BOEING

APOLLO TIE
WASHINGTON, D. C.

DOCUMENT NO. D2-117083-1

TITLE STATISTICAL TRENDS ANALYSIS OF VIBRATION INDUCED SPACECRAFT FAILURES

MODEL NO. APOLLO-TIE CONTRACT NO. NASw-1650

16 DECEMBER 1968

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ISSUE NO. ISSUED TO

ABSTRACT

This document incorporates the results of a survey on spacecraft vibration induced failures experienced during laboratory testing and the launch and boost portion of the mission. Twenty-eight different types of spacecraft, involving 83 flight models, comprised the data sample. A statistical and reliability analysis, coupled with the failure data, was made to define the spacecraft vibration flight and test failure rates according to the complexity of the spacecraft, the severity of the vibration environment, and the intensity of the qualification and acceptance vibration test programs.

Key Words

Vibration Failures
Spacecraft Refurbishment
Failure Rate

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PREFACE

This document was prepared for the Apollo Program Office, National Aeronautics and Space Administration, Washington, D. C., by the Boeing Company in partial fulfillment of the requirements of Contract NASw-1650, Technical Direction Serial No. 3, Item 3.

The following personnel contributed to this survey by providing a considerable amount of background information and data relating to spacecraft environmental testing.

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ABBREVIATIONS

CPS	Frequency in cycles per second
G	Acceleration in gravitational units
Grms	Root mean squared acceleration
PSD	Power spectral density
MAX q	Maximum dynamic pressure
S/C	Spacecraft
λ	Failure Rate
N	Number of failures

DEFINITIONS

ACCEPTANCE TEST - A test to detect workmanship deficiencies in a component, subsystem or system which is destined for use in service. The test environment may or may not simulate the maximum expected service environment, but the test level is generally less than the qualification test level.

COMPONENT - An integral package such as a camera, a valve, a battery or a transmitter.

QUALIFICATION TEST - A test on a component, subsystem or system to demonstrate design capability to withstand a critical service environment. The test environment is usually a conservative simulation of the maximum expected service environment in order to establish design margins.

SUBSYSTEM - A group of components which is part of a larger system. An example of a subsystem is the Gemini fuel cell module consisting of two fuel cell sections, a hydrogen tank, and oxygen tank, pressure regulators, valves and associated tubing and wiring.

SYSTEM - A complete or major portion of a spacecraft

TEST PATTERN - The vibration levels and duration which together comprise an acceptance or qualification vibration test.

1.0 Introduction

This survey was performed to support a concurrent survey involved with an in-depth study of vibration testing of four major spacecraft programs: Mariner, Gemini, Lunar Orbitor and Surveyor. (Reference 1)

1.1 Scope

During this survey data was obtained on spacecraft failures experienced in a vibration environment. This included the qualification, acceptance and flight environments. The data was coupled with basic reliability theory to derive, with the support of the collected empirical data, the failure rate (λ) of a spacecraft subjected to a vibration environment. This document presents the results of a statistical and reliability analysis capable of predicting spacecraft vibration flight and test failure tests according to the complexity of the S/C, the severity of the vibration environment and the intensity of the qualification and acceptance vibration test program.

1.2 Summary

This document presents the results of a survey of spacecraft vibration induced failures. The source data originated from 28 different S/C programs involving 83 flight models.

A reliability analysis was performed using reported vibration component reliability stress factors and an average S/C MTBF found from 72 flight models. The theoretical failure rate was shown to be proportional to $g^N t$, where g is the vibration acceleration level N is a constant and t is time: Analytically, N is shown to vary between 1.76 and 2.17.

When the spacecraft failure data is normalized to either piece-part count or spacecraft weight, it is shown to be proportional to $g^2 t$.

The effect of qualification and acceptance vibration testing on the reliability of a flight model has been calculated. The probability of flight model failure increases with decreasing values of $g^2 t$ in qualification testing and increasing values in acceptance testing. Refurbished qualification test models were also included in these analytical treatments.

This document concludes with a review of the application of reliability theory and the data acquired during this survey to show how the optimum S/C vibration environmental test program can be chosen commensurate with the required reliability.

2.0 BACKGROUND

Vibration of a spacecraft (S/C) or of S/C equipment may cause failures by either a fatigue process or by a resonant induced overload. Generally, a combination of these two failure modes is present in a vibration environment. The failure mode most damaging is the resonant one occurring when the natural frequency of a system or piece part, coincides with the exciting vibration frequency, when this occurs dynamic amplification factors of 10 to 100 are not uncommon. The exact amplification factor is dependent on the damping within the system since the strain hysteresis is the major process in which the applied energy can be dissipated.

Excitation frequencies below 30 cycles per second will generally only excite structural systems, the larger more massive structures such as the Saturn V, have fundamental bending frequencies at about 2 to 5 cps. Excitation frequencies between 30 and 200 cps will generally excite large piece part components such as transformers, wire bundles and panels. Small parts such as transistors, diodes, and electronic tube filaments are most sensitive to vibration frequencies ranging between 200 and 1500 cps.

It has become standard procedure to subject a prototype spacecraft to a severe vibration environmental test before finalizing the S/C design, and after fabrication, each flight unit to a low level vibration test prior to launching. The purpose of these tests, called Qualification and Acceptance Vibration Tests respectfully, is two-fold: The qualification test, where the spacecraft is subjected to a level 1 1/2 to 2 times the flight vibration level (Reference 1), is performed to find any potential failure modes caused by highly amplified dynamic loads, or by interreaction of equipment within the spacecraft. The purpose of the acceptance test, performed generally at the maximum flight vibration level, is to disclose failures due to defects in materials and/or workmanship.

2.1 The S/C Flight Vibration Environment (Reference 2)

The spacecraft flight vibration environment is generated by 3 radically different phenomena occurring during the launch and boost phase of the mission profile: they are acoustical, aerodynamic and mechanical. The spacecraft receives vibration through both mechanical and acoustical paths. Acoustically induced vibrations are generated by the booster rocket engine noise. The aerodynamic vibrations stem from boundary layer pressure fluctuations, flow separation, and oscillating shock waves. Mechanically induced vibrations result from rocket engine thrust variation, resonant burning of solid propellant rockets and/or dynamic loads generated by rotating equipment.

2.1.1 Acoustically Induced Vibration

Acoustically induced vibrations occur mainly in the frequency spectrum above 100 cps. These vibrations are characteristically broad band random in nature and extend up to several thousand cycles per second. Generally speaking, the S/C vibration level above 100 cps is directly proportional to the acoustic sound pressure level. (See Figures 2.2 and 2.3). The frequency spectrum peak is a function of the exhaust nozzle diameter and the jet exit velocity.

Total acoustic power radiated is between .5 and 1 per cent of the mechanical exhaust stream power. Thor, Atlas and Titan class boosters generate on the order of 107 watts acoustic power. Typical sound pressure levels at the S/C are 140 db, a few seconds after liftoff these levels drop by 15 to 20 db. The acoustically induced vibration data is presented in terms of power spectral density g^2/cps versus frequency since it is predominantly random. Figure 2.2 presents a typical launch and boost acoustic sound pressure level time history.

2.1.2 Lift Off

The vibration levels are severe at lift off because of ground reflected booster rocket engine noise. This phenomena is also intensified because of the effect of the flame deflectors, deflecting the exhaust 90 degrees. Since the noise sources are distributed around the exhaust stream, they are closer to the spacecraft and result in higher vibrations at lift-off than when the vehicle is airborne.

2.1.3 Aerodynamic Induced Vibrations

The S/C vibration levels decrease rapidly just after liftoff. Then, as the vehicle gains speed, aerodynamic noise becomes the predominant source of S/C vibration. In general, the vibration level increases with time as a function of the free stream dynamic pressure, q. (Reference 3, 4 and 5). The lower shaded area of Figure 2.3 shows that the S/C aerodynamic induced vibrations are maximum when q is maximum and then goes to zero as the vehicle leaves the earth's atmosphere.

The S/C vibration level in the transonic-maximum q region of the vehicle trajectory is highly dependent on the S/C shape or shroud configuration. If the payload is of a streamlined, aerodynamically "clean" shape, the vibration levels are generally equal to or less than those experienced at launch. However, if the payload or S/C shroud does not have a smooth configuration, extremely high vibration levels will occur during the transonic and max q periods of flight. This is particularly true of the bulbous payload shapes, and those shpaes consisting of blunted cone angles, (References 6 and 7). Vibration levels on the order of 5 times the launch levels have been recorded with these shroud shapes. Sometimes separate peaks show up at the Mach 1 and maximum q periods of flight; however, the usual vibration versus time curve shows a high level at lift-off and a second peak at max q.

2.1.4 Mechanically Induced Periodic Vibration

Spacecraft vibrations generated by mechanical means occur mainly at low frequencies. They are cuased either by periodic thrust perturbations and/or by dynamic loads generated by rotating equipment. The most significant low frequency mechanical vibrations (25 cps) involve the vehicle modal response coupled to a feedback interaction between the vehicle propulsion system and the structural system (Pogo). This affect occurs along the vehicle longitudinal axis and acceleration levels up to 3 g's have been measured.

A much more severe type of mechanical vibration is caused by burning irregularities, often inducing high frequency oscillations due to the acoustical characteristics of the combustion chamber and the fuel's burning properties. Several modes may occur simultaneously with frequencies ranging from several hundred to several thousand cycles per second. The X-248 solid rocket (a small 3rd stage rocket) was a primary offender in this category, producing extremely high S/C vibration levels (18g's) or combinations of several sinusoids (References 8 and 9). Current upper stage solid rockets (like X-258) produce a much less severe vibration environment because of design improvements (Reference 10).

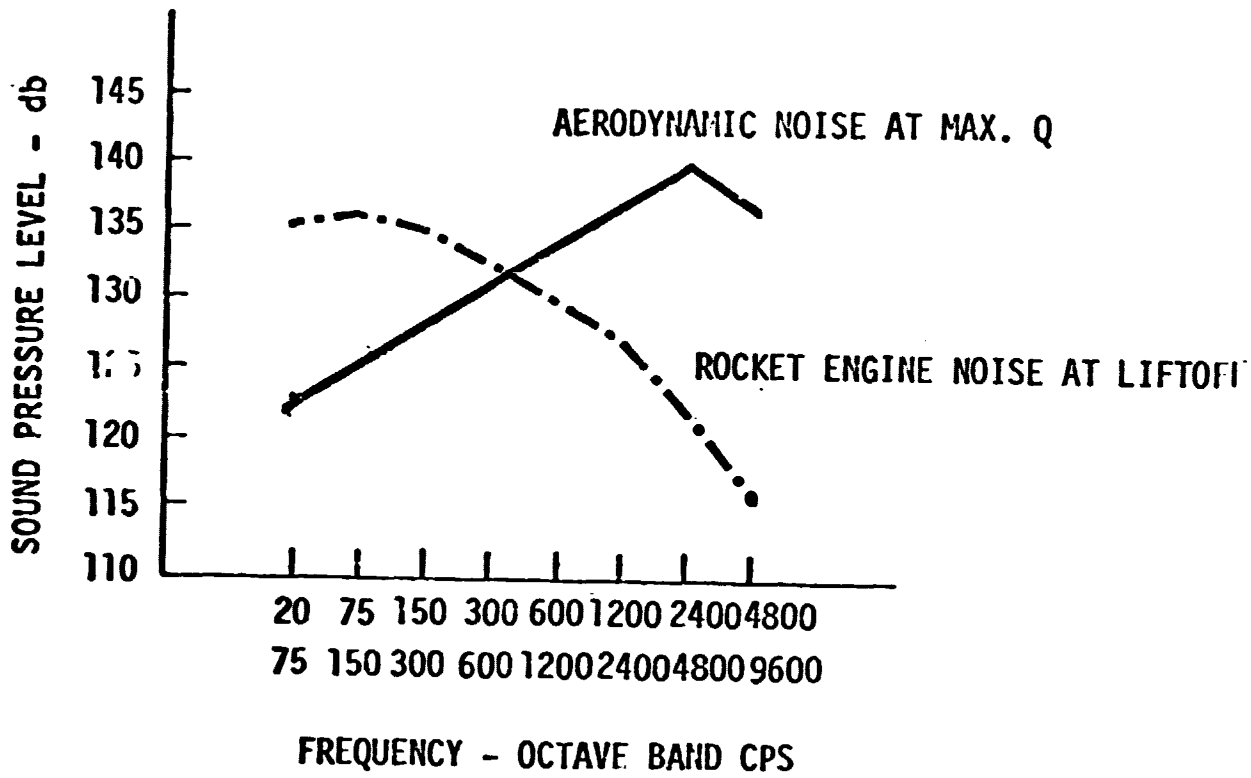


FIG. 2.1 TYPICAL OCTAVE SPECTRA EXTERNAL TO SPACECRAFT - SHROUD

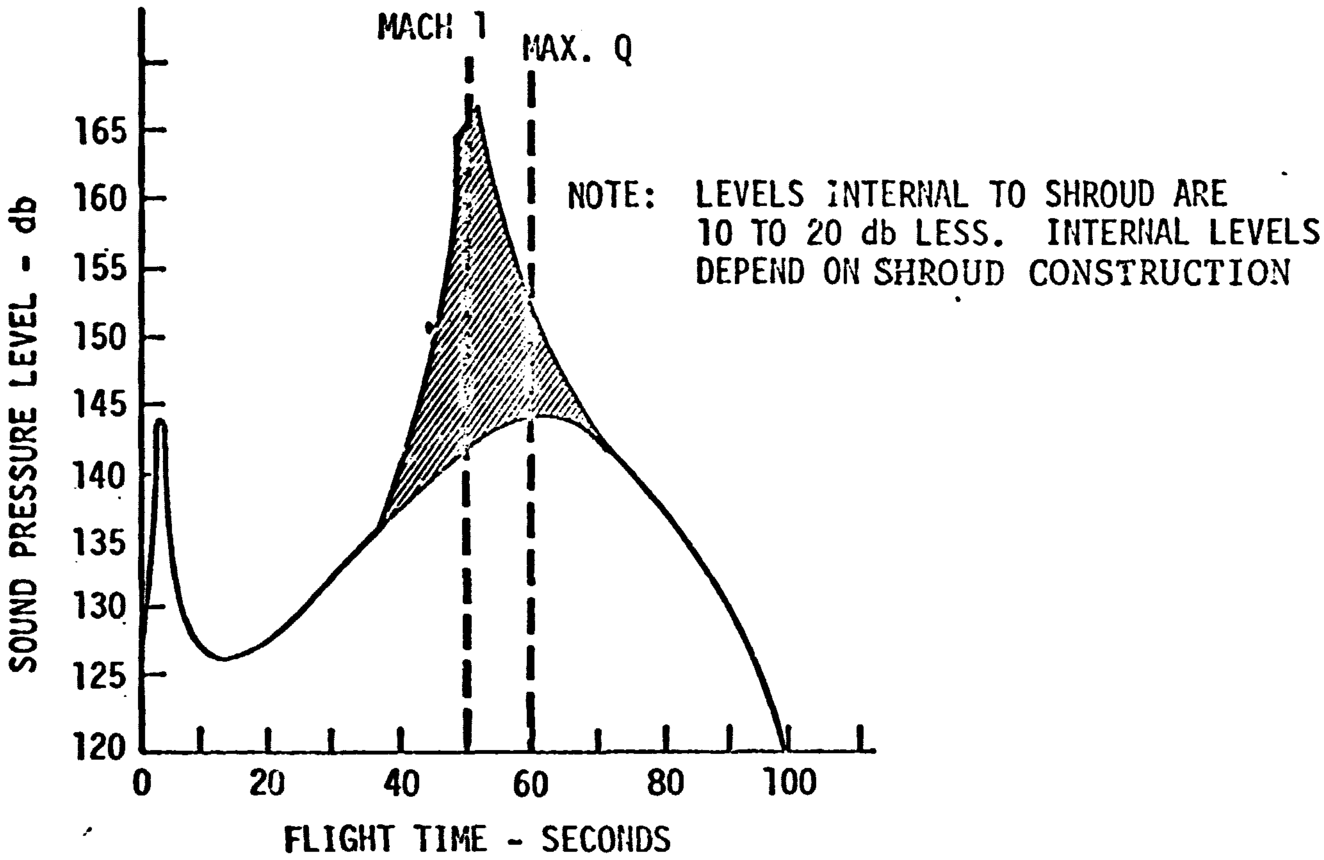


FIG. 2.2 TYPICAL TIME HISTORY OF EXTERNAL ACOUSTIC LEVELS

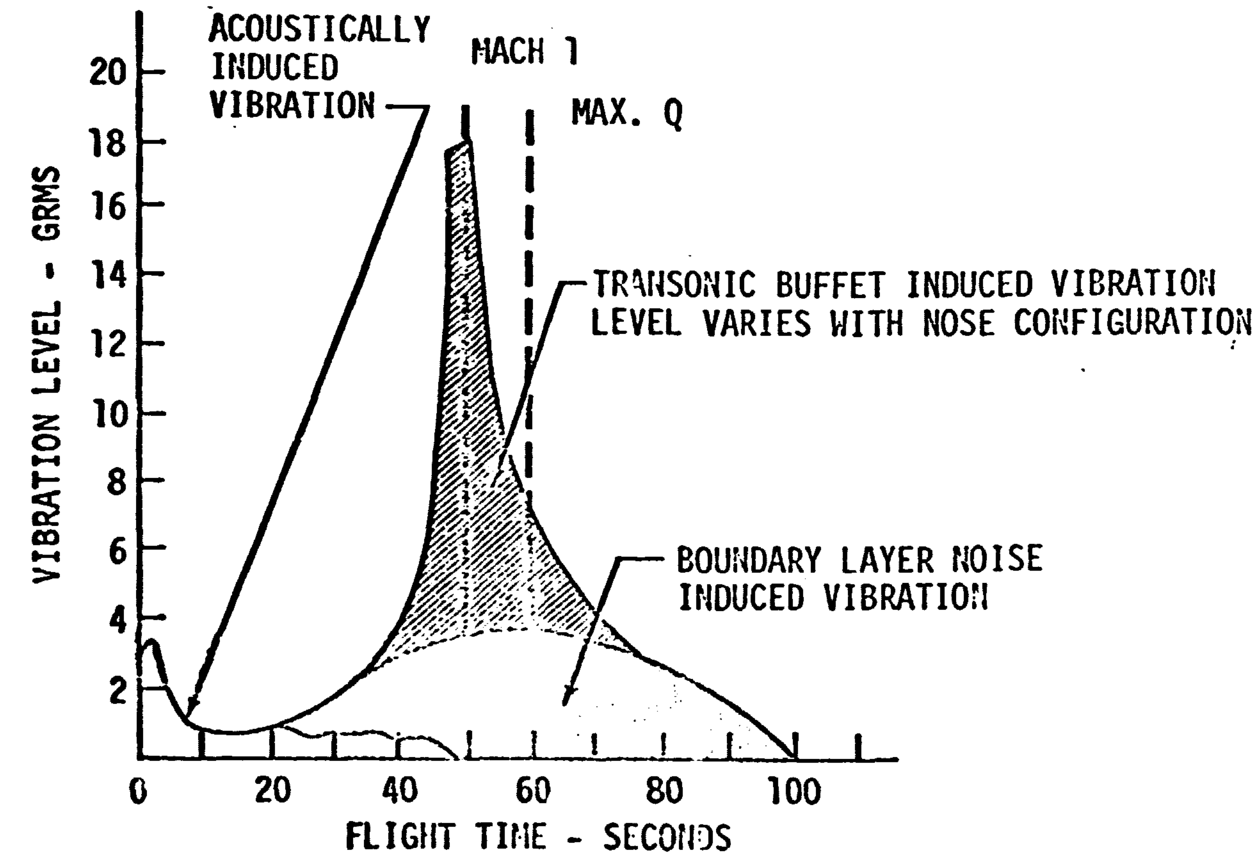


FIG. 2.3 TYPICAL TIME HISTORY OF SPACECRAFT VIBRATION LEVELS

3.0 ENVIRONMENTAL TEST PROCEDURES

The primary purpose of an environmental test is to duplicate within the laboratory the environment to be encountered during the mission in order to detect any potential failure modes or adverse environmental effects on the S/C.

Because of test equipment limitations this cannot be accomplished exactly, i.e., S/C vibration testing is performed along one axis at a time, whereas in flight it occurs simultaneously along all axis. The steady-state longitudinal accelerations, pressure decay and thermal changes are all performed during separate tests, thereby leaving a possible environmental inter-action failure mode undiscovered. A safety factor is incorporated within the test levels and duration (test pattern) in order to account for the possibility of environmental interaction and to increase confidence in the ability of the S/C to successfully withstand the expected environments.

Random vibration is used to simulate S/C vibration induced by aerodynamic and acoustic effects, sinusoidal vibration is used to simulate the vibrations generated by engine burning resonances, transportation and shipping, Pogo, and by rotating equipment located within the launch vehicle.

The 5 primary objectives of S/C environmental testing are:

1. Verification that new or improved designs meet performance requirements and have a satisfactory life expectancy.

2. Verification that samples of previously tested hardware are suitable in a new application.

3. Elimination of defects in design, material or workmanship.

4. Discovery of unexpected interactions between subassemblies when the total system is exposed to environmental stresses.

5. Generation of test data that will serve as a guide in evaluating new designs and assessing their reliability.

3.1 Design Qualification (Prototype) Tests

The qualification or prototype unit, is almost invariably the first model in which the subassemblies appear together in the near-final S/C configuration. Tests of this prototype system are directed toward detecting major design weakness. Overtesting is required since it is extremely difficult to achieve all the desired objectives with just one sample; however, because of weight limitations, S/C designs cannot be expected to have an excessive margin of strength.

In the face of these problems, prototype test levels are usually established at what might be considered the 99 percent probability level - that is, there should be no more than one chance in a hundred that a flight unit will experience an environment more severe than that employed during prototype testing.

3.2 Flight Unit Testing

Because only one prototype has been qualified, virtually no information is available on the variation to be expected between units of the same design. Flight unit testing is intended, then, to discover quality control problems, and/or defects in material or workmanship. One of the most difficult problems associated with acceptance tests is determining the duration of the tests in order for these tests to both give reasonable assurance that the flight unit can survive the launch and boost environment without seriously weakening the unit itself. The acceptance test levels are generally set at the 95th percent probability level and, as will be shown later, this is by far the most severe environment that the flight S/C will encounter.

3.3 Test Levels

3.3.1 Acoustic and Aerodynamic Vibration Levels

The random vibration used to simulate the acoustic and aerodynamically induced S/C vibrations is generally a composite function obtained by superimposing the maximum flight random vibration spectrum recorded at about maximum q and lift-off (see Figure 3.1). The envelope of these two curves is assumed to represent the 95th percentile flight vibration environment and is used for the acceptance test levels. A factor of 1.5 is used to increase the acceptance test levels to the qualification levels. The qualification or the 99th percent level is arrived at by assuming a normal gaussian amplitude distribution in which the 95th percent point is equal to 1.65σ and therefore, the 99th percent level is equal to $1.5 \times 1.65\sigma$ or 2.47σ, the 99.3 percent point.

Because of the scarcity of available data, this procedure is approximate and is not backed up in statistical terms with carefully computed standard deviations and levels of confidence. Consequently, these test levels may be increased by program direction in order to achieve a higher confidence level in the ability of the S/C to successfully withstand the imposed vibration environments.

3.3.2 Mechanical Vibrations

Mechanical vibrations are simulated by sinusoidal frequency sweep tests. During these tests the acceleration level is set at a factor of about 1.3 times the maximum expected level, for the 95th percentile level, while the frequency is slowly swept from 20 to 2000 cps. Engine resonant burn simulation is made by slowly sweeping the frequency about the resonant combustion frequency only.

3.4 Test Duration

While the test levels derived in the previous section, will differ from program to program, in general the testing philosophy used will not change the test levels by more than 50%; however, this is not true of the test duration. Some of the older S/C programs, influenced by aircraft vibration test plans, specified tests lasting up to one hour per axis; whereas, some of the most recent S/C test plans specify test durations equal to the powered flight time for the acceptance tests and twice this for the qualification test plan.

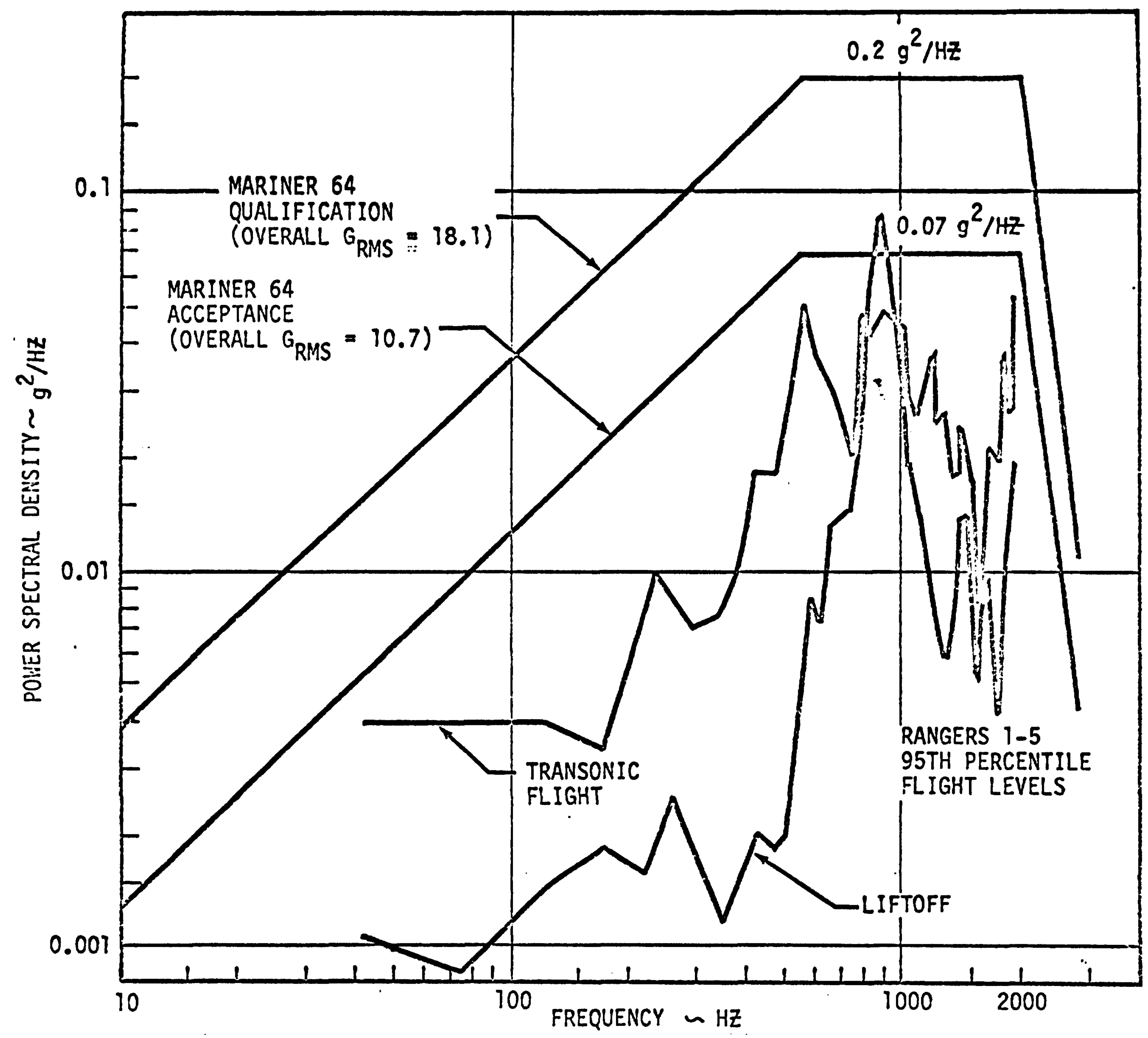


FIG. 3.1 DERIVATION OF SYSTEM VIBRATION LEVELS, MARINER 64 (REF 42)

4.0 SURVEY PROCEDURE

The objective of this survey was to determine if statistical and/or reliability theories could be used to predict spacecraft failures induced by a vibration environment. In order to accomplish this task, the S/C failures experienced during flight, acceptance and qualification tests were tabulated for a reasonably large data sample - 28 different S/C programs comprising 83 flight spacecraft. This study was oriented toward the complete S/C or system level, rather than the component level, because documented records are more readily available for complete S/C.

The failure data was collected from Environmental Test Reports, Spacecraft Final Development Reports, and from conversations with the Test and Evaluation Group at Goddard Space Flight Center (References 17 to 26).

The data sample used for this study is listed in Table 1.

- - - - -

Alouette	Beacon		Explorer XVII
Syncom	Telstar	Ranger	Explorer XVIII
OSO	Lunar Orbitor	Pioneer VII	Explorer XXV
Ariel I	Gemini	Pioneer VI	Explorer XXVIII
Ariel II	Mercury	Surveyor	Explorer XXI
Mariner Mars	Tiros	Explorer X	
Mariner Venus	OAO	Explorer XII	
Relay	Nimbus	Explorer XIV	

TABLE I - SPACECRAFT PROGRAMS SURVEYED FOR S/C VIBRATION FAILURE DATA

This sample encompasses S/C typical of the manned and unmanned designs dating between 1961 and 1967. The vibration environments associated with the acceptance, qualification and flight conditions were also collected, where documented, in order to evaluate the effect of environmental severity versus failures.

The vibration levels used within this study were those specified for the base of the spacecraft or at the attachment point between the spacecraft and launch vehicle. This was done primarily to avoid the immense amount of work that would be required to trace down and account for the transmissibility throughout each spacecraft. Furthermore, the small amount of flight data available is generally recorded only at the spacecraft attachment point.

The S/C piece part count and weight was required in order to normalize the S/C failure data so that a large complex S/C with many components, having a large number of failures occurring during the environmental test program or flight, could be rationally compared to a smaller S/C with its failure history occuring during development or flight.

A serious problem was acquiring "piece-part" count data. This was found for 23 different S/C. The remaining S/C in the data sample, had their part count estimated from Figure 4.1 (weight versus part count) using the same type of launch vehicle. Since this problem could be encountered while applying the failure rate data to other S/C, the results of this survey have been plotted both against piece part count and S/C weight.

4.1 Assumptions

The following assumptions were made for the reliability and statistical analysis presented in Sections 5 through 8. These assumptions were made primarily to simplify the study and to avoid the necessity of treating each separate S/C as a totally different entity. Eventually, when a significantly greater number of S/C have been designed, it should be possible to subdivide, at least, the different types into a more similar category, such as, Manned versus Unmanned S/c, and S/C weighting less than 400#.

4.1.1 S/C Are Similar

This assumption was made because spacecraft in general have the same objectives, that is, to survive the launch and boost environment, collect data and transmit it back to earth. In order to achieve these functions each satellite will contain the following major systems: power, guidance and control, electrical, logic, data acquisition, and telemetry.

4.1.2 S/C Failures

All failures occurring during a vibration test are assumed to be caused by the vibration environment. That is, no debugging, or "infant mortality", failures are assumed to occur. Considering the relatively short duration of most vibration tests, i.e., 1-15 minutes, and the quality of the components, it is unreasonable to credit any failures as due to substandard components.

4.1.3 Constant Part Count

To facilitate reliability calculations the assumption of a constant part population was made, i.e., when a failure occurs it was assumed that the piece was repaired and replaced and then the test continued - considering the high piece part count of most S/C and the small failure rate, this assumption does not take too much liberty with actual test procedures.

4.1.4 S/C Complexity

While the piece part count was found for the majority of S/C, the part distribution was not. Therefore, in normalizing to the piece part count the assumption is made, that in general larger more complex S/C are not made more complex due to paralleling systems, for increasing reliability by redundancy, but that the larger S/C have more complex missions requiring more systems and hence, more components than smaller S/C.

All test failures are treated as random failures occurring throughout the test at a rate proportional to the severity of the test environment and the duration of the test, i.e.,

$$F = K \lambda t$$

where: λ = System Failure Rate within a non-vibration environment

K = Vibration Environmental Stress Factor

t = test duration

F = expected number of failures

4.2 Spacecraft Code

A rather large amount of the failure data used within this report was acquired with the understanding that it be treated as proprietary information and not be disclosed. Therefore, the S/C failure data is coded so that individual programs cannot be identified.

5.0 Derivation of Equivalent Vibration Environment Factor

A review of S/C vibration environmental specifications will immediately show a considerable variation in test levels and durations. Therefore, it was desirable to devise a method which would equate the different test patterns to determine which tests were potentially more damaging, to plot failure rate data against environmental severity in order to define the K stress factor, and to transfer the failure rate data to different test patterns for future use and analysis.

5.1 Derivation of Spacecraft (System) Vibration Environmental Stress Level Factor (Reference 27)

In the simplest case, when a system is subject only to failures which occur at random intervals, and the expected number of failures is the same for equally long operating periods, its reliability is mathematically defined by the well-known exponential formula

$$R(t) = e^{-\lambda t}$$

In this formula e is the base of the natural logarithm (2.71828 ....), λ is a consistant called the chance failure rate, t is the operating time, and R the reliability of the system; the reliability is then the probability that the system, which has a constant failure rate λ, will not fail in the given operating time t.

This reliability formula is correct for all properly debugged system which are not subject to "infant mortality" failures. The region in which the above formula is valid is conventionally referred to as the "useful life" of the system.

Figure 5.1 delineates the region in which the simple exponential relationship is valid. The "burn-in" region is generally associated with new components, whose early failures can be classified as production defects. Generally S/C design policy is to use burnt-in components; furthermore, the remaining early failures are screened during the functional and performance tests performed before the vibration test sequence.

For the exponential case, the inverse of the failure rate $1/\lambda$ is defined as the mean time between failures (MTBF), and is used as a method of comparing survival or reliability rates of different systems.

The objective now is to apply this theory to gain some insight as to how a S/C is theoretically damaged, in a vibration environment. Therefore, in order to determine the vibration environment, or test pattern, which will generate equal reliability or survival probabilities the following constants must be known:

$$R(t) = e^{-K\lambda t}$$

where K = Vibration stress factor

λ = S/C failure rate in an ambient environment

t = test duration

5.1.1 Spacecraft MTBF

Only in space have complete S/C been operational for long enough periods of time, and in sufficient quantity, to provide statistically valid data on MTBF's. The space environment has been found to be benign as far as electronic and structural components are concerned (Reference 28). Planning Research Corporation (Reference 29) has performed a study of Reliability Data from Inflight Spacecraft, in which the survival rate of 72 long term spacecraft are reported. This data was plotted on Figure 5.2. The MTBF was calculated and found to be 8176 hours.

5.1.2 Estimate of K Stress Factor

A search through reliability failure rate data handbooks and literature disclosed only two sources of data applicable to this study. MIL-STD-756 lists an overall stress factor of 80 for the launch and boost environment, while the FARADA Handbook listed (Reference 27) the stress factor of basic components versus g_{RMS} level.

The FARADA failure rate data handbook lists piece-part stress factors as a function of g_{RMS}. This data was plotted on Figure 5.3 and conservatively represented by a straight line.

A study of the launch environment shows that an average level of $4.5g_{RMS}$ can be expected for a spacecraft. This data was combined with the FARADA data to generate a K factor vs. g_{RMS}. In other words, a general system level of 80 @ 4.5 g's was used. At the higher g levels, the factor of 80 was multiplied by the FARADA factors.

5.1.3 Equivalent Test Pattern

Using the in orbit S/C MTBF and the K vibration stress factors, derived in Section 5.1.2, a family of curves with equal estimated damage was derived as a function of both acceleration level and test time as shown in Figure 5.4.

i.e. @ $g_{RMS} = 10$

$$K = K_{MIL-217} \times K_{FARADA} = 80 \times 5.5 = 440$$

@ Ambient Env. MTBF = 8176 hours

@ 10 g's $\therefore$ MTBF = $\frac{8176}{440}$ (hours) = 1115 minutes

Using the exponential relationship

@ 10% damage: $R(t) = e^{-\lambda t} = .90$

or $\lambda t = .106 = \frac{t}{MTBF}$

$\therefore \frac{t}{1115 \text{ min}} = .106$ & $t = 118.2$ minutes

Similarly

@ $g_{RMS} = 20$

$K = 80 \times 225 = 1800$

@ 20 $g's_{RMS}$ MTBF = $\frac{8176}{1800}$ = 4.54 hrs. = 272 mins.

@ 10% damage

again $\lambda t = .106$

or $\frac{t}{272} = .106$

$t = 28.9$ min.

A series of lines of equal reliability or damage, generated using the above mentioned technique, were plotted in Figure 5.4. These lines have the general relationship of g^2t equals a constant for the same amount of damage. This means that if the test level is halfed the test time should be four times greater to cause an equal amount of failures to occur. This relationship is fairly insensitive to the slope of the line used to envelope the FARADA data. For example, the enveloping line was changed for two cases shown in

Figure 5.3 as two dashed lines with the resultant $g^N t$ relationship calculated for each case. Furthermore, a conversation with the original source of the FARADA data, disclosed that the reported failure rates were estimated to be accurate to only $\pm 25\%$ and therefore the g^2t relationship derived is as accurate or exact as the source data.

An interesting relationship may be found by performing a dimensional analysis on the term g^2t. If it is assumed that the input force can be expressed as F=ma, instead of the more usual vibration relationship $F=F_0 \sin wt$, and assuming no sinusoidal motion then the number of failures N_F is proportional to the applied power per unit mass.

i.e. if $N_F = g^2t$ where N_F = Number of Failure

and F = ma

then: $N_F = \frac{Fgt}{M} \approx \frac{F}{m}\frac{X}{t}$

since work = F·X

and power = work/time

we have $N_F \approx \frac{\text{Applied power}}{\text{Unit mass}}$

The results of the reliability study, presented in this section, have predicted that an equivalent amount of damage or failures will occur when a S/C is subjected to a vibration environment with the same g^2t value. The next section uses the g^2t values coupled with actual S/C failure data to empirically determine the failure rate coefficients.

5.1.4 Sinusoidal vs. Random Vibration

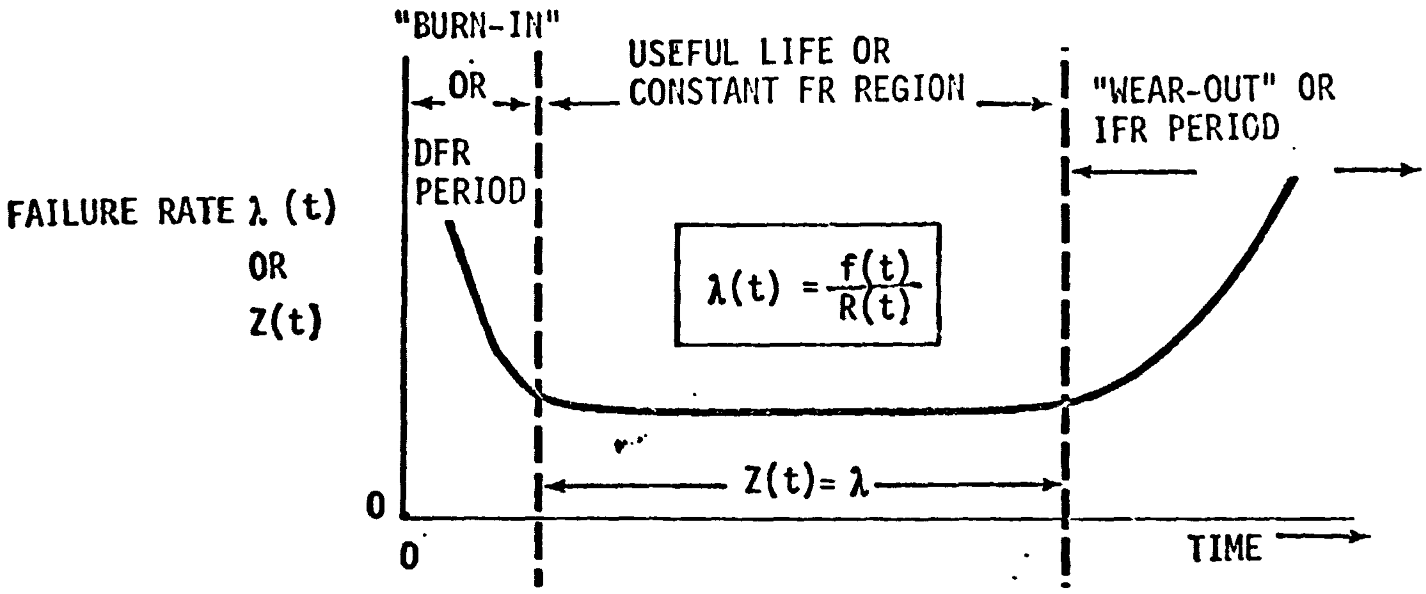
An experimental study (Reference 30) comparing fatigue life of a simple aluminum link under a random excitation and sinusoidal loading respectively has shown that the fatigue life of the randomly loaded specimens was one order of magnitude less than that of the sinusoidally loaded specimens (See Figure 5.5). The fact that random excitation is more damaging to electronic components, than a sinusoidal one, is also corroborated in the FARADA Handbook (Reference 27).

During spacecraft vibration testing the sinusoidal sweeps are generally less severe than the random excitation portion of the test program; however because the frequency is swept from 20 to 2000 cps there will be brief intervals where a resonant response will be excited. To quantitatively account for the sinusoidal resonant effects and the overall fatigue sensitivity to the random excitation the following relationship was used when comparing test patterns:

A - For random excitation the g^2t values were made using the g_{RMS} level.

B - For the sinusoidal portion of each test the peak acceleration level was used in calculating the g^2t function.

C - For both cases t is in minutes.



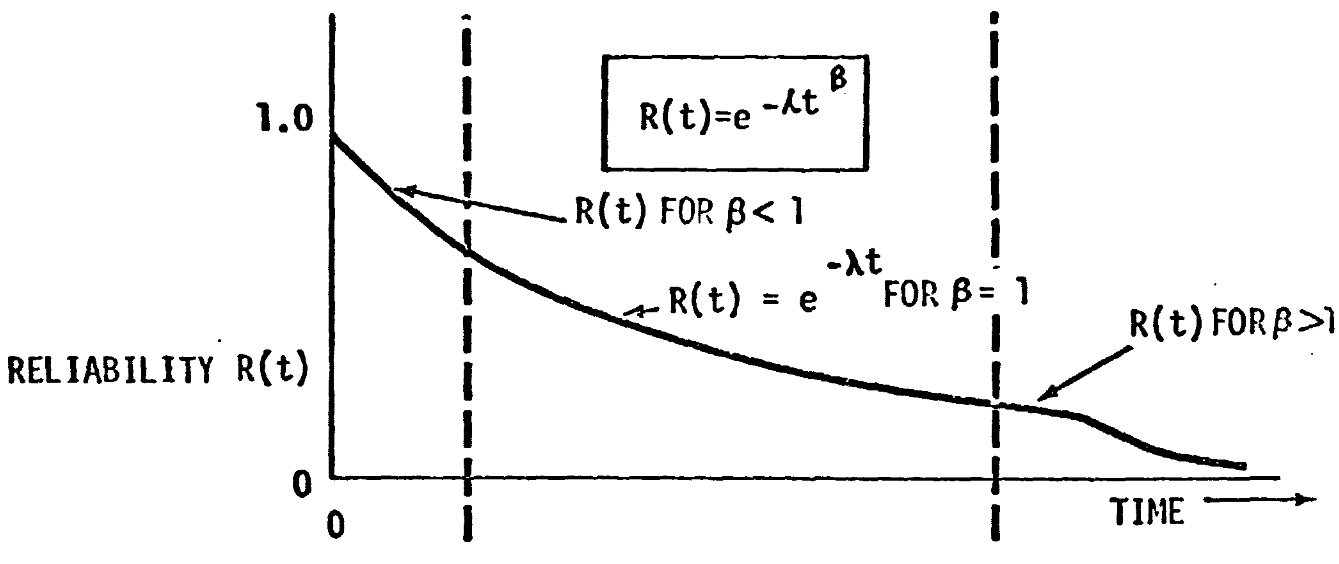


FIGURE 5.1 - TYPICAL SHAPES OF THE FAILURE-RATE, AND SURVIVAL FUNCTIONS FOR THE DFR (DECREASING FAILURE-RATE), CONSTANT FAILURE-RATE, AND IFR (INCREASING FAILURE-RATE) PERIODS.

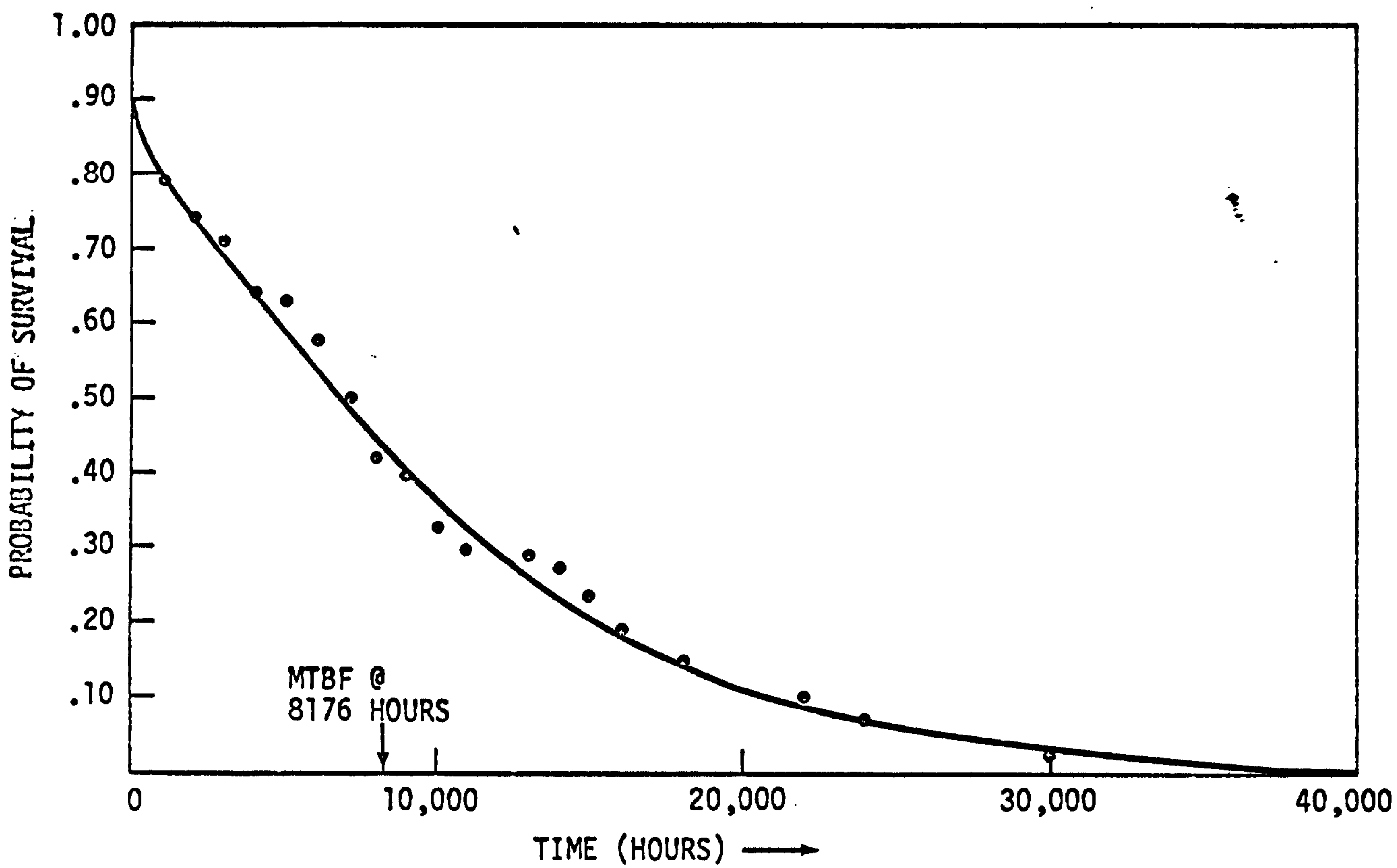


FIG. 5.2 SURVIVAL RATE OF 72 LONG TERM SPACECRAFT

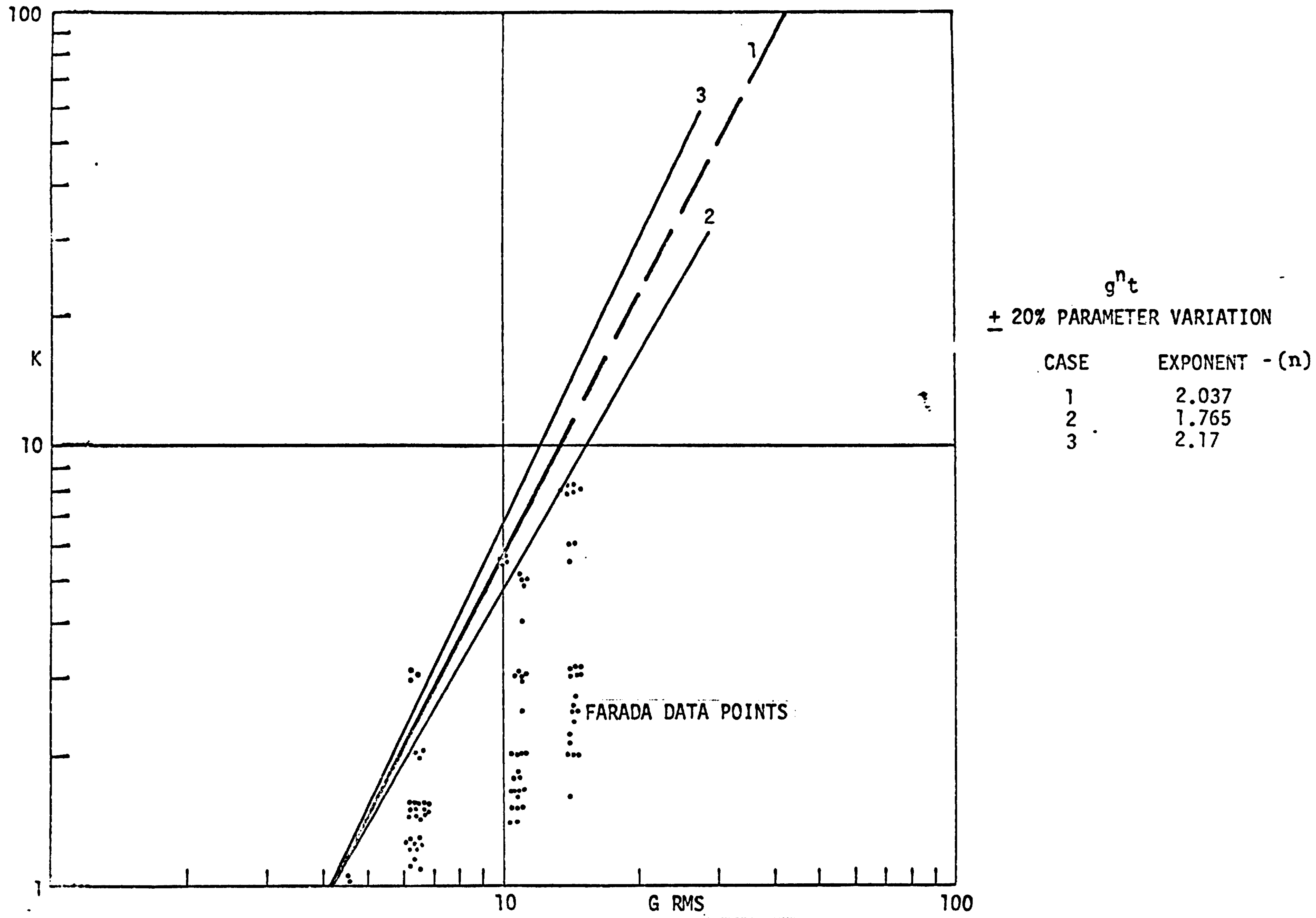


FIG. 5.3 FARADA ENVIRONMENTAL STRESS FACTOR K VS g R.M.S.

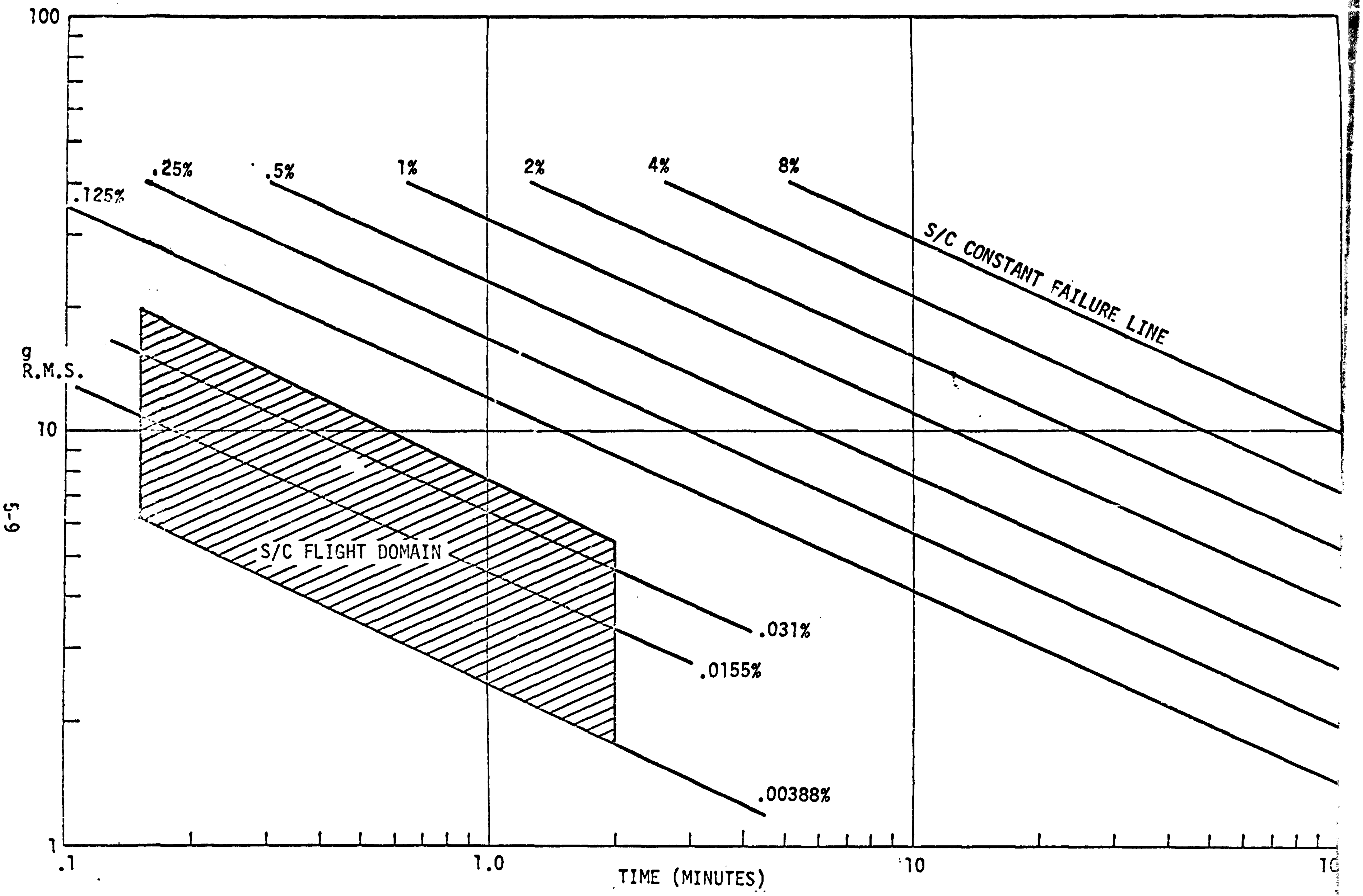


FIG. 5.4 PREDICTED S/C FAILURES VS v R.M.S. & TIME

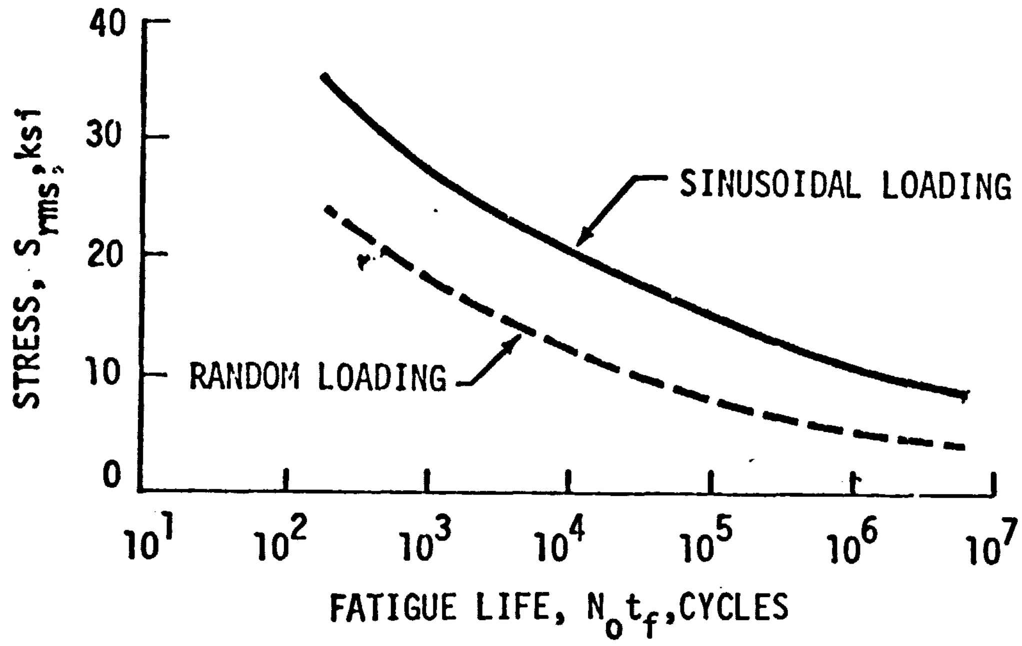


FIGURE FIG. 5.5 COMPARISON OF FATIGUE LIVES FOR RANDOM AND SINUSOIDAL LOADING

6.0 The Spacecraft Vibration Environment

The following three sections summarize the g^2t severity factor for the spacecraft flight, acceptance and qualification vibration environment. These g^2t values were obtained from flight performance reports and from test specifications. Reference (31-32).

6.1 The Flight Environment

The severity factor of the spacecraft launch and boost environment was claculated by converting flight vibration data (g_{RMS} vs. time) to g^2_{RMS} versus time curves and integrating the area under the curves (See Figure 6.1). Reference 35-48. The flight environment severity factor was found by this technique to vary from a minimum of about 6 to a maximum of 60 except in the case where an X-248 solid propellant third stage rocket motor developed a "resonant burn" condition equivalent to an additional g^2t value of about 165. See Reference 8.

An interesting variation of vibration acceleration levels, on the same launch vehicle, is caused by different shroud or nose cone configurations and by the launch pad cooling techniques. (See Figure 6.2). That is, if a "wet pad" is used the cooling water will absorb some of the acoustic energy, resulting in about a 20% reduction in the liftoff vibration levels by comparison to the case where a hard, acoustically reflective surface (concrete) is used.

As mentioned previously in Section 2, the shroud configuration can affect the aerodynamically generated S/C vibration even more significantly. The cone-cylinder nose shroud may have a g^2t factor 1/7 as high and a peak acceleration level 2/3 less than other more irregularly shaped shrouds such as the "Hammer Head" and "Boat Tail".

6.2 Acceptance Vibration Environment

The vibration acceptance tests are performed because even though a prototype model has been qualified, virtually no information is available on the variations that can be expected between units of the same design. The test level for a time is generally equal to the launch and boost duration per each of the three main orthogonal axis.

As would be expected the acceptance test level severity factor is significantly higher than the flight g^2t levels. This is because the 95th percentile levels only occur for a relatively short period during powered flight. The S/C acceptance test severity factors were found from environmental test specifications to vary from a low of 270 to a maximum of 1916. The higher values corresponded to launch vehicles using the X-248 rocket motor. Table II, comparing the flight and acceptance test levels, shows the relationship between the severity factors:

Flight (g^2t)	Acceptance Test (g^2t)
6-60	270 - 500
171-225 (X-248 Motor)	500 - 1916

TABLE II - S/C g^2t FLIGHT & ACCEPTANCE TEST LEVELS

At either extreme then, the acceptance test can be actually 2 to 83 times more hazardous to the S/C than the launch environment: the average factor being 11 times more severe for S/C not using the X-248 motor and 6 times more severe for those S/C using the X-248 rocket motor.

6.3 The Qualification Environment

The spacecraft vibration qualification test is performed at the 99th percent probability level ($\approx$1.5 times greater than the acceptance test levels) for about twice the duration. Consequently the g^2t qualification test severity factors are about 4.5 times higher than the acceptance g^2t factors.

Table III compares the qualifications, acceptance and flight g^2t severity factors.

Flight (g^2t)	Acceptance Test (g^2T)	Qualification Test
6 - 60	270 - 500	1200 - 2200
171 - 225	500 - 1916	4000 - 5300

TABLE III - S/C FLIGHT, ACCEPTANCE & QUALIFICATION g^2t LEVELS

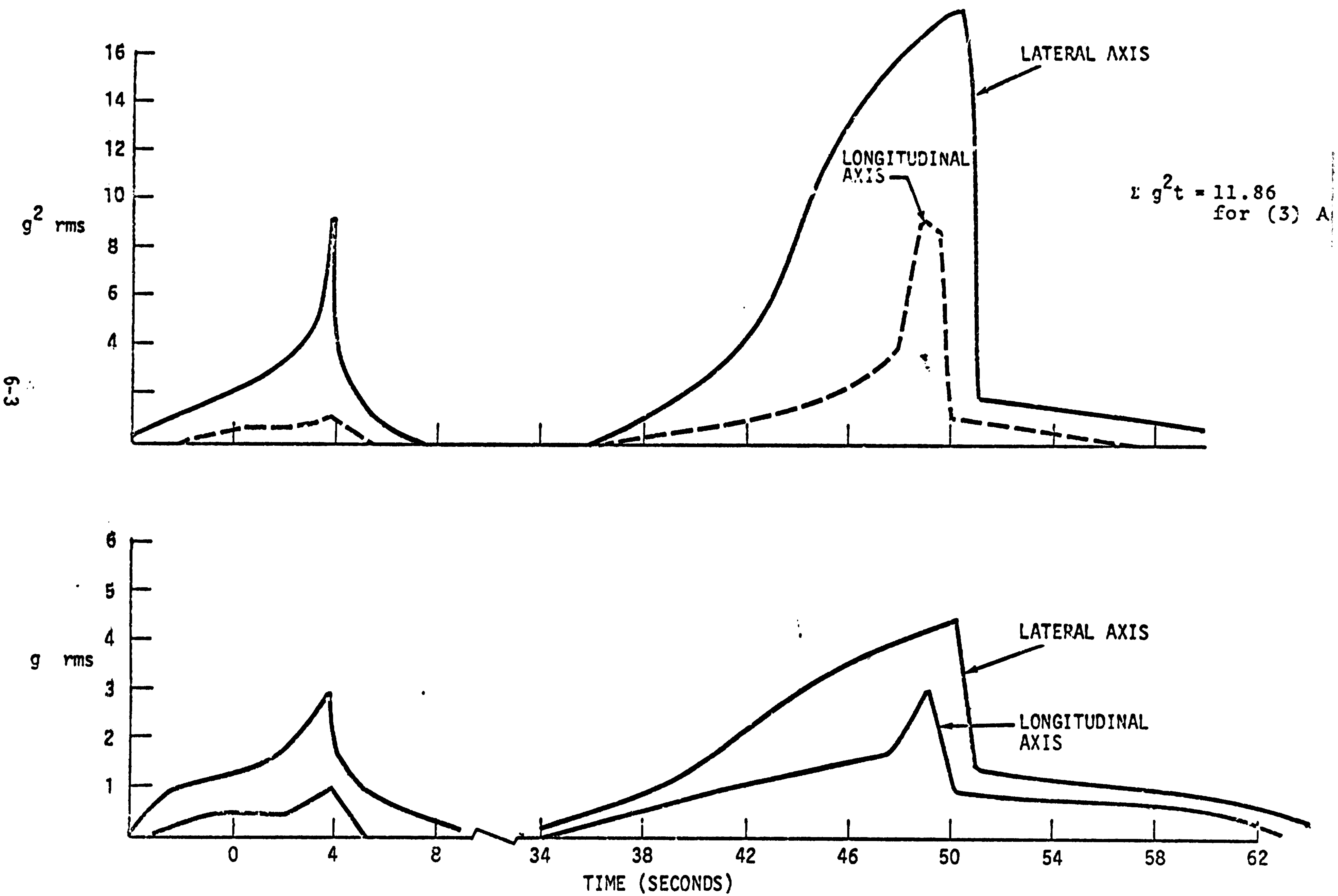


Fig. 6.1 CALCULATION OF FLIGHT g^2t

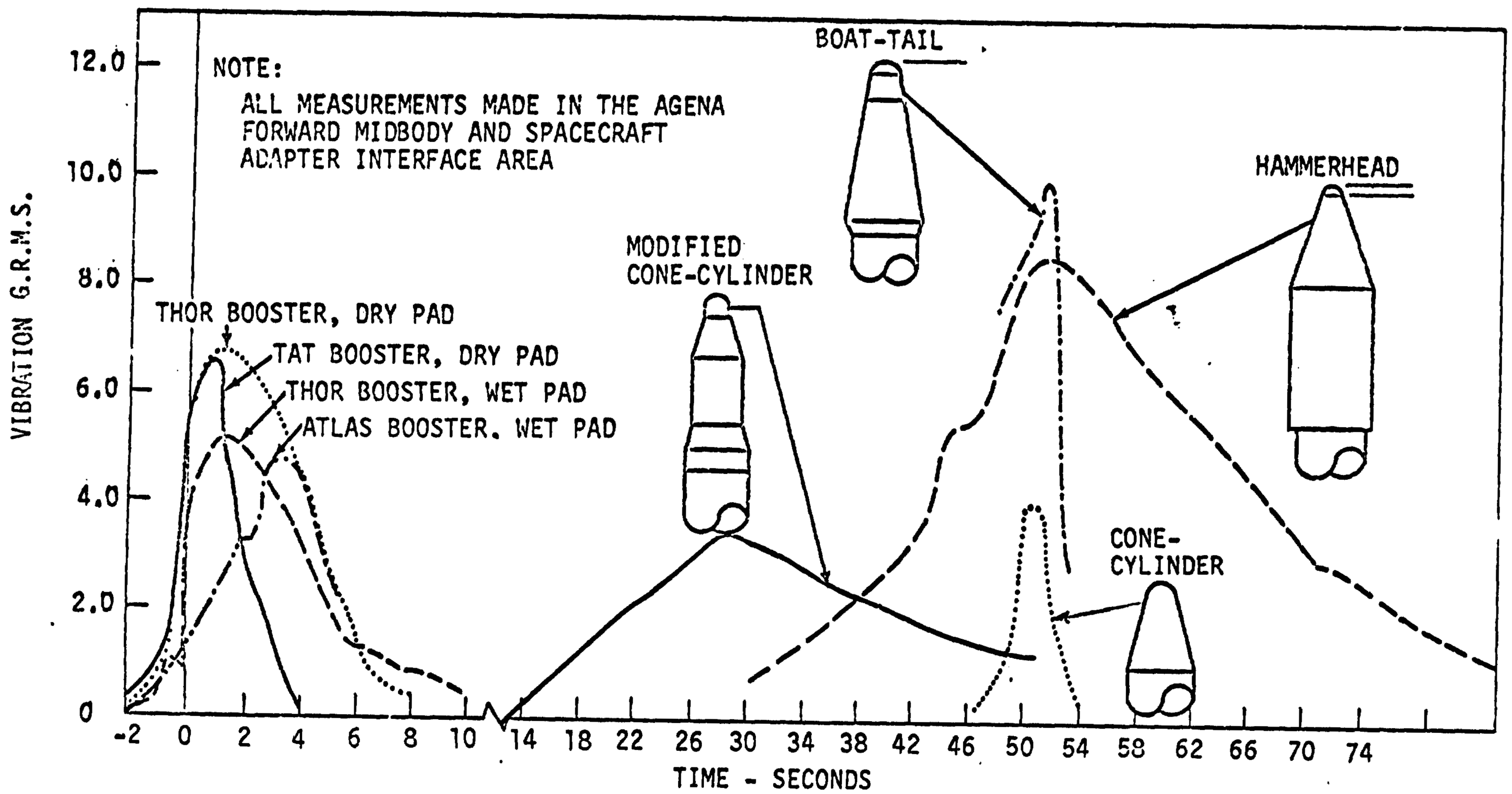


FIG. 6.2 TYPICAL SPACECRAFT VIBRATION TIME HISTORY

7.0 S/C Failure Data

7.1 Laboratory Failure Data

The S/C failure data used within this study was acquired from Environmental Test Reports, Program Development Final Reports, and from conversations with the Test and Evaluation Group at Goddard Space Flight Center.

Documented component or subsystem failure rate data is extremely scarce. The best data available on that subject is furnished in References 40, 20, 41 and 42. Spacecraft failure data, while limited is generally available as shown in References 1, 12, 13, 16, 20, 24, 25, 26, 40, 41 and 42.

The best estimates of how serious these problems or failures would be is that between 25 and 50% are "of a castrophic nature" (Reference 10 and 15). A compilation of laboratory failures occurring with 7 qualification and 20 flight S/C has shown the following part distribution.

%Electronic Equipment	%Structural	%Attitude Control	%Misc.
59	20	5	15

TABLE IV - S/C LABORATORY FAILURE DISTRIBUTION

This distribution of failures closely parallels the weight breakdown of a S/C as shown below in Table V for 6 S/C.

Spacecraft	% Electronic	% Structural	% Attitude Control	% Misc.
Explorer XII	65.6	27	N/A	7.2
" " XIV	""	"	"	"
" " XV	""	"	"	"
Aluette I	74	19	"	N/A
Sert	56	21	"	"
Mariner (Venus)	59	17	13	10

TABLE V - S/C WEIGHT DISTRIBUTION

The failures listed in Table IV, whenever possible, represent significant problems requiring either a design or fabrication "fix". A half a dozen screws loosening would at most be counted as 1/2 failure, generally most of the S/C failures consist of serious problems such as structural and component failures.

7.2 Flight Failure Data

Flight failure data is not only extremely limited but because of the inaccessability factor some failures may go unnoticed. The most complete and best source of flight failure data is contained reference 29.

The distribution of flight failures is shown in Table VII, the sample source is 198 successfully launched S/C with 665 documented anomalies and failures:* 26% of the vibration induced failures were catastrophic or severely effected the S/C performance.

%Electronic	% Structural	% Attitude Control	% Misc.
82.5 **	2.3	12.4	3

TABLE VI - FLIGHT FAILURE DISTRIBUTION*

*51% of these failures occurred during the launch phase of the mission.

**The higher than expected electrical-electronic failures is probably caused by the fact that only electronic communication is available with the S/C and consequentially an electrical malfunction is more likely to be sensed than any other.

The flight failure data was obtained from Planning Research Corporation, Reference 29, and from conservations with the TEst and Evaluation group at GSFC. Each reported flight failure was reviewed and sorted into two groups. The first were those failures which were very probably caused by vibration, while the second group represents those failures which were possibly induced by the flight vibration environment.

7.3 Normalization of Failure Data

The S/C failure data was normalized to the S/C weight, Reference 43, and, whenever known, the piece-part count. This was done because a S/C with a high component count would be expected to have a larger number of failures than a smaller S/C with less components

In order to estimate the piece-part count of S/C when the talley was not available, a plot of known component pieces vs. weight was made up and used (See Figure 4.1). The piece-part count, used for this study, did not include the number of solar cells, and soldered connections or screws, bolts, washers contained within the S/C.

27 S/C Programs
83 S/C

Table VII
SPACECRAFT VIBRATION FAILURE DATA

S/C Type	Weight (Pounds)	Number of Components	Vibration Test Environment: Qualification g^2t	Qualification Failures Per S/C	Acceptance g^2t	Acceptance Failure Per S/C	Flight Failures g^2t	Probable F	Possible F	No. of Flt S/C
1	320	6,911	2,719	2	1,000	1	6-60	0	0	1
2	86	4,174	4,500	4	1,800	1.5	60-160	1	0	3
3	458	12,000	4,262	15	"2000"	N/A	60-160	0	0	4
4	83	5,858	5,235	2	1,163	1	60-160	N/A	N/A	-
5	89	5,858	N/A	N/A	1,163	1	N/A	N/A	N/A	-
6	138	12,002	5,235	7	2,000	2	60-60	0	1	1
7	405	9,600	5,235	11	1,916	2	60-160	N/A	N/A	-
8	132	5,114	2,165	10	759	2.0	60-160	0	1	1
9	150	"5800"	2,165	11	759	3.0	60-160	0	0	1
10	575	37,761	4,561	9	461	.333	6-60	0	0	2
11	172	11,500	4,400	16	1,800	2.5	60-160	0	0	3
12	120	8,000	2,719	2	--	N/A	--	N/A	N/A	-
13	79	5,200	5,235	1	--	N/A	--	N/A	N/A	-
14	172	5,600	5,235	4	1.916	.4	60-160	N/A	N/A	-
15	860	21,500	1,438	3	476	.2	6-60	1	1	5
16	8,000	58,000	6,500	N/A	138	1.25	6-60	0	3	8
17	3,000	8,874	8,000	N/A		N/A		1	0	24
18	285	4,800	4,000	N/A	1,500	N/A	60-160	2	3	8
19	830	29,300	1,965	N/A	525	N/A	6-60	0	0	1
21	447	13,775	4,561	3	461	1	6-60	0	0	2
22	700	15,717		N/A		N/A	6-60	1	2	6
23	140	9,684	"2700"	N/A	636	N/A	6-60	0	0	1
24	140	8,343	"2700"	N/A	636	N/A	6-60	1	0	1
25	136	12,016	"2278"	N/A	"494"	N/A	6-50	0	0	1
26	130	9,000	"2278"	N/A	"494"	N/A	6-60	0	1	1
27	90	5,000	"2278"	N/A	"494"	N/A	6-60	0	0	2
28	2,200	60,000	1,700	N/A	372	N/A	6-60	0	0	7

N/A - NOT AVAILABLE

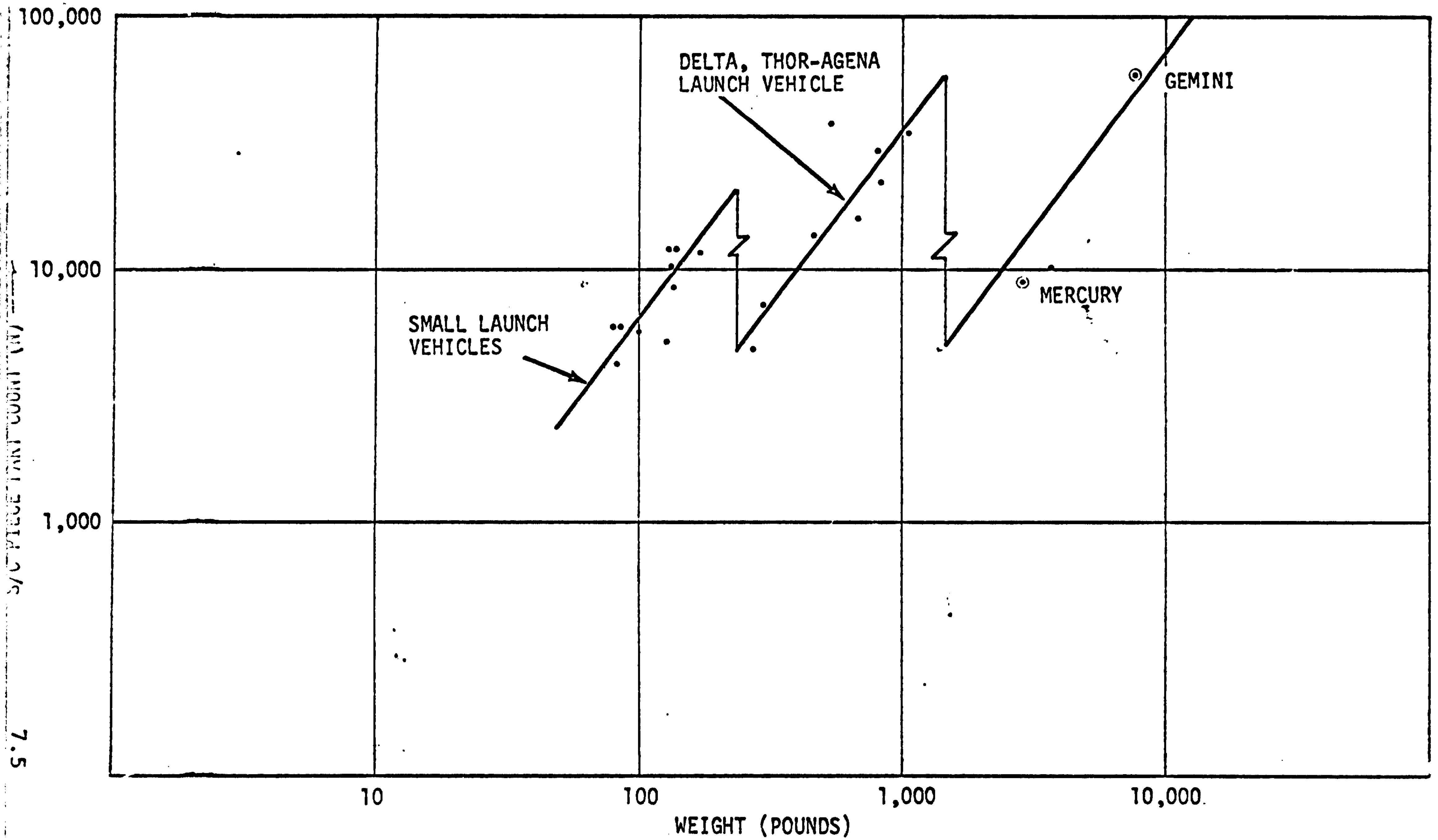


FIG. 7.1 S/C WEIGHT VS PIECE PART COUNT

8.0 S/C Failures vs. Weight and Components

Table VII is a tabulation of the survey data sample. Included is the weight, piece-part count, the failures per S/C which occurred within the qualification, acceptance and flight environments, and the environmental severity factors (g^2t). As indicated, this information was not available for the complete data sample. Estimates of the environmental test levels were made using the applicable Environmental Test Level Specifications. The flight levels were estimated, when data was not available, from known S/C using similar launch vehicles and shroud configurations.

8.1 S/C Failures vs. g^2t

Figures 8.1 and 8.2 are plots of S/C failures versus g^2t, normalized to the spacecraft weight and the piece-part count. As would be expected from Figure 4.1, a greater spread of the failure data was found, when the data is normalized to weight rather than the piece-part count.

The spacecraft failure data of Figures 8.1 and 8.2 is plotted with either an open or closed circle. The open circled symbols represent S/C in which each subsystem was qualified and tested prior to system level testing, the closed point symbols represent S/C failure points for those S/C where no previous vibration testing was performed prior to S/C system testing. This explains why the majority of open circled data points lie to the left of the arithmetic mean of Figures 8.1 and 8.2. The flight failure data was plotted an an open box with the horizontal sides representing the g^2t extremes and the vertical extremes representing the limits of the possible and probable flight vibration induced failures.

The S/C failure rate data has also been plotted in Figures 8.3 and 8.4 as failures versus acceleration and time as a function of S/C weight and piece-part count.

A comparison was made between seventy-eight spacecraft to see if the flight failure rate of pre-tested spacecraft differed from those S/C which were not tested prior to system level tests. As would be expected no measurable difference was noted since it should not make any difference to the spacecraft if a "weak link" or potential failure mode is corrected during a subsystem or system test. These calculations are listed in Appendix A.

8.2 S/C Flight Failure Rate vs. Qualification Test Levels

A comparison was made between 71 S/C to see if any difference in the flight failure rate could be discerned as a function of the qualification test levels. The flight failure rate for those S/C using the X-24 rocket motor were normalized to the other S/C by dividing their actual flight failure rate by the g^2t ratio of their flight environments which is equal to 6.

The results of these calculations, as shown in Appendix B, were plotted in Figures 8.5 and 8.6. The piece-part failure rates show a strong relation between test levels and flight failure rates where as the correlation between weight based failure rates is at best described as following the same pattern. This can be explained as due to the fact that the weight versus piece-part count is a discontinuous function as indicated in Figure 4.1 and that the increased failure rate, with decreasing test levels, cannot be accurately resolved with the existing spread of the failure rate data versus weight.

8.3 Subsystem Failure Data

Subystem failure data is very poorly documented and scarce. This survey found only five programs where this data was recorded. Furthermore, even for these five programs only the number of types of failures are tabulated with no reference to the number of components or weight of each subsystem: Subsystem failure data was acquired from REferences 29, 40, 41 and 42. This data was plotted in Figure 8.7 as percentage of failures versus g^2t. Though the percentage of subsystem failures might seem high, by comparison to the number of failures reported during the S/C qualification and acceptance tests, it should be remembered that these failures must be amortised over several S/C. When this is done the failure rates are of the same order of magnitude as the individual, untested spacecraft.

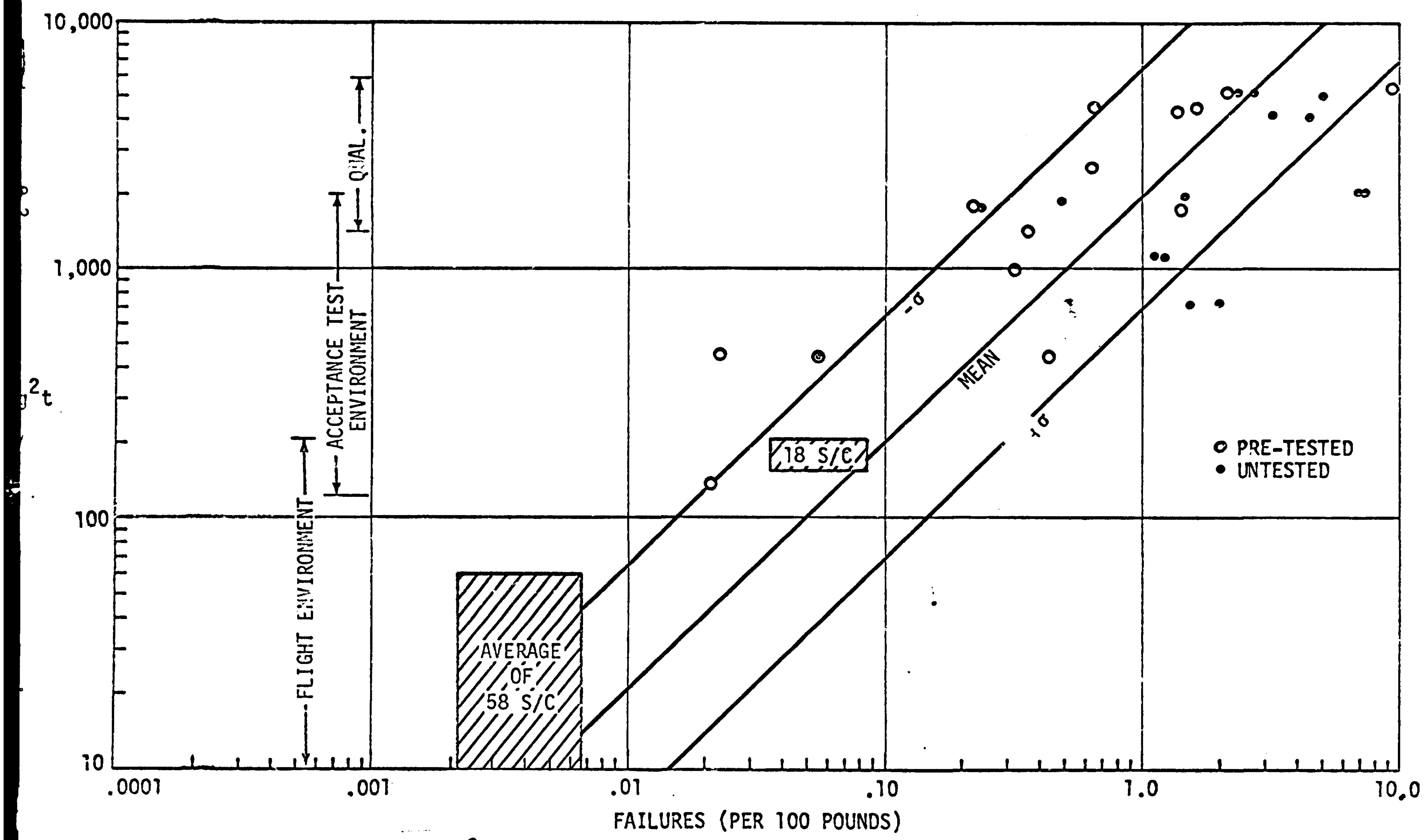


FIG. 8.1 S/C FAILURES VS g^2t

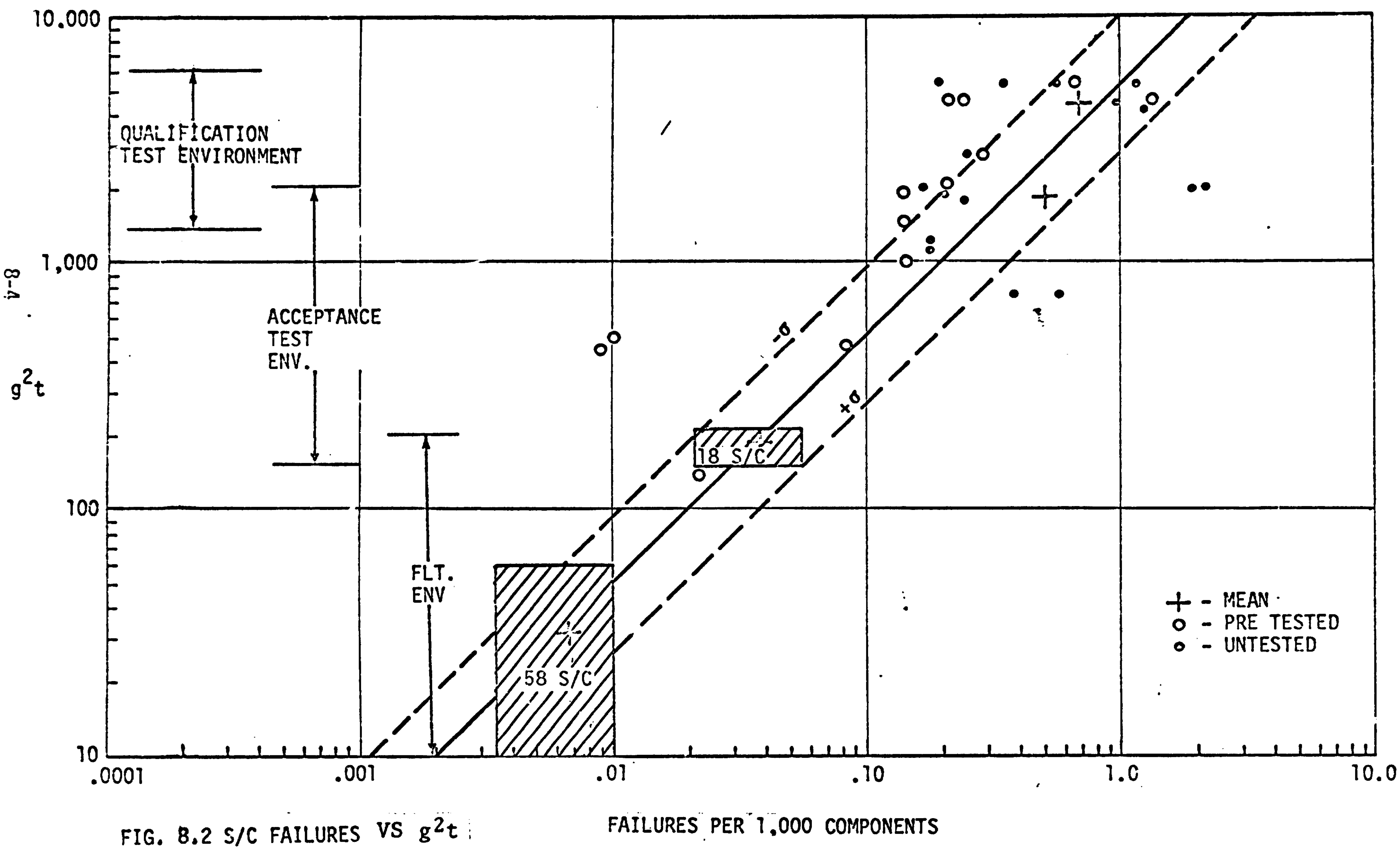


FIG. 8.2 S/C FAILURES VS g^2t

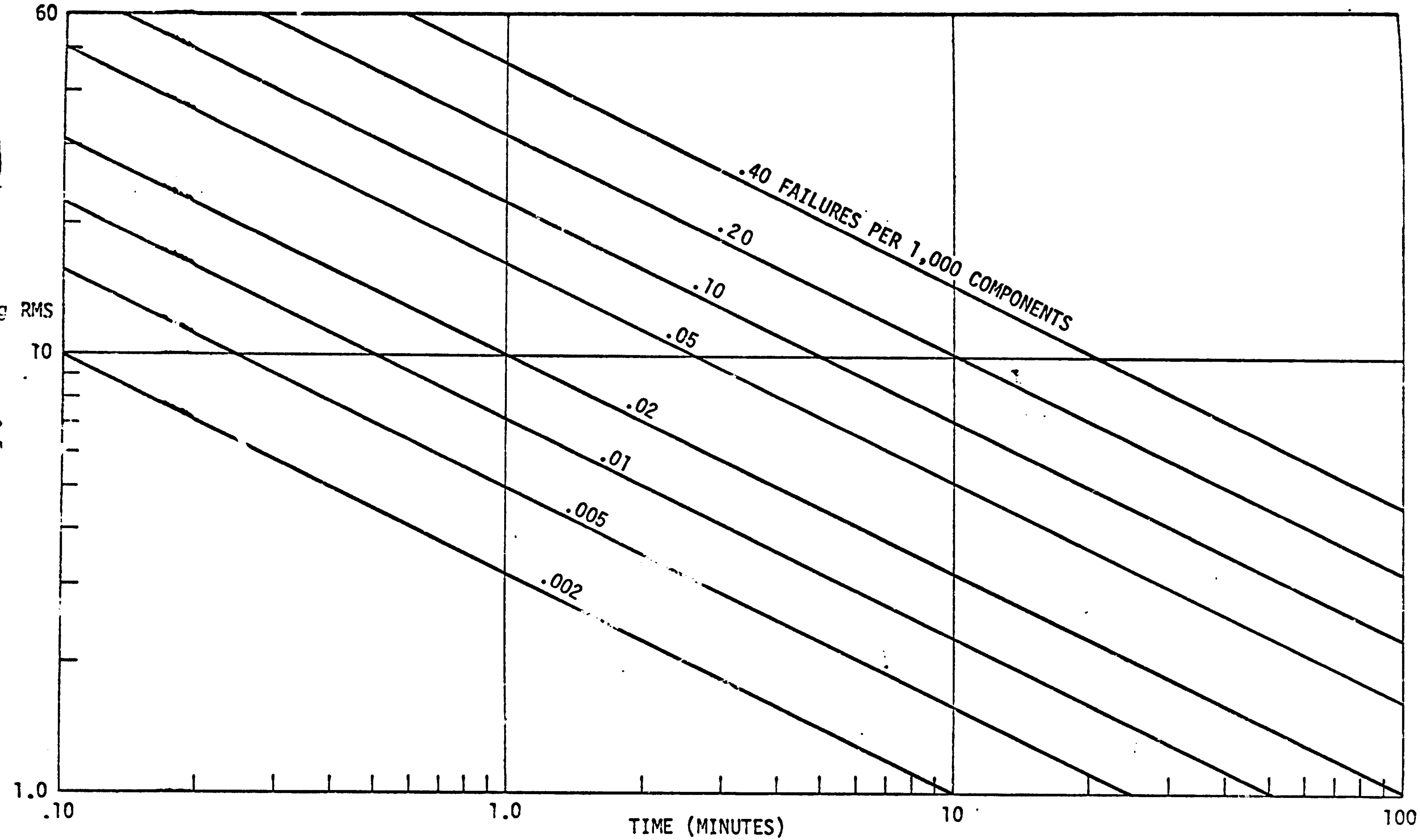


FIG. 8.3 S/C FAILURES VS g R.M.S. & TIME

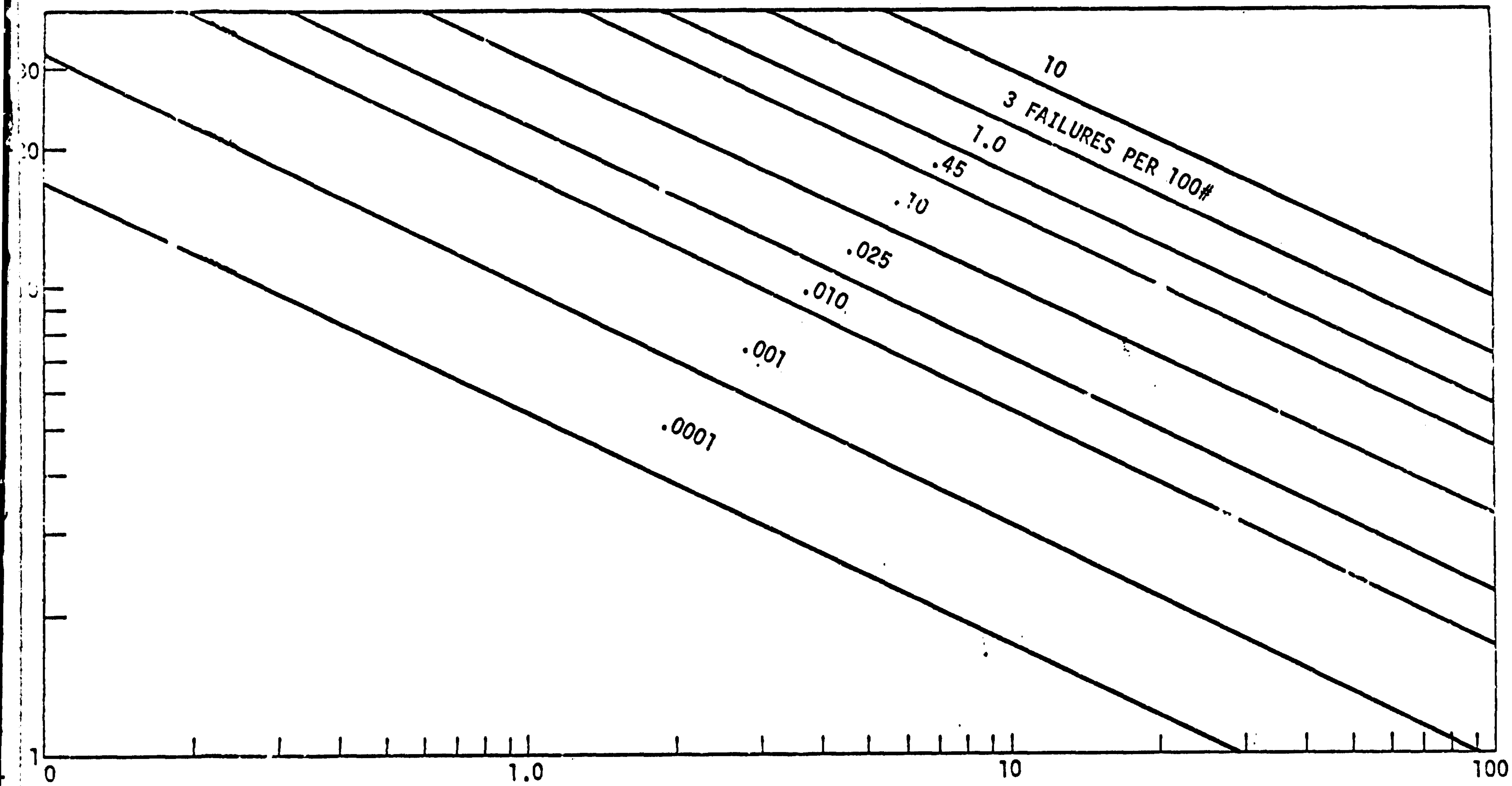


FIG. 8.4 S/C FAILURES VS g R.M.S. & TIME

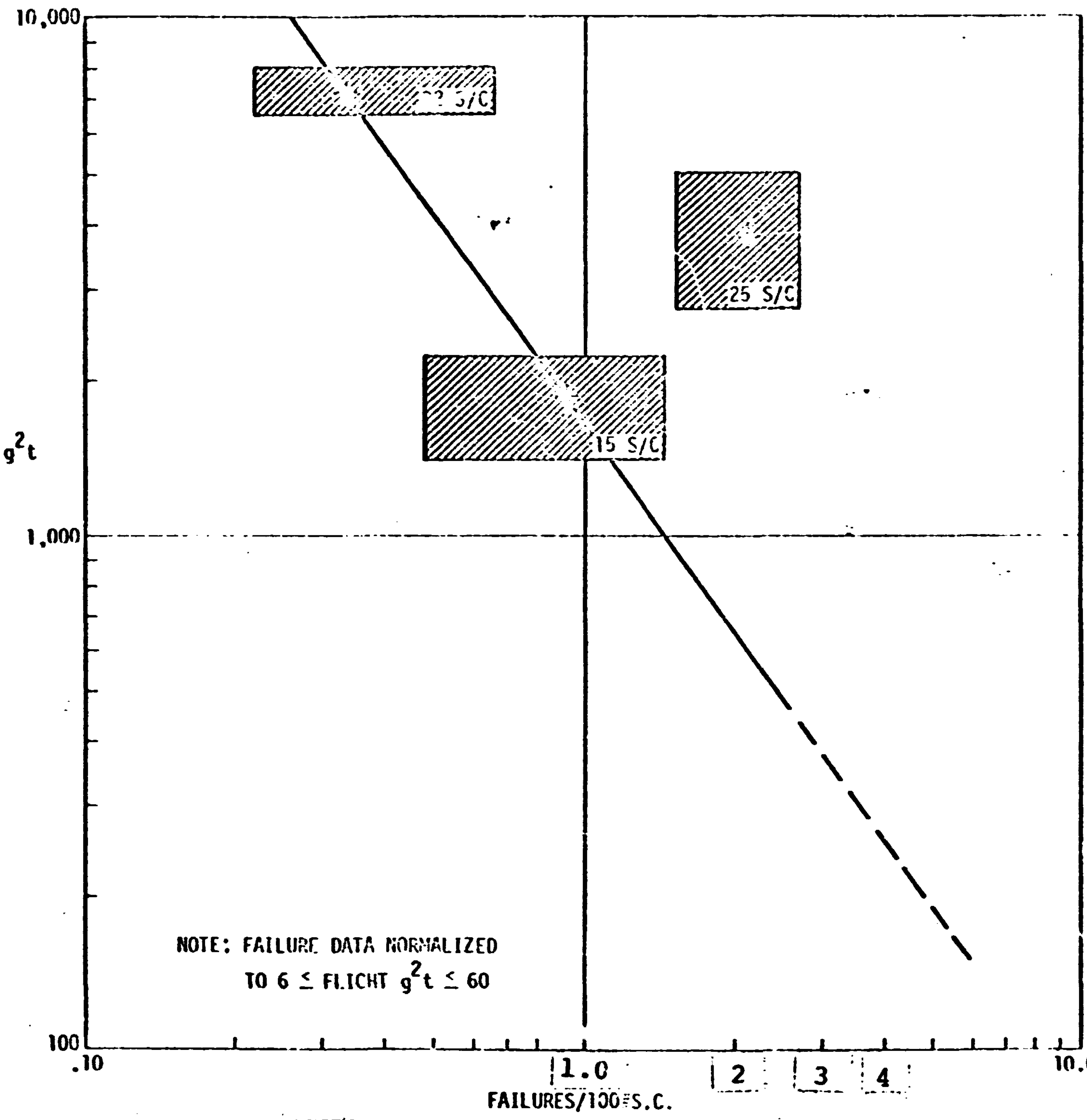


FIG. 8.5 S/C FLIGHT FAILURE RATE VS QUALIFICATION g^2t

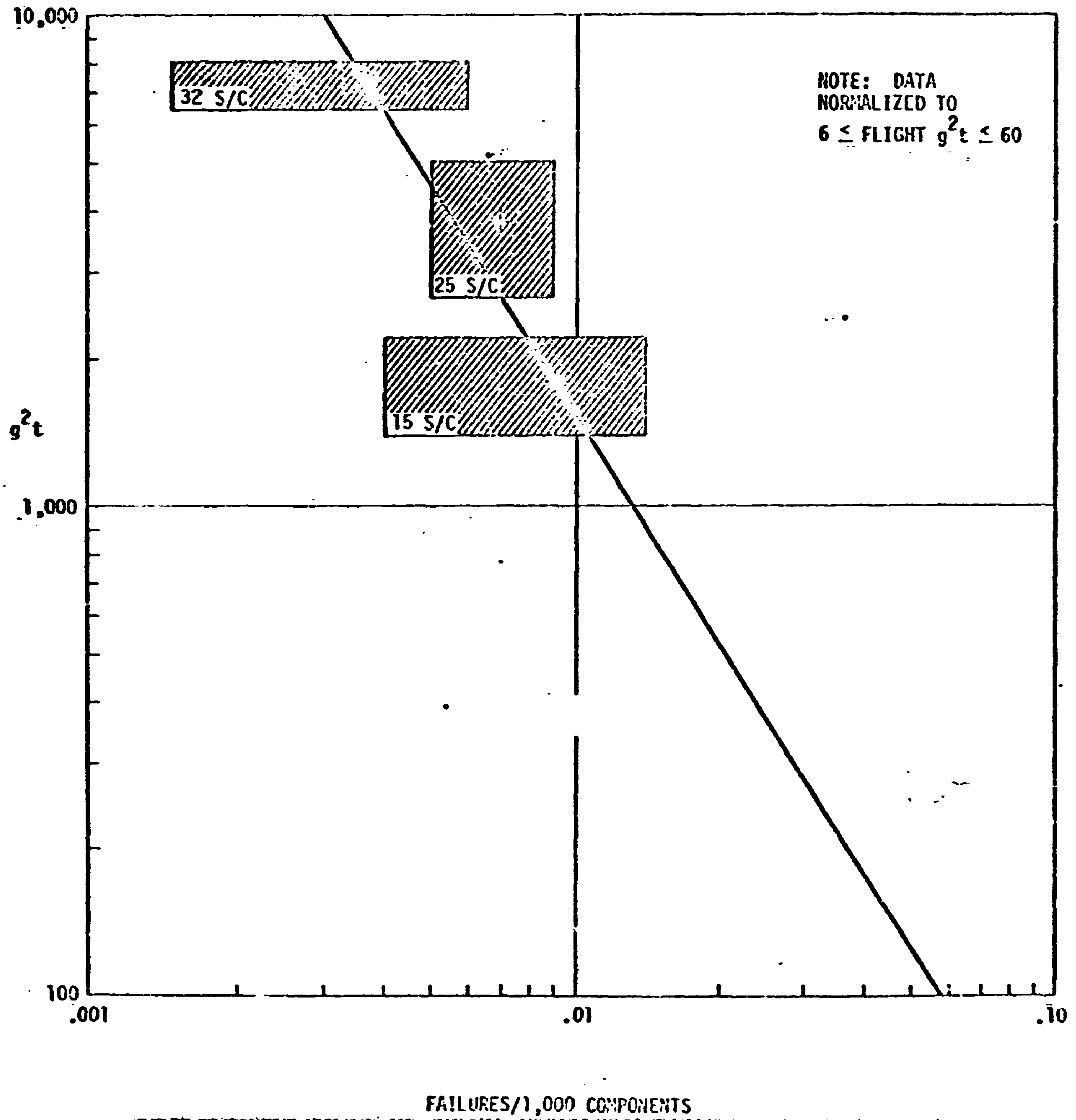


FIG. 8.6 S/C FLIGHT FAILURE RATE VS QUALIFICATION g^2t

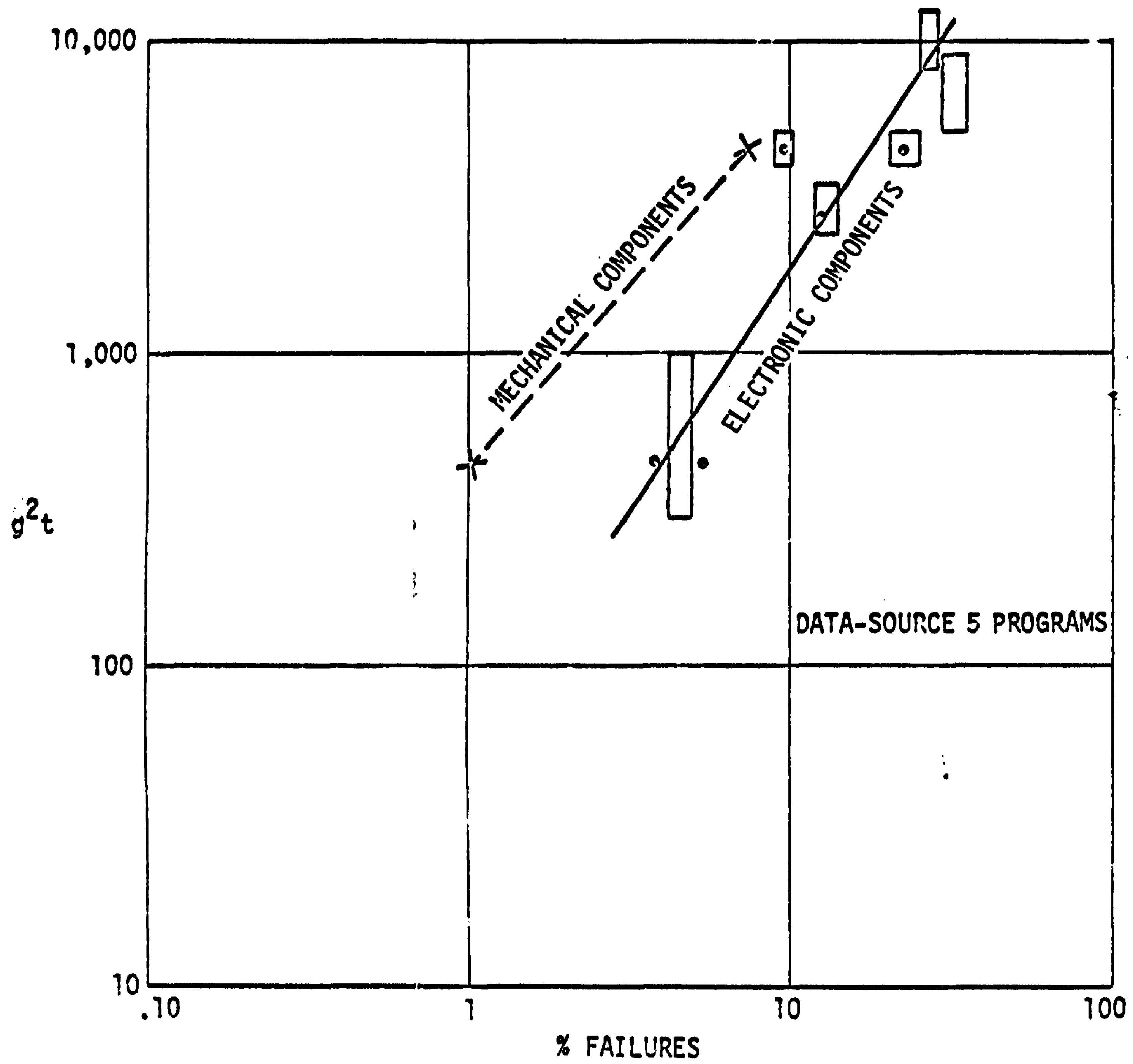


FIG. 8.7 SUB-SYSTEM FAILURES VRS g^2t

9.0 VALIDATION OF g^2t SEVERITY FACTOR

The two most important assumptions used in deriving the g^2t severity factor were:

(1) - S/C failures are proportional to g^2t

(2) - The failure rate (λ) is constant

Figures 8.1 and 8.2, in which spacecraft failures, normalized to weight and piece-part count, plotted against g^2t, show that a straight line with a slope of 45° will fit the average failure rates over a region exceeding two orders of magnitude. The forty-five degree slope, on log-log paper, indicates that the relationship is linear or therefore, that failures, on the average, are directly proportional to the g^2t value and therefore, that the failure rate is constant. Furthermore, since λt is equal to the expected number of failures then the reliability of a S/C vis a vis vibration failures can be calculated by using Figures 8.1 or 8.2, knowing either the S/C weight or part count.

A second way in which the original assumptions have been validated is to compare the failure rates of S/C with airborne equipment plotted in MIL-STD-217 from field performance data. This data is plotted in Figure 9.1 and superimposed upon it is the S/C MTBF based on a 5 g_{RMS} and 10 g_{RMS} launch and boost environment using the data plotted on Figure 8.2.

As can be seen in Figure 9.1, the S/C MTBF is slightly lower than that for airborne equipment. This is reasonable, since in general airborne equipment is subjected to a less severe vibration environment. It must be emphasized that the MIL-STD-217 curve was not used in any prior calculations and that the calculated missile environment MTBF agrees well with the empirical field data.

Because of the importance of these calculations, in helping to validate the g^2t relationship, a work sheet has been included in Appendix C delineating the manner in which the MTBF can be calculated from the failure rate data curves of Figure 8.2.

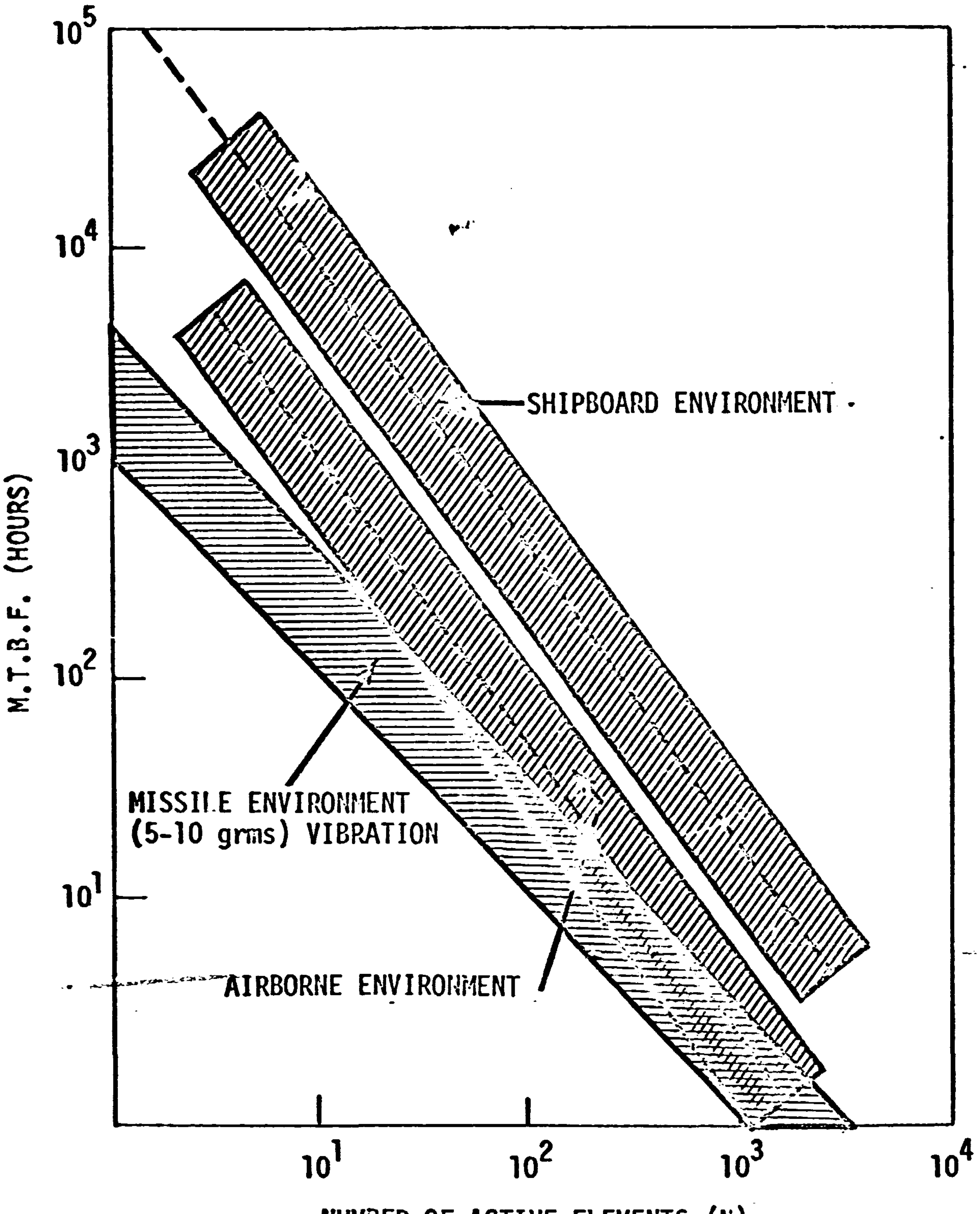


FIG. 9.1 FAILURE RATES AND MTBF VERSUS FUNCTIONAL COMPLEXITY FOR ELECTRONIC EQUIPMENT. (SYSTEM LEVEL) REPRODUCED FROM MIL-STD-756

10.0 Failure Rates

As previously mentioned in Section 7 between 25 and 50% of the problems experienced in the laboratory could be defined as catastrophic, whereas only 26% of the vibration flight anomalies are found to be severely debilitating to S/C performance . Furthermore, it was found from the subsystem failure rates that the rate of mechanical failures was 1/5 that of electronic subsystems. Therefore, the following failure rates should be used for calculating vibration induced failures:

10.1 Expected environmental Test Failures

If each subsystem has been tested prior to system tests then the failure rates shown in Figures 8.1 and 8.2 should be used lying between the $-\sigma$ and mean values according to the g^2t level: On the other hand, if no pretesting was done then the $+\sigma$ value should be used.

10.2 Expected Subsystem Environmental Test Failures

The subsystem failure rates found during this survey are shown in Figure 8.7. For this case, the failure rates are shown as a percentage of failures or demonstrated reliability. These subsystems are understood to primarily consist of small electronic subsystems weighting between 5 and 20 pounds.

10.3 Flight Failures

As previously shown in Sections 8.2 and 8.3 the flight failure rate is independent of the manner in which the vibration environmental tests were performed but not of the test pattern. Therefore, the flight failure rate should be chosen from Figures 8.1 and 8.2, with the mean value as the best estimate, 25% of these failures would be considered as seriously affecting the performance of the S/C.

11.0 EFFECT OF ACCEPTANCE TEST g^2t ON FLIGHT FAILURE RATE

General

It has long been known, at least intuitively, that pretesting a flight model can weaken it and seriously affect its flight worthiness. A method has been devised in this section showing how the S/C flight reliability changes with the acceptance test severity (g^2t). Unfortunately, there is not enough data available to determine the optimum acceptance test level; however, there is enough to determine the flight worthiness of the actual qualification test article.

11.1 100% S/C Life in a Vibration Environment

In order to calculate the reliability of a S/C after acceptance testing and refurbishment, the 100% S/C life must be known or that g^2t condition in which all systems experience at least one serious problem. Assuming that each spacecraft system contains between 500 and 1000 components then, in flight the g^2t required to generate between 1 and 2 serious failures per 1000 components, or 100% S/C life, is $40{,}000 \leq g^2t \leq 80{,}000$. (From Figure 8.2, using the pretested S/C failure rates and statistical data showing that 1/4 the flight vibration problems are serious.)

11.2 Acceptance Test Levels vs. Flight Failure Rates

The survey data sample, tabulated in Table VII, was examined to determine if the flight failure rate was sensitive to the acceptance test g^2t. The work sheet showing these calculations is in Appendix D, and the results plotted in Figure 11.1.

While the correlation between the estimated 100% S/C life and that extrapolated from the acceptance test failure rates vs. g^2t, differ by a factor of two, this difference is not considered excessive considering the spread of the basic data.

There are now two curves showing flight failure rates as a function of test levels. The first one, derived in Section 8, shows how a spacecraft type is improved with higher qualification test severity factors (g^2t) (Figure 8.5, 8.6). The second one, derived in this section shows how an individual spacecraft is affected by the prelaunch test environment.

These two curves are superimposed in Figure 11.1. The flight reliability level of any spacecraft can be calculated, using Figure 11.1 if the S/C Qualification and Acceptance Vibration Test levels and duration are known.

Flight reliability calculations can be made using the following relationship:

$$\text{Flight Reliability} = e^{-\left[\lambda t_{qual} + \frac{(g^2t\ accept\ -.22)}{g^2t\ qual}\lambda t_{accept}\right]}$$

where:

e = Base of the natural logorithm (2.7182 ....)

λt = expected failures at the qualification & acceptance level g^2t

$\frac{(g^2t_{acc}-.22)}{(g^2t_{qual})}$ = a empirical factor used to account for the fact that the qualification and acceptance failure data cannot be separated. This factor will make the flight failure rate equal to the Qual type failure rate at g^2t qual = $4.5g^2t$ accept., the most usual ratio.

As can be seen, this formula predicts that the maximum reliability for any spacecraft occurs when the acceptance test $g^2t=0$. While this is definitely true for random type failures it is not true for failures caused by defects built into the S/C which is what the acceptance test is supposed to uncover.

It would seem, therefore, that until more data becomes available on acceptance testing failures, that the acceptance g^2t value should be chosen so that they are at least twice and possibly three times more severe than the launch vibration environment. This would place the g^2t accept at 180, for S/C using liquid fueled launch vehicle.s

11.3 Flight Worthiness of the Qualification S/C

As a general rule the qualification model is not refurbished and flown. Two cases were found, however, where a qualification S/C was refurbished and launched. These S/C thereafter operated without any apparent launch induced vibration problems.

The reliability of a qualification test model may be calculated from the flight reliability formula by using g^2t ACCEPTANCE = g^2t QUALIFICATION. In this case there are two opposing factors which will determine the flight reliability of the S/C.

If the qualification test is very severe the flight reliability of the production flight S/C will be increased; however, the reliability of the test specimens will be very low. On the average, if the qualification test is not severe, the S/C flight reliability will tend to be low, whereas, the reliability of the refurbished test article will be high.

The flight reliability of a refurbished qualification S/C was calculated and plotted in Figures 11.2 and 11.3 as reliability versus the qualification g^2t for a S/C with 10,000 and 90,000 components respectively. For comparison, the flight reliability of a production S/C (where the acceptance g^2t was equal to .22 times the qualification g^2t) was plotted on the same figure. These calculations were made for the serious case where the flight failure rate is 1/4 that of the total number of flight induced vibration problems.

The flight worthiness of a refurbished S/C is highly dependent upon the size or number of components contained within it. A small S/C (N=10,000) qualification test model can have a flight reliability factor of 95.5 by comparison to the acceptance test model with a flight reliability of 97.5%. The larger S/C (N=90,000) qualification test model will have a maximum reliability of 67%, whereas the acceptance test model will have a reliability level highly dependent on the severity of the qualification test.

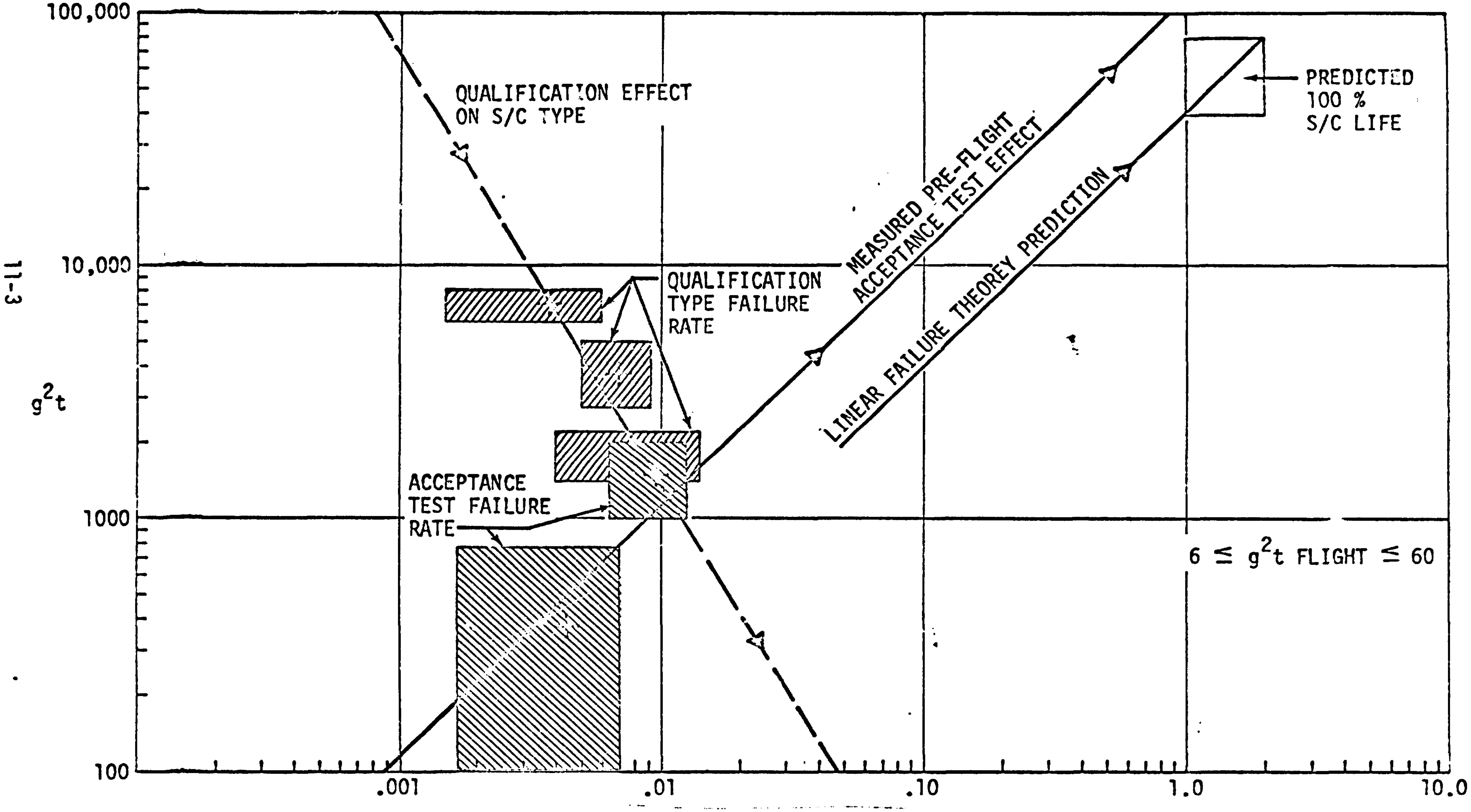


FIG. 11.1 S/C FAILURE RATE VS QUALIFICATION & ACCEPTANCE TEST LEVELS

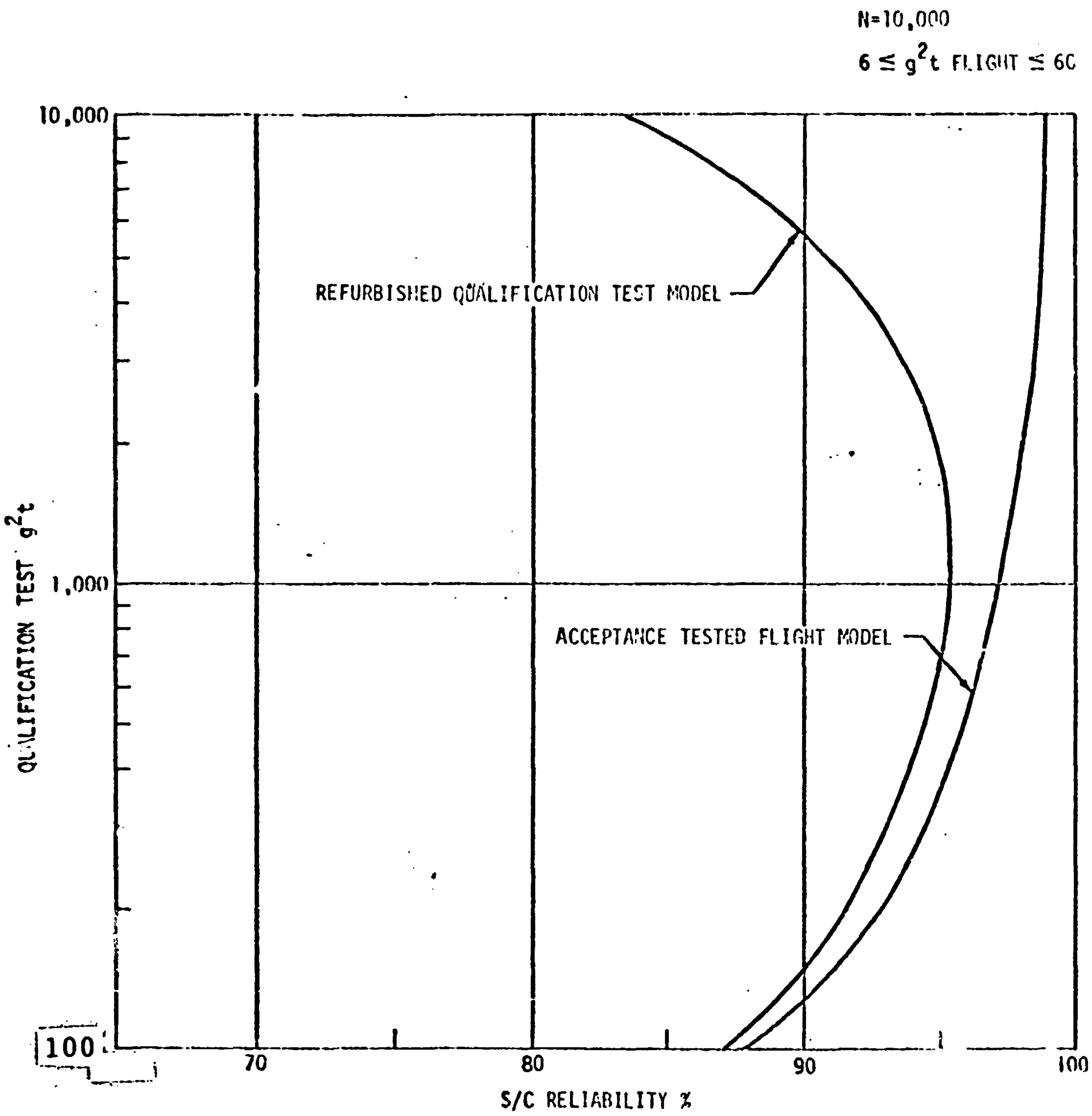


FIGURE 11.2 COMPARISON OF S/C RELIABILITY BETWEEN THE REFURBISHED QUALIFICATION TEST UNIT AND A FLIGHT MODEL (SERIOUS FAILURES)

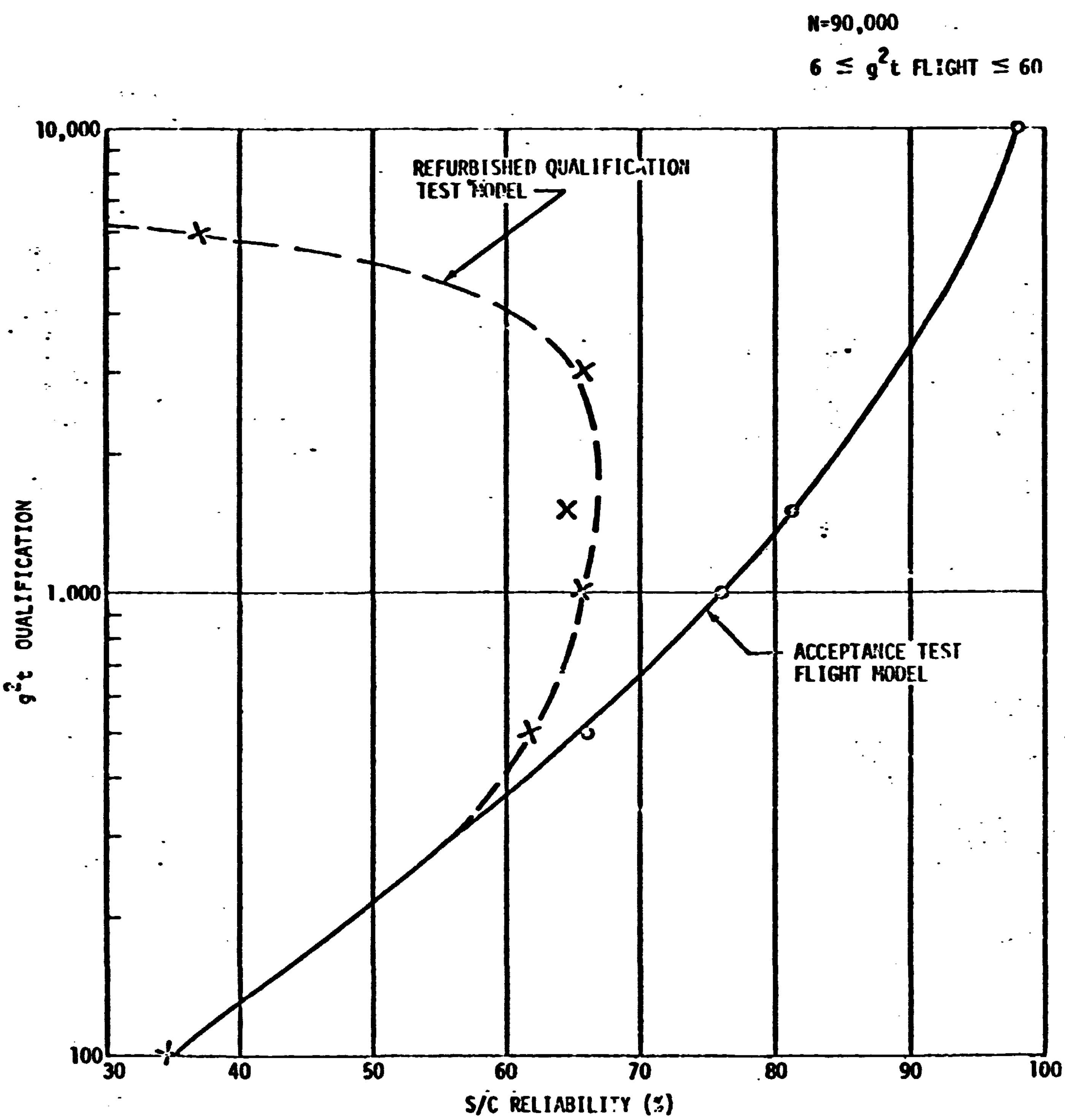


FIGURE 11.3 COMPARISON OF S/C RELIABILITY BETWEEN THE REFURBISHED QUALIFICATION TEST UNIT AND A FLIGHT MODEL (SERIOUS FAILURES)

12.0 Application of Reliability Theory to S/C Failure Data

The analytical and empirical relationships derived withing the previous sections can be used to calculate the probability of a vibration problem occurring during the powered portion of the S/C mission as a function of the qualification test levels and duration, the complexity of the S/C and the severity of the flight environment.

Laboratory development and acceptance test failure rates for both system and subsystem testing have also been presented. In fact enough failure rate date is presented to determine on a purely economical basis, the most efficient type of S/C vibration test program needed, commensurate with the desired reliability, providing that the cost of the launch vehicle, spacecraft, and laboratory rental or operational charges are available.

For example, if a spacecraft with 10,000 components is designed, what will be the required test levels and the expected laboratory failures required to achieve a specified flight reliability assuming a flight g^2t of 6-60.

Using Figures 8.2 and 8.6 and remembering that the expected fialures are equal to λt, these questions can be quickly answered and are tabulated in Table VIII. If, however, the question was directed at the probability of a serious problem occurring then the failure rate used would be 1/4 that used for any vibration problem occurring, this was done and presented in Table IX.

The last tabulation is each table is for the case in which no vibration testing was performed prior to flight. The results for this case indicate that on the average there is a 90% probability of a vibration induced anomaly occurring with a 20% probability that it will be serious.

Tables VIII and IX also show that if the normal acceptance test was only performed ($g^2t \approx 430$) then the probability of a vibration problem occurring would drop to 20% and the probability of a serious problem would be only 5%.

TABLE VIII. REQUIRED QUALIFICATION TEST LEVELS AND EXPECTED LABORATORY FAILURES VERSUS THE PROBABILITY OF THE OCCURRANCE OF A FLIGHT VIBRATION PROBLEM WHEN N = 10,000 COMPONENTS

PROBABILITY OF A VIB. PROBLEM OCCURRING DURING FLIGHT	FLIGHT RELIABILITY	REQUIRED λt	REQUIRED QUALIFICATION g^2t (FROM FIG. 8.6)	EXPECTED QUAL. FAILURES PER S/C	REQUIRED ACCEPT. g^2t	EXPECTED ACCEPT. FAILURES PER S/C
.01	.99	.01	50,000	50	12,500	12.5
.05	.95	.05	4,500	4.5	1,125	1.1
.10	.90	.10	1,500	1.5	375	.33
.20	.80	.225	430	.43	108	.0225
.40	.60	.510	120	.12	30	.03

*(from Figure 8.6)

TABLE IX. REQUIRED TEST LEVELS & EXPECTED LABORATORY FAILURES VERSUS THE PROBABILITY OF A SERIOUS FLIGHT VIBRATION PROBLEM WHEN N = 10,000 COMPONENTS

PROBABILITY OF A SERIOUS VIB. OCCURRING DURING FLIGHT	FLIGHT RELIABILITY	REQUIRED λt	REQUIRED QUALIFICATION g^2t*	EXPECTED QUAL. LAB. FAILURE PER S/C	REQUIRED ACCEPT. TEST g^2t	EXPECT ACCEPT. TEST FAILURES PER S/C
.01	.99	.01	6400	6.4	1600	1.6
.02	.98	.02	2200	2.5	550	.55
.05	.95	.05	520	.5	130	.13
.10	.90	.10	180	.18	45	.045
.20	.80	.225	60	----	----	-----

* (from Figure 8.6)

APPENDICES

A. FLIGHT FAILURE RATE VERSUS ENVIRONMENT TESTING PROCEDURES

B. COMPARISON OF FLIGHT FAILURE RATE WITH G^2T QUAILFICATION LEVELS

C. CALCULATION OF SYSTEM MTBF

D. FLIGHT FAILURES versus ACCEPTANCE TEST LEVELS

APPENDIX A

FLIGHT FAILURE RATE VERSUS ENVIRONMENT TESTING PROCEDURES

TABLE A-1. S/C SUBJECTED TO SYSTEM LEVEL VIBRATION TESTING ONLY:

SPACECRAFT TYPE	PART COUNT x 10^{-3}	NO. OF FLT. S/C	NORMALIZATION FACTOR	PRODUCT	FLIGHT FAILURE	
					PROBABLE	POSSIBLE
2	4.17	3	6	75.06	1	0
3	12	4	6	288	0	0
6	12	1	1	12	0	1
8	5.1	2	6	61.2	0	1
		10		= 436.26	1	2

$N_f/N_o = \frac{1}{436.26} =$ .00229 FAILURES/1000 COMPONENTS

$N_f/N_o = \frac{1}{436.26} =$ .00688 FAILURES/1000 COMPONENTS

TABLE A-2. S/C SUBJECTED TO SYSTEM & SUBSYSTEM VIBRATION TESTING

SPACECRAFT TYPE	PART COUNT x 10^{-3}	NO. OF FLT. S/C	NORMALIZATION FACTOR	PRODUCT	FLIGHT FAILURE	
					PROBABLE	POSSIBL
1	6.9	1	1	6.9	0	0
10	37.2	2	1	74.4	0	0
11	11.5	3	6	207	1	0
15	21.5	5	1	107.5	1	1
16	58	8	1	464	0	3
17	8.9	24	1	213.6	1	0
18	4.8	8	6	230.4	2	3
21	13.8	2	1	27.6	0	0
22	15.7	6	1	94.2	1	2
28	60X	7	1	420	0	0
19	29	1	1	29	0	0
		$\Sigma =$ 67		$\Sigma =$ 1874.6	6	9

$N_f/N_o = \frac{6}{187.4} =$ .0032 FAILURES/1000 COMPONENTS

$N_f/N_o = \frac{16}{1874.6} =$ .00800 FAILURES/1000 COMPONENTS

APPENDIX B - COMPARISON OF FLIGHT FAILURE RATE WITH G^2T QUALIFICATION LEVELS

TABLE B-1

g^2t qualification 1400-2200

S/C	wt/100	N/1000	F.C.F.*	Flight S/C	N (1000)	wt (100#)	Flt. Failures Prob.	Flt. Failures Poss.
1	1.32	5.114	1	1	5.1	1.3	0	1
2	1.50	6.00	1	1	6.0	1.5	0	0
15	8.50	21.50	1	5	107.5	42.5	1	1
19	8.30	29.30	1	1	29.3	8.3	0	0
28	22.0	8.0	1	7	56.0	154.0	0	0
				15	203.9	207.6	1	2

$t\,\lambda_{Wt} = \frac{1 \text{ to } 3}{207.62} = .00482$ to $.0144$ Failures/100#

$t\,\lambda_{N} = \frac{1 \text{ to } 3}{203.9} = .00490$ to $.0147$ Failures/1000 components

TABLE B-2

g^2t qualification 2700-5000

S/C	wt/100	N/1000	F.C.F.*	Flight S/C	N (1000)	wt (100#)	Flt. Failures Prob.	Flt. Failures Poss.
1	3.2	6.9	1	1	6.9	3.2	0	0
2	.86	4.2	6	3	75.6	15.5	1	0
3	4.58	12.0	6	4	288.0	110.0	0	0
6	.83	12.0	6	1	72.0	5.0	0	1
10	5.75	37.8	1	2	75.6	11.5	0	0
21	4.47	13.8	1	2	27.6	8.9	0	0
11	1.72	11.5	6	3	207.0	30.9	1	0
18	2.85	4.8	6	8	230.4	136.8	2	3
23	1.40	9.7	1	1	9.7	1.4	0	0
24	1.40	8.3	1	1	8.3	1.4	1	0
				26	1001.1	324.7	5	4

$t\,\lambda_{(Wt)} = \frac{5 \text{ to } 9}{324.7} = .0154$ to $.0277$ Failures/100#

$t\,\lambda_{(N)} = \frac{5 \text{ to } 9}{1001.1} = .00499$ to $.00899$ Failures/1000 components

*Flight normalization factor to account for X-248 resonant burn condition.

TABLE B-3

g^2t qualification 6500-8000

S/C	wt/100	N/1000	F.C.F.*	Flight S/C	N (1000)	wt (100#)	Flt. Failures Prob.	Flt. Failures Poss.
17	30	8.874	1	24	212.0	720	1	0
16	80	58.00	1	8	464.0	640	0	3
				32	667	1360	1	3

$t\lambda_{Wt} = \frac{1 \text{ to } 4}{1360} = .000732$ to $.00294$ Failures/100#

$t\lambda_{N} = \frac{1 \text{ to } 4}{667} = .00149$ to $.00599$ Failures/1000 components

Appendix C - Calculation of System MTBF

Basic reliability theory coupled with the FARADA vibration environmental K factors has shown that the number of failures N_f is proportional to g^2t. Actual S/C failure data, collected for the qualification, acceptance and flight vibration environments has correlated this as shown in Figure 8.2 where N_f/N_o is seen to be proportional to g^2t.

Now by definition: $\lambda = \frac{N_f}{N_o t}$

From Figure 8.2 $\frac{N_f}{N_o} = g^2 t$

$\therefore \quad \lambda \cong g^2$

From this we see that the reliability of a system $R(t) = e^{-\lambda t}$ can be calculated by either integrating the g^2 vs. t vibration environment or by dividing the g vs. t history into small Δt increments so that $R(t) = + e^{-(\lambda_1 t_1 + \lambda_2 t_2 + \ldots + \lambda_n t_n)}$

If a S/C is subjected to a 5 g_{rms} environment, then the expected number of failures (λt) as a function of time is shown in Table C-1 using the S/C failure rate data of Figure 8.2.

Table C-1 S/C Failures vs. time @ $g_{rms} = 5$

Time (minutes)	g^2t	Expected Failures per 1000Components (N_f/1000N)	$\lambda = \frac{N_f}{N_o\ t}$
1	25	.005	.005
10	250	.05	.005
100	2500	.5	.005
1000	25,000	5.0	.005

$\therefore$ @ 5 g_{rms} $\lambda_5 = .005$ $\frac{\text{Failures}}{\text{Min (1000 components}}$

Since MTBF = $1/\lambda$

then

$$\text{MTBF @ 5 } g_{rms} = \frac{\text{(Min) 1000 Components}}{(.005)N}$$

if N = 1000 then $\text{MTBF} = \frac{1000 \text{ Min}}{1000\ (.005)}$ = 200 Min. = 3.33 Hrs.

N = 100 then $\text{MTBF} = \frac{1000 \text{ Min}}{100\ (.005)}$ = 2000 Min. = 33.3 Hrs.

N = 10 then $\text{MTBF} = \frac{1000 \text{ Min}}{10\ (.005)}$ = 20,000 Min. = 333 Hrs.

Similarly @ 10 g_{rms}:

N	MTBF (HRS)
10	83
100	8.3
1000	.83

APPENDIX D

FLIGHT FAILURES versus ACCEPTANCE TEST LEVELS

g^2t Acceptance Test Levels: 0 to 760

Spacecraft	Acceptance Test g^2t	$Nx10^{-3}$	Flight S/C	F.C.F.*	1x2x3 (Product)	Flight Failure Probable	Flight Failure Possible
8	759	5.1	1	6	30.6	0	1
9	759	5.1	1	6	30.6	0	0
10	461	37.8	2	1	75.6	0	0
15	476	21.5	5	1	107.5	1	1
16	138	58.5	3	1	175	0	1.5
19	525	29.3	1	1	29.3	0	0
21	461	13.8	2	1	27.6	0	0
23	636	9.7	1	1	9.7	0	0
24	636	8.3	1	1	8.3	1	0
25	494	12	1	1	12	0	0
26	494	9	1	1	9	0	1
27	494	5	1	1	5	0	0
28	377	60	7	1	420	0	0
16_B	0	58.5	5	1	292.5	0	1.5
				SUM	1232.7	2	6

$$\text{Failure Rate} = \frac{2}{1232.7} - \frac{8}{1232.7}$$

$$\text{Failure rate} = .00162 \quad \text{to} \quad .00649 \quad \frac{\text{Failures}}{1000 \text{ components}}$$

*Flight Correction Factor used to account for the X-248 resonant burn condition.

g^2t Acceptance Test Levels 1000 - 2000

Spacecraft	Acceptance g^2t	$N \times 10^{-3}$	Flight S/C	F.C.F.	1x2x3 (Product)	Flight Probable	Failure Possible
2	1800	4.17	3	6	75	1	0
1	1000	6.9	1	1	6.9	0	0
3	1000	12	4	6	288	0	0
6	2000	5.2	1	1	5.2	0	0
11	1800	11.5	3	6	207	1	0
18	1600	4.8	8	1	38.4	2	3
			20	SUM	620.5	4	- 4

$$\text{Failure Rate} = \frac{4}{620.5} \quad \text{to} \quad \frac{8}{620.5}$$

$$\text{Failure Rate} = .0064 \quad \text{to} \quad .0128 \quad \frac{\text{Failures}}{\text{1000 components}}$$