

11. A fuselage structure according to claim 9, wherein zinc is between 9.0% and 11.0%.

12. A fuselage structure according to claim 79, wherein Cu>0.6%.

13. A fuselage structure according to claim 7, wherein Cu>0.6%.

14. A fuselage structure according to claim 12, wherein Cu 0.6-1.2% and Mg 2.2-3.0%.

15. A fuselage structure according to claim 12, wherein Cu 0.8-1.5% and Mg 2.2-3.0%.

16. A fuselage structure according to claim 79, wherein Mg 0.5-3.6%.

17. A fuselage structure according to claim 7, wherein Mg 0.5-3.6%.

18. A fuselage structure according to claim 7, wherein Mg/Cu>2.8.

19. A fuselage structure according to claim 7, wherein Mg/Cu>3.5.

20. A fuselage structure according to claim 7, wherein Mg/Cu>4.0.

21. A fuselage structure according to claim 18 wherein said alloy comprises a rolled, forged or extruded material comprising an aluminum alloy according to claim 18.

22. A fuselage structure according to claim 79, wherein Mg>1.95+0.5 (Cu-2.3)+0.16 (Zn-6)+1.9 (Si-0.04).

23. A fuselage structure according to claim 7, wherein Mg>1.95+0.5 (Cu-2.3)+0.16 (Zn-6)+1.9 (Si-0.04).

24. A fuselage structure according to claim 79, wherein the maximum mass percentage of the following elements is not exceeded:

Cr 0.40 Mn 0.60 Sc 0.50 Zr 0.15 Hf 0.60 Ti 0.15
Ce 0.35 Nd 0.35 La Eu 0.35
Gd 0.35 Tb 0.35 Dy 0.40 Ho 0.40 Er 0.40 Yb 0.40 and Y 0.20.

25. A fuselage structure according to claim 7, wherein the maximum mass percentage of the following elements is not exceeded:

Cr 0.40 Mn 0.60 Sc 0.50 Zr 0.15
Hf 0.60 Ti 0.15 Ce 0.35 Nd 0.35 La
Eu 0.35
Gd 0.35 Tb 0.35 Dy 0.40 Ho 0.40 Er 0.40 Yb 0.40 and Y 0.20.

26. A fuselage structure according to claim 79, further comprising at least one element selected from the group consisting of Ag, Sn, Cd, Ge and In, the content of each of said at least one element, if selected, being present in an amount of between 0.02% and 0.15%.

27. A fuselage structure, according to claim 7, further comprising at least one element selected from the group consisting of Ag, Sn, Cd, Ge and In, the content of each of said at least one element, if selected, being present in an amount of between 0.02% and 0.15%.

28. A fuselage structure according to claim 79, wherein said alloy is formed into an extruded product prepared from said alloy, wherein said product exhibits in temper T6511, measured on test pieces cut from a plane zone of the profile,

a) a bending angle, determined at 130° C. by means of a three point bending test according to DIN 50 111 (section 3.1) on a specimen of 1.6 mm thickness and expressed as the mean value computed from several individual measurements performed on specimens cut from different locations over the length of the profile, of at least 34°, and

b) a tensile yield strength (TYS) of at least 720 MPa.

29. A fuselage structure according to claim 7, wherein said alloy is formed into an extruded product prepared from said alloy, wherein said product exhibits in temper T6511, measured on test pieces cut from a plane zone of the profile,

a) a bending angle, determined at 130° C. by means of a three point bending test according to DIN 50 111 (section 3.1) on a specimen of 1.6 mm thickness and expressed as the mean value computed from several individual measurements performed on specimens cut from different locations over the length of the profile, of at least 34°, and

b) a tensile yield strength (TYS) of at least 720 MPa.

30. A fuselage structure according to claim 28, wherein said extruded product has a bending angle of at least 35° and a TYS of at least 750 MPa.

31. A fuselage structure according to claim 29, wherein said extruded product has a bending angle of at least 35° and a TYS of at least 750 MPa.

32. A fuselage structure according to claim 28, wherein said product exhibits in temper T76511, measured on test pieces cut from a plane zone of the profile,

a) a bending angle, determined at 130° C. by means of a three point bending test according to DIN 50 111 (section 3.1) on a specimen of 1.6 mm thickness and expressed as the mean value computed from several individual measurements performed on specimens cut from different locations over the length of the profile, of at least 37°, and

b) an ultimate tensile strength (UTS) of at least 670 MPa.

33. A fuselage structure according to claim 29, wherein said product exhibits in temper T76511, measured on test pieces cut from a plane zone of the profile,

a) a bending angle, determined at 130° C. by means of a three point bending test according to DIN 50 111 (section 3.1) on a specimen of 1.6 mm thickness and expressed as the mean value computed from several individual measurements performed on specimens cut from different locations over the length of the profile, of at least 37°, and

b) an ultimate tensile strength (UTS) of at least 670 MPa.

34. A fuselage structure according to claim 32, wherein said product has a stress corrosion resistance, determined by an EXCO test according to ASTM G34 in the T6511 temper on unmachined test pieces, of at least level EB.

35. A fuselage structure according to claim 33, wherein said product has a stress corrosion resistance, determined by...
an EXCO test according to ASTM G34 in the T6511 temper on unmachined test pieces, of at least level EB.

36. A structural aircraft member manufactured from a fuselage structure of claim 79.

37. A structural aircraft member manufactured from a fuselage structure of claim 7.

38. A structural aircraft member according to claim 36, wherein said structural member comprises a fuselage stringer.

39. A structural aircraft member according to claim 37, wherein said structural member comprises a fuselage stringer.

40. (canceled)

41. (canceled)

42. (canceled)

43. A structural member according to claim 36, having a ultimate tensile strength in the T76511 temper of at least 670 MPa.

44. (canceled)

45. A structural member according to claim 37, having a ultimate tensile strength in the T76511 temper of at least 670 MPa.

46. (canceled)

47. (canceled)

48. (canceled)

49. A structural member according to claim 36, having a tensile yield strength in the T76511 temper of at least 640 MPa.

50. A structural member according to claim 37, having a tensile yield strength in the T76511 temper of at least 640 MPa.

51. (canceled)

52. (canceled)

53. A structural member according to claim 36, having a tensile yield strength of at least 660 MPa.

54. A structural member according to claim 37, having a tensile yield strength of at least 660 MPa.

55. A structural aircraft member according to claim 36, wherein said structural member comprises a floor beam.

56. A structural aircraft member according to claim 37, wherein said structural member comprises a floor beam.

57. An aircraft comprising a fuselage assembled from a plurality of stringers and a plurality of sheets, wherein at least part of said stringers comprise structural members according to claim 36.

58. An aircraft comprising a fuselage assembled from a plurality of stringers and a plurality of sheets, wherein at least part of said stringers comprise structural members according to claim 37.

59. A fuselage structure comprising an extruded product comprising a bending angle of at least 34° in the T6511 temper determined at 130° C. by a 3 point bending test according to DIN 50111 on a sample thereof 1.6 mm in thickness cut from a plane area of said product, and said product further comprising a TYS of at least 720 MPa.

60. A fuselage structure comprising an extruded product comprising a bending angle of at least 360° in the T76511 temper determined at 130° C. by a 3 point bending test according to DIN 50111 on a sample thereof 1.6 mm in thickness cut from a plane area of said product, and said product further comprising a TYS of at least 660 MPa.

61. A fuselage structure of claim 60 wherein said extruded product further comprises a corrosion resistance rating of at least EB (EXCO test according to ASTM G34).

62. (canceled)

63. (canceled)

64. (canceled)

65. (canceled)

66. A fuselage structure of claim 79 comprising a rolled product.

67. A fuselage structure of claim 7 comprising a rolled product.

68. A fuselage structure of claim 79 comprising an extruded product.

69. A fuselage structure of claim 7 comprising an extruded product.

70. A fuselage structure of claim 79 comprising a forged product.

71. A fuselage structure of claim 7 comprising a forged product.

72. A stringer comprising a 7xxx alloy wherein after being subjected to homogenization and scalping and being extruded at 400° C. at a speed of below 0.50m/min, extrusion forces required to obtain extrusion profiles of said alloy decrease as the content of magnesium is increased in said alloy.

73. A stringer comprising a 7xxx alloy comprising up to 0.15% Sc, up to 12% Zn, and a Mg/Cu ratio of ≥2.4, said alloy possessing an Rm of from 680-700 MPa and Rm of from 680-700 MPa.

74. A fuselage structure comprising at least one stringer that is not 2024, wherein if prepared with a stringer pitch of 200 mm and a stiffening ratio of 0.25, said structure exhibits an SIF that is reduced up to 5% as compared to a fuselage structure with 2024 T3 alloy stringers.

75. A fuselage structure comprising at least one stringer that is not 2024, wherein if prepared with a stringer pitch of 200 mm and a stiffening ratio of 0.25, said structure exhibits an SIF that is reduced up to 15% as compared to a fuselage structure with 2024 plastic domain stringers.

76. A stringer that is not 2024, wherein if utilized in a fuselage with a stringer pitch of 200 mm and a stiffening ratio of 0.25, said structure exhibits an SIF that is reduced up to 5% as compared to a fuselage structure with 2024 alloy stringers.

77. A stringer that is not 2024, wherein if utilized in a fuselage with a stringer pitch of 200 mm and a stiffening ratio of 0.25, said structure exhibits an SIF that is reduced up to 15% as compared to a fuselage structure with 2024 plastic domain stringers.

78. A stringer geometry selected from one or more of the following:

<table>
<thead>
<tr>
<th></th>
<th>Z1</th>
<th>Z2</th>
<th>Z3</th>
<th>Z4</th>
<th>Z5</th>
<th>Z6</th>
<th>Z7</th>
<th>Z8</th>
</tr>
</thead>
<tbody>
<tr>
<td>Free flange width [mm]</td>
<td>12.7</td>
<td>12.7</td>
<td>12.7</td>
<td>12.7</td>
<td>12.7</td>
<td>12.7</td>
<td>12.7</td>
<td>12.7</td>
</tr>
<tr>
<td>Fastened flange width [mm]</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
<td>25.4</td>
</tr>
<tr>
<td>Height [mm]</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
<td>38.1</td>
</tr>
<tr>
<td>Free flange thickness [mm]</td>
<td>1.0</td>
<td>1.5</td>
<td>1.5</td>
<td>2.0</td>
<td>1.0</td>
<td>1.5</td>
<td>1.5</td>
<td>1.5</td>
</tr>
<tr>
<td>Fastened flange thickness [mm]</td>
<td>1.0</td>
<td>1.0</td>
<td>1.5</td>
<td>1.5</td>
<td>1.0</td>
<td>1.0</td>
<td>1.5</td>
<td>1.5</td>
</tr>
<tr>
<td>Web thickness [mm]</td>
<td>1.0</td>
<td>1.0</td>
<td>1.0</td>
<td>1.5</td>
<td>1.5</td>
<td>1.5</td>
<td>1.5</td>
<td>1.5</td>
</tr>
</tbody>
</table>
79. A fuselage structure comprising at least one stringer comprising an Al—Zn—Mg—Cu alloy, comprising (in mass percentage):
   a) Zn 8.3-14.0 Cu 0.3-2.0 Mg 0.5-4.5 Zr 0.03-0.15 Fe+Si<0.25
   b) at least one element selected from the group consisting of Sc, Hf, La, Ti, Ce, Nd, Eu, Gd, Tb, Dy, Ho, Er, Y and Yb, the content of each of said elements, if selected, being between 0.02 and 0.7%,
   c) remainder aluminum and inevitable impurities, and wherein Mg/Cu>2.4 and (7.0-0.4 Zn)>[Cu+Mg]>[6.4-0.4 Zn).

80. A fuselage structure according to claim 79, wherein if prepared with a stringer pitch of 200 mm and a stiffening ratio of 0.25, said structure exhibits an SIF that is reduced up to 5% as compared to a fuselage structure with 2024 T3 alloy stringers.

81. A fuselage structure according to claim 79, wherein if prepared with a stringer pitch of 200 mm and a stiffening ratio of 0.25, said structure exhibits an SIF that is reduced up to 15% as compared to a fuselage structure with 2024 plastic domain stringers.

82. A fuselage structure according to claim 79 wherein said at least one stringer geometry is selected from one or more of the following:

<table>
<thead>
<tr>
<th>Section [mm²]</th>
<th>Z1</th>
<th>Z2</th>
<th>Z3</th>
<th>Z4</th>
<th>Z5</th>
<th>Z6</th>
<th>Z7</th>
<th>Z8</th>
</tr>
</thead>
<tbody>
<tr>
<td>Equivalent thickness [mm]</td>
<td>1.0</td>
<td>1.1</td>
<td>1.3</td>
<td>1.3</td>
<td>1.3</td>
<td>1.3</td>
<td>1.3</td>
<td>1.5</td>
</tr>
</tbody>
</table>

| Free flange width [mm] | 12.7 | 12.7 | 12.7 | 12.7 | 12.7 | 12.7 | 12.7 | 12.7 |
| Fastened flange width [mm] | 25.4 | 25.4 | 25.4 | 25.4 | 25.4 | 25.4 | 25.4 | 25.4 |
| Height [mm] | 38.1 | 38.1 | 38.1 | 38.1 | 38.1 | 38.1 | 38.1 | 38.1 |
| Free flange thickness [mm] | 1.0 | 1.5 | 1.5 | 2.0 | 1.0 | 1.5 | 1.5 | 1.5 |
| Fastened flange thickness [mm] | 1.0 | 1.5 | 1.5 | 1.0 | 1.5 | 1.5 | 1.5 | 1.5 |
| Web thickness [mm] | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 |
| Section [mm²] | 76 | 83 | 95 | 102 | 95 | 102 | 102 | 114 |
| Equivalent thickness [mm] | 1.0 | 1.1 | 1.3 | 1.3 | 1.3 | 1.3 | 1.3 | 1.5 |

* * * *