NON-PERIODIC INSPECTIONS OF AGING AIRCRAFT STRUCTURES

Andreas Rambalakos
Mgr. of Structures Engineering - Dassault Falcon Jet Corp., NJ
Ph.D. candidate - Dept. of Civil Engineering & Engineering Mechanics, Columbia University, NY

George Deodatis, Ph.D.
Professor - Dept. of Civil Engineering & Engineering Mechanics, Columbia University, NY

ABSTRACT

This paper presents a methodology for computing the evolution and resulting gradual deterioration of the reliability of an aircraft structural component by employing a direct Monte Carlo simulation approach. Uncertainty is considered at this stage with the time to crack initiation and the probability of crack detection. The structural component is assumed to consist of a prescribed number of elements. The methodology is used to determine the required non-periodic inspections so that the reliability of the structural component will not fall below a prescribed minimum level. A sensitivity analysis is conducted to determine the effect of three key parameters on the specification of the non-periodic inspection intervals: namely a parameter associated with the time to crack initiation, the applied nominal stress fluctuation and the minimum acceptable reliability level. This is a report of ongoing research and as such its conclusions could be somehow modified in the future upon completion.

INTRODUCTION

In the last three decades, aircraft structural design has gone through several interactive changes engendered by the complex combination of external load sources, environments, human interaction and economical constraints. These parameters affect the flight regime in which aircraft operate and determine the extent of their service life. Starting initially from static strength considerations, fatigue and the formation of cracks have eventually emerged as the main challenges that have precipitated in several changes concerning aircraft design philosophies (*safe life*, *fail-safe* and *damage tolerant*). In this age of a growing number of aging aircraft fleets, aircraft manufactures and operators are tasked to maintain and extend the useful service life beyond the original design life through extensive maintenance programs. These programs involve continuous inspections of the aircraft structure or system that vary in scope and details depending on the aircraft type and mission profile.

Several studies in the past have focused on determining the reliability of an aircraft structural component through the application of periodic and non-periodic inspections. The studies conducted by Yang and Trapp [1] and Shoji *et al*. [2] examined the reliability of fatigue critical structural components using probabilistic methods and under prescribed periodic inspections. The work done by Deodatis *et al*. [3], Ito *et al*. [4] and Deodatis *et al*. [5] introduced the application of non-periodic inspections through a Bayesian approach in order to maintain or exceed a prescribed minimum reliability level through out the service life of the fatigue sensitive structural component. In those studies, the structural system was assumed to be comprised of several elements connected in "series" and in which probabilities of failure were considered before and after crack initiation.

The crack propagation rate was simplified to a closed form by neglecting the dependency of the stress intensity factor on the geometric correction factor. Furthermore, upon failure of an element, there was equal load redistribution to the surviving elements or no redistribution at all.

In this paper, the work by Deodatis *et al*. [5] is expanded to evaluate the reliability as a function of time of an aircraft structural assembly through the application of a brute force Monte Carlo simulation method. The structural assembly under study is a fuselage skin lap joint which is not modeled using any simplistic modeling assumptions in terms of system failure (i.e. series or parallel configurations), while satisfying damage tolerance requirements as specified by federal regulations ([6] and [7]). Similarly with the work in [5], an inspection is performed when the system reliability reaches a pre-specified minimum level. System failure can only occur after crack initiation in the form of multi-site damage (MSD). The crack propagation rate is more accurate since now the dependency of the stress intensity factor on the geometric correction factor is taken into account. Also, unlike previous literature, here there is unequal (and more realistic) load redistribution to the surviving elements following failure of an element.

This is a report of ongoing research and as such its conclusions could be somehow modified in the future upon completion.

STRUCTURAL ELEMENT AND SYSTEM DEFINITIONS

In this analysis, the structural system under study is the fuselage skin lap joint on a Dassault Falcon 900 jet. The system is bounded between frames C8 and C10 and along stringer No.L4 as shown in Figure (1). Furthermore, it is assumed that the central frame (No.C9) is broken and thus the system is a two bay unstiffened fuselage structure. The skins are chemically milled from aluminum 2024-T3 CLAD and have a nominal thickness of 1.2 mm (0.050”). The typical representative values for frame spacing, fastener pitch and element dimensions are shown in Figure (2). The configuration of the typical skin element was selected to be consistent with previous literature that have studied the presence of MSD [8] in fuselage skin lap joints and also the estimation of stress intensity factors for cracks emanating from fastener holes [9]. The cracked element is assumed to have a crack of length “a” on both sides of the hole and is subjected to a hoop stress, σ_{HOOP}, resulting from cabin pressurization (primarily), aerodynamic loads and fuselage torsional/bending loads. It should be noted that this methodology can be applied on any type of structural configuration and under any load spectrum. The selection of the fuselage skin lap joint subjected to MSD was chosen for its simplicity and its criticality in terms of structural integrity for aging aircraft.

FUNDAMENTAL ASSUMPTIONS/DEFINITIONS

In the course of analyzing the fuselage skin lap joint with MSD, the following assumptions and definitions were made in order to provide a framework for analysis that could be subjected to further refinement or alterations depending on the system under study.

Unit of Time

In this paper, time is measured in number of flights or equivalently in pressurization cycles where in each flight the cabin is pressurized during flight and depressurized upon landing. If it is desired to think in terms of flight hours, then on average the equivalency is 1.5 flight hours per flight.

Time To Crack Initiation

The probability distribution for the Time To Crack Initiation (TTCI) is assumed to follow a two parameter Weibull distribution ([1] and [3]-[5]) with probability density function given by:

$$TTCI(t) = \frac{\alpha}{\beta} \cdot \left(\frac{t}{\beta}\right)^{\alpha-1} \exp\left[-\left(\frac{t}{\beta}\right)^{\alpha}\right] \tag{1}$$

where α is the shape parameter and β is the scale parameter. The shape parameter is usually taken to be equal to 4.0 for aluminum structures. The scale parameter can be perceived as a value below which lie 63.2% of all observations. Several considerations have been taken into account for the selection of an appropriate value for the scale parameter:

- Parameter β can be estimated from full scale fatigue testing or, in the absence of full scale tests, the parameter can be estimated from the design mission spectrum, the pertinent S-N diagram and the cumulative damage hypothesis [1].

- Another suggestion [4] is that β is a nonlinear function of the stress intensity level that the structure is subjected to. This non-linear variation follows the typical S-N diagram of a particular alloy that can be found in any approved material handbook such as reference [10]. It should be noted that typical S-N diagrams report the *average* number of cycles to failure for a given stress level where failure is defined as the complete rupture of the test coupon.

- Finally, another way to estimate β is through its dependency on the design fatigue life of an aircraft structure for a given reliability level [11]. Many aerospace manufacturers set the design service life objectives for aircraft structures with a minimum of 95% reliability. For aluminum structures a 95% reliability level in terms of crack initiation implies that the fatigue life of the structure is twice the design service life objective. Therefore, for a design service life objective of 20,000 flights and for a Weibull distribution for fatigue crack initiation, the beta-parameter can be estimated from:

$$1 - \exp\left[-\left(\frac{N_R}{\beta}\right)^4\right] = 0.05 \tag{2}$$

where N_R is equal to 20,000 flights. Solving equation (2) for β, a value of 42,026 flights is obtained which is approximately twice the design objective service life. The value of β equal to 40,000 flights will be used in this paper as the nominal value since the design service life objective for a Dassault F900 jet is 20,000 flights.

Crack Propagation Law

Once a crack is initiated, it will propagate with a rate in accordance with the power law from Paris equation [12]:

$$\frac{da}{dt} = C_1 \cdot \left(\Delta K\right)^{m_1} \tag{3}$$

where C_1 and m_1 are material constants and ΔK is the stress intensity range. For aluminum 2024-T3 the m_1 constant assumes a value of 3.59 and the C_1 constant is equal to 1.333 x 10^{-20} (when the characteristic length is expressed in inches and the stress is measured in psi [12]). The stress intensity range, ΔK is given by:

$$\Delta K = F\left(\frac{a}{r}\right) \cdot \Delta\sigma \cdot \sqrt{\pi \cdot a} \tag{4}$$

where $F(a/r)$ is a geometric correction function that depends on the ratio of the crack length a over the hole radius r for cracks emanating from fastener holes and $\Delta\sigma$ is the nominal stress fluctuation. Specific values of the geometric correction function in terms of various a/r ratios are given in reference [9]. The tabulated values that pertain to simultaneous cracks emanating from both sides of a hole were considered from [9] and an effort was made to curve fit an exponential decay function by minimizing the error. The best curve fit with the least error yielded the following expression for $F(a/r)$:

$$F\left(\frac{a}{r}\right) = 1 + 2.36 \cdot \exp\left[-2.08 \cdot \left(\frac{a}{r}\right)\right] \tag{5}$$

Equation (5) depicts the influence that the high stress field at the hole edge has on the stress intensity factor at the crack tip through equation (4). At low values of a/r, the crack tip is subjected to the high stress zone due to the stress concentration from the presence of the hole. As the crack propagates, the tip will eventually exit this high stress zone and it will then be subjected to the far field stress, thus reducing its propagation rate. Refer to Figure (2) where the high stress zone is depicted in the typical element at the edge of the hole. Equation (3) can be integrated numerically to yield the crack length as a function of time. However, for computational expedience, the variables are separated and the crack propagation law is kept in differential form where the time increment dt has a value of 100 flights. In order to justify the use of a time increment of 100 flights the behavior of the crack propagation law was also tested with a time increment of 10 flights. The results showed no difference in the crack propagation for crack lengths allowed in this model before element failure is achieved. Therefore, from equation (3):

$$da = C_1 \cdot (\Delta K)^{m_1} \cdot dt \tag{6}$$

where ΔK and $F(a/r)$ are given by equations (4) and (5) respectively and the crack increment, da is added to the previous crack length to yield the current crack length.

Probability of Crack Detection

The probability of detecting (POD) a crack of length a during an inspection is given by a three parameter Weibull distribution [5]:

$$D(a \mid d) = 1 - \exp\left[-\left(\frac{a - a_{\min}}{d - a_{\min}}\right)^{\varepsilon}\right] \tag{7}$$

In the above formula, a_{min} is the minimum detectable crack length and d and ε are constants. The values for parameters d and ε are set to 1.57 inches (40 mm) and 1.4 respectively [5]. The minimum detectable crack length is set equal to the initial crack size [5] which in this paper will be assumed to be 0.050 inches.

Applied Nominal Stress Fluctuation

The entire system will be subjected to a nominal stress fluctuation $\Delta\sigma$ which is the result of cabin pressure differential (primarily) between the cabin pressure and the ambient atmospheric pressure at altitude, aerodynamic loads and fuselage torsional/bending loads. In this paper, $\Delta\sigma$ will be set equal to 20,000 psi as the nominal applied stress and with a stress ratio of zero. It should be noted that the maximum normal cabin pressure differential in a Falcon 900 is 0.067 MPa (9.72 psi) which translates into a hoop stress of 10,123 psi. Our assumption here is that aerodynamic loads and fuselage torsional/bending loads bring up the overall value of $\Delta\sigma$ to 20,000 psi. Although the contribution from aerodynamic loads and fuselage torsional/bending loads might not be enough to bring the overall $\Delta\sigma$ up to 20,000 psi, this value is adopted in this study in order to demonstrate the capabilities of the proposed methodology in a meaningful way (values of $\Delta\sigma$ significantly lower than 20,000 psi cannot produce failures from crack propagation because of extremely low rates of crack propagation).

Element and System Failure Definitions

Cracks will initiate on the top row of rivets on the outer (top) skin of the lap joint as shown in Figure (2). This is the row with the highest probability of crack formation due to the higher pin load experienced by the rivets and the presence of a bypass load. The same load conditions exist on the bottom row of the inner (bottom) skin. However, the top outer row is also countersunk, a fact that increases the probability for crack formation. Referring to Figure (2) again for system and element definitions, an element can fail under two possible conditions:

1. A *cracked element* will fail when the stress in the element net section reaches the yield strength of the material. This failure criterion has been proposed in [13] and [14] and will be adopted in this paper. In addition, reference [14] proposes another failure criterion by which failure of a cracked element is reached when there is contact between the crack tip plastic zones of two approaching cracks. This failure criterion is less conservative because it allows more time to progress before element failure is achieved due to the small size of the crack tip plastic zones and thus will not be applied in this paper.

2. An *uncracked element* can fail if the applied load on it causes net section yielding. This failure criterion is most probable to be observed after several cracked elements have failed and redistributed their loads in adjacent surviving elements. This could have a cascading effect leading eventually to system failure.

The *system* will fail when the surviving elements can no longer prevent net section yielding upon application of 115% of the nominal stress fluctuation, $\Delta\sigma$. This is consistent with the residual strength requirements under damage tolerance considerations in accordance with references [6] and [7] for pressurized cabin structures. If the system under analysis was other than the pressure vessel, then residual strength requirements would dictate that the damaged structure be able to withstand limit loads without significant changes in structural stiffness or geometry or both. It should be noted that the current methodology checks for system failure at every time step meaning that it is assumed that the system is experiencing 115% of the maximum normal stress at every time step (one time step is 100 flights). This is believed to be conservative because aircraft are usually not experiencing limit loads at every flight or every 100 flights for that matter.

Another system failure criterion would be when the stress intensity factor of a coalesced crack reaches the fracture toughness of the skin material. However, this failure criterion was found not to be critical for our structural configuration and thus, will not be used in this paper.

Inspection Process

During a scheduled inspection, the structural system and all its elements will be inspected for cracks and element failures. If a crack is detected, then the element is repaired or replaced. Similarly, if an element has failed due to net section yielding, then the element is repaired or replaced. It is assumed that element failure has a probability of detection equal to unity during an inspection. It must be noted that repair or replacement of just a single cracked or failed element in a fuselage skin lap joint is practically impossible, since an element cannot be isolated from its neighboring elements. This becomes obvious when repairing a failed element with a mechanical doubler as the physical dimensions of the doubler will "repair" adjacent elements as well. For the sake of demonstrating the proposed methodology, this technicality is ignored (it is assumed that we have the capability of repairing or replacing a single element).

MONTE CARLO SIMULATION

A Monte Carlo simulation technique is used in this study to determine the reliability of the fuselage skin splice joint (referred to as the "system") as a function of the number of flights when the system is exposed to MSD. An inspection is performed every time the reliability level falls below a pre-specified minimum acceptable level. The following steps provide an algorithm for the implementation of the Monte Carlo simulation technique into a computer code:

- ❖ Declare constants and initialize the variables.

- ❖ Define by NE the total number of elements in the system and by SB the total number of simulations performed as part of the Monte Carlo scheme. Generate realizations for the TTCI for every element in every simulation (for a total of $NE \times SB$ realizations of the TTCI). Initialize the failure flag of each element to zero (=0 the element survives, =1 the element has failed) and compute the initial load on each element. Find the global minimum TTCI value of all the $NE \times SB$ realizations of the TTCI and start the time loop from that value. Set the system reliability equal to unity up to that global minimum TTCI (measured in flights) for all simulations.

- ❖ At each *time step TS* in the time loop commencing from the global minimum TTCI, do the following:

  - ➢ For each *simulation, i* (i = 1, 2, …., SB) do the following:

    - Go through all elements and find which ones have TTCI (*i*,*p*) (where *p* = 1, 2,…., NE) less than or equal to the current time *TS*. Compute the crack length of all elements that are cracked.

    - Check to see if the crack length in every cracked element exceeds a critical value (for net section yielding). Check also if the load on the uncracked elements causes yielding of those elements. If failure is confirmed in either case, set the flag of the failed element(s) to unity and increase the value of NFE (*i*) accordingly (NFE=Number of Failed Elements).

- Determine the position of the failed element(s) and redistribute its/their load share to the adjacent non-failed elements on either side. If the failed element is at one of the outer boundaries of the system, then its load share will be added to the nearest surviving element.

- Check to see if NFE (*i*) exceeds a critical number for net section yielding of the entire system. If such failure occurs, set the System Status (*i*) to unity.

- Record and save the current number of failed elements (NFE (*i*)), the element failure flag FLAG (*i*,*p*) where *p* = 1, 2,…., NE and the System Status (*i*).
(System Status =0 if the system survives; =1 if the system fails).

- Compute the probability of failure (POF) by counting how many systems have failed out of the total SB systems. Determine the reliability (R) for that time step (R = 1-POF). If the reliability is less than the minimum acceptable reliability level (R_{min}) do the following:

  ⇒ Reset the current time one step backwards and reset the system attributes to the values they held at that previous time step in order to perform the inspection when the reliability is at the minimum acceptable level.

  ⇒ Perform an inspection.

  ⇒ Separate the failed systems that failed before the inspection so they are not included upon resuming the time loop towards the next inspection. This implies that after every inspection there will be fewer surviving systems than before the inspection.

  For those systems that have not failed check the following:

  ⇒ If a cracked or uncracked element fails, reset its flag back to zero (because we are repairing/replacing all failed elements) and assign a new TTCI. For the cracked elements that have not failed so far, determine the probability of crack detection (POD) through equation (7) for the crack lengths at the time of inspection. In order to account for the possibility that an inspector may miss to detect the current cracked element, generate a random number uniformly distributed between zero and one. If the random number is less than or equal to the POD for that particular cracked element, that means that the crack was detected. Therefore, repair/replace the element, reset its FLAG (i,p) to zero and assign a new TTCI. If not detected, allow the crack to continue growing in the following time steps after the inspection. Since all failed elements are repaired/replaced, the FLAG (i,p) of all elements (repaired/replaced and non-repaired) in the system is reset to zero (meaning that they survive), the system assumes the original load distribution and the overall reliability returns back to unity.

The above algorithm has been implemented into a computer code and a parametric analysis was conducted to determine the sensitivity of some key variables. The nominal values of the various parameters appearing in the problem are tabulated in Table (1).

PARAMETERS	VALUE
Design Life	20,000 flights
Extended Life	30,000 flights
Minimum Level of Reliability, R_{min}	0.95 (95%)
Number of Elements in the System: NE	50
Nominal Stress Fluctuation: $\Delta\sigma$	20,000 psi
Tension Yielding Stress for 2024-T3 CLAD	44,000 psi
Parameter α in eq.(1)	4.0
Parameter β in eq.(1)	40,000 flights
Parameter C_1 in eq.(3)	1.333×10^{-20}
Parameter m_1 in eq.(3)	3.59
Parameter d in eq.(7)	1.57 inches
Parameter a_{min} in eq.(7)	0.050 inches
Parameter ε in eq.(7)	1.4
Initial Crack Length: a_0	0.050 inches
Time Step (TS)	100 flights
Number of simulations (SB)	10,000 simulations (systems)

Table (1) Values of Parameters Used in the Monte Carlo Simulation

SENSITIVITY ANALYSIS

A sensitivity analysis is performed to determine the effect of three key parameters on the predicted intervals of the non-periodic inspections. Specifically, parameter β in eq.(1), the nominal stress fluctuation $\Delta\sigma$, and the minimum acceptable reliability level were varied individually and in combination.

Variation of Parameter β

The values of 20,000, 30,000 and 50,000 flights are considered on top of the nominal value of 40,000 flights. In all cases, the applied stress fluctuation is 20,000 psi and the minimum acceptable reliability level is 0.95. Figure (3) displays results for β = 20,000 and 30,000 flights, while Figure (4) displays results for β = 40,000 and 50,000 flights. As expected, it is easily observed that as β increases, a smaller number of inspections is required. Table (2) displays the specific number of required inspections to maintain the minimum reliability above 0.95 as a function of parameter β.

β	**20,000 flights**	**30,000 flights**	**40,000 flights**	**50,000 flights**
# of inspections	15	3	1	0

Table (2) Required number of inspections as a function of parameter β.

Variation of Nominal Stress Fluctuation $\Delta\sigma$

The scheduling of inspections was found to be highly sensitive to small variations in the value of the applied nominal stress fluctuation $\Delta\sigma$ (when all other parameters were kept at their nominal values). Referring to Figure (5) for plots corresponding to $\Delta\sigma$ values of 20, 21 and 22 ksi and Figure (6) for $\Delta\sigma$ values of 23 and 24 ksi, it is easily observed that the effect of $\Delta\sigma$ on the number of inspections is significant. Table (3) displays the specific number of required inspections to maintain the minimum reliability above 0.95 as a function of $\Delta\sigma$.

$\Delta\sigma$	**20 ksi**	**21 ksi**	**22 ksi**	**23 ksi**	**24 ksi**
# of inspections	1	2	3	5	59

Table (3) Required number of inspections as a function of $\Delta\sigma$.

Variation of Minimum Acceptable Reliability Level

The change in the requirement for minimum reliability level from 95% to 99.5% is examined next, keeping all other parameters at their nominal values. Refer to Figure (7) and Table (4) for the required non-periodic inspections as a function of the minimum acceptable reliability level.

R_{min}	**0.95**	**0.995**
# of inspections	1	6

Table (4) Required number of inspections as a function of R_{min}.

Combinatorial Variation of Parameters

Two sets of the three key parameters (β, $\Delta\sigma$ and R_{min}) were selected to demonstrate their combinatorial effect on the number of inspections. Refer to Figure (8) and Table (5) for results.

	$\Delta\sigma$ = 24 ksi β = 50,000 flights R_{min} = 0.95	**$\Delta\sigma$ = 20 ksi β = 50,000 flights R_{min} = 0.995**
# of inspections	34	2

Table (5) Required number of inspections from combinatorial variation of parameters.

CONCLUSIONS

The aforementioned methodology has demonstrated its ability to determine non-periodic inspections of a fuselage skin lap joint for a prespecified minimum reliability level using Monte Carlo simulations. Three key parameters, namely parameter β of the time to crack initiation, the nominal applied stress fluctuation $\Delta\sigma$ and the minimum acceptable reliability level were varied individually and in combination to determine their effect on the required number of non-periodic inspections in order to maintain the minimum reliability level. A general conclusion in all cases is that the frequency of inspections increases as the time increases. This is a report of ongoing research and as such its conclusions could be somehow modified in the future upon completion.

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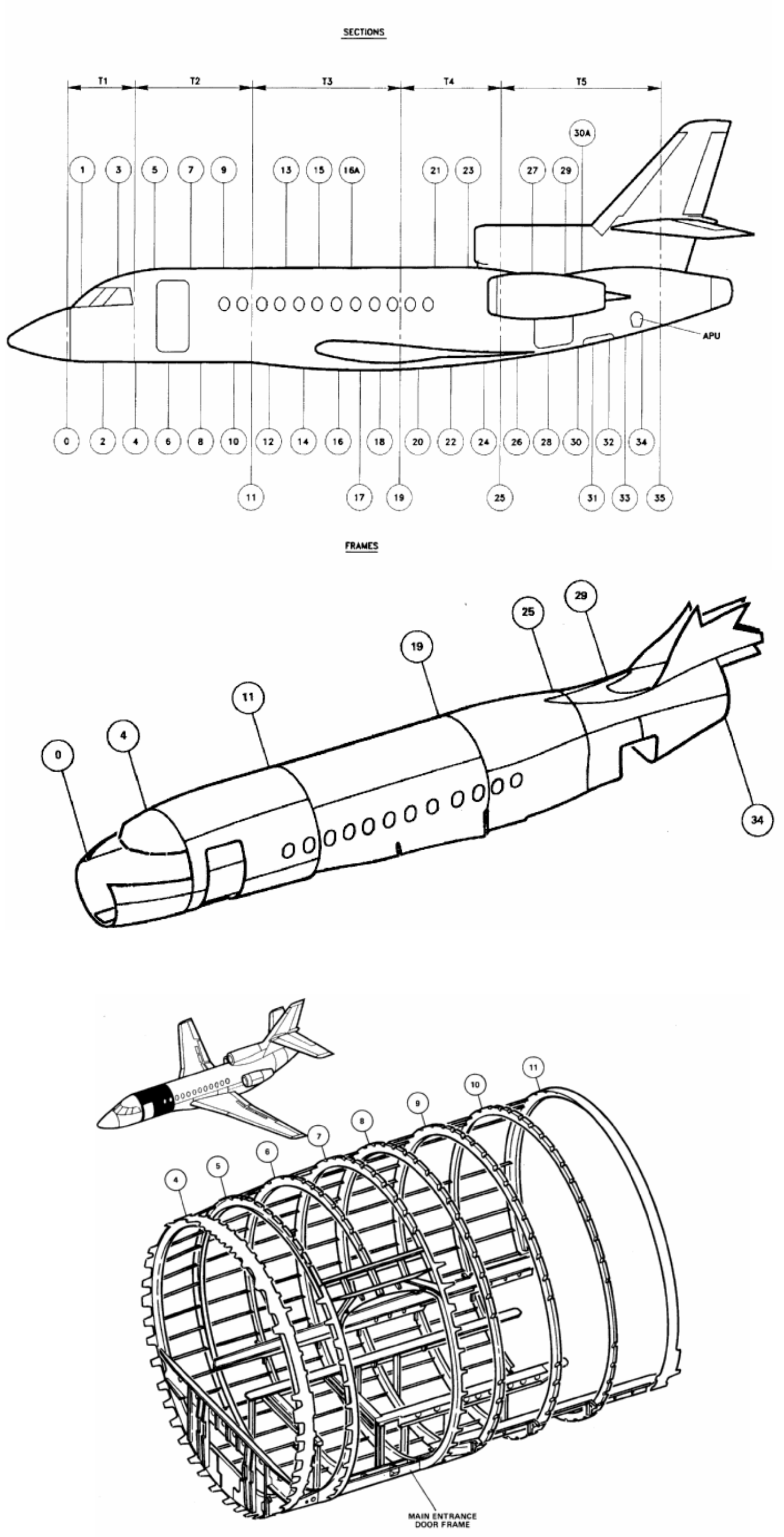
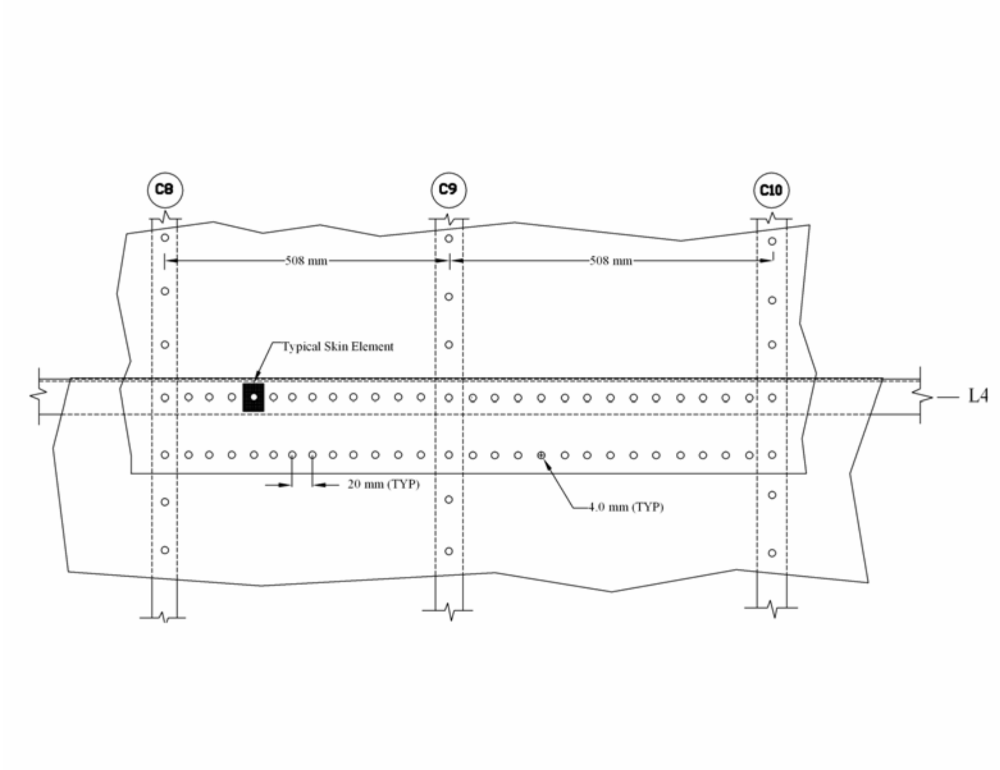


Figure (1) Frames and Skin Splice Joints for Dassault F900



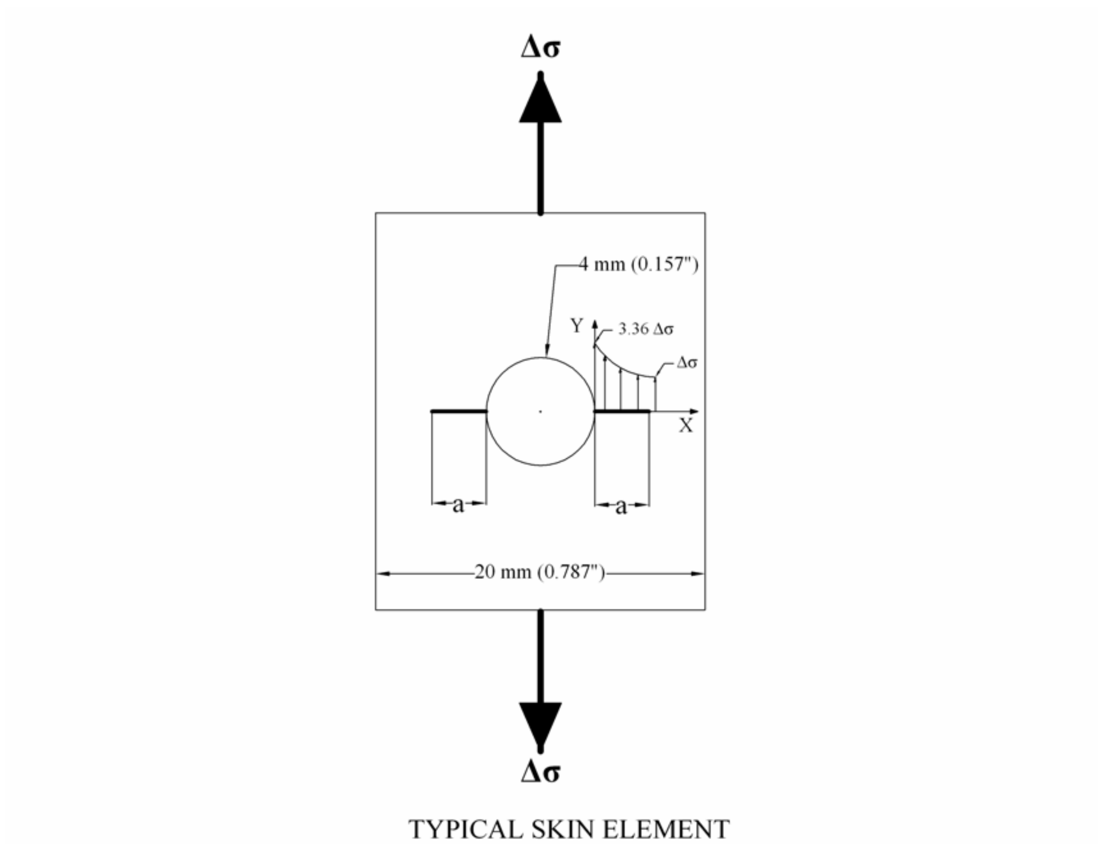
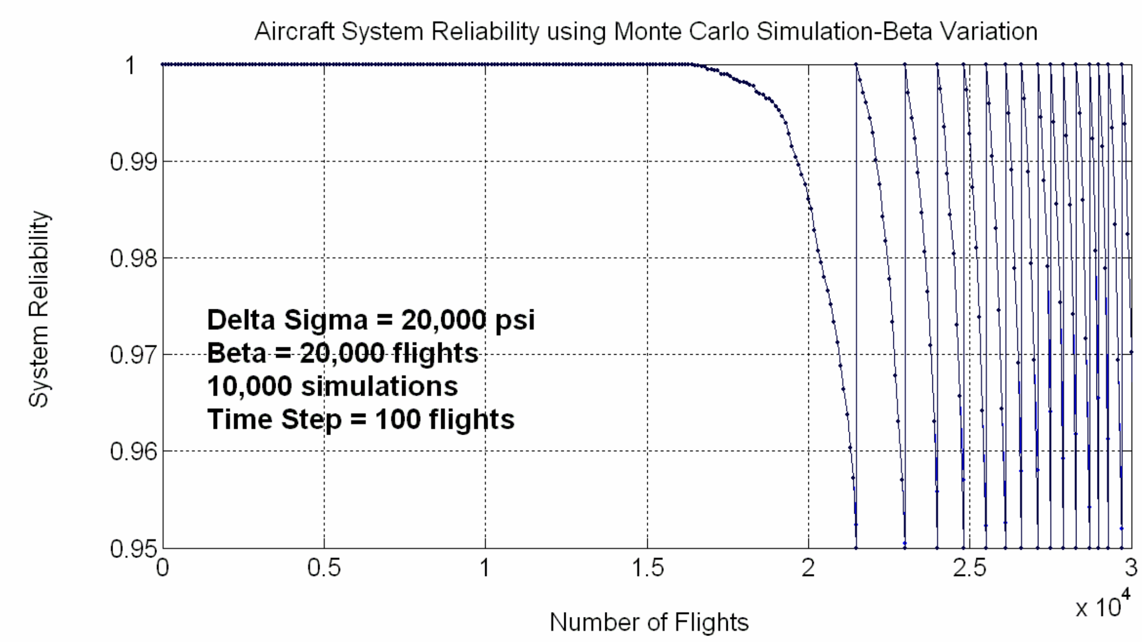


Figure (2) System and Element Configurations.



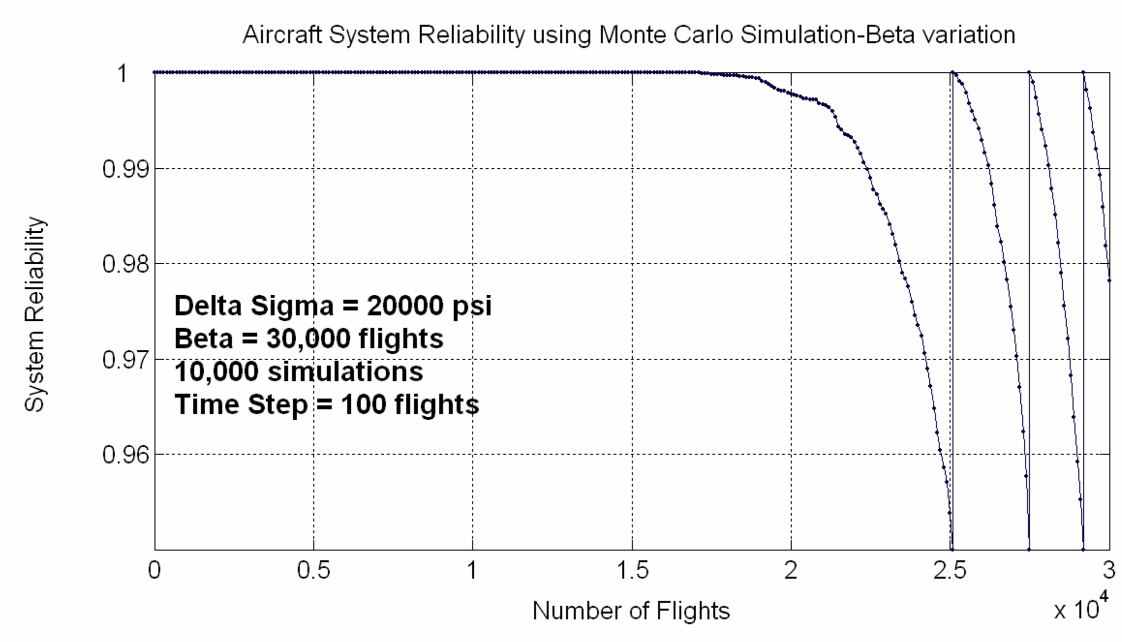
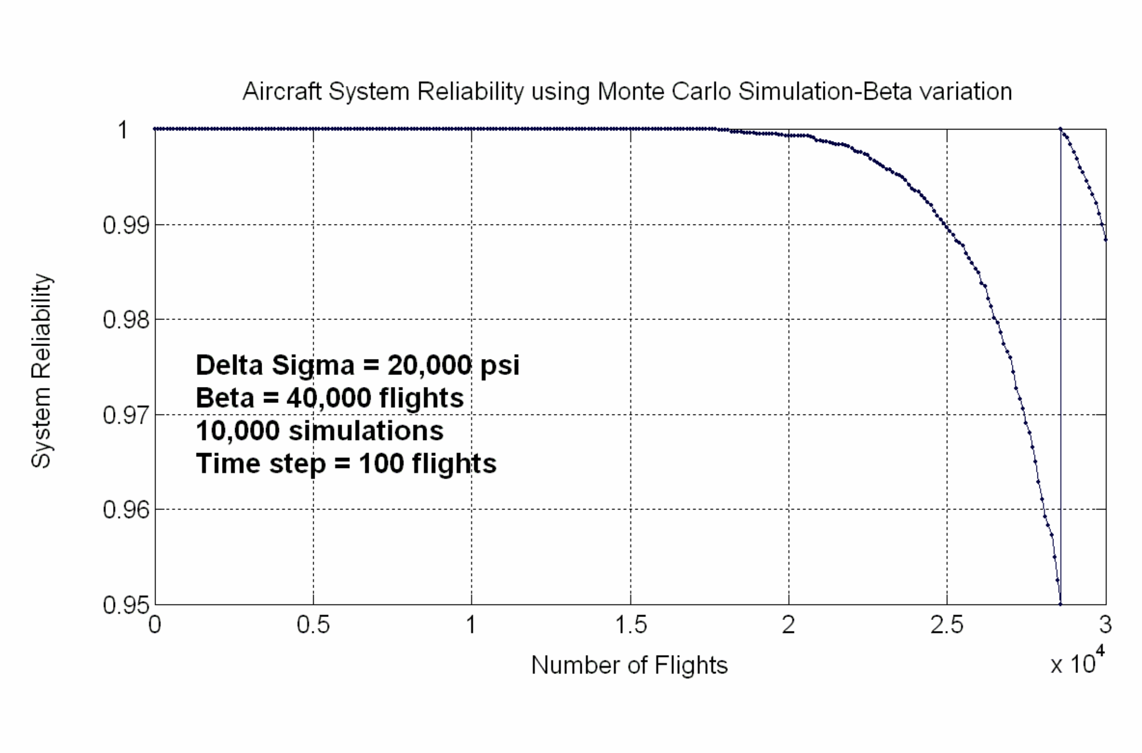


Figure (3) Parameter Beta Variation (β = 20,000 & 30,000 flights)



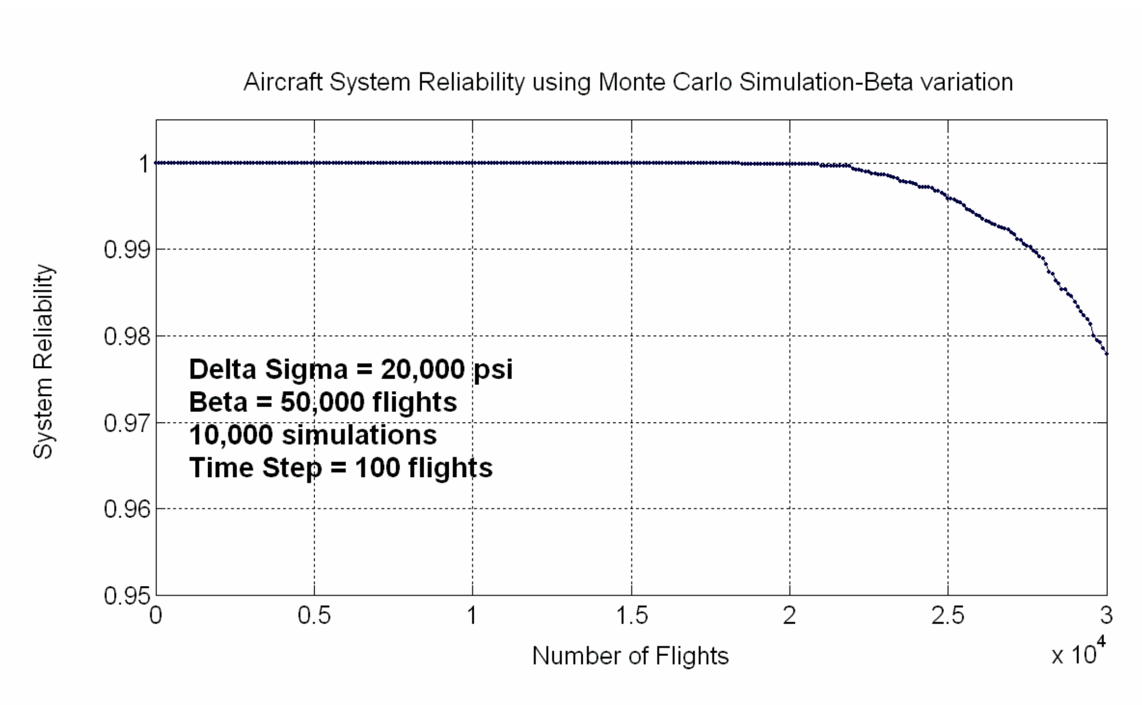
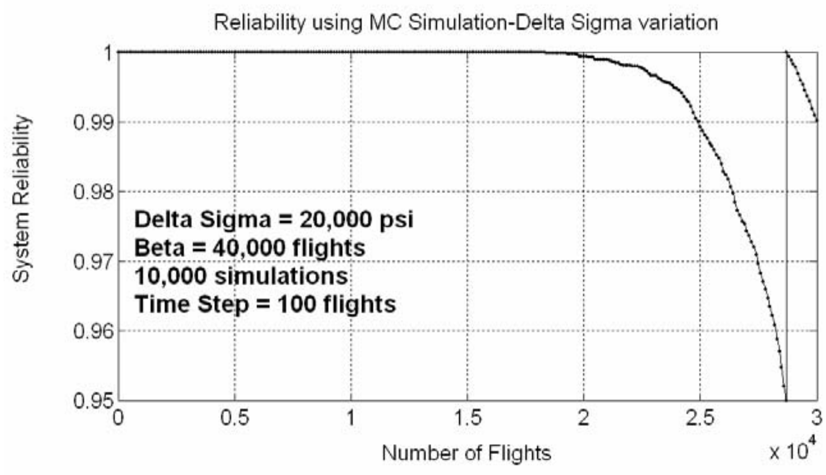
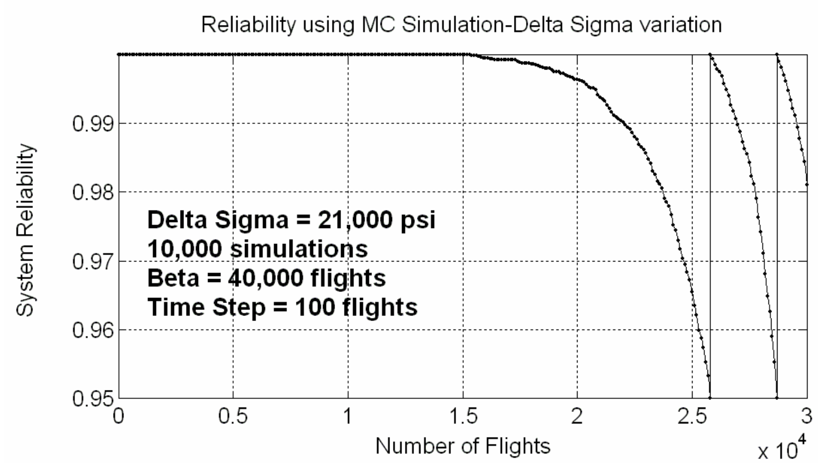


Figure (4) Parameter Beta Variation (β = 40,000 & 50,000 flights)





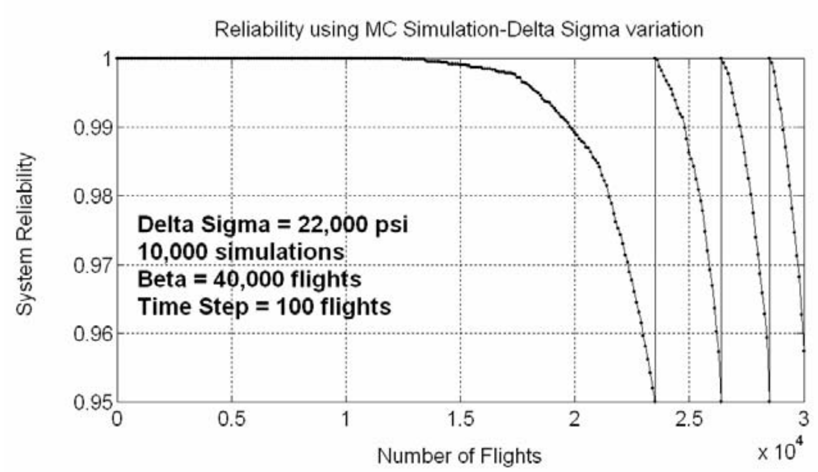
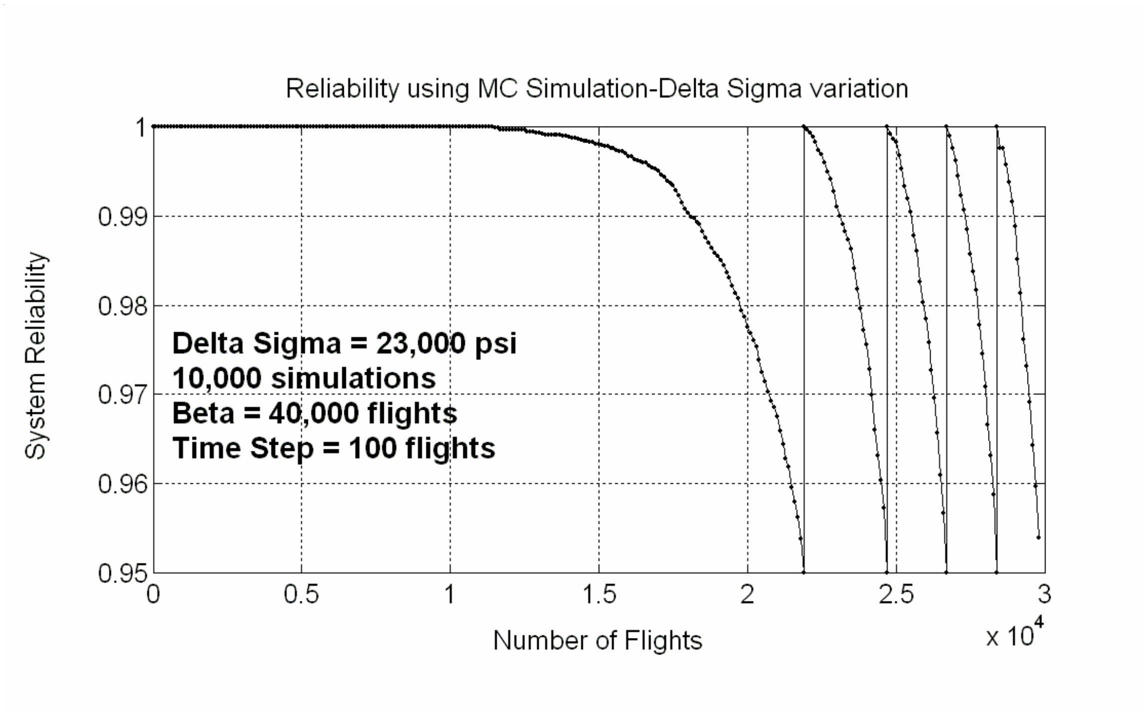


Figure (5) Parameter $\Delta\sigma$ Variation ($\Delta\sigma$ = 20, 21 and 22 ksi)



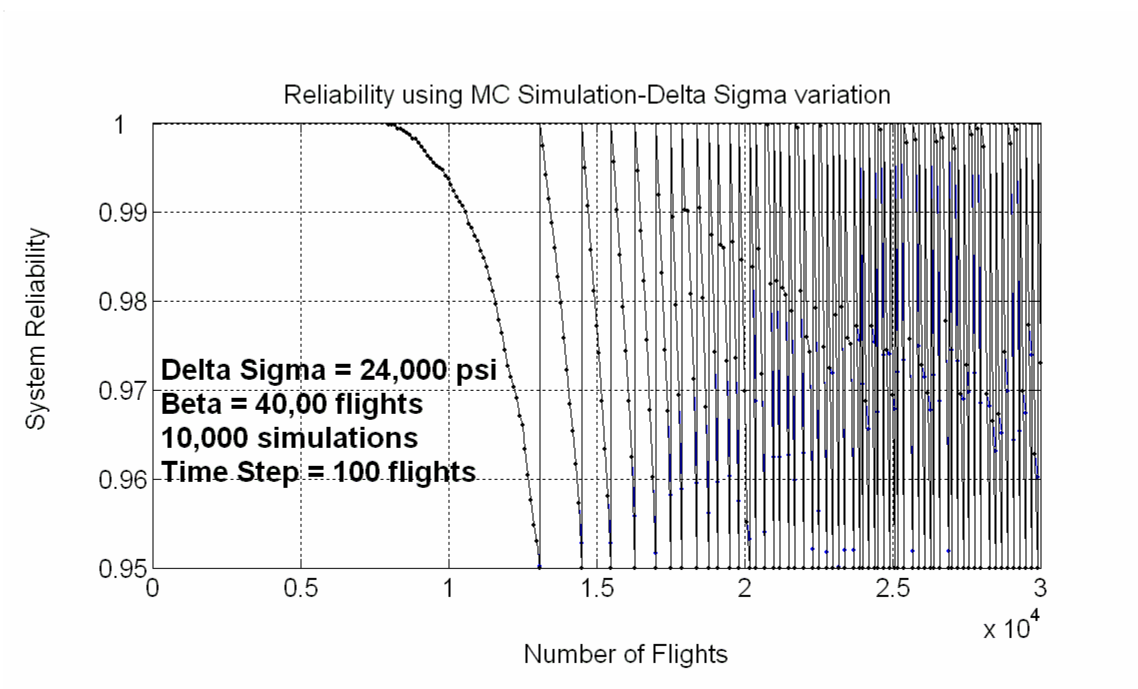
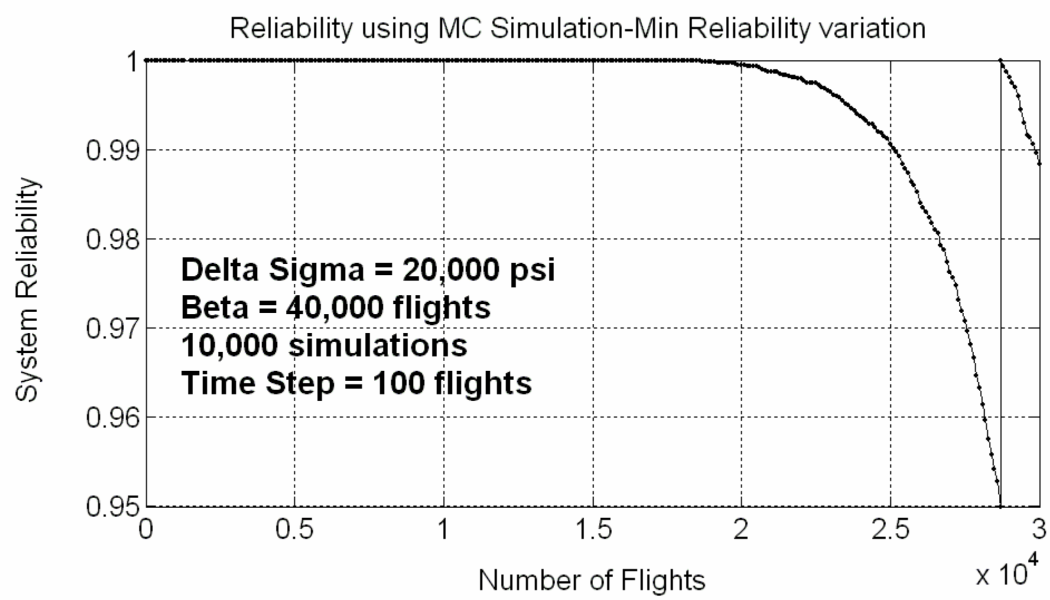


Figure (6) Parameter Δσ Variation (Δσ = 23 and 24 ksi)



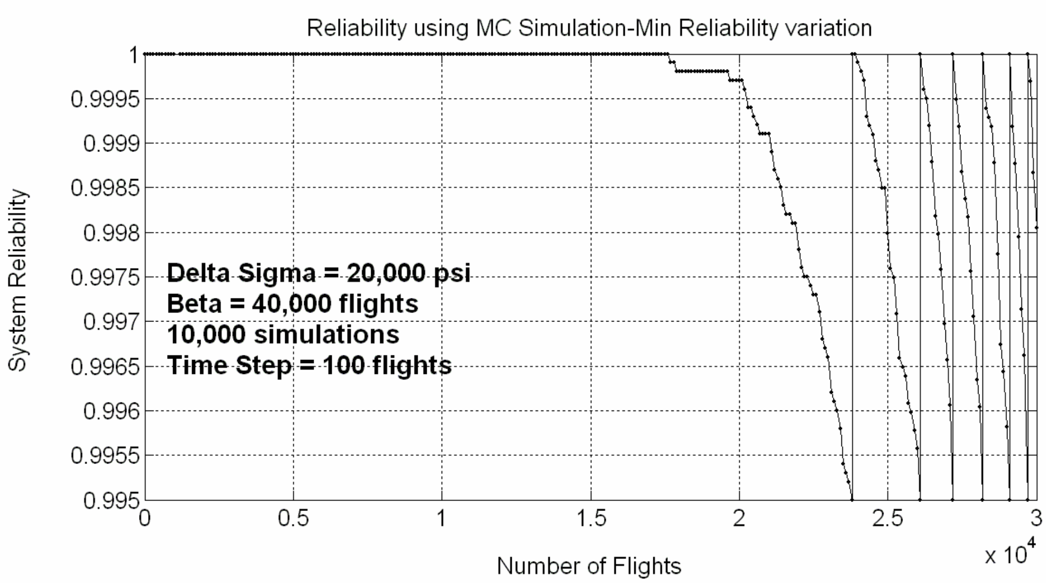
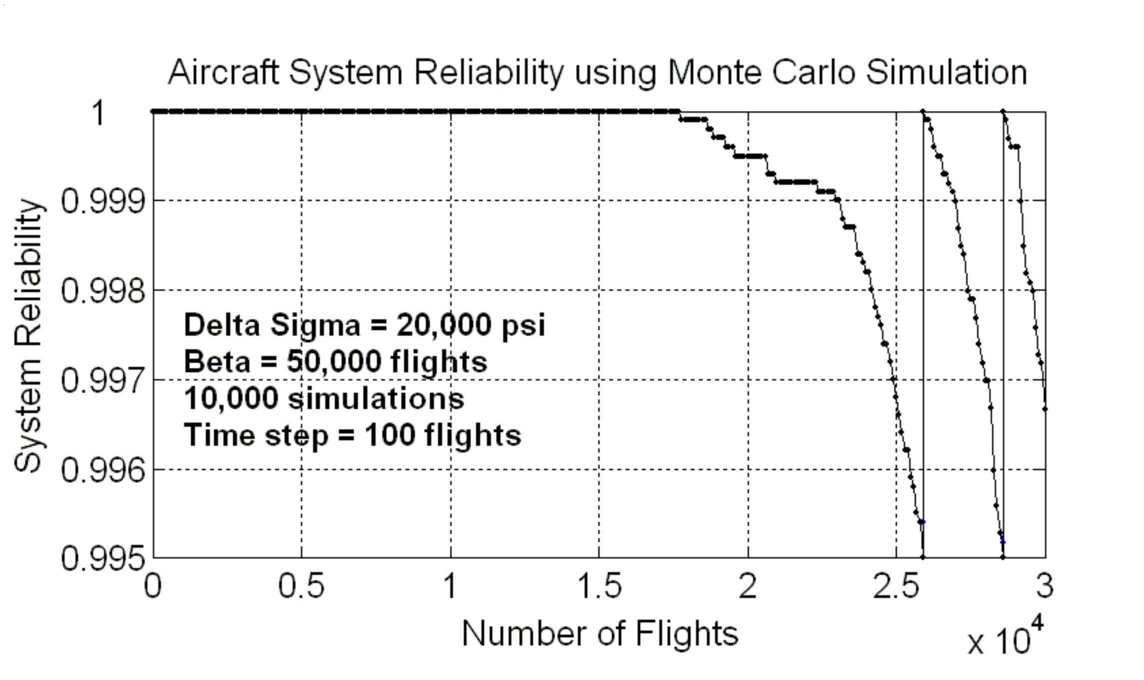


Figure (7) Minimum Reliability Variation



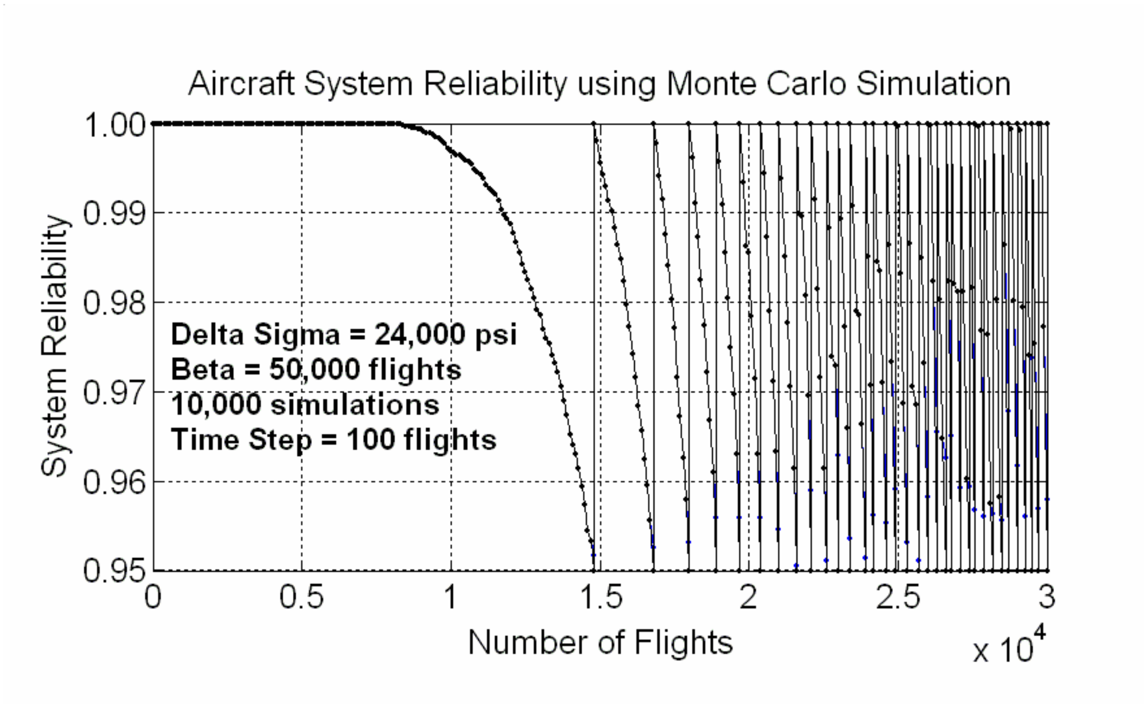


Figure (8) Combinatorial Variation of Parameters