Reduction of the Ultimate Factor by Applying a Maximum Load Concept

Otto Sensburg,* Otto Bartsch,† and Hans Bergmann‡
Messerschmitt-Boelkow-Blohm GmbH, Munich, Federal Republic of Germany

The ultimate factor for fixed-wing aircraft has not been reduced from 1.5 for more than 50 years. Many arguments can be made clearly indicate that the prediction of loads is more accurate today. A case can also be made that stresses can be determined with more accuracy by applying sophisticated finite-element methods. On the other hand, opponents can claim that, owing to these sophisticated prediction methods, the structure is much more exploited and therefore the factor of safety against failure should remain unchanged. Because these arguments cannot be settled—especially when dealing with a certification agency—a maximum load concept is proposed. The flight control system for a naturally unstable aircraft will limit the design parameters, such as acceleration, acceleration rate, velocity, attitude, in such a way that the limit design loads are not exceeded. The reliability of this load limitation is as high as that of the flight control system (FCS) itself, assuming no condition can be tolerated that will make the aircraft unstable or uncontrollable. Load cases that cannot be limited by the FCS—particularly landing cases—are still considered with an ultimate factor of 1.5. Also, loads arising from gusts must be treated with 1.5 if there is no gust load alleviation system available. Fortunately, most of the design cases for fighter airplanes arise from maneuvers, which is the reason that considerable structural weight reductions can be achieved when the proposed concept is applied.

Nomenclature

g = acceleration due to gravity
$HM_{T/E}$ = hinge moment of trailing-edge flaps due to α and $\eta_{T/E}$
i = number of applied standard deviations concerning strength
j = safety factor
j_{ik} = safety factor with regard to dispersion of applied loads and strength respectively
j_m = safety factor due to mean values of applied load and strength respectively
j_p = safety factor, proof value
j_u = safety factor, ultimate value, stated by airworthiness requirements
k = number of applied standard deviations concerning loads
$N_{F/P}$ = foreplane root shear due to α and $\eta_{F/P}$
n, n_y, n_z = load factor, general, lateral, and vertical
P = total probability of failure
$p(U)$ = probability density of strength
$p(V)$ = probability density of applied load
p = roll rate
$\dot{p}$ = roll acceleration
q = pitch rate
$\dot{q}$ = pitch acceleration
R = reliability of a structure
r = yaw rate
$\dot{r}$ = yaw acceleration
t = time parameter
U = strength
U_m = mean value of strength
V = applied load
V_m = mean value of applied load
V_E = aircraft velocity
α = angle of attack
α_U = coefficient of variation concerning strength
α_V = coefficient of variation concerning load
δ_4 = asymmetric trailing-edge flap deflection
ϵ = strain
$\eta_{T/E}$ = symmetric trailing-edge flap deflection
$\eta_{F/P}$ = foreplane deflection
σ_C = compressive stress
σ_T = tensile stress
σ_R = ultimate design value of stress (rupture) for metallic materials
$\sigma_{RTHEORETICAL}$ = theoretical ultimate value of stress for carbon fiber composites (CFC)
$\sigma_{R:1.5} = \sigma_{LIMIT}$ = limit design value of stress for metallic materials using $j_u = 1.5$
σ_{RC} = ultimate design value of compressive stress used for CFC
σ_{RT} = ultimate design value of tensile stress used for CFC
$\sigma_{RC:1.5}$ = limit design value of compressive stress used for CFC
$\sigma_{RT:1.5}$ = limit design value of tensile stress used for CFC
$\sigma_{0,2}$ = yield stress
σ_U = standard deviation of strength
σ_V = standard deviation of applied load
φ = bank angle

Received July 2, 1986; revision received April 8, 1987.

*Chief Structural Dynamicist. Member AIAA.
†Head, Dynamic Loads Department.
‡Development Engineer, Dynamic Loads Department.

Introduction

SAFETY factors were introduced into the design of aircraft structures[1] to take care of uncertainties, namely, 1) uncertainties in the theoretical or experimental determination of stresses, 2) scatter in the properties of structural materials and inaccuracies in workmanship and production, 3) deterioration of the strength of materials during the operational life of the aircraft, and 4) the possible occurrence of load levels higher than limit load.

As can be seen in Fig. 1, there were different opinions in various countries about the required safety factor. In the

United States, a factor of 1.5 was introduced in 1934, while it was not reduced to 1.5 in the United Kingdom until 1945. It could be argued whether unsafe airplanes were flying in the United States or if British airplanes were too heavy (about 20% more in structural weight) for 11 years.

A factor of 1.5 makes a lot of sense when looking at the stress/strain curve of aluminum alloy 2024 (Fig. 2), which was a widely used material at that time. The ratio of ultimate stress to yield stress was about 1.5 for that material and a structural designer was well advised to use that factor if the aircraft should not have bent wings every time the pilot only marginally exceeded the limit g. Modern materials (Fig. 2) do not exhibit this behavior and a safety factor of 1.5 cannot be justified by this consideration.

Table 1 lists the safety factors by several regulations. It shows that there is still quite a number of opinions. It is interesting to note[2] that manned spacecraft can fly with a factor of safety of 1.4 and that MIL-A87221, issued in 1985, specifies no value for the ultimate safety factor.

Generally, it can be stated that the present-day safety factor is a more or less arbitrary factor multiplied with the limit load or the load factor to make sure that the prescribed loads may be exceeded by a certain amount before failure of the structure occurs. It is also true that the presently used structural design and qualification procedure has produced structurally safe airplanes with a low probability of failure. Statistics show that structurally related accidents occur at a much lower rate than the overall accident rate. Looking at primary causes of these accidents, one finds that a large percentage is landing/undercarriage gear related.

By a probabilistic approach and a maximum load concept, it will be shown here that the same structural safety can be achieved on an airplane with a load-limiting, carefree handling, artificially stabilizing system with an ultimate factor of 1.3 as on one without such a system (stable airplanes) and a factor of 1.5. Obviously, a factor of 1.4 (with a load-limiting system) will give a safer aircraft standard than that reached for contemporary combat airplanes.

Prediction of Safety Factor by a Probabilistic Approach

In aircraft construction, stress analysis is based on the so-called limit loads, which are the maximum loads to be expected in service, and their basic conditions are laid down in the design requirements. These loads are multiplied by the ultimate safety factor j_u. Among other points, the most important one is to prove that no failure will occur up to and including these increased loads. To determine the ultimate safety factor j_u, the distance between the mean value of the applied load V_m and the mean value of the structural strength U_m has to be provided in such a way that the probability of an applied load is greater than V_m as,

$$V = V_m + k\sigma_V$$

together with the probability of the occurrence of a structure with a strength less than U_m,

$$U = U_m - i\sigma_U$$

result in a reliability R, which means a failure of the structure during its service life can be deemed improbable. In this case, reliability is defined to be the probability that the irregular changing strength U of the structures is greater at all times than the irregular changing of maximum loading V acting on these structures. If the probability density distributions for applied load and structural strength are known (see Fig. 3), the probability of failure $P(U>V)$ can be determined according to Ref. 3 in a generally valid form,

$$P(U>V) = \int_{V=0}^{V=\infty} p(V) \cdot \int_{U=0}^{U=V} p(U)\mathrm{d}U \cdot \mathrm{d}V$$

with the probability of load lying between V and $V+\mathrm{d}V$

$$p(V) \cdot \mathrm{d}V$$

and the probability of strength levels lower than V

$$P(U<V) = \int_{U=0}^{U=V} p(U)\mathrm{d}U$$

Correlation between reliability R and the failure probability reads as

$$R = 1 - P$$

If the density functions $p(V)$ and $p(U)$ represent Gaussian distributions and if there are known mean values of U_m and V_m, as well as known standard deviations of σ_U and σ_V, the failure probability can be determined through reliability ac-

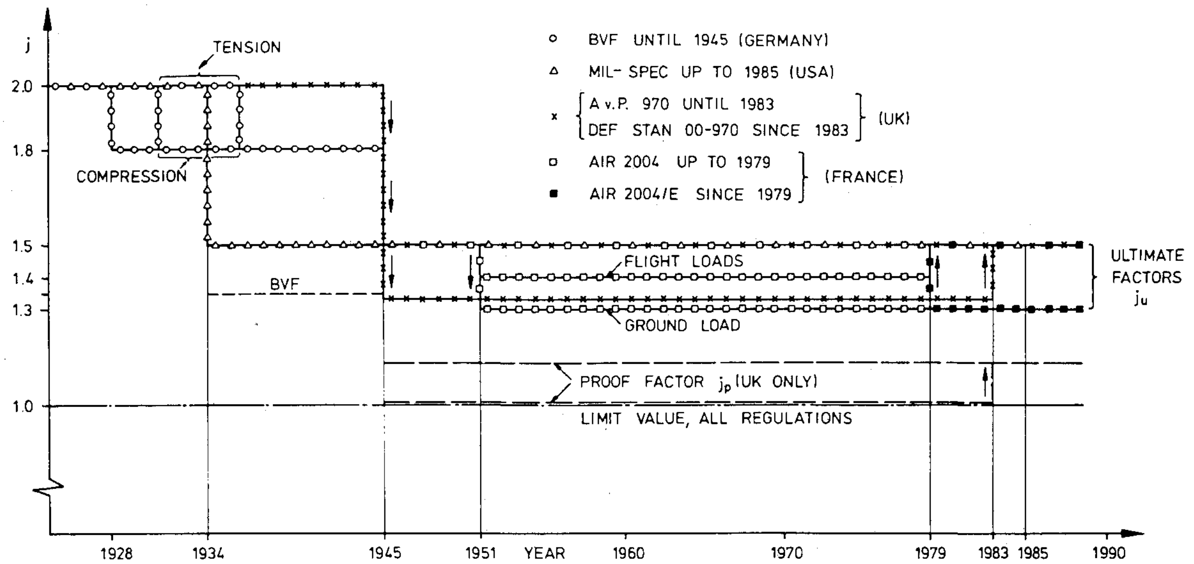


Fig. 1 History of safety factor.

Table 1 Load definition and safety factors for flight conditions from regulations valid in 1986

Country/ Regulation	Definition of limit load	Definition of ultimate load	Ultimate safety factor j_u	Definition of proof load	Proof factor j_p
USA MIL-A87221 (USAF) 28.2.85	Maximum loads which can result from authorized flight use of the air vehicle including maintenance activity. Requirement: no detrimental deformation shall occur.	Limit load × ultimate factor Requirement: no structural failure shall occur.	No value specified	Not defined	–
USA FAR Part 25	Maximum load to be expected in service. Requirement: no detrimental permanent deformation of the structure. Deformation may not interfere with safe operation.	Limit load × ultimate factor Requirement: no failure of the structure for at least 3 seconds	1.5	Not defined	–
UK DEF Stan 00-970	The greatest load that will occur during specified life under an average spectrum of loads. Requirement: see definition of proof load.	Limit load × ultimate factor Requirement: a complete item shall not collapse and stress at each feature; shall not exceed the allowable value for that feature.	1.5	Limit load × proof factor Requirement: a complete item shall not sustain deformation detrimental to safety, moving parts essential to safety shall function satisfactorily. No effects of loading after removal of the load that reasonably cause the aeroplane to be deemed unserviceable.	≥1.125
UK BCAR	Maximum load anticipated in normal conditions of operation. Requirement: see definition of proof load.	Limit load × ultimate factor Requirement: the primary structure shall be capable of supporting the ultimate load.	≥1.5	Limit load × proof factor Requirement: primary structure shall be capable of supporting all loads up to and including the proof load. Elastic and permanent deformation occuring during application of this load shall not interfere safe operation of aeroplane. Permanent deformation as remains after removal of the load shall not significantly reduce airworthiness. Moving parts essential to safe operation of aeroplane shall function satisfactorily up to and including this load.	1.0
France AIR 2004/E	Maximum loads allowable in service. Requirement: limit load may not cause permanent deformations larger than 5% of deformation under this load. Moving parts esential to safety may operate satisfactorily. Permanent deformations may not significantly reduce the flying qualities.	Limit load × ultimate factor Requirement: no failure and no inadmissible deformation of the structure may occur.	1.5	Not defined	–

cording to the equation,[6]

$$R = \Phi\left(\frac{U_m - V_m}{\sqrt{\sigma_U^2 + \sigma_V^2}}\right)$$

Applying the coefficients of variation,

$$\alpha_V = \sigma_V / V_m, \quad \alpha_U = \sigma_U / U_m$$

reliability R reads as

$$R = \Phi\left(\frac{(U_m/V_m) - 1}{\sqrt{\alpha_U{}^2 (Um/Vm)^2 + \alpha_V{}^2}}\right)$$

The safety factor j_{ik} due to the structural strength diminished by $i\sigma_U$ and the applied load increased by $k\sigma_V$ is defined by

$$j_{ik} = \frac{U_m - i\sigma_U}{V_m + k\sigma_V}$$

Applying the coefficients of variation α_V and α_U, the safety factor is

$$j_{ik} = \frac{U_m}{V_m} \cdot \frac{1 - i\alpha_U}{1 + k\alpha_V} = j_m \cdot \frac{1 - i\alpha_U}{1 + k\alpha_V}$$

and, finally, the safety factor demanded can be written as

$$j_u = j_m = \frac{U_m}{V_m} = j_{ik} \frac{1 + k\alpha_V}{1 - i\alpha_U}$$

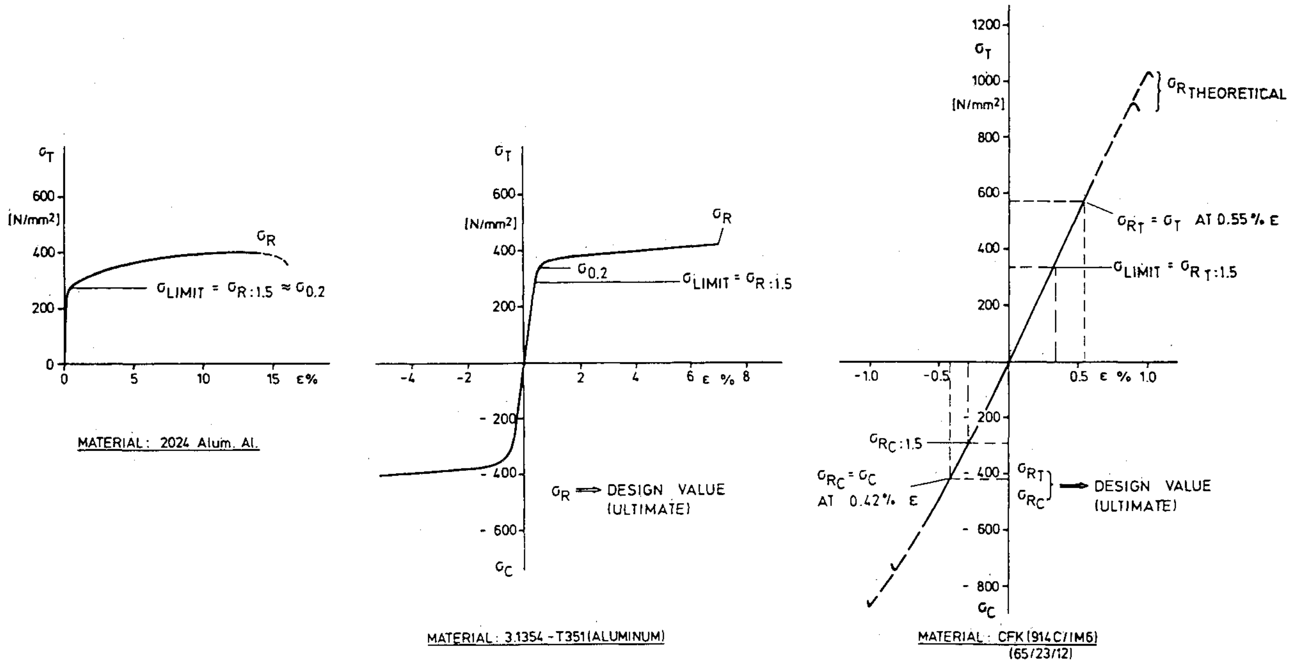


Fig. 2 Comparison of various materials under consideration of ultimate safety factor j_u = 1.5

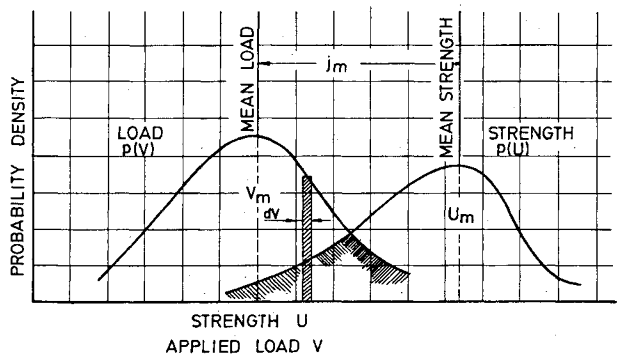


Fig. 3 Probability of failure due to load and strength variations.

Now it can be seen clearly that at constant reliability and failure probability, respectively, the ultimate safety factor j_u can be diminished when the dispersion of the applied loading represented by α_V can be reduced. This relation can generally be seen in Fig. 4 showing the failure probability vs the ultimate safety factor for different coefficients of variation, as well as in Fig. 5 showing the ultimate safety factors caused by probability density distribution $p(V)$ due to various dispersions.

The two factors mainly influencing the safety factor level, structural strength and applied loading, are now discussed separately.

Concerning strength, three main variation factors can be identified:

1) *Uncertainties in the theoretical or experimental determination of stresses.* Even if errors are made in structural analyses, the major static test will uncover them. The only remaining uncertainties will stem from the differences between the test and service aircraft structures. This aspect can be dealt with through rigorous quality control. This view was expressed by experts of three major aircraft military airworthiness authorities, namely France, Sweden, and Germany.[4]

2) *Scatter in the properties of structural materials and in accuracies in workmanship and production.* Quality control,

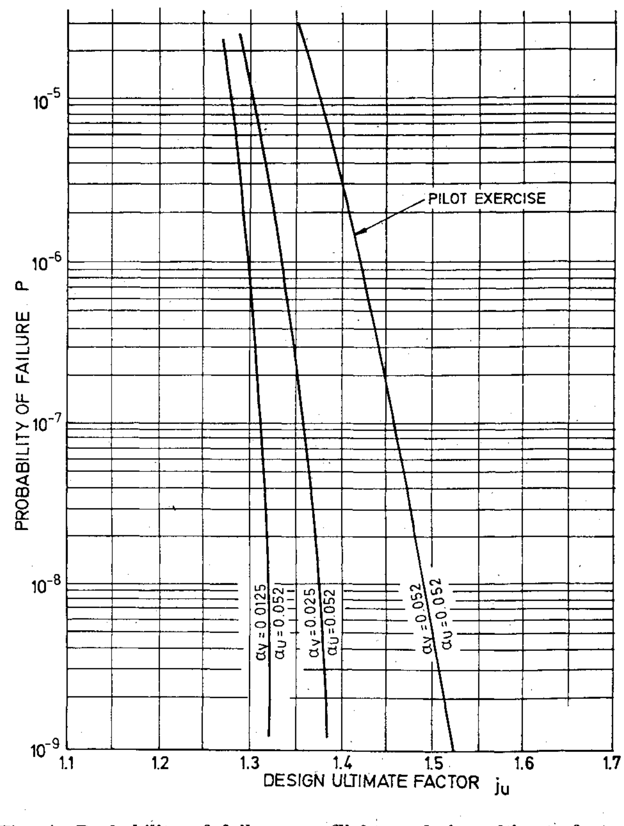


Fig. 4 Probability of failure per flight vs design ultimate factor, normal distributions of loading and strength assumed.

allowable design values (A-B values) together with inspection techniques will keep this scatter low.

3) *Deterioration of the strength of materials during the operational life of the aircraft.* This is mainly a fatigue problem and is not related directly to the factor of safety on ultimate strength. Three considerations should be mentioned:

a) Forty percent of the structure is made of carbon fiber composites (CFC), which are not fatigue critical.

b) Whatever dimensions are necessary for fatigue-critical parts will be installed in the aircraft (this is one of the reasons why the weight savings due to ultimate factor reductions are not proportional to such reductions).

c) A major fatigue test will be performed.

The use of carbon fiber material does not increase the scatter noted above because CFC allowances already reflect the disadvantages of the composites compared to metals such as barely visible impact damage, hot/wet reductions (see Fig. 2 where, for instance, 0.55 of strain is used instead of 1.2% to cover impact damage), or higher variability.

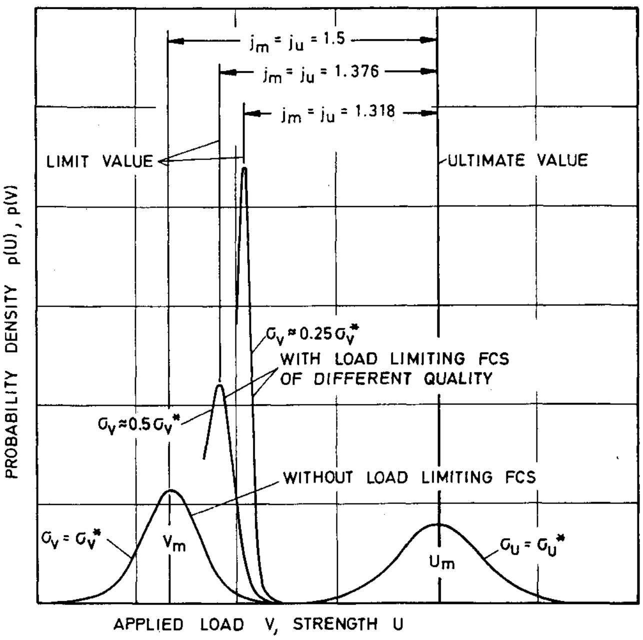


Fig. 5 Dependence of safety factor on dispersion of applied loads with and without load limiting flight control systems (FCS) (probability of failure = const).

Repair of battle damage is also a consideration for using the reduced strain allowable with CFC's, thus enabling holes with a diameter of up to 6 mm to be drilled into major CFC structures for repair rivets. Metal corrosion, which reduces strength, must be avoided by proper design.

Summarizing, it can be said that the scatter factor on strength is low and no argument can be brought forward that it is increased with the use of carbon fibers. If it was more dangerous to use CFC materials, then airplanes built with these materials, which are already flying, should have been certified with a higher safety factor.

Concerning applied loads, there can be large excesses over the limit commanded by pilots on conventional airplanes. (See Figs. 6 and 7.) The maximum load concept means that if these excesses can be reliably limited by the flight control system, then the ultimate factor can be reduced. This possibility was indicated in Ref. 1. Also, the British regulation Av.P. 970 reads: "Group (b) loading case: Cases where the severity of the loads, which may arise in service, is limited by some reliable means to a definite value, which cannot be exceeded." For these cases, it states: "The conception of a specified ultimate load is hardly applicable here." It is also said in Ref. 4 by German military airworthiness authorities that if a different factor can be applied when an operational loads-limiting system is built in the aircraft, "If in special cases it can be proven, that 1.5 limit load physically is not achievable (e.g. limiting device) a lower factor of safety could be accepted."

A calculation was performed in Ref. 3 assuming a normal distribution for load and strength probabilities with a standard deviation of 0.052 times the mean value (Fig. 4). The coefficient of variation 0.052 is reduced for loads to 0.0250 and 0.0125, which is interpreted as the effect of a load-limiting system. The same probability of failure of 4.6×10^{-9} can be reached with a factor $j_m = 1.5$ (coefficient of variation 0.052), $j_m = 1.37$ (coefficient of variation 0.025), or $j_m = 1.32$ (coefficient of variation 0.0125). So, clearly, three airplanes having different load-limiting capabilities have the same safety standard. The related load distributions are shown in Fig. 5. Two important things must be pointed out:

1) A maneuver load control system can limit maneuver loads to a wider extent – gust loads cannot be limited to the

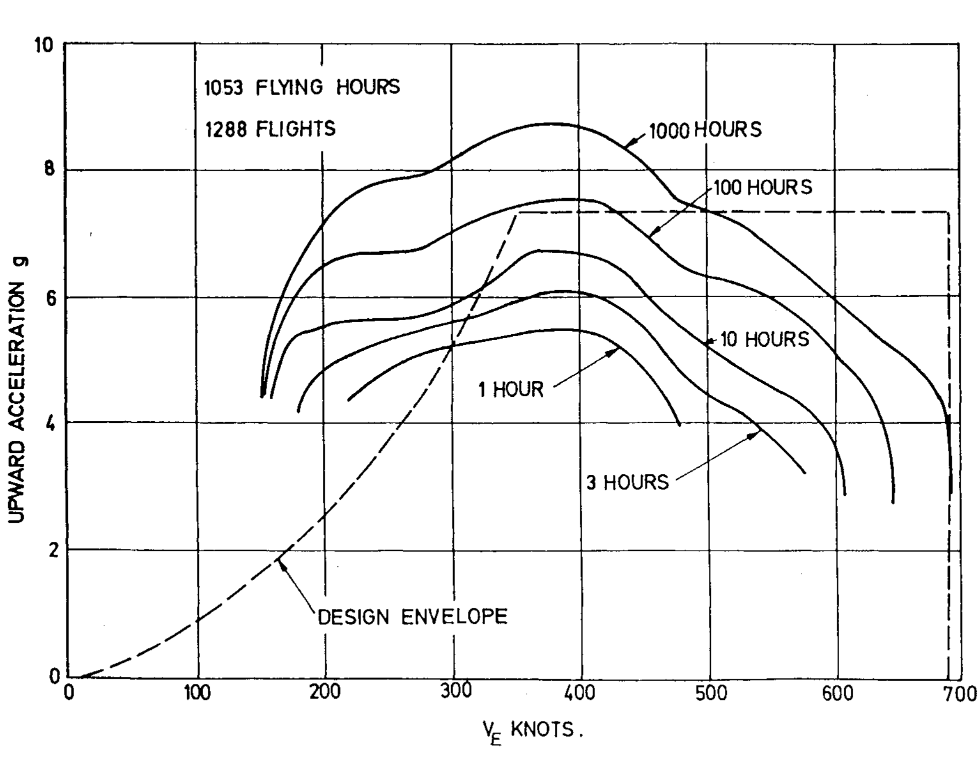


Fig. 6 F-100A Ellis Air Force Base composite *V-g* envelopes for all missions.

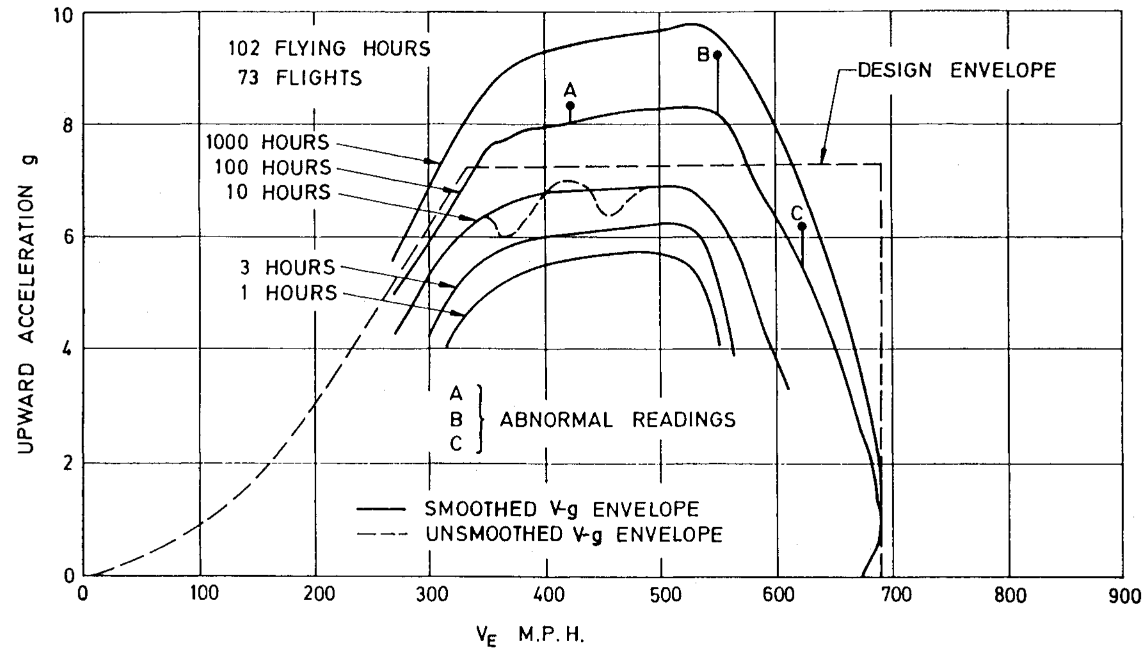


Fig. 7 F-86A combat engagements (Korea) *(V-g)* envelopes for air-to-air missions.

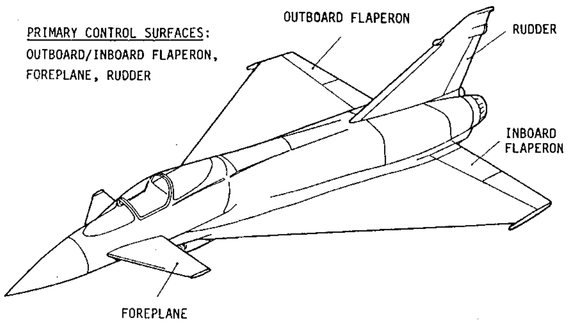


Fig. 8 Aircraft control surfaces.

same extent. On fighter airplanes, all of the design loads result from maneuvers. The superposition of gust cases and maneuver cases is excluded by military specifications. Either maximum gust cases with level flight conditions or reduced gust speeds with 0.6 $n_{z\max}$ are considered.

2) Loads that cannot be safely limited by the system should still have ultimate factor of 1.5 on the limit load or 1.225 on the design landing speed. This will also maintain the same safety standard on a large percentage of structural failure cases.

In the next section, an attempt is made to explain the carefree maneuvering system and how design loads are produced with such a system.

Carefree Handling and Maneuver Load Control

A fighter aircraft with its control surfaces is shown in Fig. 8. All of its primary flight control surfaces are interfaced and controlled in their motion through the Flight Control System (FCS) and the pilot commands maneuvers with his stick and pedals. As will be shown in the following figures, the way in which the maneuvers are flown is very different from the stable command-augmented airplanes that use only one control surface and the design loads depend very much on the FCS. This is explained in more detail in Ref. 5.

As shown in Fig. 8, the primary control surfaces are the inboard flaperons, outboard flaperons, foreplane, and rudder. Inboard/outboard flaperons *and/or* the foreplane can, therefore, be used for trimming and controlling longitudinal aircraft motion. Such control depends on the allocation of the stick inputs to these control surfaces.

Figure 9 shows an example of how much the foreplane and wing trailing-edge flap loads can be affected by an appropriate choice of the initial trim contribution. This applies for the subsonic region (where the aircraft is unstable longitudinally), as well as for the supersonic region (where the aircraft is stable).

It is interesting to note that the aircraft needs only small control deflection angles for the initiation of a maneuver—as expected—but the control deflection must be immediately checked in order to stop the effect of the instability. Therefore, no similarity of the stick input in comparision to the actual control surface deflections can be seen subsonically, while supersonically the usual increasing control deflection is seen to command a steady maximum *g* condition. If the foreplane/flap schedule is chosen from only the point of view of handling and performance, one may run into problems with the design loads on *both* surfaces, as one gains advantages on *both* surfaces by choosing an optimum loads concept. As demonstrated for trim, we tried to show the effect of controlling the aircraft alternatively by foreplane *or* trailing-edge flaps. Figure 10 shows this effect for the pitch maneuver starting at the nearly optimal foreplane of -5 deg. Again, it can be seen that the contribution of either foreplanes *or* trailing-edge flaps can strongly affect the respective control surface loads, at least for the supersonic case. It must be noted that, to demonstrate these cases, a given control system has been degraded by changing the assignment of foreplane/flap command and feedback paths, which may be seen in the nonoptimal motion of the *g* time history (Fig. 10). Figure 11 illustrates the problem that military specification requirements no longer represent generally usable structural design conditions for a carefree handling aircraft. It can be seen that the MIL-type triangular stick displacement initiates a full *g* maneuver with associated high positive and negative pitch rates and operationally unacceptable acceleration rates, resulting in high inertia loads, for the pilot. The carefree handling aircraft on the right side of the diagram is controlled to its maximum *g* by a full back stick and it can be seen that both the maximum *g*

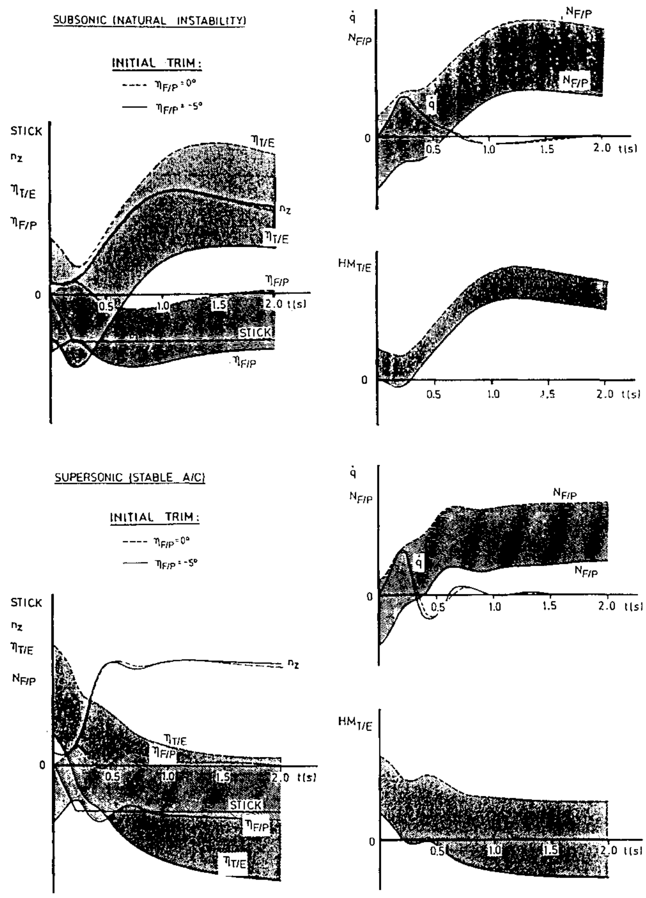


Fig. 9 Adaption of trim control for load optimization.

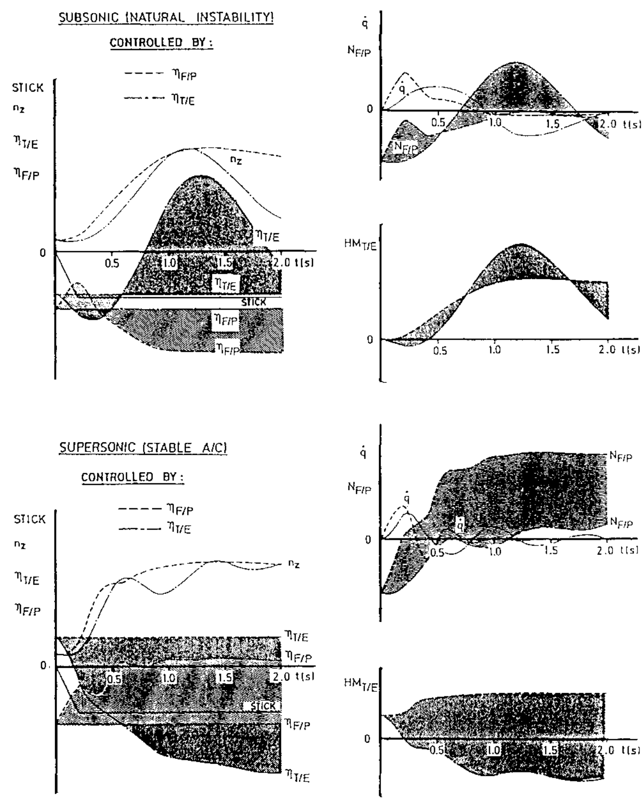
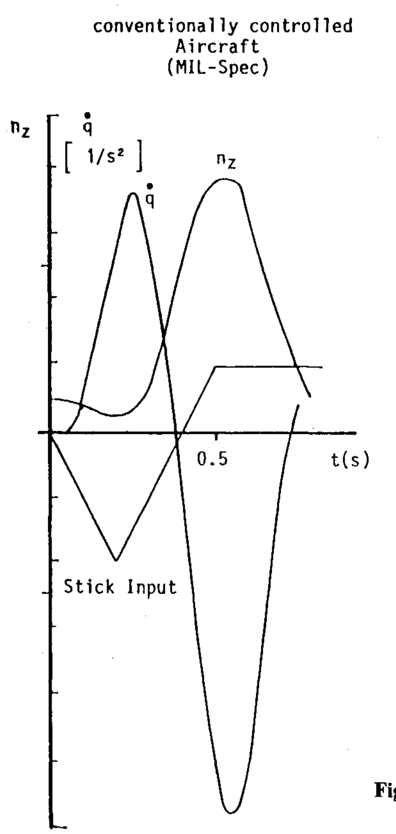


Fig. 10 Adaption of dynamic maneuver control for load optimization.



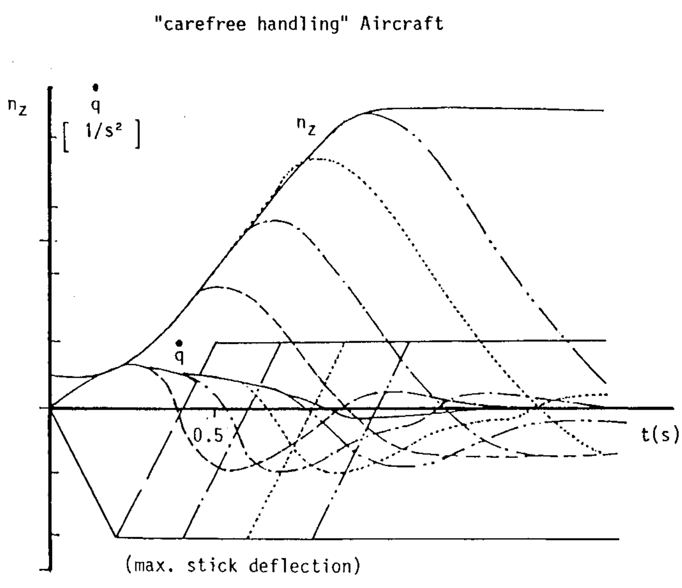


Fig. 11 Comparison of MIL-specification and carefree handling maneuvers.

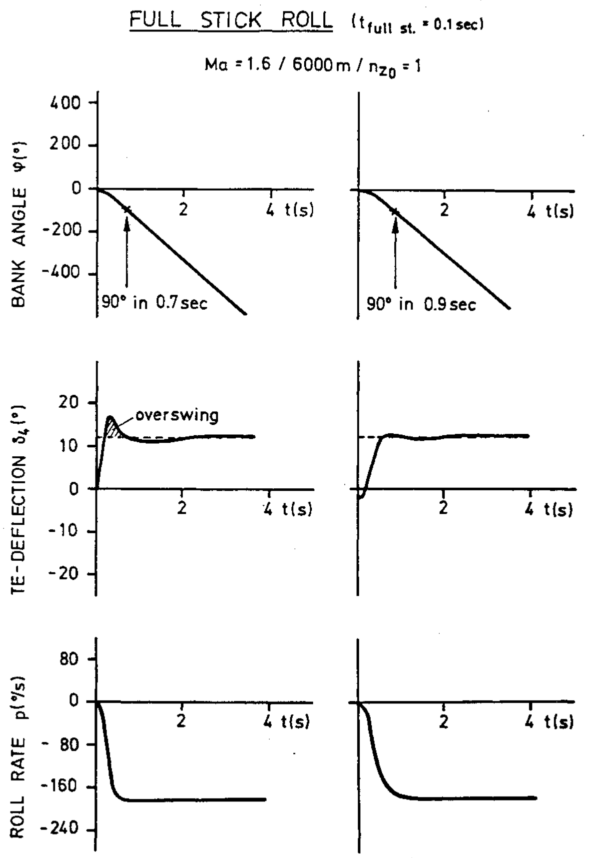


Fig. 12 Adaption of control system in order to achieve 36% reduction (trailing-edge flap).

and pitch rates are reduced to operationally meaningful levels by the control system. Naturally, an immediate triangular reversal is stopping the maneuver initiation and one will not reach maximum g as with a conventional aircraft, where the pilot has to be very careful in performing a MIL-type triangular maneuver without exceeding the g limits.

Finally, it should be emphasized that in designing the control system it is very important to include the design loads at a very early stage, as can be explained by Fig. 12. It may be accepted from a handling/performance point of view to overfulfill the time-to-bank requirement by an overswinging control deflection. This clearly increases the control hinge moments, so load reductions (in this case about 36%) are achieved by simply adopting the maximum acceptable time to bank.

All the examples shown clearly illustrate that the aerodynamic loads produced by maneuvers depend on carefully chosen control laws/trim programs of the flight control system.

Real-time simulations with measured aerodynamic derivatives, aerolastic efficiencies, and optimized control laws must be performed and maximum response parameters selected from time histories (Fig. 13). These response parameters are expressed as accelerations and rates in all axes and altitudes. A proper functioning flight control system also has to consider aircraft weight, c.g., and fuel state. From these, the actual design loads are derived and they are dependent on the FCS design, for the flying aircraft, on performance.

Weight Savings with a Reduced Ultimate Factor

The possible weight savings are shown in Fig. 14 for a delta-wing aircraft. There is is a much smaller percentage of structural weight saved than the percentage by which the

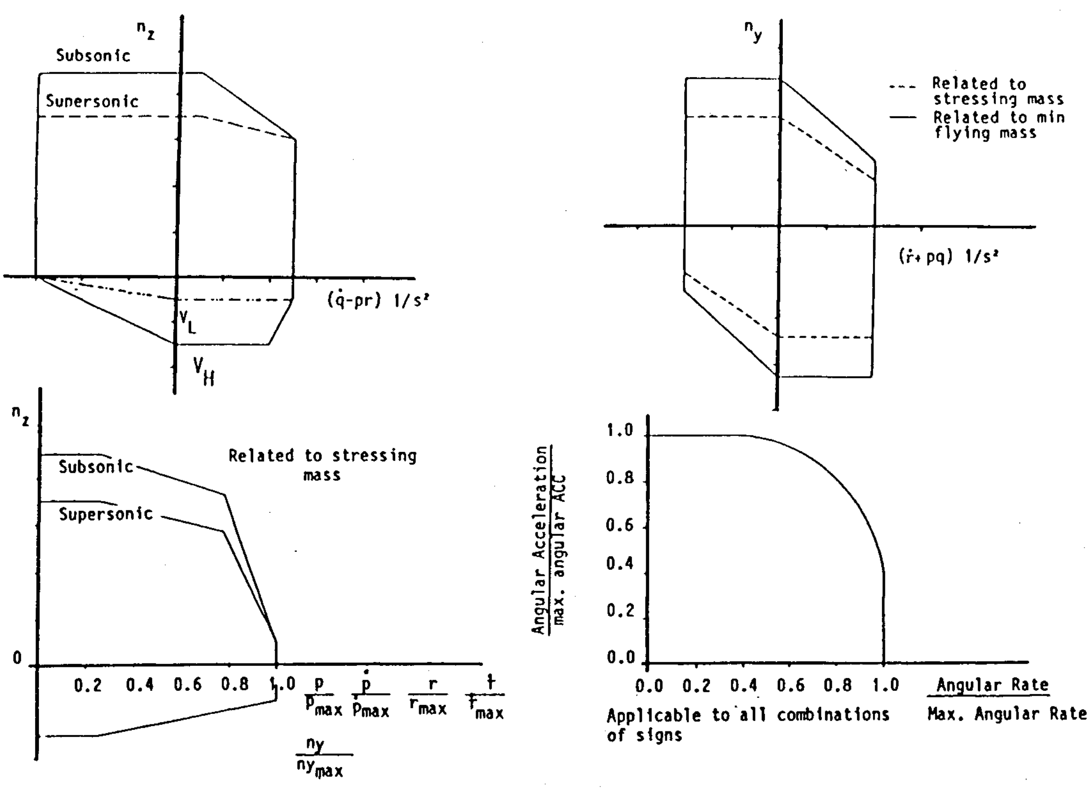


Fig. 13 Flight parameter envelopes for structural design.

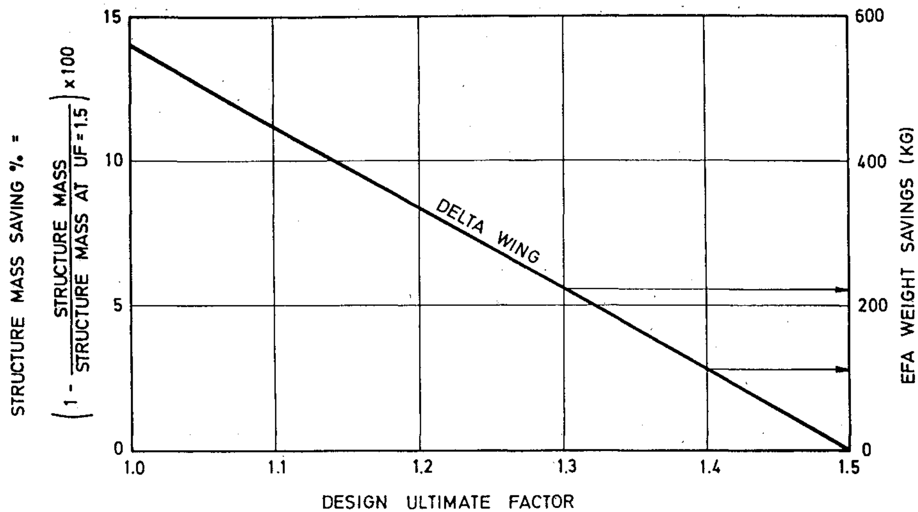


Fig. 14 Savings in structure mass for variations of ultimate factor (fully resized aircraft).

ultimate factor is reduced, which illustrates that a large portion of structure is designed with other constraints such as fatigue, minimum gages, etc.

Conclusions

1) Unstable aircraft are completely dependent upon the integrity of the flight control system.

2) The applied loads are also dependent upon the flight control system. Therefore, the design loads cannot be exceeded and the ultimate factor can be reduced to 1.4 or less.

Acknowledgment

The authors want to express their gratitude for the very useful ideas contributed to the subject by Mr. R.J. Sellars of BAE-Warton and Sig. L. Casalegno of AIT-Torino.

References

[1]Struck, H., "Factor of Safety/Limit Load Concept," AGARD Rept. 661, Nov. 1977.

[2]Muller, G.E. and Schmid, C.J., "Factor of Safety – USAF Design Practice," AGARD Rept. 661, Nov. 1977.

[3]Heath, W.G., "Factors of Safety – Should They be Reduced?," AGARD Rept. 661, Nov. 1977.

[4]"Factors of Safety Related to Structural Integrity," AGARD Rept. 677, June 1981.

[5]Sensburg, O. and Schnidinger, G., "Integrated Design of Structures," Messerschmitt-Boekow-Blohm GmbH, Munich, FRG, Rept. S/PUB/240, April 1986.

[6]Oery and Kattge, "Zuverlässigkeit von Strukturen," ERNO Rept. 204/66, Dec. 1966.