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VOLUME 1

SUMMARY

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TABLE OF CONTENTS

INTRODUCTION

This Final Report covers the work performed by the General Electric Company for the Jet Propulsion Laboratory under Contract No. 951112, Modification No. 3. The Report also covers Company-sponsored work considered to be pertinent or closely related to the objectives of the Contract study tasks. Abbreviated statements of the four study tasks follow:

- Perform a preliminary design study of a Voyager spacecraft, powered by radioisotope thermoelectric generators (RTG's). Examine the impact of the use of RTG's upon the spacecraft and mission design.
- Study the manner of application and practical extent of the use of redundancy in the Voyager missions at both the spacecraft system and mission levels.
- Determine the impact of the planetary quarantine constraint upon the design and fabrication of the Voyager spacecraft and upon the Voyager mission.
- Study the Voyager data management and project control requirements and describe approaches to meeting these requirements.

The Final Report is presented in five volumes, this Summary Volume, plus one volume for each of the above tasks. In addition, there are 11 Data Management appendixes. This Summary volume presents a very brief description of the activities, results, and conclusions of each study.

SECTION 1

RTG-POWERED SPACECRAFT DESIGN STUDY

1.1 SUMMARY

1.1.1 OBJECTIVES

Long-life earth orbiting and interplanetary spacecraft designs to date have employed arrays of silicon solar cells to provide spacecraft electrical power.

The continuing development of radioisotope thermoelectric generators (RTG's) has provided spacecraft system designers with a possible alternative power source, free from the limitations of spacecraft-sun distance and spacecraft-sun orientation.

The objectives of this study were to perform a preliminary design of an RTG-powered Voyager spacecraft bus and to:

a. Examine potential spacecraft system interface problems resulting from the integration of the RTG's.

b. Determine the potential effects of the RTG's upon the mission, identifying both additional constraints and flexibility.

c. Perform an overall comparison of the Voyager RTG-powered spacecraft with the solar-powered version.

The design study was constrained to the use of tested and proven technologies wherever possible. The General Electric Voyager Task B spacecraft design was the baseline system into which the RTG power system was to be integrated. State-of-the-art RTG performance characteristics were to be used. Specifically excluded from the scope of this study were the considerations of lander design.

1.1.2 RESULTS

The accomplishments during this 16-month study are summarized below:

1. A parametric study was conducted on the design characteristics of the RTG units. From the results of this study, an RTG design was selected.

2. A preliminary design of a Voyager spacecraft incorporating the selected RTG's was performed.

3. An extensive radiation effects study was performed to determine the effects of the RTG neutron and gamma radiation upon the piece parts and materials of the spacecraft bus and upon typical science instruments.

4. Thermal analyses were conducted to determine the effect of the 15 thermal kilowatts of the RTG's upon the Voyager planetary vehicle. The analyses covered the conditions of pre-launch, powered flight, parking orbit, and space flight operations.

5. The question of nuclear safety was examined. To ensure the containment of nuclear fuel in the event of near-earth mission abort and subsequent earth re-entry, two design approaches to the RTG units were investigated, one based upon separating the RTG units from the spacecraft and one based upon the RTG units remaining with the spacecraft during re-entry.

6. A solar-powered version of the spacecraft was designed with the objective of providing an "interchangeable" design in which changes necessary in converting from solar to RTG-powered versions were minimized.

7. A comparison study of the two types of spacecraft was conducted, considering factors such as weight, reliability, mission flexibility, and development problems.

1.1.3 CONCLUSIONS

Principal conclusions of the study are as follows:

a. Based upon existing technology, an RTG-powered spacecraft for Voyager missions is feasible. Additional development work is necessary to develop and prove out designs for fuel containment in the event of mission abort.

b. Again, based upon existing technology, the RTG-powered spacecraft is about 200 pounds (or 8 percent) heavier than the solar-powered version. Current improvements now under evaluation offer promise of significantly reducing this weight penalty.

c. The RTG-powered spacecraft is more reliable and offers greater mission flexibility.

d. Certain types of scientific instruments (primarily those intended to measure radiation) are adversely affected by the radiation from the RTG and would require special consideration in design and/or suitable shielding.

e. Voyager spacecraft design should anticipate the later incorporation of RTG power.

1.2 STUDY DESCRIPTION

1.2.1 STUDY ORGANIZATION

The study was organized into elements as follows:

a. Basic studies

    1. Influence of nuclear safety considerations

    2. RTG parametric design study

    3. Radiation sensitivity of spacecraft and science equipments

b. Spacecraft trade studies

    1. Generation of alternative spacecraft configurations

    2. Thermal, magnetic, field-of-view, and articulation studies

    3. Configuration selection

c. RTG spacecraft design definition

    1. Thermal and radiation mapping studies

    2. Structural, weight, and mass properties analysis

    3. Subsystem design

d. Design evaluation

    1. Safety

    2. Reliability

3. Mission flexibility

e. Comparison study: RTG versus solar spacecraft

The preliminary design of an RTG-powered spacecraft constituted the principal activity of this study and is described in Section 1.2.2. This is followed by a description of the spacecraft converted to solar power (Section 1.2.3). Sections 1.2.4 and 1.2.5 describe the principal results of RTG sizing and radiation sensitivity studies, respectively.

1.2.2 RTG SPACECRAFT DESCRIPTION

1.2.2.1 General Arrangement

A photograph of the spacecraft is shown in Figure 1-1. The insert shows the support arrangement of two spacecraft/lander combinations in the shroud. A feature of this design is the high truss support, chosen to satisfy the use of an over-the-nose shroud. The shroud separation joint is located at an intermediate point between the largest spacecraft diameter and the largest lander diameter. By reducing the axial distance between these diameters, i.e., by using the high truss design, low angular tipoff requirements are avoided.

In other configuration respects, the design is similar to the Task B result. The diameter of the equipment bay torus ring has been reduced to permit a larger projected area of the truss support slant panels. The increased area provides for potential growth in RTG requirements as well as array area for the solar-powered version discussed later. The bays have been reduced in number from 16 to 12 and increased in length to provide the same volume and radiation area as the Task B design.

Since RTG power eliminated the requirement for spacecraft-sun orientations, the 90-inch high-gain antenna is fixed to the spacecraft rather than articulated with a two-axis gimbal. Earth-pointing of the antenna is accomplished by biasing the spacecraft central axis from the Sun-pointing direction by means of wide-angle digital sun sensors which operate in null fashion against a stored signal program. The antenna mounting angle is such that about 10 degrees of bias occurs before encounter, zero degrees at encounter, and a maximum of 30 degrees

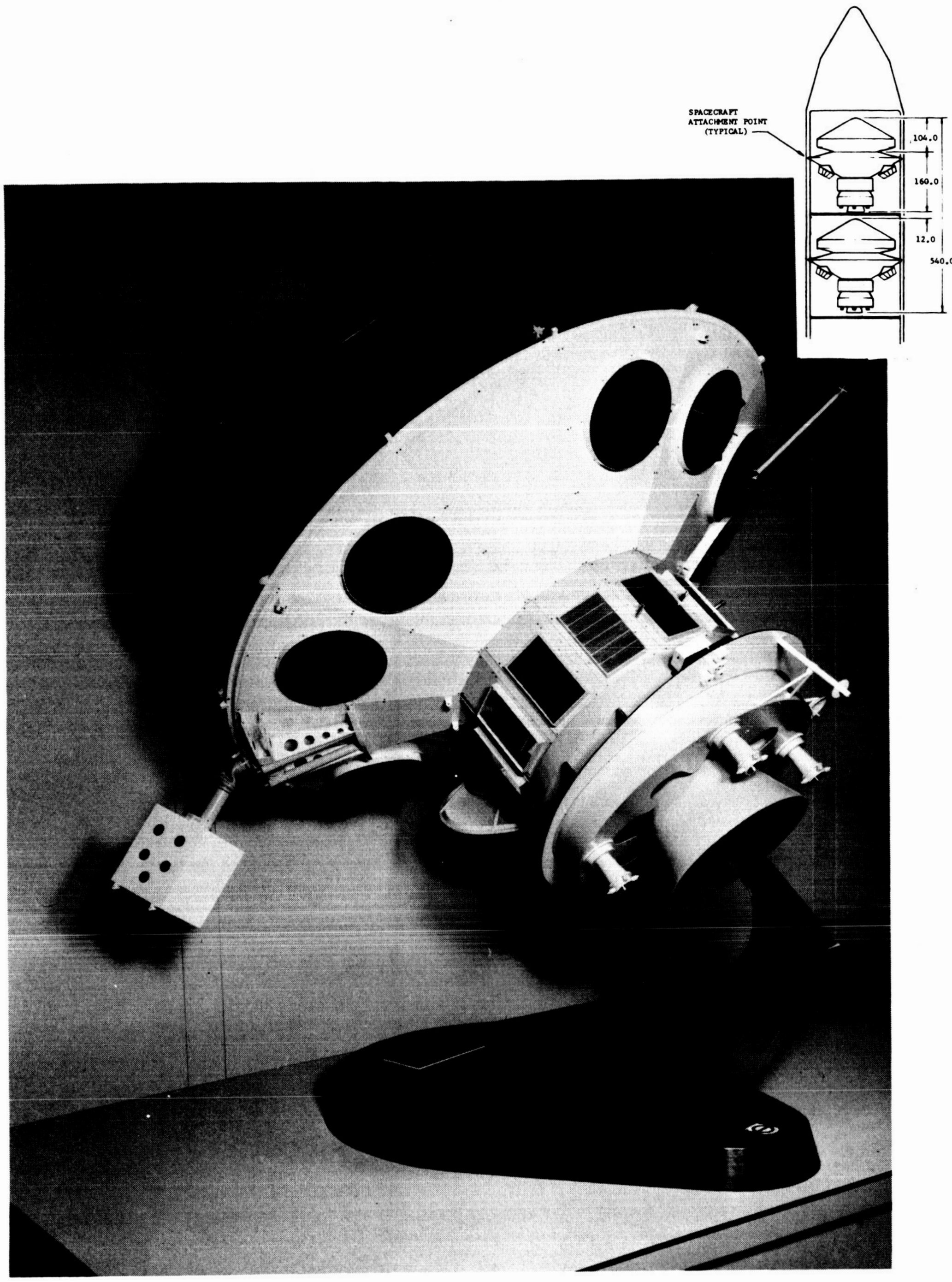


Figure 1-1. RTG-Powered Voyager Spacecraft

after encounter. A thermal barrier shelf is provided to prevent solar heating of the equipment bays during bias operation.

Because the high-gain antenna is fixed, no backup medium gain antenna is required. However, a medium-gain antenna is used to avoid excessive spacecraft bias during the midcourse phase.

1.2.2.2 RTG Integration

Eight RTG's are mounted on 8 of the 12 shear panels which form part of the truss support structure. This location was favored for several reasons. First, since they are above the more temperature-sensitive spacecraft equipment, namely the electronic bays and propulsion units, there is a reduced tendency to transport the RTG waste heat by convective means during prelaunch operation. Second, since they are located near the largest periphery of the spacecraft, more mounting space is available. A 50 percent growth is possible by using larger RTG's on the same mounting location. Third, with the RTG's canted as shown, the high location avoids concentrated heating of the Saturn V Instrument Unit which is located near the 20-foot-diameter base of the lower spacecraft shroud. Because the RTG's are canted, they reject their wast heat over a broad region of the internal shroud wall, resulting in more gradual temperature gradients. Since the eight RTG's are more or less distributed uniformly around the spacecraft, the temperature distribution on the shroud wall is somewhat axisymmetric, resulting in acceptable shroud thermal stress levels.

Each RTG is rated for an end-of-life power of 75 watts and uses a lead-telluride thermopile operating at a hot-junction temperature of 1050^{o}F and a cold-junction temperature of 500^{o}F. This thermopile is identical in all essential features to that developed in the SNAP-27 Program. As shown, the RTG's are mounted to the shear panels at the peripehry of their heat rejection fins, although a variety of other mounting methods is available. A headlight-shaped thermal barrier presents heat transport toward the inboard side of the mounting panels; heat rejection is predominantly in the outward axial direction.

The selection of eight RTG's was based on available mounting space and current RTG technology. The parametric study showed little variation in total weight within the range of 4 to

12 RTG's, so that the selection was not critical in terms of weight. The 75-watt size resulting from the selection of 8 RTG's is close to the capability of the SNAP-27 RTG, and therefore advantage can be taken of existing technology.

The RTG's are intended to be fueled prior to enclosure by the shroud. This is done in an explosive safe facility before transport of the encapsulated spacecraft/lander for assembly to the launch vehicle. The orientation of the RTG's on the spacecraft permits easy accomplishment of this operation. The fuel capsules are inserted along the RTG axis and locked in place.

After spacecraft encapsulation in the shroud, some form of internal and shroud wall cooling is required to remove the RTG waste heat. This must continue until liftoff. Cooling is required to limit the temperature rise of the electronic and propulsion equipment rather than the RTG's. Without cooling, the RTG temperature rise is less than 30°F and is within the design margin. The electronic equipment temperatures, however, could rise to levels in excess of 170°F for the case of no cooling on a hot day. Internal cooling by air alone is not adequate, due to the poor convective heat transfer coefficient of air at reasonable velocities. Analysis indicated that some other means of cooling the shroud walls is required. This can be accomplished by installing cooling coils in the shroud walls which serve as a radiation heat sink.

The time from liftoff to spacecraft separation may be as long as 90 minutes, during which no active cooling is available. With shroud heating by aerodynamic, solar, earth, and albedo fluxes, electronic temperatures during this phase will increase about 12°F. This result is based on an electronics temperature of 80°F at liftoff. In free space flight, the electronics operate at an average temperature of 60°F, with no temperature problems attributable to the RTG's.

1.2.3 SOLAR SPACECRAFT DESCRIPTION

A solar-powered spacecraft configuration was defined which minimized the changes necessary in converting from one type of spacecraft to the other.

The solar-powered version of the spacecraft is shown on Figure 1-2. Substitution of the RTG's and their mounting panels with solar panels results in a projected area in the central axis direction (solar direction) of 229 square feet. This is comparable to that of the Task B design. Principal changes in the spacecraft electronics occur in the (1) power conditioning equipment, because of differences in the source characteristics of RTG's and solar panels, and (2) the increased battery capacity in the solar-powered version. In either version, the power system electronics is contained in the same two bays.

The equipment in the other 10 bays remains relatively unaffected. In other general respects, the design is similar to the Task B result, including the use of a two-axis gimbaled high-gain antenna.

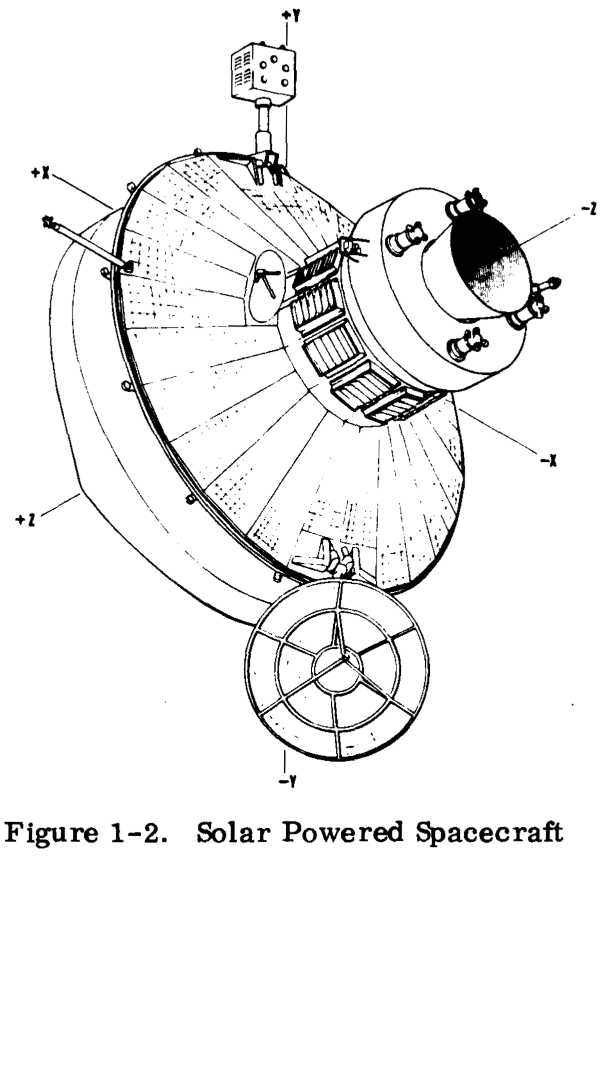


Figure 1-2. Solar Powered Spacecraft

1.2.4 RTG SIZING STUDY

This parametric study for the selection of an RTG design resulted in data on RTG weight and dimensions for the following independent variables: hot- and cold-junction temperatures, the use of Pb-Te and Si-Ge thermopiles, the use of Pu-238 isotope fuel, a power range from 50 to 200 electrical watts, and variations in heat rejection view factors. Typical results of the study are shown in Figure 1-3, which summarizes the nominal characteristics for a 75-watt RTG requirement. More comprehensive sizing results are presented in Volume 2, Section 5.1 of this Final Report. The sketches near the bottom of Figure 1-3 indicate the influence of heat rejection capability on the RTG configuration. The RTG on the left rejects heat omnidirectionally, whereas the RTG on the right rejects heat in a predominantly rearward direction. The RTG shown at the center represents an intermediate case. From the weight variations shown, it is evident that an optimum selection must be made in conjunction with achieving tolerable thermal conditions on the rest of the spacecraft. Integration of the RTG described earlier has taken this into account.

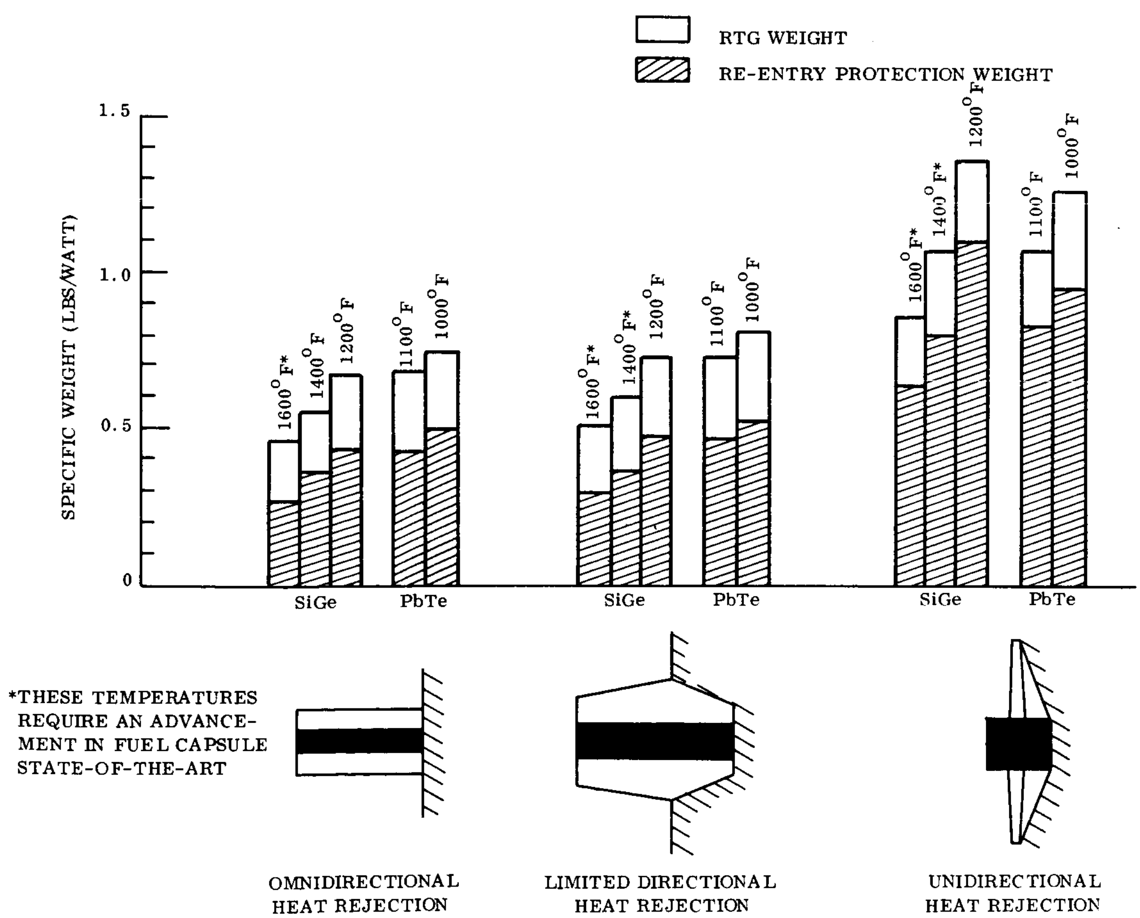


Figure 1-3. RTG Sizing Summary

Figure 1-3 also shows the influence of thermopile type and hot-junction temperature. There are practical limits on the RTG elements to be considered in interpreting the results. A maximum hot-junction temperature of $1100^{\circ}F$ was considered for lead-telluride thermopiles, since current data indicate that this temperature is consistent with the Voyager-Mars mission time. This does not apply to silicon-germanium thermopiles, which can operate at much higher temperatures. Although the data of Figure 1-3 indicates decreased weight for Si-Ge RTG's, this advantage is of a conjectural nature, since the higher-temperature fuel capsules required have not been developed. One major item favoring silicon-germanium thermopiles is that hermetic sealing of the thermoelements in an inert gas environment is not required as in the case of Pb-Te thermopiles.

Estimates of abort re-entry protection weight penalties are also shown on Figure 1-3. Several methods of re-entry protection were considered. The separated method permits directed re-entry through the Earth's atmosphere with the use of available aerodynamic re-entry shapes, but requires RTG separation from the spacecraft before the onset of re-entry heating. It only appears applicable to a limited number of abort possibilities. The integral method does not require RTG separation but depends on protection cladding which completely surrounds the vulnerable portions of the RTG. It was initially estimated to be too costly in weight, but subsequent analysis has shown promise for graphite cladding techniques and those using refractory metals with oxidation-resistant coatings. From the standpoint of proven technology, fuel containment under re-entry conditions remains to be demonstrated and constitutes the greatest uncertainty on the RTG weight estimates. On the other hand, the basic RTG weight figures (excluding re-entry protection) are considered to be reasonably accurate, since they are based on the established performance of the SNAP-27 generator.

1.2.5 RADIATION SENSITIVITY STUDY

This study was conducted to determine equipment tolerance to the RTG radiation environment consisting principally of neutron and gamma rays. An initial phase of the study determined damage threshold levels of representative spacecraft equipment as defined by the Voyager Phase IA, Task B spacecraft design. Figures 1-4 and 1-5 summarize these threshold levels. Similar threshold damage studies were conducted on 22 science instruments considered to be possible candidates for use on the spacecraft.

A later phase of the study determined the spatial distribution of gamma doses and neutron radiation fluxes emanating from the RTG's. Taking the effect of impeding masses into account, the spatial dose and flux distribution were determined. This resulted in an integrated gamma dose between 50 to 100 rads (C) and an integrated neutron flux between 1 x 10^{10} and 2 x 10^{10} neutron/cm^2 at the equipment bays equivalent to a mission period of 1 year. These are shown on Figures 1-4 and 1-5 and indicate that no problems exist as far as gamma effects are concerned; threshold and moderate effects due to the neutron flux are evident in several areas. The results indicate the need for careful piece-part selection and derating to achieve adequate insensitivity. These are considered to be appropriate techniques for solving this problem.

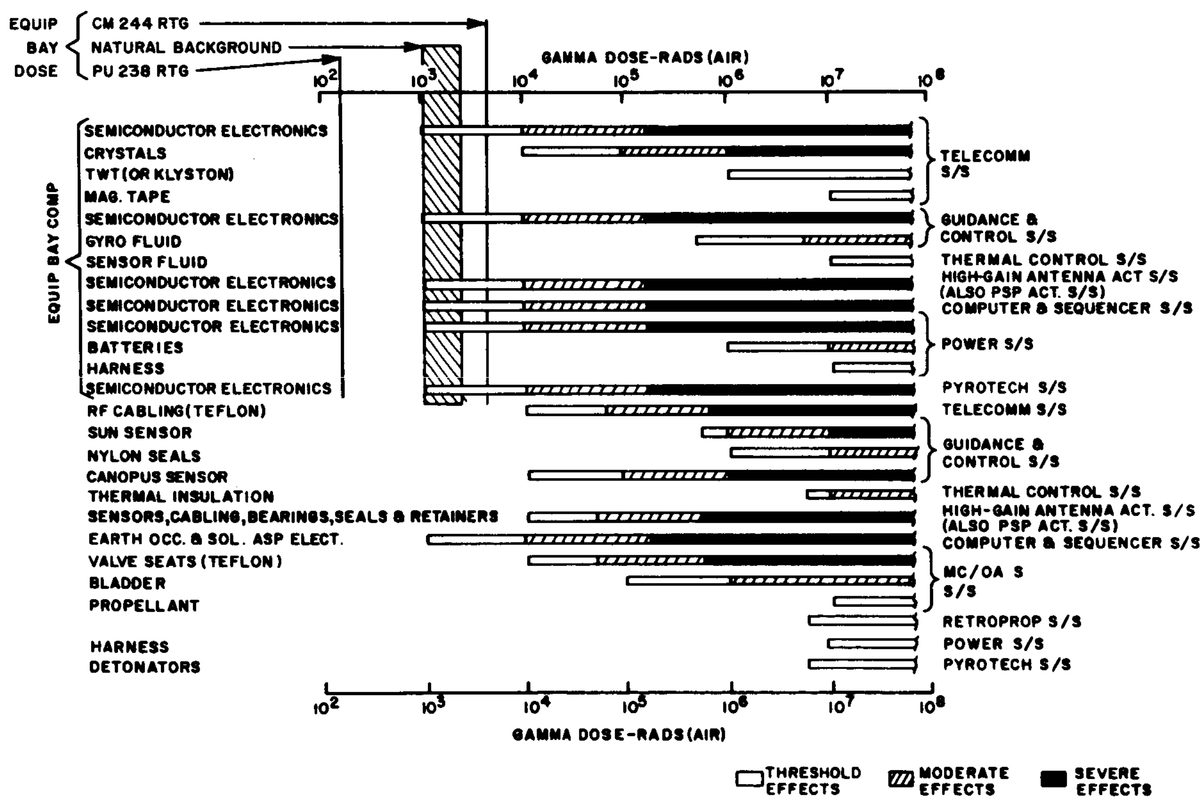


Figure 1-4. Gamma Effects Summary

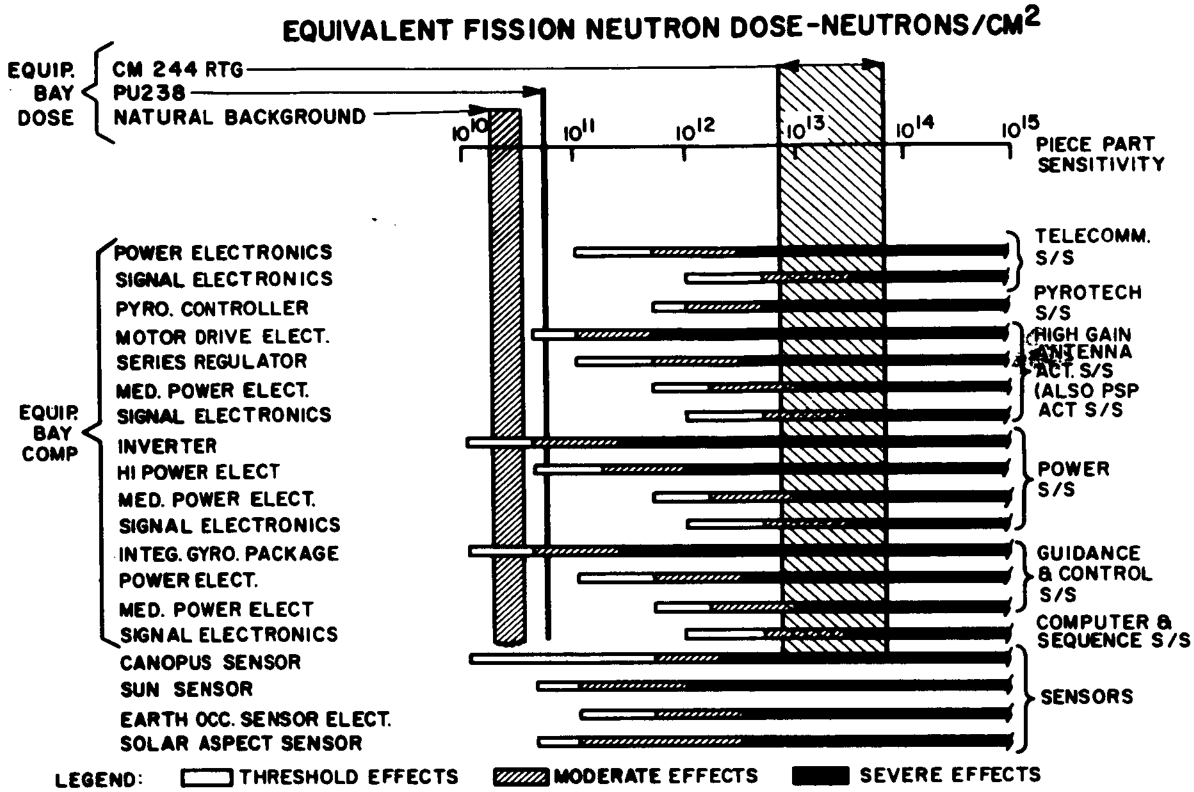


Figure 1-5. Neutron Effects Summary

A similar evaluation was conducted by using Curium-244 as the isotope fuel. Resulting neutron fluxes of 10^{13} to 10^{14} neutron/cm^2 would cause severe effects in practically all subsystems. On this basis, Curium was dropped from further consideration.

The effects on science instruments are generally of a different nature, involving dynamic interference rather than permanent damage. Twenty-two instruments were investigated which can be classed into three categories: (1) a camera group, (2) a spectrometer and radiometer group, and (3) a fields and particles group. Problems were noted for the six instruments falling in the fields and particles group; these instruments are designed to detect natural particles similar to those emanating from the RTG's.

Instruments in the other categories were considered to be generally immune to radiation effects, though this would have to be ascertained on a specific instrument basis. Recent tests at JPL, for example, indicated dynamic interference of a radioactive source with the photomultiplier tube of a UV spectrometer.

1.2.6 COMPARISON—RTG VERSUS SOLAR-POWERED SPACECRAFT

The final activity on the RTG study was the performance of a comparison study summarized below as the advantages and disadvantages of RTG power relative to solar power.

1.2.6.1 Advantages

The advantages of RTG-powered spacecraft are as follows:

a. RTG's are believed to provide higher overall reliability for a number of reasons. First, they operate at relatively constant conditions throughout their life. This is contrasted with temperature cycling experienced by both solar arrays and batteries. Second, they are relatively immune to environmental degradations such as that resulting from ionizing radiation on solar arrays. Third, they permit significant reductions in battery requirements and associated problems of reliable cycle charge/discharge operation. (Batteries that are required with RTG's to provide peak load capability are only used intermittently with correspondingly decreased cyclic stress.) Fourth, with respect to spacecraft failure mode operation, they provide many additional options for corrective action. Fifth, due to removal of the sun-pointing constraint, they permit the use of a fixed high-gain antenna, with gains in reliability resulting from elimination of stepping motors, linkages, bearings and RF rotary joints.

b. They provide greater mission flexibility for missions beyond Mars because of their independence of solar energy.

c. They permit improved ground test verification with respect to predicted space output. In fact, ground power performance is identical to flight power performance. They also permit under-the-shroud power system verification just prior to launch.

1.2.6.2 Disadvantages

The disadvantages of RTG-powered spacecraft are as follows:

a. They are heavier; an improvement of about 20 to 25 percent in specific power (watts per pound) is needed to make them comparable with solar power. This disadvantage must be viewed in the context of present technology. On the basis of developments presently underway, particularly those related to higher thermoelectric efficiency, there is a high expectation that the weight differential will be appreciably narrowed.

b. Development is required relative to isotope fuel containment re-entry protection. This constitutes the largest source of RTG weight uncertainty and should receive early priority in Voyager RTG developments.

c. Certain radiation detection instruments require either special design and/or shielding to be compatible with the RTG environment.

d. Sufficient quantities of Pu-238 isotope may not be available to fill the needs for RTG-powered Voyager spacecraft.

e. Although a detailed cost comparison was not performed under this study, cursory consideration indicates that RTG power is several times more costly than solar power for Mars missions.

1.3 RECOMMENDATIONS

If the recommendation on the use of RTG power were to be made on the limited considerations of (1) only Mars missions and (2) missions without capsules, then it is difficult to provide adequate justification for RTG's when one considers the flight-proven performance of solar power in Mars fly-by and Earth orbiting missions. However, the decision and the timing will, of course, be influenced by (1) the desired flexibility of Voyager to perform missions beyond Mars and (2) by the capsule program. Incorporation of RTG's in Voyager capsules will of necessity solve many of the problems common to both capsule and spacecraft, e.g., abort re-entry protection, science compatibility, and general nuclear safety problems.

Thus, it seems prudent to provide for the possibility of eventual incorporation of RTG power in the Voyager spacecraft. By designing a "convertible" spacecraft, later modifications may be minimized. Specific recommendations are:

a. Design the shroud to be consistent with the incorporation of shroud wall cooling.

b. Design spacecraft subsystems for operation in an RTG radiation environment.

c. Design solar power subsystem output characteristics such that user subsystems are relatively unaffected by the substitution of an RTG subsystem.

d. Where possible, design science instruments for operation in an RTG radiation environment.

SECTION 2
PLANETARY QUARANTINE STUDY

2.1 GENERAL

In recognition of the importance of preserving the natural microbiological state of the planets during initial exploratory flights, National Policy on Planetary Quarantine has been established. This is to ensure that planetary microbial contamination due to accidental implant of Earth life forms will not interfere with attempts to detect the existence and/or nature of extraterrestrial life.

For the Voyager Mars exploration missions, measures to comply with the National Policy on Planetary Quarantine will consist of: (1) enclosing the sterile landing capsule in an impermeable biological barrier in order to maintain its isolation from all possible sources of microbial contamination and (2) identifying all other mechanisms of possible microbial contamination from nonsterile sources and assuring that these mechanisms are adequately understood and controlled.

Using a systems engineering approach, the Task C Planetary Quarantine Study has identified contamination sources, developed a basic quarantine mathematical model, conducted several experimental and analytical investigations to furnish supporting data, outlined the uncertainties which remain to be determined, and correlated the conclusions available at this time with the requirements of the Voyager Mission.

2.2 STUDY OBJECTIVES

The basic objective of the Planetary Quarantine Task was to perform analytical and experimental studies in order to show the effects of the Planetary Quarantine requirements on the Voyager Program. Emphasis was placed on the possible ways of contaminating Mars via sources which had not previously been studied in great depth, such as various ejecta leaving the unsterilized spacecraft and carrying viable organisms to Mars. Meeting the basic objective included showing the effect of the Quarantine requirement on: (1) the design of the orbiting spacecraft hardware elements, (2) the manufacturing and facilities requirements, and (3) the operating mission.

A further goal was to recommend a satisfactory, but not unnecessarily costly, solution for each potential contamination problem. Reliability, monetary and schedule penalties associated with over-solving the problem appeared so great as to warrant detail study and characterization of each potential contamination source.

In conjunction with the studies, two other goals were considered of major importance: (1) to thoroughly document the study activities so that various interested parties could critically assess the study results and (2) to develop flexible analytical tools and experimental approaches so that the work performed on the study could be updated and/or modified as necessary.

2.3 STUDY DESCRIPTION

Figure 2-1 shows the subtasks undertaken on this study and their interrelationship. Basically, the Group I tasks include the development of the computerized analytical tools necessary to calculate the probability of contaminating Mars. The Group II tasks are basic experimental and/or analytical studies needed to develop the input information necessary in using the analytical tools. The tasks in Group III represent the main system analysis wherein the effects of the Quarantine requirement are studied and evaluated.

The mathematical model necessary to calculate the probability of contaminating Mars must, in effect, be a representation of the physical pehnomena associated with the various contamination sources. In general, these physical phenomena consist of certain major elements.

The first element involves the number of viable microorganisms initially present in each source. The second element concerns the transport phenomena — the means by which these initially present microorganisms travel from the spacecraft to the Martian surface. The third element concerns the effect of various lethal environmental factors or kill mechanisms which tend to reduce the number of viable organisms.

One method of organizing the elements necessary in the calculation of the probability of contaminating Mars is a matrix, as shown in Figure 2-2. The rows of the matrix represent the

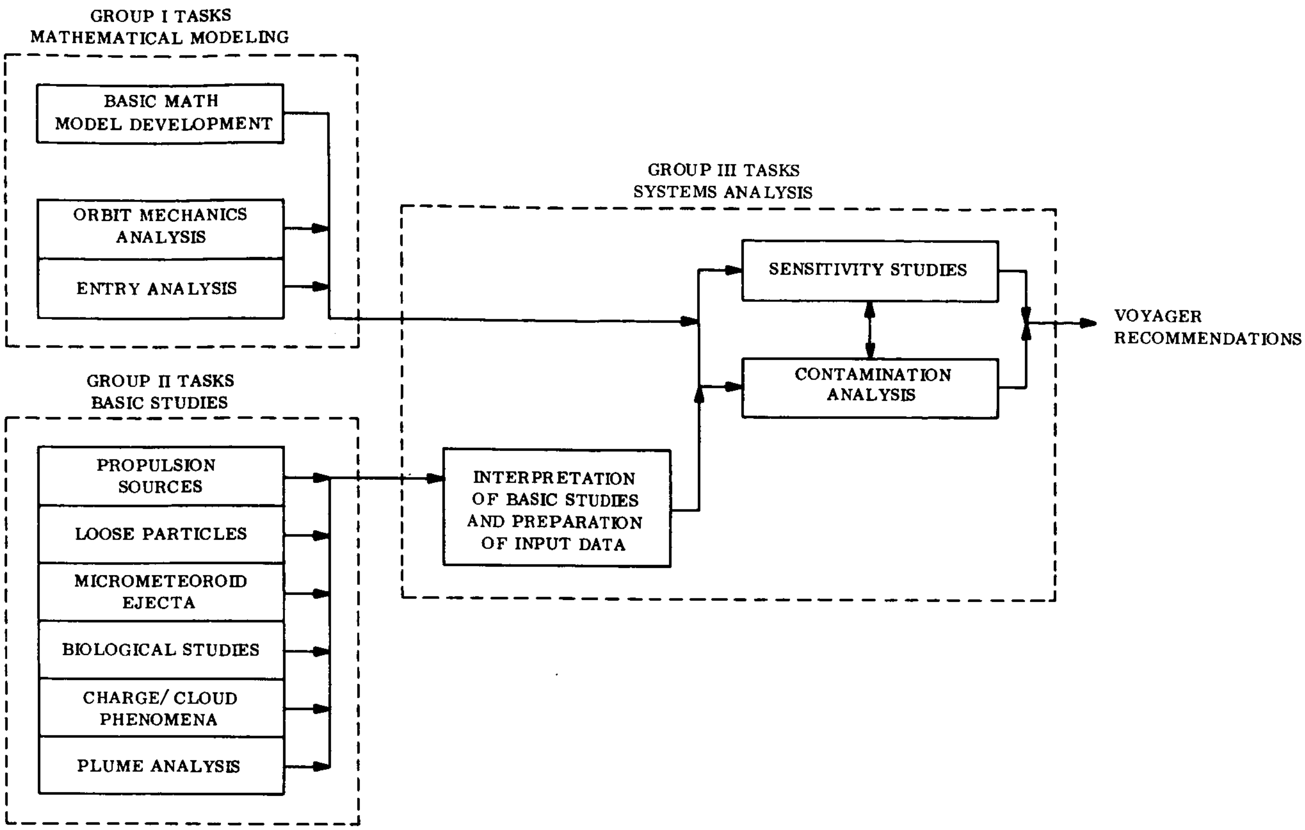


Figure 2-1. Planetary Quarantine Study Simplified Work Flow Diagram

ROUTE TO MARS →

SOURCE OF CONTAMINATION	1 INITIAL LOADING-V.O.	2 SURVIVE DURING TRIP	3 EJECTION PROCESS	4 TRANSPORT PROCESS	5 SURVIVE DIE-OFF	6 SURVIVE VACUUM	7 SURVIVE UV	8 SURVIVE OTHER SOLAR RADIATION	9 SURVIVE ENTRY HEATING	10 SURVIVE MARS ENVIRONMENT	11 NUMBER V.O.'s TO MARS SURFACE PRIOR TO TIME T
ATTITUDE CONTROL GAS SYSTEM											
ORBIT INSERTION ENGINE											
LOOSE PARTICLES											
MICROMETEOROID EJECTA											

Figure 2-2. Mathematical Model Format (Contamination Analysis Matrix)

various potential sources of Mars contamination. The columns of the matrix describe how particles may find their way to the surface of Mars and the effects of various lethal environments on these sources.

A summary of the various possible sources of Mars contamination is shown in Figure 2-3*. Most potential sources of contamination within Categories 2 through 5 in the figure were investigated in the present study, although major emphasis was placed on source Category 5 - Flight Spacecraft Ejecta/Efflux Impacts since this category had received little previous investigation. The objectives of the present study did not include the investigation of the first source, concerning the initial sterilization of the capsule.

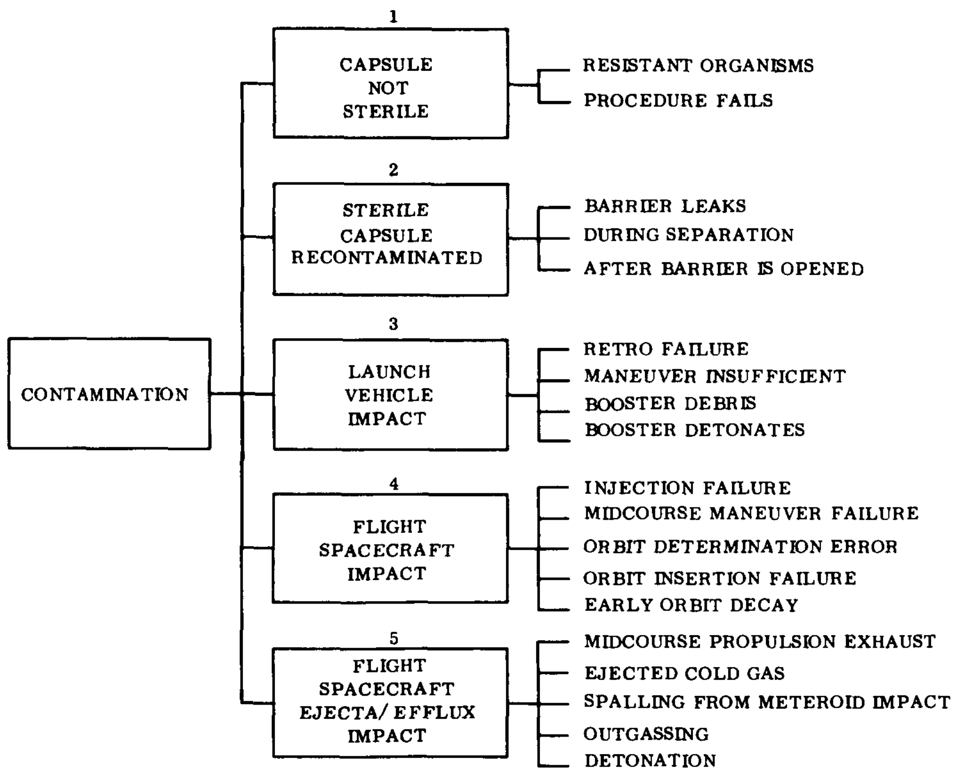


Figure 2-3. Sources of Contamination

*Planetary Quarantine Plan, Voyager Project, 15 March 1966, 3rd Revision, 1 June 1967, NASA-OSSA, Voyager Project Office, Doc. No. 818-11-PQ-001.

A series of computer programs was developed which essentially performs the mathematical analysis represented by the matrix shown in Figure 2-2. Both the input and output information for the matrix is treated in the form of probability distributions, rather than simply average or worst-case values.

Three fundamental types of calculations are represented by the contamination analysis matrix. The first type of calculation is solely within a column and is the calculation of the probability of a single viable organism surviving the process described by the column heading. Two examples of this are the orbit mechanics (transport) and the entry heating effects.

The second type of calculation is that of starting with an initial number of organisms (actually a distribution) in Column 1 and successively operating on this distribution with the probability of survival specified, or calculated, for each column. Hence, as this operation proceeds from left to right, the number of viable organisms is reduced until the final number getting to Mars is obtained. This operation is not a simple multiplication of the distribution by the column probability of survival. Rather, it is a probability calculation which consists of operating on an initial probability density function by a set of conditional probability density functions, thus generating a marginal probability density function as the output. Details of this operation and the computer program developed for recursively performing the operation any number of times are presented in Volume 3 of this Final Report.

The third fundamental calculation consists of adding distributions - the operation needed for combining the row answers in Column 11.

The necessary input data for Column 1 concerning the number of microorganisms associated with the various sources were primarily obtained by collecting existing data on the number and types of microorganisms associated with spacecraft hardware manufactured under different conditions. A Biological Burden Catalog, for various types of cleanroom manufacture, was developed. Experimental activities were undertaken for the several propulsion systems to provide input data concerning the number of microorganisms associated with these potential contamination sources.

The activities consisted of bioassaying solid propellants and attitude control gas systems.

An analytical study was conducted to evaluate the potential lethal effects of the several interplanetary environmental factors, principally the ultraviolet radiation and the interplanetary thermal environment. This study provided partial input information for Columns 2, 5, 6, 7, 8, and 9. An experimental program was conducted to determine the sporicidal/bacteriocidal characteristics of several typical rocket propellants: aluminized polybutadiene, nitrogen tetroxide, Aerozine-50 and hydrazine. These experiments provided information on the various propulsion sources for Column 2.

Column 3 represents the effect of the ejection process on the contamination source and contains information of two types: (1) the rate at which microorganisms are ejected from the system and (2) whether viable organisms are killed in the process. Input data for this column were virtually nonexistent at the initiation of this study, so several experimental tasks were undertaken to provide the needed data.

A large experimental program was undertaken wherein small-scale rocket motors, inoculated with a known quantity of test spores, were fired into heat exchanger/collection chambers and the entrapped effluent biologically assayed to determine the effects of combustion environments on microorganism viability. Solid propellant, bipropellant and monopropellant propulsion systems were evaluated and the time-temperature in the experiment closely simulates the actual space time-temperature profile for each system. Eighty-two test firings were performed during the development, calibration, and lethality test phases of this activity.

A breadboard single-axis cold gas attitude control system was designed, utilizing typical hardware and components. The system was tested on a typical interplanetary mission duty cycle, incorporating simulated vibration loads. In addition, a portion of the attitude control system of an operational spacecraft configuration was similarly evaluated. In both cases, the gaseous effluent from the exhaust nozzles was collected by biological filter elements and subsequently assayed.

High-velocity micrometeoroids impinging on an unsterile spacecraft surface may cause the ejection of viable and nonviable particles from the surface material. A study was undertaken to determine the mechanisms for particle ejection and to define the environment created by the impingement/ejection phenomena, to determine the physical characteristics of the ejected particles, and to determine the number of viable organisms surviving the impingement/ejection environment. Both experimental and analytical tasks were conducted. The experimental effort consisted of: (1) firing simulated micrometeoroids, at a velocity of 30,000 feet per second, at targets which had been inoculated, on the top and bottom, with a known number of microorganisms; (2) catching the target ejecta; and (3) bioassaying as well as counting and measuring ejecta. The analytical effort involved activities such as extrapolation of velocities used in the experimental effort to cover the full range of the actual micrometeoroid environment and the compilation and analysis of related work by other investigators.

Column 4 represents the analysis of the trajectories of microorganisms leaving the spacecraft. This analysis was primarily concerned with the parameters which may influence the particle flight paths, ejection velocity, ejection direction, ballistic coefficient (M/C_DA), and the time in the mission at which ejection occurs. For the analysis the mission was divided into three parts: (1) the heliocentric transfer orbit from Earth to Mars, (2) the orbit insertion maneuver from heliocentric to areocentric orbit, and (3) long-life orbital operations about Mars. Each of the three phases of the mission consists of events which are either favorable or unfavorable to the quarantine constraint. For the heliocentric phase the favorable result occurs when the particle trajectory is sufficiently perturbed so that its separation from the spacecraft at encounter is great enough to avoid the possibility of planetary capture. Particles which do not achieve this favorable separation distance are assumed to enter the Martian atmosphere and encounter the possibility of sterilization by entry heating. The orbit mechanics study, therefore, provides the initial conditions for the entry heating analysis.

In the Mars orbital phases of the mission, results favorable to planetary quarantine occur when the particle orbits are not greatly perturbed from the orbiting spacecraft orbit, so that they remain in orbit throughout the quarantine period. When the particle orbits decay within the quarantine period, conditions for atmospheric entry are again provided, so that entry heating may be considered.

The entry heating analysis in Column No. 9 was performed to evaluate the potential thermal kill of microorganisms during entry through the Mars atmosphere. In order to contaminate Mars, these viable organisms would have to survive the entry phenomena. Survivability has been quantified in terms of the entry time-temperature response of the microorganisms. Two analytical studies were conducted as a part of this study. First, the trajectories of bacteria were evaluated to define the time-temperature histories of entering microorganisms as a function of the Mars atmosphere, the ballistic coefficient of the particle, its emissivity, absorptivity, velocity, initial temperature and starting altitude. An analytical model was developed to represent the thermal history of an entering microorganism as a function of the above parameters. This model is based on free molecular flow, which has been found appropriate for the small particle sizes and the aerodynamic environment under consideration.

The second study concerned the development of an expression for the thermal kill of entering microorganisms. An analytical expression was developed to represent the probability of survival as a function of the temperature history, the concept of a "lethality integral" having been defined to represent the cumulative kill effect for varying temperature. An algorithm was developed to compute the lethality integral, hence the survival ratio, for any given set of four critical parameters: the ballistic coefficient, the entry angle, the entry velocity and the particle emissivity.

An experimental program was conducted to determine the critical thermodynamic properties of a specific type of microorganism. Bacillus subtilis var. niger was used, as this spore form is presently the standard being used in the sterilization and quarantine fields. The solar absorptance and hemispherical emittance of the test spore were measured and the results subsequently incorporated into the entry heating analysis.

Column No. 10 provides for a "growth and contamination" factor as discussed in the Voyager Planetary Quarantine Plan*.

*Planetary Quarantine Plan, Voyager Project, 15 March 1966, 3rd Revision, 1 June 1967, NASA-OSSA, Voyager Project Office, Doc. No. 818-11-PQ001.

Column 11 provides the desired answer — the probability of contamination for each particular source.

The various analytical and experimental tasks discussed above, as well as others, are described in detail in Volume 3 of this Final Report.

2.4 STUDY RESULTS AND RECOMMENDATIONS

2.4.1 MARS ORBITAL ALTITUDES

Primarily, two Mars orbits were studied: a 1,000 x 10,000 km nominal orbit and a 500 x 10,000 km orbit. For these two orbits the difference in the contamination probability, for several sources (those having low velocity ejecta), was between 1 and 3 orders of magnitude, which in some cases is enough to cause a problem.

Studies concerned with the accidental impact of the spacecraft or biobarrier concluded that periapsis altitudes of from 300 to 1000 km would be safe, depending on the atmosphere model; the lower altitudes are associated with the current best estimate of the atmosphere*. However, three sigma guidance errors could be on the order of 300 km; hence, considerable margin is required.

The recommendation of this study is that planetary quarantine constraints should be placed on the minimum value of orbit periapsis; the recommended value at this time is 1,000 km. Further studies might, if deemed strongly desirable for other reasons, reduce this constraint by a few hundred kilometers.

2.4.2 SPACECRAFT MANUFACTURING

Based on the study of the various potential sources of contamination which are affected by the spacecraft surface biological and particulate loads, it is concluded that the planetary quarantine requirements can be met, with an adequate safety margin, if the maximum bioload

*Hess, D.S., and E. Pounder, "Voyager Environmental Predictions Document," JPL-SE003 BB001-1B28, 26 October 1966.

is essentially as shown in Figure 2-4 and the maximum particulate load is as shown in Figure 2-5. These loads should be achievable if the spacecraft manufacturing includes the three elements of: (1) good cleanroom facilities, (2) good cleanroom operating procedures, and (3) good hardware cleaning operations. These three items are also consistent with what most likely would be desired from a standpoint of manufacturing for high-reliability hardware.

The study revealed that the mere specification of the cleanroom class is not sufficient to control the biological and particulate loads. The second and third steps mentioned above are also required. The study also revealed that steps used to minimize either the biological load or the particulate load are effective in reducing the other as well.

The specific recommendation made is that the bioload and particulate loads shown in Figure 2-4 and 2-5 be achieved. These loads should be achievable by the above discussed three steps within the present state of the art without ethylene oxide (ETO) decontamination.

2.4.3 ATTITUDE CONTROL GAS SYSTEM

The study revealed that the attitude control gas system posed a potential hazard to contamination and that some degree of bioload reduction was necessary. Certainly heat sterilization would meet the requirement; however, the magnitude of the problem does not at all indicate the need for sterilization. Similarly, ETO decontamination would more than meet the requirement. The study also indicated that the use of an onboard filter (3-micron biological type), even if it were between the storage tank and the pressure regulator, would maintain the probability of contamination from this source well within its allotment (e.g., the calculated probability of contamination, using filters, was 2.5×10^{-9} versus an allocated probability of contamination of 1×10^{-6}).

The possibility also exists that the attitude control gas system would not need any bioload reduction, if a larger allocation were assigned to this one source. It is felt prudent, however, to recommend that some steps be implemented to reduce the bioload of the attitude control gas system over that which would normally be expected, if the system were manufactured according to present practice and the gas was biofiltered when loaded into the system. The use of the onboard filter — if practical — is recommended.

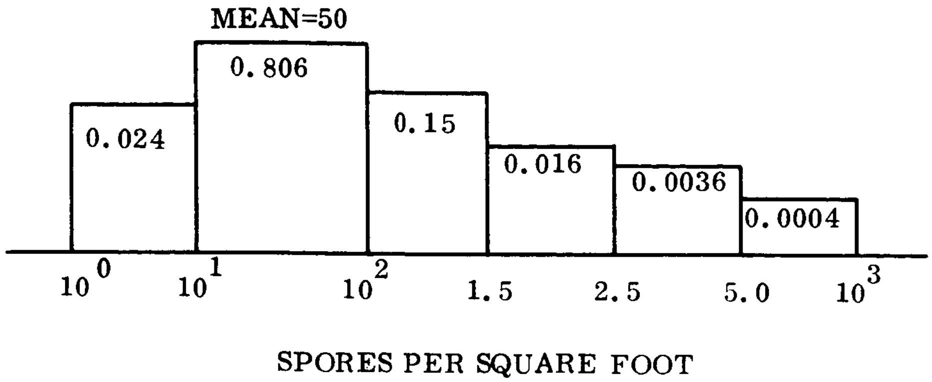


Figure 2-4. Recommended Maximum Biological Surface Load

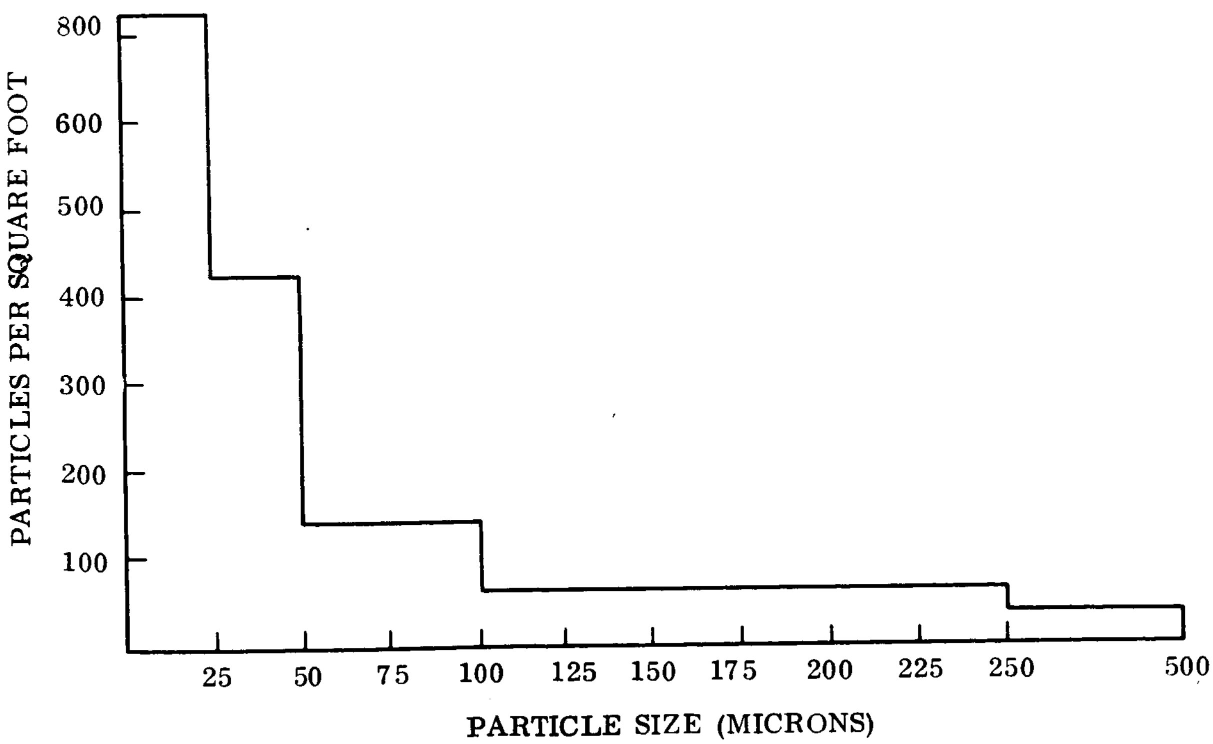


Figure 2-5. Recommended Maximum Particulate Surface Load

2.4.4 PROPULSION SYSTEM CONTAMINATION SOURCES

To satisfy the planetary quarantine requirement, the midcourse, orbit insertion, and orbit trim propulsion systems must be studied from two main viewpoints. The first concerns the contamination threat due to the ejecta from the engines carrying viable organisms directly to Mars. The second concerns the accidental impact of Mars by the unsterilized spacecraft as a result of the maneuver. The study results for the ejecta case are treated in this section. Accidental impact will be discussed in Section 2.4.5.

a. Midcourse/Orbit Trim Propulsion

The analysis indicates that no contamination threat exists, from the combustion exhaust ejecta, for any possible midcourse maneuver. Furthermore, the orbit trim maneuvers studied, including both the raising and lowering of spacecraft orbit, were favorable from an ejecta standpoint. However, the number of potential orbit trim maneuvers makes an all inclusive study impractical, and final determinations concerning the orbit trim contamination source must await better mission definition. Consequently, the recommendation is that, from the standpoint of combustion exhaust ejecta, no planetary quarantine constraint be placed on the midcourse maneuvers. The orbit trim maneuvers must be reexamined when a mission design is selected.

The propellant pressurant gas for liquid propulsion systems (all midcourse/orbit trim systems studied employed liquid propellant systems), though not specifically studied, represents a cold gas ejecta source very similar to the attitude control gas system previously discussed. It is recommended that the specific pressurant system selected be studied, and, if necessary, solutions similar to those indicated for the attitude control system (e.g., on-board filters or decontamination) be implemented.

b. Orbit Insertion Propulsion

Both liquid bipropellant and solid propellant orbit insertion systems were studied. Two specific orbit insertion maneuvers were examined to determine the effect on the probability of contaminating Mars via the engine ejecta. The best-case maneuver, from a quarantine standpoint, was one which occurred past hyperbolic periapsis. No contamination threat existed for either the bipropellant or solid propellant engines. The worst-case maneuver studied occurred well before hyperbolic periapsis. For this case, the bipropellant engine was safely within its contamination allocation. Many of the maneuvers most desirable, for reasons other than quarantine, appear to be reasonably satisfactory.

The above conclusions were based on engines manufactured under normal conditions. Consequently, the recommendation for the orbit insertion engine proper and the orbit insertion maneuver is to establish a requirement that the selected orbit insertion (including pointing failure modes) be analyzed from a combustion exhaust ejecta viewpoint and be approved from a planetary quarantine standpoint.

The propellant pressurant gas, associated with a bipropellant orbit insertion system, is very similar to the attitude control gas system and should be studied and treated as described above, under the Attitude Control Gas System.

The contamination potential of thrust vector control (TVC) fluids, associated with a solid propellant orbit insertion system is, again, similar to the attitude control system. The filtration and decontamination recommendations offered for the ACS are certainly applicable, although design solutions (e.g., choice of TVC fluid) may be equally suitable. Consequently, it is recommended that quarantine requirements be established as a TVC design constraint and that the selected design be analyzed from a standpoint of contamination potential.

2.4.5 ACCIDENTAL IMPACT OF LAUNCH VEHICLE/SPACECRAFT/BIOBARRIER

The probability of impacting Mars with the last stage of the launch vehicle, the spacecraft bus, or the biobarrier can be made to meet any reasonable planetary quarantine allocations.

The two major variables are: (1) design of the maneuver (whether launch trajectory, midcourse guidance, orbit insertion, or orbit trim), and (2) the corresponding hardware accuracy and reliability required (for example, the reliability associated with being able to shutdown the propulsion engine at the end of the maneuver).

The many conflicting mission requirements, only one of which is planetary quarantine, concerning the design of these various maneuvers are such that continual analysis and tradeoff studies must be undertaken throughout the program to ensure that the probabilities of contamination are within specified allocations for these sources.

The results of the present study indicate that, for the assumed guidance errors and atmosphere model:

a. Launch vehicle retrofire is not required.

b. Aim point biasing and three midcourse corrections will satisfy the quarantine requirement.

c. The biobarrier can be safely released in a 1,000 x 10,000 km orbit.

d. Orbit insertion maneuvers exist which are not overly sensitive to pointing and timing errors.

e. An orbit trim maneuver which lowers the spacecraft periapsis is potentially dangerous because of a possible failure of the engine to shutdown.

However, all of these items must be continually studied and traded off during the program.

2.4.6 CAPSULE RECONTAMINATION

The study of the capsule recontamination concentrated on: (1) providing a framework for analyzing the probability of contaminating Mars via a recontaminated capsule, (2) identifying the possible mechanisms for transferring viable organisms from the spacecraft to the capsule, and (3) qualitatively assessing several mission and hardware alternatives.

The lack of design definition for the capsule and biobarrier, in addition to the absence of data concerning mechanisms for transporting organisms from the spacecraft to the capsule, precluded generating any meaningful quantitative results. However, the study did indicate that the problem appears to be of sufficient magnitude that steps should be taken to minimize the contamination probability. Consequently, the tentative recommendation is made to maintain the biobarrier intact as long as possible to minimize the capsule exposure time. This implies carrying the biobarrier into orbit. In addition, it is recommended that in-depth studies of capsule recontamination be undertaken. These studies require investigation of the phenomena involved with the transport of viable organisms from the unsterilized spacecraft to the capsule. Also, better hardware definition for the capsule is required for the studies to be most useful.

2.5 GENERAL STUDY CONCLUSIONS

During the study, the various individual investigators have identified, in their appropriate reports, several areas wherein additional work might be useful. This section sets forth a few important general conclusions concerning planetary quarantine which have evolved as a result of this study:

a. Presentation of the probability of contamination from ejecta type sources, in the form of a probability distribution, provides new information not previously available. The results show that even those sources which did not meet the probability of contamination criteria for "one or more" viable organisms would be safe, by many orders of magnitude, if the real concern with contamination were with "a few" or more organisms getting to Mars. This question becomes important when the dollar and/or reliability cost of meeting a "one or more" criterion greatly exceeds that for meeting, say, a "two or more" criterion.

b. Continuing analysis, along the lines performed in this study, should be a regular part of the Voyager program. This will permit the continuing assessment, from a quarantine viewpoint, of the effects of modified or new data in all areas of the program.

c. Additional effort in several biological areas would be very desirable to help advance the state of the art of the relatively new field of spacecraft-related biology. Two examples are: (1) development of more rapid procedures to determine biological loading of spacecraft hardware to permit meeting the tight hardware schedules for a Voyager-type program, and (2) more data concerning the lethal effects (singly or combined) of several environmental factors, e.g., solar radiation.

d. Continued progress in many areas of the overall space program will provide better data, which in turn, will permit some of the uncertainties in the analysis to be reduced. An example of this would be improved knowledge concerning the micrometeoroid environment and the Mars atmosphere.

e. The development of an overall mechanized computer program for calculating the probability of contaminating Mars for a mission might be very desirable after the program has progressed sufficiently downstream into the hardware stage. Such a program could then be routinely exercised with the latest input information on all sources. It would provide the analysis needed as a basis for certification for launch from a planetary quarantine standpoint. Additionally, it could be used during the flight program to provide rapid assessment of mission alternatives which might be required by design or by failure of some part of the spacecraft system.

SECTION 3

APPLICATION OF REDUNDANCY STUDY

3.1 SUMMARY

The objective of this study was to develop logical methods for applying redundancy to Voyager at both the project and system level. The questions to be answered were:

a. Within the spacecraft system, with a given weight budget, how should the spacecraft system manager select redundancy to obtain the highest probable mission return per pound of redundant hardware?

b. At the project level, what strategy should the project manager follow in selecting the configurations to be launched at each opportunity? Should he, for example, fly simple, reliable probes during early missions, or should he attempt more valuable planetary landers, with their attendant higher risk?

The principal accomplishments of this study were the successful development of two computerized tools to assist the system manager and the project manager in their roles as decision makers. The basic approaches are similar at each of the two levels.

Briefly, at the systems level, the approach is as follows (Figure 3-1):

a. First, the mission profile is described with functional flow diagrams to a level of detail such that each function can be associated with equipments which can perform the function. Then, for each function, different equipments with varying degrees of redundancy are described in terms of weight, reliability, and failure modes.

b. Simultaneously, the possible outcomes of the mission and their relative values are defined. In contrast to the usual concept of maximizing system reliability, this study emphasized the selection of redundancy to maximize the probability of achieving valuable mission outcomes.

c. Then, system hardware failure modes are related to the probabilities of achieving the various mission outcomes.

d. Given, (1) the hardware alternatives and their weights, (2) the probabilities of arriving at the various mission outcomes, and (3) the values of each possible outcome, the technique developed determines, for any total system weight allocation, that spacecraft configuration which yields the maximum expected mission worth.

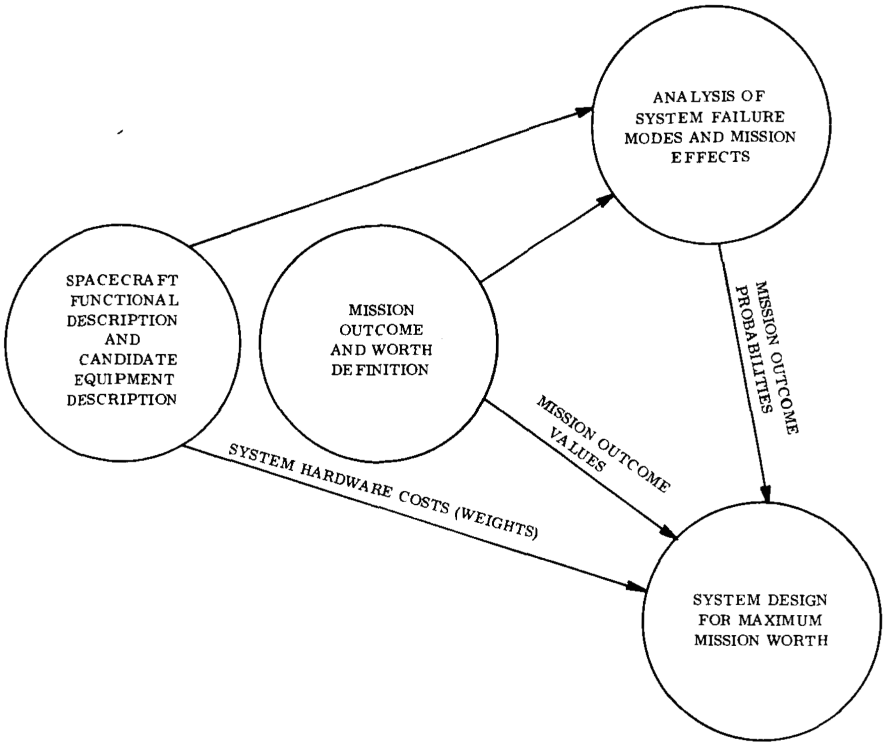


Figure 3-1. Selection of System Redundancy

Using the above approach, the spacecraft system was described functionally in considerable detail. A comprehensive system failure mode and mission effects analysis was performed. To handle the massive load of computation (some 30,000 tabulating cards are required to describe the hardware alternatives above), a computer program system was developed and exercised, resulting in a set of spacecraft configurations, each optimized for maximum mission worth, for a range of varying weight constraint.

At the project level (Figure 3-2), the problem is that of deciding upon mission configurations, specifically, the number and types of launch vehicles, spacecraft, and capsules to be launched at the first opportunity and at subsequent opportunities.

Like the system level, mission configurations are selected by combining - (1) the values of possible project outcomes, (2) the probabilities of potential configurations achieving these outcomes, and (3) the costs of each configuration - to determine, for any project dollar

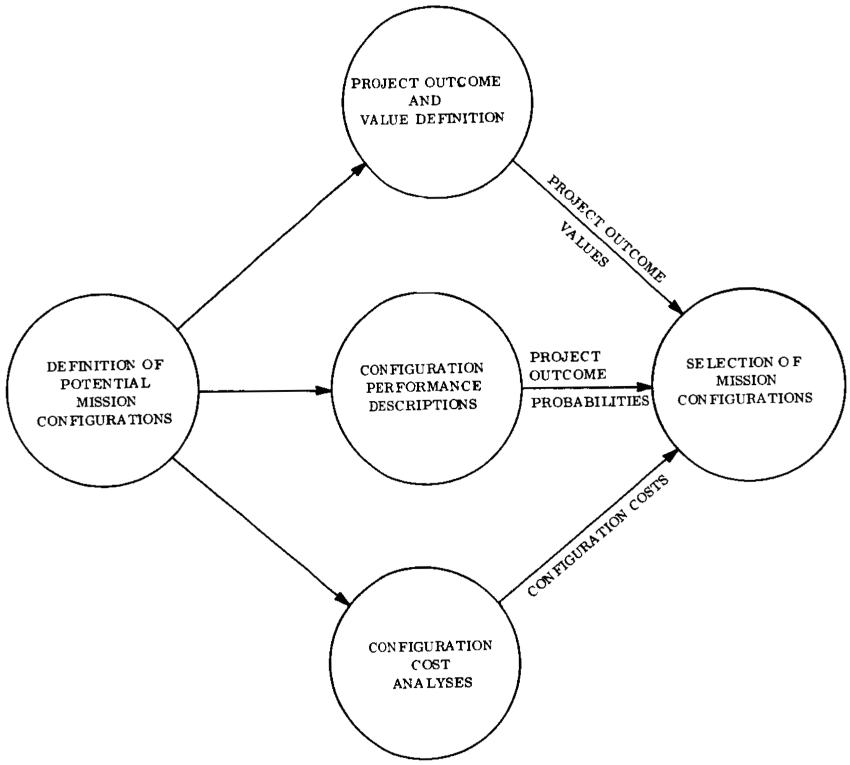


Figure 3-2. Selection of Mission Configurations

budget, that sequential strategy of mission configurations which offers maximum return of project value. The project selection effort marked the first application of decision analysis to the space program.

The real value of the Redundancy Study lies, not so much in the specific system and mission configurations generated as results, but rather in the development of two versatile tools to serve as aids to the system and project manager. In the design of both tools, the required input data was kept in a form and at a level familiar to both decision makers.

It is recognized that there will always be some degree of uncertainty in some of the inputs to the decision process, e.g., the subjective assignment of value to various mission or project outcomes. One of the important uses of these tools is that of determining the sensitivity of the decision to the uncertain input, over any specified range of uncertainty.

It is felt that the techniques developed under this study can be of significant value in the design of various system and mission elements of the Voyager project or, more generally, in the design of any complex space project.

Some 300 pages are devoted to this study in Volume 4 of this Final Report. In this section, the highlights of this study are covered first for the system problem, and then for the project problem.

3.2 SELECTION OF SPACECRAFT SYSTEM REDUNDANCY

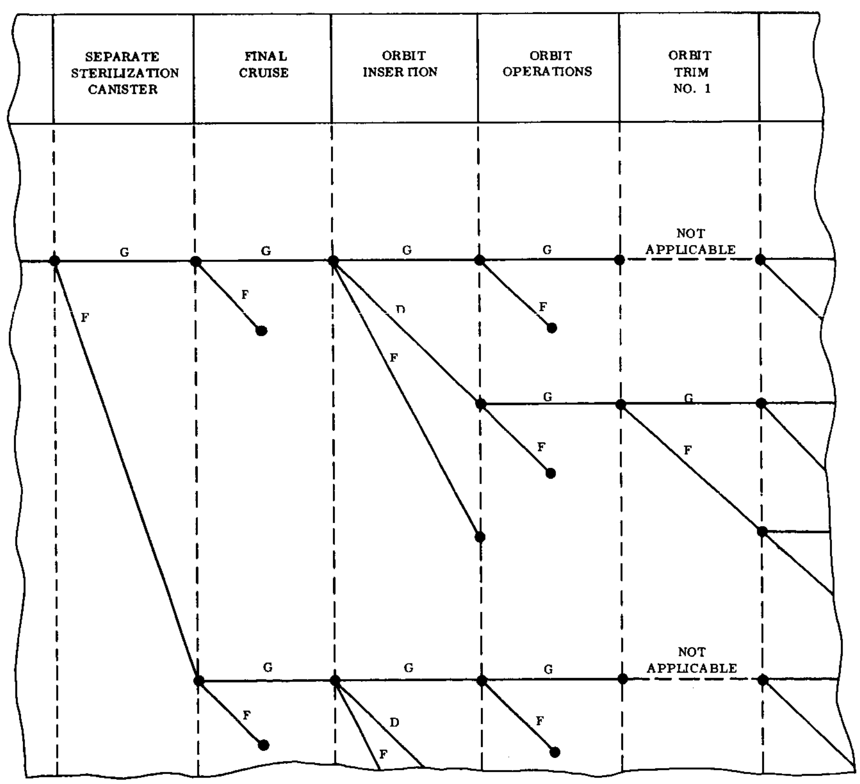
As a starting point in the study, a typical 1973 mission configuration and mission profile were selected as a reference. The reference mission configuration consisted of one Saturn V vehicle with two planetary vehicles, each planetary vehicle consisting of one spacecraft and one capsule. The reference mission profile was defined by functional flow diagrams of sufficient detail such that the functions to be performed could be correlated with discrete elements of spacecraft hardware. This was a major accomplishment of the study, requiring some 8000 different functions.

The identification and description of system hardware at the lowest level of the mission flow diagrams was basically a two-step process. First, the elements of the single-string (nonredundant) spacecraft system were defined. Then, for selected functions, alternate equipments with varying degrees of redundancy were defined. It was from this pool of candidate redundancy that the actual system redundancy was selected. Even though the redundancy pool was not large, it was calculated that, by applying all possible combinations to the single-string system, some 10^{21} possible system configurations could be formed! Finding the best configurations was a task of the first order.

Failure rates or reliabilities for the various single-string and potentially redundant components were then established. To accomplish this, in a uniform manner, a standard failure data base at the piece-part level was developed to serve as the reference for assessing hardware reliabilities.

In parallel with the mission and system definition, several trade studies were performed during the study. These trade studies were performed to assist in the definition of the reference mission configuration and profile and the single-string spacecraft system, and to identify candidates for redundancy, with special emphasis on functional redundancy. Abstracts of these studies are contained in Volume 4 of this Final Report.

Identification of the possible outcomes of the mission started with the mission functional flow definition. A logical outcome tree was developed based on the performance of the first-level functions (subphases) of the mission profile. A fragment of this tree is illustrated in Figure 3-3 for the end of transit and the beginning of orbit operations. The terminal nodes on the mission outcome tree define a set of some 1000 different mission outcomes.



LEGEND
G: GOOD
D: DEGRADED
F: FAILED

Figure 3-3. Mission Outcome Tree

Worth was assigned to each of the branches of the mission outcome tree. The worth of a particular outcome was simply the sum of the worths of the branches which had to be traversed to reach that outcome. The expected worth of the mission (the index of the optimization procedure) was defined as the worth of each possible outcome multiplied by its probability of occurrence and summed over all mission outcomes.

Given the system definition and mission outcomes, the various modes of failure and degraded performance of the single-string and potentially redundant system hardware were then related to mission outcomes. There resulted a comprehensive system failure modes and mission effects matrix.

Depicted graphically in Figure 3-4, the goal of the optimization computer program was to extract from all possible system configurations that subset of configurations, each member of which yields the maximum mission expected worth (MEW) of all configurations of comparable weight. Since the sheer magnitude of some 10^{21} configurations rendered impractical the consideration of each possible configuration, the optimization routine employed a variation of dynamic programming to extract this set of optimal configurations.

Figure 3-5 illustrates the basic MEW-weight curve for the case of a single planetary vehicle and launch vehicle mission for the potential redundancy defined during the study. Mission expected worth was normalized to a maximum of 100 percent , and the MEW of the single-string system was calculated at about 42 percent. The weight of the single-string system was calculated at 16, 970 pounds.

Nearly all of the increase in MEW in Figure 3-5 occurs within the first 80 pounds of weight. Furthermore, study of the compositions of the systems within this range reveals that this occurs in two stages. The configurations of the first 40 pounds are made up by gradually adding redundancy to the following hardware:

Computer & Sequencer:	PSP science control, solar aspect sensor, power conversion, countdown chain
Guidance & Control:	Attitude control sensors, electronics, and power supply
Power:	Logic and switching
Mechanisms:	Pyro controller

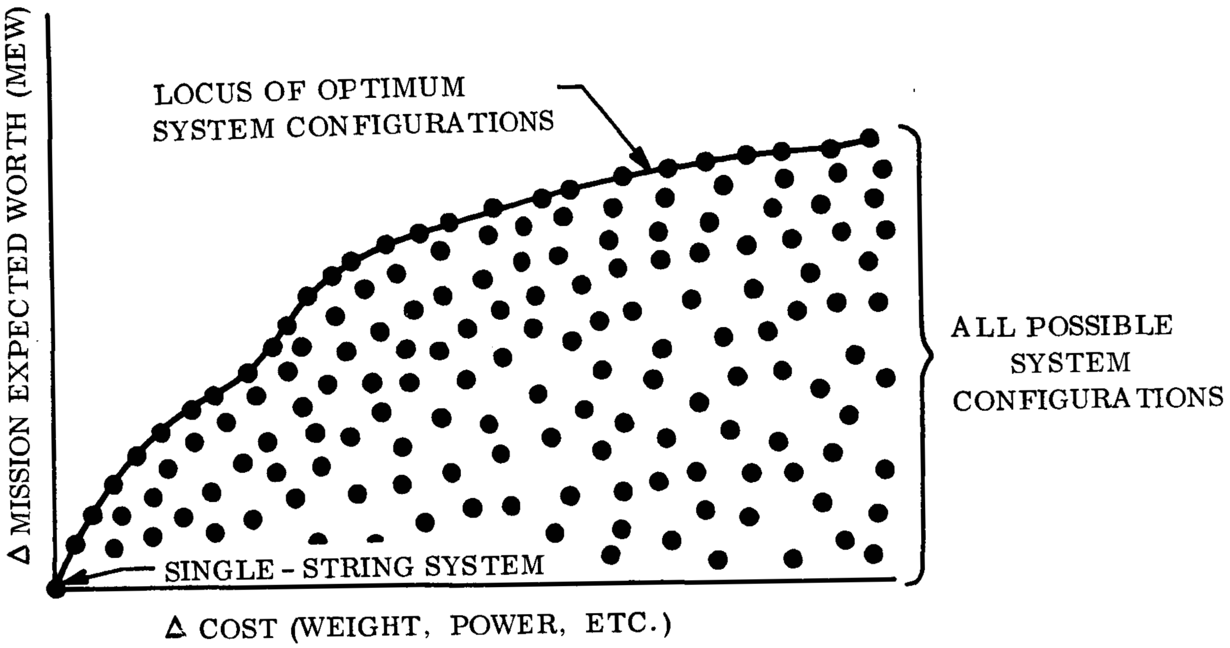


Figure 3-4. The Goal of the Optimization Process

At the about the 45th pound of weight, the single-string radio and command group is replaced by an alternate very similar to the GE Task B design, while the rest of the spacecraft reverts to single string. The configurations for the next 40 pounds essentially duplicate the pattern prior to the addition of radio and command redundancy.

Beyond the first 80 pounds of weight differential, the expected worth of the curve of Figure 3-5 levels off to a maximum of about 67 percent. This asymptotic behavior is due, not to any inherent limitations in the spacecraft system, but rather due to the limitations on the reliabilities of the redundant alternates in certain key spacecraft hardware groups. Postulation of more reliable redundant alternatives in these key groups would, in effect, push the knee of the curve to the right and upward.

A correlation with the classical system reliability figure of merit was performed by placing all of the mission worth on system success, defined for the 1973 mission as the completion

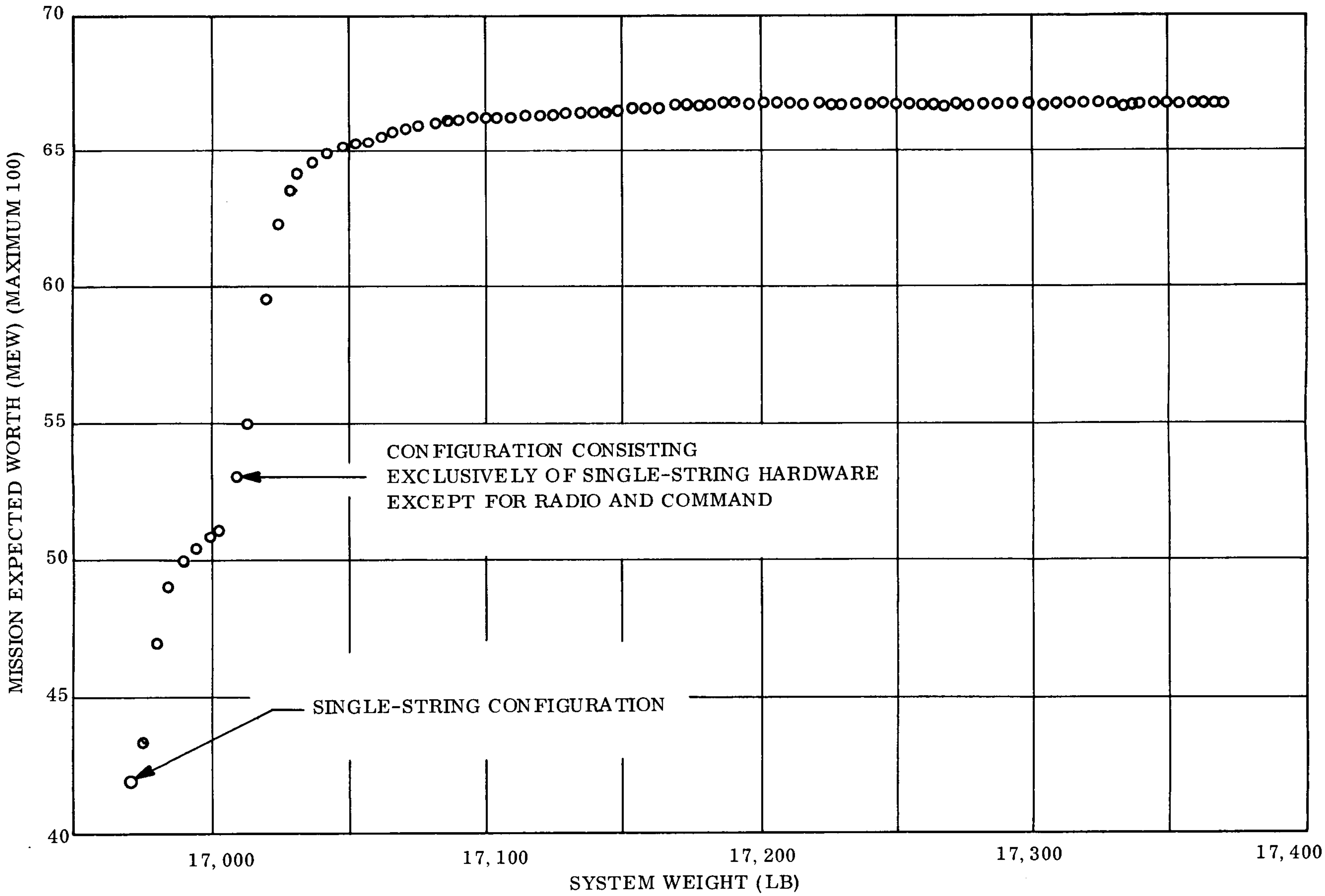


Figure 3-5. Typical System Redundancy Results

of 1 month of orbital operation about the planet. By this definition, the reliability of the single-string spacecraft system was calculated at about 0.38; with the proposed redundant alternates, the reliability rose asymptotically with weight to about 0.66.

Figure 3-5 depicts the expected worth of a mission consisting of a single planetary vehicle and launch vehicle. Exercises with the nominal mission configuration of two planetary vehicles and one launch vehicle revealed that the expected worth of the nominal mission is about 81 percent of the maximum attainable.

In summary, a computerized tool was developed to permit the system manager to allocate redundancy to a spacecraft system, not on the conventional criterion of maximizing system reliability, but on a broader criterion of maximizing the spacecraft system contribution to the expected worth of the mission. The results have provided considerable insight into the amount and type of redundancy which should be applied to the spacecraft system. Although designed for the Voyager mission and the Voyager spacecraft system, this tool is adaptable to other spacecraft and other system elements in complex mission environments.

For this study, the input definition was generally performed within the framework of the General Electric Phase 1A, Task B design. Since the Task B design is nearly 2 years old at the time of this writing, it would seem appropriate to re-examine the input definition in light of the system design evolution. It is recommended that the mission and system definition input to the tool be remodeled and the optimization system be re-exercised to reflect evolutionary changes in the mission profile and system design since Task B, with special reference to the system design update of Phase 1A, Task D.

3.3 SELECTION OF MISSION CONFIGURATIONS

It is in this area of activity that the principles of decision analysis, an applied extension of statistical decision theory, were employed. Prior to this study, decision analysis had been applied primarily to industrial and business decisions. The decision analysis of space projects is a significantly new application.

To develop the application of decision analysis to the Voyager mission configuration problem, a two-phase program was adopted. The first phase (pilot phase) consisted of defining a simplified version of the Voyager project. This smaller problem allowed easier development of the modeling approach, insight into the level of detail required in structuring the inputs to the decision, and a mechanism for discussion of the model and the usefulness of its results.

Based upon the promising results of the pilot phase, a more comprehensive model was developed. Five potential capsule systems were defined:

a. A small, nonsurvivable atmospheric probe capable of returning entry data and atmospheric profiles.

b. A small, nonsurvivable probe capable of atmospheric and entry experiments and descent television.

c. A soft lander, of medium size, capable of atmospheric and entry experiments, optional descent television, surface television, and simple surface experiments.

d. A large soft lander, capable of entry and atmospheric measurements, optional descent television, sophisticated life detection experiments, surface television, and some surface experiments. It was assumed that this lander could not be made available in time for the first Voyager launch in 1973, so a fifth capsule type, available in 1973, was proposed.

e. A large soft lander, identical to d above, except that the payload devoted to biological experiments is replaced by a complete set of sophisticated, nonbiological, surface experiments.

Employing the above 5 capsule types, some 14 potential mission configurations were defined, as summarized in Table 3-1. Additionally, some 56 project outcomes were hypothesized, consisting of all combinations of four outcomes of the orbiting spacecraft and 14 outcomes from the capsule.

The heart of decision analysis is the decision tree. The decision tree contains two types of nodes and two types of branches. Emanating from decision nodes are alternative branches, each branch representing one of the configurations available for selection at that point in the project. Chance nodes are followed by outcome branches, one branch for each outcome

Table 3-1. Potential Mission Configurations

Configuration No.	Launch Vehicle		Spacecraft		Capsules (See Text)			Years of Availability
	No.	Type	No. per LV	Type	No. per SC	First Capsule Type	Second Capsule Type	
1	STOP THE PROJECT							1971-1981
2	SKIP THE MISSION OPPORTUNITY							1971-1981
3	2	Atlas-Centaur	1	Mariner '71 Flyby	1	No. 1-Direct Entry		1971
4	1	SIB	2	Voyager Orbiter	0			1973
5	2	SIB	1	Voyager Orbiter	0			1973
6	1	SV	2	Voyager Orbiter	0			1973
7	1	SV	2	Voyager Orbiter	1	No. 1-Orbital Entry		1973-1975
8	1	SV	2	Voyager Orbiter	1	No. 2-Orbital Entry		1973-1975
9	1	SV	2	Voyager Orbiter	1	No. 3-Orbital Entry		1973-1981
10	1	SV	2	Voyager Orbiter	1	No. 5-Orbital Entry		1973-1981
11	1	SV	2	Voyager Orbiter	1	No. 4-Orbital Entry		1975-1981
12	1	SV	2	Voyager Orbiter	2	No. 1-Direct Entry	No. 1-Orbital Entry	1973-1975
13	1	SV	2	Voyager Orbiter	2	No. 1-Direct Entry	No. 2-Orbital Entry	1973-1975
14	1	SV	2	Voyager Orbiter	2	No. 1-Direct Entry	No. 3-Orbital Entry	1973-1977

that may be achieved at that point in the project. Probabilities of occurrence and values are assigned to each outcome. Costs are assigned to each decision alternative. The full-scale decision tree contained some 3200 nodes and 22,000 branches.

To generate the probabilities of the 14 potential mission configurations achieving all combinations of the four orbiter outcomes and 14 capsule outcomes during any year of opportunity and from any current level of project achievement, a comprehensive probability model was developed.

The method for determining the cost of each alternative involved estimating a baseline cost for each potential mission configuration and then modifying it as a function of the year of opportunity and prior mission configuration development history. The cost model results were found to be in general agreement with known Voyager budgetary planning for those configuration sequences which have been considered to date for Voyager.

A value model was developed to provide a medium for encoding the value preferences of the decision maker and converting these preferences into value assignments for orbiter and capsule outcomes. For a set of subjectively assigned nominal values which were employed in numerous exercises, some 30 percent of the project value is provided by the orbiter, suggesting that the spacecraft is more than just a bus to carry the capsule to the planet.

The full-scale decision model and the three submodels for probabilities, values, and costs were programmed within the framework of a versatile program system. Exercising the programs with nominal input data, some 20 project policies were found to be dominant as a function of project cost. No attempt was made to examine the sensitivity of the resulting policies to variations from nominal probabilities, values, and costs. In this respect, the results and their implications must be considered preliminary in nature. Nevertheless, several trends seem to be strong enough to draw a few general conclusions.

First, for nominal costs, probabilities, and (particularly) values, an ambitious unmanned Mars project seems in order. In 1971 a Mariner flyby with an atmospheric probe (configuration 3) is suggested. In 1973, Voyager with large surface landers with extensive television and physical experiment capability (configuration 10) is recommended. From 1975 through 1981, landers of the VBL class are desirable (configuration 11). This project sequence has an expected project value of 69 percent of maximum.

This project profile is not too different from the current (July, 1967) NASA plan for Mariner/ Voyager Mars. Differences exist principally in the 1973 mission, where a medium lander (like configuration 9) is now planned, and in the deletion of the 1981 mission. Interestingly enough, if a more conservative approach relative to risk is taken, and if good mission outcomes are valued higher if they occur earlier in the program, then policies even more similar to the unmanned Mars project profile result. The configuration 10 choice in 1973 reverts to 9, and the last mission of the series can be earlier than 1981, provided that satisfactory outcomes have been realized.

Inference can be drawn from the results that the use of single nonsurvivable capsules with Voyager spacecraft is usually not desirable. Dual capsule configurations (per spacecraft) with nonsurviving capsules, where the first capsule enters prior to orbit insertion and the second after achieving orbit are occasionally recommended in tradeoff policies.

For Saturn V launch vehicle reliabilities (through interplanetary injection) above 0.8, additional exercises with the decision model showed that multiple Saturn V vehicles per opportunity cannot be justified economically. Since current SV reliability estimates are usually greater than or equal to this figure, it would appear that a project profile employing single Saturn V's per opportunity is the more desirable.

It must be emphasized that the results obtained thus far are from the initial exercising of a newly developed complex program. As mentioned above, the sensitivity of the optimal policies to input apportionments of value and probability has not been examined. Thus, the early results must be assessed with these qualifications in mind. Furthermore, the "optimal" policies resulting from this technique can be evaluated only after a fairly thorough understanding of the many inputs to, and the structuring of, the programs which generated these policies.

The objective of this effort was to develop a logical procedure for selecting Voyager mission configurations which reflected technical feasibility, NASA project objectives, and the economic environment of the project and which was dynamically adaptive to project history. The natural continuation of this work is to further exercise and develop this decision-making tool. In order to carry this out most profitably, it is important that NASA Voyager personnel in responsible decision-making positions and their staff become familiar with the potential applications of this model. In this way, important feedback can be provided for further development, and equally important, understanding can be developed in NASA so that this decision tool can ultimately be fully utilized.

SECTION 4

DATA MANAGEMENT AND CONTROL TASK

4.1 SUMMARY

The broad objective of the Study was to support NASA/JPL in the development of the Voyager Data Management System.

To accomplish this objective, a core Data Management Study Team was organized consisting of Project Management/Data Management/Information System personnel. Throughout the Study, this team was actively supported by the Valley Forge Voyager Spacecraft functional organization (which identified information flow and requirements) and by Division management personnel and consultants (who reviewed operational aspects of the System).

During the Study, 34 technical documents were issued which described the Data Management System, identified contractor data requirements and information flow, and analyzed functions of the Data Management System. Included in these documents is a series of studies relating to project management control of contractor activities.

The primary results of these studies are that:

a. All the functions critical to providing Voyager Management and operating personnel with the... "minimum essential data to do their jobs" have been identified and selected functions were both studied and tested.

b. An extensive field test to determine how a typical spacecraft development organization would establish Voyager data requirements-- and what these requirements would be at the Contractor level was successfully conducted within the General Electric Valley Forge Voyager Organization (utilizing the participation of 30 senior technical personnel).

4.2 OBJECTIVES

The primary objectives of the Data Management Study were to support NASA/JPL in the development of the Voyager Data Management System by:

a. Delineating the basic functions of the Voyager Data Management System throughout its Operational phases;

b. Developing Voyager Contractor Level Information Flow and Data Requirements for the Design and Acquisition Phases (C and D);

c. Analyzing Contractor implementation of selected functions of the Voyager Data Management System;

d. Determining Data Requirements and reporting systems for project management control of contractor activities.

4.3 APPROACH

The basic approach utilized during all phases of the study--to assure the practicability of the Data Management System--was to augment the core Study Team with Voyager Spacecraft Functional personnel to develop data flow and requirements, with Information System personnel to develop methods and media of information flow, and with Division Management Practices personnel and consultants to review operational aspects of the program as illustrated in Figure 4-1.

This Study was organized into 4 basic phases as described below.

Phase (1) - The Data Management System Study - resulted in the preparation of a series of system flow diagrams which delineated the Data Management System, a Glossary which identified Voyager Data Management terms and a Data Standards Study.

Phase (2) - Contractor Data Requirements Study - resulted in the definition of Voyager Phase C and D Data Requirements by Functional Managers at the Valley Forge Space Technology Center. This activity included the preparation of Data Item User Matrices, 345 Data Requirement Descriptions, User Flow Diagrams, Document Relationship Trees, Frequency and Phasing Charts, and a Subcontractor Data Item Study.

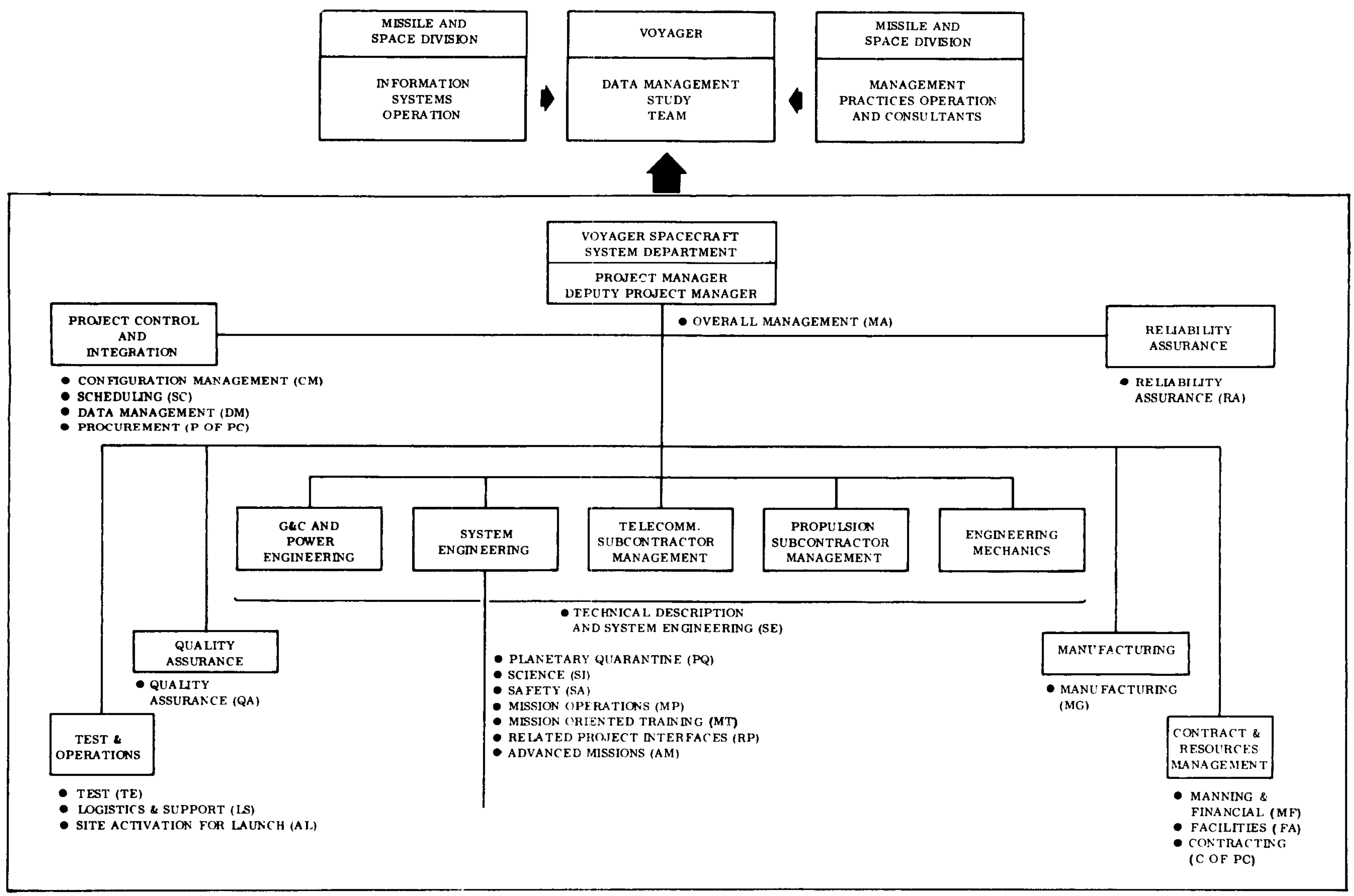


Figure 4-1. Data Management Study Team

Phase (3) - Contractor Implementation Studies - resulted in a series of studies which analyzed how a Voyager Contractor would implement functions of the Data Management Program, and included the preparation of an Information System Equipment Handbook, Microfilm Compendium, Automatic Data Processing Plan, Indentured Numbering System Study and Data Cost Study.

Phase (4) - The Identification of Contractor Management Information Studies - resulted in the generation of Data Requirements for Project Management Control, an Automatic Data Processing Plan for Project Control, a Technical Performance Monitoring Study, and a Project Control Room Study.

4.4 DATA MANAGEMENT SYSTEM STUDY - PHASE 1

Working closely with JPL, a series of Flow Diagrams delineating the proposed Voyager Data Management System was prepared during the initial phase of the study.

These Diagrams apply to the management of formal data, both "hard-copy" and that maintained by means of Automatic Data Processing Equipment. (Formal data, which is generally utilized by multiple project elements, is prepared and processed in accordance with project - established requirements, procedures and controls; informal data, in contrast, is generally utilized within--and managed internal to--a particular Project element.)

Additionally, a "Glossary" defining Voyager Data Management Terms and a report identifying existing Specifications and Standards relating to the preparation, submittal, and review of data were prepared.

Key Features of the Voyager Data Management System (identified in the flow diagrams) which are vital to its successful project-wide application include:

a. User establishes data requirements; and conversely, data responses are prepared only in response to these requirements.

b. Data Review Boards and Data Management Office activities exist within each organization level.

c. Emphasis is on communication of information utilizing both "hard copy" documentation and Automatic Data Processing Equipment, as appropriate.

d. Recognition is made of Specialized User Systems and integration of these into the overall Data System.

e. Data Management starts during the establishment of requirements phase and provides tracking capability throughout the Program.

f. Traceability is provided of data items (and certain data elements) as to generator, related hardware item, contract, etc., throughout the Program (including other Project applications).

4.5 CONTRACTOR DATA REQUIREMENTS STUDY - PHASE II

The Contractor Data Requirements Study had the dual objective of:

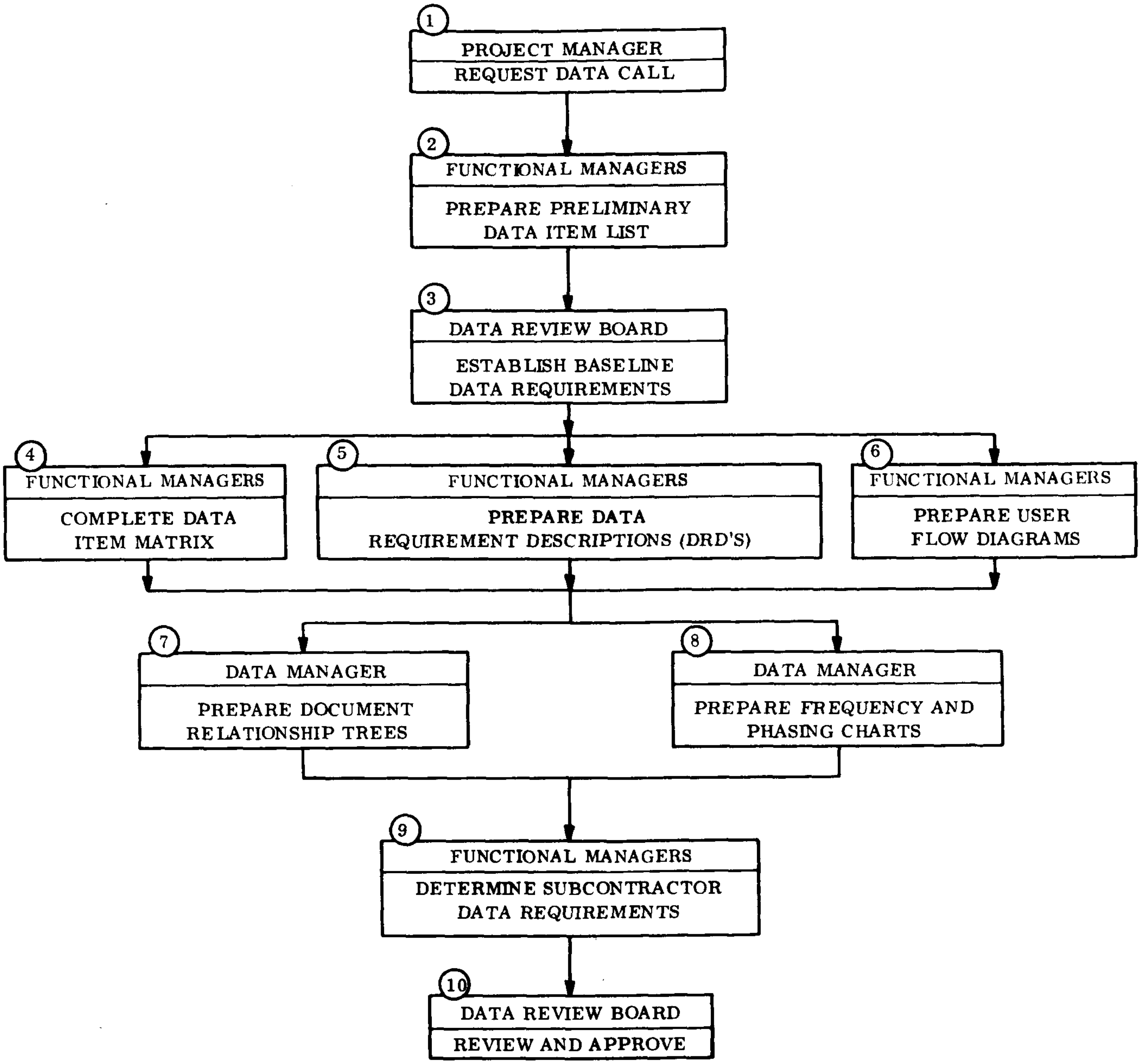
a. Developing an approach (including tools) for conducting a Contractor-level data call, and

b. Identifying the data which a typical Spacecraft Contractor would manage during Phases C and D of the Voyager Program.

To accomplish this, an actual data call was conducted within the General Electric Voyager Project organization at the Valley Forge Space Technology Center. (This organization contained approximately 120 senior professional personnel at the time of the data call.)

Although this data call was a hybrid in that it developed a complete Contractor data base in one cycle (rather than responding to System Office imposed data requirements as would be normal during actual program implementation) the tools and approach developed -- as well as the data base -- are considered basically applicable for use on Voyager Phases C and D.

The 10 steps comprising the data call activity are shown schematically in Figure 4-2. A brief description of this activity follows.



NOTE: CONTINUOUS INTEGRATION, REVIEW, AND SUPPORT BY THE CONTRACTOR'S DATA MANAGEMENT OFFICE (NOT SHOWN) IS REQUIRED THROUGHOUT THE DATA CALL.

Figure 4-2. Key Activities--Contractor Data Call

The Project Manager initiated the data call, established and chaired a Data Review Board, and assigned responsibility for each data Functional Management Category to the senior member of his staff responsible for that area of activity. The senior staff members (functional managers) then generated preliminary data item lists, which were submitted to the Data Review Board.

The Data Review Board integrated these inputs, considered the justification for each item, eliminated overlapping requirements, and , in some cases, combined two or more data items. The output of the review was a baseline data requirements list.

The functional managers then generated: (1) a Data Item Matrix, relating data item "approvers, reviewers, and users" (see Figure 4-3); (2) data requirement descriptions covering the required contents, the use, and relationship of each data item to other items; and (3) user flow diagrams relating the data item to project activity, e.g., design.

The functional managers responsible for planned major subcontracts reviewed the data requirements list and identified those which would be applicable to subcontractors. Meetings were held with potential subcontractors, and their participation in this activity was obtained.

The Data Manager prepared the document relationship tree and the data frequency and phasing charts. The charts identify the timing of data items relative to the Voyager Project Schedule.

4.6 CONTRACTOR IMPLEMENTATION STUDIES - PHASE III

To determine how the Contractor would utilize the Data Management System (delineated in Phase I) to process the data developed in Phase II, studies were performed and documented in the reports described below.

An Information System Equipment Handbook, describing performance characteristics and cost trends of equipment such as computers, input/output equipment, data storage devices, reproduction and copying equipment, etc.

DATA ITEM LIST/USER MATRIX

DATA ITEM NUMBER	DATA ITEM (1 of 7) TECHNICAL DESCRIPTION AND SYSTEM ENGINEERING	DESCRIPTION	APPLICABILITY TO FUNCTIONAL USERS AT CONTRACTOR LEVEL																				APPLICABILITY AT SUBCONTRACTOR/VENDOR/SUPPLIER LEVELS						APPLICABILITY TO PROJECT BOARDS										
			SE	PO	MG	CM	QA	TE	RA	LS	MA	SC	PC	MF	MP	DM	FA	SA	AL	SI	RP	AM	PRINCIPAL SUB-CONTRACTORS	MAJOR SUB-CONTRACTORS	KEY SUBCON AND VENDORS	OTHER VENDORS	KEY SUPPLIERS	KEY SUBSUPPLIERS	CONFIGURATION CONTROL	CONFIGURATION MANAGEMENT	DATA REVIEW	DESIGN REVIEW	FAILURE REVIEW	INTEGRATED SAFETY	INTEGRATED TEST	MAKE OR BUY	MATERIAL REVIEW	SOURCE EVALUATION	SOURCE SELECTION
	PLANS																																						
SE-001	Plan, System Development	Defines the system analysis, design, test, integration and control activities in developing the spacecraft system and the relationship between subsystem development plans and integration and control plans.	R	U	U	U	U	U	R	U	PM	U	U	-	U	-	U	U	U	U	-	U	NA	NA	NA	NA	NA	NA	U	U	-	U	-	-	U	-	-	-	-
SE-002	Plan, Interface Integration	Defines the analysis, integration and control activities required to interface the spacecraft with the capsule, shroud, etc.	R	U	U	U	U	U	U	U	PM	U	U	-	U	-	U	U	U	U	-	U	NA	NA	NA	NA	NA	NA	U	U	-	U	-	-	U	-	-	-	-
SE-003	Plan, Subsystem Development	Defines the analyses, tests and design to be performed in developing the subsystem and its components. Each plan references or incorporates the pertinent provisions of the various integration and control plans.	R	U	U	U	U	U	R	U	PM	U	U	-	-	-	U	U	-	U	-	U	S	(S)	(S)	NA	(S)	NA	U	U	-	U	-	-	U	-	-	-	-
SE-004	Plan, Magnetic Cleanliness Control	Covers the approach for implementing appropriate general engineering specifications, which, in turn, are incorporated into hardware specifications.	A	-	R	-	U	U	U	-	U	-	-	-	-	-	U	U	-	U	-	U	C/S	C/S	C/S	NA	C/S	NA	U	U	-	U	-	-	U	-	-	-	-
SE-005	PLAN, ELECTROMAGNETIC COMPATIBILITY CONTROL	COVERS THE APPROACH FOR IMPLEMENTING APPROPRIATE GENERAL ENGINEERING SPECIFICATIONS, WHICH, IN TURN, ARE INCORPORATED INTO HARDWARE SPECIFICATIONS.	A	-	R	-	U	U	U	-	U	-	-	-	-	-	U	U	-	U	-	-	C/S	C/S	C/S	NA	C/S	NA	U	U	-	U	-	-	U	-	-	-	-
SE-006	Plan, Cleanliness Control	Covers the approach for implementing appropriate general engineering specifications, which, in turn, are incorporated into hardware specifications.	A	R	R	-	R	U	U	U	U	-	-	-	-	-	U	U	-	U	-	U	C/S	C/S	(C/S)	NA	C/S	C/S	U	U	-	U	-	-	U	-	-	-	-
SE-007	Plan, Mass Properties Control	Covers the approach for implementing appropriate general engineering specifications, which, in turn, are incorporated into hardware specifications.	A	-	U	-	U	U	-	-	U	-	-	-	-	-	-	U	-	U	-	U	(S)	NA	NA	NA	(S)	NA	U	U	-	U	-	-	U	-	-	-	-
	SPECIFICATIONS																																						
SE-008	Specification, System Performance/Design Requirements	Defines system configuration and performance, allocates performance and design requirements to each subsystem including OSE, and defines system constraints and interfaces.	R	U	U	U	U	U	R	U	PM	U	U	-	U	-	U	U	U	U	-	U	NA	NA	NA	NA	NA	NA	U	U	-	U	-	-	U	-	-	-	-
SE-009	Specification, Subsystem Design Requirements	Defines the subsystem configuration, allocates performance and design requirements to each component, including OSE. Not used for procurement.	A	U	U	U	U	U	R	-	R	-	-	-	-	-	U	U	U	U	-	U	(C)	(C)	NA	NA	(C)	NA	U	U	-	U	-	-	U	-	-	-	-
SE-010	Contract End Item (CEI) Detail Specification (Prime Equipment) Part I (Requirements)	Specifies performance and design requirements of a contract end item (flight hardware or operational support equipment).	A	-	U	U	R	U	R	U	R	-	-	-	-	-	U	U	U	U	-	U	C/S	C/S	(C)	NA	(C)	NA	U	U	-	U	-	-	U	-	U	-	-
SE-011	Contract End Item (CEI) Detail Specification (Prime Equipment) Part II	Specifies configuration details for procurement of a contract end item (including acceptance testing).	A	-	U	U	R	U	R	U	R	-	U	-	U	-	U	U	U	U	-	U	S	S	(S)	NA	(S)	NA	U	U	-	U	-	-	U	-	U	-	-

* KEY INFORMAL DATA

U - USE   R - REVIEW AUTHORITY   A - APPROVAL AUTHORITY   PM - PROJECT MANAGER APPROVAL   C - PREPARED BY CONTRACTOR   S - PREPARED BY SUBCONTRACTOR/VENDOR/SUPPLIER   - OPTIONAL APPLICATION   NA - NOT APPLICABLE

Figure 4-3. Excerpt from Data Item List/User Matrix-Contractor Level

An Automatic Data Processing Plan, which recommends and describes an ADP system which provides both central coordination of, and flexibility within, the ADP subsystem for: Project Control; Engineering Development; Configuration Management; Purchasing and Materials Control, Fabrication, Assembly and Test; Test and Environmental History Accounting; and Document Management.

A Data Item Indentured Numbering System report, which describes a numbering system for relating hardware and software items, and which permits retrieval of data concerning them. The system identifies each data item as to functional category, imposer's code and responder's code, and provides for retrieval of the data item when requested against such subjects as work breakdown structure, type and kind of data item, and hardware identification.

4.7 CONTRACTOR MANAGEMENT INFORMATION STUDY - PHASE IV

This phase was concerned with data requirements, data handling and with data display for Management Project Control.

4.7.1 FORMAL DATA REQUIREMENTS:

These include the Data Item Matrices, the Data Requirement Descriptions, and the User Flow Diagrams developed for the Overall Management (MA); Scheduling (SC); Manning and Financial (MF); and Procurement and Contracting (PC) Categories during the Phase 2 (Contractor Data Requirements) portion of the Study.

4.7.2 AUTOMATIC DATA PROCESSING SYSTEM FOR PROJECT CONTROL

Description of an Automatic Data Processing System to handle Management Data is included in the Automatic Data Processing Plan and consists of the following subsystems:

a. Cost and Schedule Subsystem (Project Control uses this subsystem to determine project costs, budgets, schedules, resource utilization, and projections.)

b. Operations Analysis Subsystem (This is a real-time subsystem which provides Project Control with the ability to gather data across all project activities, to ascertain its significance to project performance, and to initiate action based on the analysis of the data.)

c. Contact/Action Item Subsystem (This subsystem provides a means of collecting total project external contact information and action item requirements and status.)

d. Facilities Subsystem (This subsystem provides Project level information relative to space requirements, equipment needs and associated activities.)

e. Subcontract Procurement Subsystem (Subcontract Management uses this subsystem to help control subcontractor activity.)

f. Resource Support Subsystem (This subsystem generates administrative support data such as personnel lists, telephone directories, organization charts, and similar items.)

g. Project Control Room Subsystem (The Project Control Room Subsystem provides uniform data for base line analyses, displays, and reports for direct use by Project Management, project control, functional management and customer personnel.)

4.7.3 TECHNICAL PERFORMANCE MONITORING STUDY

This report presents an approach to provide "monitoring" of Technical Performance for the Voyager Project. In the context of this Study, "monitoring" is considered to involve analysis by Project Management of available data to determine: (1) conformance with established technical requirements, and (2) required action to solve indicated problems and prevent potential problems.

The approach recommended for Voyager is to incorporate essentially three complementary viewpoints to establish a realistic Technical Performance Monitoring System.

The viewpoint of technical management, reinforced as required by external experts, is obtained through the Design Review mechanism at discrete points.

The viewpoint of the Cognizant (or Lead) Engineer is obtained through the Technical Adequacy Report at regular (biweekly) periods. This Report provides for assessment of status and trend with respect to selected system and subsystem parameters.

The viewpoint of the performing functional organizations is obtained through their regular assessment of technical status by reporting milestones completed (or percent of completion) into a Schedule-Cost Coupling System.

During the Study, effort was concentrated on the development of a Technical Adequacy Reporting Technique, defining the data requirements and flow for Design Reviews, and identifying means to enhance the value of Schedule-Cost Coupling Systems in highlighting technical problems.

4.7.4 PROJECT CONTROL ROOM STUDY

This study identified key characterisitcs of a successful Project Control Room. These included the capability of (1) highlighting "the exceptions," (2) identifying the information from which conclusions were drawn, (3) having available the broad-based background information that makes the exceptions understandable and permits the Program Manager to be fully informed, and (4) assessing the impact of changes.

The Work Breakdown Structure was selected as the primary tool for ensuring the continuity and traceability of Project Control Room data. Typical displays were developed for Plan/ Status Data for the following types of information:

a. Facilities utilization schedules

b. Major Hardware utilization schedules

c. Summary networks at subtask, task, and project level

d. Detailed schedules of activities by subtask and functional component

e. Summary schedules at subtask, task, and project level

f. Interface event schedules - customer/contractor; contractor/subcontractor

g. Special event schedules, e.g., all Proof Test Model components complete.

h. Total cost curves by work breakdown level and functional level and by standard action.

i. Labor cost curves by work breakdown level and functional level and by standard action.

j. Material cost curves by work breakdown level and functional level and by standard action.

k. Total manpower curves by work breakdown level and functional level.

l. Categorized manpower curves by work breakdown levels and functional level.

m. Technical requirements/status at subtask, task and project levels unresolved problem lists

n. Detailed networks of special and topical subtask activities.

Measurement Data

a. Value curves by work breakdown level (work package, subtask, and task project) and by functional level (component, operation, section, and department).

b. Planned time versus actual time by activity.

c. Planned cost versus actual cost by work package.

Special Data

a. Open action items lists

b. Superseded revisions of all schedule, cost, and manpower plans

c. Work breakdown structure by customer, contractor, and subcontractors

d. Organization interface

e. Overtime control

4.8 FINAL REPORT

Volume V of this Final Report contains summaries and excerpts from the 34 Technical Documents issued during the Study. In addition, a series of 11 Appendixes (A through K) are included which present Contractor Data Packages (Data Item Matrix, Data Requirement Descriptions, User Flow Diagram, Document Relationship Tree, and Frequency and Phasing Charts) for the following Functional Management Categories:

a. Technical Description and System Engineering (SE)

b. Planetary Quarantine (PQ)

c. Manufacturing (MG)

d. Configuration Management (CM)

e. Quality Assurance (QA)

f. Test (TE) and Mission Operations (MP)

g. Reliability Assurance (RA)

h. Logistics and Support (LS)

i. Overall Management (MA); Scheduling (SC); and Manning and

j. Financial (MF)

k. Procurement and Contracting (PC)

l. Data Management (DM)