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RELIABILITY PREDICTION FOR SPACECRAFT

SoHar Incorporated

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This study provides the basis for improving the utility of Mil-Hdbk-217 for reliability prediction of spacecraft components and systems. The reliability performance histories of 300 satellite vehicles, which were launched between the early 1960's through Jan 84, were reviewed and analyzed during the course of the study. Analysis of over 2500 reports of malfunctions indicated strong evidence of a decreasing failure rate with time in orbit. The cause for the decreasing hazard was found to be traceable primarily to design and environmental causes. In general, however, it was found that current predictions over-estimate the failure rate by at least a factor of two and that the excess of predicted over observed failures increases with time in orbit.

Three methods are provided for spacecraft reliability prediction. In order to account for the decreasing hazard, two of the procedures use a Weibull model with parameters based upon similar spacecraft mission types. A third method uses modifications of Mil-Hdbk-217
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procedures to account for the overestimation of failure rates which result when current procedures are used.

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EXECUTIVE SUMMARY

This study was undertaken to improve the utility of MIL-HDBK-217 for reliability prediction of spacecraft components and systems. As part of this effort over 3,000 reports of anomalous incidents affecting U. S. spacecraft (plus a small number of foreign spacecraft) were analyzed. Slightly over 2,500 of these reports were sufficiently detailed to permit assignment of the failure to a mission time and a specific subsystem, and in approximately 80% of these further analysis was possible to determine the underlying cause of the failure (design, quality, etc.) and the specific part in which the failure originated. The data were obtained from over 300 satellites comprising 96 programs which were launched between the early 1960s through January of 1984.

A primary motivation for this effort were earlier reports that indicated that the hazard (failure rate normalized with respect to the surviving population) decreased with time on orbit. Reliability prediction based on MIL-HDBK-217 assumes an exponential failure law which corresponds to constant hazard. If there is strong evidence that hazard is indeed decreasing this should be taken into account in the reliability model in order to permit realistic predictions and improved allocation of reliability resources.

As shown in Figure 0-1, this study has produced very strong evidence for the existence of a decreasing hazard. The cause for this apparent deviation from conventional reliability experience has been traced to failures due to design and environmental causes. These occur with decreasing frequency with time on orbit, corresponding to the decreasing probability of encountering an environment that is more stressful than a previously encountered one. The classical parts, quality, and operational failures do not deviate significantly from the exponential failure distribution after an initial period dominated by infant mortality. From the distribution of causes of failure, shown in Figure 0-2, it is seen that design and environment together account for about 45% of the failures, and that parts, quality, and unknown causes together account for about an equal percentage. (Chapter 2)

The study found a significant difference in failure rates among subsystems as shown in Figure 0-3 which can be explained in terms of relative complexity. The number of anomalous incidents per spacecraft is higher in post 1977 spaceprograms than in earlier ones, but the severity of failures is significantly less. The increased complexity of recent satellite designs (many of them multi-mission) accounts for the greater number of failures, and the higher redundancy and ruggedness of the subsystems accounts for the lesser severity of incidents. (Chapter 3) Failure rates were affected by the mission type with communication satellites generally having the lowest failure rates and navigation satellites having the highest ones. This seems to reflect the relative maturity of the technologies employed in the satellite design. Orbit altitude did not by itself have a major effect on the failure rate, but orbit dependent equipment selection (e. g., the need for tape recorders on low altitude missions) produced an apparent altitude related effect. (Chapter 4)

Based on these observations a reliability prediction procedure has been developed in which satellite reliability is composed of two factors that account for mission and parts effects, respectively. The general model is

$$R = R_{parts} * R_{mission}$$

where the first factor comprises an exponential reliability prediction based on MIL-HDBK-217 procedures while in the second factor a Weibull model is used to account for the decreasing hazard associated with design and environment failures. This model is validated by a comparison of predicted and demonstrated reliability from two spacecraft programs. The new model will in general predict a higher reliability for long mission durations. Use of this model in trade-offs and design decisions will lead to more realistic assessment of space mission reliability and permit a better allocation of

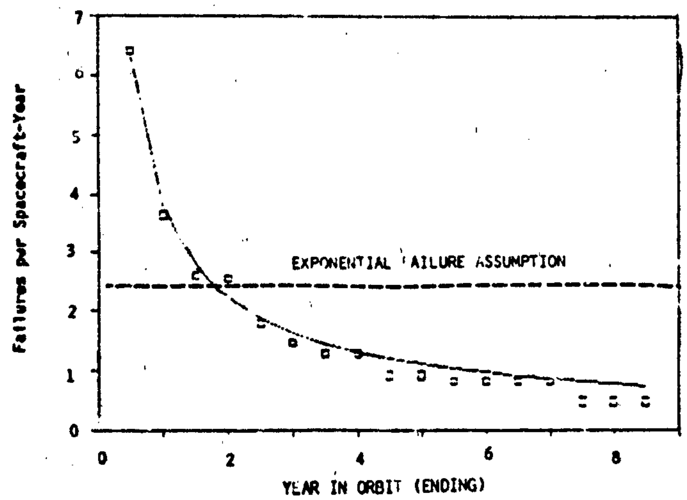


FIGURE 0-1. SUMMARY FAILURE EXPERIENCE IN PRESENT STUDY

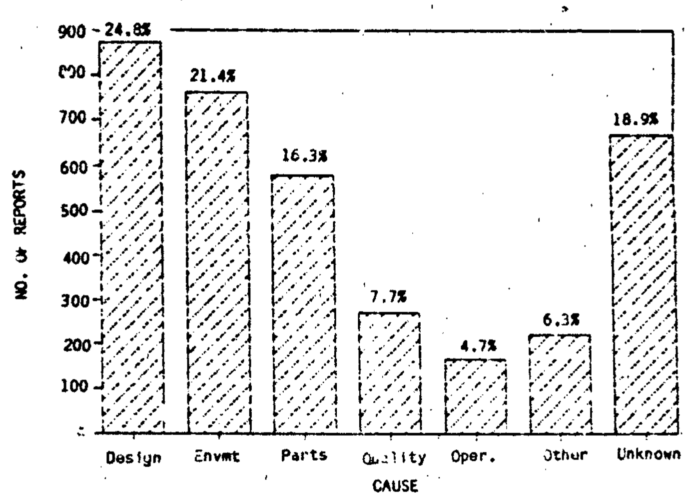


FIGURE 0-2 DISTRIBUTION OF CAUSES

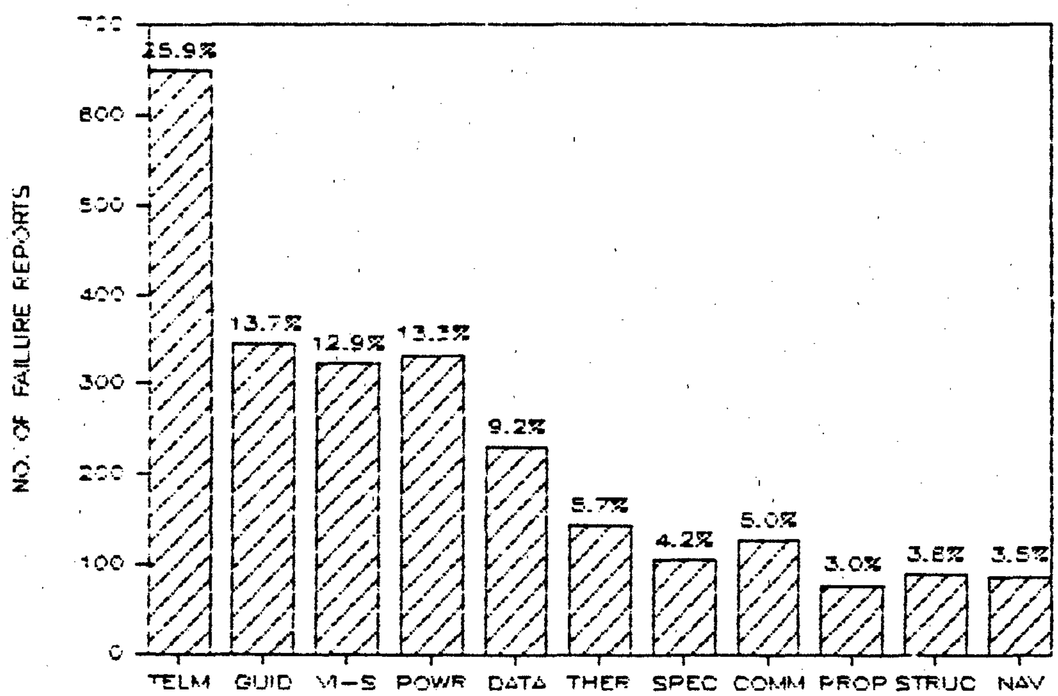


FIGURE 0-3. DISTRIBUTION OF FAILURES BY SUBSYSTEM

resources in satellite design. Alternative models for situations where the parameters of the two-part model cannot be obtained are also provided (Chapter 5)

Two alternate prediction methods are provided. One method is applicable for the subsystem and component designer who needs failure rate information for part selection and reliability design decisions. Although not as accurate as the primary method, the procedure is simple to apply and involves a modification of the space environment factor (S_F) in MIL-HDBK-217 by a factor of .5. Time dependency effects for the failure rate are not directly considered by use of the S_F modifier. However, current MIL-HDBK-217 methods tend to overestimate space environment failure rates and use of the S_F modifier results in overall improvement in predictions. A piecewise exponential model is also provided to account for time dependency effects when the S_F factor is not modified.

The second method is applicable for the mission planner and the spacecraft designer in those cases where the prediction must be based upon similar spacecraft missions and extrapolations to longer mission durations are necessary. A single term Weibull model is used where the beta parameter has been empirically determined to give a workable fit to the observed spacecraft reliability data.

Chapter 1

INTRODUCTION

1.1 OBJECTIVES OF THIS REPORT

Reliability prediction for spacecraft is practiced on three levels

- mission planning and spacecraft specification
- spacecraft design
- spacecraft subsystem and component design

The findings of this report are of interest at all three levels.

For the mission planner is interested in determining the satellite lifetime which results in the lowest cost per year, and the prediction of the failure rates is obviously an important input to that analysis. The time dependence of failure rates investigated in Chapter 2 and historical satellite reliability trends discussed in Chapter 3 respond to that need. Also, gross mission failure rates and the effect of subsystem and orbit parameters on component reliability which are discussed in Chapter 4 will be of importance at that level. The single term Weibull reliability prediction model described in Section 5.3.2 is particularly suited for mission planning.

The spacecraft designer is faced with the need for determining fault tolerance and redundancy requirements for subsystems and major components. Predicted subsystem failure rates discussed in Chapter 4 are the major data input to these decisions. The spacecraft designer must also provide environmental protection for the equipment, select duty cycles for some of the spacecraft functions, and must plan for testing of the satellite as a whole as well as for its components. The analysis of causes of spacecraft failures presented in Chapter 3 will be helpful in these decisions. The reliability prediction procedures of Chapter 5 address a direct need of the spacecraft designer.

The subsystem and component designer needs failure rate information for parts selection and internal redundancy decisions. The reliability prediction procedures found in Chapter 5 are applicable to this environment, and the piecewise exponential model described in Section 5.3.1 may be particularly suitable. Causes of component failures discussed in Chapter 3 and detailed analyses presented in Chapter 4 are also pertinent to the design decisions made at this level.

1.2 OVERVIEW OF METHODOLOGY

Selected data on satellite failures were transcribed from existing data bases (see following section) into a dedicated data base for this study which contained for each incident

- Satellite Program
- Flight Number
- Month and Year of Launch
- Failure Time (in months on orbit)
- Severity Classification
- Cause of Failure (up to three classifications)
- Subsystem Affected

- Part Affected (where applicable)

Subsets of the data base could be extracted for any combination of logical and quantitative conditions. The estimates of quantitative parameters presented in the body of the report were in most cases derived by multivariate regression. Tests of hypotheses were used to support qualitative findings, such as distinctions between contributions to the failure rate by various causes. Statistical aspects of the methodology are discussed in Appendix A.

1.3 DATA SOURCES AND OTHER CONTRIBUTIONS

The two major data sources utilized in this study were the Orbital Data Analysis Program (ODAP) at The Aerospace Corporation and the On-Orbit Spacecraft Reliability (OOSR) data compiled by Planning Research Corporation for NASA.

The study started with an ODAP compilation as of December 1982, received a major update in June of 1983, and was finally brought up to date as of July 31, 1984 at which time most failures that had occurred during 1983 and a few later ones had been captured. Dr. Max Weiss, Dr. F. D. Maxwell, and Mr. Jay Leary were particularly helpful in furnishing this material and associated documents, and by critqueing preliminary findings that were discussed with them.

The OOSR study was completed in January 1983 and no updates were obtained during the conduct of the effort reported on here. Mr. Bloomquist and Ms. Graham were generous of their time in explaining their methodology and in permitting us access to original files to explore details that were not available in the published documents.

Further details on the data bases are presented in Appendix B.

Mr. Myron Lipow and Mr. Sam Lehr of TRW were very helpful by discussing their methodology for spacecraft reliability prediction and by furnishing data utilized in that process.

The RADC Project Engineer for the study, Mr. Eugene Fiorentino, provided much constructive guidance throughout the investigation. His review of the draft of this report helped us to provide needed clarifications and to avoid inconsistencies. The formulation of the simplified exponential approximation for reliability prediction in Section 5.3.1 is due to his suggestion.

We want to express our gratitude for this assistance while at the same time asserting that the conclusions presented here are exclusively the responsibility of the authors.

Chapter 2

TIME DEPENDENCE OF THE FAILURE RATE

Standard methods of reliability prediction, including those described in MIL-STD-756 and MIL-HDBK-217, are based on an exponential failure rate assumption. This implies that the probability of failure over some fixed finite time interval among the survivors at the beginning of that interval is constant and independent of prior service. Because of this characteristic the exponential failure distribution is sometimes called "the distribution without memory". The exponential failure rate assumption has been found consistent with experience in many terrestrial electronic applications, and it leads to mathematically tractable reliability models. It has therefore also been adopted for spacecraft reliability prediction. However, for a number of years there has been evidence that space applications experience a decreasing hazard, and the data collected in the present effort confirm this finding.

This chapter first synopsizes prior investigations into the decreasing hazard phenomenon, then presents the results of the current investigation and analyzes the possible processes that can cause a decreasing hazard, and finally it discusses the implications of the decreasing hazard for spacecraft reliability prediction.

2.1 Historical Perspective

Early evidence of decreasing hazard can be found in a study of satellite failures during the decade ending 1970 sponsored by the Navy Space Systems Office [BEAN71]. A particularly significant illustration from that report is reproduced in Figure 2-1. It is seen that the number of anomalous incidents decreases much faster than the number of (operational) spacecraft in the sample. A quantitative analysis of these data for successive 10,000 hour periods is presented in Table 2-1.

TABLE 2 - 1 DECREASING HAZARD IN EARLY SPACECRAFT

Period ending (hours)	Avg. Failures per 1000 hrs.	Avg. Operat. Spacecraft	Hazard (see note)
10,000	74	96	0.77
20,000	12	48	0.25
30,000	3	22	0.14

Note: Hazard is expressed as number of failures per 1000 operating spacecraft-hours

Investigation of this phenomenon was not a specific objective of the referenced report and it is not further commented on (Table 2-1 was compiled as part of the current investigation). However, a few years later researchers at NASA Goddard addressed the constant hazard assumption and found that "it does not occur until 90 (or probably more) days in space" [TIMM75]. That study also introduced normalized failure rates (dividing the observed failures during a given period by the number of spacecraft contributing to the observations). This technique is continued in the present investigation and the term failure ratio is used for the failure rate that is normalized in this manner. The failure ratio is used as an approximation for the hazard (for definitions of hazard see [LLOY77, p. 135] or [VANA64, p. 61]; 'hazard function' or the shorter 'hazard' used in the former reference seems preferable to 'hazard rate' used in the latter).

The normalized malfunction rate computed in the NASA Goddard study is illustrated in Figure 2-2. The definitions used in connection with this figure are

Failure	the loss of operation of any function, part, component, or subsystem, whether or not redundancy permitted recovery of operation
Problem	any substandard performance or partial loss of function which is not sufficient to be classed as a failure

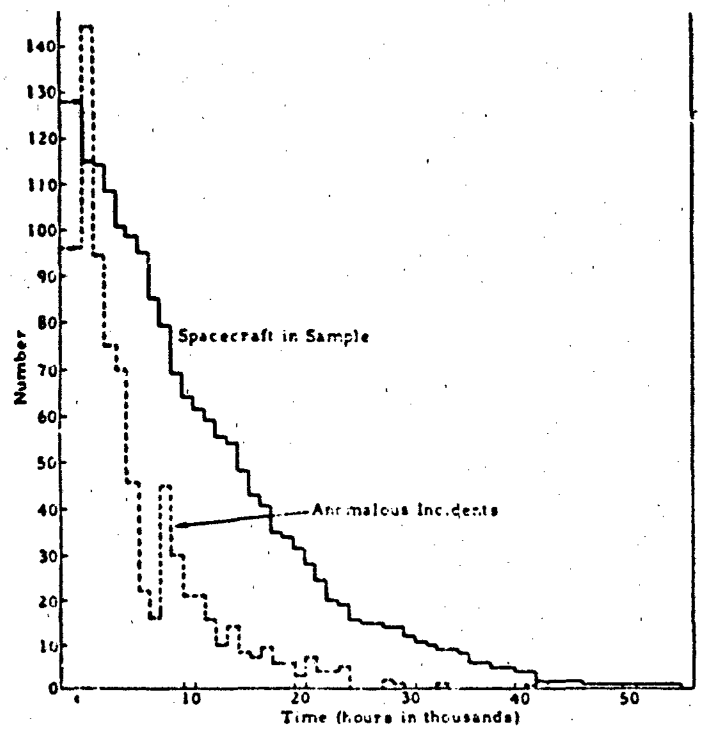


FIGURE 2-1. NUMBER OF SPACECRAFT UNDER OBSERVANCE AND NUMBER OF INCIDENTS

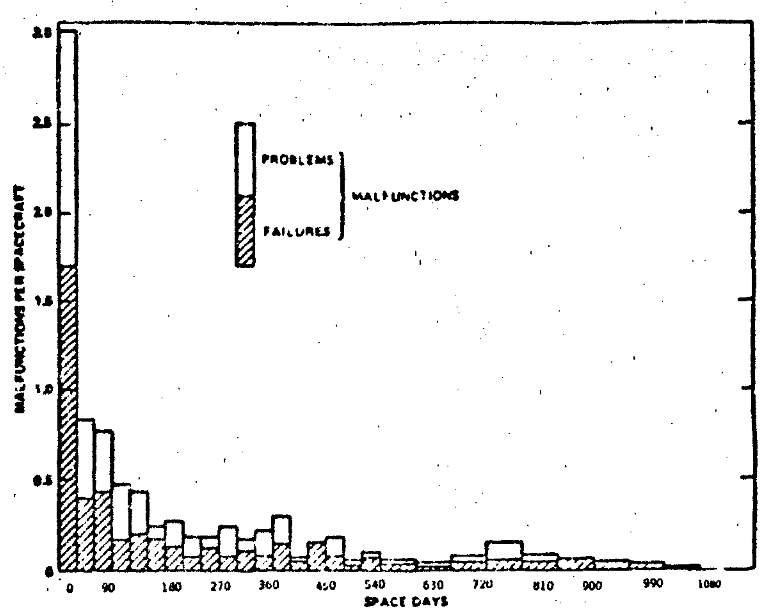


FIGURE 2-2. TIME DISTRIBUTION OF NORMALIZED SPACE MALFUNCTIONS FROM 57 GSFC SPACECRAFT

In a later publication [NORR76] the same research group fitted Duane and Weibull models to their results and found a decreasing hazard function over the entire time span covered by the data (roughly three years). They also found a very good fit to the Duane model at the component failure level as shown in Figure 2-3 (examples of components are tape recorders and transmitters). Excess failures observed during the very early life, specifically during the first 30 days on orbit, were found to be related to inadequacies of spacecraft and component testing. No other explanation for the decreasing hazard is offered in these reports.

The NASA Goddard studies as well as all others discussed in this chapter counted as a malfunction any observation of nonconforming behavior, whether it occurred in a spare or in an active unit. Therefore, the entire spacecraft equipment can be modeled as being in a series configuration for evaluating the failure rate. If the exponential failure law applies at the component or lower level, the total failures observed should therefore also follow the exponential distribution.

An update of the Navy Space Systems study prepared in 1978 showed further evidence of decreasing hazard [BLOO78]. That report includes many spacecraft with lifetimes in excess of three years, and further decreases in hazard are implied for these. Excerpts from Exhibit 3 of the reference are shown in Table 2-2. Each row summarizes the data for the first 10 spacecraft that exceed the lifetime shown in the first column; in most cases the longest lifetime included is within 2,000 hours of the threshold. The hazard is an average value because the reference does not provide incremental data.

TABLE 2 - 2 HAZARD EXPERIENCE IN 1978 REPORT

Spacecraft Life (Hours)	Hazard Failures per 1000 Spacecraft-Hours
4,000	1.20
8,000	0.60
16,000	0.48
32,000	0.27

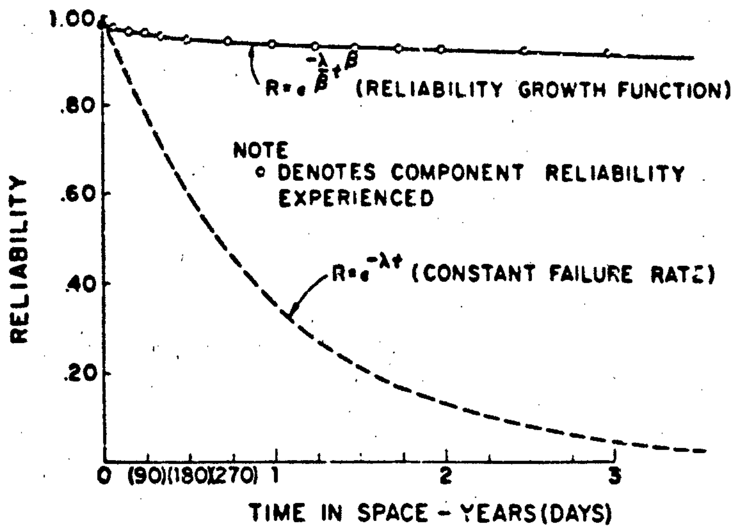


FIGURE 2-3. GSFC COMPONENT RELIABILITY IN SPACE

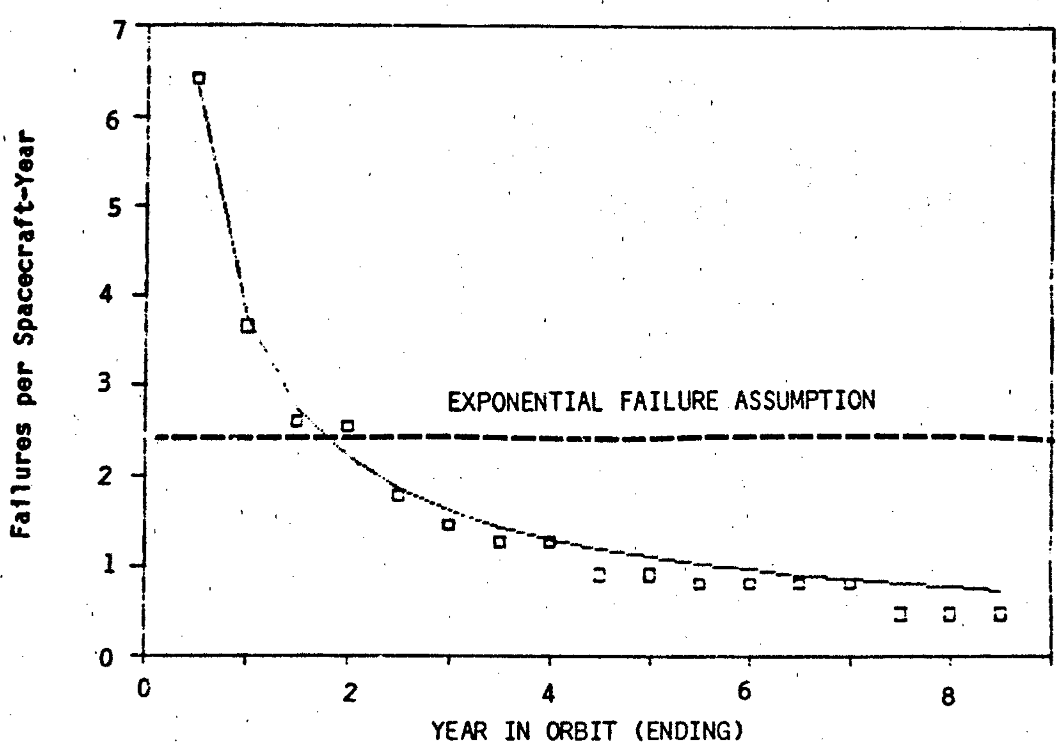


FIGURE 2-4. SUMMARY FAILURE EXPERIENCE IN PRESENT STUDY

At the time that the report became available a number of Air Fore satellite programs that could benefit from long mission times (e. g., communication and navigation programs) were in the early implementation phase. It was realized that these satellites could be designed in a more economical manner if advantage were taken of the lower hazard at prolonged on-orbit periods but because no clear cause for the decreasing hazard phenomenon could be identified it was decided to stay with the exponential failure rate assumption as a "conservative" approximation of the true reliability function. However, a technical need for improved knowledge of spacecraft electronic failure rates was recognized, and this need is addressed in the present study. The findings and analysis of this part of the study will be found in the immediately following sections. The implications of the decreasing hazard for various aspects of spacecraft reliability prediction are discussed in the final section of this chapter.

2.2 Decreasing Hazard Findings

As shown in Figure 2-4 the failure ratio (defined as an approximation of hazard in the previous section) decreases throughout the satellite life with the greatest decrease during the first three years. During the second year the failure ratio is approximately one-half of the average for the first year (and slightly over one-third of the average for the first six months). At the end of the third year it has decreased to about one-third of its average value during the first year, and at the end of eight years it is down to about one-tenth of the failure ratio during the first six months. This has very significant implications on the mission planning and redundancy provisions as shown in the last section of this chapter. However, before this finding can be accepted at face value a number of possible objections must be resolved. Two factors may cause the observed failure ratio to decrease while the true failure ratio remains constant

- shadowing and

- decreased user interest or funding

Shadowing designates the loss of observability for parts that are associated with a failed component. As an example of this process consider a failure in a tape recorder or multiplexor, components for which most satellites carry spares. As soon as a disabling failure in the primary or active unit occurs, it is switched out and the spare unit is activated. Because no further use is made of the original unit, subsequent parts failures will not be detected. Even more significant can be the termination of an entire mission package, such as the cessation of all optical weather observations when a vidicon fails. There is no doubt that the reports used in Figure 2-4 are affected by shadowing but it is not believed to account for a significant part of the decrease in hazard because

- The data presented in Figure 2-3, which are on a component basis and therefore not subject to shadowing. The source for these data computed a Weibull shape parameter (b in the notation used in the present report) of 0.311, indicative of a decreasing hazard

- Failures of severity that disable components but not an entire subsystem (severity classifications 2 and 3) occur at the rate of approximately 0.5 per spacecraft-year between the second and eighth year on orbit. The average component population under observation is at least 65 (this figure is given in [NORR76] for the comparatively simple satellites launched prior to 1970). Thus, the decrease in hazard accounted for by shadowing is less than 1% per year whereas the decrease in hazard shown in Figure 2-4 is over 10% per year between the second and eighth year. Failures that disable a major subsystem but not the entire satellite occur at a rate less than 0.1 per year. Assuming that such a failure will remove five components from observation, the effect is comparable to or less than that due to component failures.

The lack of interest or funding as a satellite operates past the initially planned period may cause failures to go undetected or unreported, thereby creating the illusion of a decreasing hazard. The lower rate of reporting is especially likely to affect minor discrepancies, transient failures, and conditions which could be easily corrected by operational procedures. The ratio of minor malfunctions reported to the total failures is therefore expected to decrease if there is systematic underreporting of the former for longer mission durations. There is some evidence of this effect in the data as discussed below.

The database used in this study classified criticality as follows:

1. Mission critical

2. Single point failure (affecting a major subsystem)

3. Redundant unit

4. Work around

5. Degraded performance

6. Temporary

7. All others

Classifications 4 - 7 are in the following grouped together as low criticality failures. The observed ratio of low criticality failures to all failures shown in Figure 2-5 is almost constant for the first five years on orbit and exhibits a slightly decreasing trend thereafter at the rate of about 3% per year. Since failures of low criticality comprise initially somewhat less than two-thirds of the total, this effect translates to underreporting at a 2% per year rate for the total population. This can account for some but by no means all of the decreasing hazard observed after the first five years on orbit.

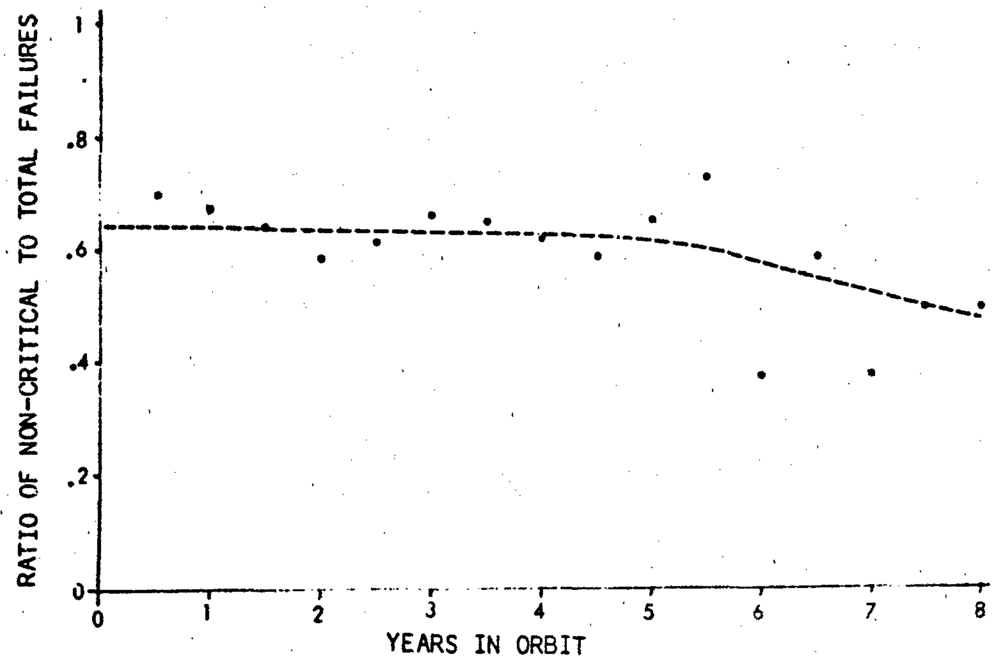


FIGURE 2-5. DECREASING REPORTING TREND

A final reservation about acceptance of decreasing hazard arises from the incompatibility of such a characteristic with the established and observed failure patterns of electronic parts. It will shortly be seen, however, that conventional parts failures account for only a fraction of the total failures that affect spacecraft in orbit, and that other causes of failure are compatible with a decreasing hazard.

That many spacecraft systems employ redundancy does not affect the conclusions presented here since failures in all equipments (active and standby) were monitored and reported. As far a failure reporting is concerned, a simple series model of all equipments can therefore be assumed.

Weibull hazard plots were fitted to the observed failure ratios by a least squares method that has been in use for many years [KAO56]. The form of the

Weibull hazard function used here is

$$Z(t) = b*t^{b-1}/a$$

where b (beta is used in most texts) is the shape parameter and a (or alpha) is the scale parameter. The corresponding reliability function is

$$R(t) = \exp(-t^b/a)$$

The best fit for the total failure population is obtained for a = 255 hours and b = 0.28 and the curve shown in Figure 2-4 represents this relation. The methodology for fitting a Weibull hazard function to the failure ratio data is also applied to subsets of the failure data. The fits are not always as good as that discussed above. The reason is not only that the subsets have smaller populations and that greater dispersions therefore have to be expected but also that some failure processes seem to follow another distribution. Nevertheless, the Weibull fit was used as a standard procedure because

- it fitted the majority of the failure populations quite well
- it is widely used in other reliability prediction literature
- the Weibull parameters permit a concise quantitative comparison of individual populations.

In some practical applications of reliability prediction other mathematical representations of the time dependence of the failure ratio may be preferable, and alternative procedures discussed in Chapter 5 address that need.

2.3 Basis for the Decreasing Hazard

The most likely sources of the decreasing hazard observed for the overall failure population are failures due to design and environmental causes. To explore this important issue it will be necessary to examine causes of failures briefly here, while a more detailed discussion, including definitions of the categories, is deferred until Chapter 3.

Causes of failure were grouped under seven major headings:

- Design
- Environment
- Parts
- Quality
- Operational
- Other known causes
- Unknown

The distribution of failures among these classifications is shown in Figure 2-6. Failures caused by design show a consistently decreasing failure ratio as illustrated in Figure 2-7. Note particularly that the failure ratio for the eighth year and later is less than 5% of that observed during the first six months, and that it is approximately one-half of that reported at the end of the fifth year on orbit. The failure ratio for environmental causes, shown in figure 2-8, exhibits approximately similar tendencies, though the dispersions are greater.

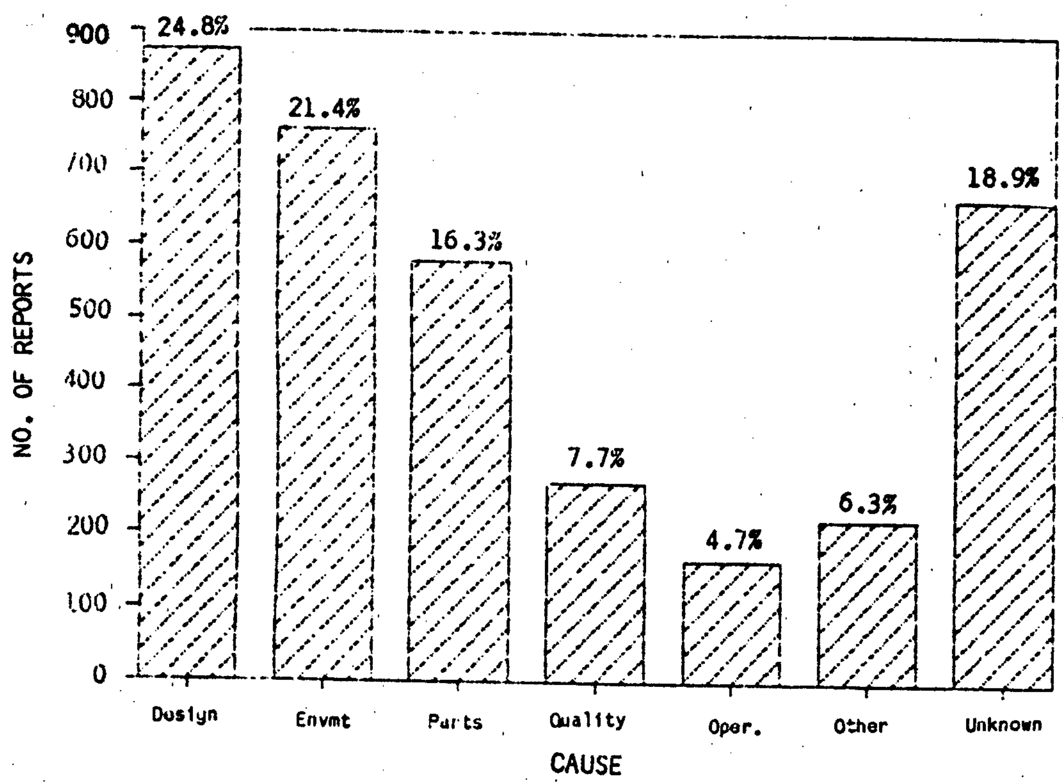


FIGURE 2-6. DISTRIBUTION OF CAUSES

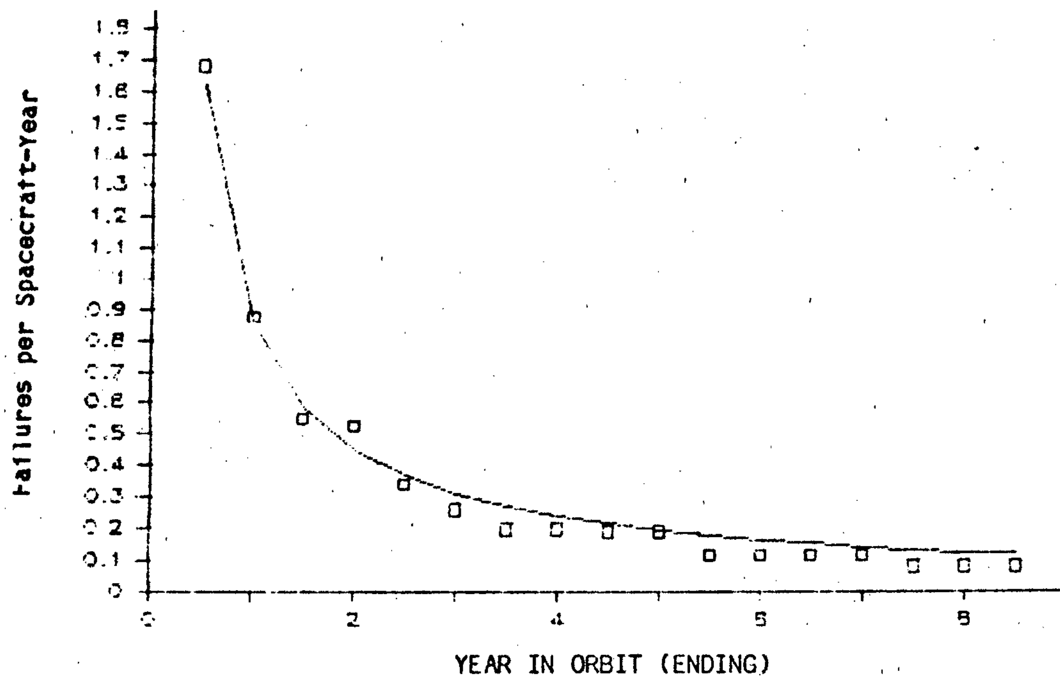


FIGURE 2-7. TIME TREND OF FAILURES CAUSED BY DESIGN

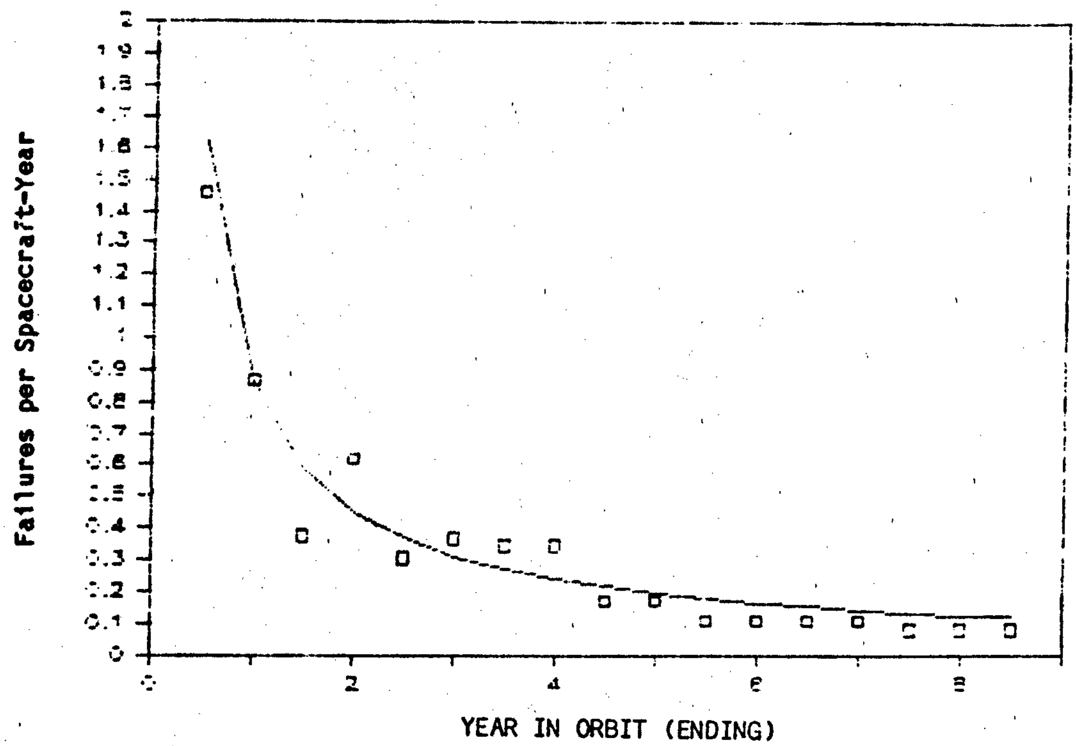


FIGURE 2-8. TIME TREND OF FAILURES CAUSED BY ENVIRONMENT

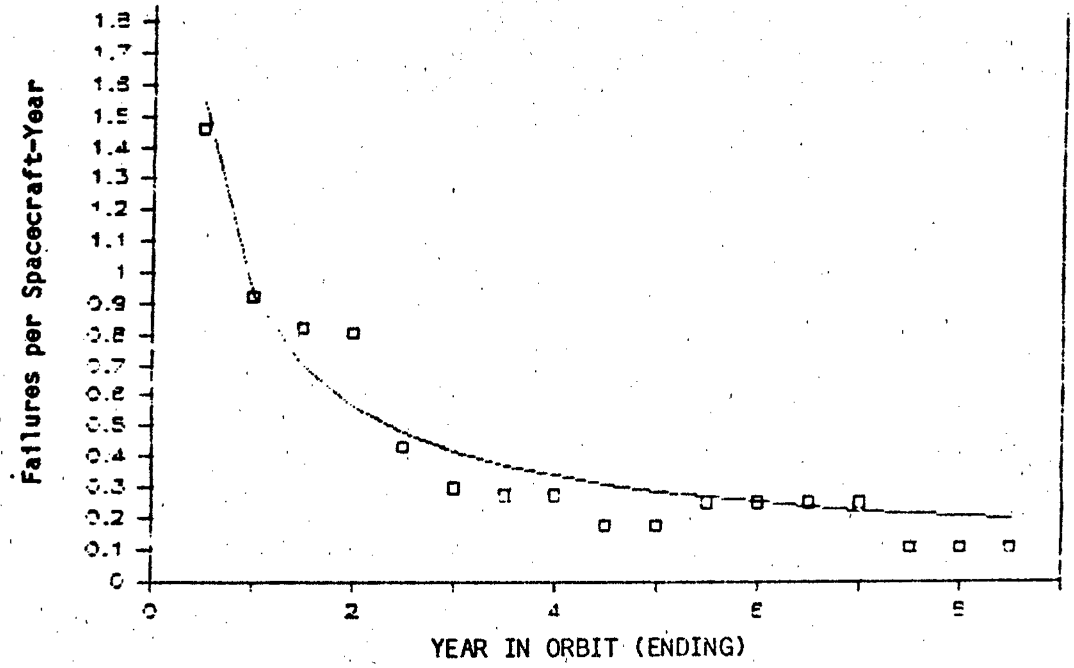


FIGURE 2-9. TIME TREND OF FAILURES CAUSED BY PARTS + QUALITY

In constrast, the failure ratio associated with parts and quality causes, shown in Figure 2-9, provides only a small decrease beyond the end of the third year. The rate of this decrease is only slightly more than can be accounted for by the shadowing and loss of interest effects discussed in the previous section. Thus, for failures attributable to parts there does indeed appear to be a constant hazard region after an initial period of sharply decreasing failures. The nature of that initial period is discussed later in this section. Failures due to operational causes come closest to a constant hazard of all the categories considered here. These are illustrated in Figure 2-10. Miscellaneous other known causes follow a similar pattern.

Failures due to unknown causes, illustrated in Figure 2-11, show an overall hazard pattern that is consistent with that found for parts and quality causes, but there is evidence of a continuing decrease through the eighth year. As demonstrated in Figure 2-8, unknown causes are a significant contributor to the total failure ratio. The shape of the failure ratio plot suggests that there is a greater fraction of parts related failures in that category than design related failures. The Weibull coefficients for the total failure population and for individual cause classifications are shown in Table 2-3. The parameter designated a is a scale factor, similar to MTBF for the exponential distribution. The b parameter is the shape factor which determines whether there is a decreasing, constant, or increasing hazard. Values of b less than unity correspond to a decreasing hazard, while the exponential distribution can be represented as a special case of the Weibull with b = 1. As is seen in the table, design and environment show the sharpest deviation from the constant hazard condition, while operational and other known failures show the closest approximation to it.

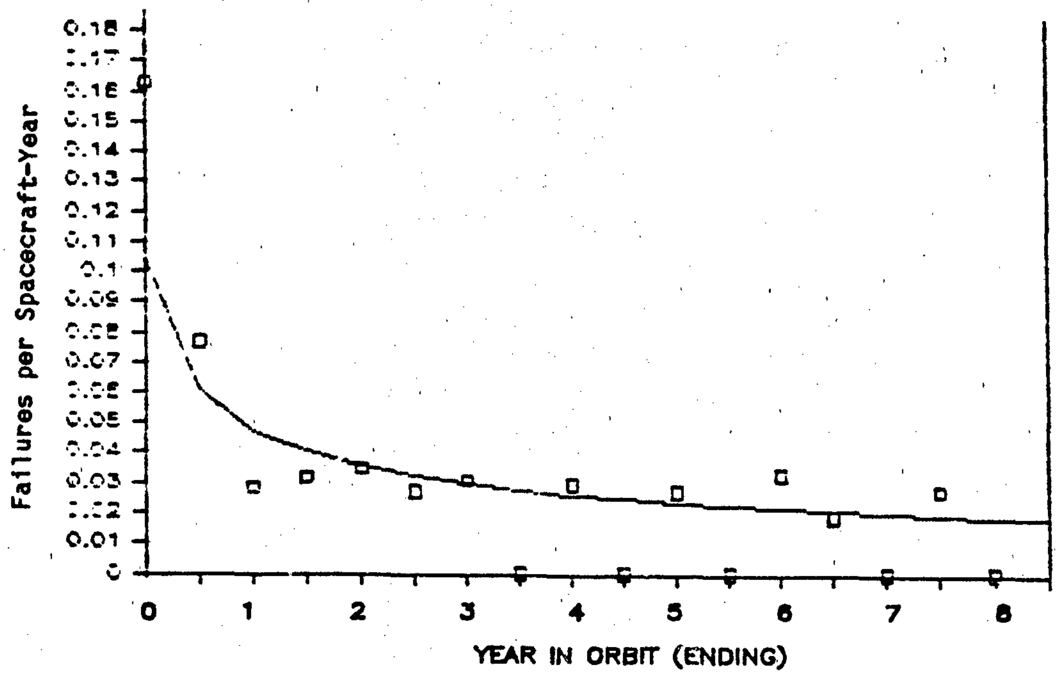


FIGURE 2-10. TIME TREND OF FAILURES CAUSED BY OPERATION

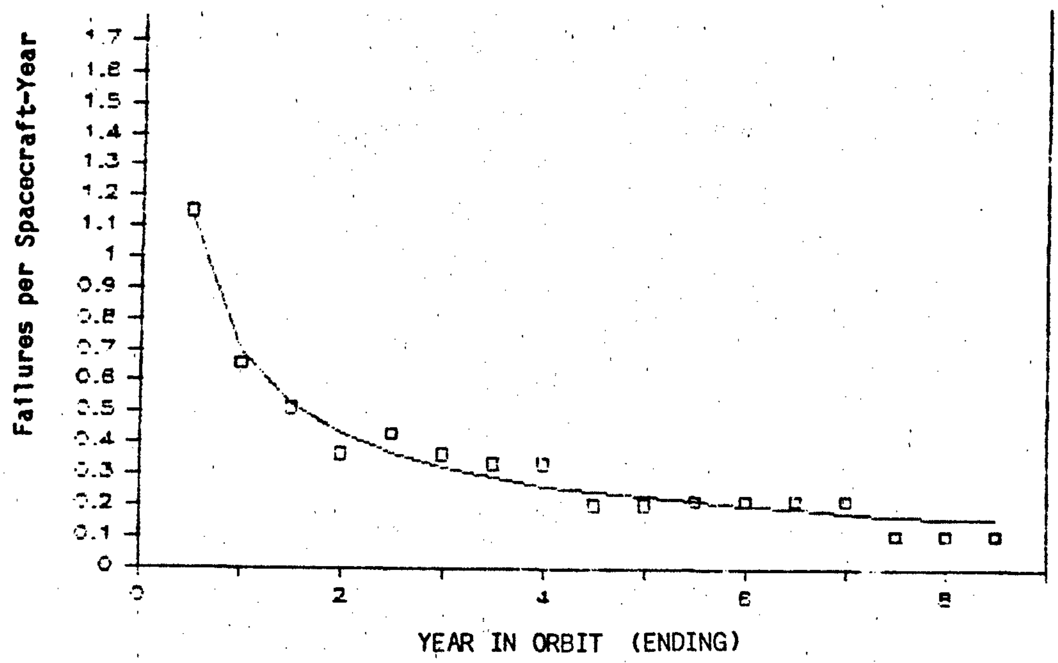


FIGURE 2-11. TIME TREND OF FAILURES DUE TO UNKNOWN CAUSES

TABLE 2 - 3 WEIBULL PARAMETERS BY CAUSES OF FAILURE

Cause Classification	Weibull Parameters a (10^6 Hrs)	b
All causes	0.000255	0.28
Design	0.000036	0.06
Environment	0.000047	0.07
Parts and Quality	0.001035	0.28
Operational	0.113796	0.51
Other known	0.115081	0.57
Unknown	0.002156	0.32

The findings presented thus far have identified failures due to design and environmental causes as the most significant factor in producing a decreasing hazard beyond the initial break-in period of the satellites. However, this runs counter to the conventional assumption that design failures will become manifest very early in the operational life of a component and that a period of successful operation of several years should virtually preclude that any further design failures will occur.

For an understanding of this phenomenon it is instructive to turn to the stress-strength concept of reliability that was initially developed for mechanical structures like bridges [FREU45] but has also been found applicable to electronic and electromechanical equipment [LUSS57, KECE64]. The basic relationships for determining the failure probability according to this approach are shown in Figure 2-12. The upper part of the figure illustrates the relation between a constant load and variable strength, such as might apply to the failure probability (due to dielectric breakdown) of a capacitor connected across the output of a constant voltage power supply. The dielectric strength of the capacitors is assumed to be a random variable whose distribution is determined by the material and process attributes. By standard design practices the average value of the strength is placed well above the deterministic level of the applied load (the rated output voltage of the power supply). Due to the variable nature of the strength a small fraction of the product, given by the value of the strength distribution at x_L, will fail. These failures will occur almost immediately after the power

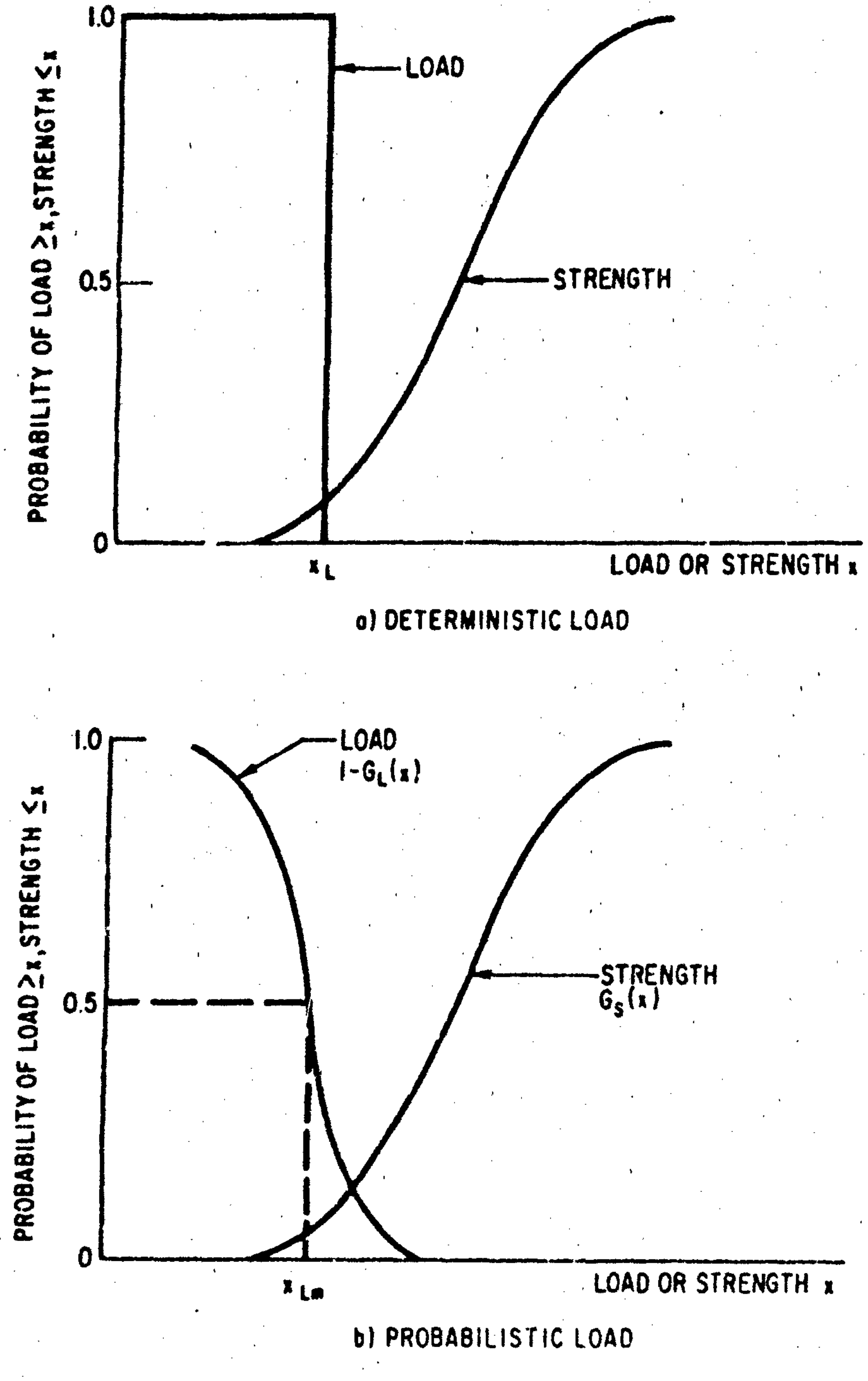


FIGURE 2 - 12 STRESS-STRENGTH RELIABILITY MODEL

supplies that incorporate the low strength capacitors have been placed into use.

The lower part of the figure represents the case when both the load and the strength are variable. The load curve represents the probability that the load will exceed the abscissa value x, whereas the strength curve, as before, is the probability that the strength will exceed the value of x. This illustration will apply where the capacitor is placed across an unregulated power supply, the output voltage of which varies as a function of the line voltage and of load fluctuations. Although the average value of load is the same as in the previous example, it is intuitively seen that a greater fraction of the product will fail. The value of the failure probability in this case must be computed by a convolution integral [PAPO65] but this procedure is not necessary for the understanding of the long term decreasing failure rate. Instead, the focus is on the time of occurrence of the failures

Returning to the example of the power supply capacitor, the <u>initial</u> failure rate for the unregulated supply may not differ markedly from that of the regulated supply. However, whereas in the former case no failures were expected after the initial period, there is clearly a mechanism for continuing occurrence of failures under variable load. The probability that the output voltage will exceed some value, y, above the nominal level during the first hour of operation may be extremely small but the probability of that value being exceeded over a period of one year will certainly be greater. The capacitors with dielectric strength between the nominal output voltage and y will fail when that exceedance occurs, and therefore failures must be expected during the entire period of operation.

The investigation of the occurrence of unusually large or small values of a random variable was pioneered by E. J. Gumbel and is called statistics of extremes after the title of his definitive work in that field [GUMB58]. It deals with phenomena for which no firm upper (or lower) limit can be established, such as the discharge volume of a river (an early application was in the investigation of floods), the lifespan of man, or, of particular interest to spacecraft failures, the intensity of magnetic fields caused by solar storms. A very terse description of the central problem of the statistics of extremes was made before the discipline became established; "However big floods get, there will always be a bigger one coming" [PRES50]. In terms of spacecraft reliability, that the equipment has survived under the environmental stresses experienced during a period of m years on orbit does not preclude the occurrence of a phenomenon during year m + 1 that produces a greater stress and hence leads to failure. However, the likelihood that greater stresses will be encountered decreases over successive intervals, and that leads to the decreasing hazard. A brief numerical exposure to the methodology is presented below.

The probability density of the largest value of n = 1..10 samples drawn from a standardized normal distribution is shown in Figure 2-13 which is taken from Gumbel's book. For n = 1 the density of the sample is of course equal to that of the parent distribution. For a sample of two, the mode for the largest value is approximately 0.5 standard deviations above the mean of parent distribution, but then it takes a sample size of 5 to move the mode to 1 standard deviation above the parent mean, and even at n = 10 it is only at 1.3 standard deviations. (This discussion has centered on the mode, the highest point on each of the curves, because it is the easiest characteristic to point out; except for n = 1, the mean and median of the extreme value distribution are not exactly equal to the mode.)

In terms of spacecraft reliability, each year of operation can be equated to one observation on the basis that many of the stresses are seasonal (other interpretations are of course also possible). Table 2-4 lists the probability of exceeding a previously observed stress level during a given year on orbit under the above assumption. The data are based on median values of the extremes for normal variates taken from Graph 4.2.2(2) in [GUMB58].

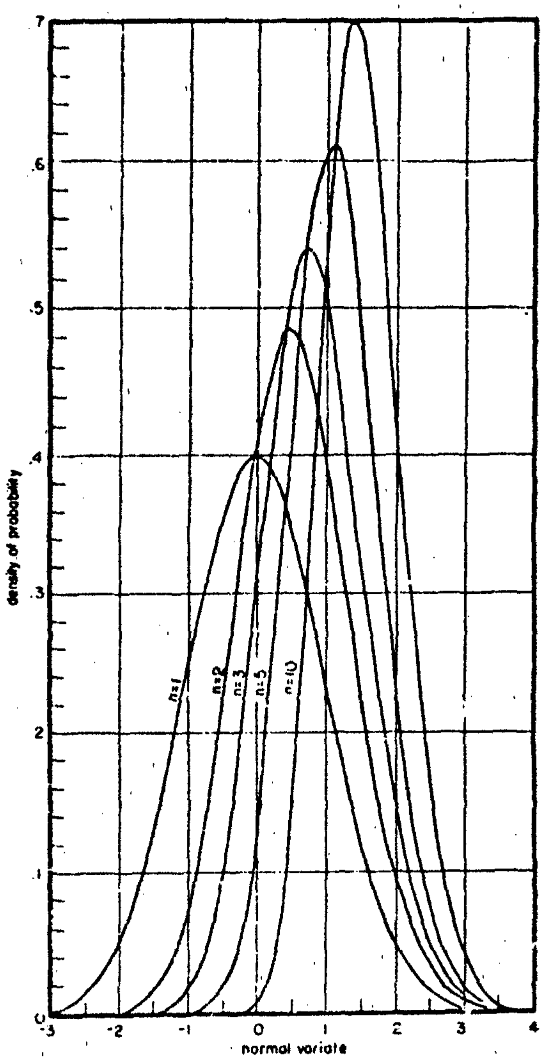


FIGURE 2 - 13 PROBABILITY DENSITY OF NORMAL EXTREMES

TABLE 2 - 4 PROBABILITY OF EXCEEDING A PREVIOUSLY OBSERVED VALUE

No. of Years	Median Extr. Value*	Increment over Prev. Val*	Increment/ Year*
2	0.5	0.5	0.5
4	1.0	0.5	0.25
6	1.25	0.25	0.125
8	1.4	0.15	0.075
10	1.5	0.1	0.05

* in multiples of standard deviations

The increment values are plotted together with the time trend of environmentally caused failures (Fig. 2-8) in Figure 2-14.

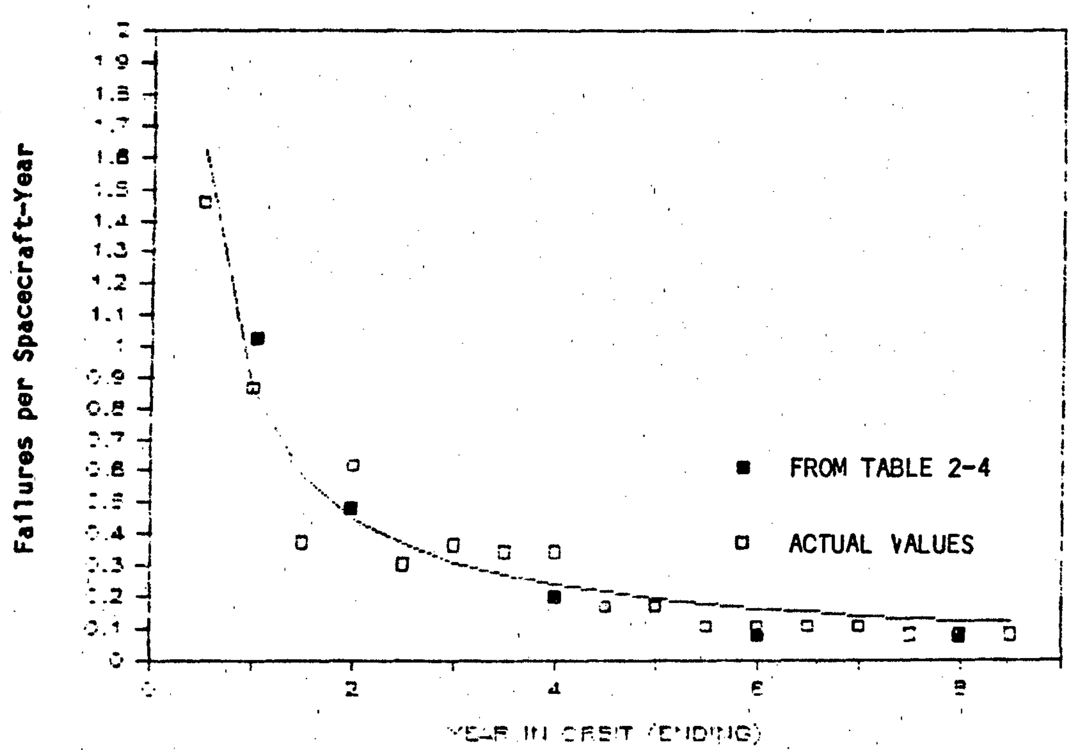


FIGURE 2 - 14 EXTREME VALUES FITTED TO ENVIRONMENTAL FAILURES

For this comparison is was assumed that the median of the normal distribution from which the extreme values were derived was a stress that caused 1 failure per spacecraft-year and that each exceedance of this stress by one standard deviation also caused 1 failure per spacecraft year. Both of these assignments are arbitrary and were made to produce a reasonable fit to the curve in a simple manner (better results could have been obtained by curve fitting techniques). That exceedance of the stress by one standard deviation causes 1 failure per spacecraft year can be interpreted in two ways

- Spacecraft equipment strength is uniformly distributed so that for each unit increase in stress the same fraction of failures will be encountered

- Spacecraft equipment strength is normally distributed, and the normal distribution from which the extreme values were drawn was obtained as the convolution of a normally distributed environment variable and the normally distributed strength variable (the probability of failure of a given system is under these conditions normally distributed, and the probability of system failure over a number of years or for a number of systems will follow the extreme value distribution).

This brief excursion into the field of statistics of extremes has thus provided a rationale for experiencing a long term decreasing hazard for failures associated with the intensity of natural phenomena.

It remains to be explained why the time trends for failures due to parts and quality causes, for which a constant hazard is postulated in the out years still shows a pronounced decreasing trend during the initial two years. Several causes are probably responsible for this

- parts defects that were not properly eliminated by test -- these defects need not cause immediate failure in the post-launch environment because (a) many spacecraft components do not become operational until sometime after orbit is achieved, and (b) the failures occur only at elevated stress levels (an application of the statistics of extremes on a smaller scale)

- parts failures due to unrecognized design deficiencies, either in the parts themselves or in portions of the equipment that cause overloads or otherwise induce the observed failures

- underestimation of the shadowing and loss of interest effects.

The first of these factors is identified as the most significant one, both in terms of the number of failures caused, and also because it is the one most under control of project management [TIMM75]. This aspect of the time trend of spacecraft failures is also closely related to the employment and effectiveness of screening techniques, as subject that is receiving increasing attention in the reliability literature [SAAR82].

For reliability prediction at the spacecraft level a single Weibull model, such as the one shown in Figure 2-4, will be quite suitable. For reliability prediction at the subsystem and lower levels it is necessary to distinguish between the two contributions to failure probability as is explicit in the procedures described in Chapter 5. Examples of the application of these findings are presented in the next section.

2.4 Examples of Applications

The confirmation of the decreasing hazard phenomenon and the formulation of a Weibull model for reliability prediction is not merely of theoretical interest. The following examples show that significant decisions in mission planning and spacecraft design can be affected by the acceptance of a decreasing hazard model. In other areas, the distinction between random (parts and quality) and correlated (design and environmental) failures may affect reliability related design decisions.

The examples presented here are necessarily simplified and the parameters are selected to emphasize the difference between the constant and decreasing hazard assumptions. In a practical case the effects may be less than in these examples but they are usually quite significant. Additional research in this area will therefore be found beneficial.

2.4.1 Mission Planning

A satellite mission may terminate for one or more of the following reasons:

- catastrophic failure

- exhaustion of consumables such as attitude control gas or propellants for orbit maintenance; the degradation of solar cells is a related item because is requires allocation of additional capacity to sustain a long life on orbit

- technological obsolescence

The latter factor does not usually enter into the detailed trade-off decisions but it sets a time horizon beyond which benefits in the other areas are immaterial. Trade-offs between reliability (failure prevention) and consumables are necessary because both make demands on the same resources (funding and satellite weight). It is intuitively seen that it may be inefficient to provide consumables for more than 10 years when the predicted reliability of the prime mission equipment is very low at that point in time. Conversely, a reliability improvement to extend the satellite MTBF to eight years may not be warranted if consumables are provided for only five years.

The following example is a simplified mission planning investigation that highlights the effect that the choice of the failure distribution can have on optimum mission duration. It is assumed that spacecraft equipment design is fixed and that a reliability estimate at the 2 year point is 0.67. The spacecraft equipment configuration is modeled as two independent redundant strings of reliability R, so that the spacecraft reliability R_s becomes

$$R_s = 1 - (1 - R)^2$$

Consumables are to be provided until the time when the reliability drops to 0.4. In the first data column of Table 2-5 the time for R_s to reach 0.4 is computed under the exponential assumption and in the second data column it is computed using the Weibull distribution for the shape factor of 0.28 which was found to give a good fit to the total failure population in our sample.

TABLE 2 - 5 MISSION TIME FOR A SPECIFIED RELIABILITY

Parameter	Exponential	Weibull
Failure probability at 2 yrs.	0.33	0.33
Failure prob. equation at 2 yrs.	$(1 - e^{-2L})^2$	$(1 - e^{-1.21/a})^2$
Evaluation of parameter (L or a)	0.43	1.43
Failure probability at x years	0.6	0.6
Failure prob. equation at x yrs.	$(1-e^{-.43x})^2$	$\{1-\exp(-x^{.28}/1.43)\}^2$
Value of x	3.5 years	11 years

The time for which consumables are to be provided is much longer for Weibull than for the exponential assumption. To determine the potential benefit of this longer life to the mission planner assume that the mission value, V, is given by

$$V = v * T' - C(T)$$

where T = nominal mission time (to exhaustion of consumables)

T'= effective mission time or mean mission duration[1]

----

1. This is equivalent to the MMD truncated at the depletion of expendables as defined in MIL-STD-1543(USAF) "Reliability Program Requirements for Space and Missile Systems"

v = value to the user per actual year on orbit

and C(T) = cost of providing consumable for duration T.

T' will be approximated by (1 + R(T))*T/2. Since the mission termination is defined by R(T) = 0.4, the expression for T' simplifies to 0.7*T. For C(T) assume 0.05*s*T where s is the basic spacecraft cost. Also, assume that s/v = 3 (this means that the effective mission time must be at least three years before the program becomes economically justified. The following data are required to compute the mission value, V.

	Exponential	Weibull
Nominal mission duration, T	3.5	11
Effective mission time, T'	2.6	7.7
Value in terms of satellite cost*	0.87s	2.55s
Cost of consumables, C	0.18s	0.55s
Value excl. satellite cost, V	0.69s	2.00s
Net mission value	- 0.316s	1.00s

* making use of the relation v = s/3.

It is seen that a mission that had at a submarginal value under the exponential assumptions became soundly effective when the Weibull distribution was used.

2.4.2 Subsystem Design

A subsystem consists of three components that have the following mission reliability (for 5 years) and weight

Component	Reliability at 5 yrs	Weight lbs.
A	0.90	100
B	0.80	200
C	0.70	300

It is required that the entire subsystem reliability at 5 years be at least 0.70. The 5 year component reliabilities were computed under the exponential assumption. As in the previous case, the reliability prediction for a 2 year mission duration was the best validated data point and any Weibull model must be tied to the same 2 year values.

For the exponential assumption the subsystem reliability requirement can be met by

- The entire subsystem can be made redundant, requiring only a single reconfiguration provision, but incurring a weight penalty of 600 lbs. The resultant reliability will be 0.75, neglecting the failure probability of the switching circuits.

- Individual components can be made redundant, each with its own reconfiguration provisions. The minimum weight system that meets the requirements uses redundancy for A and C, with a reliability of 0.72, again without allowance for failures in the switching provisions. The weight penalty is 400 lbs.

In both cases it was assumed that active and standby systems had the same reliability. The reliability of a redundant system or component, R_r, was computed from

$$R_r = 1 - (1 - R)^2$$

where R is the reliability of the non-redundant unit.

In order to apply the Weibull model, the reliability at the 2 year point must first be computed. The hazard, L, is obtained from the five year reliability, R_5 as

$$L = (\ln R_5)/5$$

and then the two year reliability under exponential assumptions becomes

$$R_2 = e^{-2L}$$

These values are tabulated together with the computed Weibull 'a' parameter in Table 2-6. The other entries show the predicted Weibull reliability at 5 years, R'_5, and the reliability obtainable when individual components are made redundant.

TABLE 2 - 6 SUBSYSTEM PARAMETERS USING WEIBULL ASSUMPTIONS

Component	Reliab. at 2 yrs	Weibull 'a' param. (years)	Weibull Reliab. at 5 yrs.	Reliab. for redund. component
A	0.96	28.81	0.95	0.99+
B	0.91	13.60	0.89	0.99
C	0.87	8.51	0.83	0.97

The series reliability for the three components is 0.702 which just meets the minimum requirements. If just component A is made redundant, the reliability becomes 0.74, comparable with the configurations discussed for the exponential case, and at a weight increment of only 100 lbs.

Chapter 3

CAUSES OF FAILURE

By understanding the causes of failure the users of this report may be able to modify the baseline reliability prediction procedures in the light of their mission or equipment characteristics. If conditions that cause a specific class of failures are absent for a given application, then the failure prediction can be correspondingly reduced. Conversely, if a cause of failures is more pronounced, then the failure prediction will have to be increased. One of the most constructive uses of reliability prediction is as a design tool: to identify the configurations that yield the highest reliability within given constraints. In this connection, knowledge of the causes of failure can be effectively employed to improve the reliability of new as well as existing designs.

By way of providing background for the treatment of causes of failure, the first section of this chapter describes how failures on spacecraft are diagnosed. The classification of causes that was already briefly described in the preceding chapter is then explained in detail and examples of each type of failure are provided. Next, differences in the relative frequency of certain causes between pre-1977 missions and later ones are analyzed and some significant trends are identified. Finally, the association of spacecraft subsystems with the major causes of failure is investigated.

3.1 Diagnosis of Spacecraft Failures

The principal tools for diagnosis of spacecraft failures are

- Telemetry

- Analysis of spacecraft operation

- Retrospective analysis after subsequent anomalous events are observed

When spacecraft are returned to the earth there is of course an opportunity for direct diagnosis of the failure. Only for very few of the failures reported here was the latter course applicable. Because of the large economic and national security implications of spacecraft failures, many supporting investigations are usually carried out as soon as any off-nominal operation is observed.

Most spacecraft are heavily instrumented in order to permit monitoring of their operation, taking corrective measures when unusual events are observed, and detecting design weaknesses that can be avoided in future launches and designs. Instrumentation takes the form of

- Measurements of the environment (primarily temperature and radiation levels) and of supporting functions, such as electric power, common time bases, and attitude control

- Normal outputs of each payload function, e. g., sensor outputs from meteorology and earth observation satellites

- Specific diagnostic measurements in both the payload and supporting functions, including intermediate outputs of all sensor processing and of housekeeping functions (e. g., attitude error), and local temperature, vibration, and pressure measurements for pressurized components.

Satellites which are in continuous contact with a ground station can use direct telemetry for sending the data to the monitoring facility. Satellites which are not in continuous ground contact (this includes most missions in low orbits) must first record the data for later downlinking in a compressed time frame when they are in station contact. The tape recorders required for this procedure were themselves a very frequently failing component.

As a result of the availability of monitoring data, anomalies are often diagnosed before they affect the operation of the spacecraft or of the mission. In many cases, procedures can be initiated to prevent further progression of the malfunction, and in some cases even remedial action is possible, e. g., when a high battery temperature is noted, the load on the battery can be reduced and the battery might be reconditioned by subjecting it to controlled charge and discharge cycles.

Analysis of spacecraft operations is another important source of failure information. Examples are loss of power in a communication link, incoherent sensor output, or failure to execute a command that had been stored or sent. Tracking data can be used to diagnose malfunctions in propulsion and attitude control subsystems. The combination of spacecraft operations and telemetry can be a very effective diagnostic tool, e. g., by sending commands to the spacecraft that exercise functions believed to be implicated in the malfunction, and by correlating out-of-spec telemetry data with spacecraft rotation, spacecraft orbital position (relative to the sun or to the earth), or other periodic spacecraft activities.

Retrospective analysis can be used to assign causes to malfunctions that had originally gone undiagnosed. The most common occurrence is that one or more similar malfunctions are observed in other spacecraft. Just the multiple observation of identical events will usually indicate that a design-related cause is involved. Multiple observations will also permit identification of common features of the anomalies, e. g., all occurring on exiting from an eclipse or all following transmission of a specific command. Finally, the diagnosis of one malfunction based on telemetry and/or spacecraft operations can furnish clues for retrospective assignment of causes to previously observed occurrences of the same type.

Ground-based support of satellite failure diagnosis consists of analysis of the on-orbit data (telemetry, tracking, and operational), simulations (based on analytical models or utilizing suspected hardware components), and re-inspection of residual hardware (e. g., components procured for future launches or excess inventory for a current satellite) or of equivalent hardware (components or parts of the same type and date of manufacture). The results of such inspections sometimes show defects in parts, workmanship, or procedures that become candidates for further diagnostic activities of narrower scope. Sometimes procedural deviations are discovered, e. g., that parts did not undergo all required tests or that the test might have overstressed the part.

3.2 Classification of Causes

As indicated in the preceding section, the diagnosis of spacecraft failures is unique in that

- a sizeable effort by high level technical personnel is devoted to the diagnosis of most failures
- because of the inaccessability of the spacecraft the *corpus delicti* can only rarely be recovered

The latter factor suggests that the diagnosis of any one malfunction may be subject to some uncertainty. On the other hand, the comprehensive nature of the data collection, analysis and reporting effort makes aggregations of spacecraft failure data a very valuable basis for statistical evaluation. In order to facilitate meaningful statistical results, fairly broad cause classifications have been selected so that a population of at least 100 failures exists in each category. This is particularly important when subclassifications are evaluated, e. g., the distribution in time to failure after launch by causes that was presented in the preceding chapter. The following cause classifications were selected on this basis

- Design
- Environment

- Parts
- Quality
- Operation
- Other known
- Unknown

In some evaluations failures due to parts and quality are treated as a single entity, and the same is true in some instances for failures due to operation and other known causes. In ODAP the cause of failure is expressed in key words as well as in prose. The key words are either equivalent to those used here or could be easily translated into them. In the OOSR reports the failure is described in prose and an "Incident Type" is derived from this which is classified in two ways

- Electrical, mechanical, other, and unknown
- Catastrophic part failure, other part-related incident, non-part-related, and unknown

The mapping of OOSR reports into the cause classifications shown above relied primarily on the prose descriptions.

The classifications which are of primary importance for the reliability prediction of electronic components are design, environment, parts, and quality. The conceptual distinctions between these causes are shown in Figure 3-1. Random parts failures, which are the core subject of the MIL-HDBK-217 reliability prediction procedures, are in the present data collection usually characterized by

- the failure is traced to a part or to a small aggregation of parts
- there is no evidence of a design deficiency, excessive environmental stress, or of a quality related problem

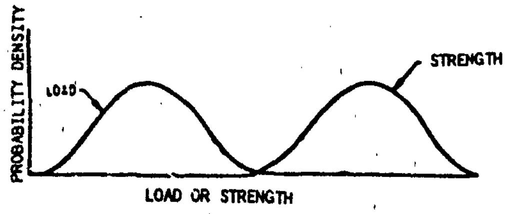
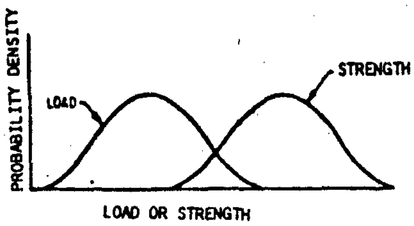
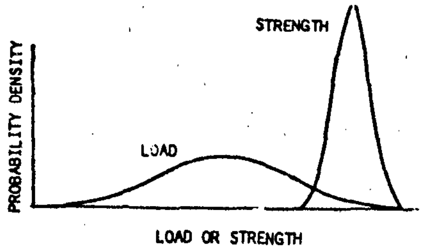
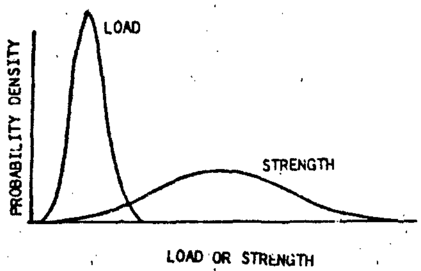
CAUSE	ASSUMED LOAD/STRENGTH RELATION	HOW DIAGNOSED
PARTS (RANDOM)		NON-REPETITIVE NO OTHER CAUSE LIKELY
DESIGN		REPETITIVE ANALYSIS ESTABLISHES THAT STRENGTH IS INADEQUATE IN SOME CIRCUMSTANCES
ENVIRONMENT		USUALLY REPETITIVE ANALYSIS SHOWS LOAD DUE TO ENVIRONMENT TO EXCEED ORIGINAL SPECIFICATION
QUALITY		USUALLY REPETITIVE ANALYSIS SHOWS THAT VARIATION OF STRENGTH EXCEEDS SPECIFICATION

FIGURE 3 - 1 REPRESENTATION OF FAILURE MECHANISMS

- there is no pattern of similar failures

Typical part failure synopses are "Decryptor B-side power supply failure; suspect intermittent open in transistor of power converter" (ODAP incident 2426), or "Solar array temperatures appear abnormal but no effect on power output; due to array thermistor failure" (IUE incident 9 in OOSR).

Design failures can be of two types: selection of parts that do not possess sufficient strength as indicated in the figure, or not allowing for the full range of spacecraft operations. An example of the former is "Sensor circuit reset while using back-up encoder; the detectors within the optical decoder are sensitive to Van Allen belt energetic particles." (ODAP incident 466) This failure occurred in 1979 when the characteristics of the Van Allen belt were well known and should have been considered in the design. The report on this incident also references another problem of the same type. An example of a more operations related design deficiency is "Sunlight entered sensor of electrons and photons experiment, causing loss of about 50% of the experiment data; design error or oversight -- the sensors were light sensitive" (ISEE-1 incident 1)

Environment is listed as a cause of failure where unanticipated environmental effects were encountered or where the magnitude of anticipated events was greater than specified or expected. As indicated in the figure, the load due to the environment frequently has a very long right tail which causes occasional failures even in parts or components which were correctly designed according to the original mission specification. Although the load distribution is shown here as normal, it may actually be more closely approximated by an exteme value distribution as discussed in the previous chapter. The significant feature in either case is a long right tail. Examples are "Ionospheric plasma monitor data is degraded, apparently caused by static charge build-up on spacecraft" (ODAP Incident 500), and "Delayed restart of Operational Linescan System (2 minute compared to normal 15 - 40 seconds). May be due to unusual pattern of proton effects" (ODAP Incident 508). The component involved in the first example had been designed when spacecraft charging was a little understood phenomenon, and therefore the problem is not classified as improper design. In the second example the environment is specifically described as unusual.

Quality is assigned as a cause when there are repeated failures in the same part or assembly that cannot be attributed to design or environment or which correlate with quality defects found in populations of similar parts. An example of the former type is "Shunt voltage in power conditioning assembly indicates erratic fluctuations. Probable cause is opening of collector resistor in shunt driver circuitry. Similar problems were encountered on two previous flights" (ODAP Incident 1342). Correlation with ground observations governed the classification of Viking Lander 1 Incident 1: "Telemetry indication of reduction in internal pressure of radiothermal generator 1. Traced to leakage of gases into the pressure transducer reference cavity. Suspected prior to launch based on pre-launch pressure data." Failures that were traced to improper test or that were test induced were also placed into the quality category. An example of this cause is "Mass deployment telemetry switch did not indicate that boom had been deployed. Attributed to deformation of actuator during ground system test. Revised tooling and installation procedures" (ODAP Incident 43). The representation of this cause of failure in Figure 3-1 by a standard distribution of strength with large variance is a very general indication of the failure process. In practice, it is more likely that there is a bimodal distribution and failures occur only in the (anomalous) low strength portion of the population.

As had already been indicated in the previous chapter, failures classified into the unknown category were most likely due to parts. This is consistent with the diagnostic key for parts failures indicated in Figure 3-1 -- non-repetitive and no other cause likely. The primary criterion that led to placement into this category rather than into parts was insufficient data in the reports. Examples are "Faulty multiplexer no. 1 channel caused loss of some narrow coverage driver TWTA temperature data. Switched to redundant multiplexer" (ODAP Incident 25) or "Manifold pressure increased out-of-limits following simultaneous firings of + pitch and - roll. Returned to normal within one orbit" (Nimbus-7 Incident 48).

Failures classified as due to operation involved sending improper commands to the spacecraft or faulty ground software. Examples are "Temporary drop in battery power. Improper reconditioning (by ground command) caused cell failure. Recovered by using new deep charge reconditioning technique" (ODAP Incident 1130) or "Address accept was not transmitted during upload. If a message is sent to spacecraft within 6 seconds of receiver turn-on, the message is not accepted. Corrective action: wait at least 7 seconds after receiver turn-on before uploading" (ODAP Incident 1194). Very few of the failures (less than 1%) were due to faulty on-board software. This is not too surprising because only two of the major missions utilized significant computer programs (contrasted with stored telemetry or timing routines). An example of an on-board software failure is "Large yaw error while switching central processors. Traced to software fault; rewrote procedure" (ODAP Incident 1326).

The classification of other known failures includes early depletion of consumables (attitude control gas, orbit make-up propellant), wearout failures, and wiring. Examples are "Radiometer scan drive motor showed signs of periodic loss of speed after 18 months on orbit, may be due to old age" (ODAP Incident 1527) or "Sensor lost lock on limb due to increased detector temperature caused by depletion of the cryogen" (Nimbus-7 Incident 29).

It is probably evident from this discussion that the classification involved some judgement. In ODAP this led to the assignment of multiple causes for some failures, a practice which was also followed in this report (the data base allows for up to three causes but this limit was only infrequently utilized). One result of the multiple classification is understatement of the relative frequency of the unknown category which is only rarely used together with any other cause while failures in the remaining categories may be counted more than once (but only for the purpose of classification).

3.3 HISTORICAL TRENDS IN CAUSES OF FAILURE

For the purpose of reliability prediction it is of interest to investigate historical trends in the causes of failure. If the recently launched spacecraft exhibit a drastically different failure pattern, then this should be taken into account in the prediction methodology. For the investigation of historical trends the spacecraft were divided into two categories:

- Early programs -- where the first launch took place prior to 1977

- Late programs -- where the first launch took place in 1977 or later

Spacecraft in the latter category are likely to utilize medium to large scale integrated semiconductors and are therefore more representative of the designs addressed by future reliability studies. It must be recognized, however, that reliability prediction based on interpretation of field data has inherent limitations in dealing with new part types or design methods.

The distribution of causes in the two chronological divisions is shown in Figure 3-2. It is seen that failures caused by design and environment constitute a considerably greater proportion among the late programs, and that failures due to parts, quality, and unknown causes are a correspondingly smaller proportion. A summary of aggregated causes is shown in Table 3-1.

TABLE 3 - 1 VARIATION OF CAUSES WITH DATE OF FIRST LAUNCH

Cause	Fraction of All Causes	
	Early Programs	Late Programs
Des & Env	.424	.565
P, Q & Unkn	.458	.338
Oper & Other	.118	.097

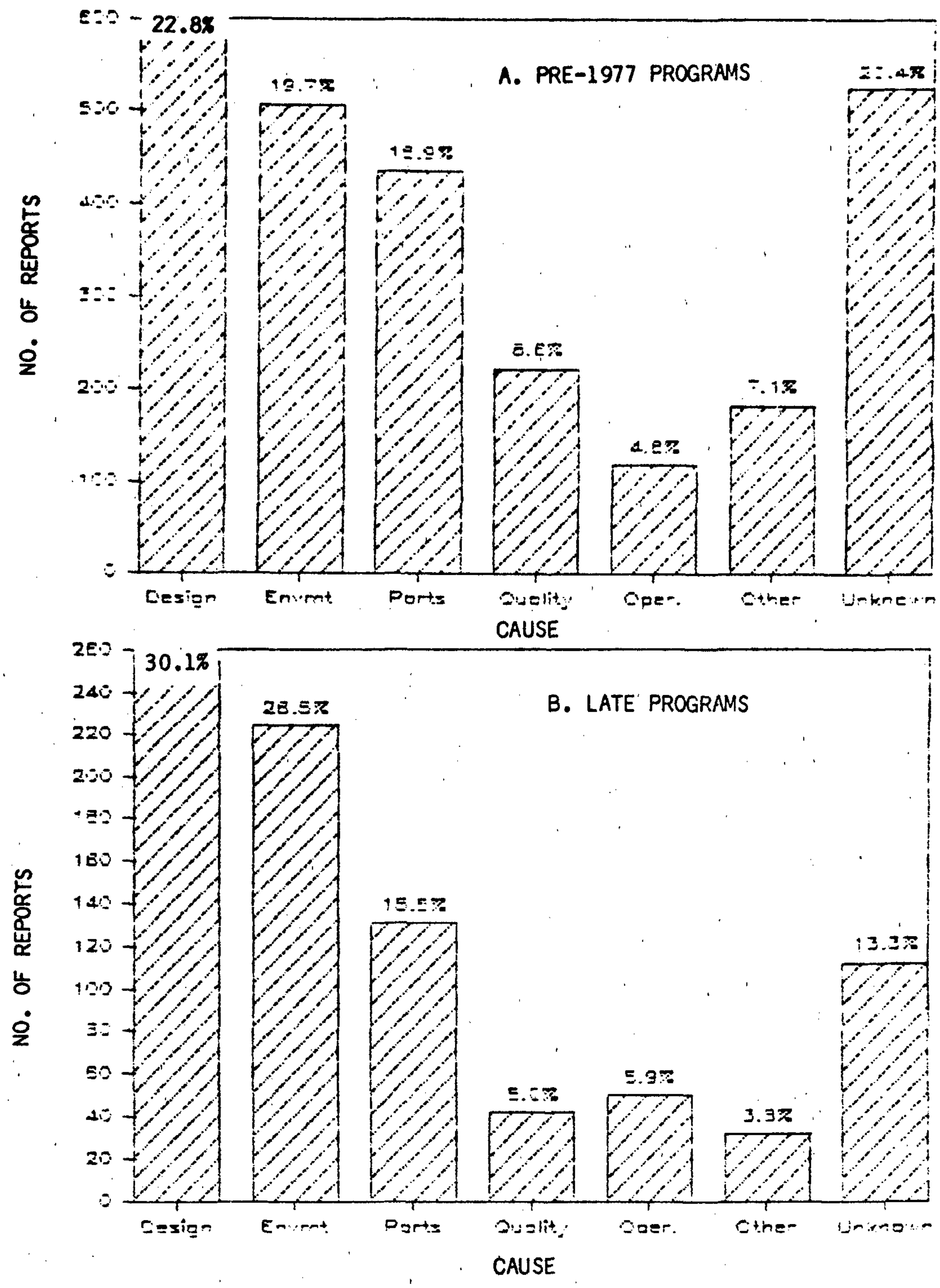


FIGURE 3 - 2 DISTRIBUTION OF CAUSES

A positive conclusion from this summary is that improved parts selection and quality control for space applications seems to have borne fruit. A more surprising finding is that advances in design and environmental studies do not seem to have kept pace with the demands of space missions. One explanation is that a number of new mission types, such as navigation, have been introduced and that many of the design problems are associated with these. New part types, particularly large scale semiconductor memories, also saw their first use in space in the late programs, and some of the environmental failures are due to radiation effects on these. These effects are readily seen in the distribution of failures by subsystem shown in Figure 3-3 for design and in Figure 3-4 for environment.

Further, a part of the increase in design and environmental causes is due to improved instrumentation, observation and analysis. Failures due to unknown causes have decreased from over 20% in early program to less than 15% in late ones. As a result of greater experience and better data, failures that would have been undiagnosed or assigned to random parts failures are now recongized as due to design problems.

In the preceding chapter it was seen that design and environment caused a much more pronounced and continuing decrease in the failure ratio than all other causes. Due to the increased proportion of failures caused by design and environment it might be expected that the failure ratio for late programs would show a more sharply decreasing trend than the pattern discussed in Chapter 2 (particularly Figure 2-4). However, this could not be verified partly because differences in the mission mix made it difficult to isolate effects due to causes, and partly because the late programs yielded insufficient data for times on orbit in excess of three or four years. Since the cut-off date for this report was January 1984, no spacecraft launched after January 1977 could have accumulated more than 7 years in orbit and only a very small number had accumulated five or more years.

The overall failure ratio for late programs is about twice as large as for early programs. This should not be interpreted as a decrease in reliability

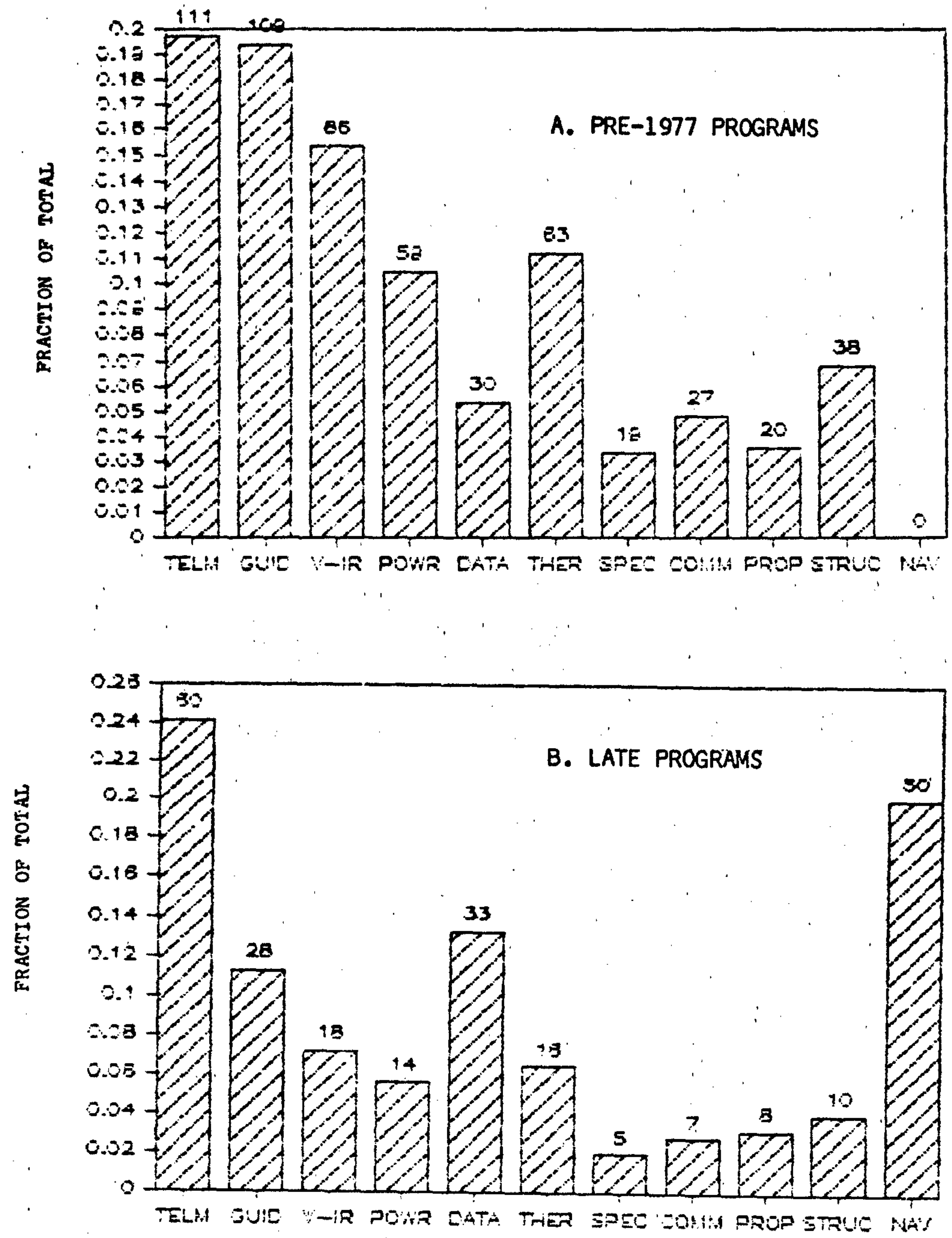


FIGURE 3 - 3 DESIGN PROBLEMS BY SUBSYSTEM

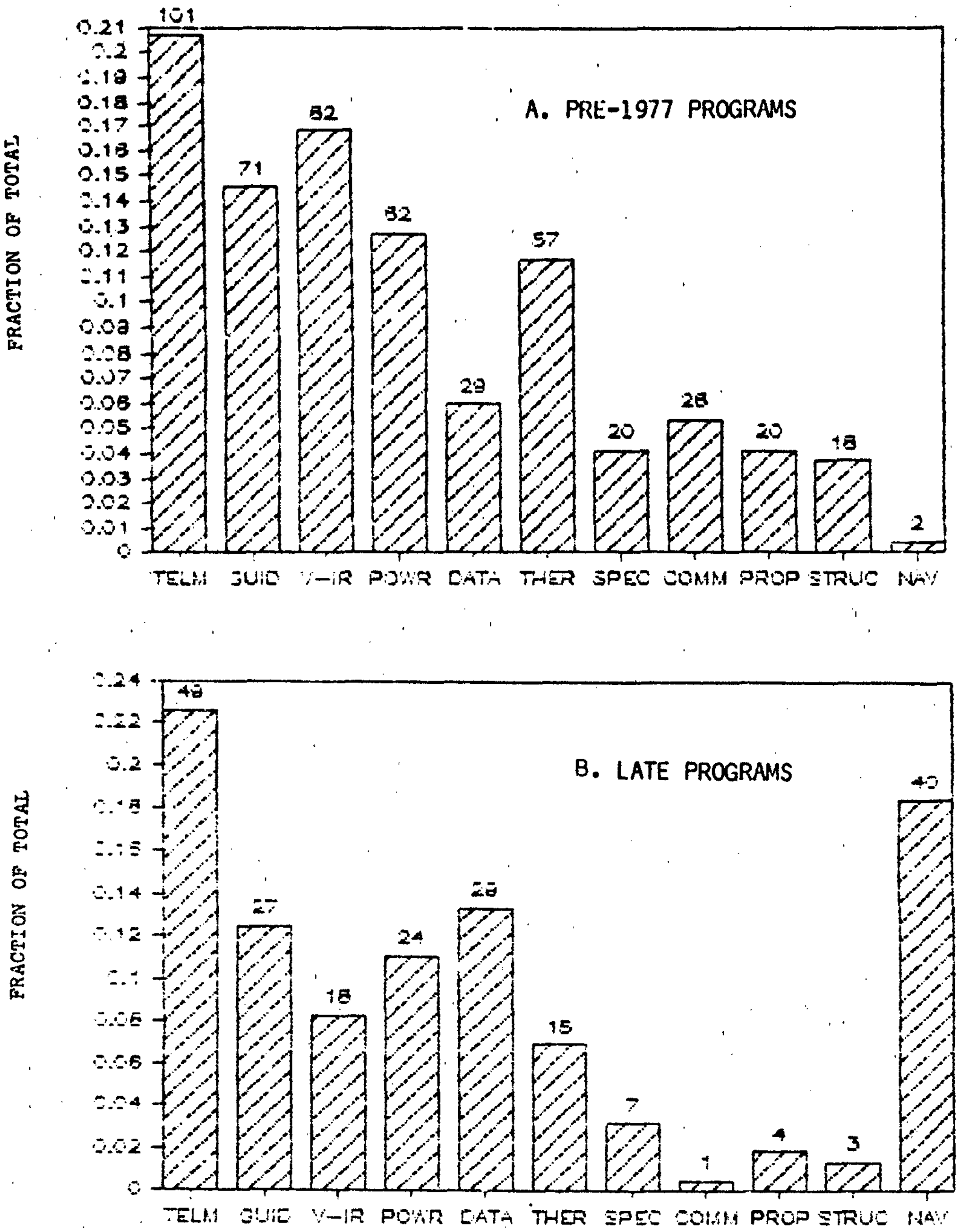


FIGURE 3 - 4 ENVIRONMENT PROBLEMS BY SUBSYSTEM

for either satellites or parts. The major cause of the higher failure ratio is the much greater complexity of the satellites launched by the late programs. At least three factors contribute to the increase in complexity

- Multi-mission satellites, e. g., combining earth observation and meteorology or providing several types of communications on one satellite)

- Higher performance and accuracy of individual missions, e. g., more channels and higher signal-to-noise ratio for communication payloads, increased accuracy and ease of use for the navigation function)

- Increased use of redundancy to support longer mission durations

It is difficult to quantify the increase of complexity in terms of component or parts counts, partly because the data are difficult to obtain but mostly because the definition of parts and components has undergone very major changes, particularly in the electronics field. The improved ruggedness of recent satellites as a whole can be seen from the greatly reduced fraction of failures that are in the high severity categories (see Chapter 4 for a further description of the severity classifications).

TABLE 3 - 2 SEVERITY OF FAILURE FOR EARLY AND LATE PROGRAMS

Classification		Early Programs		Late Programs	
Code	Description	Count	Percent	Count	Percent
1	Critical failure	186	10	18	3
2	Single point failure	160	8	28	5
3	Redundant unit	353	18	68	12
4	Work-around req'd	339	18	101	17
5	Degraded performance	499	26	117	20
6	Temporary failure	334	17	225	38
7	Others	52	3	32	5

Critical failures, which terminate the operation of the entire satellite or a major function, represent a much smaller percentage of the total for late programs. Conversely, failures which have only a temporary effect on satellites operation represent a much higher fraction of the total for late programs.

Although there are noticeable differences in causes, failure ratio and severity between early and late programs, the advantages of utilizing the entire data base for reliability prediction outweighed those of restricting it to the late programs. The advantages considered in this connection included

- the incident population available is approximately four times as large
- hazard trends could be evaluated through the eighth year after launch
- meaningful sub-analyses could be investigated

The detailed evaluation of failure ratios by subsystems and missions in the next section and in the following chapter permits tailoring of the reliability prediction for the equipment population and orbit characteristics of newer satellites. A specific case is the evaluation of navigation satellites, a mission type that was only rarely encountered prior to 1977.

3.4 FURTHER ANALYSIS OF CAUSES OF FAILURES

This section analyzes for each of the major cause classifications (a) where the failures arise (primarily by subsystem) and (b) whether there are significant differences in the locale of the failures between early and late programs. The data presented here identify the baseline population for the reliability prediction procedures of Chapter 5. This information may be used to tailor prediction for new satellite types in which the mix of subsystems and functions differs significantly from previous designs but specific tailoring procedures are not provided as part of this report.

In each of the following subsections the distribution of causes of failures among spacecraft subsystems is illustrated by means of bar graphs. The ordering of the subsystems along the horizontal axis is by decreasing failure contribution in the total satellite population. If the representation of subsystem failures within a given cause corresponds to that within the total data base, the height of the bars will decrease from left to right. Any deviation from a strictly decreasing pattern indicates an atypical contribution of subsytems within a given cause. Only the most important ones of these deviations are commented on.

3.4.1 Design

Because failures caused by design constitute the largest category (almost 25% of the total), non-conformance to a decreasing pattern among the bar graphs is particularly significant. In Figure 3-5A which encompasses the early programs two subsystems have a clearly excessive representation: thermal and structures. The leading causes of design failures in the thermal subsystem were inadequate thermal models during the first decade of space flight and failure to account for deterioration of thermal coatings in the space environment. Most of the design failures in the structures subsystem were associated with deployment mechanisms (latches, articulated booms, and separation devices).

As can be seen in Figure 3-5B, which illustrates the same relation for late programs, improved modeling and better understanding of the characteristics of coatings have greatly reduced the incidence of design failures in the thermal subsystem. There has also been a considerable improvement in the structures area although the design of deployment devices continues to be a source of failures. The data management subsystem which made only a very small contribution in the early programs has become a very significant cause in late programs. The main reason for this is that there were very few data management functions in satellite designs that saw their first launch prior to the mid-1970s. Data management systems will continue to increase in importance and complexity in future satellites, and the contribution of design failures in these should be an area of concern. Redundancy which is

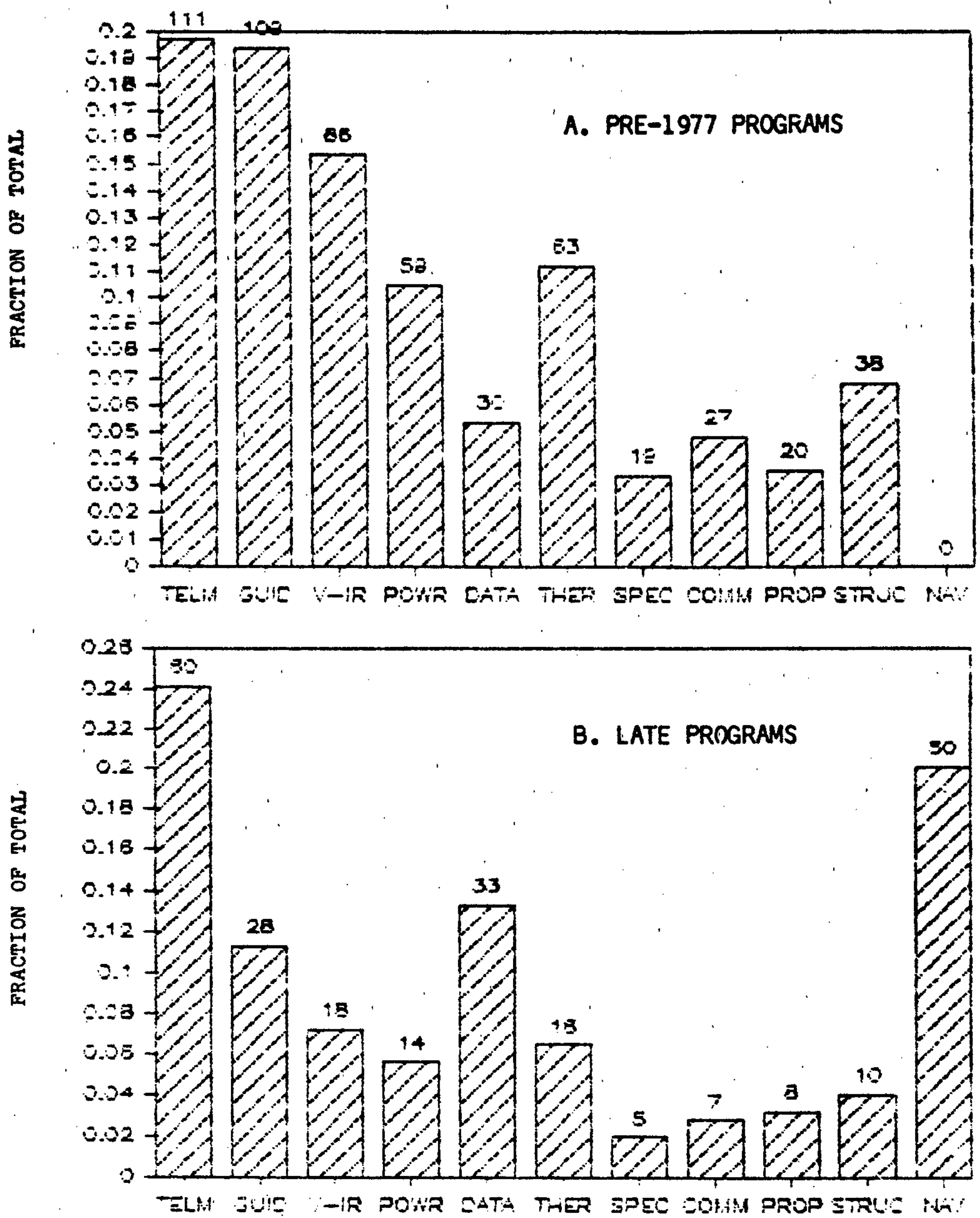


FIGURE 3 - 5 ANALYSIS OF DESIGN FAILURES

widely used to permit digital equipment to be used in critical applications provides only limited protection against failures due to faulty design. The other very significant change from the pre-1977 experience was associated with the navigation payload subsystem which constituted the second largest number of failures in Figure 3-5B. Here, again, the change in satellite functions is a major factor in the difference between the early and late programs. However, there have been some unusual reliability problems in the navigation payloads as further discussed in Section 4.4.1.

The telemetry subsystem is the largest contributor to design causes during both periods covered in Figure 3-5. The percentage of total design failures due to this subsystem has increased somewhat in late programs. The telemetry, tracking and command functions in recent satellite designs are very complex and there is no indication that this trend will abate. In the context of reliability prediction the telemetry subsystem is one of the more stable spacecraft components. The relative contribution of the guidance and visual/infrared sensor subsystems to the design failures is much less in late programs than in early ones. In both cases there has been a considerable maturation in system design and a very marked improvement in component technology which permits more conservative design.

3.4.2 Environment

The general trend for failures due to environmental causes shown in Figure 3-6 is very similar to that found for design causes. In early programs the thermal subsystem contributes a disproportionately large number of failures but this tendency is much reduced in late programs. The visual/infrared sensor subsystem has the second largest number of failures in early programs, largely due to lack of knowledge of space effects on optics and sensitive sensor elements. The contribution of this subsystem to environmental failures in late launches is much less. The navigation and data subsystems show up as the second and third most frequent cause of failures due to the environment, and this is again related to the greater representation of these systems on recent designs and the lack of experience on space effects on the components.

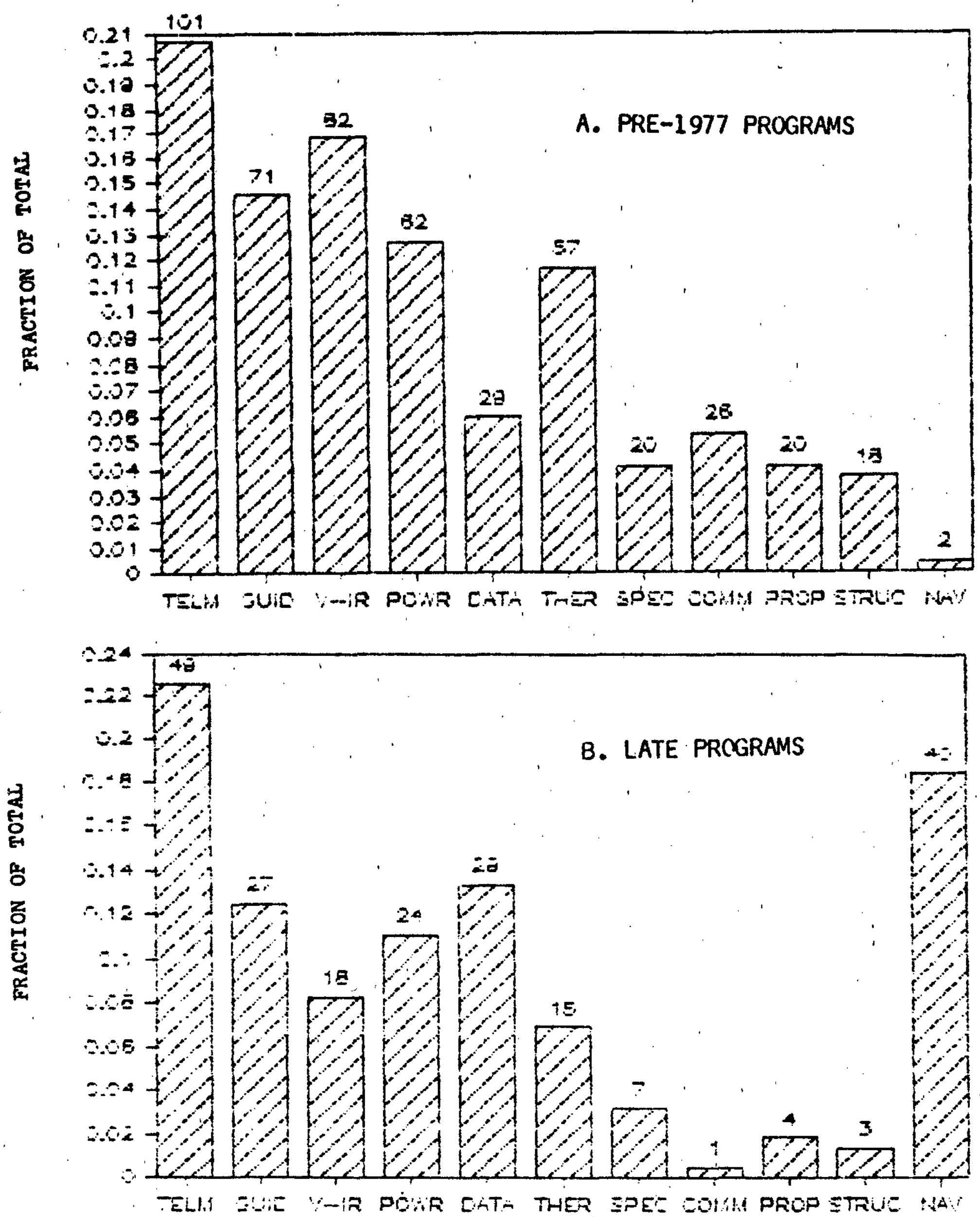


FIGURE 3 - 6 ANALYSIS OF ENVIRONMENT FAILURES

The environmental failures in the telemetry subsystem are partly due to unusual space effects such as solar flares, but another significant segment is due to electromagnetic interference. Some of the latter arises from equipment aboard the spacecraft but a large amount comes from terrestrial and unknown sources. Fortunately many of these failures affect the spacecraft only temporarily. The power subsystem accounts for about one-eighth of all environmental failures during both periods. Most of these failures are associated with solar cells and battery charging circuits.

3.4.3 Parts and Quality

As indicated in Figure 3-7, the data management and navigation payload subsystems are particularly significant contributors to failures due to parts and quality. The navigation function has the largest number of failures due to this cause among late programs while data management account for approximately 15% of the failures in both time periods. The communication payload is a significant factor in early programs but much less so in late ones. Telemetry, data management, and the navigation payload are the largest users of semiconductors on the spacecraft, and therefore the distribution of parts and quality failures shown in Figure 3-7B is not too surprising.

3.4.4 Unknown Causes

It is seen in Figure 3-8A that for early programs the telemetry subsystem accounts for 35% of all failures due to unknown causes, a proportion that is markedly higher than seen in any other cause. Part of the reason may have been lack of instrumentation in this function in the earlier satellites. Figure 3-8B shows that in late programs the unknown failures due to telemetry represent only about one-half of that fraction, more in line with the representation of telemetry in the remaining causes. The power subsystem is a large contributor in both time periods but particularly among recent programs. Many of these failures are associated with power conversion

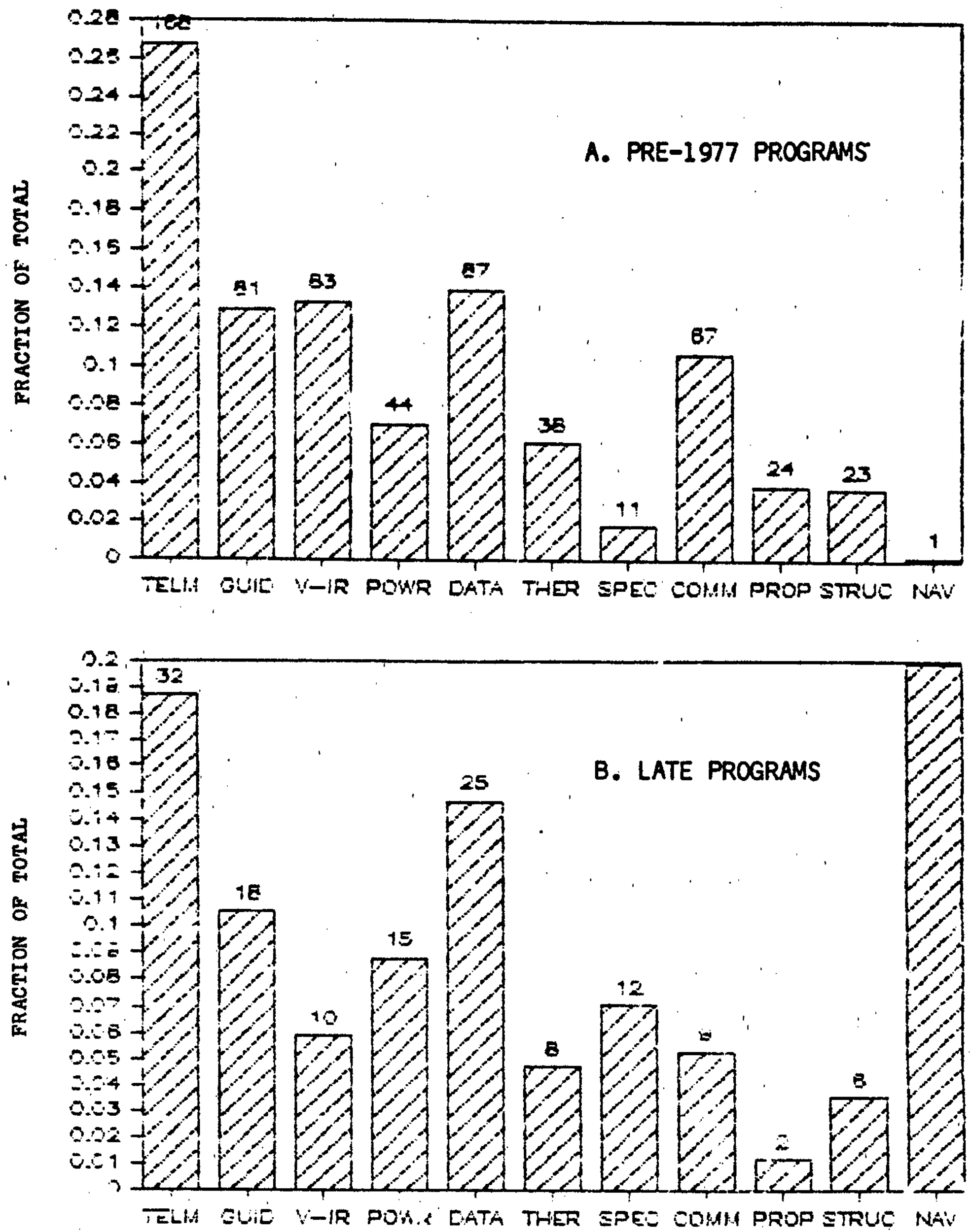


FIGURE 3 - 7 ANALYSIS OF PARTS AND QUALITY FAILURES

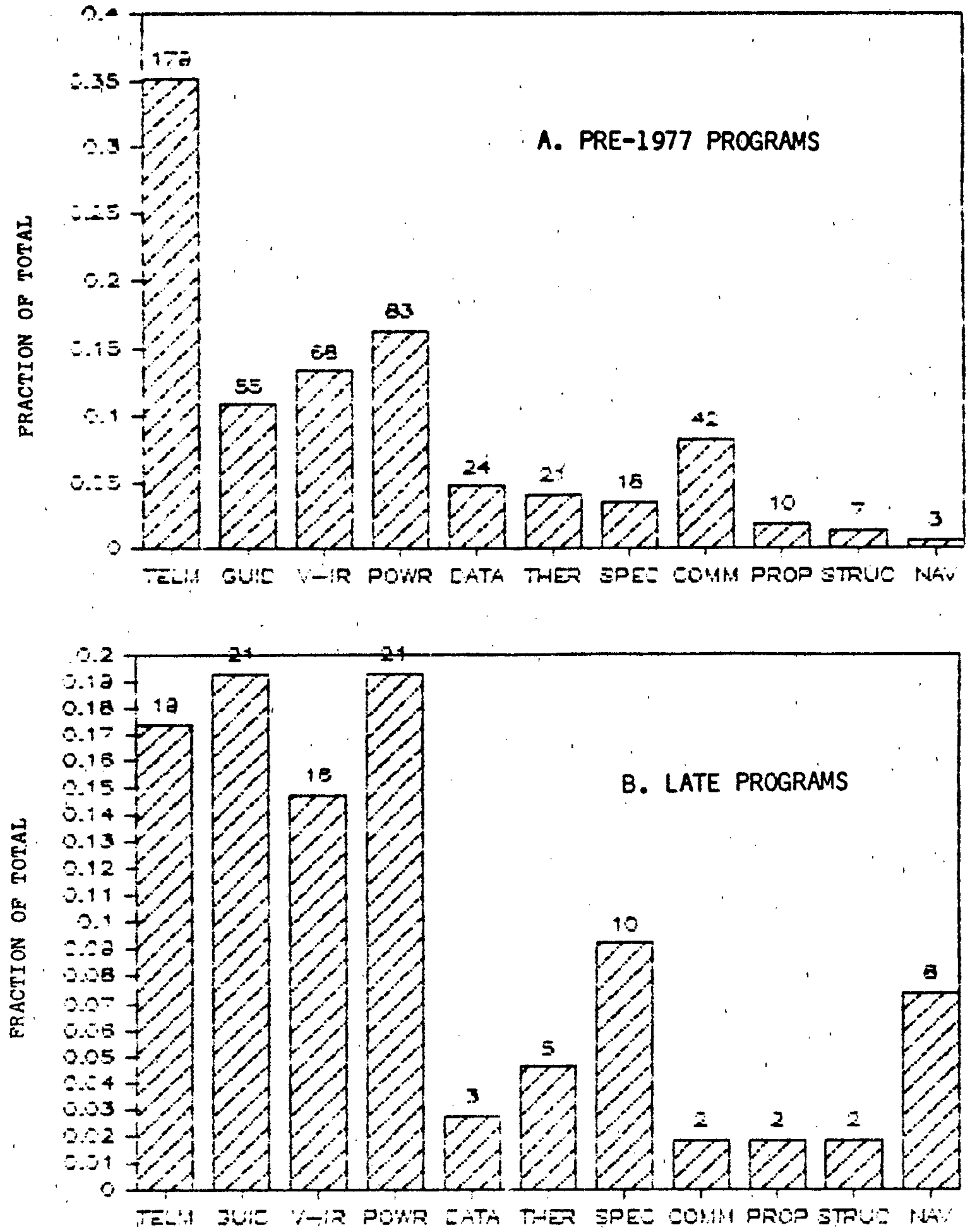


FIGURE 3 - 8 ANALYSIS OF UNKNOWN FAILURES

electronics, a function that is apparently not well instrumented.

The guidance subsystem, visual/infrared sensors, and special payloads are other major contributors to unknown causes in recent programs. Among the guidance and sensor failures are many that cause only minor disturbances and which might conceivably have been overlooked on earlier flights. The increased contribution of special payloads is largely due to a higher representation of this category in recent programs.

Chapter 4

DETAIL EFFECTS AND FACTORS

This chapter presents analyses of the failure severity, of the effects of complexity, and of failure rates in a number of partitions of the total satellite population. The conclusions are summarized in Table 4-1. Section 4.1 discusses failure distributions by severity; Section 4.2 examines partitions based on subsystems, Section 4.3 analyzes complexity effects, Section 4.4 mission effects, and Section 4.5 orbit effects.

TABLE 4-1. RESULTS OF ANALYSES PRESENTED IN THIS CHAPTER

PARTITION	EFFECTS
SEVERITY	Frequency of occurrence is inversely related to severity
SUBSYSTEMS	Telemetry, guidance, and electrical power are the largest sources of failures; thermal and structural subsystems are among the lowest. Differences in the failure distributions in pre- and post-1977 programs reflect maturing technologies in some subsystems (e.g., guidance, communication payloads) versus increasing complexity in others (e.g., data management, visual-IR)
	The failure rates of electronic and electromechanical subsystems generally decrease as a function of time whereas mechanical subsystems do not exhibit such behavior.
	The importance of parts and quality causes increases with the maturity of subsystems, particularly in electronic subsystems
COMPLEXITY	Indicators of complexity can demonstrate statistically significant differences in failure rates.
MISSION	Significant differences in failure rates are evident for different classes of missions.
ORBIT	Low orbit (i.e., perigee less than 200 km) satellites have a higher failure rate than higher orbit satellites. However, such differences can be accounted for by payload and specific subsystems characteristics rather than by environmental differences.

3.1 Severity

For the purposes of this study, the severity of failures was categorized as follows:

1. Critical Failure -- entire satellite or a major mission function fails. Example: Loss of S-band and instrument operation due to spacecraft power problems. Attempted work-around but to no avail (AEM-1, incident 11)

2. Single Point Failure -- major assembly or component failure. Example: No output from sensor 25, band 5 and degraded output from sensor 26. Loss of IR data causes significant mission impairment. Periodic outgassing performed to clean sensors but not successful in long run (Landsat-3, incident 8)

3. Redundant Unit Failure -- requires activation of a back-up component or system. Example: Command clock power supply #2 failed; switched to redundant power supply but only one command link now open (Landsat-2, incident 16)

4. Work-around -- failure requires change in operating procedures and may cause degraded performance. Example: Auxiliary command memory halted due to fixed core checksum error. Checksum modified to accommodate the error (Landsat-3, incident 7)

5. Degraded Performance -- failure degrades performance of a mission function. Example: Threshold problems in coastal zone color scanner cause loss of data in channels 1-4, reducing water coverage from 90% to 50 - 60% (Nimbus-7, incident 1).

6. Temporary failure -- full capability restored spontaneously or after recovery procedure. Example: Stratospheric sounder scan shifted 43 counts and there were other irregularities in the command logic. Mission effect was small, and the problem has not recurred (Nimbus-7, incident 13)

7. All other failures -- usually not affecting a mission function. Example: Earth resource budget scanhead went into a forbidden zone. Attributed to gimbal motor torque margin and lubricant viscosity. Negligible effect on mission (Nimbus-7, incident 17)

Figure 4-1 shows the distribution of all failures by severity. It is seen that failure frequency is inversely related to severity, i.e., serious failures occur less often than trivial ones. That the distribution peaks at category 5 rather than at 7 is probably due to the tendency not to report all failures that result only in a temporary anomaly or that have no significant effect on the mission.

Figures 4-2 through 4-5 illustrate the distribution of failures by severity over orbital life segments, starting with the first month on orbit and going out to lifetimes of five years or more. It is seen that this distribution remains roughly the same for the first three intervals investigated. However, for failures occurring after 5 years, there is a marked drop in the proportion of reported failures in severity categories above 3. As already discussed in Section 2.2, the major reason for this appears to be the decreasing thoroughness of the failure reporting procedures, particularly for missions which had considerably surpassed the initially estimated lifetime and for which operating staff may have been reduced. A clear indication of this phenomenon is that the mode shifts from category 5 to category 3. The ratio of severity 4 and higher failures to those of severity 1 - 3 is 2.4 in Figure 4-3 and only 1.2 in Figure 4-5. The total data loss due to this process is unlikely to be more than 60 failures.

4.2 Subsystems

This section discusses the location of failures in terms of subsystems. The following 11 subsystems, listed in order of decreasing failure frequency, are analyzed (definitions were adapted from [ODAP84]):

1. Telemetry, tracking, and control: used for commanding the satellite by receiving ground commands and decoding and distributing them to other satellite subsystems. It directs steerable antennas and transmits state-of-health, tracking, and payload data to ground stations. It includes tape recorders where these are required in connection with ground communication. The name of this subsystem is sometimes shortened to 'telemetry' but is always meant to include the total functions just described.

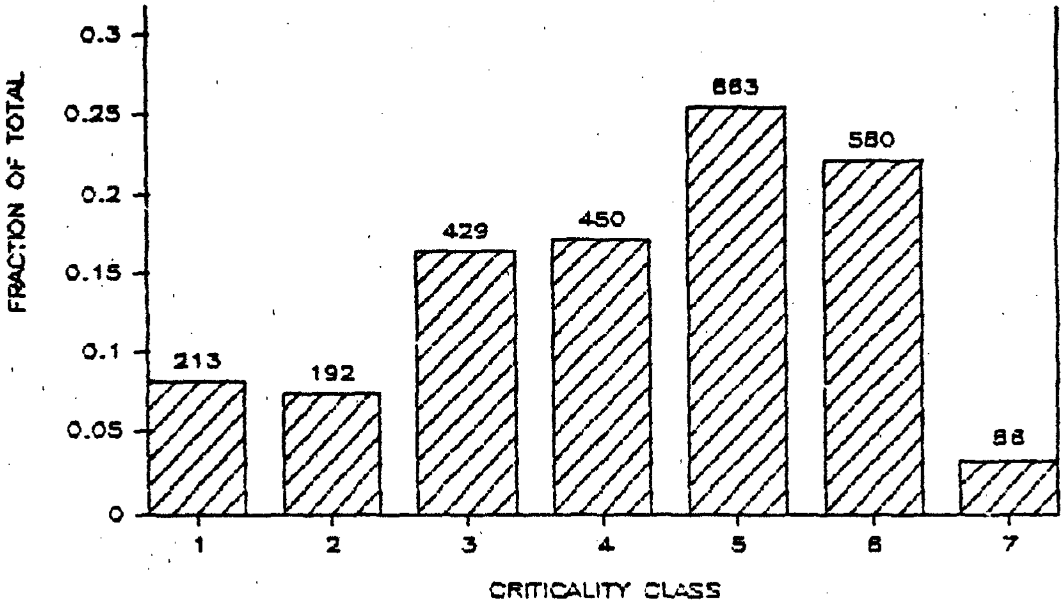


FIGURE 4 - 1 DISTRIBUTION OF ALL FAILURES BY CRITICALITY

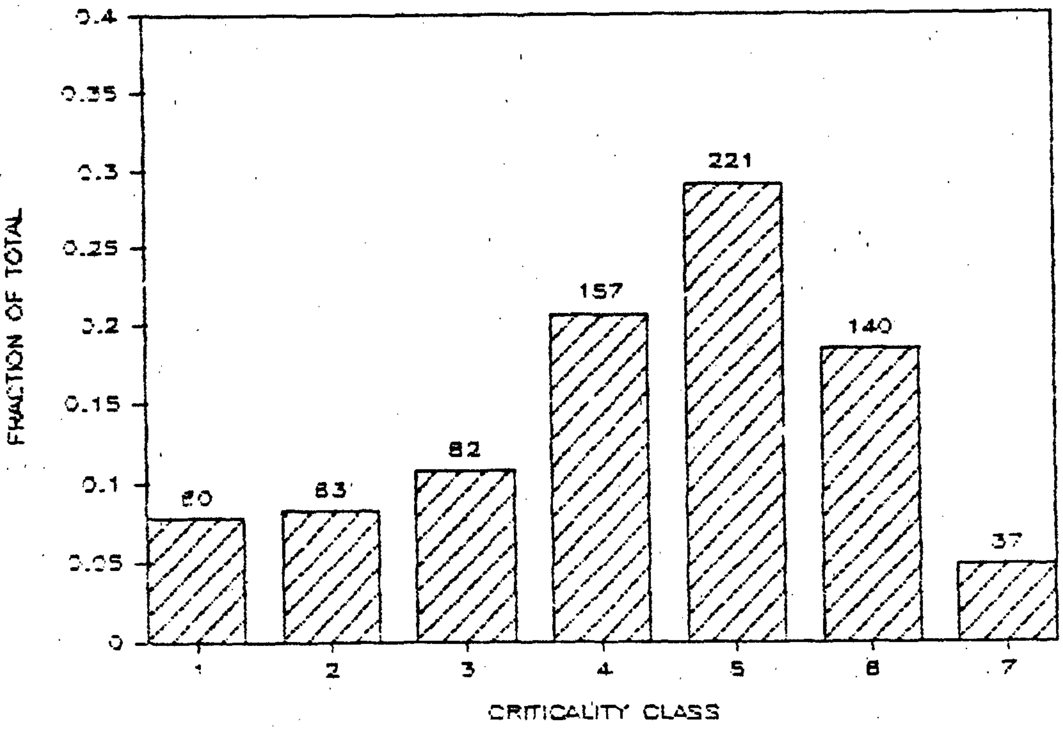


FIGURE 4 - 2 DISTRIBUTION OF FIRST MONTH FAILURES BY CRITICALITY

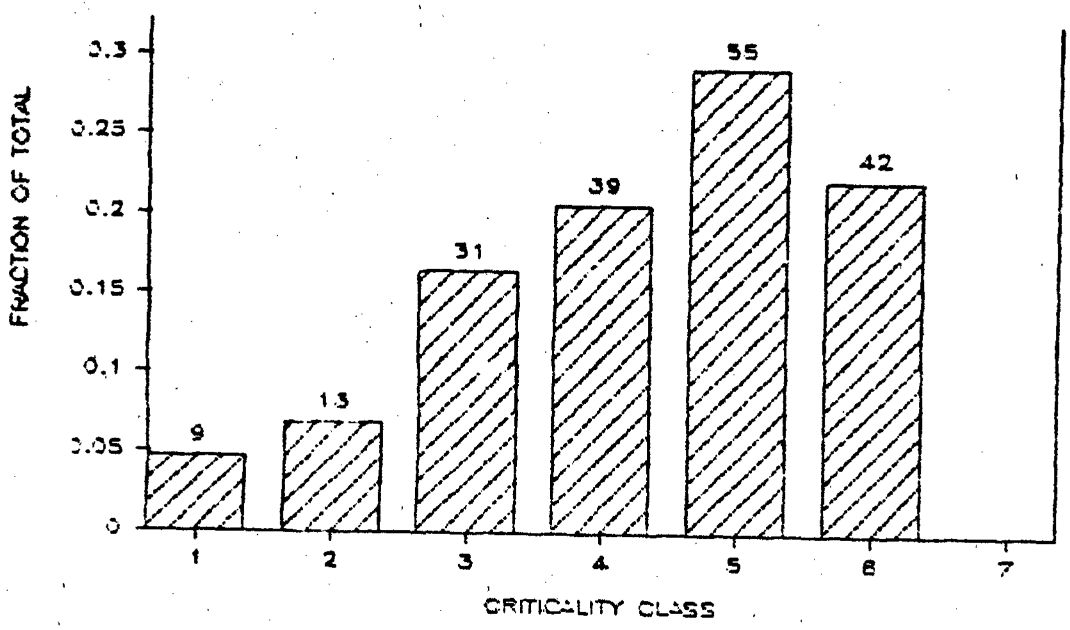


FIGURE 4 - 3 DISTRIBUTION OF SECOND MONTH FAILURES BY CRITICALITY

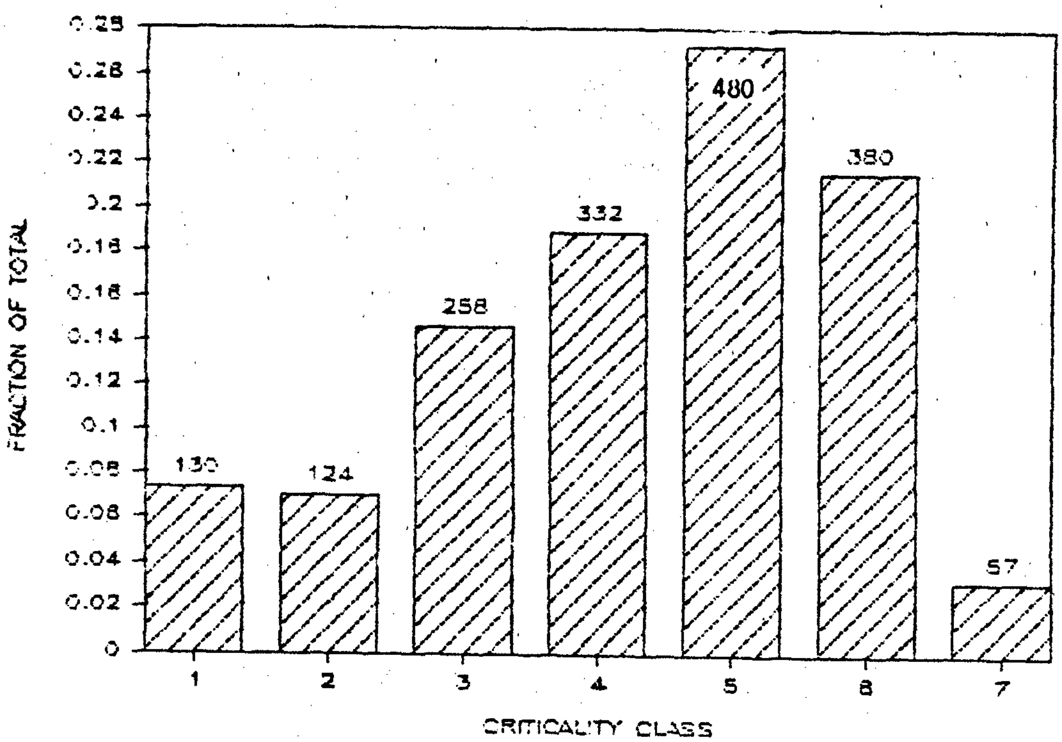


FIGURE 4 - 4 DISTRIBUTION OF FIRST YEAR FAILURES BY CRITICALITY
(Excluding First Month Failures)

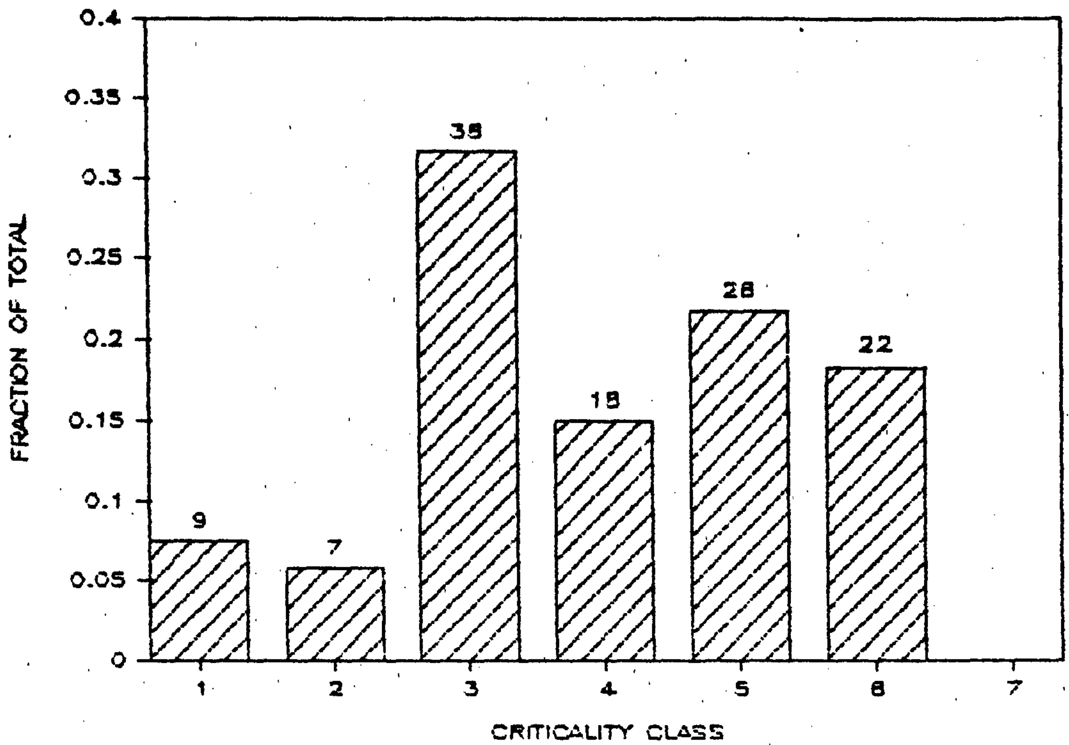


FIGURE 4 - 5 DISTRIBUTION OF FAILURES AFTER 5 YEARS BY CRITICALITY

2. Guidance and stabilization: used for initial satellite guidance in the ascent phase and orbit acquisition. It may then be used for keeping a despun platform stationary with respect to the earth and to avoid sensor lock on the sun and moon. It provides firing pulses to the propulsion subsystem and is involved in spacecraft stabilization, orbital drift corrections, and mid-course corrections.

3. Electrical power and distribution (including solar cells, batteries and thermionic power supplies): generates, stores, conditions, and distributes electrical power to the other subsystems.

4. Visual-IR sensors: Earth measurement and observation in the IR and visual spectrum (e.g., spectrophotometers, radiometers, scanning and chopping interferometers, and vidicon cameras

5. Data management (including CPUs, timers, and memory): stores and processes instructions, data, constants, and other parameters. It also includes software packages and timing functions.

6. Thermal: regulates the temperature in various compartments of the satellite by means of thermostats, heat pipes, louvres, heaters, coatings and cryogenics.

7. Communication payload: payload on board communication satellites, including antenna pointing and de-spin provisions

8. Specialized payloads: Primarily scientific and surveillance payloads not included in other payload categories.

9. Propulsion: furnishes thrust for orienting the spacecraft and correcting orbital drift.

10. Structural: consists of the primary structure, protective coverings, separation mechanisms, deployment devices, and ordnance.

11. Navigation payload: payload on board navigation satellites

The telemetry, power distribution, guidance, thermal control, and propulsion subsystems are present on all missions. Visual-IR sensors and special payloads were deployed on scientific, meteorological, reconnaissance, earth resources, and surveillance satellites. The communication and navigation payload subsystems were used on communication satellites and navigation satellites, respectively.

Section 4.2.1 discusses the distribution of failures among these subsystems section 4.2.2 analyzes the time-dependence of failures by subsystems, section

4.2.3 investigates causes within some subsystems, and section 4.2.4 looks at groups of subsystems which are characterized by the predominance of electronic, electromechanical, or mechanical equipment. Table 4-2 summarizes the results of these analyses.

TABLE 4-2. SUMMARY OF SUBSYSTEM ANALYSES

SUBSYSTEM	ACRONYM	PREDOMINANT EQUIPMENT	DECREASING FAILURE RATE*	PRIMARY FAILURE MECHANISM**
Telemetry	TELM	Electronic	Yes	Design/Envmt
Guidance	GUID	Electromechanical	Yes	Design/Envmt
Power	POWR	Electromechanical	Yes	Design/Envmt Parts/Quality
Vis.-IR Sensors	VI-S	Electronic	Yes	Design/Envmt Parts/Quality
Data Mgmt.	DATA	Electronic	Yes	Design/Envmt Parts/Quality
Thermal	THER	Mechanical	No	Design/Envmt
Comm. Payload	COMM	Electromechanical	No	Parts/Quality
Special	SPEC	Electromechanical	Yes	Design/Envmt Parts/Quality
Propulsion	PROP	Mechanical	No	Design/Envmt
Structural	STRUC	Mechanical	No	Design/Envmt
Nav. Payload	NAV	Electronic	No	Design/Envmt

* Failure rate (i.e., no. of subsystem failures per mission per year) that shows a statistically significant decrease over time as measured by a correlation coefficient above 0.7 (see section 4.2.2)

** Known failure mechanisms were divided into three overall categories: design/environment, parts/quality, and other (see section 4.2.3)

The characterization of the commuication payloads as an electromechanical system may at first appear puzzling. The payload includes in many cases the sliprings which provide the connection to the despun portion of the satellite and in other instances the steering mechanism for antennas.

4.2.1 Distribution of Subsystem Failures

Figure 4-6 shows the distribution of all failure reports by subsystem. The subsystems which have the most failures are all complex electronic or electromechanical systems. The low failing subsystems include several that are active for only a small portion of the total mission time, such as propulsion and the deployment portion of the structural subsystem.

Figures 4-7 and 4-8 show the initial (first month on orbit) and first year distribution of subsystem failures. Failures in the visual-IR subsystem make up a larger portion of the earlier failures than in the total population whereas the power distribution and communication payload failures comprise a smaller fraction. In the structural subsystem the failure of deployment mechanisms is clearly responsible for the unreliable operation during the initial month on orbit.

Figure 4-9 shows that excess contributions to failures after five operational years are distributed in the opposite way: power and communications payload are high and the structures subsystem is low. Wearout effects in batteries, solar cells, and traveling wave tubes are believed to be responsible for much of the unreliability of the former two subsystems. Wearout or depletion effects may also be responsible for the relatively large number of failures associated with the propulsion subsystem. The small contribution of the structures subsystem is due to the static role of the structural components in the steady state orbital phase. In Figures 4-10 through 4-13 the failure contribution of subsystems are divided into pre-1977 and later programs (see Section 3.3). Because failures from the pre-1977 programs make up approximately 75% of the data base, the similarity of their failure distributions to the overall sample is not surprising. The failures from the late programs show a higher proportion associated with the visual-IR, data management, special, and navigation payload subsystems. The former two can be explained by both the larger number and increasing complexity of such systems on later spacecraft; the latter two can be explained by the larger

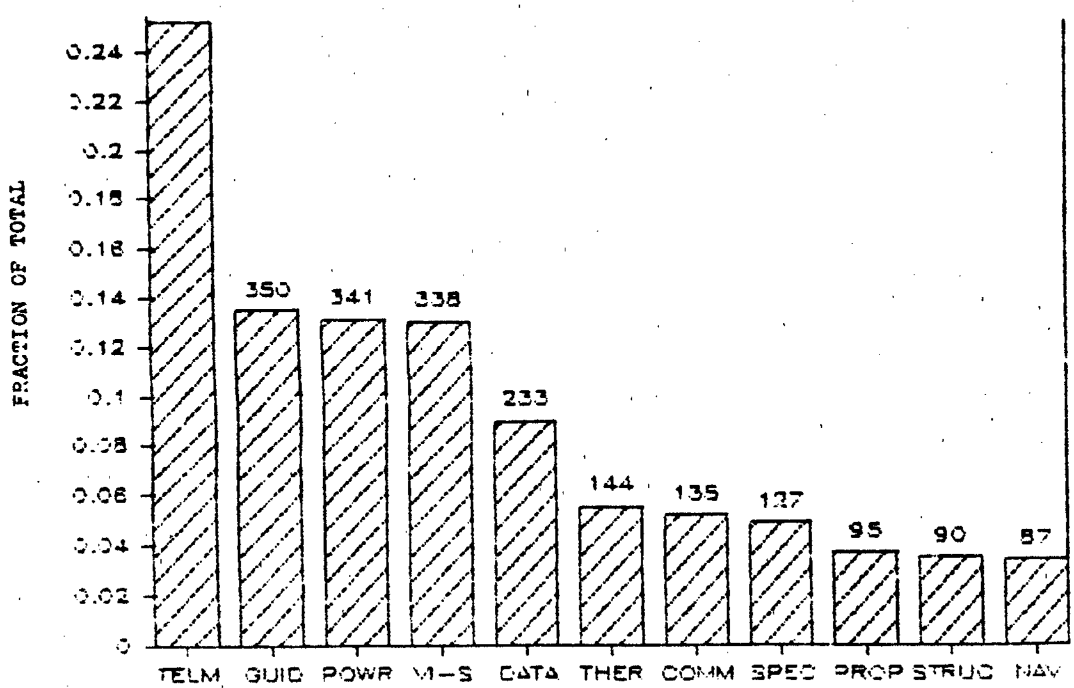


FIGURE 4 - 6 DISTRIBUTION OF ALL FAILURES BY SUBSYSTEM

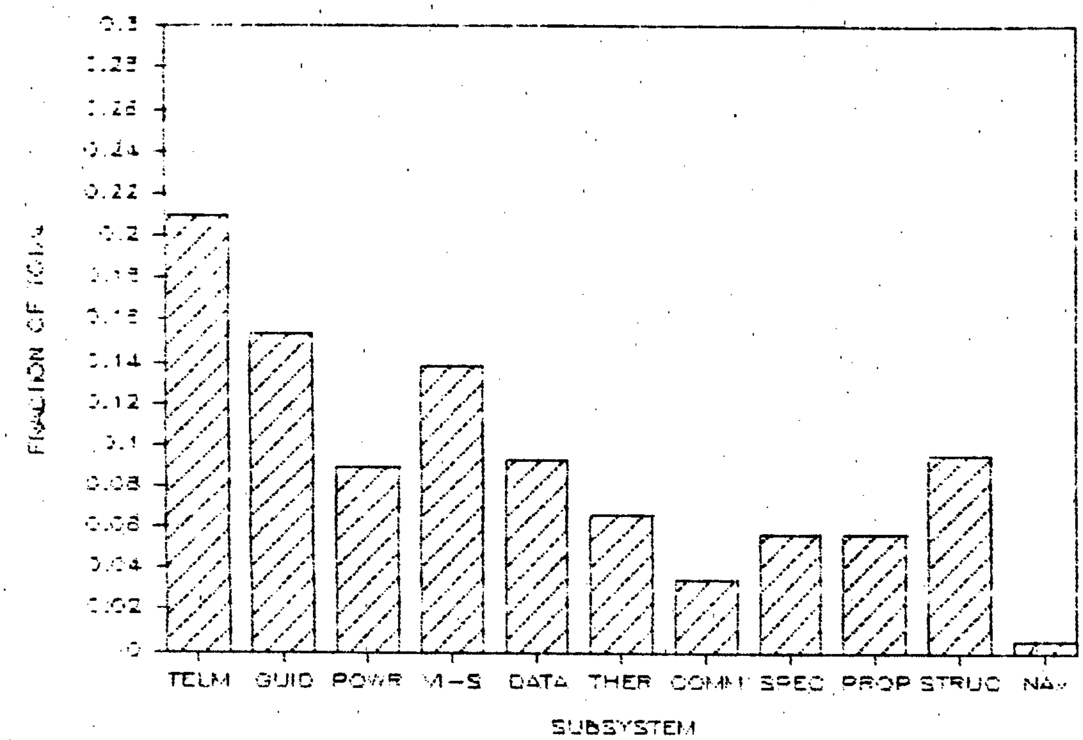


FIGURE 4 - 7 DISTRIBUTION OF FIRST MONTH FAILURES BY SUBSYSTEM

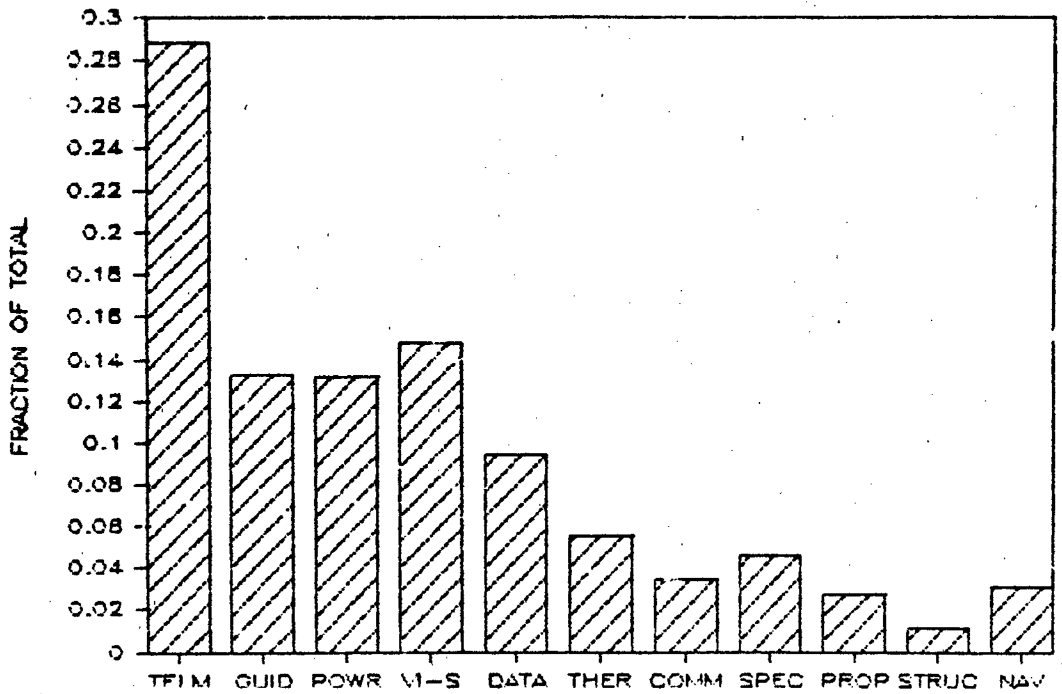


FIGURE 4 - 8 DISTRIBUTION OF FIRST YEAR FAILURES BY SUBSYSTEM

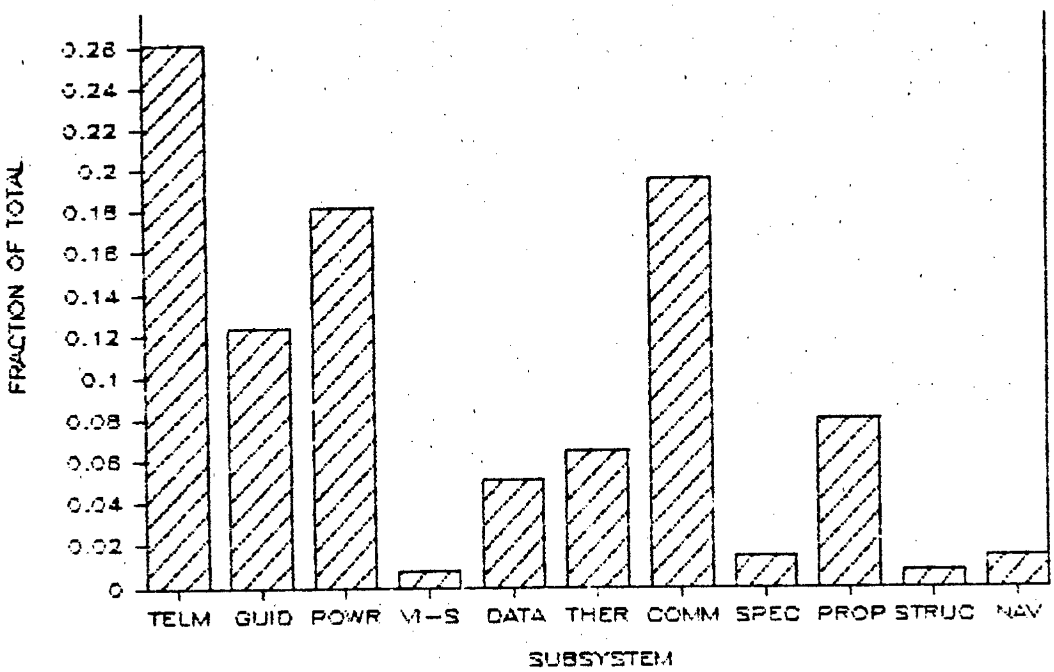


FIGURE 4 - 9 DISTRIBUTION OF FAILURES AFTER 5 YEARS BY SUBSYSTEM

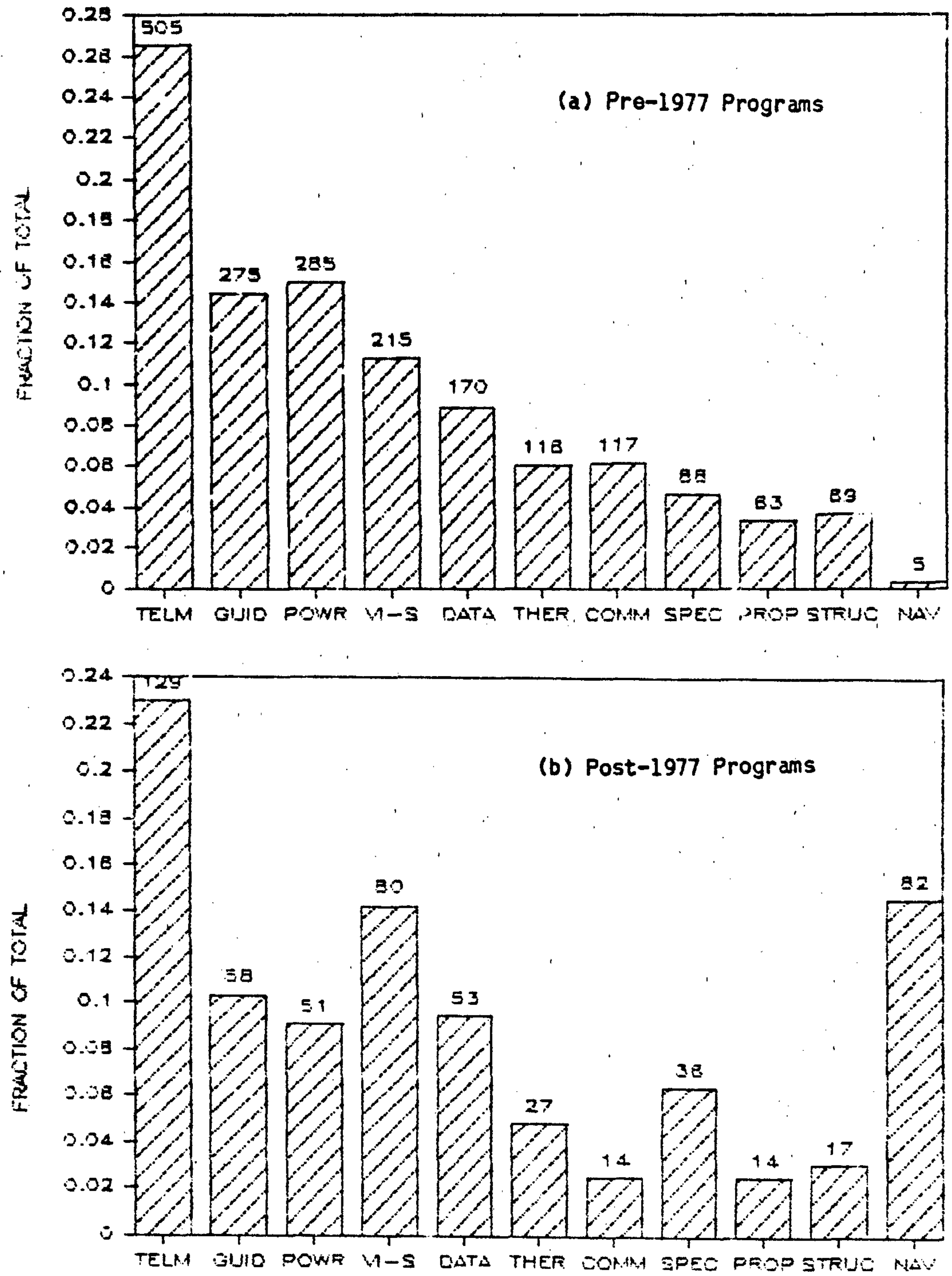


FIGURE 4 - 10 DISTRIBUTION OF ALL FAILURES BY SUBSYSTEM AND LAUNCH DATE

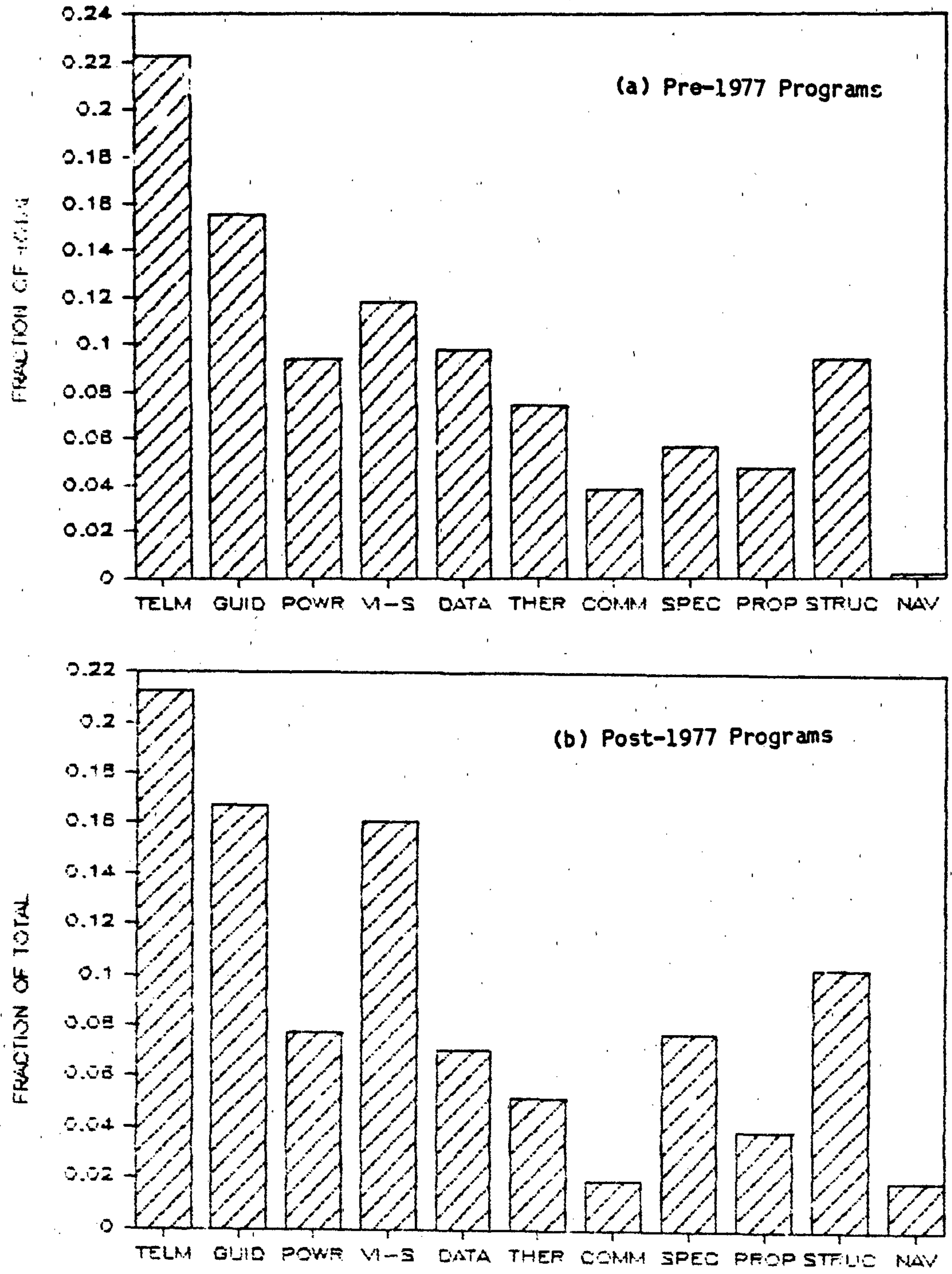


FIGURE 4 - 11 DISTRIBUTION OF FIRST MONTH FAILURES BY SUBSYSTEM AND LAUNCH DATE

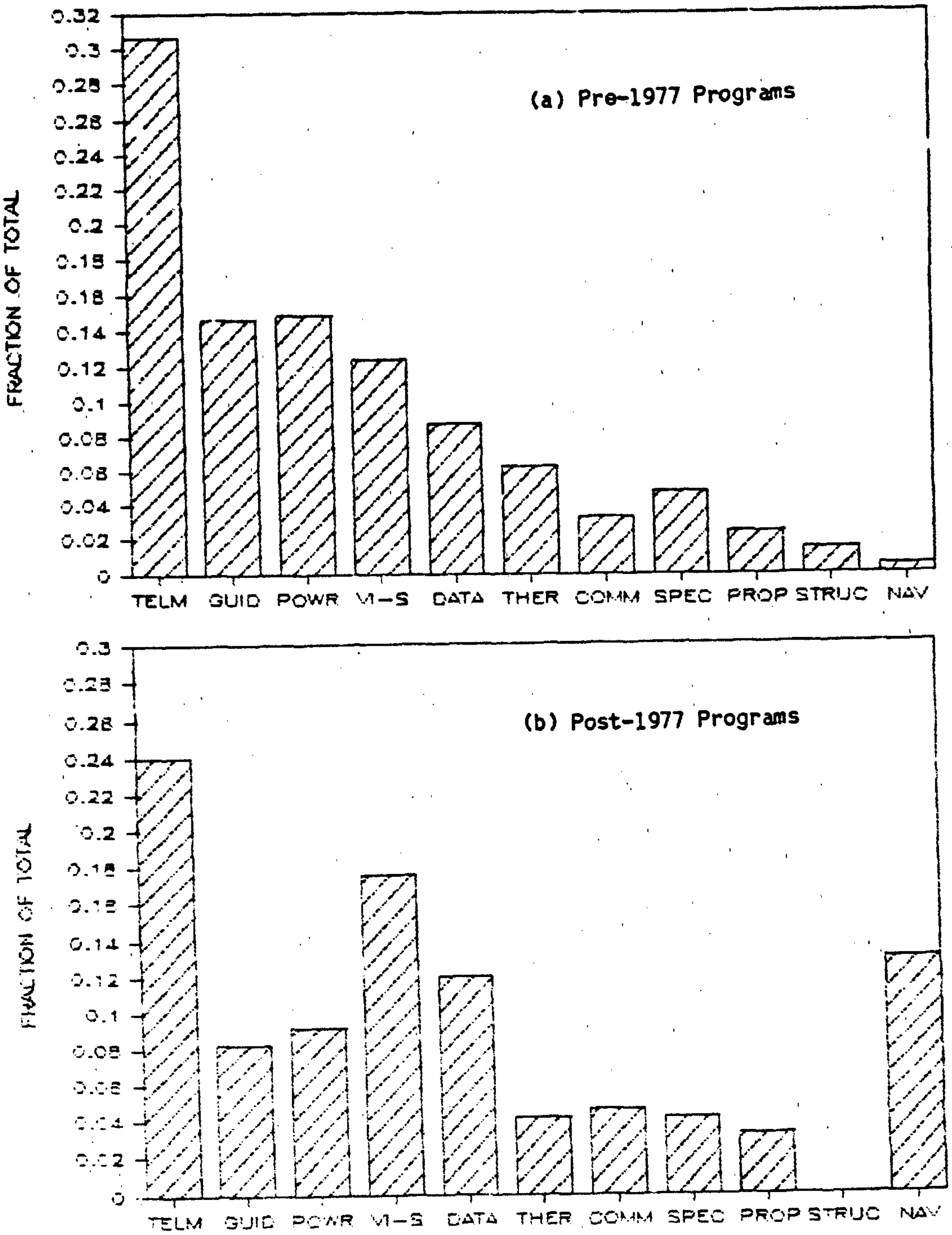


FIGURE 4 - 12 DISTRIBUTION OF FIRST YEAR FAILURES BY SUBSYSTEM AND LAUNCH DATE

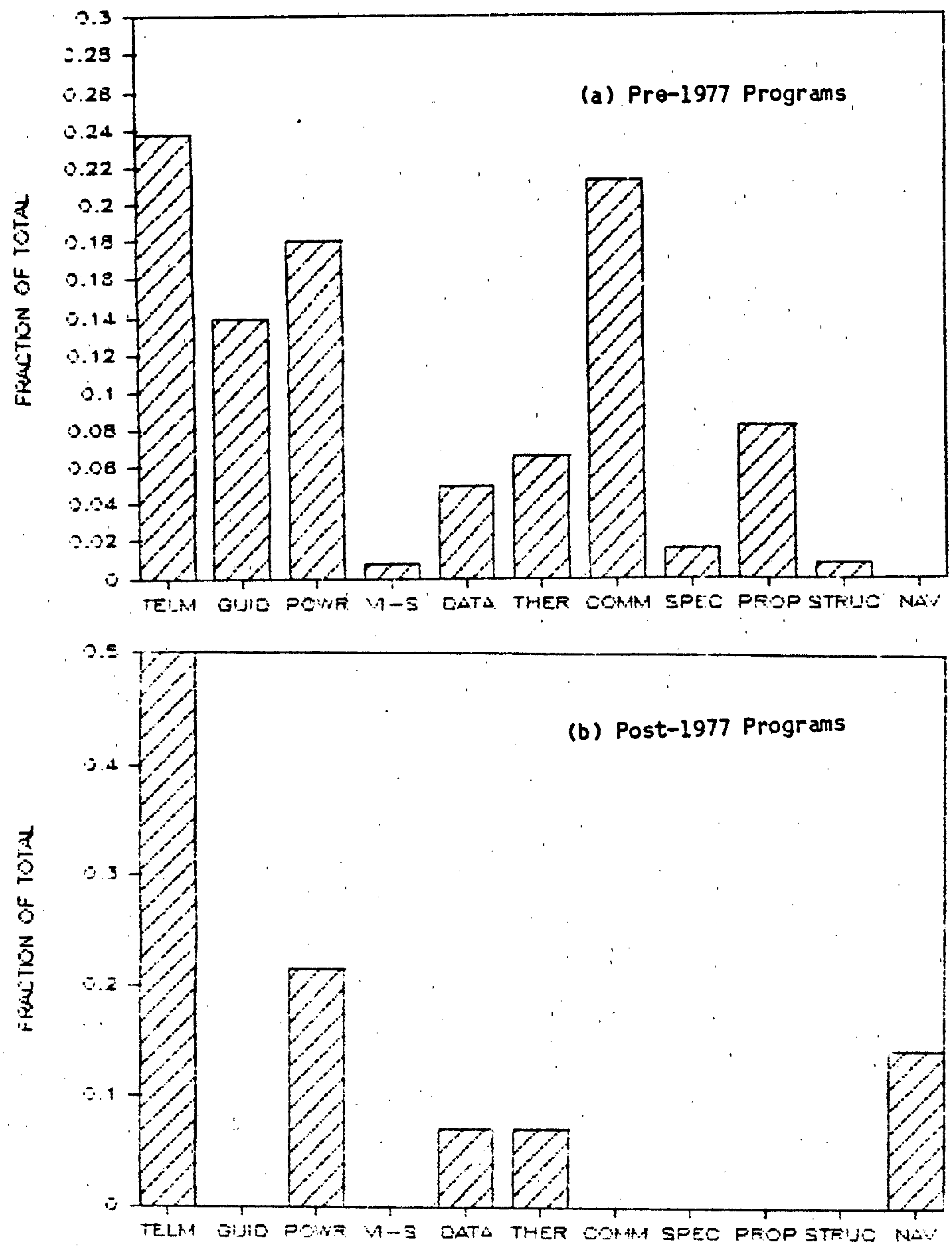


FIGURE 4 - 13 DISTRIBUTION OF FAILURES AFTER 5 YEARS BY SUBSYSTEM AND LAUNCH DATE

number of relevant missions (i.e., the GPS constellation).

The later programs demonstrate the progress made in the implementation of mature subsystems such as guidance, electrical power supply and distribution, or communication payloads. The fraction of these failures is lower than in the pre-1977 programs. The distribution of failures in other subsystems is approximately the same for both the earlier and later programs. The lack of failure reports from some subsystems in the late programs for orbit life of five years or more (Figure 4-13B) may be due to the small number of satellites that have completed the required lifetime.

4.2.2 Time Dependence

Figures 4-14 through 4-24 display failure ratios (failures per mission per year) of individual subsystems as a function of time on orbit. The data have been calculated and smoothed as described in Appendix A. The more frequently failing subsystems (telemetry, guidance, electrical power, visual/IR sensors, and data management) have decreasing failure ratios, i.e., their reliability improves over time. However, most other subsystems (thermal, communication payloads, propulsion, and navigation payloads) do not exhibit such behavior.

Table 4-3 shows data for linear regressions within each subsystem on failure ratio versus time. The table shows that where the slopes are statistically significant (defined as a coefficient of determination, R^2, of 0.5 or greater) they are always negative. Furthermore, statistically significant negative slopes are primarily found among the subsystems with the greatest number of failures.

The communication payload (Figure 4-20) and the propulsion subsystem (Figure 4-22) exhibit wearout effects which are consistent with the known equipment characteristics of these functions. Several other subsystems show no significant time dependency of the failure ratio after an initial period of high failures. The thermal subsystem (Figure 4-19), the structural subsystem (Figure 4-23), and the navigation payload (Figure 4-24) are among these.

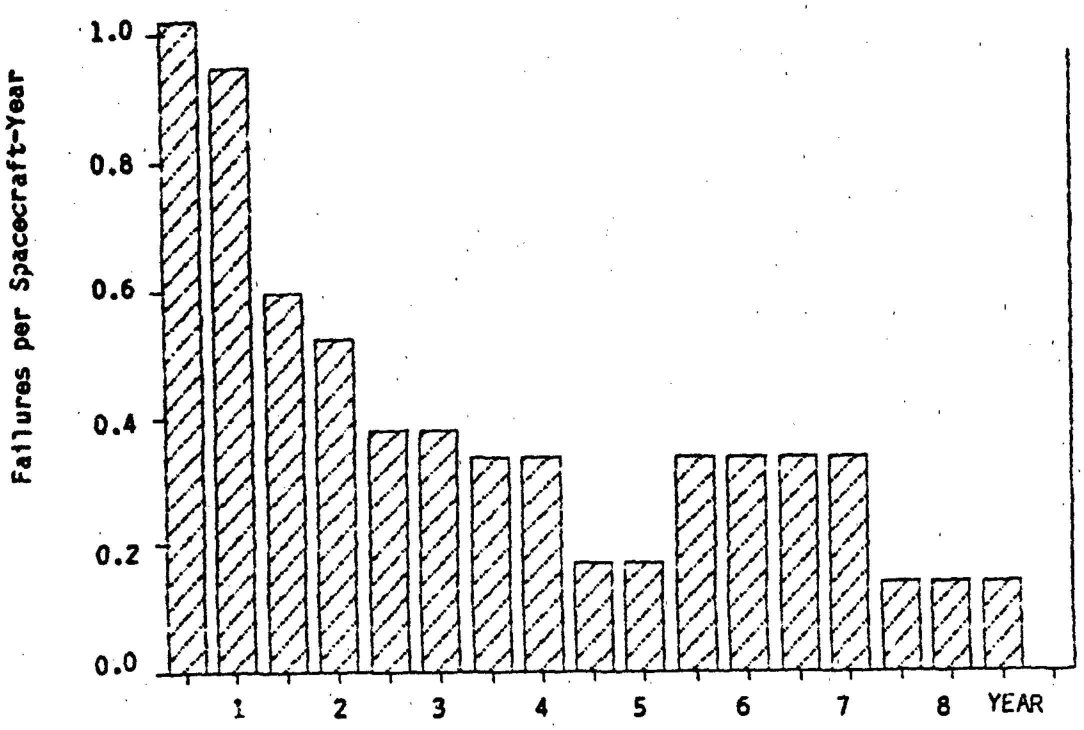


FIGURE 4 - 14 TIME TREND OF TELEMETRY SUBSYSTEM FAILURES

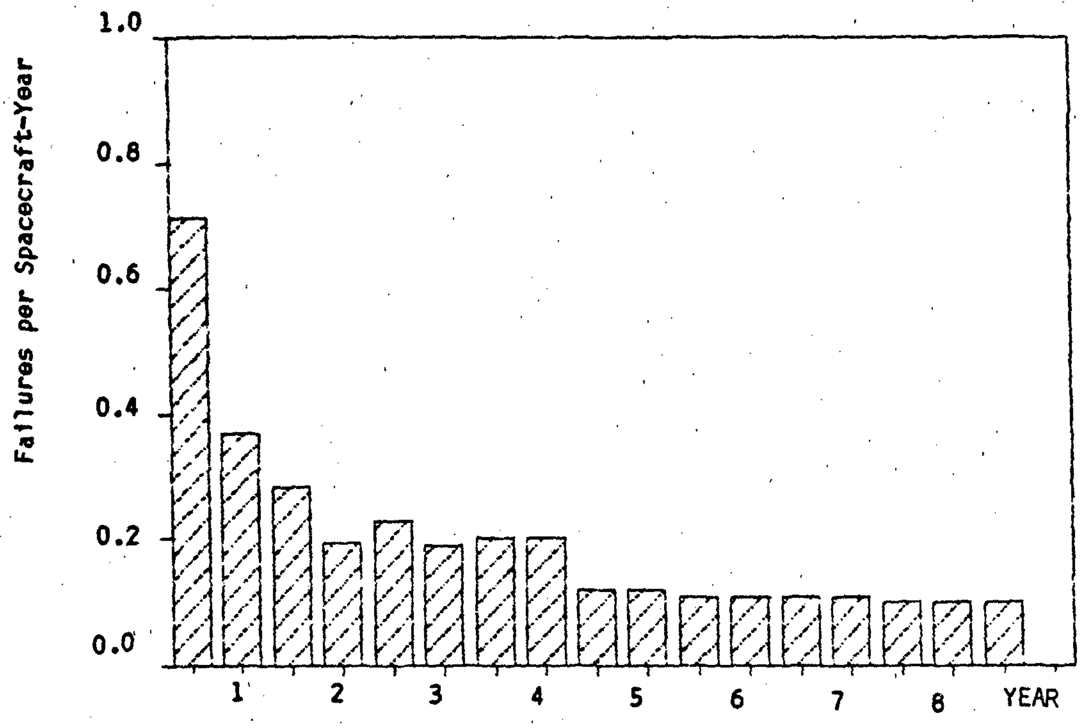


FIGURE 4 - 15 TIME TREND OF GUIDANCE SUBSYSTEM FAILURES

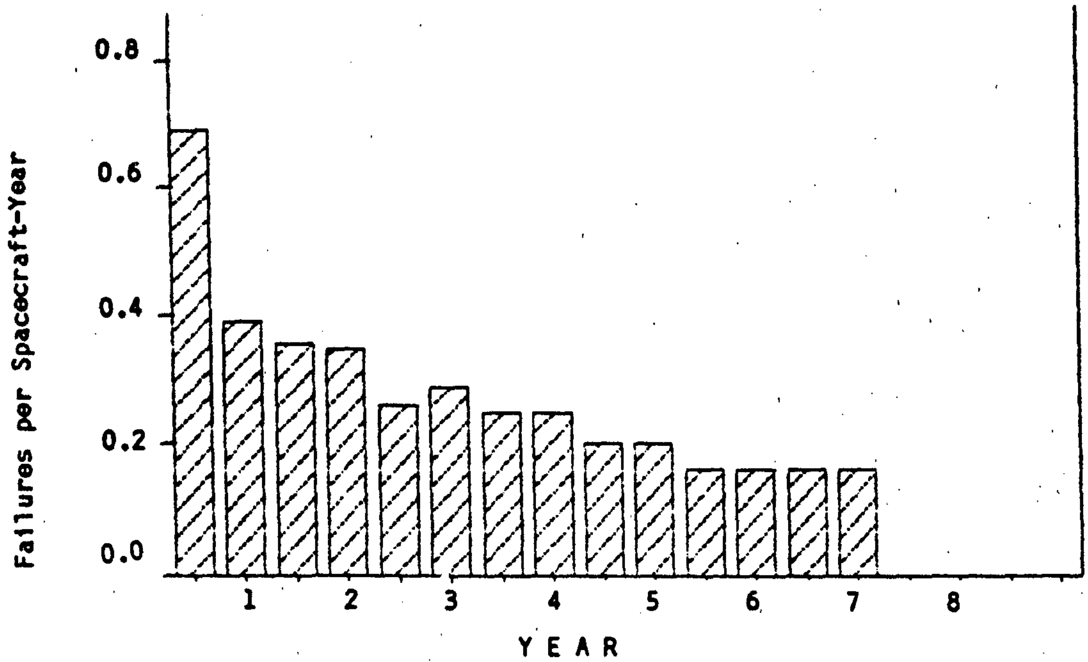


FIGURE 4 - 16 TIME TREND OF ELECTRIC POWER SUBSYSTEM FAILURES

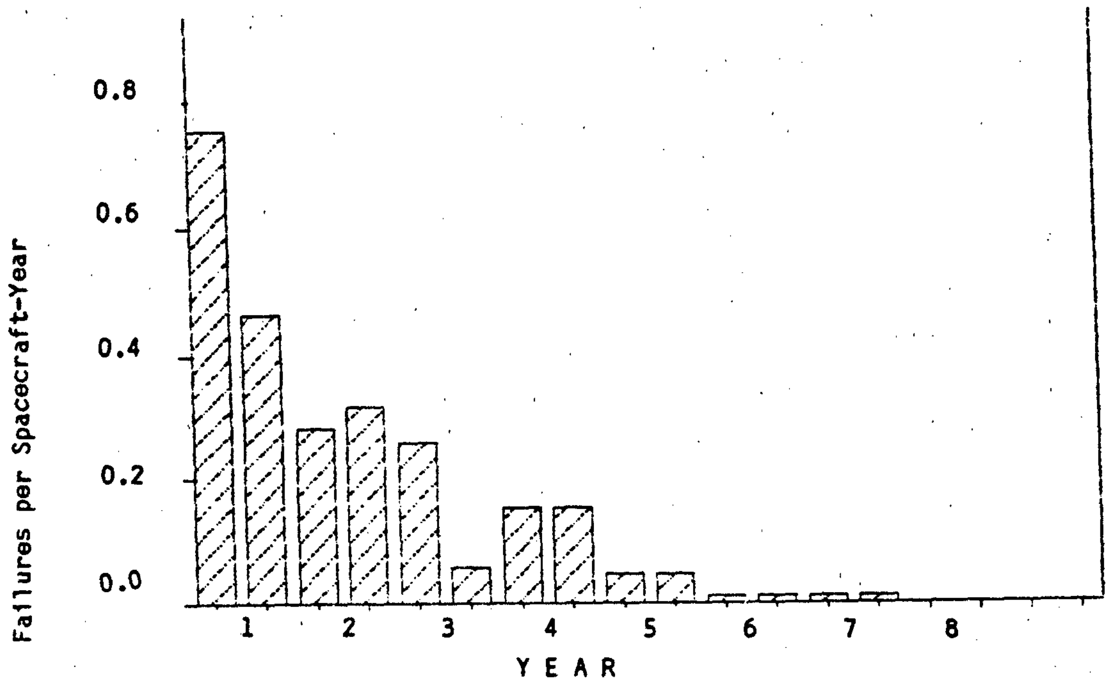


FIGURE 4 - 17 TIME TREND OF VISUAL & IR SENSOR SUBSYSTEM FAILURES

FIGURE 4 - 18 TIME TREND OF DATA MGMT. SUBSYSTEM FAILURES

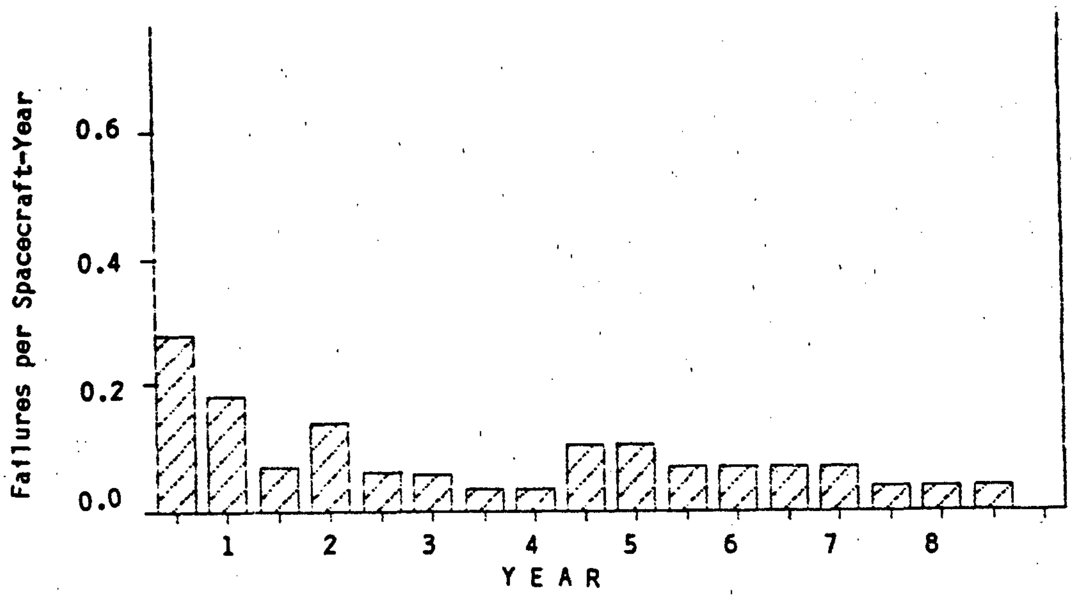


FIGURE 4 - 19 TIME TREND OF THERMAL SUBSYSTEM FAILURES

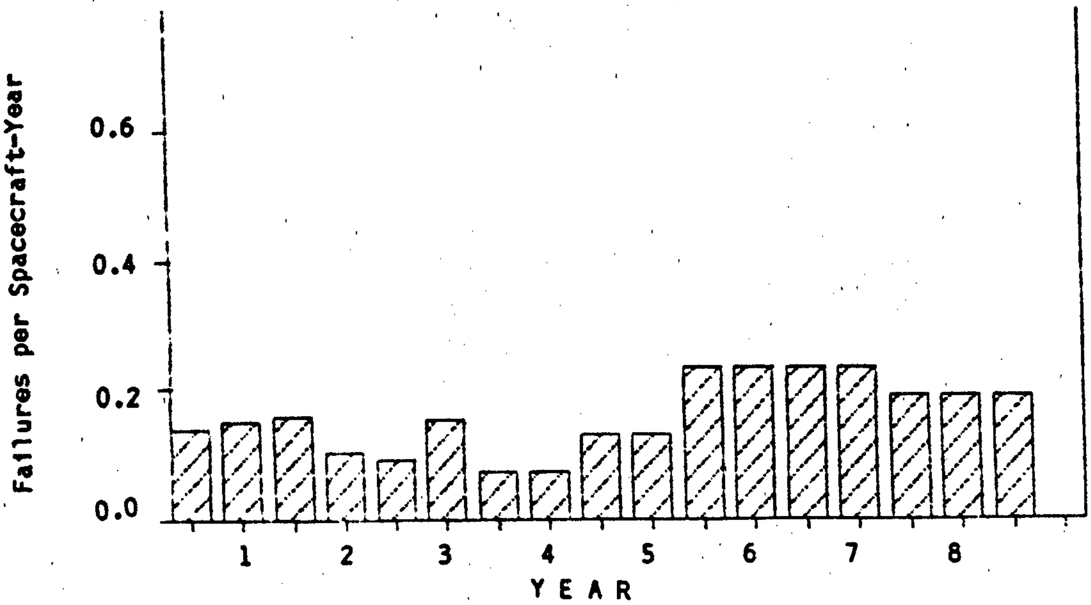


FIGURE 4 - 20 TIME TREND OF COMMUNICATION PAYLOAD FAILURES

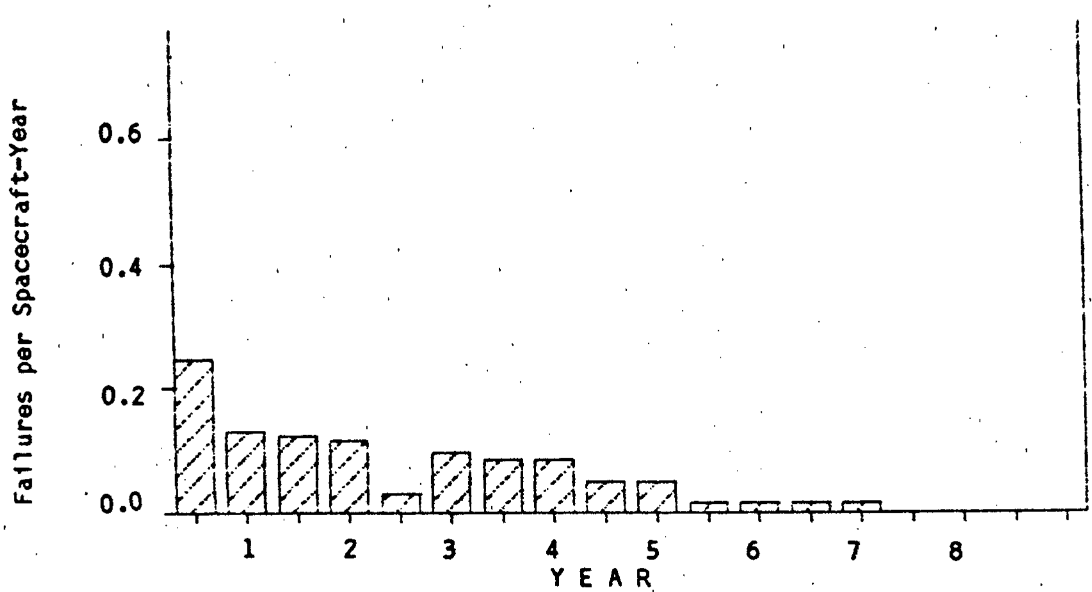


FIGURE 4 - 21 TIME TREND OF SPECIAL PAYLOAD FAILURES

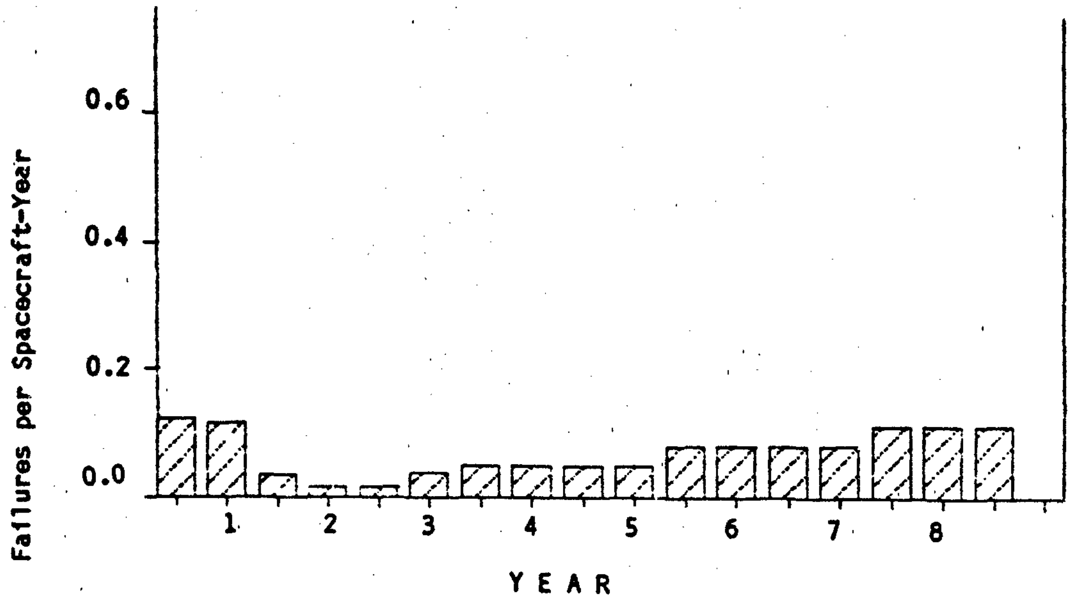


FIGURE 4 - 22 TIME TREND OF PROPULSION SUBSYSTEM FAILURES

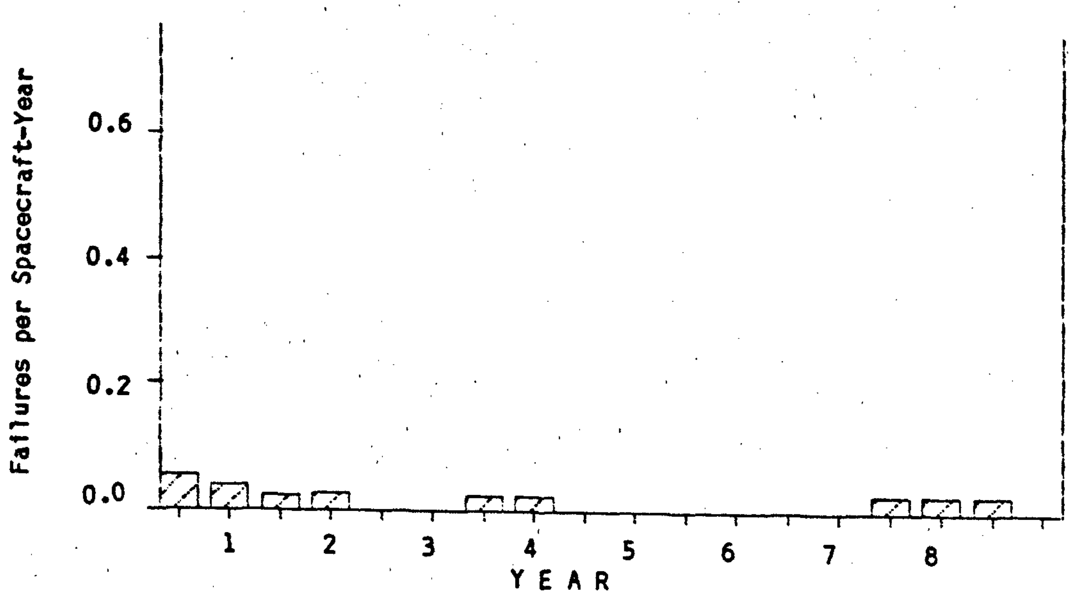


FIGURE 4 - 23 TIME TREND OF STRUCTURAL SUBSYSTEM FAILURES

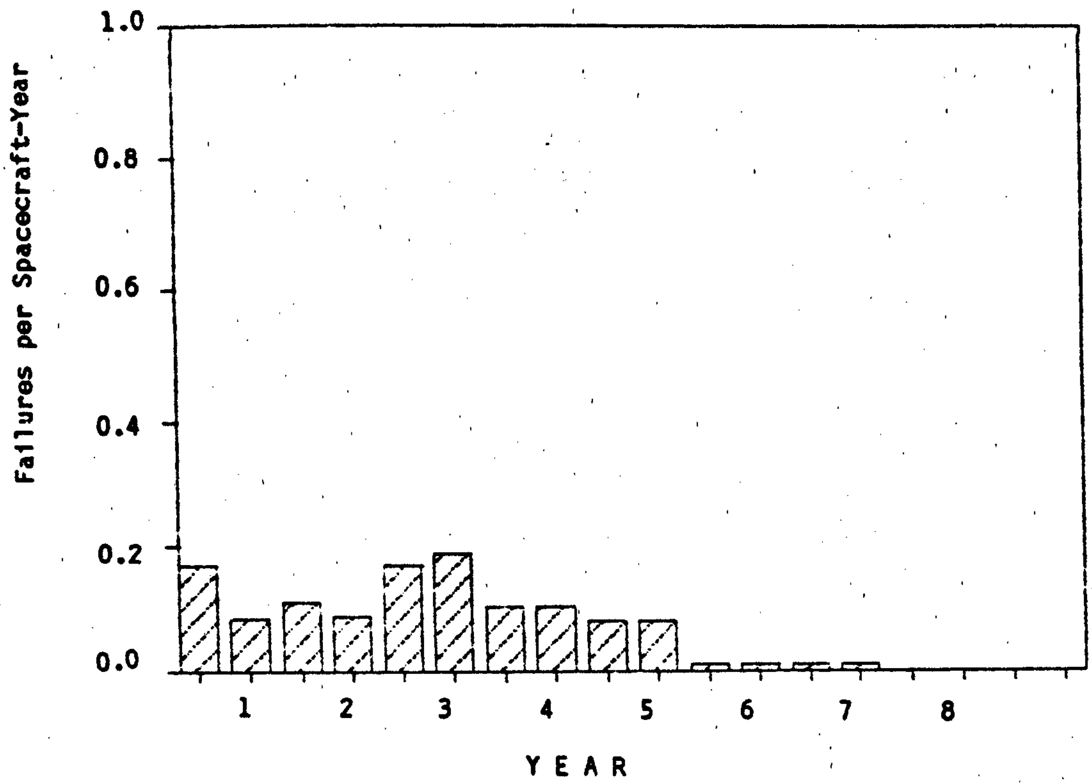


FIGURE 4 - 24 TIME TREND OF NAVIGATION PAYLOAD FAILURES

TABLE 4 - 3 LINEAR REGRESSION ON SLOPE OF FAILURE RATIO

SUBSYSTEM	R	Coeff of det.	Std Err.	Signf.*	Intercept**	Slope**
Telemetry	-.77	.60	.11	.0001	.43 (.05)	-.024 (.005)
Guidance	-.70	.49	.07	.0009	.22 (.03)	-.011 (.003)
Power	-.85	.73	.06	.0000	.24 (.03)	-.017 (.002)
Visual/IR Sensors	-.88	.78	.04	.0000	.24 (.04)	-.014 (.002)
Data Management	-.71	.51	.05	.0006	.15 (.02)	-.009 (.002)
Thermal Subsystem	-.67	.45	.03	.0018	.09 (.01)	-.005 (.001)
Communic. Payload	-.49	.24	.06	.0363	.04 (.03)	.006 (.003)
Special Payload	-.88	.77	.02	.0000	.09 (.01)	-.006 (.001)
Propulsion	.05	.00	.04	.8460	.04 (.02)	.000 (.001)
Structural	-.45	.20	.04	.0543	.05 (.02)	-.004 (.002)
Navig. Payload	-.65	.42	.03	.0028	.07 (.01)	-.004 (.001)

* F-distribution probability that such results could have been due to chance

** Quantities in parentheses are standard errors of the estimate. Units are failures per 6-months per mission

4.2.3 Causes of Failure within Subsystems

The following discussion is concerned with causes of failure within each subsystem. It supplements Section 3.4 in which the contribution of subsystems to each cause category was investigated. The percentage contributions of major causes to failures within each subsystem are shown in Table 4-4. Design and environment failures are the most important contributors in most cases. The parts and quality cause is the most significant one for the communication payload and data management subsystems, both of which employ a large number of complex electronic components. The same pattern might be true for telemetry, visual/IR sensors and special payloads if the large percentage of unknown failures in these subsystems is mostly composed of parts and quality causes.

TABLE 4-4. COMPOSITION OF SUBSYSTEM FAILURES BY CAUSE

SUBSYSTEM	DESIGN	ENVMT	PARTS/ QUALITY	OPER/ OTHER	UNKNOWN
Telemetry	21.0%	18.3%	24.2%	9.7%	26.8%
Guidance	28.7%	20.7%	20.7%	11.1%	18.7%
Power	21.4%	20.4%	19.8%	17.8%	20.6%
Visual/IR	20.5%	23.0%	15.2%	8.9%	32.4%
Data Mgt	18.8%	17.3%	33.7%	19.9%	10.3%
Thermal	33.5%	30.5%	19.5%	3.8%	12.7%
Communic.	7.3%	13.8%	38.8%	6.1%	24.0%
Spec Pyld	19.0%	24.4%	21.4%	10.1%	25.0%
Structures	43.0%	17.4%	24.0%	5.8%	9.9%
Nav. Pyld	31.6%	26.6%	22.2%	10.8%	8.9%

4.2.4 Electronic, Electromechanical and Mechanical Subsystems

In order to investigate whether the time-dependent failure behavior and the causes of failures are affected by the predominant component type or function, subsystems were grouped into the following three categories:

ELECTRONIC	Telemetry, Command, and Control Visual-IR Sensors Data Management Navigation Payload
ELECTROMECHANICAL	Guidance Special Payloads Power Communication Payload
MECHANICAL	Thermal Propulsion Structural

Figure 4-25 shows the contribution of each of the groupings to the failure causes. Figures 4-26 through 4-28 show the relative importance of various causes within each of the groupings. Although design and environment failures were the primary causes in all categories, they were most important for mechanical subsystems. Unknown causes were an important contributor to the electronic category and reflect their more complicated failure modes.

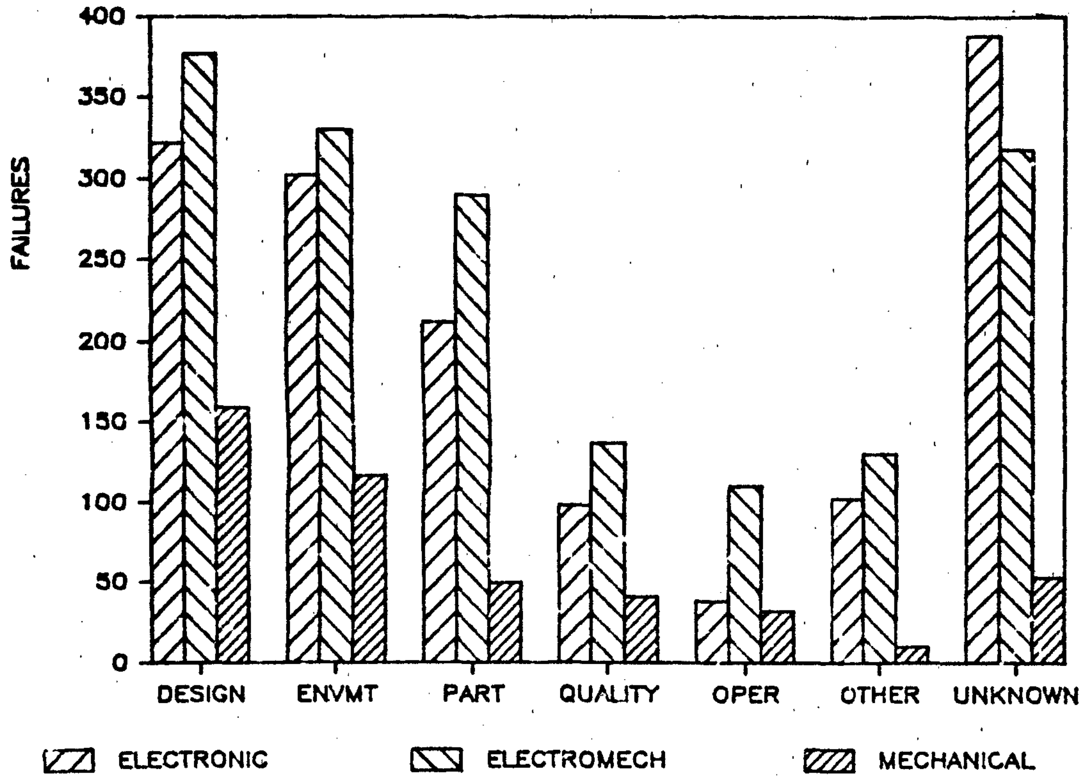


FIGURE 4 - 25 FAILURE CAUSES BY SUBSYSTEM CATEGORY

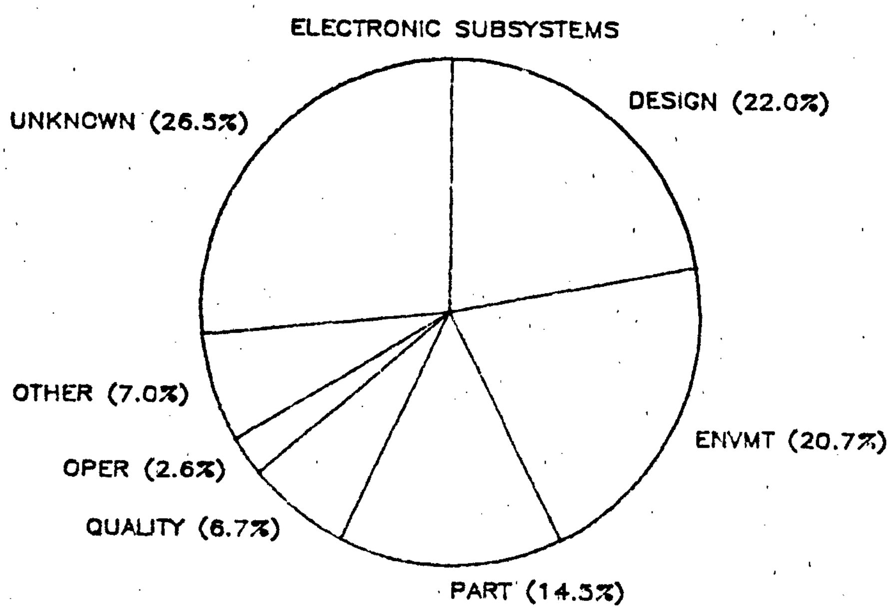


FIGURE 4 - 26 FAILURE CAUSES IN ELECTRONIC SUBSYSTEMS

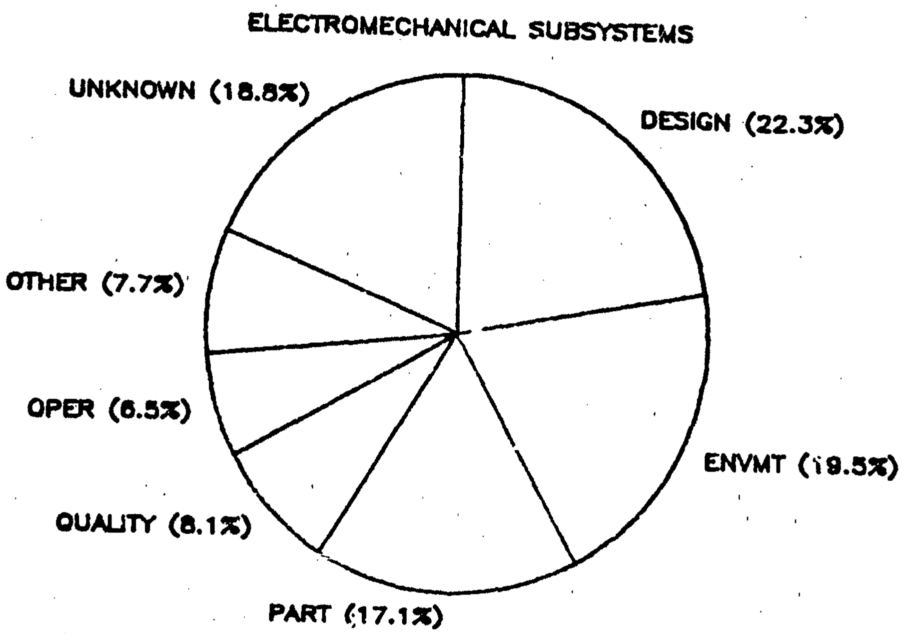


FIGURE 4 - 27 FAILURE CAUSES IN ELECTROMECHANICAL SUBSYSTEMS

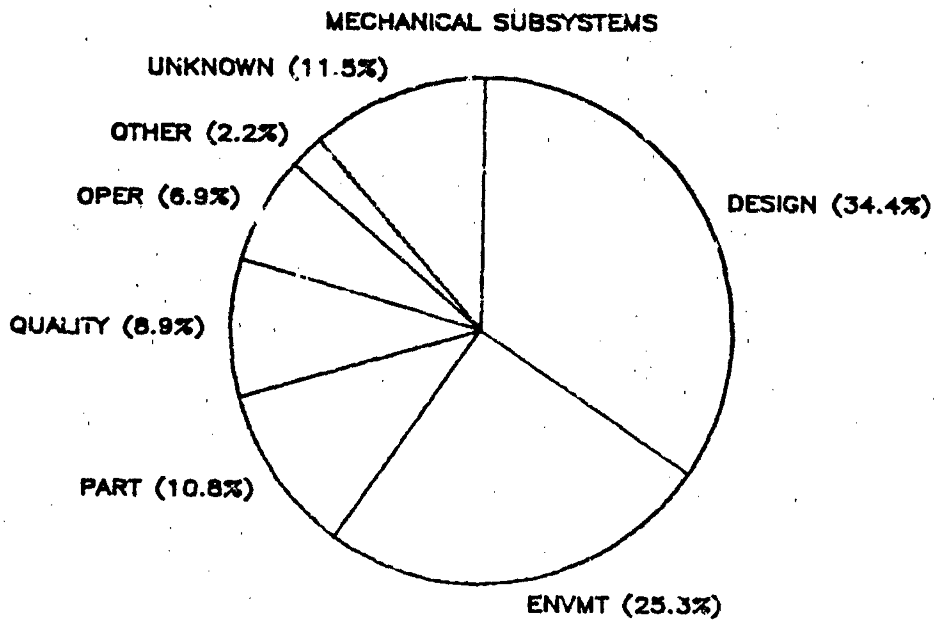


FIGURE 4 - 28 FAILURE CAUSES IN MECHANICAL SUBSYSTEMS

Figures 4-29 through 4-31 show the time dependent behavior of electronic, electromechanical, and mechanical subsystems. There is no statistically significant slope in the mechanical category (this is consistent with earlier results), but a definite negative slope is present in the case of electronic subsystems which indicates a decreasing hazard. A smaller, but still decidely negative slope) is evident for electromechanical subsystems. This result can be explained by the presence of both electronic (decreasing hazard) and mechanical (non-decreasing hazard) failure mechanisms. There is some evidence of wearout among the mechanical systems, much of it apparently due to the propulsion components.

Since the primary purpose of this investigation is to to provide a basis for improved reliability prediction for spacecraft within the scope of MIL-HDBK-217, and since the latter deals specifically with electronic equipment, the question arises whether the time dependency aspects of the prediction procedures should be based on the total population of failure reports or specifically on those dealing with electronic equipment. In this connection it is necessary to make a distinction between electronic equipment and electronic systems or subsystems as classified in the earlier portions of this section. Electronic equipment is the preponderant contributor to failures in both the electronic and the electromechanical subsystems described here, but it is not a significant contributor to failures in the mechanical subsystems. A comparison of the failure ratio for electronic and electromechanical systems with that for the entire population is shown in Figure 4-32. It is seen that the general time trend (which determines the b parameter of the Weibull distribution) is identical for both populations. On detail inspection it will be noted that the difference between the two graphs in Figure 4-32 is greater at the beginning and at the end than in the middle. This is due to the large proportion of failures during the first year and to the wearout effects that can be seen starting after the fourth year in Figure 4-31.

The primary reliability prediction procedure described in the following section is based on the Weibull b parameter derived for the entire population

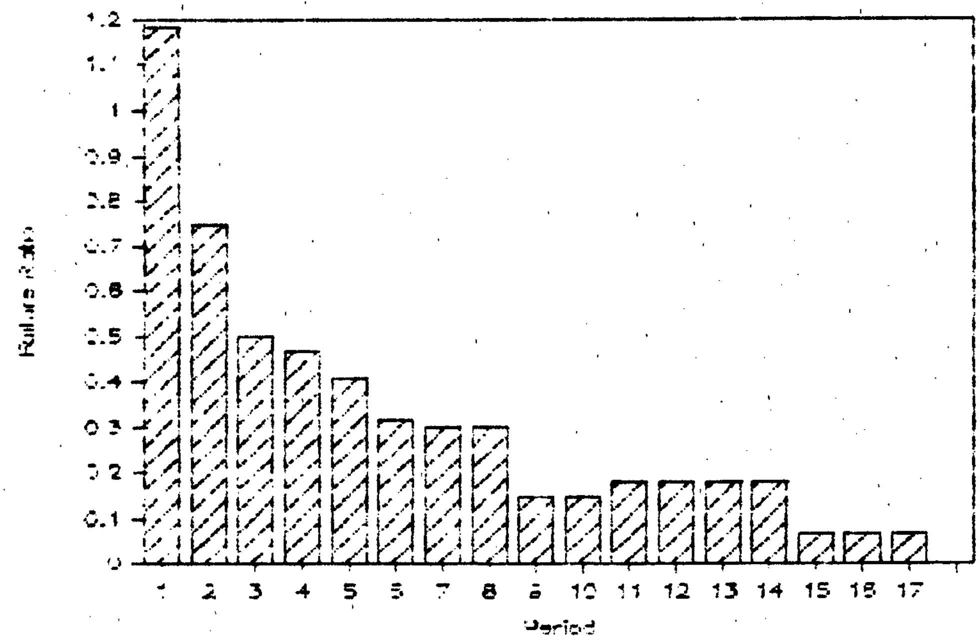


FIGURE 4 - 29 TIME TREND OF FAILURES IN ELECTRONIC SUBSYSTEMS

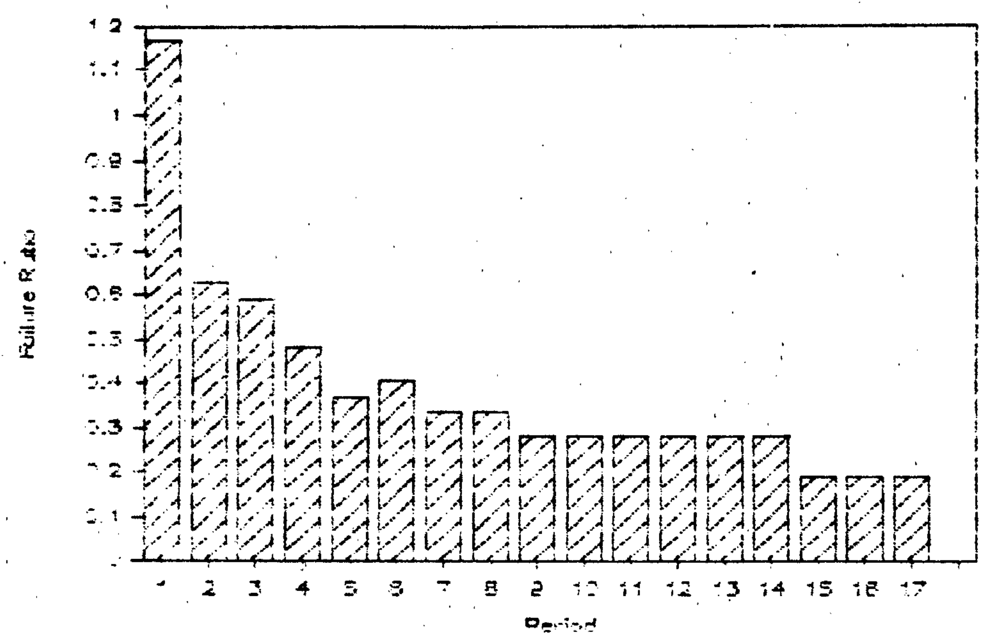


FIGURE 4 - 30 TIME TREND OF FAILURES IN ELECTROMECHANICAL SUBSYSTEMS

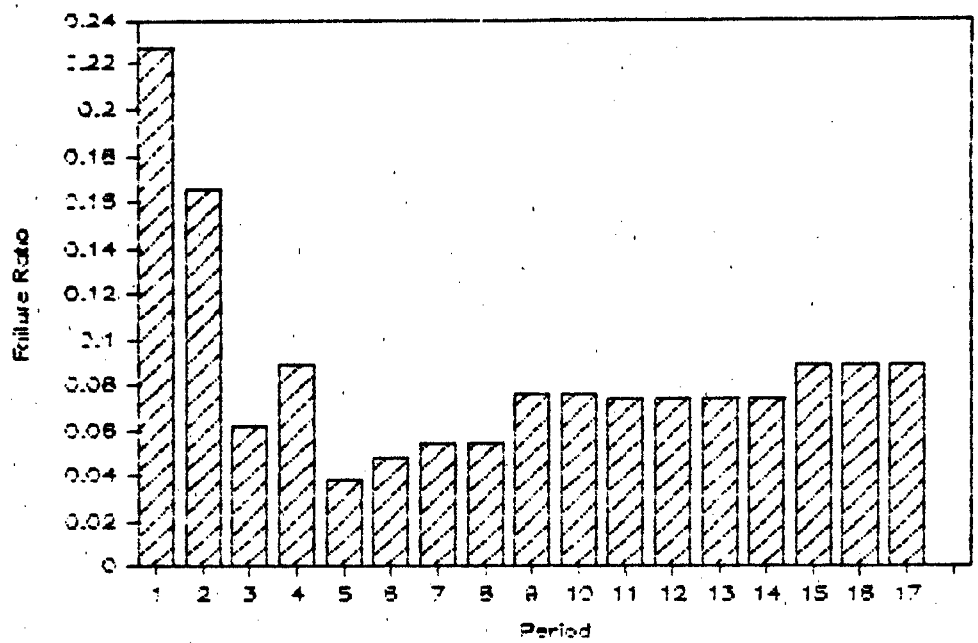


FIGURE 4 - 31 TIME TREND OF FAILURES IN MECHANICAL SUBSYSTEMS

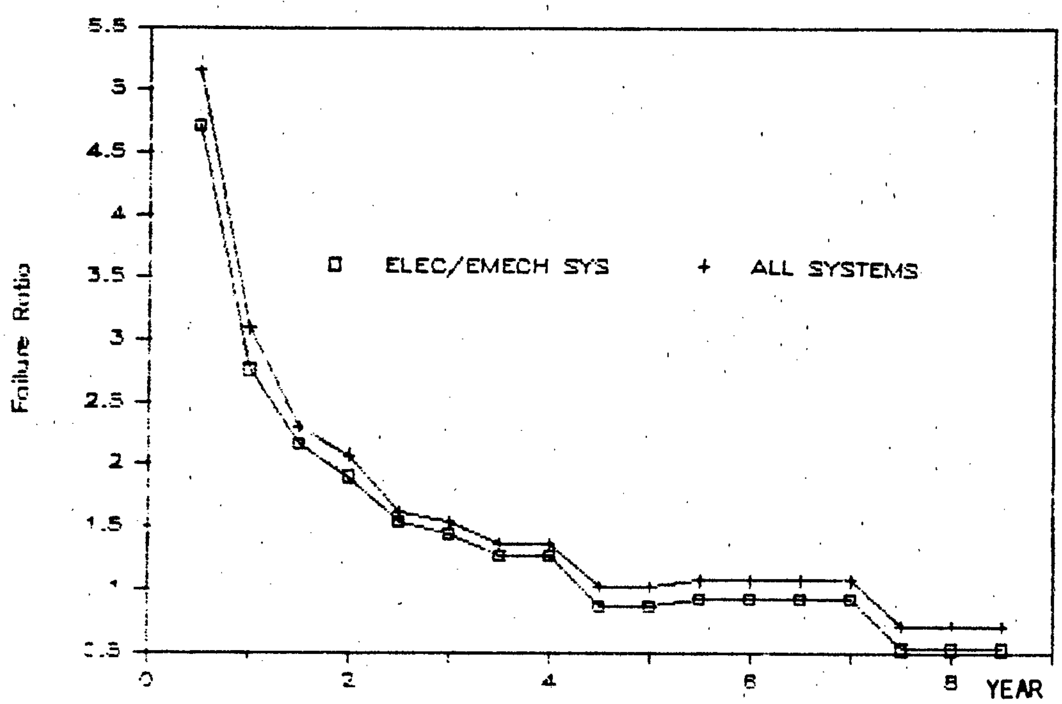


FIGURE 4 - 32 TIME DEPENDENCY OF ELECTRONIC FAILURES COMPARED WITH ALL FAILURES

because this was simpler to implement and because the difference in the time relationships between the two graphs in Figure 4-32 is so small. A further consideration is that the validation of the model had to be made against data for an entire spacecraft for which no breakdown between electronic and other equipment was available.

4.3 Complexity Effects in Selected Subsystems

This section explores the relationship between reliability and subsystem complexity. Because complexity involves many factors (e.g., number of components, interconnections, constraints), it is difficult to develop a direct measure that is unambiguous. However, other more easily determined indicators may serve as useful surrogates. Table 4-5 shows such indicators for subsystems where design and environment were the most important causes.

TABLE 4-5. COMPLEXITY INDICATORS FOR SELECTED SUBSYSTEMS

TELEMETRY	Presence of a computer
GUIDANCE	Nature of stabilization (i.e., 3-axis, spin-stabilized, or gravity stabilized)
POWER	Capacity
THERMAL	Active or passive

To determine whether the presence or absence of complexity indicators had a statistically significant effect on the failure ratio the following tests were performed:

- For discrete variables, missions were grouped into those using or not using the indicator. Upper and lower 90% confidence bounds on the failure ratio were computed for each group. If these intervals did not overlap, then the indicator was considered significant. The technique used to determine these confidence intervals is taken from Epstein [EPST60].

- For continuous variables, a linear regression of MTBF versus the value of the variable was performed. If the coefficient of determination (R^2) was greater than 0.5, then the indicator was considered significant.

In most of the subsystems investigated the more complex implementation was associated with a much higher failure rate. It is realized that the complexity is introduced because it is essential for functional or accuracy requirements of the mission. Nevertheless, the significantly lower reliability of the more complex subsystems should be considered in any trade-offs.

4.3.1 Telemetry

Data on whether the Telemetry, Tracking, and Control system included either an on-board CPU or a hardwired encoder/decoder unit was available for a total of 101 flights comprising almost 3800 orbital months. As shown in Table 4-6, CPU-based systems had more than five times the failure rate of the hardwired systems (based on point estimates of lambda). Because many computer-related failures are less severe than those occurring on totally hardwired systems, a second analysis was performed on only failures of the three most critical classes. These results, also shown in Table 4-6, confirm that there is a significant difference between the failure rates of CPU-based and hardwired systems although the difference (in both relative and absolute terms) is not as large as when all failures are considered. The results of this analysis demonstrate that complexity, as manifested by the presence of an onboard CPU, affects the failure rates of the telemetry, tracking, and control subsystem.

Table 4 - 6 EFFECT OF COMPLEXITY ON TELEMETRY SUBSYSTEM

	ALL FAILURES		CRITICALITY 1-3 FAILURES	
	CPU	Hardwired	CPU	Hardwired
Time in Orbit, years	42	272	42	272
No. of Failures	98	110	32	45
No. of Flights	19	82	19	82
Lambda, per year				
Point Estimate	0.19	0.034	0.063	0.014
Lower Limit*	0.16	0.030	0.049	0.011
Upper Limit*	0.23	0.038	0.077	0.016

*90% Confidence Interval

4.3.2 Guidance

The complexity of guidance and stabilization subsystems was characterized by the satellite stabilization method, three-axis stabilization being the most complex, gravity stabilization being the least complex, and spin stabilization being of intermediate complexity. Data on the nature of the satellite stabilization system were available on a total of 180 flights and 6600 orbital months. Table 4-7 summarizes the results of the analysis. Using the decision rules defined at the beginning of this section, one can state that subsystems using 3-axis stablization had a significantly higher failure ratio than those using spin stabilization, and that the latter in turn had a much higher failure ratio than those using gravity stabilization.

TABLE 4 - 7 EFFECT OF COMPLEXITY ON THE GUIDANCE SUBSYSTEM

	3-axis	Spin	Gravity
Time in Orbit, years	132	363	53
No. of Failures	81	78	2
No. of Flights	56	78	46
Lambda, per year			
Point Est.	0.61	0.216	0.038
Lower Limit*	0.53	0.18	0.0072
Upper Limit*	0.70	0.24	0.090

*90% Confidence Interval

4.3.3 Power

The capacity of the power supply and distribution subsystem was not a good indicator for its failure rate. The coefficient of determination (R^2) was 0.0007, and the significance of the F-distribution was well below the 90% decision point. The probable explanation is that larger power supplies were placed on later satellites and therefore represented a more mature technology. Another factor is that larger capacity power systems do not necessarily involve a larger number or more complex components. Finally, the percentage of the power system capacity utilized may be less for large systems, thereby promoting higher reliability.

4.3.4 Thermal

The use of active thermal control (e.g., thermal louvers, heaters, etc.) versus total reliance on passive measures (reflective and insulating coatings, etc.) was the basis for determining thermal subsystem complexity. The sample consisted of 83 flights comprising close to 2900 orbital months. Table 4-8 shows that the point estimate of the failure rate for the passive systems was about one-quarter of that of the active subsystems. These conclusions are significant at well over the 90% level.

TABLE 4 - 8 EFFECT OF COMPLEXITY ON THE THERMAL SUBSYSTEM

	ACTIVE	PASSIVE
Time in Orbit, years	108	131
No. of Failures	35	11
No. of Flights	48	35
Lambda		
Point Estimate	0.32	0.084
Lower Limit*	0.25	0.053
Uppper Limit*	0.40	0.012

*90% Confidence Interval

4.4 Mission Effects

This section discusses the results of analyses by mission type based on the following four mission classifications:

NAVIGATION	Operational navigation satellites (excluding experimental launches such as NTS).
OBSERVATION	Meteorology, Earth Resource, Reconnaissance and Surveillance satellites — excludes experimental and research missions (which are included in the next category).
SCIENTIFIC	Experimental and Scientific launches including both NASA and DoD research (as opposed to operational) missions.
COMMUNICATION	Commercial and military communication satellites.

Table 4-9 shows the major satellite programs that were included in each category, the number of flights and failure reports.

TABLE 4 - 9 CLASSIFICATION OF PROGRAMS BY MISSION TYPE

MISSION CLASSIFICATION	PROGRAMS	NO. FLIGHTS	NO. FAILURES
NAVIGATION	TRANSIT GPS	22	241
OBSERVATION	DMSP ITOS / NOAA / TIROS N LANDSAT METEOSAT NIMBUS SEASAT SMS TIROS VELA	79	912
SCIENTIFIC	ANNA ARIEL ATS BIOSAT DYNAMICS EXPLORER ESAA EXPLORER GEOS GOES HCMM (AEM 1) HEAO HERMES INJUN ISS IUE LES MAGSAT MARINER LUNAR ORBITER OAO OGO PEGASUS PIONEER RANGER SAGE (AEM 2) SOLAR MAX SURVEYOR TDRS USAF SPACE TEST PROGRAM VANGUARD VIKING VOYAGER	120	703

TABLE 4-9 (continued) CLASSIFICATION OF PROGRAMS BY MISSION TYPE

MISSION CLASSIFICATION	PROGRAMS	NO. FLIGHTS	NO. FAILURES
COMMUNICATION	DSCS (II AND III) FLTSATCOM IDCSP INTELSAT (II, III, IV, and V) MARECS NATO (II and III) SKYNET (I and II) TELSTAR SYNCOM INSAT MARISAT SATCOM	100	490

Three major categories of subsystems were established for this analysis:

COMMON ELECTRONIC & ELECTROMECHANICAL	Telemetry Guidance Power Data Management
COMMON MECHANICAL	Thermal Structural Propulsion
PAYLOAD	Visual/IR sensors Navigation Payload Communication Payload Special Payloads

Figure 4-33 depicts the results of the failure rate determinations by mission type and subsystem type. Navigation satellites show the highest failure rate, results which reflect primarily the GPS constellation (the only other navigational satellite program was the relatively simple TRANSIT). Earth observation satellites had the next highest failure rate, a reflection of the complexity of the instrumentation, telemetry, and guidance systems on many of these missions. The lowest failure rates were in the communication satellites. This reliability can be attributed to their previously noted technological maturity. This rank ordering of failure rates is consistent

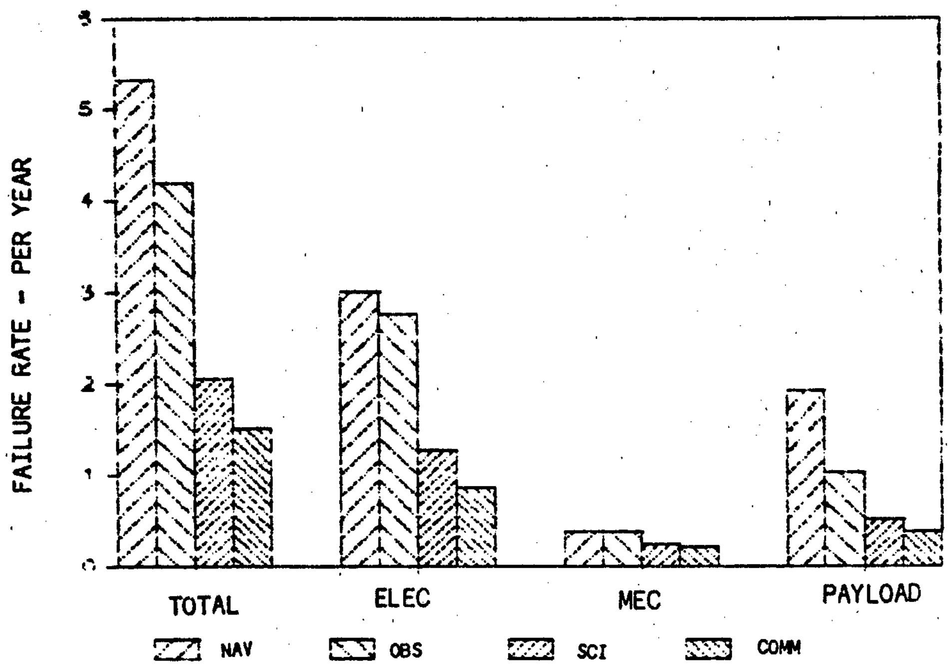


FIGURE 4 - 33 FAILURE RATES BY MAJOR SUBSYSTEM AND MISSION CATEGORIES

across the subsystem groupings defined above.

Table 4-10 and Figures 4-34 and 4-35 show the two Weibull parameters for missions and subsystems. Navigation and communication satellites have beta values of greater than 0.5 (i.e., failure rates are not decreasing significantly through the mission life) on both an overall basis and for most subsystems. With the exception of scientific/experimental missions, the beta values for mechanical subsystems are also greater than 0.5. These results are discussed for the individual mission types in the following subsections.

TABLE 4 -10 WEIBULL PARAMETERS FOR MISSIONS AND SUBSYSTEM GROUPINGS

	ALPHA (a)		BETA	
	Est	Std. Err	Est	Std. Err
NAVIGATION				
Total Mission	0.160	0.159	0.916	0.069
Electronic/Elmech.	0.261	0.267	0.876	0.089
Mechanical	6.166	4.317	1.894	0.106
Payload	0.495	0.443	1.155	0.114
OBSERVATION				
Total Mission	0.087	0.061	0.389	0.044
Electronic/Elmech.	0.153	0.099	0.470	0.061
Mechanical	1.875	0.996	0.738	0.100
Payload	0.217	0.196	0.243	0.055
SCIENTIFIC				
Total Mission	0.080	0.044	0.175	0.084
Electronic/Elmec.	0.150	0.073	0.220	0.091
Mechanical	1.426	0.752	0.255	0.120
Payload	0.611	0.238	0.378	0.125
COMMUNICATION				
Total Mission	0.397	0.279	0.668	0.095
Electronic/Elmech.	0.419	0.346	0.463	0.043
Mechanical	4.674	3.806	0.926	0.372
Payload	3.024	1.690	1.117	0.125

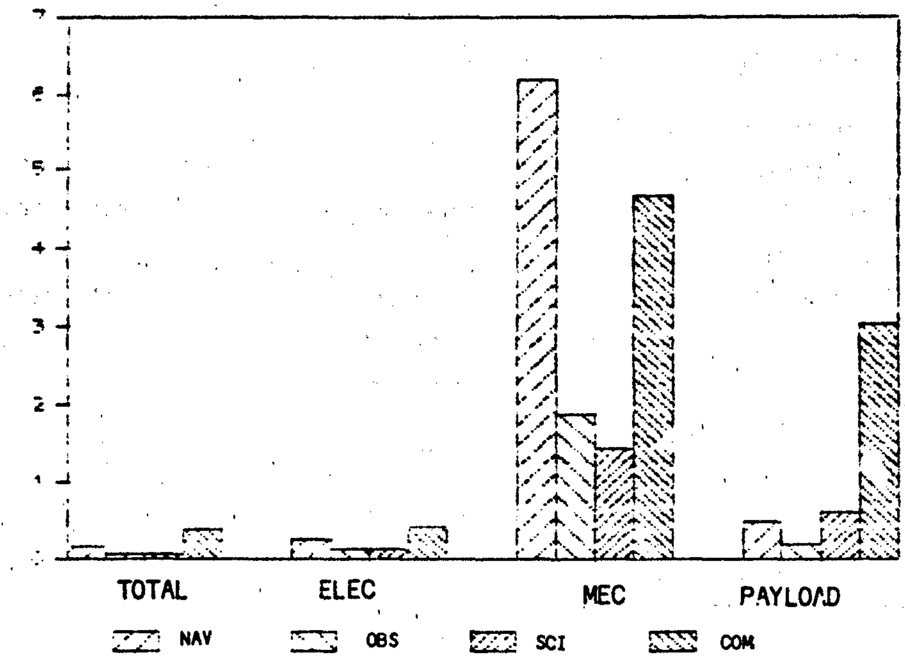


FIGURE 4 - 34 WEIBULL ALPHA BY MAJOR SUBSYSTEM AND MISSION CATEGORIES

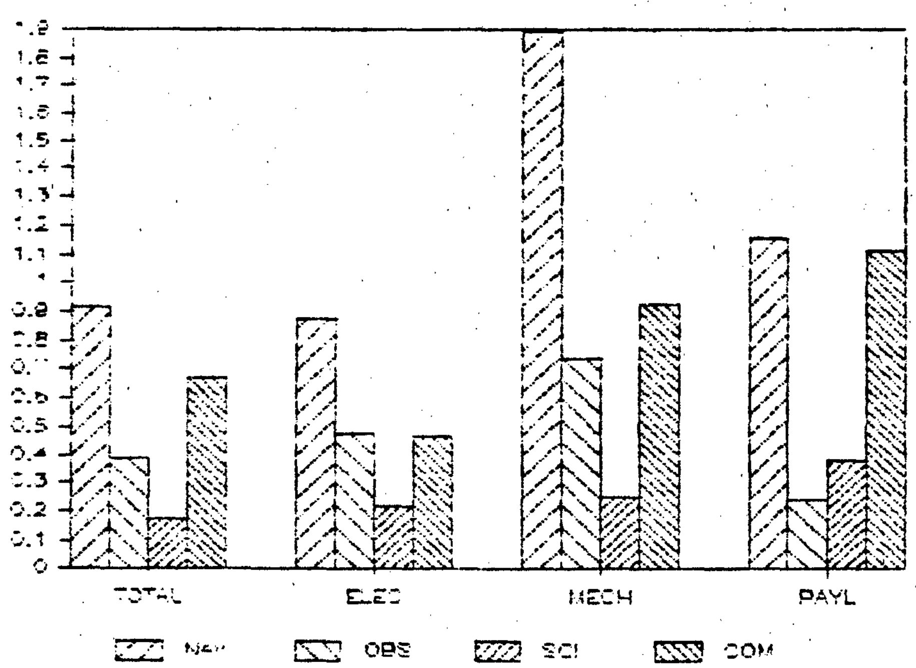


FIGURE 4 - 35 WEIBULL BETA BY MAJOR SUBSYSTEM AND MISSION CATEGORIES

4.4.1 Navigation Missions

Figure 4-36 shows the distribution of failure reports by subsystem in navigation satellites. The largest single contributor is the navigational payload subsystem. Other important factors are the telemetry and data management subsystems. Table 4-11 shows the results of failure rate calculations for the entire mission and major subsystem groupings.

TABLE 4 - 11 FAILURE RATES FOR NAVIGATION MISSIONS

	TOTAL MISSION	COMMON SUBSYSTEMS Electronic/ Electromech.	COMMON SUBSYSTEMS Mechanical	PAYLOAD SUBSYSTEMS
Years in Orbit	45			
No. of Failures	241	136	18	87
Failure Rate				
Point Estimate	5.33	3.01	0.40	1.92
Lower Limit*	4.89	2.68	0.28	1.66
Upper Limit*	5.77	3.34	0.52	2.19

*90% Confidence Interval

Figure 4-37 shows the behavior of failure ratios (failures per mission-year) over time. The overall mission and major subsystem groupings do not exhibit reliability growth, a fact confirmed by the Weibull curve fitting whose results are shown in Table 4-10.

4.4.2 Earth Observation Missions

The distribution of failures by subsystems in earth observation missions is shown in Figure 4-38. Common electronic/electromechanical subsystems are the most significant source of failures; telemetry accounts for more than 30% of the total. Mission payloads (visual/IR sensors and special payloads) make up about one-quarter of the failure reports. The high percentage of telemetry

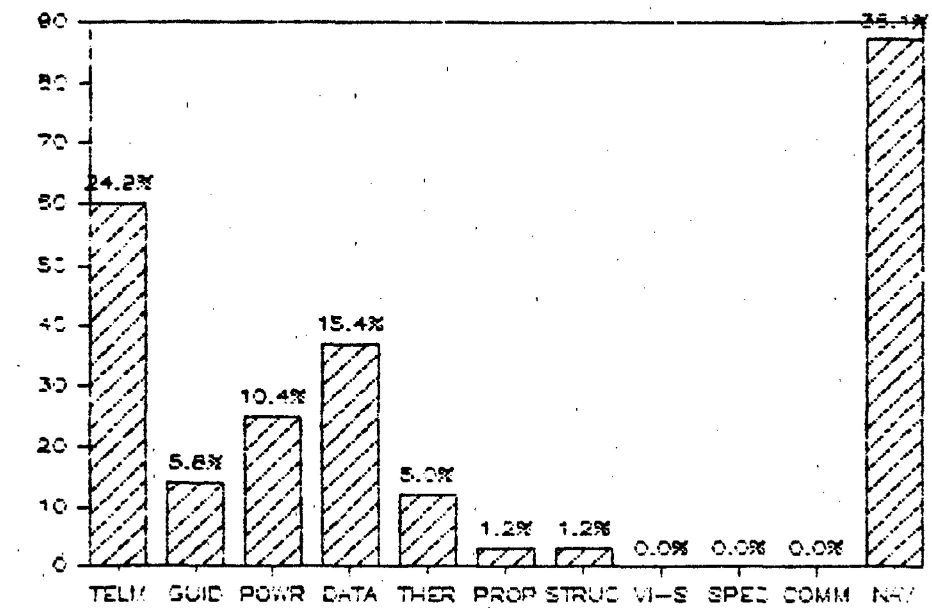


FIGURE 4 - 36 FAILURES BY SUBSYSTEMS IN NAVIGATION MISSIONS

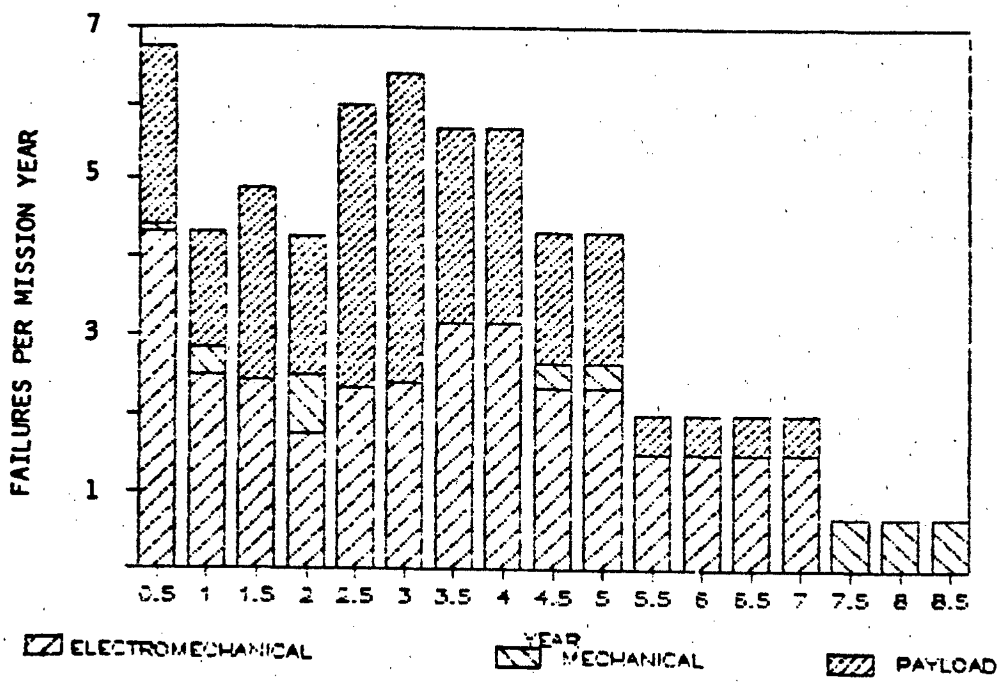


FIGURE 4 - 37 TIME TREND OF FAILURES FOR NAVIGATION MISSIONS

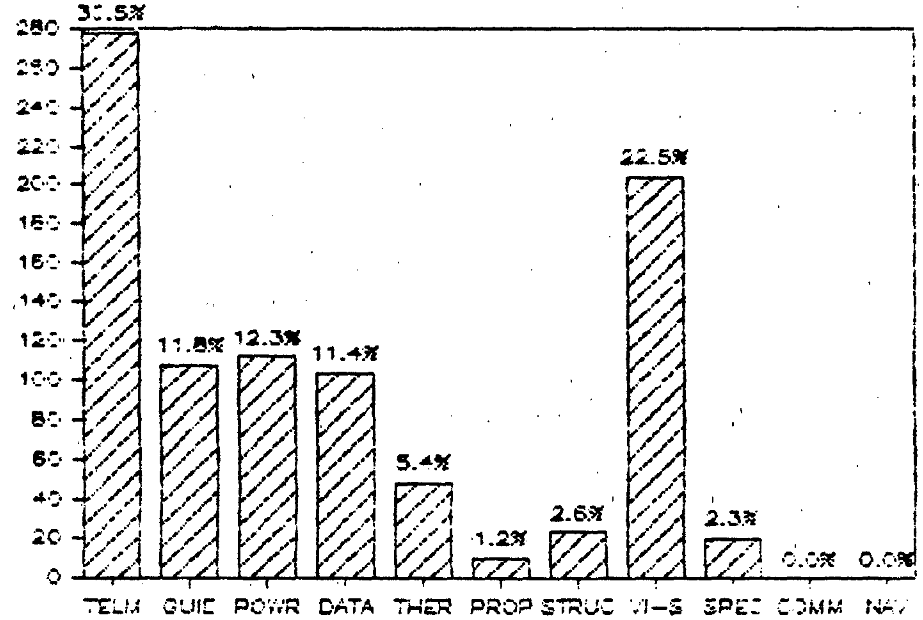


FIGURE 4 - 38 FAILURES BY SUBSYSTEMS IN OBSERVATION MISSIONS

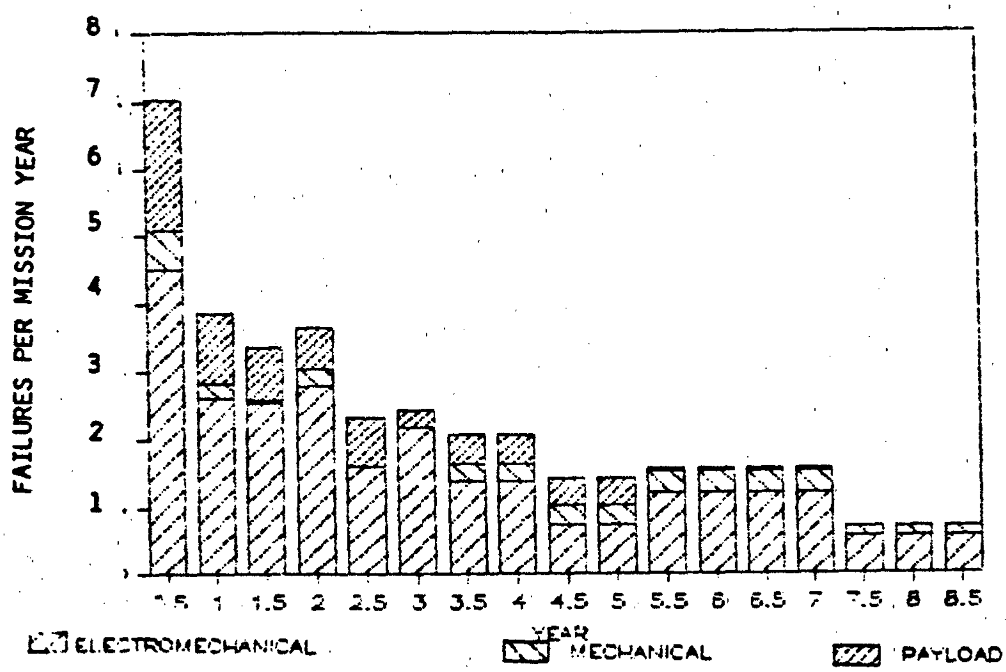


FIGURE 4 - 39 TIME TREND OF FAILURES FOR OBSERVATION MISSIONS

and data management failures is related to the number of missions which are in low orbits and therefore include magnetic tape recorders. Table 4-12 shows the results of failure rate calculations for earth observation missions.

TABLE 4 - 12 FAILURE RATES FOR EARTH OBSERVATION MISSIONS

	TOTAL MISSION	COMMON SUBSYSTEMS		PAYLOAD SUBSYSTEMS
		Electronic/ Electromech.	Mechanical	
Years in Orbit	218			
No. of Failures	912	602	84	226
Failure Rate				
Point Estimate	4.18	2.76	0.38	1.04
Lower Limit*	4.00	2.62	0.33	0.95
Upper Limit*	4.36	2.90	0.44	1.12

*90% Confidence Interval

4.4.3 Scientific and Experimental Missions

As shown in Figure 4-41, scientific and experimental satellites exhibit a strongly decreasing failure ratio. The primary explanation is the importance of electronic and electromechanical subsystems in both the mission and the payload. The nature of scientific and experimental satellite missions is such that design and environment related failures are also much more significant than in other mission classifications. The decreasing failure ratio is consistent with this explanation. Table 4-10 shows Weibull parameters for this mission class.

4.4.4 Communication Satellites

The distribution of failures by subsystems for communication satellites is shown in Figure 4-42. The common electronic/electromechanical subsystem grouping accounts for the largest fraction of all failures, but the mission

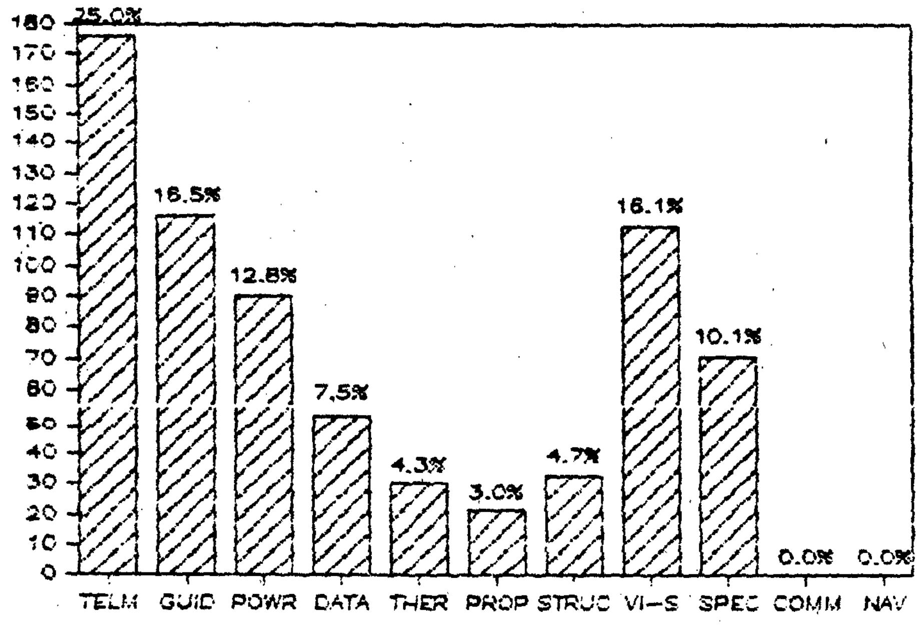


FIGURE 4 - 40 FAILURES BY SUBSYSTEMS IN SCIENTIFIC MISSIONS

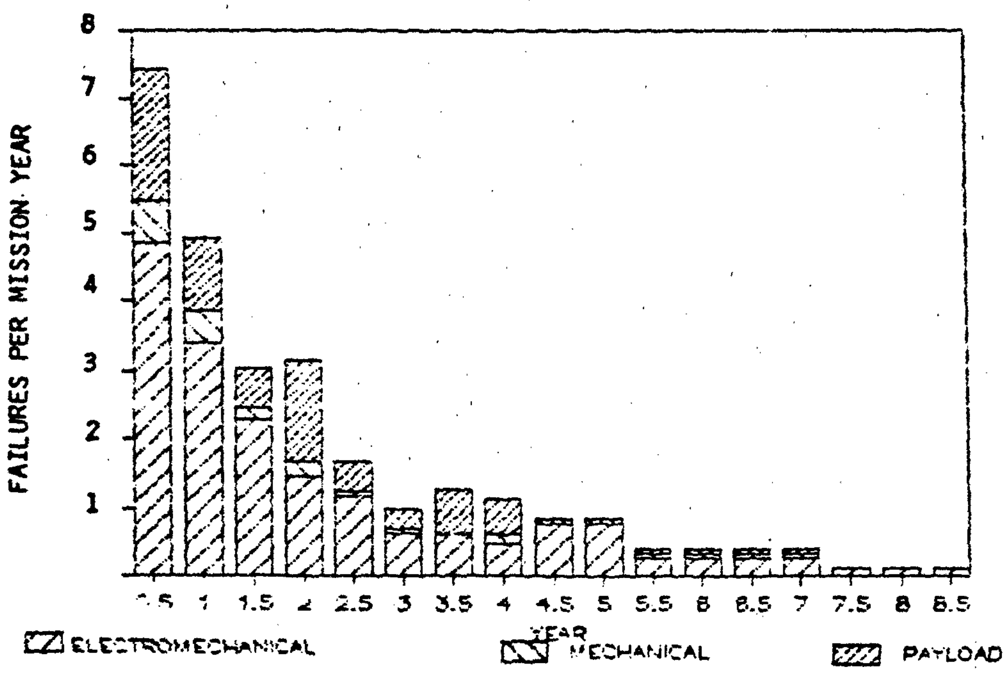


FIGURE 4 - 41 TIME TREND OF FAILURES FOR SCIENTIFIC MISSIONS

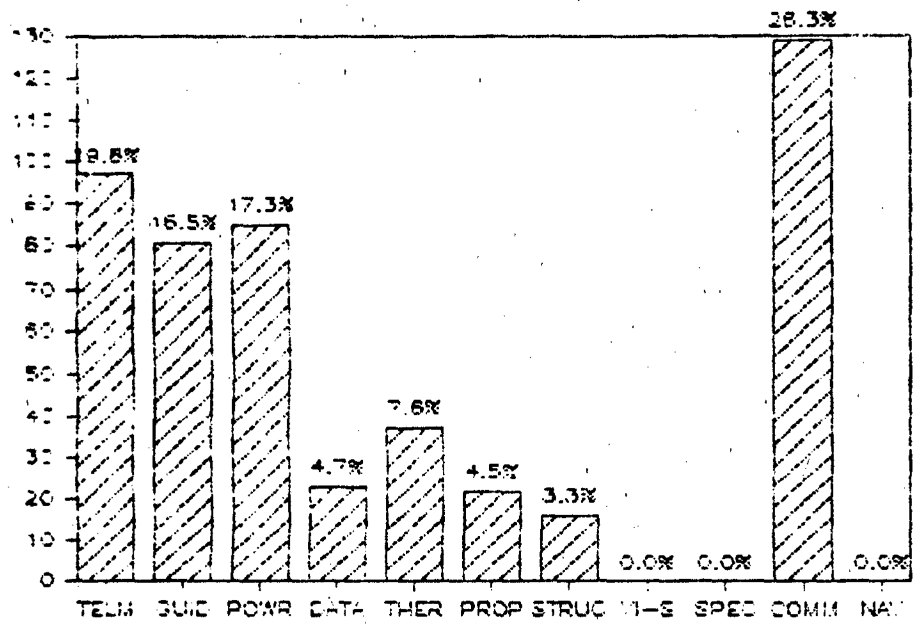


FIGURE 4 - 42 FAILURES BY SUBSYSTEMS IN COMMUNICATION MISSIONS

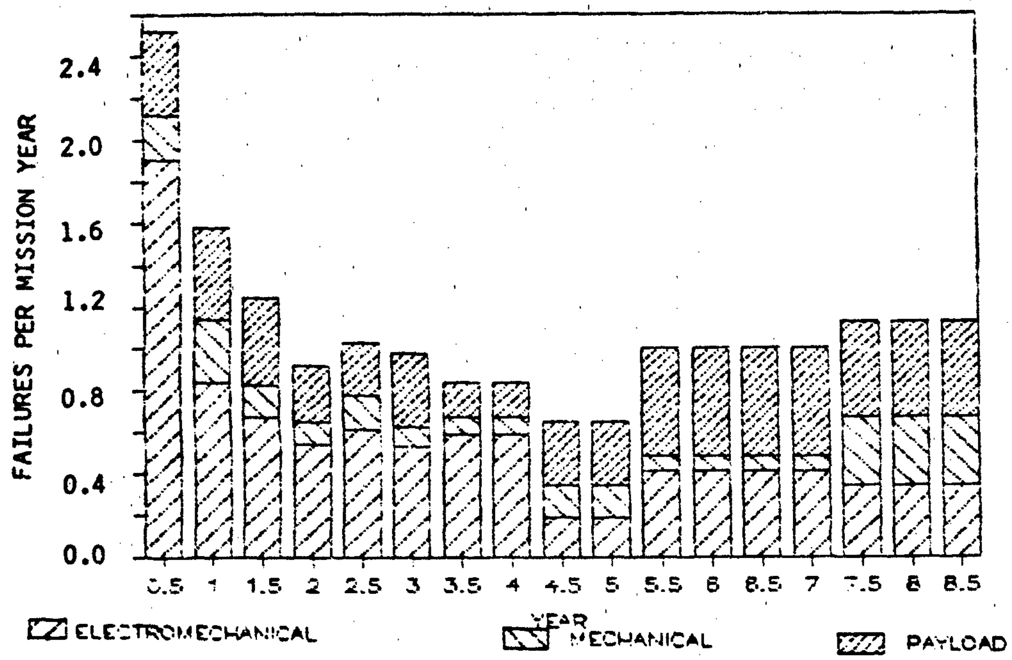


FIGURE 4 - 43 TIME TREND OF FAILURES FOR COMMUNICATION MISSIONS

payload is the single most important contributor. That the telemetry subsystem is less important is evidence of both the well understood nature of communication satellite telemetry and control and the fact that most of these vehicles are in geostationary orbits. Table 4-13 shows the result of failure rate and MTBF calculations.

TABLE 4 - 13 FAILURE RATES FOR COMMUNICATION MISSIONS

	TOTAL MISSION	COMMON SUBSYSTEMS		PAYLOAD SUBSYSTEMS
		Electronic/ Electromech.	Mechanical	
Years in Orbit	324			
No. of Failures	490	286	75	129
Failure Rate				
Point Estimate	0.66	1.13	4.32	2.51
Lower Limit*	0.62	1.05	3.76	2.25
Upper Limit*	0.70	1.22	5.06	2.83

*90% Confidence Interval

As shown in Figure 4-43, communication satellites do not have a strongly decreasing failure ratio overall, but the failure ratio of electronic and electromechanical subsystems do show the usual reliability growth. The non-decreasing failure ratios in the mission payload can be attributed to (a) the presence of non-electronic components (e.g., mechanical despin assemblies, pointing mechanisms, and antennas) and (b) wearout in travelling wave tubes. Table 4-10 shows the results of Weibull parameter estimations for these missions.

4.5 Orbital Effects

The following orbit classifications were used for this analysis:

LOW	Perigee of less than 200 Km
MEDIUM	Perigee between 200 and 2000 Km
HIGH	Geostationary (perigee of 35,000 Km) and extraterrestrial missions.

Because so many factors are related to orbit (e.g., mission type, satellite complexity, etc.), analyzing all mission failures with respect orbit could lead to erroneous results. For example, any differences between geosynchronous and extraterrestrial satellites are more likely to be due to mission differences (primarily communications versus scientific) than to effects of the trajectory.

Therefore, the analysis of orbit effects was restricted to the telemetry subsystem which had a considerable commonality between missions. Table 4-14 summarizes the result of the analysis. When all failure reports were included, there were statistically significant differences in the failure rates of the low orbit satellites on one hand, and medium and high orbit satellites (between which there was little difference) on the other. Further investigation of these trends revealed that magnetic tape recorder (MTR) related incidents accounted for approximately one-third of the low orbit failures, one quarter of the medium orbit failures, and less than 5% of the high orbit failures. The need for MTRs is a consequence of the satellite orbit but is not a directly related physical effect. Thus, the analyses were repeated for low and medium orbit missions with the MTR-related failure reports censored. Table 4-15 shows that the failure rates without the MTRs are practically the same for low and medium orbit satellites, and that the failure rate for high orbits is only slightly higher. From this limited investigation it is concluded that orbit parameters do not have a significant effect on the failure rate of telemetry subsystems.

TABLE 4 - 14 FAILURE RATES FOR TELEMETRY SUBSYSTEM BY ORBIT

ORBIT	WITH MTR (per year) Point Estimate	Upper Limit*	Lower Limit*	WITHOUT MTR (per year) Point Estimate	Upper Limit*	Lower Limit*
LOW	1.00	1.11	0.91	0.54	0.62	0.47
MEDIUM	0.76	0.84	0.68	0.54	0.60	0.47
HIGH	0.67	0.72	0.62	approx. same as with MTR**		

*90% confidence interval

**only 11 out of 269 total failure reports were related to MTRs

Chapter 5

USE OF MIL-HDBK-217 FOR SPACECRAFT RELIABILITY PREDICTION

From the spacecraft reliability experience and its analysis contained in the preceding chapters a reliability prediction methodology consistent with MIL-HDBK-217D is formulated. The primary procedure combines the conventional parts-based model with a Weibull term to account for the decreasing hazard phenomenon that has been described in the preceding chapters. Alternate procedures are provided for special situations.

5.1 SUITABILITY OF MIL-HDBK-217 METHODOLOGY

In reliability prediction for electronic equipment it is usually taken for granted that system failures are due to failures at the part level, and that the latter are a random phenomenon governed by the exponential failure law. These assumptions are also the basis for both of the prediction procedures in MIL-HDBK-217. As discussed in sections 2.2 and 2.3 of this report, only about one-half of the failures observed on spacecraft conform to this classical failure pattern (those classified as due to parts, quality and unknown causes). The other one-half, primarily due to design and environmental causes, exhibits a hazard that shows a pronounced decrease with time on orbit.

A challenging part of the work reported on here was to develop a reliability prediction methodology that was consistent with the experience of the space programs and yet was compatible with the overall approach of MIL-HDBK-217. Adherence to MIL-HDBK-217 procedures is important because of

- the familiarity of Government and industry personnel with that methodology

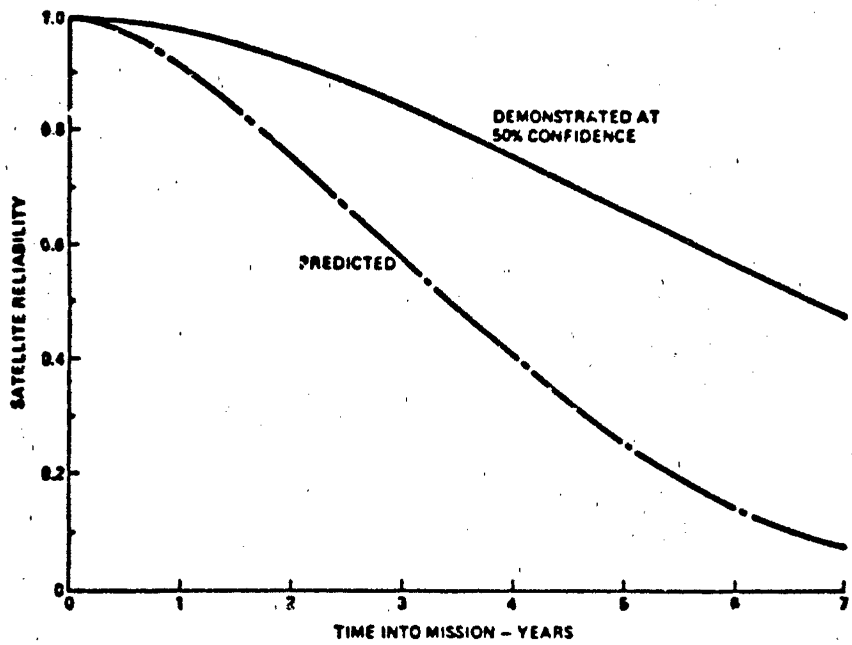
- the considerable investment in computerized procedures based on MIL-HDBK-217

- the need to utilize the parts reliability experience that is being accumulated in applications other than on spacecraft

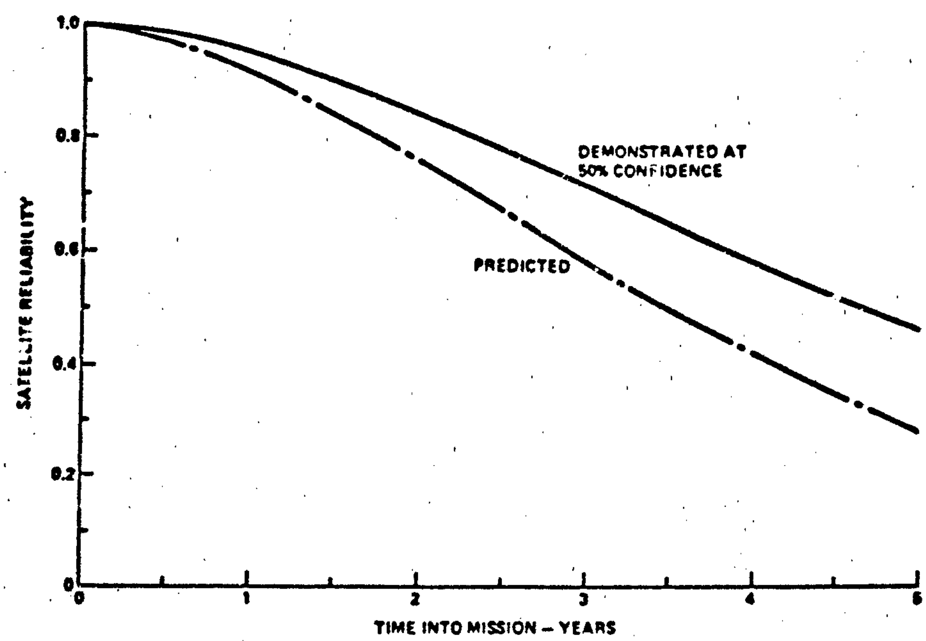
The latter point is due to the comparatively small number of electronic parts failures that are observed on spacecraft and the even smaller number which can definitely be associated with a specific part type. There were fewer than 30 permanent failures in memory devices for the entire spacecraft population surveyed here, and none of these devices could be made available for a post-failure physical analysis. In contrast, the RAC MDR series of publications reports on failures of several thousand memory devices each year, most of which occur in known and controlled environments and at least some of which are subjected to a detailed post-failure analysis. An unpublished study by a major supplier of spacecraft reported a total of 9.62 failures in microcircuits in 892×10^6 part hours during twelve years prior to 1983. The fractional number of failures is due to allocation among part types where the failure was attributable to one of several parts. For all other part types the number of attributed failures is even less. In most cases the observed orbital failure rates are within a factor of 5 of those predicted by MIL-HDBK-217D procedures. Since the base failure rates for some part types changed by approximately the same ratio between the C and D versions of MIL-HDBK-217 (published in 1979 and 1982, respectively), the parts failure predictions generated for space applications appear to fall within broadly acceptable limits.

Nothing in the data studied as part of this effort indicates that the electronic parts failure process in space differs from that in other applications once the proper environmental model is known. Thus, improved reliability prediction for spacecraft seems to depend much more on the development of accurate thermal and radiation models than on the modification of the parts failure rates in MIL-HDBK-217 for a given environment.

An important consideration in applying MIL-HDBK-217 procedures and data to space applications is how the past predictions compare with the achieved reliability. Accurate data for this purpose are very difficult to obtain because in most cases a complete reliability prediction is made only very early in the development phase (in many cases in connection with a proposal) and the launched configuration differs markedly from that which was analyzed. Some spacecraft contractors maintain updated files of the predicted reliability but these are usually considered proprietary data. One major systems company made data without attribution available to this study which permit a comparison of predicted (by existing MIL-HDBK-217D procedures) vs. achieved ("demonstrated at 50% confidence") reliability at the spacecraft level. Failure of the major mission function was equated to total spacecraft failure in this analysis. A graphical representation of the data for two programs, each involving multiple satellites, is shown in Figure 5-1. Because of the extensive redundancy provisions neither the predicted nor the demonstrated reliability follow the exponential relation. At an earlier time (ca. 1975) a comparison of observed vs. predicted reliability had been made for a number of programs with time on orbit as the independent variable. A summary plot from that study is shown in Figure 5-2.



A. PROGRAM A



B. PROGRAM B

FIGURE 5-1. COMPARISON OF PREDICTED AND DEMONSTRATED RELIABILITY

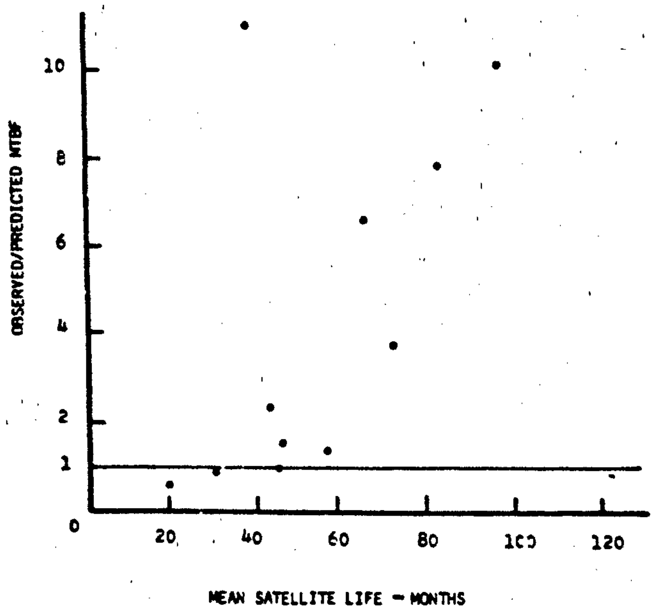


FIGURE 5-2. RATIO OF OBSERVED/PREDICTED MTBF VS. MISSION DURATION

The experience on these programs is in agreement with other programs for which summary data were obtained as part of the investigation leading to this report but were not made available for publication. From the composite of Figures 5-1 and 5-2 and other comparisons that were examined during the course of this project it is concluded that present reliability predictions for spacecraft overestimate the failure rate by at least a factor of two, and that the excess of predicted over observed failures increases with time on orbit. The reliability prediction procedures proposed below provide corrections for both of these difficulties.

5.2 Proposed Prediction Procedure

The primary procedure which is discussed here requires knowledge of the spacecraft mission and of failure rates at a non-redundant level (typically parts or subassembly). In the two alternate procedures, which are described in a later section, one or the other of these requirements is waived.

5.2.1 Derivation

The key element of the primary procedure is to regard the on-orbit reliability as a product of two factors, of which the first represents the conventional parts failures and the second the failures due to mission effects which include design, environmental and other causes. Thus, the reliability of a non-redundant part or subassembly is obtained from

$$R = R_{parts} * R_{mission} \qquad 5\text{-}1$$

where $R_{parts} = \exp(-Mt)$ and $R_{mission} = \exp(-t^b/a)$. Reliability predictions for higher levels of spacecraft systems, which typically include redundancy, can be generated for a fixed mission time from the above reliability prediction by conventional methods, described by MIL-STD-756B, Method 1001.

The data presented in Figure 2-6 indicate that the two factors (parts/quality/unknown and design/environment) make an equal contribution to the total spacecraft hazard, and Figure 5-2 suggests that the cross-over between the exponential and Weibull components occurs between 20 and 30 months on orbit (2 years has been used in the following). To obtain an overall reliability prediction that results at the spacecraft level in one-half of the failure probability obtained by current methods, the first step is to determine the parameter of the exponential distribution, here M. It is assumed that the failure probability of electronic and electromechanical systems at the spacecraft level is dominated by redundant functions[1]. Thus, if the new prediction for M is to result in one-half the failure probability obtained by using the existing methodology with parameter L

----

1. Very few of the essential spacecraft and mission functions do not employ redundancy (this excludes experimental equipment, sensors, etc.). For small equipment segments, such as the memory within a computer, a higher level of redundancy may be employed but these segments do not make a significant contribution to the total spacecraft reliability. See also the discussion of Figure 5-5.

$$[1 - \exp(-Lt)]^2 = 2\,[1 - \exp(-2Mt)]^2 \qquad 5\text{-}2$$

The factor of 2 in the exponent on the right side of the equation is necessary because the exponential part, represented by M, constitutes nominally one-half of the total failure probability[2] Thus

$$1 - \exp(-Lt) = 1.41\,[1 - \exp(-2Mt)] \qquad 5\text{-}3$$

Using only the first two terms for the series expansion of the exponential, $e^{-x} = 1 - x$, one obtains the approximation

$$M \simeq L/2.82 \qquad 5\text{-}4$$

Next, the parameters of the Weibull term, $R_{mission}$, can be determined. Where the specific mission type is not known, a generic assignment of b = 0.12 may be made (see Table 2-3, this assignment is based on combining design and environmental causes). Where information about the mission type is available, more accurate assignments of b can be made from the following table. The values for the b parameter shown here differ from those in Table 4-10 because the latter were computed for all causes, whereas those in Table 5-1 were computed only for failures due to design and environment causes.

TABLE 5 - 1 PREDICTION FACTORS BY MISSION TYPE

Mission Type	b	Ma*
General	0.12	0.54
Communication	0.4	0.66
Navigation	0.9	0.93
Observation	0.13	0.55
Scientific	0.09	0.53

* See below for the use and calculation of this factor

----

2. Equation 5-2 makes use of the approximation $R \simeq 1 - Lt$ which is valid only for $Lt \ll 1$. This is justified because the prediction procedure is applied at the parts or subassembly level at which the Lt product is indeed much less than one.

Once b is known, a is computed to make the failure probability of the Weibull term equal to that of the exponential term at t = 2 years. For the generic case of b = 0.12

$$\exp(-2M) = \exp(-2^{0.12}/a) = \exp(-1.09/a)$$

Hence,

$$Ma = 1.09/2 = 0.54$$

where M is related to the hazard computed by the current MIL-HDBK-217D methodology as indicated in equation 5-4. For the general spacecraft category, the statement of the proposed method of reliability prediction is therefore

$$R = \exp(-Mt) * \exp(-t^{0.12}) = \exp\{-M(t+t^{0.12}/0.54)\}$$

5.2.2 Procedure

The following four step procedure implements the proposed modification of the MIL-HDBK-217 reliability prediction for spacecraft. The Military Handbook for Reliability Prediction of Electronic Equipment, MIL-HDBK-217, describes two methods for reliability prediction:

- Part Stress Analysis which accounts for the detailed thermal, electrical and operational stresses to which each part is subjected

- Parts Count Method which is based on an average stress exposure of the parts

In both approaches the application environment (such as space flight) is an element of the reliability prediction. The stress analysis utilizes an explicit environment factor π_E while the parts count provides a distinct grouping of failure rates for each application environment, thus including an implicit allowance for the environment factor. These differences affect only the first step of the following procedure.

1. Obtain adjusted exponential hazard

   - if the parts stress method is used, divide S_F by 2.82 (ref. equation 5-4)

   - if the parts count method is used, divide the individual failure rates by 2.82

2. Enter Table 5-1 to select b and Ma

3. Compute a from a = Ma/M

4. The complete prediction can then be computed from

   $R = \exp(-Mt) * \exp(-t^b/a)$ Ref. eq. 5-1

5.2.3 Validation

To validate this prediction methodology, it will be applied to the two spacecraft programs for which predicted and demonstrated (achieved) reliability had been depicted in Figure 5-1. Figure 5-3 is a repeat of that figure with the addition of point predictions based on the proposed methodology. It is not precisely known what satellite types are represented in these figures but reasonable guesses lead to predictions that match the demonstrated reliabilities rather closely. The following procedures were used to generate these predictions:

1. the original prediction, which is based on the exact redundancy structure and component count used in the spacecraft is approximated by a prediction for a hypothetical spacecraft consisting of five major subsystems, each of equal complexity and each being redundant. The reliability of this hypothetical spacecraft is given by

$$R = 1 - (1 - e^{-Lt})^5$$

2. L was selected to give a good fit to the exact prediction curve; this was achieved at L = 0.14 for both parts of the figure (this indicates that the spacecraft were of nearly equal complexity). The computed points are identified as open circles in the figure. Note that the calculations based on dual redundancy provide an extremely good match to the prediction based on the actual redundancy. This validates the statement made in connection with the derivation of equation 5-2 that the spacecraft reliability function is dominated by dual redundant elements.

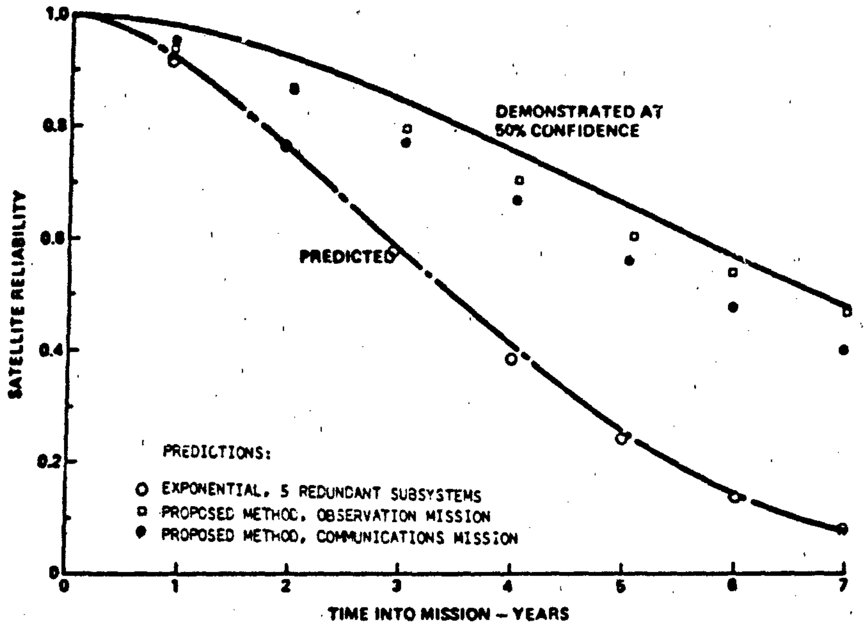
3. Beta was selected from Table 5-1 and a was then computed as discussed above. The satellites shown in part A of Figure 5-3 were evaluated as communication and observation satellites, with the latter giving a much better fit. The satellites shown in part B of the figure were evaluated as communication and navigation satellites, with the latter giving a better fit.

The methodology proposed here requires only minor modifications of the existing MIL-HDBK-217 data and procedures. Its chief advantage is that it removes the systematic underestimation of reliability for long mission durations which is inherent in the exponential assumption. A further advantage is that it distinguishes between the principal types of space missions and thus permits more appropriate estimates to be generated for each.

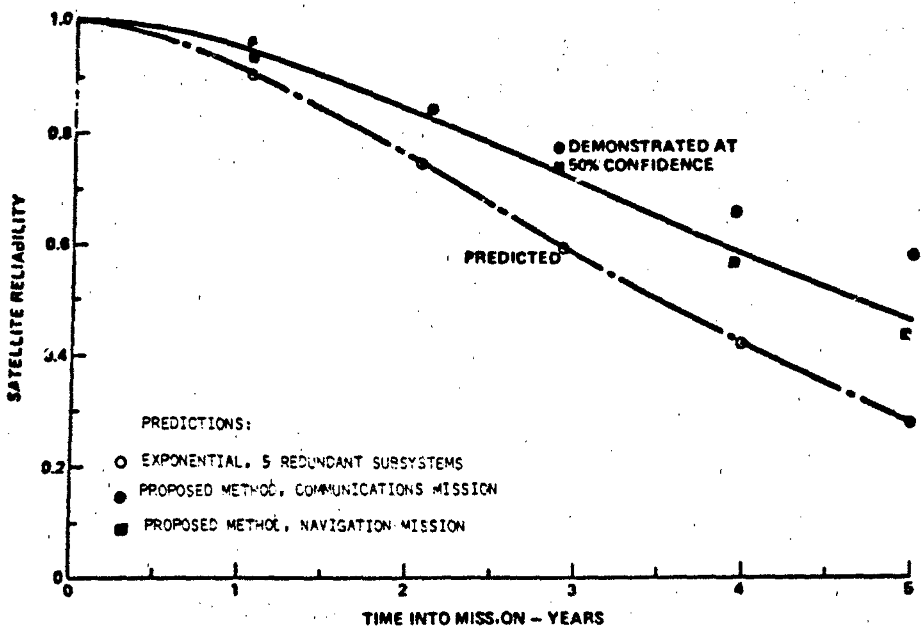
5.3 Alternate Procedures

The preferred reliability prediction procedure described above may be inconvenient or difficult to perform in two environments

- for the general electronic equipment manufacturer



A. PROGRAM A



B. PROGRAM B

FIGURE 5-3 PREDICTION BY PROPOSED METHOD

- in the early stages of space mission planning

Examples of general electronic equipment are telemetry components, power supplies, and audio frequency amplifiers. Although units destined for spacecraft applications may represent specialized designs and will receive special care in assembly and test, the manufacturer's reliability organization may find it very difficult to implement a completely separate reliability prediction procedure for products which represent only a small fraction of their total output. In many cases they will not be aware of the exact satellite application category in which the units will be used. In this environment the approximate exponential model described below will be preferred.

In the early stages of mission planning the exact equipment complement and redundancy provisions are usually not known. Therefore, the single string reliability prediction is not useful, and in addition, the R_{parts} term of equation 5-1 will be difficult to obtain. In this environment it is customary to base reliability prediction on the achieved reliability of similar spacecraft or major subsystems, and extrapolating for longer mission durations where that is necessary. The single-term Weibull model described below is suitable for these purposes.

Because the alternative procedures will in general yield less accurate predictions of space systems reliability and provide less insight into the effects of mission or component changes, the primary procedure should be used wherever possible.

5.3.1 Exponential Approximations

Piecewise Exponential Prediction

The basis for the piecewise exponential approach is that the conventional reliability prediction methodology is generally regarded as workable for spacecraft operations for the first year but that a reduced hazard is applicable to subsequent years. Because reliability prediction is based on the hazard-time product (Lt or Mt in the notation used here) it is simpler to work with a modified time rather than to modify the hazard for each component type. Thus, an approximate reliability prediction can be generated by reducing the 'chargeable' mission time. A good approximation to the primary prediction is obtained by using 40% of the mission time after the first year. The reliability equation then becomes

$$R = e^{-M(1 + .4(t-1))} \quad \text{for } t \geq 1$$

As seen in Figure 5-4 this approximation yields a fairly good fit to the primary prediction for the parameters used (M = 0.25, t in years). Because of the small differences at the single string level good agreement can be expected at major system and spacecraft levels.

A convenient implementation of the piecewise linear model is by means of a conversion table as shown below. The table is entered with the actual mission time for which the prediction is to be generated. The chargeable mission time is then obtained and is used in generating the prediction. The hazards (lambdas) or hazard-time products can be added, subtotaled, etc. in the same manner as hazards and hazard-time products of the conventional reliability prediction procedures. Likewise, the existing relations for redundancy remain applicable (with chargeable time substituted for actual time).

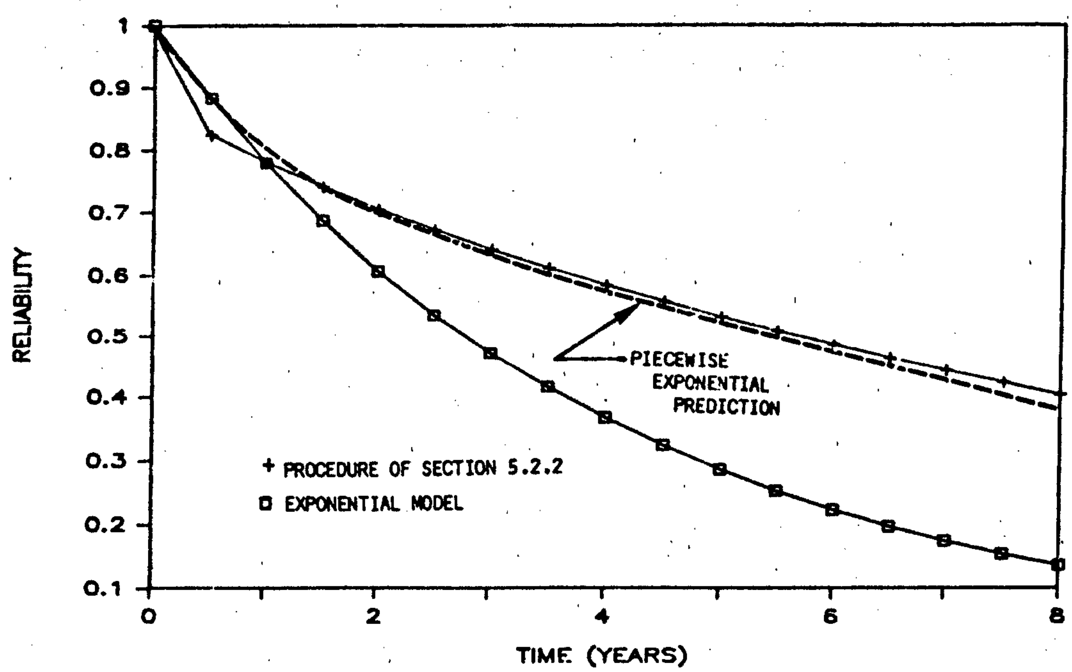


FIGURE 5-4 PIECEWISE EXPONENTIAL PREDICTION FOR SINGLE STRING

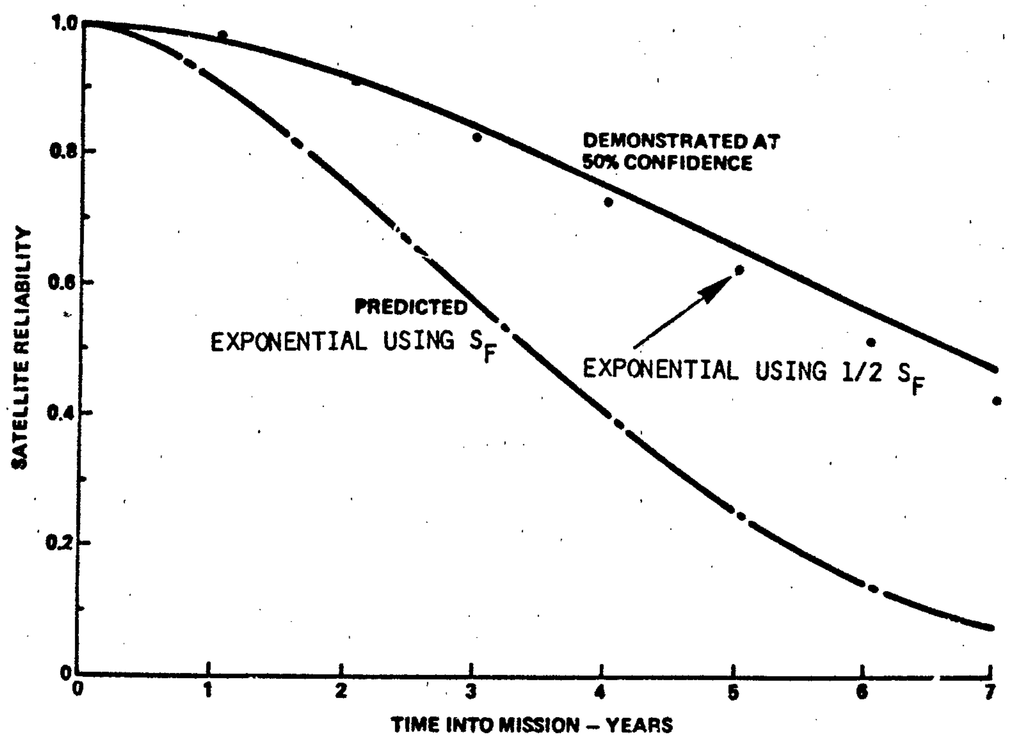


FIGURE 5-5 SIMPLE EXPONENTIAL APPROXIMATION AND DEMONSTRATED RELIABILITY

TABLE 5 - 2 TIME CONVERSION FOR PIECEWISE EXPONENTIAL PREDICTION

Actual Time (Yrs)	Chargeable Time (Yrs)	Actual Time (Yrs)	Chargeable Time (Yrs)
0.5	0.5	4.0	2.2
1.0	1.0	4.5	2.4
1.5	1.2	5.0	2.6
2.0	1.4	5.5	2.8
2.5	1.6	6.0	3.0
3.0	1.8	6.5	3.2
3.5	2.0	7.0	3.4

Simple Exponential Approximation

An even simpler procedure, suitable for missions of up to 5 years, is to reduce S_F as listed in MIL-HDBK-217D to one-half of the stated value. This permits use of all existing procedures without even the time conversion of Table 5-2. The comparison of using one-half of the given S_F with both the original exponential prediction and with the demonstrated reliability for Program A is shown in Figure 5-5. The 1/2 S_F correction is applied at the single string level and propagated to the spacecraft level by using the assumption of five redundant segments discussed in Section 5.2.3. This approximation provides very good agreement with the observed reliability until the fifth year. As in any pure exponential assumption, the incremental failure probability for long mission durations is substantially overestimated, and this presents a problem in the use of this method for mission planning.

5.3.2 Single Term Weibull Prediction

There are times when it is necessary to generate or to modify reliability predictions for spacecraft segments which incorporate redundant components but where the exact structure of the redundancy provisions is not known. A single term Weibull reliability model of the form

$$R = \exp(-t^b/a)$$

can be used in this connection. The beta parameter is selected at 0.75 which has been empirically determined to give a workable fit to the demonstrated reliability of redundant spacecraft functions or entire spacecraft. The a parameter is selected to fit a known reliability at a specified time, or to agree with a prediction arrived at for short orbit times by the exponential assumptions.

An example of the former approach is shown in Figure 5-6. The solid curves represent the reliability of a redundant spacecraft function by the exponential model and by the methodology described in Section 5.2. The broken line is the single term Weibull approximation with T = 0.08 which was selected to agree with the proposed prediction at 7 years. As an example of the use of this procedure consider a guidance system for which the reliability on an existing satellite has been demonstrated to be 0.85 for two years on orbit. What will be the reliability of this same system for seven years on orbit on a similar type of satellite?

From the basic equation for the single term Weibull model,

$$0.85 = \exp(-2^{0.75}/a) = \exp(-1.68/a)$$

$$\ln 0.85 = -\ 0.163 = -\ 1.68/a$$

$$a = 10.3$$

Now, making use of the basic equation for the seven year prediction:

$$R = \exp(-7^{0.75}/10.3) = 0.42$$

Using the exponential assumption for a redundant configuration will yield a seven year reliability of 0.33.

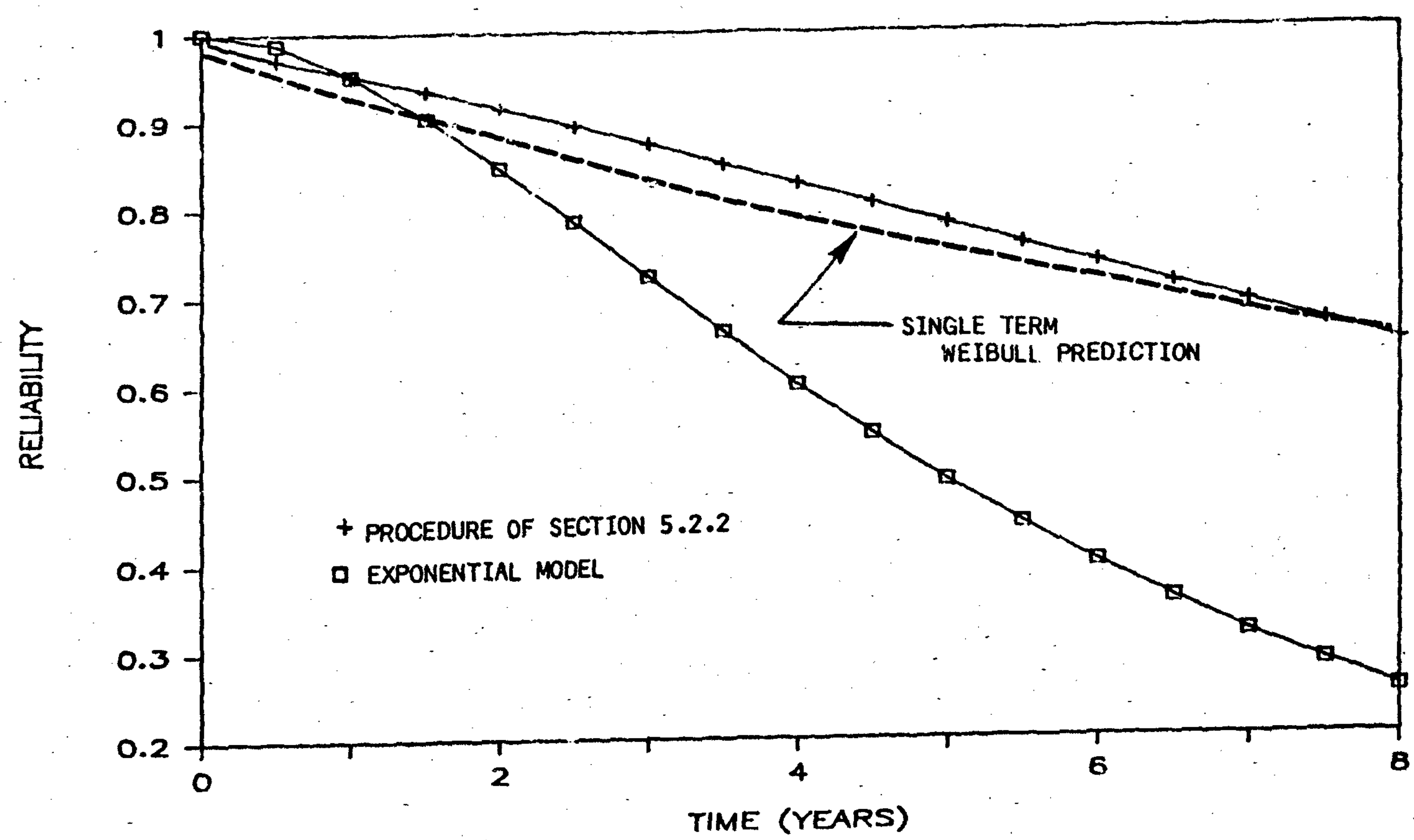


FIGURE 5-6 SINGLE TERM WEIBULL PREDICTION FOR A REDUNDANT SYSTEM

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Appendix A

METHODOLOGY

This appendix discusses the calculation of reliability parameters from failure report data. The first Section discusses the formulation of the Failure Ratio, the key parameter that is used for the description of spacecraft reliability experience. Section A.2 describes the generation of point and confidence interval estimates for the Weibull model.

A.1 Failure Ratio

The key reliability parameter utilized in the body of this report is the failure ratio, defined as the number of failures reported during a period divided by the number of operational satellites for which failure reports were obtained at the beginning of this period. All periods are referenced to the launch date. The standard period is six months, but this was modified in some instances as discussed below. Thus, period 2 extends from 7 months after launch to 12 months after launch.

The failure ratio is an approximation of the hazard which is defined as the failure density function divided by the survivor function

$$h(t) = f(t)/\{1 - F(t)\}$$

The number of surviving satellites was computed as the number launched minus the number that had become non-operational. Since all failure reports evaluated as part of this effort included an identification (sometimes coded) of the satellite from which they were obtained there was no problem in determining the total number of satellites launched. Satellites lost as result of a launch vehicle malfunction were censored from this study. The end of operational life of a satellite was recorded for approximately two-thirds of the population in one or more of the chronologies listed under Data Sources - Mission and Satellite Data in Appendix B. For the other one-third of the population it was quite difficult to determine when a satellite was no longer operational. The following criteria were adopted for declaring a satellite non-operational when specific reports were no available

- if the last reported failure was mission critical (criticality 1), the satellite was declared non-operational as of the date of that failure

- if the last reported failure was not mission critical, the satellite was assumed to have survived for the average time to next failure of satellites which incurred a non-critical failure during the period of its last reported failure.

This procedure gives creditable results except at the longest orbital life times for which special procedures were adopted as outlined below.

As the on-orbit time increased, the number of operational missions decreased. For example, there were 297 missions initially; after 9 years, there were only 17. Thus, the sample size decreased by a factor of 15 with a resultant wide variations in failure ratios which resulted strictly from the normalization procedure and had no physical reality. Figure A-1 shows an example of such fluctuations. These spurious results were undesirable because they (1) increased uncertainty in parameter estimation and (2) made visual assessments of the data more difficult. Therefore, after 5 operational years, sampling intervals were increased first to 1 year (i.e., 2 periods) and then, after 7 operational years, to 2 years. Figure A-2 shows the results of such smoothing.

A second undesired effect of normalization was an apparent depression of the initial failure ratio and an increase of failure ratios in late periods. The apparent initial depression was due to the large number of infant mortalities which made the failure ratio at the beginning a poor estimator of the average number of operational spacecraft. In the late operational periods, the fluctuations due to smaller sample sizes noted previously may cause an apparent increase in the failure ratios. Increasing the interval sizes is not possible because there are too few surviving flights. In order to resolve both problems, failures occurring before the first month and after the 102nd month were censored.

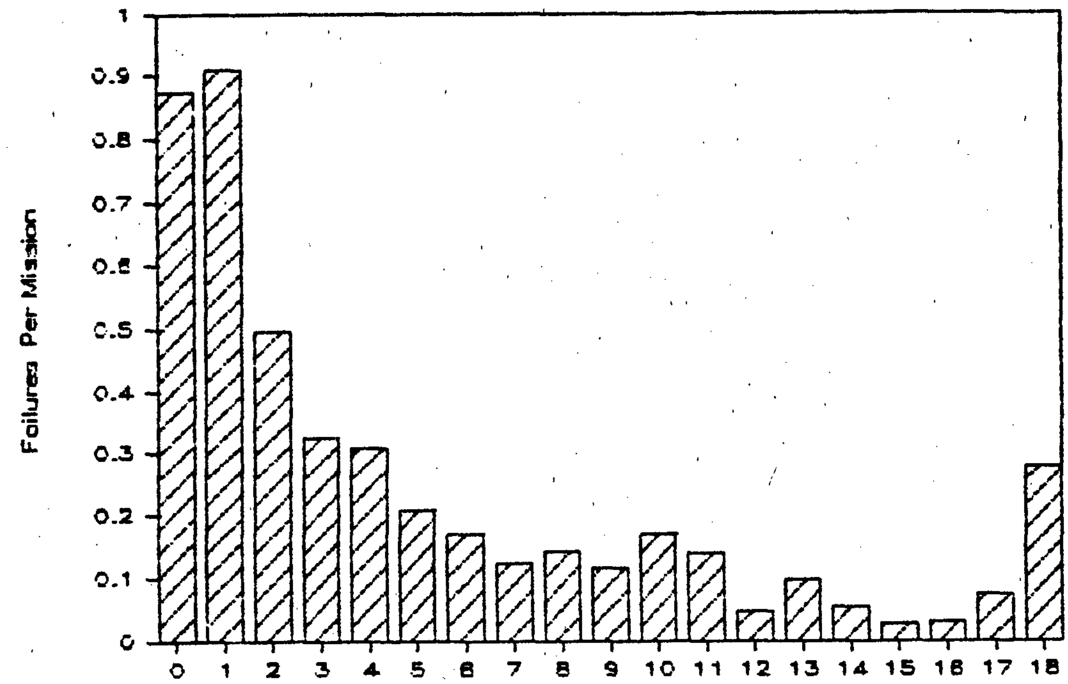

FIGURE A-1 UNSMOOTHED FAILURE RATIOS

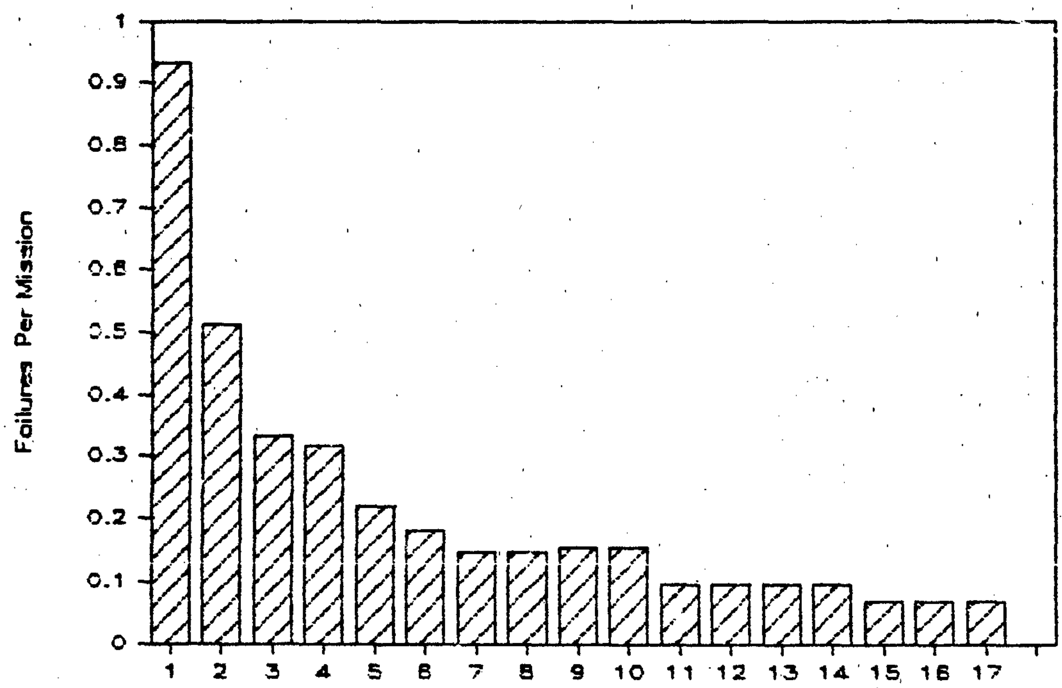

FIGURE A-2. SMOOTHED FAILURE RATIOS

A.2 WEIBULL MODEL

The Weibull distribution parameters are estimated without any assumptions on the time distribution of failures. One parameter, known as the shape factor, determines whether the failure rate decreases, remains constant, or increases over time. The second determines the frequency of failures. The Weibull distribution can be expressed as[1]

$$F(t) = 1 - \exp[-t^{b}/a] \qquad \text{(A-1)}$$

where F(t) is the cumulative failure probability, alpha is the scale parameter, and beta is the shape parameter. The hazard function, h(t) of the Weibull distribution is [LLOY77]

$$h(t) = \frac{b\ t^{\,b-1}}{a} \qquad \text{(A-2)}$$

The logarithm of equation A-2 results in the following expression:

$$\ln h(t) = \ln b - \ln a - (b - 1) \ln t \qquad \text{(A-3)}$$

In other words, the logarithm of the hazard function is linear with the logarithm of operational time. The hazard function is simply the number of failures per unit time, i.e., the normalized failure ratio defined in section A.1. Thus, a regression of ln h(t) against the ln t will yield a slope and

----

1. the parameters of the Weibull distribution are listed as a and b in the equations but are sometimes referred to as alpha and beta in the text

intercept which can be used to calculate the Weibull shape and scale factors as follows:

$$b = \text{slope} + 1 \qquad \text{(A-4)}$$

$$\ln a = \ln b - \text{intercept} \qquad \text{(A-5)}$$

Standard errors used for confidence intervals can be calculated by using the propagation of errors method. If equation A-4 is differentiated, the result becomes

$$d\, b = d\, (\text{slope})$$

squaring both sides and substituting the squares of the variances for these differentials as described in [BEVI72]:

$$\text{var}^2(\, b) = \text{var}^2\, (\text{slope}) \qquad \text{(A-6)}$$

The standard error can be substituted for the variance in this expression, and then it is seen that the standard error of beta is the same as the standard error of the slope. A similar procedure can be used to determine the standard error of alpha, the scale parameter, based on the slope and intercept. Differentiating and squaring equation A-5 results in

$$\frac{(da)^2}{a^2} = \frac{(db)^2}{b^2} - 2\, \frac{db}{b}\, d\, (\text{intrcpt}) + d(\text{intrcpt})^2$$

or

$$\text{std err}\, (a) = a^2\, [\, \text{var}^2(b) - 2\, \text{cov}\, (b, \text{intrcpt}) + \text{var}^2(\text{intrcpt})]^2 \qquad \text{(A-7)}$$

REFERENCES

BEVI72 P. R. Bevington, "Data Reduction and Error Analysis", McGraw-Hill, New York, 1972

LLOY77 D. K. Lloyd and M. Lipow, "Reliability: Management, Methods, and Mathematics", Second Edition, by the authors, 1977

Appendix B

DATA SOURCES AND UTILIZATION

This appendix describes the origins, nature and format of the data used in this report. Section B.1 discusses data sources and section B.2 describes the organization of the SoHaR space data base.

B.1 DATA SOURCES

This section discusses sources of both failure data and mission descriptions.

Failure Reports

Failure reports were obtained from two sources: ODAP and OOSR. ODAP -- Orbital Data Acquisition Program generated by The Aerospace Corporation. A sample of an ODAP report is shown in Figure B-1.

OOSR -- On-Orbit Spacecraft Reliability, PRC Rreport R-1863, 30 September 1978, and Analysis of Spacecraft On-Orbit Anomalies and Lifetimes, PRC Report R3579, 10 February 1983. A sample of an OOSR report is shown in Figure B-2. The PRC reports also contain a number of statistical summaries of these data.

```
INCIDENT    378          PROGRAM -DMSP          FLIGHT NO. -5B1-29          LAUNCH DATE-73 AUG
 SUBSYSTEM            -TELEMETRY TRACKING AND COMMAND
 ASSEMBLY             -TRANSMITTER
 SUBASSEMBLY OR TYPE-OUTPUT STAGE
 MODULE OR TYPE       -
 CAUSE              -TRANSISTOR,PROCESSOR CONTROLS
 FAILURE TIME - 43     DUTY CYCLE -100          CLASS      -ELECTRICAL
 REPORT NO.   :29-13      FEEDBACK-GAP          CRITICALITY-MISSION          ORITAL PHASE-STEADY ST
          ***  SYMPTOM - THREE SECONDS AFTER THE VERY HIGH PESOLUTION DATA PLAYBACK     ***
          ***    COMMAND FUNCTIONED, THE RF SIGNAL DROPPED.  THE P/L DATA TRANSMITTER   ***
          ***    OUTPUT POWER EXHIBITED SPIKING THEN DROPPED TO 0.13WATTS, FAILURE OF   ***
          ***    FINAL DATA TRANSMITTER OUTPUT STAGE                                    ***
          ***  CAUSE - GOLD AND ALUMINUM MIGRATION IN THE HIGH POWER OUTPUT             ***
          ***    TRANSISTOR                                                             ***
          ***  RECOVERY METHOD - NONE, MISSION TERMINATED                               ***
          ***  CORRECTIVE ACTION - CHANGED SCREENING METHODS, NEW TRANSISTOR AND        ***
          ***    REDUNDANT TRANSMITTERS                                                 ***
          ***  COMMENT - FAILED TEMPORARILY ON 12/75, STARTING 9/76 SITES HAVING        ***
          ***    FADEOUTS, 10/76 TERMINATED COMPUTER USAGE OF HIGH RESOLUTION AND       ***
          ***    HIGH RESOLUTION INFRARED DATA                                          ***
          ***  COMMENT - SR12-76                                                        ***
```

FIGURE B-1. EXAMPLE OF ODAP REPORT

Index	Anomaly Time (hours)	Anomalies: Description	Anomalies: Cause	Anomalies: Mission Effect	Corrective Action (if known)	Remarks
...	8970	High data rate tape recorder will not play back.	Bearing failure in the planetary gear.	Significant loss of data.		
118	8976	Startracker inner gimbal sticking in position.	Unknown	Not too serious.	Later units checked thoroughly.	
119	8980	One of three subcarrier oscillators hung up at lower band edge after completing a tape recorder playback.	The end of tape switch did not activate the beacon data relay as required.	None, corrected by ground command.		

FIGURE B-2. EXAMPLE OF OOSR REPORT

Mission and Sa ite Data

The following publications were used for descriptive data on programs and individual flights:

L.J. Abella and M.B. Hollinger, "U.S. Navy in Space: Past, Present, and Future", IEEE Trans. on Aerospace and Electronic Systems, Vol AES-20, No. 4, July, 1984, p. 325

Launch dates, orbital parameters, launch vehicles, and disposition of Navy related satellites including TRANSIT, FLTSATCOM, and Space Test missions.

F.W. Buehl and R.E. Hammerand, A Review of Communication Satellites and Related Spacecraft for Factors Influencing Mission Success, Vol. II, Aerospace Corp. Report No. TOR-0076(6792)-II, November, 1975

Volume II of this report contains detailed descriptions of many pre-1977 space programs. Included were parts counts, satellite weights and power, subsystem descriptions, and program histories.

E.S. Epstein, et. al., "NOAA Satellite Programs", IEEE Trans. on Aerospace and Electronic Systems, Vol AES-20, No. 4, July, 1984, p. 325

Mission histories payload descriptions, and current status of NOAA Satellites.

R.F. Gould and Y.O. Lum, eds., A Review of Satellite Systems Technology, IEEE Press, 1976.

Descriptions and mission histories of major communication satellite programs including INTELSAT, ANIK, and ATS.

Charles Hall, "The Pioneer 10/11 Programs: from 1969 to 1994", IEEE Trans. Reliability, Vol R-32, No. 5, December, 1983, p. 414

Mission history and payload description of Pioneer 10/11 and Pioneer Venus programs.

National Aeronautics and Space Administration, NASA Pocket Statistics, 1979

A brief description of all NASA missions including launch dates, vehicles, orbital parameters, and mission disposition.

TRW Corp., TRW Space Logs, 1972, 1978, and 1983 editions

TRW space logs provided both descriptive information on selected satellites and an extensive listing of launch dates, launch vehicles, and orbital parameters.

TRW Corp., A Compendium of TRW Spacecraft Reliability Data, Vol. I, 1974

This report contained detailed mission histories, failure data, and parts descriptions for 5 TRW programs including VELA, INTELSAT III, OAO, Interplanetary Pioneer, and Pioneer Jupiter.

B.2 ORGANIZATION OF THE DATA BASE

The spacecraft failure data base which supported the analyses of this report consisted of three types of files:

- Failure Reports: Data on the time, severity, cause, and affected subsystem and components of individual failures.

- Descriptive Data: Data on mission types, launch dates, duration, orbital parameters, space vehicle characteristics, launch vehicles, and information sources.

- Glossaries: Descriptions of codes used for causes, subsystems, parts, orbits, and launch vehicles.

All failure reports were contained in a single file consisting of 2613 records. Table B-1 describes the fields in the file. Descriptive data were contained in two files. The first contained data generic to all missions in a given program. Table B-2 describes this program file, which had a total of 92 records. The second descriptive file contained data on individual missions (spacecraft) as shown in Table B-3.

TABLE B-1. DESCRIPTION OF THE FAILURE REPORT FILE

FIELD NAME	DESCRIPTION
PROGRAM	Program identification number
FLIGHTNO	Flight number
SUBSYSTEM	Affected subsystem
CAUSE1	Primary (or most important) cause
CAUSE2	Secondary cause
CAUSE3	Tertiary cause
PART	Affected part or assembly
FAILTIME	Operational month in which failure occurred
CRITICAL	Criticality level (1-7)
INCIDENT	Incident number in ODAP or PRC data base for traceability

TABLE B-2. DESCRIPTION OF THE PROGRAM DESCRIPTION FILE

FIELD NAME	DESCRIPTION
PROGRAM	Program identification number
NAME	Program name
AGENCY	Project managing agency
PROGSTART	Year in which program was initiated
FIRSTLAUNCH	Launch year of first mission
LASTLAUNCH	Launch year of final mission
PROGTYP	Program type (i.e., navigational, earth observation, scientific/ experimental, or communication)
DESIGNR	Prime contractor
DESLIFE	Design life
PARTS	Total parts count
WEIGHT	In orbit weight (excluding expendables) in kg
POWER	Beginning of life power in watts
GUIDANCE	Stabilization technique (i.e., 3-axis, gravity, or spin) — if different techniques were used on some vehicles in the program, it was noted in the comments
TELM	Presence or absence of telemetry, tracking, and control computer
COMMENT	60-byte field for Comments

TABLE B-3. DESCRIPTION OF THE INDIVIDUAL MISSION FILE

FIELD NAME	DESCRIPTION
PROGRAM	Program identification number
FLIGHTNO	Flight number
INTDESG	International designation number
YEAR	Year of launch
MO	Month of launch
LIFE	Mission life time (if known) in months
LASTRPT	Time of last failure report in months
CRITFAIL	Time of last critical report
TERMTIME	Time of mission failure used for analysis (set equal to LIFE if known, otherwise the procedure described in appendix A is used)
COUNT	Number of failure reports in this mission
ENDCODE	Final mission disposition (i.e., operational, terminated in orbit, launch failure, landed, decayed)
ERRSOURC	Source of failure reports (i.e., ODAP, PRC or other)
ENDSOURC	Reference on termination time and disposition
LAUNVEH	Launch vehicle
PERIGEE	Perigee in Km
APOGEE	Apogee in Km
INCLINATION	Orbital inclination in degrees
COMMENT	60-byte field for comments