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SUMMARY REPORT

OCTOBER, 1967

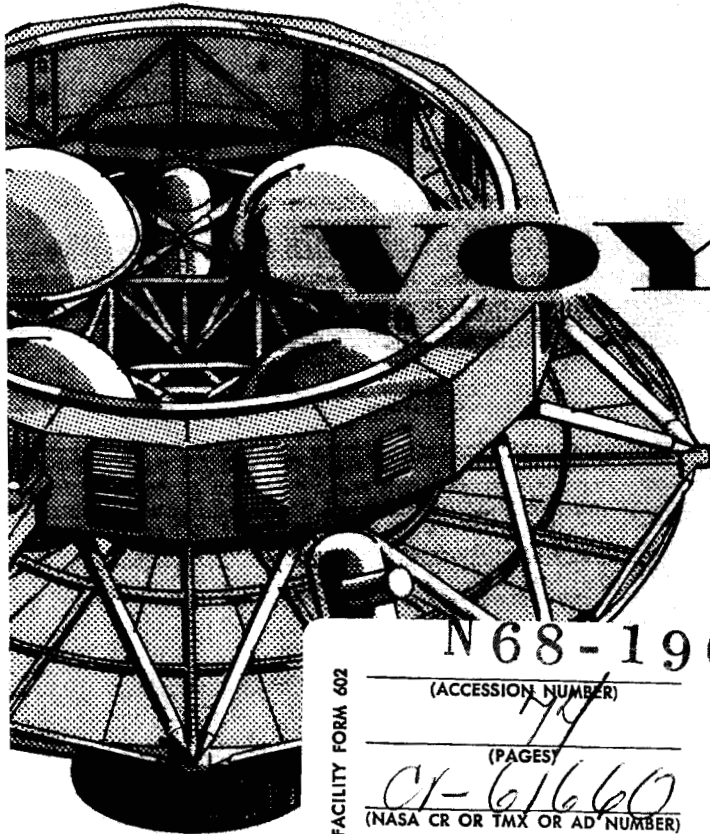
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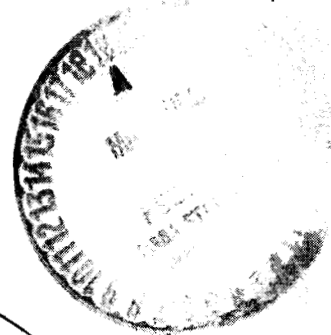
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Prepared for
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

THE **BOEING** COMPANY
SPACE DIVISION
SEATTLE, WASHINGTON

VOYAGER
SPACECRAFT SYSTEM STUDY

66

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FINAL TECHNICAL REPORT

PHASE B, TASK D

VOLUME I
SUMMARY REPORT

D2-115002-1

OCTOBER 1967

Prepared For:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

UNDER
CONTRACT NO. NAS8-22602

THE BOEING COMPANY • SPACE DIVISION • SEATTLE, WASHINGTON

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1.0 INTRODUCTION

This document summarizes the work performed under the George C. Marshall Space Flight Center contract NAS 8-22602, entitled "Voyager Spacecraft System, Phase B, Task D." The work was performed over the period June 16 through October 16, 1967.

The contracted work consisted of engineering studies leading to definition of a Voyager spacecraft system capable of performing the 1973 Mars mission. To ensure flexibility of design, additional analyses were conducted to determine the adaptability of the 1973 spacecraft to perform the 1975-1977-1979 Mars missions. The 1973 flight spacecraft definition was used to identify the operational support equipment, including mission-dependent equipment requirements and the software necessary to satisfactorily conduct the 1973 mission operations. Logistics considerations from point of manufacture through launch operations were identified for the 1973 system. Highlights of the results of the work include:

- The spacecraft system concept requires only minor changes due to updating from 1971 to 1973 requirements.
- A single design for spacecraft structure and propulsion accomplishes all missions in the 1970's.
- Modularization of major elements, such as propulsion, facilitates testing and maintenance requirements.
- A wide variety of 1973 orbits are achievable to fulfill planned scientific objectives.
- Growth capability exists in the 1973 design to accommodate future mission requirements.

The contract also required the completion of five selected engineering tasks that were designed to highlight key areas and lead to specific conclusions and recommendations.

This document presents a summary of the results of the study as well as the significant conclusions and recommendations. The detailed results may be found in the following reports:

<u>Mission/System Requirements and Analyses</u>	Volume II D2-115002-2
<u>Spacecraft Functional Description</u>	Volume III D2-115002-3
<u>Selected Engineering Tasks</u>	Volume IV D2-115002-4

This document summarizes, for each of the three volumes above, the objectives, the results, and significant conclusions and/or recommendations.

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2.0 MISSION/SYSTEM REQUIREMENTS AND ANALYSES

2.1 BACKGROUND AND OBJECTIVES

The Voyager Mars program has been an approved NASA program since 1964. The first mission opportunity has been evaluated for a 1971 Mars mission considering both the Saturn 1B/Centaur and the Saturn V launch vehicle. The decision in late 1965 to defer the first Voyager Mars mission to 1973 was disappointing from the scientific viewpoint of obtaining detailed data surrounding the nature and history of this interesting planet. The ensuing time period has, on the other hand, allowed for incorporation of technology advancements that will make the mission even more scientifically productive. The purpose of this task was to update the 1971 concept to a 1973 system utilizing applicable technology advances.

The development of a new system concept requires an iterative process starting with program objectives and ending in system requirements. First, a set of preliminary requirements is developed from which all the studies, definitions, and tradeoffs can evolve. This preliminary set of requirements stems from the program objectives and is supported by top-level mission trade studies. Once a system concept is identified, supporting system-level analyses and tradeoffs can be conducted that will confirm some of the preliminary requirements, change others, and add to them. The mission/system analysis task leads all other activities and is iterative in nature since detail system and subsystem level studies may determine that one or more of the mission or system-level requirements is not cost-effective to achieve. The preliminary set of requirements is then updated to reflect the best compromise of program objectives and state-of-the-art capability.

The Voyager Mars 1971 mission objectives were developed to a great depth of detail, leading to a definitive set of mission and system requirements. Since the first Voyager missions are to Mars, it is important to use all applicable data developed to date regardless of the timing of the first mission. This is the essence of the objectives of the mission and system requirements and analysis task. These objectives are summarized below:

- 1) Develop a preliminary set of mission and system requirements for the Voyager Mars 1973 mission.
- 2) Develop the mission and system requirements for subsequent Mars missions (1975-1977-1979).
- 3) Establish the total 1973 mission and system requirements considering the adaptability of the 1973 spacecraft system in meeting the subsequent mission requirements (1975-1977-1979). The results of the science payload evolution task were used to establish science payload requirements for all missions.
- 4) Conduct mission and system-level analyses and tradeoffs to support the establishment of requirements for all mission opportunities.

The approach to meeting the first objective is summarized in Figure 1. The first step was to examine the established requirements for the 1971 Voyager Mars

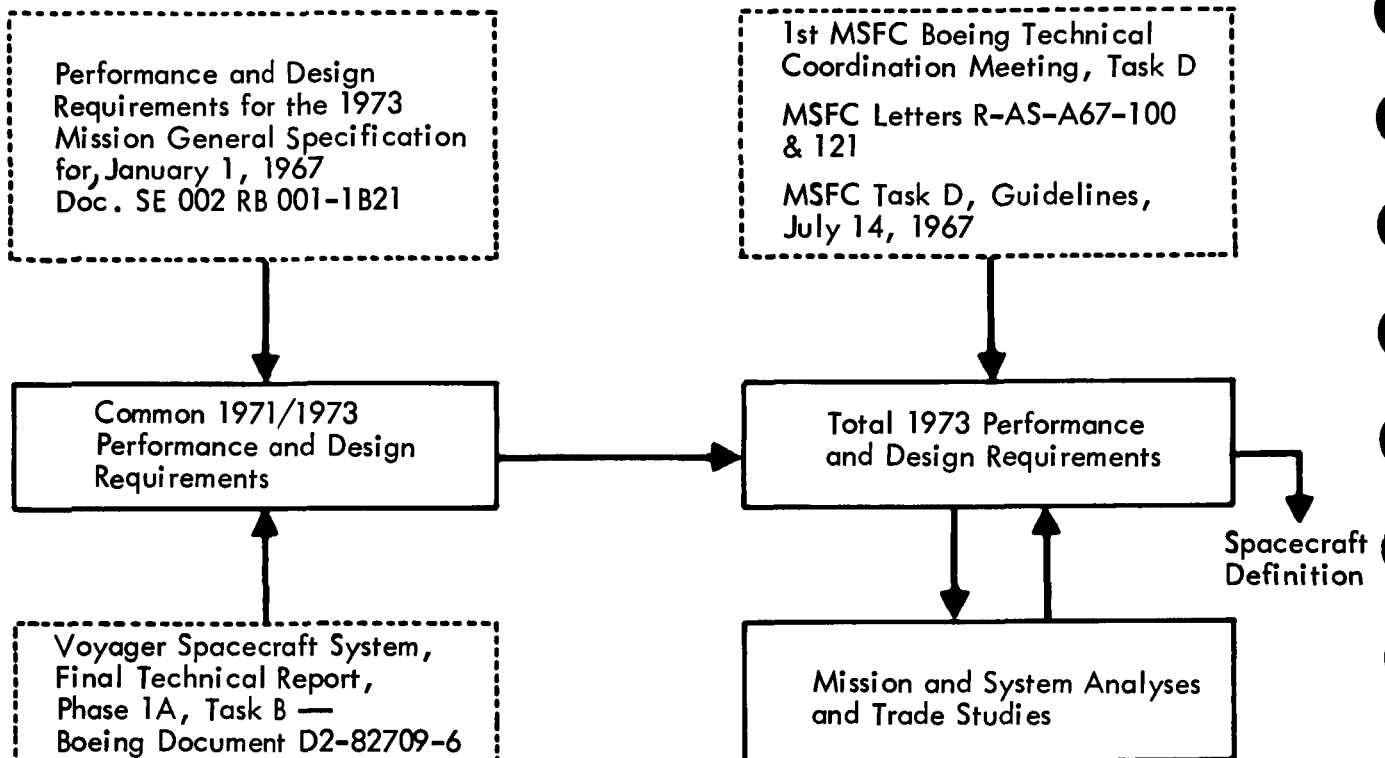


Figure 1: MISSION/SYSTEM REQUIREMENTS AND ANALYSES

mission to determine which of these were common to a 1973 mission. Source data to accomplish this task included the final report from the Phase 1A, Task B contract and the NASA mission specification released in January 1967. This review would eliminate the necessity for analyses in all areas where common requirements could be identified. The set of 1971-1973 common requirements were to be augmented with planning data received from MSFC and by contract tasks to establish the total set of preliminary 1973 performance and design requirements.

The second objective (1975-1977-1979 requirements) was met in a manner similar to the first and is outlined in Figure 2. The source data was the same NASA specification as before but augmented by the preliminary set of 1973 requirements described above. Supporting mission/system analyses and tradeoffs were conducted to support accomplishing this objective.

The third objective was accomplished by evaluating the 1973 spacecraft system toward meeting the requirements of subsequent missions. This process is also depicted in Figure 2. One other feature becomes evident from an examination of Figure 2--the spacecraft system concept defined to meet the preliminary set of 1973 requirements may not be flexible or readily adaptable to meet the subsequent mission requirements. This necessitates tradeoffs that may modify the preliminary 1973 requirements and/or the spacecraft system concept to optimize the program definition. This is indicated in Figure 2 by the block entitled "Modify 1973 Performance and Design Requirements."

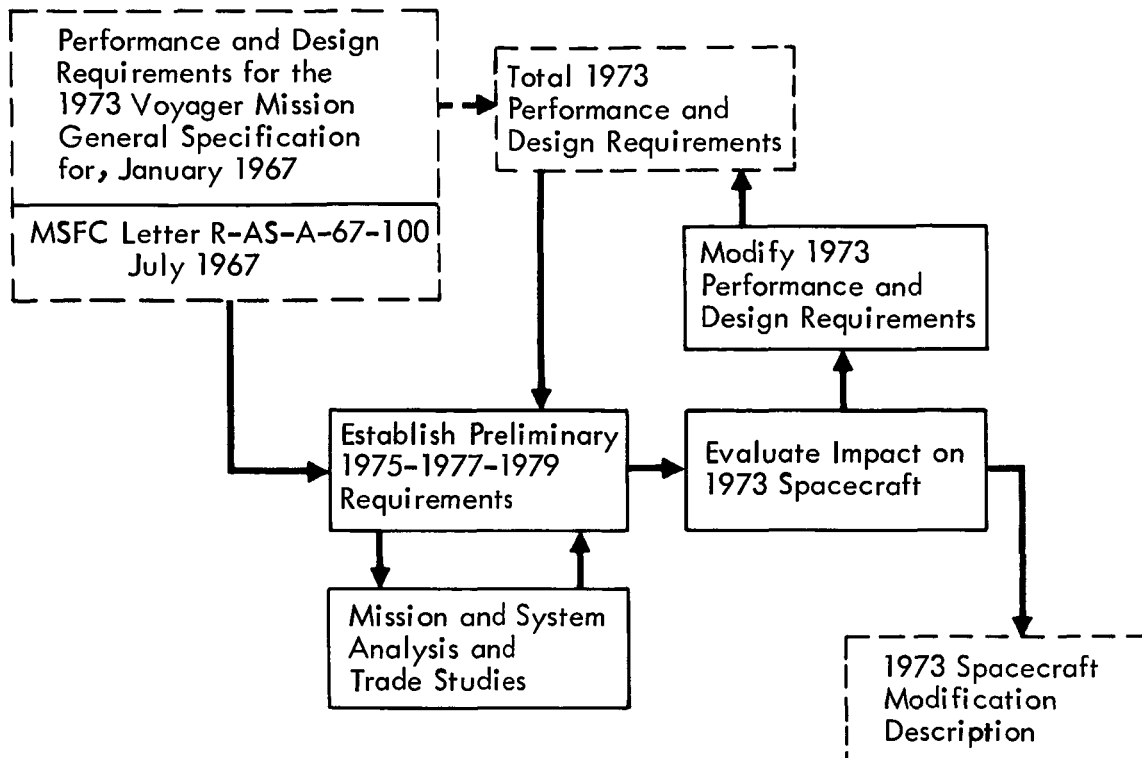


Figure 2: ADAPTABILITY TO THE 1975-1977-1979 MISSIONS

Once the preliminary set of spacecraft requirements are established and the spacecraft concept selected, it is possible to specify requirements for the operational support equipment (OSE) to support the mission. This objective was met by establishing a set of spacecraft system functional flow diagrams from point of final assembly, to processing at Kennedy Space Center, through launch and mission operations. The same set of flows was used to identify requirements for OSE, mission-dependent equipment, and software, and to identify logistics considerations.

The last objective dealing with mission and system-level analyses and tradeoffs supports the establishment of performance and design requirements for all mission opportunities in the 1970's by providing parametric data from which optimum points may be selected. These analyses, both for the 1973 spacecraft system and its adaptability for subsequent missions, are key elements of the system engineering process.

2.2 SUMMARY OF RESULTS

The mission/system requirements and analysis task was examined on the basis of a breakdown structure as shown in Figure 3. The 1971-1973 requirements comparison task, the total 1973 requirements and the requirements for subsequent missions were all keyed to this breakdown structure.

After conducting the 1971, 1973 requirements comparison, incorporating the MSFC guidelines and conducting mission and system analyses, the total 1973 performance and design requirements were tabulated, a summary of which is shown in Figure 4. Significant changes from the the 1971 requirements were in the areas of flight capsule weight and mission duration.

- Mission Definition and Profile
- Mission Requirements — Operational Life
- Trajectory Parameters — Interplanetary Transfer
- Trajectory Parameters — Operations Planning
- Trajectory Parameters — Mars Orbit
- Target Weight Allocations
- General Design Requirements
- Propulsion Requirements
- Capsule Support Requirements
- Data Return Requirements
- General Capsule Requirements
- Electrical Power Requirements
- Reliability Allocations
- Spacecraft Stabilization Requirements
- Science Payload Requirements

Figure 3: BREAKDOWN STRUCTURE
Performance and Design Requirements

An example of the data generated for requirements for the subsequent missions is shown in Figure 5, which shows interplanetary trajectory parameters for the various missions. Type II trajectories were selected in 1975 and 1977 because Type I trajectories for these missions do not result in the required payloads in Mars orbit. This type of data is used as inputs for the mission and system analysis task discussed later and also to establish boundaries for mission planning.

Two of the key mission constraints from Figure 5 are Earth-Mars distance and Mars-Sun distance. The Earth-Mars distance versus time for each of the mission opportunities is shown in Figure 6. The variation in arrival periods is indicative of the broad scope of mission operations planning that must occur. The longest communication distance at encounter is about the same for all opportunities, and all missions will experience the maximum Earth-Mars distance some time during spacecraft orbital operations. The other significant constraint is the Sun-Mars distance relationship shown in Figure 7. The distance in astronomical units (1 A.U. = 150 million km) is shown versus time for encounter and subsequent orbital operations. This figure shows that the 1973 arrival period occurs prior to Mars aphelion which means that the availability of solar energy for generation of electric power drops off during the early part of orbital operations. On the other hand, the 1975, 1977, and 1979 encounters occur after Mars aphelion which indicates an increase in solar energy available for early orbital operations.

CATEGORY	KEY REQUIREMENTS
MISSION DEFINITION AND PROFILE	● Saturn V Launch Vehicle ● Two Identical P/V's (Minor Science Payload Differences Optional)
MISSION REQUIREMENTS — Operational Life	● Six Months Life in Orbit
TRAJECTORY PARAMETERS — Interplanetary Transfer	● Type I Transfers ● Minimum Launch Period: 20 Days ● Maximum C3: 28.8 KM²/Sec² ● Maximum VHP: 3.4 KM/Sec ● Maximum DLA: + 360
TRAJECTORY PARAMETERS — Operations Planning	● Constant Arrival Date — April 2, 1974 ● Capsule Canister Separation After Orbit Insertion ● SA 506 Payload Performance — 5000# Project Contingency
TRAJECTORY PARAMETERS — Mars Orbit	● 500 KM x 18,500 KM — 45° Inclination ● 150 Initial Periaapsis Illumination — Near Evening Terminator ● Orbit Lifetime ≥ 11 Years ● No Sun Occultation for First 30 Days — Not to Exceed 1.5 Hours or 12.5% of Orbit Period Thereafter
TARGET WEIGHT ALLOCATIONS (CASE B)/(CASE A)	● Total Planetary Vehicle: 23,000#/21,000# ● Flight Capsule: 6,000#/5000# ● Total Dry Flight Spacecraft: 5,800#/5650# ● Total Science Payload: 390 / 390# (Instruments and DAE)
GENERAL DESIGN REQUIREMENTS	● Propulsion Module and Spacecraft Structure Sized for Most Demanding Mission (Including a 7,000# capsule)
PROPULSION REQUIREMENTS	● Total ΔV — 1950 M/Sec (Six Maneuvers) ● LMDE for All Maneuvers
ELECTRICAL POWER REQUIREMENTS	● Maximum Average Power — 800 Watts ● Maximum Intermittent Peak — 900 Watts
DATA RETURN REQUIREMENTS	● Average Single Spacecraft-To-Earth Data Rate: 12,500 Bits/Sec
RELIABILITY ALLOCATIONS	● Flight Spacecraft: 0.747 ● Science Payload: 0.322 ● Total 1 of 2 Flight Spacecraft (Including Science): 0.422

Figure 4: KEY 1973 PERFORMANCE AND DESIGN REQUIREMENTS

	MISSION YEAR			
	1973	1975	1977	1979
Trajectory type	I	II	II	I
Attainable launch period (days) for fixed arrival date	50	63	66	30
Shortest communication distance at encounter ($\text{Km} \times 10^6$)	158	352	300	242
Longest communication distance at encounter ($\text{Km} \times 10^6$)	260	380	365	268
Maximum communication distance during orbit ($\text{Km} \times 10^6$)	393	380	365	360
Earliest arrival date	1/23	8/3	7/21	7/18
Latest arrival date	4/2	11/12	11/17	8/14
Shortest trip time (days)	157	342	306	222
Longest trip time (days)	231	422	408	266
Shortest Sun-Mars distance at arrival (AU)	1.573	1.525	1.465	1.575
Longest Sun-Mars distance at arrival (AU)	1.640	1.630	1.610	1.535
Longest possible Sun-Mars distance (AU)	1.670	1.630	1.610	1.575
Shortest possible Sun-Mars distance (AU)	1.573	1.380	1.380	1.380

Figure 5: TRAJECTORY PARAMETER RANGES

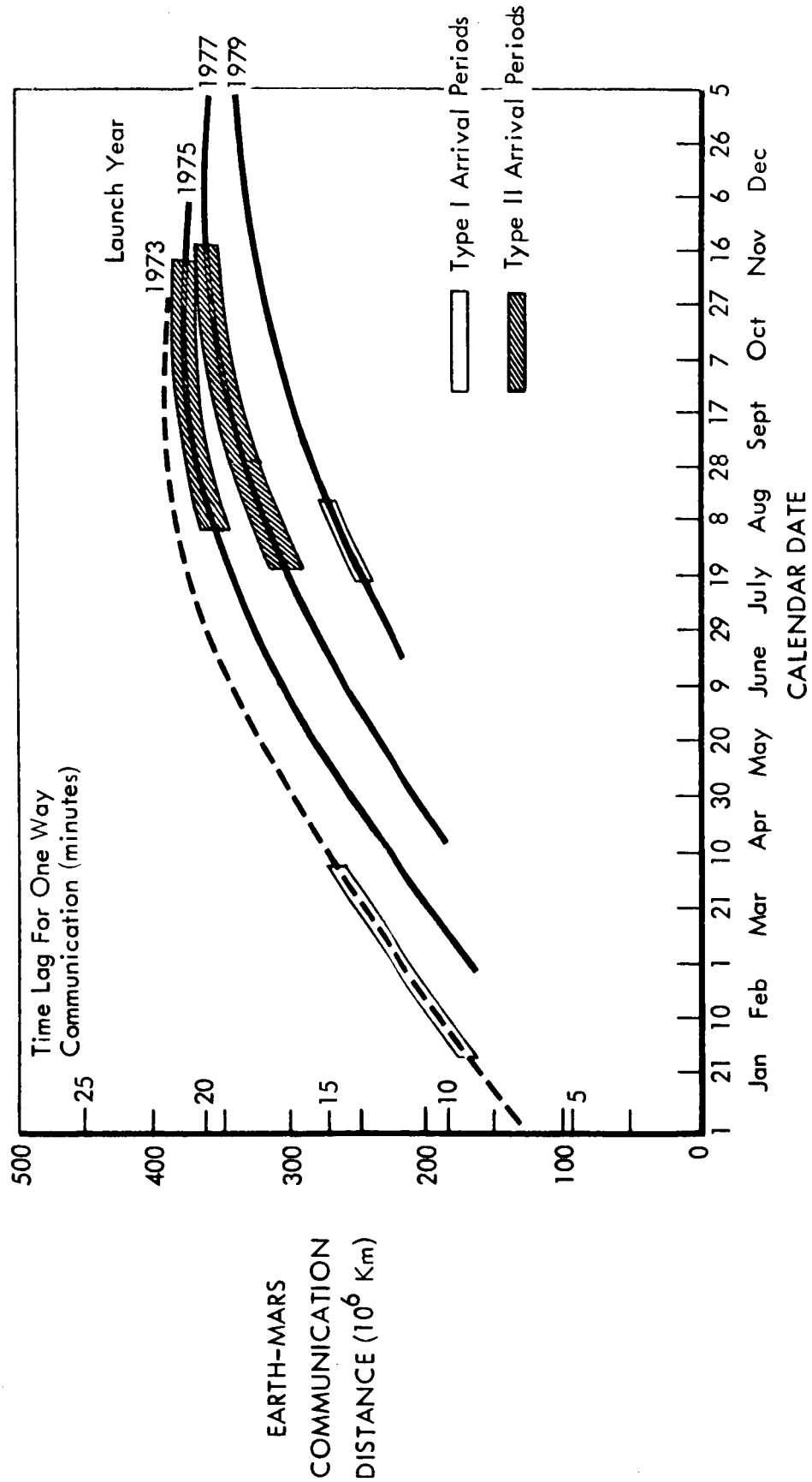


Figure 6: EARTH-MARS SEPARATION RANGE

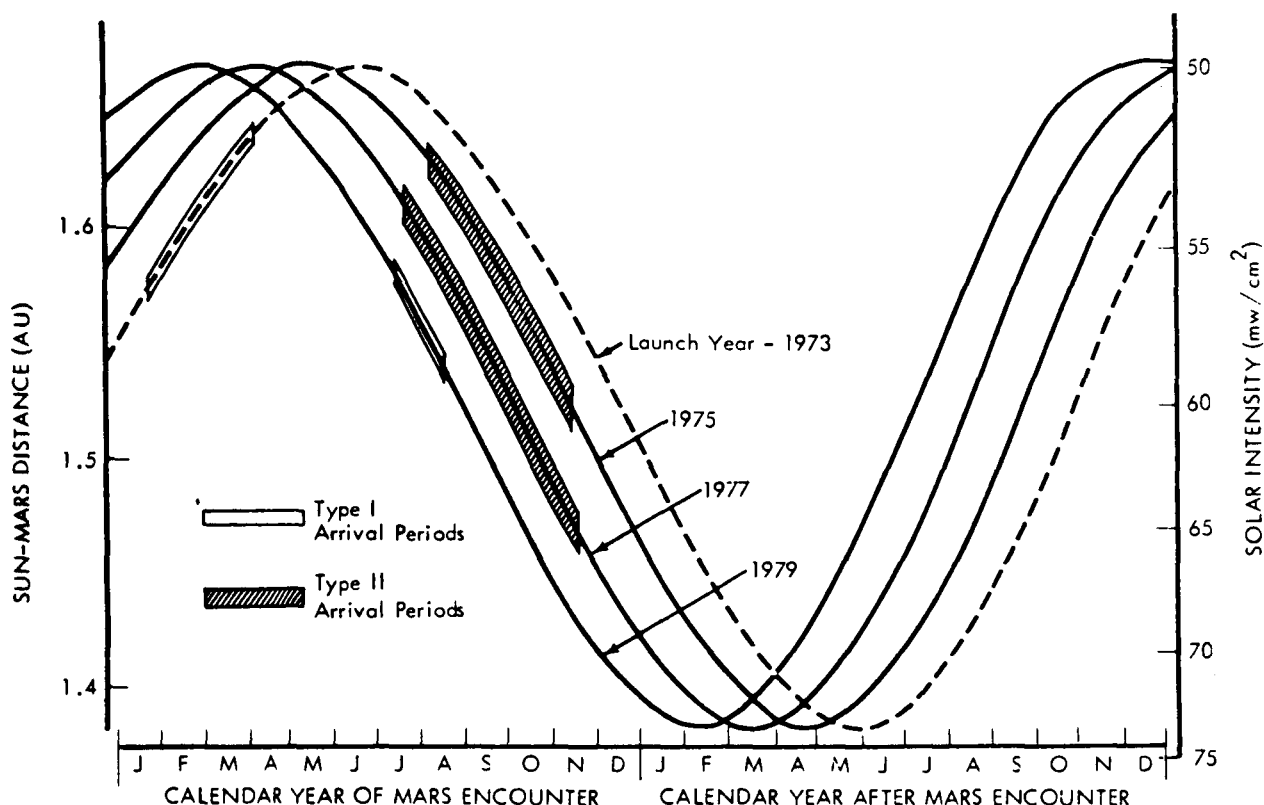


Figure 7 : SUN-MARS SEPARATION RANGE

The above effort resulted in a preliminary set of mission/system requirements for the 1973-1975-1977-1979 Mars opportunities. These were considered preliminary until the supporting analyses and tradeoffs were completed.

The ground system analyses and studies, using functional flow diagrams, established the requirements for OSE, MDE, and software. Operational support equipment consists of four categories of equipment: launch complex equipment (LCE), system test complex (STC), assembly, handling, and shipping equipment (AHSE), and fueling and servicing equipment (F&SE). Requirements and concepts for each of these were established. One of the key items is the STC which is shown in Figure 8. The core of the STC is the computer which interfaces with the spacecraft via the digital data equipment. The STC concept depicted can be van-mounted which provides flexibility and increased confidence in test activities. Flexibility is provided in that the van can be recycled to follow-on flight hardware or to support a spare spacecraft. Added assurance in test results is provided in that the flight hardware is always tested with the same ground hardware from the point of manufacture through KSC operations.

The mission and system analyses and tradeoffs conducted during Task D are indicated in Figure 9 which relates these studies to the appropriate part of the mission profile. The results of these studies are summarized below.

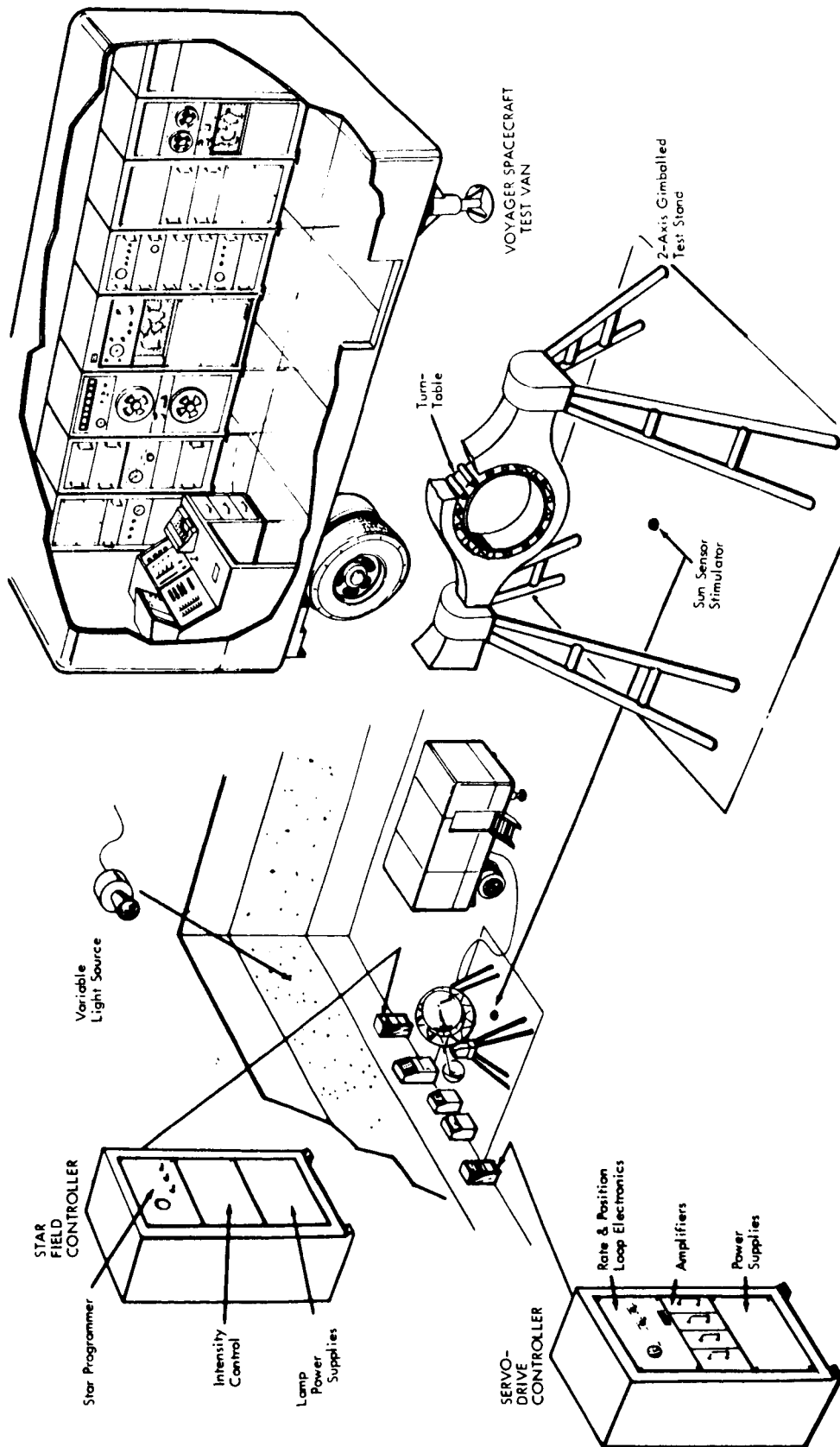


Figure 8: SYSTEM TEST COMPLEX CONCEPT

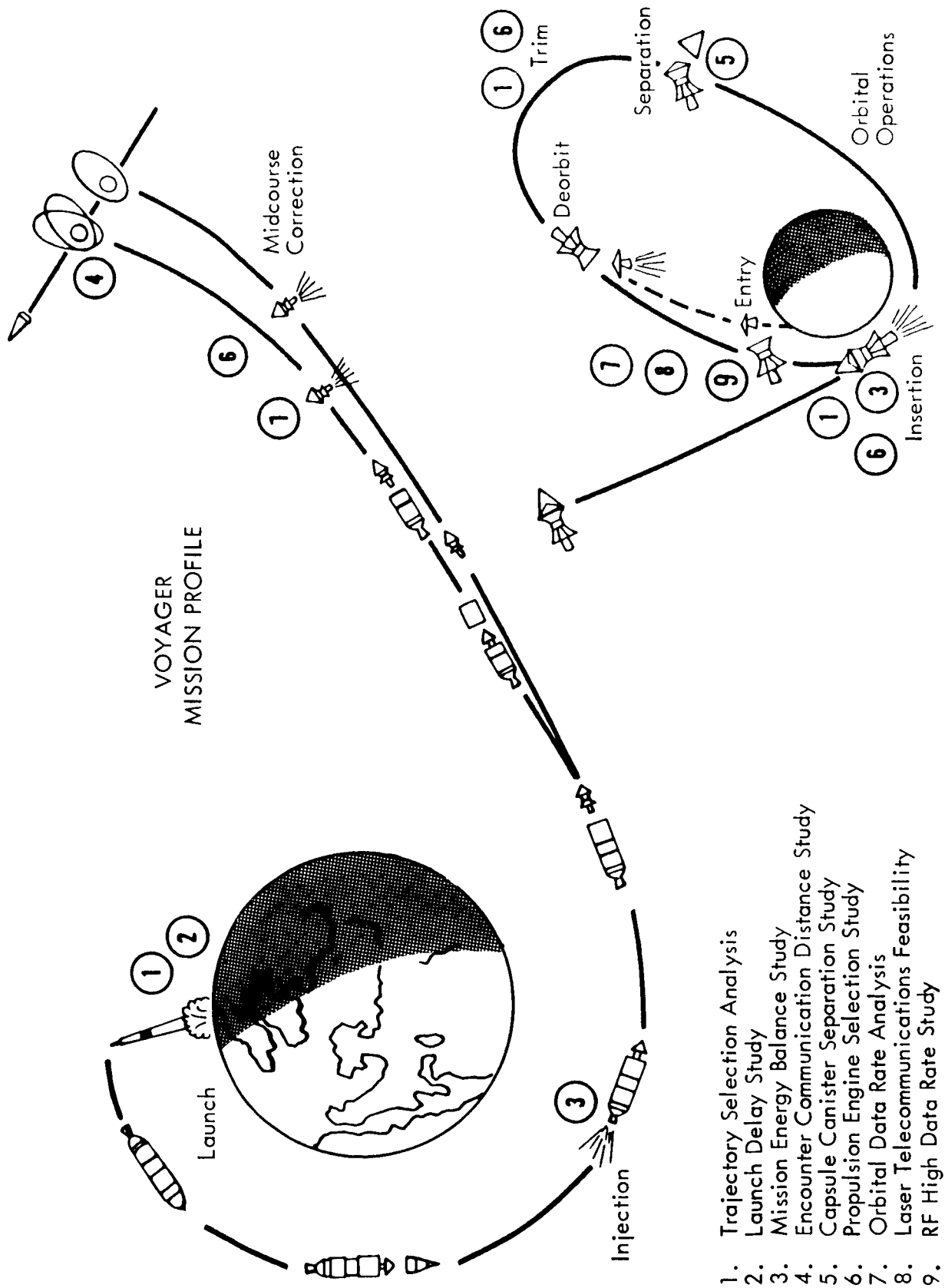


Figure 9 : SYSTEM ANALYSES AND TRADE STUDIES IDENTIFICATION

Trajectory Selection Analysis

The objective of this study was to establish the Mars mission parameters for the 1973-1975-1977-1979 Voyager missions. The orbital parameters were to be specified, including arrival date, orbit lifetime, orbit trim and insertion ΔV , and the usual parameters of altitude, period, and inclination. In all cases, the scientific objectives, established elsewhere in this study, were the governing parameters in establishing the requirements of the selected orbits. Having established the desired orbital conditions at Mars, the interplanetary trajectory was defined to meet the requirements of the Mars orbit. The interplanetary trajectory portion of the study included consideration of aim point biasing, midcourse corrections, arrival time separation, and launch system capability.

The significant results of this study are summarized below:

- 1) The design orbit for the 1973 mission envelope, shown in Figure 10, has periapsis and apoapsis altitudes of 500 km and 18,590 km respectively, a period of 12.4 hours, and an inclination of +45 degrees to the Martian equator (initial periapsis location in the northern hemisphere).
- 2) The probability that this orbit will decay to impact before 1984 is less than 8×10^{-6} . This probability calculation takes into account the errors in the orbital elements after orbit trim and the probability that orbit trim will fail, and presumes that the capsule canister is carried into orbit and jettisoned after orbit trim.
- 3) The trajectory from Earth-to-Mars can be aimed for direct insertion into the design orbit about Mars. The probability that this interplanetary trajectory will hit the planet is less than 2×10^{-6} which is consistent with planetary quarantine allocations.
- 4) The 45-degree orbit permits taking science data at subvehicle points lying between 45 degrees north latitude and 45 degrees south latitude. For imaging experiments, an illumination angle between 0 degrees and 40 degrees is desired. These angles are attained for latitudes between 29 degrees north and 45 degrees south and for altitudes between 500 and 1500 km. This altitude-latitude-illumination angle domain is traversed during the first 120 days in orbit. During this time period, the "optimum" illumination angle, 15 degrees, is attained at altitudes between 500 and 735 km.
- 5) The nodal separation between adjacent orbit ground tracks is 2.5 degrees. If photographs covering 100×100 km of the planet surface are considered, contiguous longitudinal coverage at the equator can be achieved in approximately 74 days. The overlap attainable increases with latitude to the latitude limits of ± 45 degrees.
- 6) An orbit trim velocity allocation of 150 m/sec for the 1973 mission will satisfy all March and April arrival dates within the mission operation envelope.
- 7) A total velocity allocation of 210 m/sec and three midcourse corrections are adequate for correcting trajectory dispersions and for providing each vehicle with 4-day arrival date bias capability.

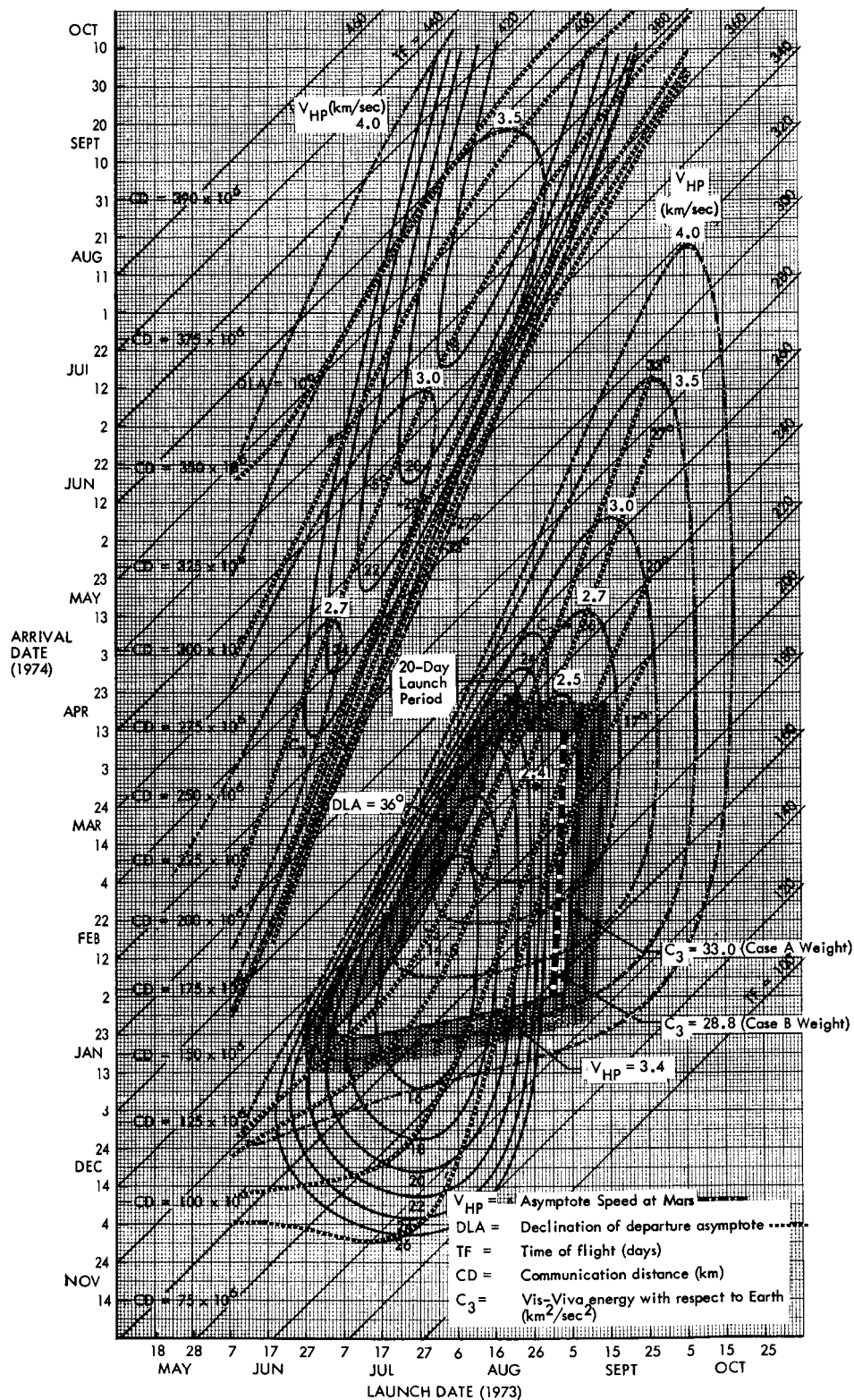


Figure 10: EARTH-MARS TRANSFER CHARACTERISTICS - 1973 TYPES I AND II

Launch Delays

The objective of this study was to present a set of mission alternatives that might be used in the event that the initial launch period could not be met. The study considered offloading the payload in various combinations, including a single flight spacecraft, to establish a wider launch period. In addition, a less desirable orbit was to be considered if this could extend the launch period. Figure 11 shows the summary data from this study. The left hand column is a reference 1973 mission. The launch period increase in days is noted at the bottom for various hardware and operations planning options. The nominal orbit had a periapsis altitude of 1000 km and a period of 13.8 hours. The marginal orbit had the same periapsis altitude but a period of 60 hours. As can be seen, trip time can be traded for launch period. Note that the 5000 pound project payload contingency is proposed to be used when determining alternate missions due to unforeseen launch delays. By offloading the payload to carry only a single spacecraft, a total extension of 67 days in launch period is possible over the reference condition.

Mission Energy Balance

The objective of this study was to evaluate, for the 1973-1975-1977-1979 opportunities, the tradeoff between launch vehicle performance allocation and spacecraft propulsion performance to maximize the weight in Mars orbit.

For each mission, there is a range of trajectory parameters which are bounded by constraints as shown in Figure 12. The DLA boundary is set by launch azimuth restrictions due to range safety considerations. The minimum launch period is set to provide assurance that a launch can be made considering probabilities of wind delays and hardware anomalies. The C_3 boundary is established by the launch vehicle capability to launch various payloads to specified Earth escape velocities. The V_{HP} boundary is set considering the spacecraft propellant required to slow the spacecraft for capture in Mars orbit. This trajectory envelope can be subdivided into launch-period envelopes as shown. For any launch period and related arrival date, a maximum C_3 and V_{HP} can be determined from the curve. This is illustrated in Figure 13 which shows that, for the 20-day launch period, early arrival dates result in high values of V_{HP} which imply larger propellant weights for the spacecraft. Late arrival dates give lower values of V_{HP} but at the expense of longer trip time as well as the high energy required of the launch vehicle.

Converting Figure 13 to a plot of arrival date versus payload in Mars orbit results in Figure 14. Here it is seen that the payload in Mars orbit is sensitive to launch period for the 1973 and 1979 opportunities. Selecting a 1973 arrival date of February 15 rather than April 4 results in a significant increase in payload in Mars orbit. The 1973 spacecraft structures and propulsion are sized to carry the largest of the science payloads (1000 pounds in 1977 and 1979) and for the largest capsule (7000 pounds). Using this design approach, a Case A flight capsule (5000 pounds) can be accommodated using Type I trajectories on all four missions. For Case B capsule weights (6000 and 7000 pounds), Type II trajectories are required in 1975 and 1977. All data on Figure 14 are based on a 5000 pound project payload contingency.

REFERENCE MISSION	ALTERNATE MISSION MODES							
NUMBER OF SPACECRAFT	2	2	2	1	2	1	1	1
NUMBER OF CAPSULES **	2	2	2	1	0	0	0	0
ORBIT PERIOD, HOURS	3.1	13.8	13.8	13.8	60.0	60.0	60.0	60.0
TRIP TIME, DAYS	192	151	252	159	160	167	370	
TOTAL ΔV , KM/SEC **	2.12	1.875	1.732	1.875	1.410	1.410	2.979	
PLANETARY VEHICLE WEIGHT, LBS. ***	21,700	20,012	19,086	20,012	9,018	9,018	15,273	
V_{HP} , KM/SEC	2.54	3.25	3.00	3.25	3.25	3.25	5.5	
C_3^* , KM^2/SEC^2	32.5	43.5	46.0	74.5	75.5	97.5	83.5	
LATEST LAUNCH DATE	9/6/73	9/11/73	9/22/73	9/28/73	9/29/73	10/1/73	11/12/73	
TOTAL LAUNCH PERIOD,	58	63	74	80	81	83	125	
LAUNCH PERIOD INCREASE, DAYS	0	5	16	22	23	25	67	

*SA-506 LAUNCH VEHICLE

** 5,000 POUND CAPSULE

LAUNCH AZIMUTH 110°

*** DOES NOT INCLUDE 5000 POUND PROJECT PAYLOAD CONTINGENCY

PERIAPSIS ALTITUDE 1,000 KM

Figure II: SUMMARY OF ALTERNATE MISSION MODES

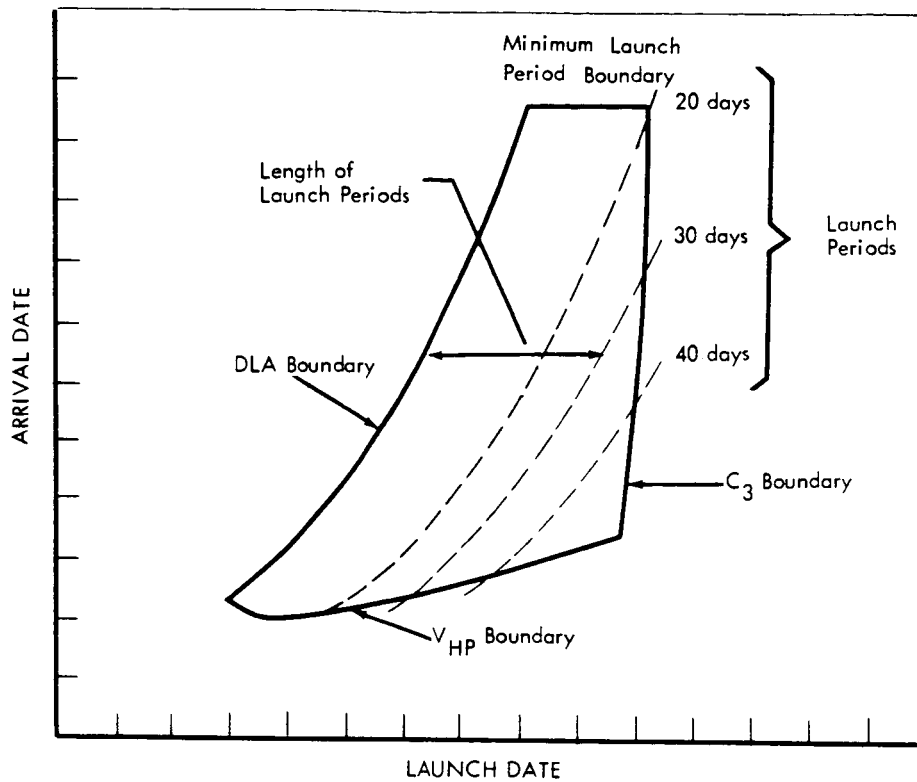
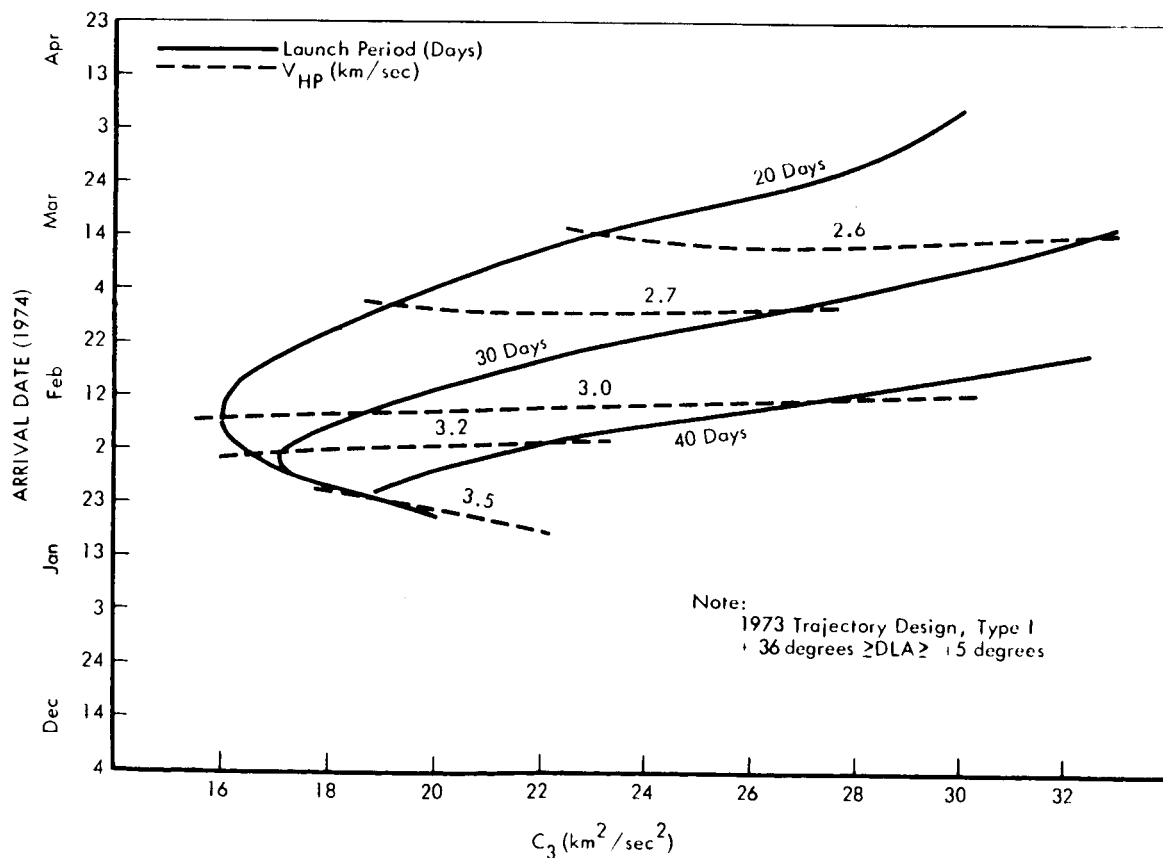
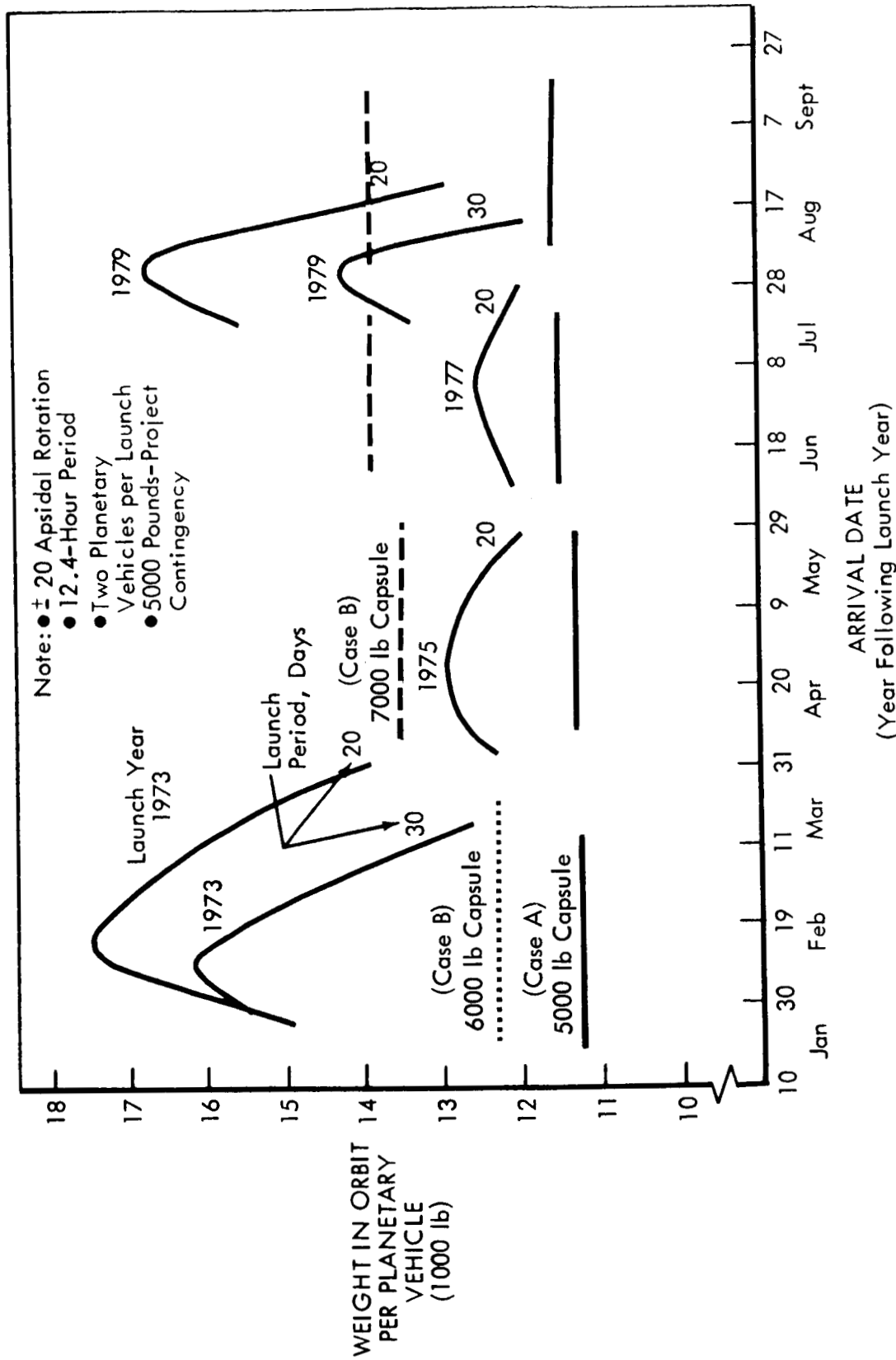


Figure 12: LAUNCH PERIOD ENVELOPES

Figure 13: MAXIMUM VALUES OF C_3 AND V_{HP} -- 1973 MISSION



Encounter Communication Distance

The purpose of this study was to examine operations planning alternatives that would result in minimizing the Earth-Mars communication distance at encounter. Minimizing the communication distance enhances the probability of data return since signal margins are increased. Decreasing the communication distance at encounter over the baseline system involves extending the envelope of transit trajectories in the direction of increasing the hyperbolic excess velocity at Mars, V_{HP} . Figure 15 shows the dependence of arrival date on V_{HP} . The resulting communication distance is also shown on the ordinate. This plot shows the value of launching early in the launch period as far as minimizing communication distance is concerned. Some additional savings in communication distance can be realized by first inserting into a long period orbit (60 hours), separate the lander capsule, and then transfer to a lower apoapsis altitude (reduced orbit period).

Capsule-Canister Separation

The biological barrier (canister) can be separated, exposing the capsule, prior to planetary vehicle orbit insertion or can be separated after orbit insertion. Separating the canister prior to Mars orbit insertion reduces the planetary vehicle mass, thereby reducing the propellant required for orbit insertion. For a 400-pound canister, about 190 pounds of spacecraft weight is saved by canister separation before Mars orbit insertion.

Retaining the canister cover as long as possible decreases the probability that a viable organism from the spacecraft finds its way to the capsule. The sources of potential contamination are (1) micrometeoroid spalling of the spacecraft impinging on the capsule and (2) viable organisms ejected from the spacecraft reaction control system impinging on the capsule.

Although the weight advantage of canister separation prior to Mars orbit insertion is attractive, the excessive time exposure of the sterile capsule under this condition leads to the conclusion to separate the canister after Mars orbit insertion. This becomes even more attractive if mission operations are such that the capsule is to remain attached to the spacecraft for a long time period before separation. This might be the case if landing site reconnaissance is required prior to capsule separation.

Propulsion Engine Selection

The contract requirements stipulated that the propulsion module was to be a single design for all mission opportunities (1973-1975-1977-1979). Three candidate liquid engines were to be considered: the LMDE, the Agena, and the Titan III Transtage engine, with most emphasis to be placed on the LMDE.

The spacecraft configuration studies considered installation of each of the three engines. The baseline design was selected using the LMDE and then this configuration was examined to note any problems with the installation of the other engines. The installation comparison is shown on Figure 16. Since the Agena and Transtage are smaller in size, they can be accommodated within the envelope specified for the LMDE if desired.

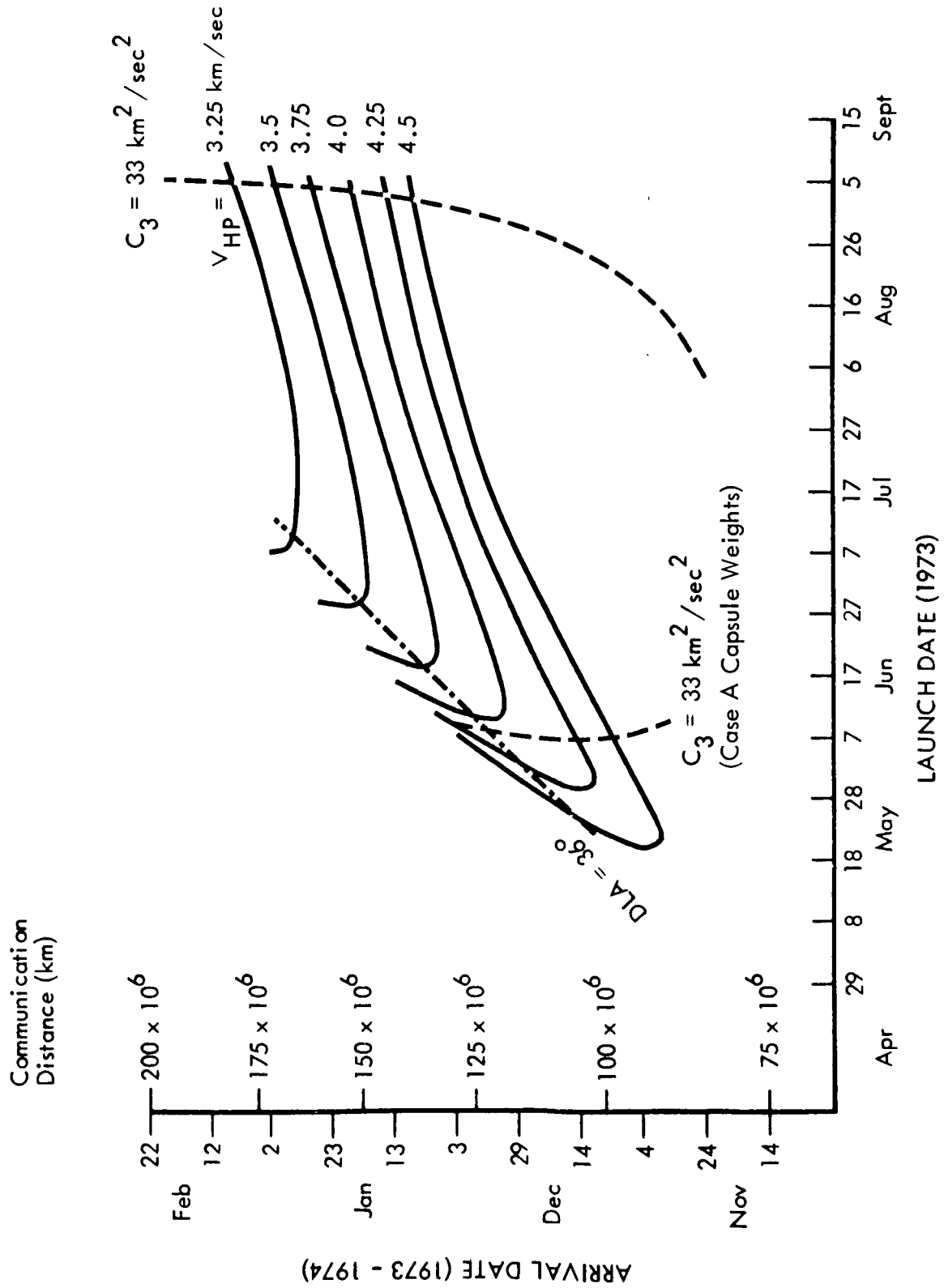


Figure 15: V_{HP} - COMMUNICATION DISTANCE RELATIONSHIP

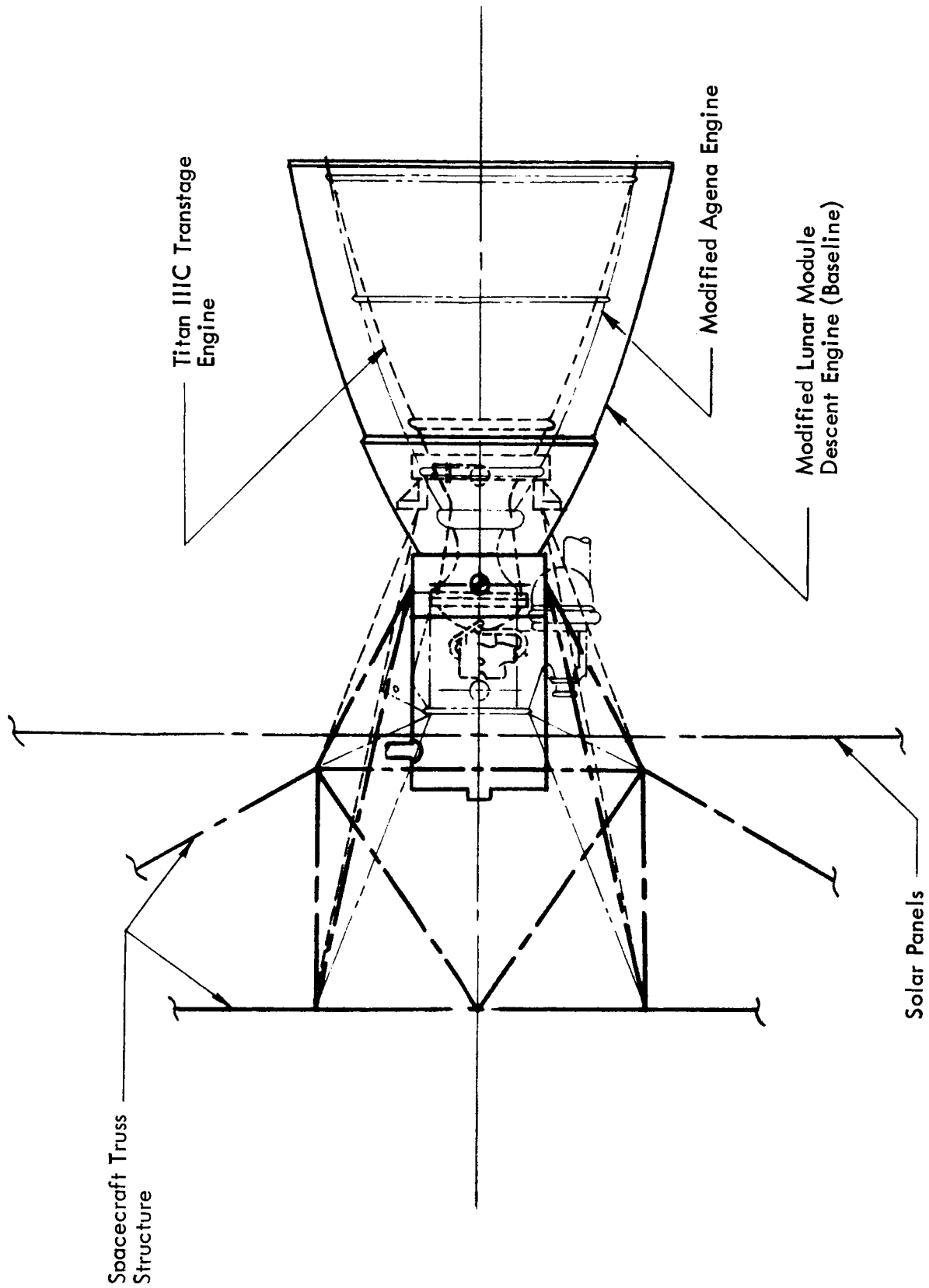


Figure 16: TRANSTAGE AND AGENA ENGINE INSTALLATION OVERLAY

Data Rate Requirements

The return of scientific data from a Mars orbiting mission requires a tradeoff between total data bits, transmission bit rate, and science objectives. The strongest contributor to data gathering is the imaging experiment since very large numbers of data bits are required to transmit pictures of reasonable resolution. There is a requirement for medium resolution pictures of Mars for purposes of mapping the planet and for capsule landing site selection. To establish the desired planet coverage and resolution requirements, an evaluation of the Lunar Orbiter program was conducted and is summarized in Figure 17. From this data, it was postulated that each Voyager Mars mission should cover 0.1% of the planet (four missions = 0.4%) at high resolution (5 to 10 meters), and over 75% of the planet at medium resolution (100 to 300 meters).

RESOLUTION (METERS)	MISSIONS 1, 2 & 3		MISSIONS 4 & 5	
	AREA (km ²)	LUNAR SURFACE (%)	AREA (km ²)	LUNAR SURFACE (%)
1	26,400	0.07	*	*
8	132,000	0.35	*	*
70 - 100	*	*	1.9×10^7	} 99.4%
1200 - 1600	*	*	1.9×10^7	

Figure 17: LUNAR ORBITER PHOTOGRAPHY RESOLUTION AND COVERAGE RESULTS

Four possible alternates are shown in Figure 18 for obtaining high and medium resolution pictures. Alternate 2 was selected as the best compromise between a conservatively designed telecommunications system for 1973 and the desire for high resolution pictures. Higher data rates, and hence more and better pictures, are attainable at the expense of development time and money.

Laser Telecommunications

A laser telecommunication subsystem on the spacecraft would provide at least an order of magnitude higher data rate than that postulated above. The laser technique is under development at this time, but a substantial increase in effort would be required to support the 1973 mission. It would seem prudent to carry the laser system as a development item on the earlier missions retaining the proven RF system as the prime subsystem. Switchover to the laser as prime could then take place in an orderly fashion.

An example of the laser technique applied to a Mars mission is shown in Figure 19. An Earth-based laser beacon illuminates the entire orbit of the spacecraft at Mars. The telescope aboard the spacecraft detects this signal and locks on to the beam. The laser beam generated aboard the spacecraft is transmitted through the same telescope using an appropriate lead angle to account for laser beam travel time. The Earth-based receiver is located as close as practical to the Earth beacon to compensate for atmospheric refraction of laser beams. A schematic of the system is shown in Figure 20.

Alternate	High Resolution		Medium Resolution		Total Science Data (bits)	Average Data Rate (bps)
	Coverage of Planet %	Resolution* at Periapsis (meters)	Coverage of Planet %	Resolution* at Periapsis (meters)		
1	0.50	10	75	100	2.66×10^{11}	21,200
2	0.10	5	75	150	1.57×10^{11}	12,500
3	0.20	5	75	150	8.82×10^{10}	7,000
4	0.01	10	100	500	4.22×10^{10}	3,400

* Average resolution is twice value shown since imaging is taken from periapsis (500 km) to 3x periapsis (1500 km).

Figure 18 : ALTERNATE IMAGING DATA OPTIONS

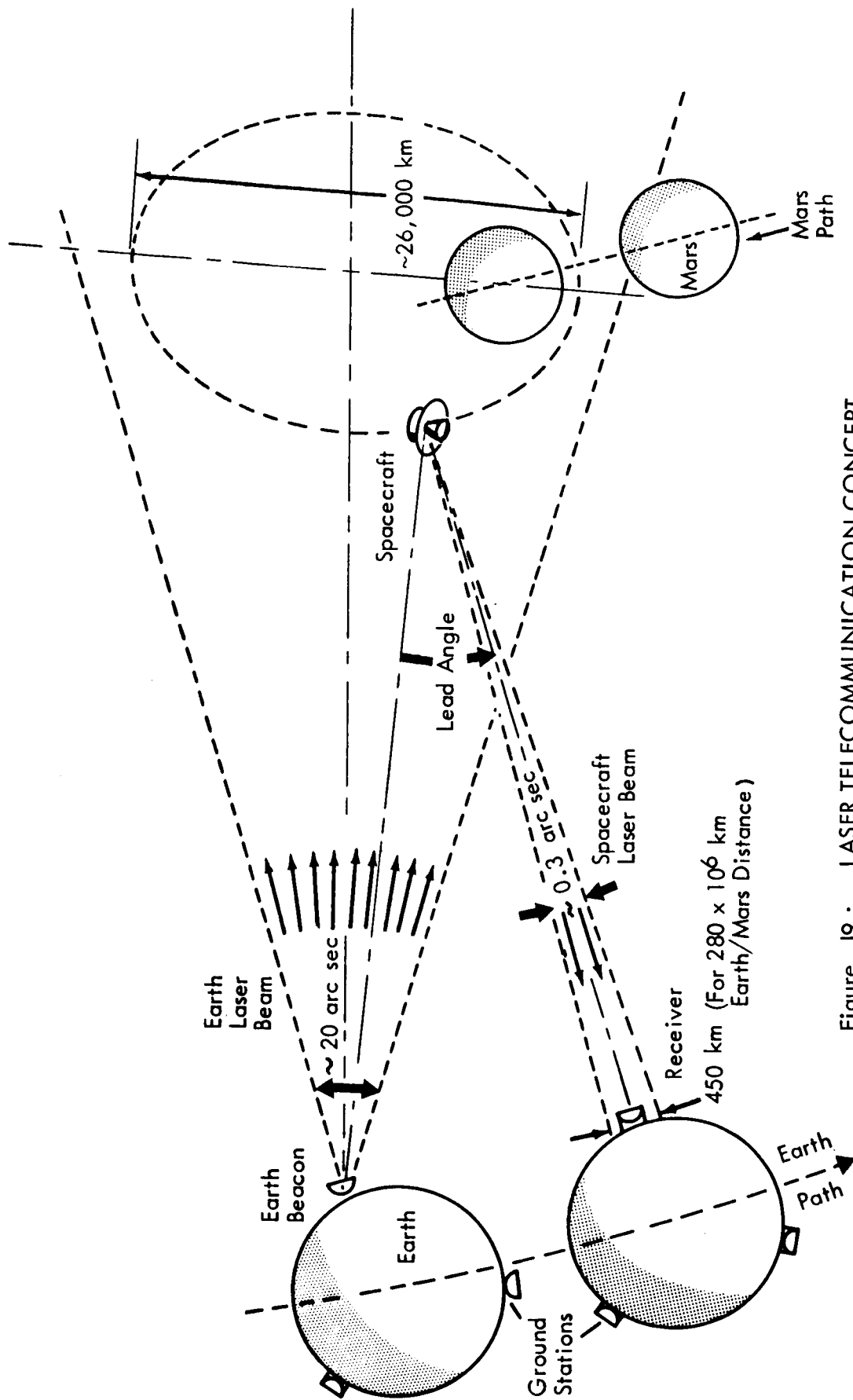


Figure 19 : LASER TELECOMMUNICATION CONCEPT

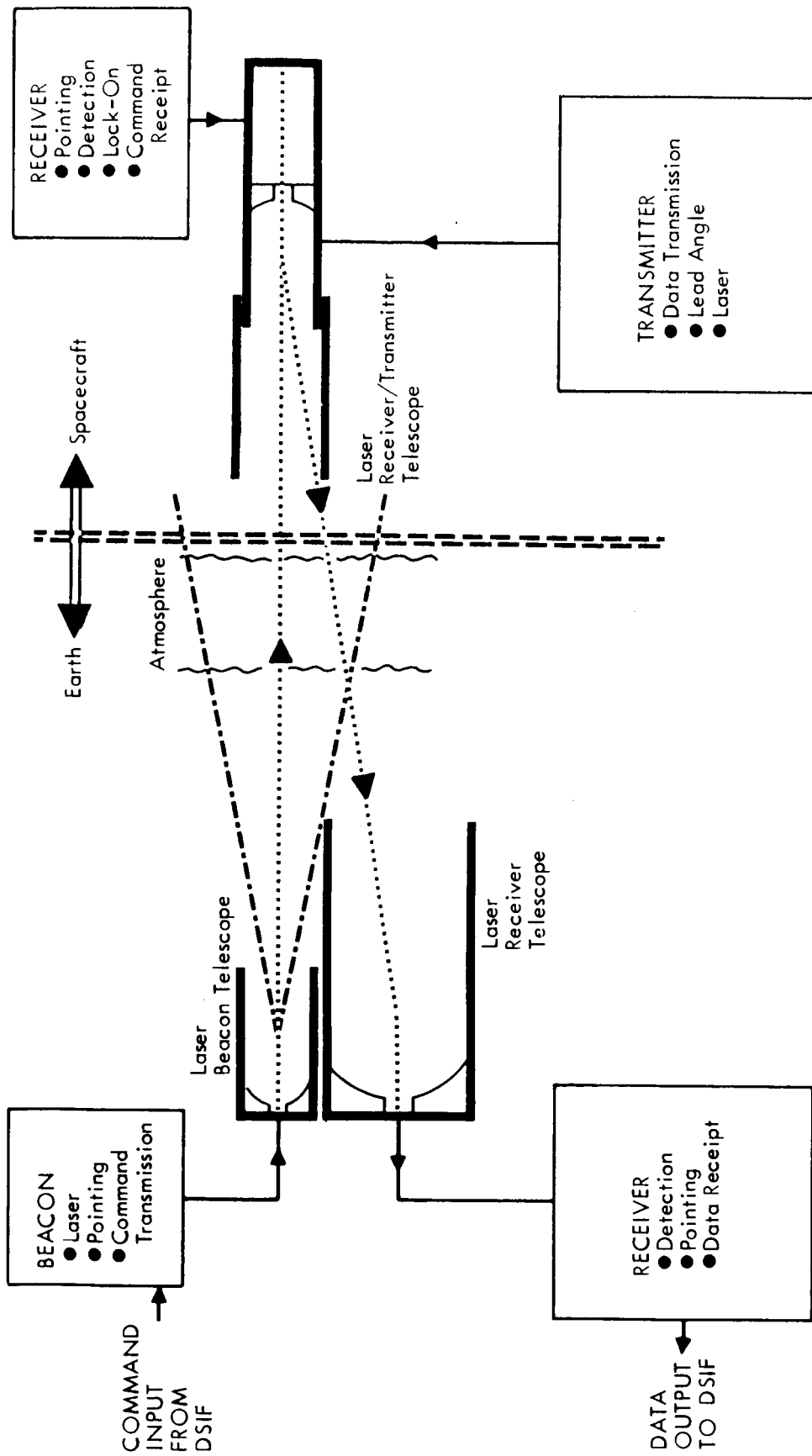


Figure 20: LASER TELECOMMUNICATION SYSTEM
FUNCTIONAL CONCEPT

Figure 21 shows a comparison of data rate capability for laser and RF systems based on 150 watts of input power. The bit rate achievable is 5 to 10 times higher for the laser system than for the RF system based on daytime operation. The nighttime performance is much higher due to less atmospheric effects. This operation is based on a helium or neon laser transmission system. Substantial increases in data rate, for the same input power, could be theoretically achieved by using a carbon dioxide nitrogen laser system.

TYPE	SPACECRAFT INPUT POWER (Watts)	SPACECRAFT TRANSMITTER DIAMETER (Ft.)	DATA BIT RATE, (bps)	
			CD * (260x 10 ⁶ KM)	CD * (380 x 10 ⁶ KM)
1973 Mission Baseline Configuration (Rf)	150	13.7 (Antenna)	2.4×10^4	1.2×10^4
Laser	150	2 (Telescope)	2.4×10^5	5.5×10^4
Laser	300	2 (Telescope)	9.5×10^5	2.2×10^5

* CD = Communication distance.

Figure 21: RF AND LASER PERFORMANCE COMPARISON

RF High Data Rate

The increased data rate requirements for the 1977 and 1979 missions resulted in a study to find ways of increasing the information bandwidth of the spacecraft-to-Earth data link for these missions. Two approaches were considered: (1) providing higher output power amplifiers with a higher gain antenna and (2) increasing the efficiency of utilization of the received signal power through different modulation techniques.

Analog and digital modulation techniques were examined. Digital techniques are more amenable to systems having requirements for multiple data rates and multiplexing of many different data sources. In addition, if data is recorded prior to transmission, signal fidelity is easier to maintain in the digital format. These are the primary reasons that digital telemetry systems have been used for deep space missions. Analog techniques offer a potential improvement in modulation efficiency and merit consideration where the payload is predominantly a single analog sensor, such as a high resolution film camera system.

Digital and analog techniques were compared on the basis of the impact on the power subsystem to (1) give an equivalent amount of data return and (2) for a given power level, the difference in data return obtained by the two modulation techniques.

This comparison is summarized below:

	Fixed Transmission Rate		Fixed Spacecraft ERP
	Digital (16,5) Coded PSK/PM	Analog VSB-AM/PM or FM/PM	Digital (16,5) Coded PSK/PM
Transmitter Power (watts)	360	140	140
DC Power (watts)	1200	465	465
Antenna Diameter (ft)	19	19	19
Effective Radiated Power (dbm)	92.4	88.3	88.3
Data Bandwidth (kHz)	64	27	10.3
Data Rate (bps)	320,000	320,000	124,000

As indicated above, for a fixed amount of data return, the analog technique requires much less power than the digital technique. For a given power input, the analog system would return over two times the data than that of a digital system.

The use of digital techniques is advantageous when a wide variety of data rates and multiplexing of data sources exist. This situation is characteristic of the Voyager science payload for the 1973 mission. An analog system in 1977 and 1979 for the large film camera appears very attractive. An approach might be to retain the digital system for the science and engineering data, but to add the analog system to support the large film camera system.

2.3 CONCLUSIONS AND RECOMMENDATIONS

The significant conclusions and recommendations resulting from the mission/system requirements and analyses task are summarized below:

- 1) Few of the 1971/1973 requirements are common but many are similar.
- 2) The spacecraft system concept requires minor modifications due to changing from a 1971 to a 1973 mission.
- 3) The establishment of the 1975 requirements did not affect the 1973 system concept.
- 4) The addition of a large film camera in 1977 and 1979 caused increases in solar array area and changes to the telecommunications system to accommodate the higher resolution pictures.
- 5) Unforeseen launch delays can be accommodated by offloading the launch vehicle payload to provide a longer launch period at the expense of a decrease in mission objectives. By offloading to a single spacecraft and allowing additional trip time, the launch period can be extended by 67 days.
- 6) The Earth-Mars communication distance at encounter can be reduced by about 35% by selecting early launch, early arrival conditions, rather than the latest, April 2, 1974, arrival distance.
- 7) Separating the capsule biological barrier (canister) after Mars orbit insertion, rather than before insertion, is recommended since it may be desirable to retain the capsule attached to the spacecraft for long periods of time to accommodate landing site reconnaissance.
- 8) An examination of the tradeoff between launch vehicle performance and spacecraft propulsion requirements showed that a significant increase in weight in Mars orbit could be achieved by selecting a February 15, rather than an April 4, arrival date for the 1973 opportunity.
- 9) All three candidate liquid engines (LMDE, Agena, and Transtage) can meet the Voyager Mars mission requirements. The propulsion module is designed such that any of the three engines can be used with minor modifications.
- 10) A 12.4-hour orbit period, 500-km periapsis orbit inclined at +45 degrees, can be used to define the 1973 spacecraft system consistent with presently understood scientific objectives.
- 11) An average data rate of 12,500 bps is recommended as a reasonable compromise between conservative design for 1973 and a desire for high resolution imaging. Higher data rates, and hence higher resolution pictures, are attainable at the expense of development time and dollars.
- 12) It is recommended that additional emphasis be given to the development of a laser telecommunication system for the 1977 and 1979 Mars missions. The laser system offers attractive gains in data rate over an RF system and would allow increased photo coverage at high resolution.

- 13) It is recommended that a laser engineering experiment be carried on the 1975 mission to verify its performance before incorporation in the 1977-1979 spacecraft system.
- 14) Although the Apollo automatic checkout equipment can be used to support Voyager ground testing, it appears that a van-mounted concept is a better choice when considering flexibility of operations and confidence gained by mating the spacecraft and its ground test hardware from point of manufacture through launch.
- 15) Logistics considerations for the spacecraft resulted in a recommendation to fuel the spacecraft on the launch pad using the existing Apollo Lunar Module fueling system modified for Voyager application.
- 16) Although many options are available, it appears that performing payload-module (spacecraft, capsule, shroud segment) integration testing at the contractor's facilities is advisable to reduce the impact on KSC facilities and personnel. Shipment of the integrated payload module to KSC would be by air, and the module would be in the horizontal position.

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3.0 FLIGHT SPACECRAFT CONCEPT

The objective of this task was to define a spacecraft system concept that could meet the 1973 Mars mission/system requirements and constraints. The final concept was selected after the implications of the 1975-1977-1979 mission requirements on the spacecraft concept were determined. The rationale was to ensure the 1973 spacecraft concept had the flexibility to grow to meet the subsequent mission requirements without major changes.

The flight spacecraft resulting from these studies is shown in Figure 22. The overall dimensions are 170 inches long and 240 inches in diameter. Provisions were made to allow mounting the flight capsule to form a planetary vehicle. All interface dimensional requirements were met.

The spacecraft concept is based on an objective of attaining modular construction, where practical, to achieve flexibility in maintenance operations and to facilitate the ground test program. The modularization is indicated in Figure 23 where it is seen that several basic modules exist. This concept allows assembly and testing of all the spacecraft electronics subsystems without the propulsion module and also allows completely separate propulsion module testing.

The major subsystem concepts were selected as a result of mission and system level analyses. The remainder were selected as a result of detail studies below the system level. The major subsystem concepts are:

- 1) A liquid propulsion system utilizing the lunar module descent engine (LMDE).
- 2) A photovoltaic (solar array) electric power source with batteries for off-Sun operations.
- 3) A celestial reference, attitude-stabilized vehicle.
- 4) A passive temperature control system consisting of insulation, radiators with louver control, and suitable thermal paints or coatings.
- 5) A computing and sequencing system that stores and executes desired mission events and is reprogrammable by ground command.
- 6) A communications system that can store all data received during each orbital period and transmit to Earth all data received each orbit. The system is compatible with the DSN.

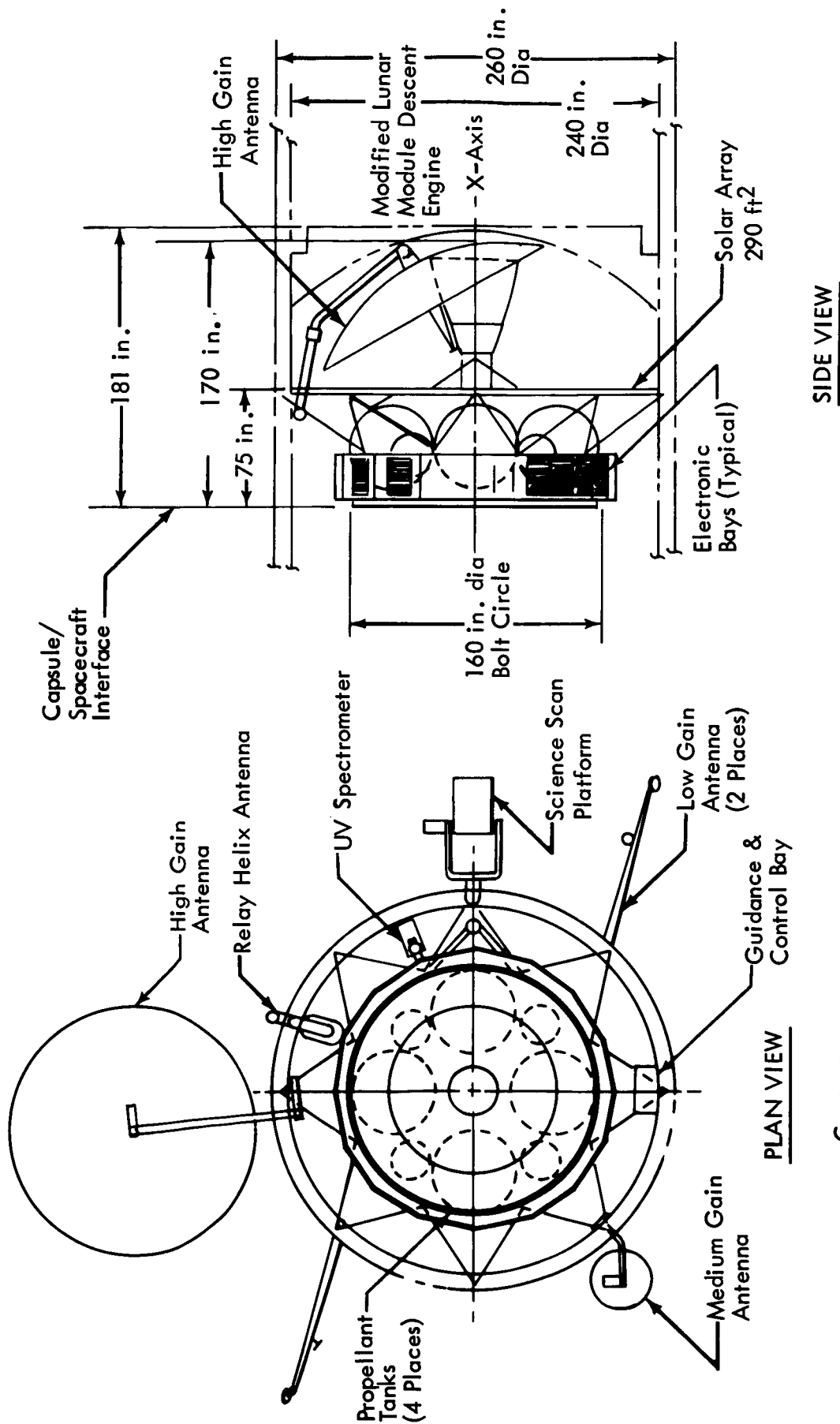
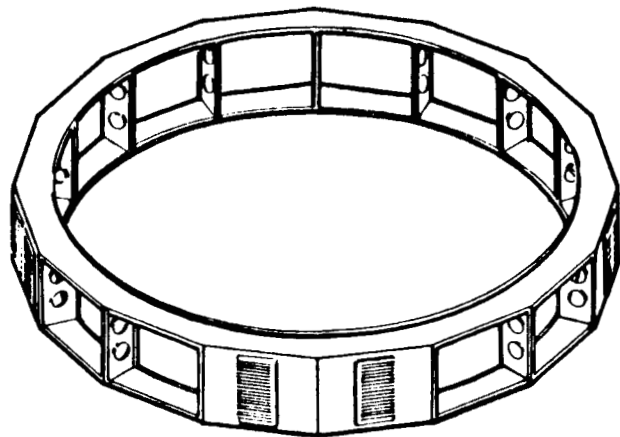
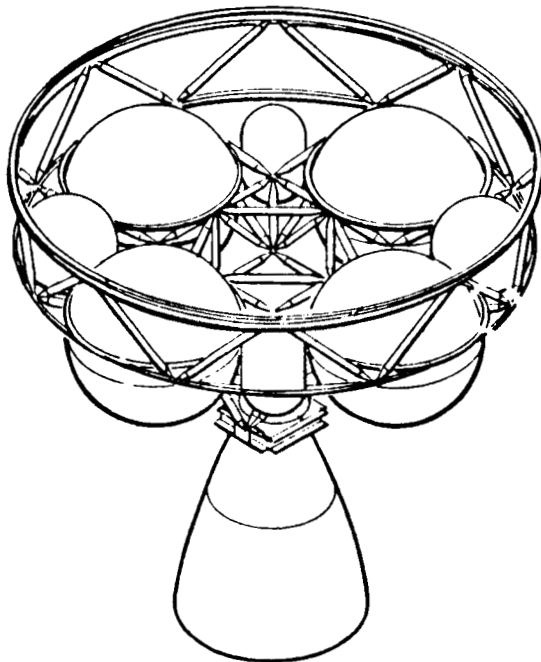


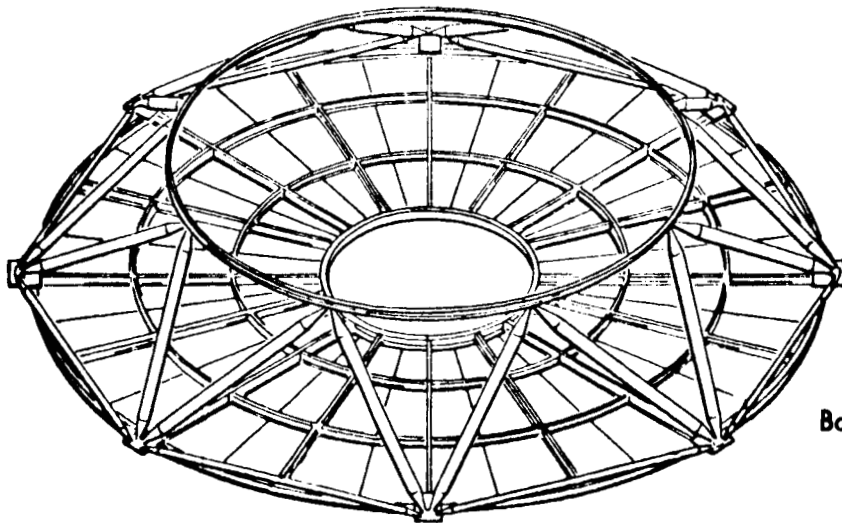
Figure 22: BASELINE CONFIGURATION



Equipment Module



Propulsion Module



Basic Module

Figure 23: BASELINE MODULES

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4.0 SPACECRAFT FUNCTIONAL DESCRIPTION

The flight spacecraft definition is based on the requirements and constraints established by the mission/system analyses and requirements task. The system concept established by that task was the starting point of detailed studies resulting in the flight spacecraft definition. Major emphasis was placed on the 1973 mission/system requirements. However, the definition for 1973 was not finalized until the requirements for the subsequent missions were evaluated as to their impact on the 1973 spacecraft. The 1973 spacecraft structure and propulsion are sized for the largest capsule weight (7000 pounds, Case B) and the largest science payload (1000 pounds).

The end result of this task was the detail definition of the 1973 Voyager Mars spacecraft and its subsystems.

The spacecraft inboard profile drawing is shown in Figure 24. The overall dimensions are within the established interface requirements. Electric power is supplied by a 290 square foot solar array consisting of 16 panels on which are mounted a total of 60,480 solar cells. The solar cells are 2 by 2 cm, 12 mils thick, 2 ohm-cm resistivity, with 6-mil cover glass and have an air mass zero efficiency of 11% based on active cell area. Under nominal conditions for early Mars orbital operation, the solar array will provide 904 watts of d.c. power to the spacecraft. The operating temperature of the solar array is from 128°K (-145°C) (Sun-occulted) to 388°K (+115°C) (near earth). The voltage output will vary over the range of 62 to 108 volts d.c.

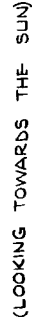
Three batteries provide power to the spacecraft when the solar array is off-Sun for maneuvers, or is occulted from the Sun. Each battery is composed of 38 series-connected, 20 ampere-hour, nickel cadmium cells. The minimum battery voltage is 38 volts, and up to 500 charge-discharge cycles may be required. Each battery is rated at 911 watt-hours and has its own charger.

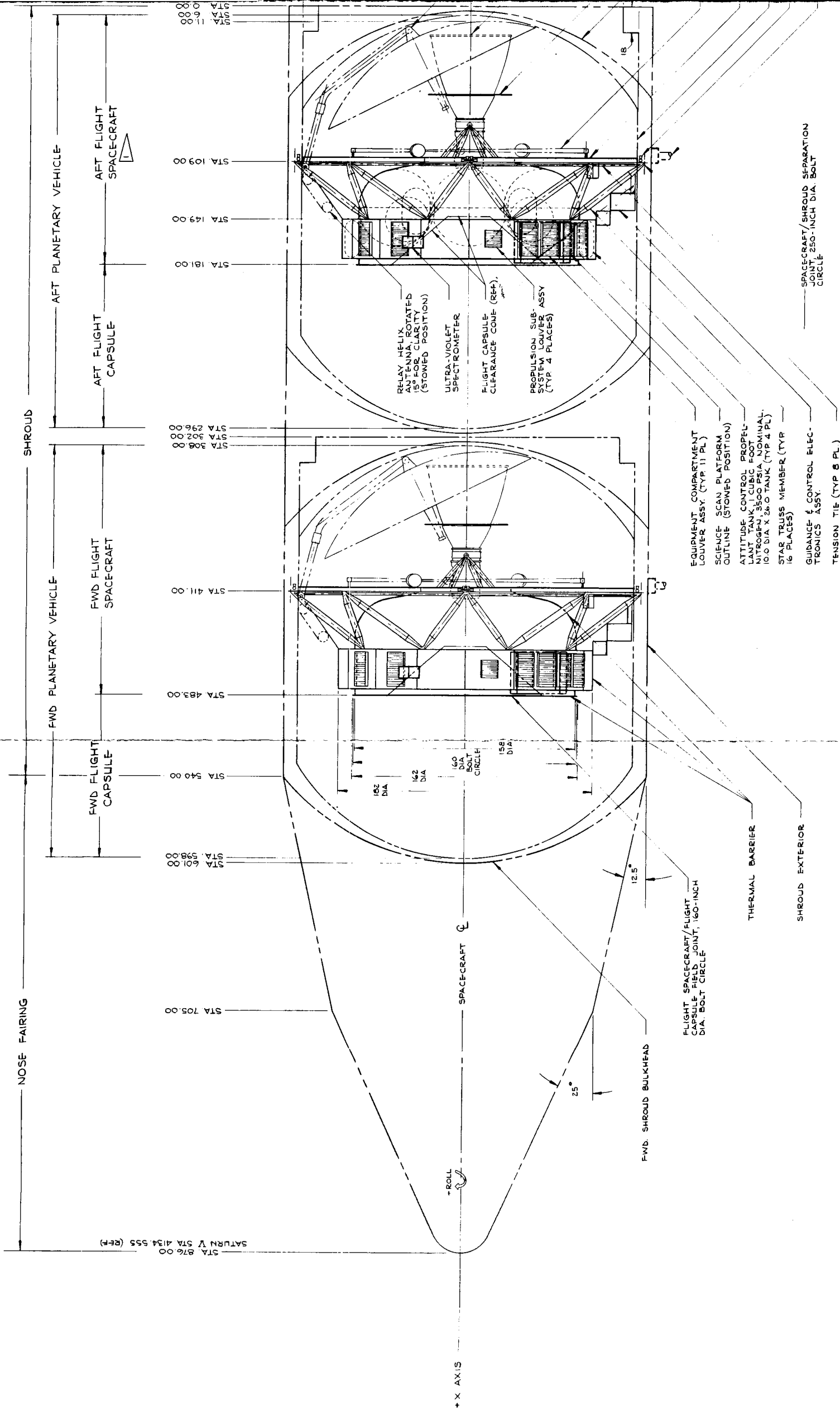
Electric power is distributed in the form of regulated 50-volt rms, 2400-Hz, single-phase power to a.c. loads. The power is obtained from regulated 230-watt inverters. A.c. distribution is provided by three buses. Redundant loads are connected to two of these buses. Nonredundant loads have relays for transfer to another bus in case of failure of their normal bus. Variable voltage d.c. is provided to the capsule, traveling wave tube amplifier (TWTA), propulsion solenoid valves, and thrust vector control actuators.

The guidance and control subsystem provides three-axis attitude stabilization to the spacecraft. The Sun-Canopus celestial references are augmented by an inertial reference system for initial rate damping, off-Sun maneuvers, and occultations. Control signals from these references command cold-gas nitrogen thrusters which apply control torques to the spacecraft. To control the spacecraft during the propulsive portion of a maneuver, the thrusting engine is gimballed by electro-mechanical actuators commanded from the inertial reference unit and attitude control electronics. Roll is controlled by the reaction control jets.

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SIDE VIEW
LAUNCH CONFIGURATION
TO SUN →

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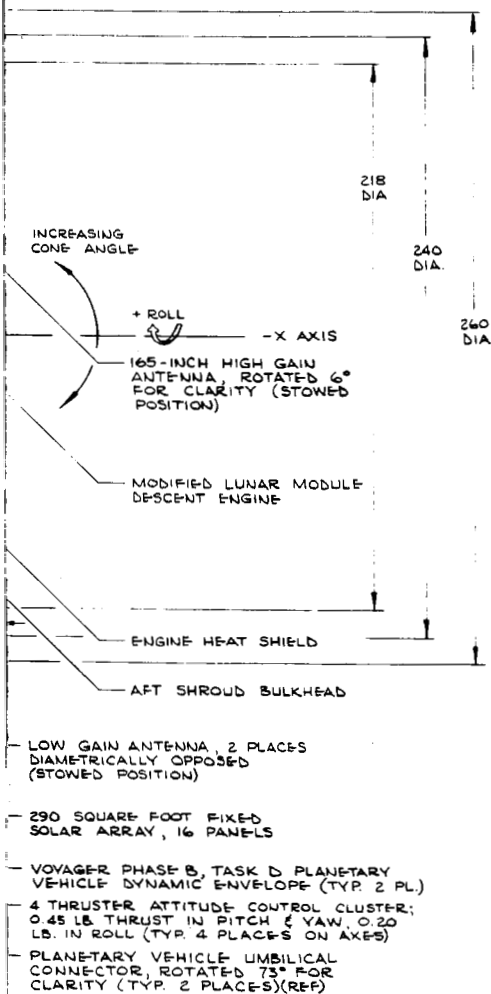
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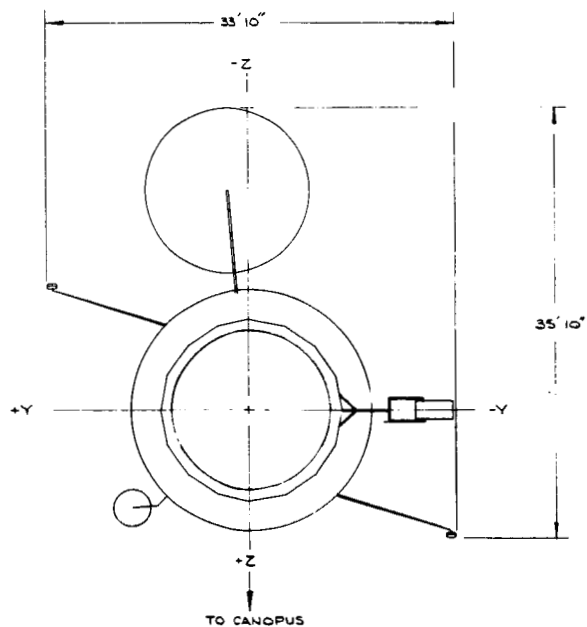
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SATURN V STA 3250 555 (REF)



2 TELECOMMUNICATION & EQUIPMENT LOCATIONS ESTABLISHED FOR 1973 THROUGH 1979 MARS OPPORTUNITIES. TELECOMMUNICATIONS EQUIPMENT DEFLECTION AXES ORIENTATION & SCIENCE EQUIPMENT LOCATIONS OPTIMIZED FOR MARS 1973 MISSION OPPORTUNITY, 4-4-74 ARRIVAL DATE, -30° INCLINATION & FLIGHT CAPSULE TOUCHDOWN AT TRUE ANOMALY OF 200 TO 270° AS APPLICABLE

1 THE FORWARD & AFT SPACECRAFT ARE IDENTICAL



SPACECRAFT DEPLOYED DIMENSIONS
(NO SCALE)
LOOKING TOWARDS THE SUN

Figure 24: SPACECRAFT GENERAL ARRANGEMENT DRAWING

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36-E

37 & 38

The Canopus sensor is an image-dissector type similar to that used on Mariner IV, Mariner Venus 1967, and Lunar Orbiter. The sensor has a wide field of view (32 degrees pitch and 4 degrees roll) which requires no mechanical gimbaling. Photo-voltaic Sun sensors provide Sun-spacecraft reference. The coarse sun sensor is used for acquisition, and an acquisition signal is generated any time the sunline is within 3 degrees of the fine Sun sensor null. The fine Sun sensor provides pitch and yaw axis control signals for normal limit-cycle operation.

Two inertial reference units are provided for redundancy. Each has three orthogonally mounted, single-degree-of-freedom, floated, rate-integrating gyros. Each gyro is individually switchable to provide redundancy in each axis.

An accelerometer is used to measure the spacecraft velocity change along the thrust axis during engine firing. The velocity increments are counted by the computer and sequencer subsystem, and thrust is terminated when the velocity change reaches the required value.

Two limb and terminator sensors are provided to detect crossings of the Mars illuminated limb and the terminator. The sensors are directed 90 degrees from the sunline on either side of the spacecraft and have a narrow fan-shaped (1.5 x 92 degrees) field of view to cover a range of orbits.

Two standby redundant cold-gas nitrogen systems provide reaction control; a total of 62 pounds of gas is stored in four 6Al-4V titanium tanks. Each system has eight thrusters which operate in force couples; 0.45-pound thrust in pitch and yaw, 0.20 pound thrust in roll. A maneuver rate of 0.2 deg/sec and a minimum angular acceleration of 0.01 deg/sec² in all three axes have been selected.

The data storage subsystem records spacecraft engineering data, cruise science data, high rate planetary science data, and capsule data received via the relay link. The selected design consists of eight magnetic tape recorders/reproducers with a total storage capacity of 1.242×10^9 bits. The bit error rate for each recorder is less than $1 \text{ in } 10^5$. The eight recorders are:

- 1) A flare recorder of 10^6 bits capacity.
- 2) A maneuver recorder of 10^6 bits capacity which also acts as a backup to the flare recorder.
- 3) Two identical recorders, 2×10^7 bits each, for IR scanner data, IR-UV spectrometer data, or postlanding capsule data.
- 4) Four identical recorders, 3×10^8 bits each, for TV data or for the capsule during its descent phase.

Operating modes for all eight recorders can be controlled by stored commands from the computer and sequencer or by ground command.

Each recorder has continuous reel-to-reel tape transport and associated electronics. All transports use two motors to cover the various record and playback speeds. Speed change of the two-phase hysteresis-synchronous motors is accomplished by power supply frequency division and attenuation of the supply voltage. A 25 to 1 maximum speed ratio is used for the single 12-pole motor.

The concept of three recorder types for the 1973 baseline can be replaced by a concept employing only one recorder type, reducing the total number required from eight to six recorders. Some additional multiplexing and formatting of science data is required; however, the concept of a single type (3×10^8 bits) of recorder is attractive. The weight change would be small, but the increase in redundancy capability is very attractive. A more detailed analysis of mission profile requirements is needed before final selection is made.

The telemetry subsystem provides acquisition, conditioning, and formatting of science and engineering data, and generation, modulation, and mixing of subcarriers for transmission of this data to Earth via the radio subsystem. The design incorporates time and frequency multiplexing, biphase modulation, and biorthogonal block encoding of science data. System flexibility is provided through implementation of six telemetry modes using the high, medium, and low gain antennas in proper relationship to provide optimum data rates for each mission phase.

The maximum data rate is 48,000 bps nominal for early orbit conditions out to 300×10^6 km. This rate drops to 24,000 bps nominal for operation out to 393×10^6 km.

The radio subsystem, operating at S-band, processes telemetry, command, tracking, and ranging signals between the spacecraft and the Deep Space Network. Redundant traveling wave tube amplifiers (TWTAs) rated at 50 watts worst case provide power for transmission. A germanium tunnel diode preamplifier with a 4.5-db nominal noise figure is used to lower the overall receiver system noise level.

Four S-band and two VHF antennas are provided to accomplish mission requirements. A 13.7-foot diameter, parabolic dish, two-axis controlled high gain antenna is the prime link with DSN. A single-axis control, 3-foot diameter, medium gain antenna is a backup to the high gain antenna. Two fixed, low gain antennas provide coverage during trans-Mars injection and early cruise as well as a backup for the end-of-mission phase.

The computing and sequencing (C&S) subsystem automatically sequences all routine spacecraft operations. It contains a cycled, special purpose computer that stores the mission sequence of commands in a 1024-word magnetic core memory. It can be updated by ground command to provide flexibility to handle anomalies or to change mission plans. The computer and sequencer contains the oscillator/register (master clock) for controlling all timed events. For example, spacecraft time is accurate within ± 20 seconds for orbit insertion without ground updating. Four kinds of electric signals are used between the computer and sequencer and other subsystems. These include a 6-volt, 4.8-kHz pulse train, for 100 milliseconds, for firing electroexplosive devices or latching devices; a 6-volt, 4.8-kHz square wave used for on-off control; a 6-volt clock reference pulse train of uniform pulses for synchronization and timing; and a 6-volt pulse train of binary data for information between the computer and sequencer and other subsystems.

The propulsion subsystem is a liquid bipropellant concept utilizing the Apollo lunar module descent engine (LMDE) for all propulsive maneuvers. The propulsion stage was sized for the worst-case mission in the 1970's to preclude changes. The LMDE operates in two thrust modes: 1050 pounds thrust for midcourse correction, arrival time biasing, and orbit trim, 9850 pounds thrust for orbit insertion. The tanks are sized to handle 12,625 pounds of usable propellants which provide a total ΔV of 1.950 km/sec required for the worst-case mission (1979). The tanks can be offloaded for all other missions.

The propellants, monomethylhydrazine and nitrogen tetroxide, are loaded into four main tanks plus two start tanks. Two helium tanks provide propellant expulsion. The system will provide 210 m/sec ΔV for midcourse and arrival biasing, 1590 m/sec for orbit insertion, and 150 m/sec for orbit trim for the 1973 Mars mission.

The LMDE is started in the low thrust mode (1050 pounds thrust) to settle the propellants prior to operation in the high thrust mode. The start tank propellants only are used for velocity corrections below 10 m/sec.

The structural subsystem provides support to all other spacecraft subsystems and is the prime physical interface with the launch vehicle shroud. A field joint is provided to accept the flight capsule. The primary structure is welded 2219 aluminum truss. It is designed for the maximum load condition encountered in any mission opportunity in the 1970's. The critical design load condition occurs during boost and results in a combined ultimate load condition of 5.94 g axial and 2.5g lateral with an ultimate safety factor of 1.25.

Deployable appendages are stowed until planetary vehicle release from the S-IVB. The appendages are released by the actuation of pyrotechnic pin pullers. Following release, they are deployed by spring-operated linear actuators using silicone oil for stroke damping. Pointing control for movable items such as the high gain antenna is achieved by pulse motor reduction gear mechanisms incorporating position feedback signals.

The planetary vehicle is attached to the launch vehicle shroud by eight explosive bolt assemblies. Upon release, compression springs located at each attachment point provide separation velocity.

The thermal control subsystem consists of thermal control coatings and paints, multilayer insulations, and bimetallic-actuated louvers. The spacecraft is used as a heat sink during transient conditions. Electronic equipment bay temperatures are controlled by space-facing louvers which react to the temperature of a radiator plate to open or close to maintain radiator plate temperature. Equipment mounted to the back side of the radiator plate transfers heat by conduction to the radiator which transmits heat by radiation to space through the louvers. The radiator temperature control range is 280°K (45°F) to 297°K (75°F). The louver assemblies are 35.6 by 61 cm (14 x 24 inches) having an effective area of 0.202 square meters (2.17 square feet). Each louver blade is supported and stabilized by bearings and is actuated by a bimetallic spring. The louver-radiator heat dissipation capability varies from 3.2 to 32.6 watts/square foot for the equipment bays.

Multilayer insulation assemblies are a maximum of 1.22 meters by 1.22 meters (4 feet by 4 feet) to facilitate manufacture and assembly. Depending on the application; the assembly is either multiple layers of 1/4-mil mylar aluminized on both sides, or multiple layers of 1/4-mil mylar aluminized on both sides with an outside cover layer of 2-mil H-film aluminized on one side, or multiple layers of 1/4-mil H-film aluminized on both sides with the outside cover of 2-mil H-film aluminized on one side.

The propulsion module is thermally isolated from the spacecraft subsystems to facilitate the ground test program and to account for a different operating temperature range, 278°K (40°F) to 317°K (110°F). Particular attention is given to the thermal conditions associated with LMDE firing. By insulating the nozzle skirt extension, the nozzle exit can be located 86 inches vertically from the solar array without overheating the solar array as a result of engine firing.

A weight statement for the 1973 configuration is shown in Figure 25. Based on a requirement of 23,000 pounds, established by the mission/system requirements and analysis task, there is a 240-pound margin for the 1973 spacecraft.

The requirements of subsequent missions (1975-1977-1979) have been considered in the 1973 spacecraft design. The impact of these requirements is summarized as follows:

Item Affecting Configuration	Mission Year			
	1973	1975	1977	1979
Science Payload - pounds	371	592	1001	1056
Capsule Weight - pounds (Case B)	6000	7000	7000	7000
Propulsion System	Designed for 1979 mission			
Antenna Diameter - feet	13.7	13.7	19	19
Structure	Designed for 1979 mission			
Attitude Control	Cold gas	Cold gas	Hydrazine	Hydrazine
Power - Solar Panel Area - square feet	290	352	414	414

The principal impacts of the subsequent missions are due to (1) large capsule, (2) large film camera, (3) larger electric power requirement, (4) more total data return. Both the propulsion module and the spacecraft primary structure have been sized initially to handle the most stringent mission requirements (1979).

The major implication is the addition of a large film camera for the 1977-1979 missions. Although the requirement for tape recorders decreases because the film is the storage device, the overall data requirements increase which effects the telecommunications system. The increased data rate, due to a larger total quantity of bits required, results in a change from a 13.7-foot to a 19-foot high gain antenna. The prime communication link is changed from digital to analog to handle the large quantity of film data. The future requirements appear so large that it appears wise to carry a laser communication experiment on the 1975 mission to place the program in a position to switch to laser communication for 1977-1979 in the event the development program is successful. The increase in maneuvering requirements due to the fixed installation film camera along with the heavier capsule leads to a potential change from a cold-gas nitrogen reaction control system to a hydrazine system to conserve weight. More stringent spacecraft stabilization requirements due to the large optics may result in the use of additional stabilization equipment such as control moment gyros. The fixed solar array can be augmented by deployable solar panels.

43

#1

CASE B		CASE A	
STRUCTURE		WEIGHTS (Pounds)	WEIGHTS (Pounds)
Capsule Support Truss			539
Upper Ring/Field Joint Flange			
Compression Struts & Fittings (16)		129	
Miscellaneous		69	
		54	
		6	
Star Truss			
Upper Ring		130	318
Compression Struts & Fittings		158	
Tension, Ties and Fittings		15	
Miscellaneous		15	
Special Purpose Fittings			
Capsule Truss/Star Truss		50	92
Tiedown/Separation Pads		20	
Miscellaneous		22	
PROPULSION			1,799
Lunar Module Descent Engine			
		409	
Fuel System			
Tanks (Main)		271.5	438.5
Tanks (Bellows)		38.5	
Tank Support		45.0	
Plumbing		77.5	
Miscellaneous		6.0	
Oxidizer System			
Tanks (Main)		271.5	464.5
Tanks (Bellows)		38.5	
Tank Support		71.0	
Plumbing		77.5	
Miscellaneous		6.0	
Pressurization System			
Tanks		440	475
Tank Supports		4	
Plumbing		31	
Engine Supports			
		12	2595
EQUIPMENT & INSTRUMENTATION			
Structure (Equipment Component and Thermal)			
Radiators		189	
Stiffeners		77.8	
Shear Webs & Chords		14.9	
Cable Tray		18.9	
Miscellaneous		52.7	
		24.7	
Guidance & Control			
Sensors		20	208
Computing & Sequencing		69	
Inertial Reference Unit		28	
Attitude Control Electronics		12	
Accelerometer Unit		5	
Gimbal Actuators - LMDE (2)		36	
Guid. & Control Electronics Ass'y. &			

Figure 25: DETAILED WEIGHT STATEMENT 1973 VOYAGER

FOLDOUT FRAME

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It is concluded that the 1973 Voyager Mars spacecraft designed in support of Task D will meet the specified requirements and constraints. Program cost savings result from sizing the propulsion and structural subsystems for the most stringent of the four missions. The modifications of the other subsystems to meet subsequent missions are considered to be a reasonable extension of the 1973 Task D baseline.

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5.0 SELECTED ENGINEERING TASKS

Since the five tasks are not directly related, the summary of each will contain the task objectives, the significant results, and conclusions of recommendations as appropriate. The five tasks in order of appearance in the statement of work are:

- Mission-Dependent Equipment Definition
- Voyager Program Test Flight
- Science Payload Evolution Considerations
- Particulate Contamination Considerations
- Photoimaging Considerations

5.1 MISSION-DEPENDENT EQUIPMENT DEFINITION

The objective of this task was to define the mission-dependent equipment (MDE), both hardware and software, required to support the Voyager Mars 1973 mission.

The existing deep space network (DSN) is designed to support unmanned missions to lunar and planetary distances. Since different programs result in differing interface requirements at the DSN, MDE hardware and software are added to the sites to make them compatible with the various spacecraft and missions. This is depicted in Figure 26.

The approach to this task was to develop the data measurements for the downlink and break this down into types, quantities, and real-time performance assessment data. Mission contingency actions were then identified and the requirements for uplink commands established. The downlink data requirements are shown in Figure 27 related to various mission phases. Over 460 separate measurements have been identified.

The extreme communication distance from Mars to Earth results in an approximate 12-minute delay between the sending of a signal and its reception. It can take over an hour to accomplish the following sequence: analyze a received signal, react to its implications, achieve communications system lockup, send the command, and then verify it has been received. This condition is shown by Figure 28. This communication delay time is one of the key parameters in deciding how to handle contingency items. It is not practical to store, on board the spacecraft, all planned missions events and all possible combinations of anomalies and their corrective action. Proper structuring of the mission software and allowing reprogramming of the spacecraft computer will alleviate this problem.

Mission-dependent software consists of computer programs added to the DSN to accomplish the Voyager mission. Figure 29 shows the functions to be performed by the software. At the DSIF's, the software edits telemetry data if required and verifies commands. At the space flight operations facility (SFOF) at Pasadena, the software performs various functions. First, the command telemetry handling routine formats data, time-tags incoming data, performs alarm monitoring,

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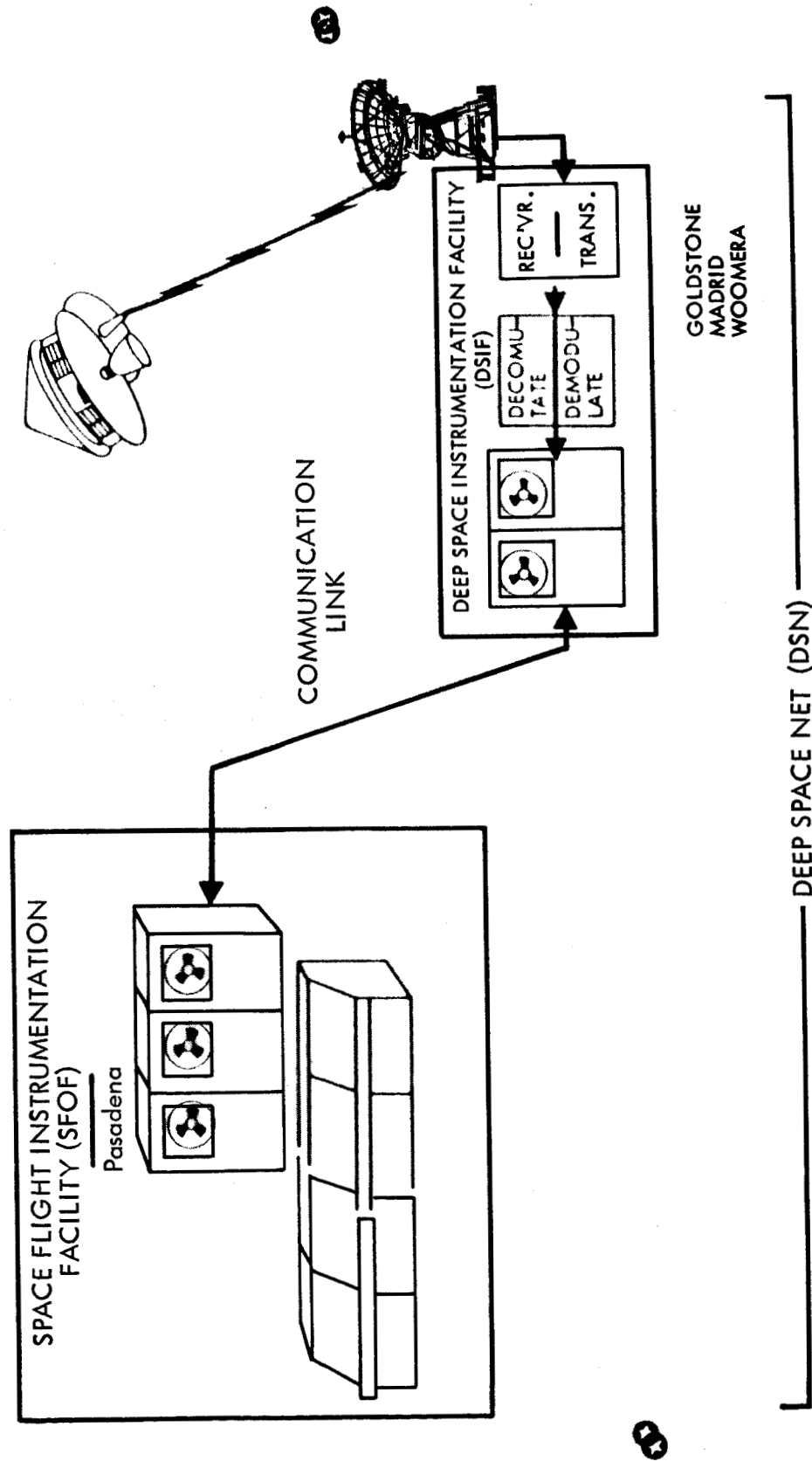


Figure 26: MISSION DEPENDENT EQUIPMENT

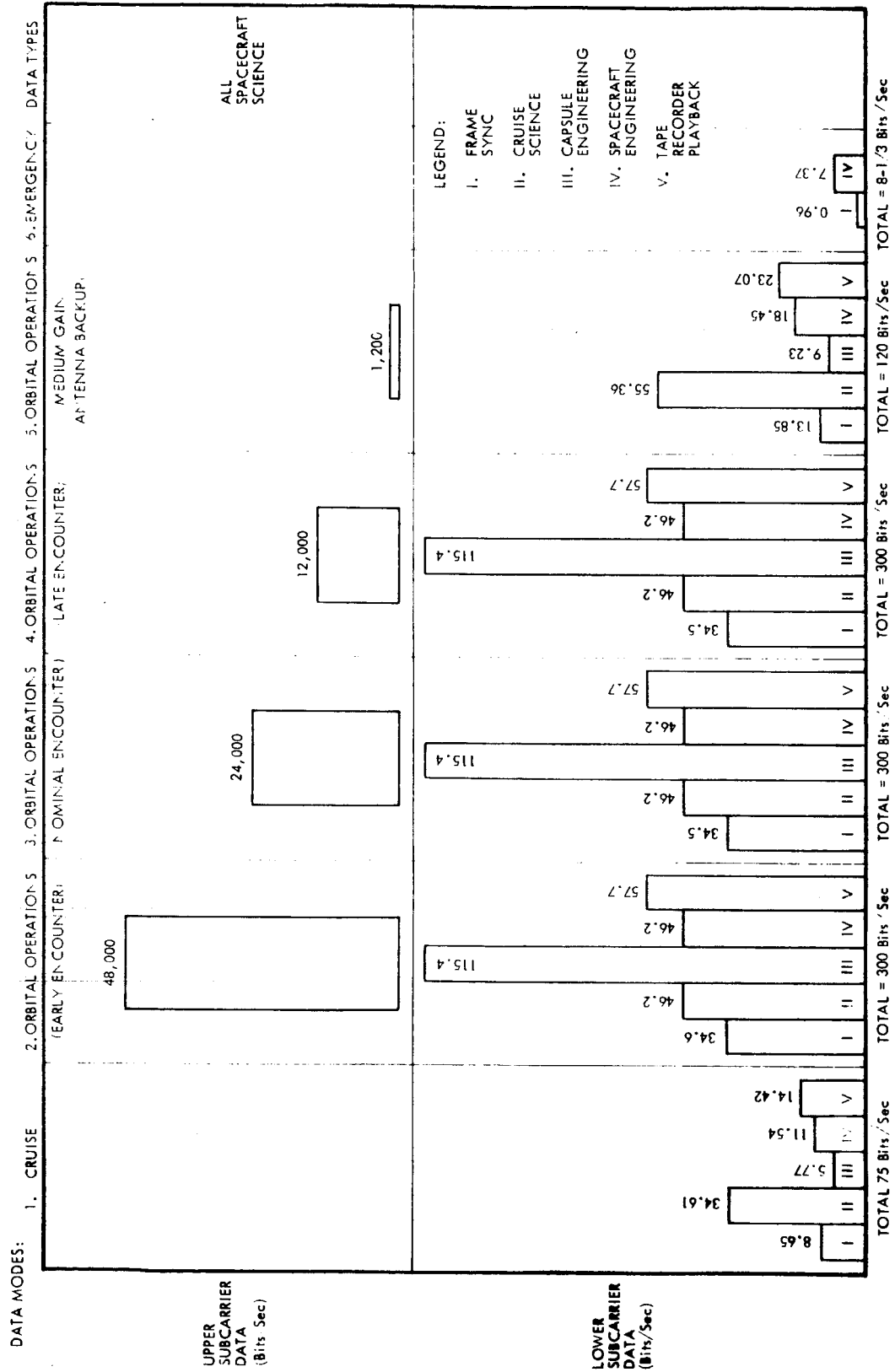


Figure 27: TYPES AND RATES OF SPACECRAFT - GENERATED DOWNLINK DATA

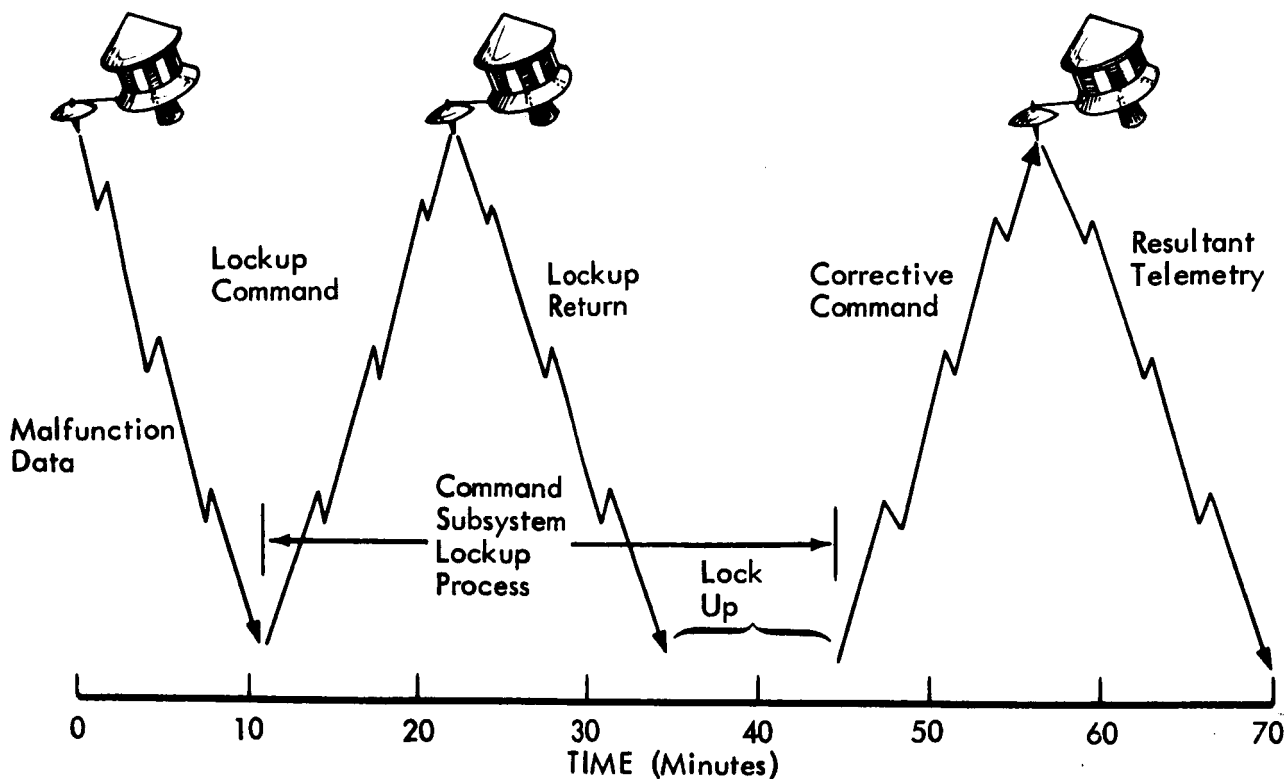


Figure 28: CONTINGENCY COMMAND TIME SEQUENCE

and selectively displays telemetry data. The flight path analysis and command (FPAC) software provides spacecraft positions and velocities, past and predicted, and computes maneuver sequences and trajectories. The spacecraft performance analysis and command (SPAC) software monitors spacecraft subsystem performance and provides predicted status and diagnostic subroutines. Mission integration and control ties the software programs together and provides the primary interface with data banks for use by FPAC and SPAC. It also provides for simultaneously handling data from two spacecraft and capsules. Checkout, training, and simulation software is used for operations personnel training preparatory for a mission.

The primary considerations for implementing MDE and software are the communications delay time at Mars distance and the requirements to handle two spacecraft simultaneously. These lead to the following recommendations:

- 1) Separate displays and command consoles should be implemented at the SFOF.
- 2) The telemetry data at each DSIF should be tagged by frame to indicate the originating spacecraft.
- 3) There should be a display that allows comparison of computed spacecraft status with current telemetry status.

Software Nomenclature	Function Performed	Where Used		Software Development		
		SFOF	DSIF	Mission Independent (80 to 100% Existing)	Modified Mission Independent	Mission Dependent (80 to 100% New)
Telemetry and Command Data Handling (TCD)	DSIF Command Processing		X			X
	DSIF Command Verification		X	X		
	DSIF Telemetry Data Handling		X			
	Input Processor Format Edit Output Processor			X		X X
	SFOF Real-Time Telemetry Processing Input Processor Telemetry Processor Output Processor	X		X	X X	
	SFOF Telemetry and Tracking Data Processor Input Processor Telemetry Processor Past Time Data Display	X		X	X	X
Flight Path Analysis and Command (FPAC)	Orbit Determination	X				X
	Guidance and Maneuver Analysis	X			X	
	Mission Information	X				X
Spacecraft Performance Analysis and Command (SPAC)	System Monitor and Control	X				X
	Contingency Aids	X				X
Mission Integration and Control (MIC)	Master Timing	X			X	
	Command Generation	X			X	
	Mission Events	X			X	
	Data Bank	X			X	
Simulation for Training and Software Checkout	Simulated Real-Time Telemetry Generation	X				X
	Simulated Real-Time Tracking Generation	X				X

Figure 29: MISSION SOFTWARE REQUIREMENTS

5.2 VOYAGER PROGRAM TEST FLIGHT

The objective of this task was to determine the benefits of a test flight in terms of added assurance and the effects on a postulated ground test program.

The approach to this task was to identify a Voyager program predicated on the use of ground testing only. Then a flight test to be added to the ground test program was considered. If the flight test could replace ground tests at no loss of mission assurance, then the costs for the deleted portions could be subtracted from program cost. Added assurance due to the test flight was to be identified where that occurred.

The test flight was projected as a launch of either a spacecraft or a planetary vehicle by a Saturn 1B. The test flight would necessarily be an Earth orbit flight unless the spacecraft propulsion system were used to augment the launch vehicle.

A list of candidate test requirements was established and a comparison made between the degree of confidence gained by a flight test or by a ground test. An example of one of the requirements is to demonstrate payload separations. The next step was to develop a matrix of payload configurations versus circular and elliptical Earth orbits that could be attained using the Saturn 1B (see Figure 30). From these orbital conditions, it was then determined if the spacecraft propulsion system could add enough velocity to achieve an Earth synchronous orbit or a solar orbit that would simulate Mars encounter. The analysis indicated that a planetary vehicle test flight is limited to Earth orbit only and that a synchronous Earth orbit or Mars encounter can be achieved for the spacecraft by utilizing the spacecraft propellants.

CONFIGURATION AND WEIGHT IN ORBIT	MAXIMUM CIRCULAR ORBIT ALTITUDE (n mi)	MAXIMUM ELLIPTIC ORBIT APOGEE (100 n mi PERIGEE) (n mi)
1 PLANETARY VEHICLE FULL FUEL LOAD	1020	2750
1 PLANETARY VEHICLE 10% FUEL LOAD	2300	7650
SPACECRAFT ONLY (NO CAPSULE) FULL FUEL LOAD	1630	4800
SPACECRAFT ONLY (NO CAPSULE) 10% FUEL LOAD	3490	15120

- Nose Shroud: 4880 lb, 3860 lb Jettisoned at 100 n mi
- Performance based on due east launch, SA-212 vehicle
- S-IVB Assumed Restartable

Figure 30 : VOYAGER/SATURN 1B FLIGHT TEST CAPABILITIES

There are several key advantages to considering a deep space flight rather than an Earth orbital flight.

- 1) The DSN performance cannot be verified for Earth orbital flights because of doppler rate, acquisition, tracking, coverage limitations, and communications delay time differences. Checkout of the Voyager system using the DSN is desirable because it would realistically simulate a mission and because signal strength margins could be evaluated.

- 2) The thermal environment near Earth is very different from that of inter-planetary space.
- 3) Realistic performance of the star tracker cannot be achieved in Earth orbit.

The major conclusions from the test flight study are:

- 1) Considerable program success assurance is added by the inclusion of a test flight. However, studies have shown that the costs of such a test flight considerably exceed the savings realized by deletion of some ground test simulations.
- 2) The best of the candidate flights was one in which the planetary vehicle was boosted to a maximum elliptical Earth orbit, the capsule was released, and then the spacecraft propellants were used to inject to a Mars encounter condition.
- 3) If money is available for a test flight, it is concluded that a deep space flight provides more program assurance than an Earth orbital test flight.

5.3 SCIENCE PAYLOAD EVOLUTION CONSIDERATIONS

The objectives of this task were to determine the ways in which science payloads may evolve from the 1973 Voyager Mars mission to the subsequent 1975-1977-1979 missions and to determine the impact of this evolution on the spacecraft.

The approach to this task involved starting with the basic objectives for planetary exploration established by the initial NASA charter and to construct a roadmap space experiment rationale that eventually leads to the definition of the experiments to be carried for the Voyager Mars missions. This overall rationale is shown in Figure 31. The rationale starts with the three stated fundamental questions. Scientists have developed a number of competing theories concerning the fundamental questions. The purpose of the measurements to be made by the experiments is to either confirm or deny the theories. To continue the chain, a theory cannot be measured, but models describing the theory can be mathematically constructed. From these models, certain implications can be drawn. The implications lead to specific questions that can be converted to experiments to answer the questions. It becomes evident that no single experiment will validate or invalidate a theory. Hardware limitations also result in an ever expanding desire for more and better experimental data to support or deny a theory. This trend is indicated in Figure 32 where a 1973 mission science payload is shown to evolve in many possible ways for the subsequent missions. The science support equipment also grows to handle the increased demand for measurements in subsequent missions. The exact path that will be followed depends in part on what is discovered in the first mission.

A postulated evolution of instruments is shown in Figure 33. This type of chart is used to plan the science missions for the various opportunities and to provide source data to specify spacecraft system interfaces such as weight and power. Using data from Figure 33, it is possible to establish the science subsystem characteristics as shown in Figure 34. As can be seen, the science payload increases from 371 pounds in 1973 to over 1000 pounds in 1977 and 1979. This trend is based primarily on the evolution of photoimaging equipment from television in 1973 to a large film-type camera in 1977. The theory here is that early missions would

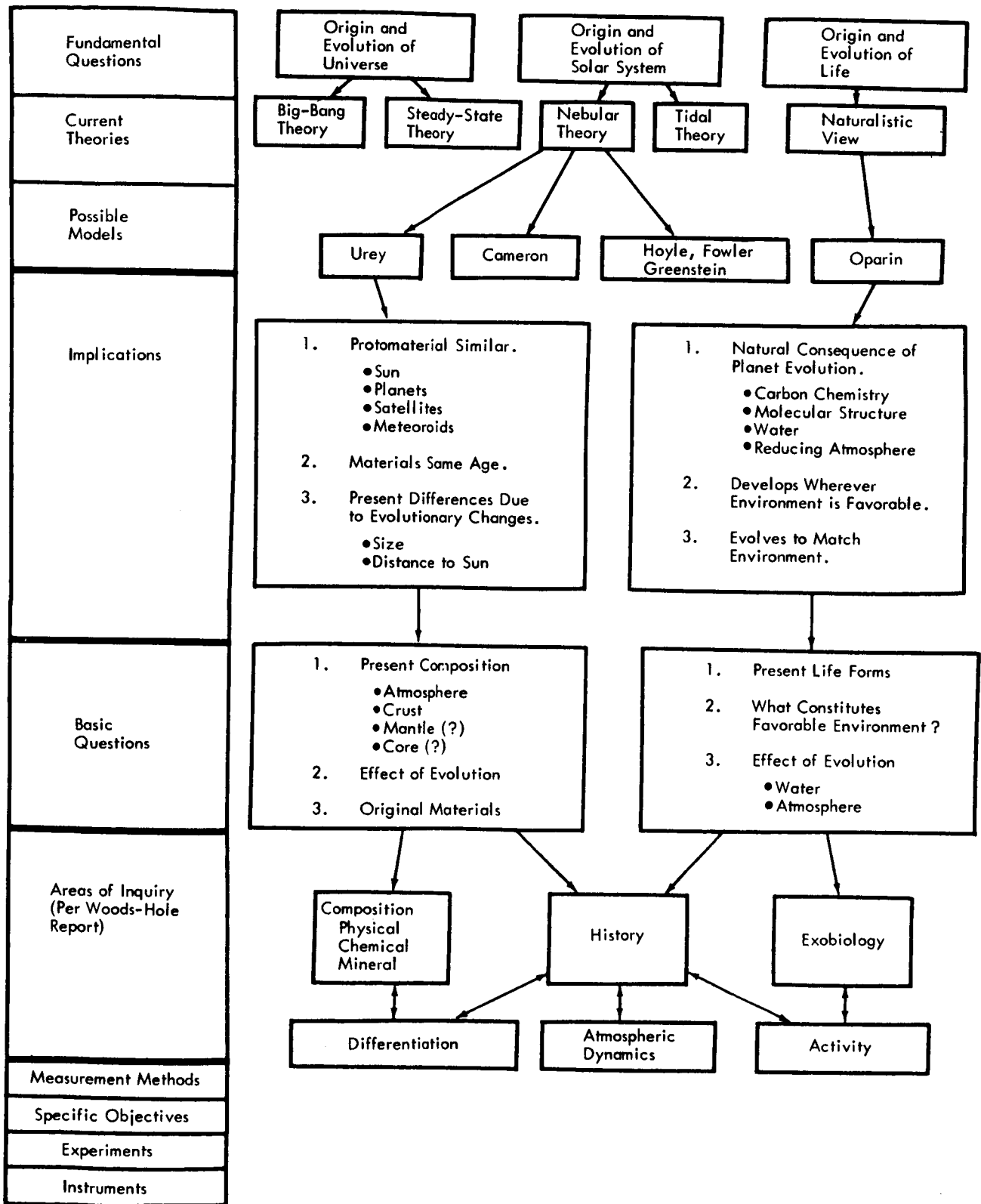


Figure 31: AREAS OF INQUIRY FOR PLANETARY EXPLORATION

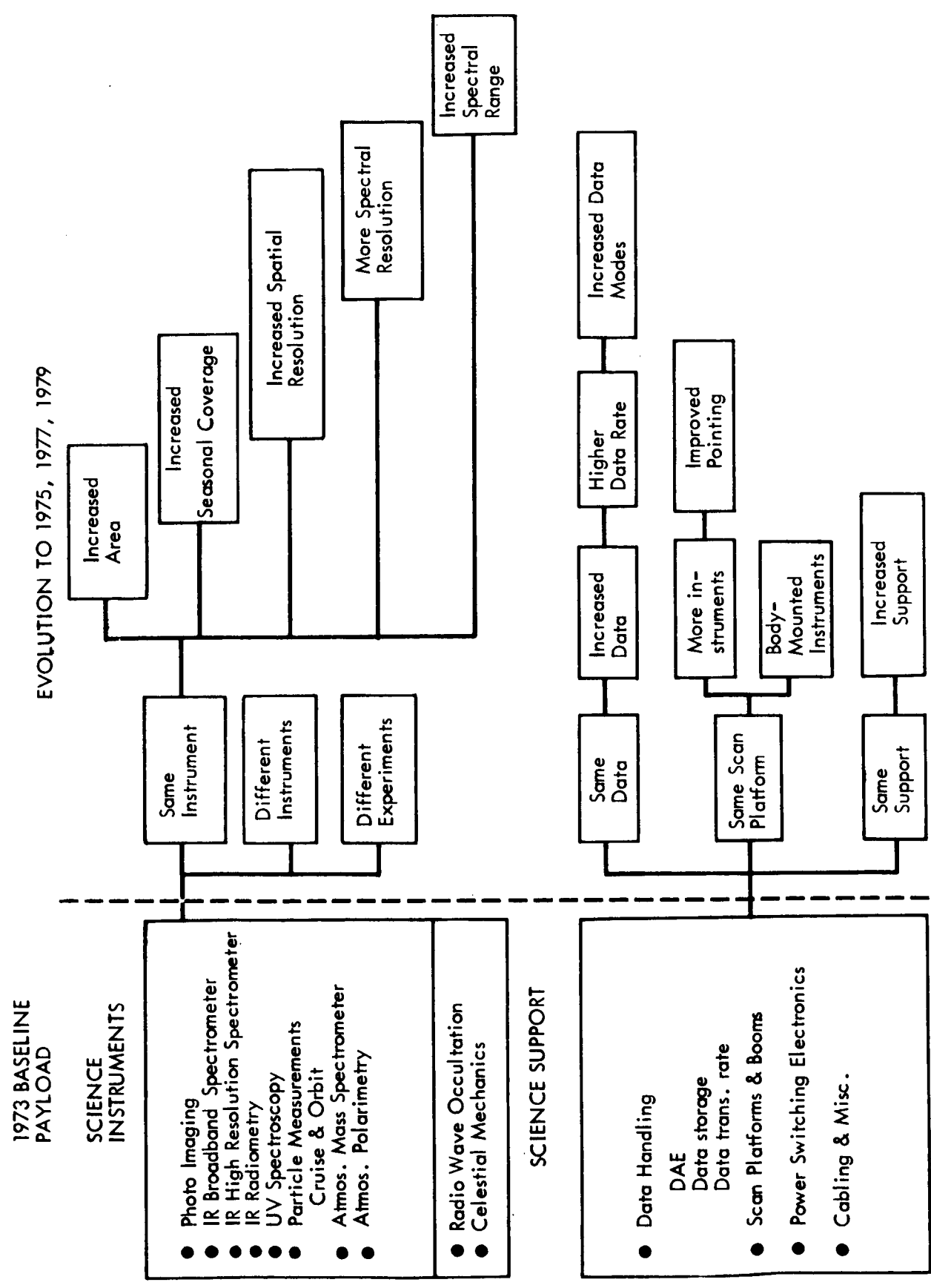


Figure 32 : SCIENCE EXPERIMENTS EVOLUTION TRENDS

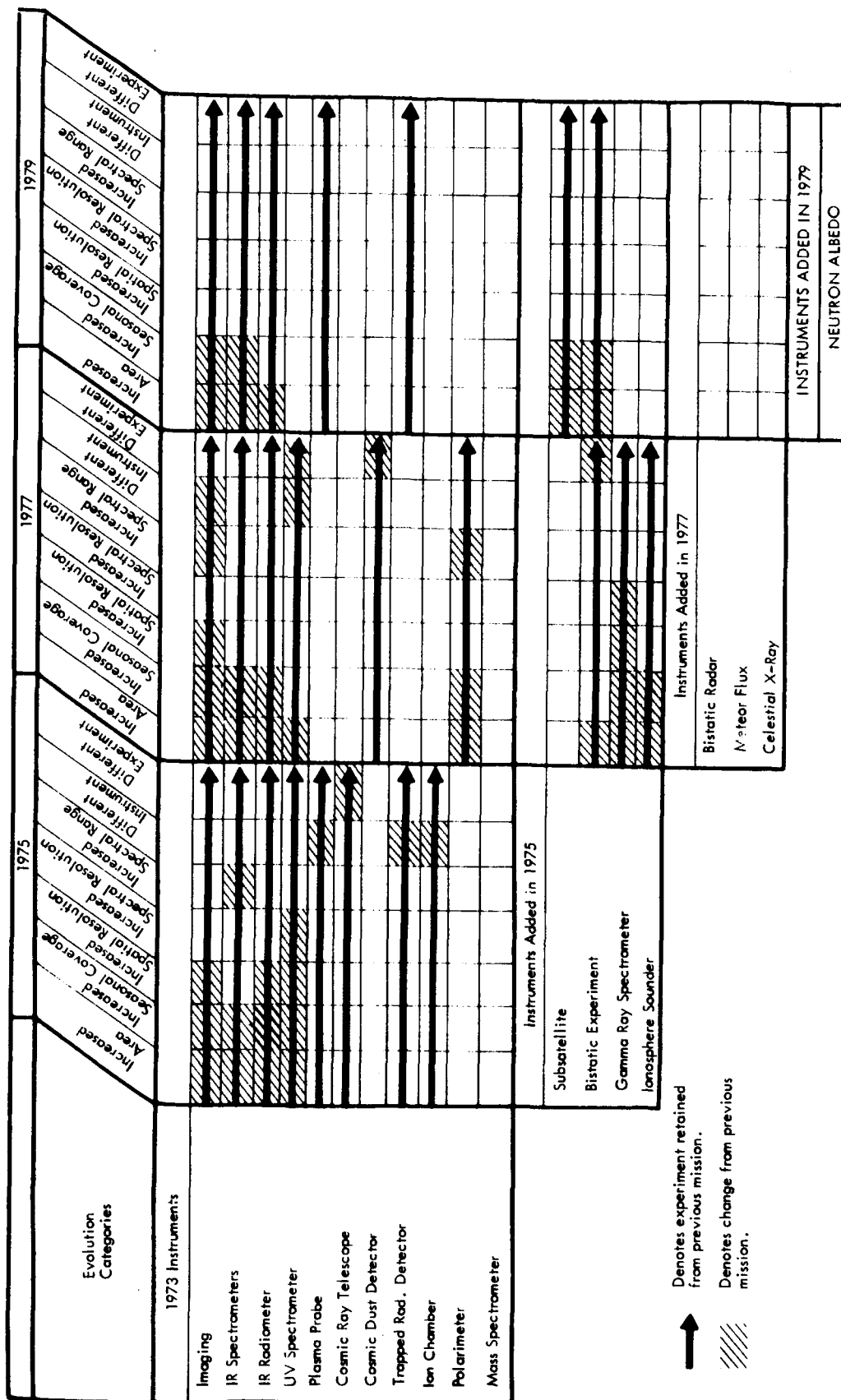


Figure 33: SCIENCE INSTRUMENTATION EVOLUTION

MISSION EXPERIMENT	1973			1975			1977			1979		
	Weight (lb)	Power (watts)	Data* (bits/sec)	Weight (lb)	Power (watts)	Data* (bits/sec)	Weight (lb)	Power (watts)	Data* (bits/sec)	Weight (lb)	Power (watts)	Data* (bits/sec)
Imaging	150	60	5×10^5	165	100	5×10^5	660	200	2.5×10^7	660	200	2.5×10^7
Broadband IR Spectrometer	25	5	1330/665									
High Resolution IR Spectrometer	30	14	1000	41	20	1500	41	20	1500	41	95	1500
IR Radiometer	20	6	2300	25	10	46,000	25	10	46,000	25	10	46,000
UV Spectrometer	30	12	4000	40	20	8,000						
Plasma Probe	10	3	23	(In Subsatellite)						(In Subsatellite)		
Cosmic Ray Telescope	8	3	46	8	3	46						
Cosmic Dust Detector	5	1	1.6				5	1	1.6			
Trapped Radiation Detector	3	1	15.5	(In Subsatellite)						(In Subsatellite)		
Ion Chamber	3	1	15.5	3	1	15.5						
Atmospheric Polarimeter	9	3	300				9	3	600			
Atmospheric Mass Spectrometer	8	10	60									
Subsatellite****				160	5	500				160	5	500
Bistatic Experiment**				40	2	10	45	2	10	40	2	10
Bistatic Radar***							20	(20)	4000			
Solar Occultation							30	25	4000			
Gamma Ray Spectrometer				15	8	50	20	12	200			
Neutron Albedo												
Meteor Flux							6	5	10	30	1	1
Celestial X-Ray							15	10	1500			
Ionosphere Sounder				25	10	100	25	10	100			
Earth Occultation***	0	0	0	0	0	0	0	0	0	0	0	0
Celestial Mechanics	0	0	0	0	0	0	0	0	0	0	0	0
Subtotal Science Instruments	301	119		522	179		901	298		956	313	
DAE And Misc.	70	44		70	44		100	54		100	54	
Total Science Payload	371	163		592	223		1001	352		1056	367	

* Average During Data Acquisition Period

** Spacecraft — Subsatellite Or Spacecraft — Spacecraft Experiment

*** Earth — Spacecraft Experiment

**** Includes Plasma Probe, Trapped Radiation Detector, Bistatic Radar and Magnetometer

 Bistatic Radar Off During Peak Power

Figure 34 : PROPOSED 1973-1975-1977-1979 SCIENCE PAYLOAD EXPERIMENTS AND CHARACTERISTICS

map the planet at medium resolution and identify several desirable landing sites for the capsule in subsequent missions. The high resolution obtained in the later missions would be for detailed investigation of selected areas that were found to be of scientific interest from the earlier missions.

The impact on the power subsystem due to the increased science payload is shown in Figure 35, where the power is seen to increase from 163 watts in 1973 to over 360 watts in 1977-1979. Very significant increases in science data bits also occur between the 1973 and 1979 missions. Figure 36 is included to provide a total picture of the impact of science payload evolution on the spacecraft. As shown, more science data requires increased data storage (at least for vidicons) and a higher data transmission rate to handle the increased data. This suggests a need for a larger antenna and higher transmitter power. The larger antenna, and a large film camera in 1977 and 1979, will result in tighter pointing accuracy requirements which in turn affects the attitude control stabilization requirements. Significant conclusions resulting from this task are:

- 1) The evolution from vidicons in 1973 to a large film camera in 1977 has the largest impact of any experiment on the spacecraft.
- 2) The evolution of the total science payload will increase its approximate weight from 400 pounds in 1973 to 600 pounds in 1975 and 1000 pounds in 1977 and 1979.
- 3) As data rate requirements increase, it will be necessary to spend development time and dollars to upgrade data recorders to match the capability of advanced vidicons.

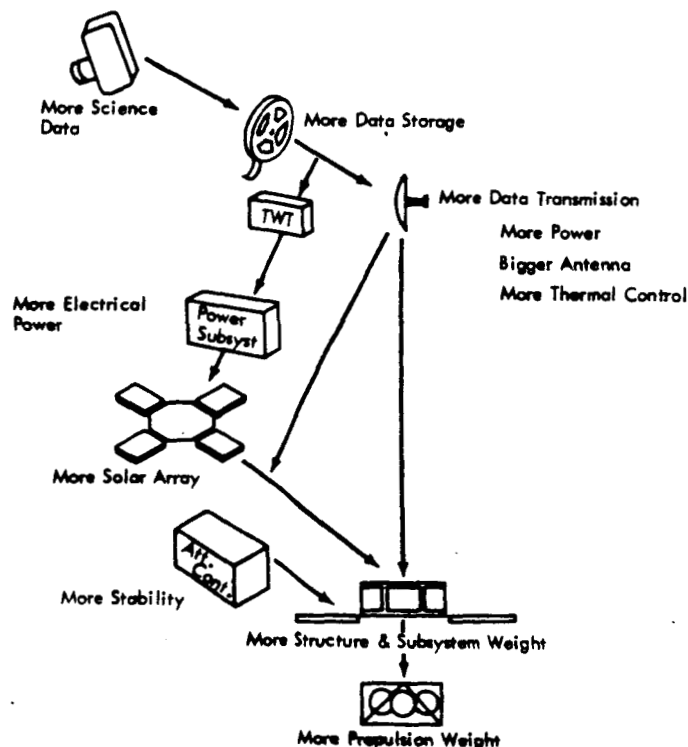


Figure 36: IMPACT ON SPACECRAFT OF INCREASED SCIENCE DATA

Mission Experiments	1973		1975		1977		1979	
	Power (watts)	Time of Oper. (min)	Power (watts)	Time of Oper. (min)	Power (watts)	Time of Oper. (min)	Power (watts)	Time of Oper. (min)
Imaging	60	-54 to -5	100	-54 to -5	200	-54 to -5	200	-54 to -5
Broadband IR Spectrometer	5	-59 to +10	20	-59 to +10	20	-59 to +10	95	-59 to +10
Hi Res. IR Spectrometer	14	-59 to 0	10	-59 to +10	10	-59 to +10	10	-59 to +10
IR Radiometer	6	-59 to +10	20	-59 to 0				
UV Spectrometer	12	-59 to 0	3	Continuous				
Plasma Probe	3	Continuous	3	Continuous	1	Continuous		
Cosmic Ray Telescope	3	Continuous						
Cosmic Dust Detector	1	Continuous	1	Continuous	3	-65 to +10		
Trap. Radiation Detector	1	Continuous						
Ion Chamber	1	Continuous						
Atmos. Polarimeter	3	-65 to +10	5	Continuous	2	Continuous	5	Continuous
Atmos. Mass Spectrometer	10	-45 to +10	2	Continuous	(20)	-80 to -20	2	Continuous
Subsatellite			8	Continuous	25	-10 to +50		
Bistatic Experiments					12	Continuous	1	Continuous
Bistatic Radar					12	Continuous		
Solar Occultation					5	Continuous		
Gamma Ray Spectrometer					10	Continuous		
Neutron Albedo					29	Continuous	29	Continuous
Meteor Flux					10	-59 to +10		
Celestial X-ray					5	-59 to 0		
Ionosphere Sounder	29	Continuous	10	Continuous				
DAE & Power Switch Elec.	10	-59 to +10						
Scan Platform No. 1	5	-59 to 0						
Scan Platform No. 2								
Standby Power, Lge. Camera								
Total Power Required	163		223		352		367	

△ Time in minutes before and after evening terminator; includes 30-minute warmup time.

2△ Alternate mode: UV spectrometer operates from 30 min before to 60 min after evening terminator

3△ Bistatic radar off during peak power.

Figure 35 : DETAILED SCIENCE PAYLOAD POWER REQUIREMENTS

5.4 PARTICULATE CONTAMINATION CONSIDERATIONS

The objective of this task was to determine a practical level of particulate contamination control (nonbiological) for the Voyager flight spacecraft and its subsystems.

Experience from other programs, such as Mariner and Lunar Orbiter, have shown that loose fragments can influence operational reliability. Evidence from the Mariner IV flight indicates that interference with the Canopus sensor and consequent loss of lock was caused by loose fragments that caused confusing reflections to be seen by the optical sensors.

Particulate contamination has long been of concern in the fabrication and assembly of subsystems where close-tolerance values are used. Contamination can cause short circuits and characteristic changes in electronic equipment.

Particulate contamination is controlled by conducting manufacturing and assembly operations in clean rooms where the air is filtered and temperature- and humidity-controlled. The end objective of this task was to show the impact of practical levels of particulate control on program costs.

This task was accomplished by first establishing allowable particulate contamination levels for the subsystems and components of the spacecraft. Proposed processes for achieving these levels were established. The facility requirements were established for accommodating the hardware and associated processing. These steps are summarized in Figure 37 for two of the seven critical subsystems. The contamination limits are given in an allowable number of particles of various sizes.

CRITICAL SUBSYSTEM	TYPE OF OPERATION	CONTAMINATION LIMITS	PROPOSED TECHNIQUE & EQUIPMENT REQUIRED	TYPE OF FACILITY REQUIRED	REMARKS & PRECAUTIONS
PROPULSION SUBSYSTEM	1. SUBASSEMBLY OF PARTS & COMPONENTS	NONE ABOVE: 5 μ METALLIC 25 μ NONMETALLIC (INTERNAL SURFACES)	TECHNIQUES AND EQUIPMENT SIMILAR TO THOSE DESCRIBED IN LUNAR ORBITER D2-100411-1. CLEAN AS ASSEMBLE. MAINTAIN CLEANLINESS LEVELS.	CLASS 100; CLEAN BENCHES OR BOOTH (DOWNFLOW); POSSIBLE USE OF HORIZONTAL FLOW ROOM.	WILL REQUIRE TRAINED PERSONNEL
	2. ASSEMBLY INTO SUBSYSTEMS				
	3. SUBSYSTEM QUAL. TESTS				
POWER SUBSYSTEM - SOLAR ARRAY ASSEMBLY	4. SPACECRAFT ASSEMBLY	$< 5 \mu$ NO LIMITS $5-25 \mu$ < 220 $28-100 \mu$ < 10 NONE ABOVE 100μ	CLASS M BAC 5703 NO CONTROL CLEAN AS ASSEMBLE MAINTAIN CLEANLINESS LEVELS	CLASS 100,000 ----- CLASS 100,000	OPERATION NO. 1- ENVIRONMENTAL CONTROLS BEING CALLED OUT ARE LESS STRINGENT THAN THOSE FOR STEPS NO. 4, 5, 6 & 7. REQUIRES EXTENSIVE USE OF TRAINED PERSONNEL.
	5. SUBSYSTEM INTEGRATION TESTS				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS				
	1. SUBASSEMBLY OF PARTS & COMPONENTS				
	2. ASSEMBLY INTO SUBSYSTEMS				
	3. SUBASSEMBLY QUAL. TESTS				
	4. SPACECRAFT ASSEMBLY				
	5. SUBSYSTEM INTEGRATION TESTS				
	6. SPACECRAFT ACCEPTANCE TESTS				
	7. SPACECRAFT CHECKOUT & LAUNCH OPERATIONS				

 Maximum Number of Particles Per Square Foot of Significant Surface

Figure 37: PARTICULATE CONTAMINATION REQUIREMENTS AND FACILITIES

Clean rooms are classified on a standard basis by Federal Standard 209a, so that it is possible to relate the established allowable levels to a particular type of clean room. Figure 38 shows characteristics of three types of clean rooms. The data shows that a Class 100,000 clean room contains no more than 100,000 particles 0.5 micron and larger per cubic foot and contains no more than 100 particles 10 microns and larger per cubic foot. This figure also implies that the largest cost would be for operation in a Class 100 room since it allows the least number of particles of any selected size. Referring back to Figure 37, the solar array assembly, when subjected to integration and acceptance testing, has contamination limits of:

Less than 100,000 particles	0.5 micron and larger
Less than 700 particles	5 microns and larger
20 particle	25 microns or larger

These limits lie close to the capability of a Class 100,000 clean room in Figure 38. On the other hand, the propulsion subsystem components have more stringent contamination limits and must be handled in a Class 100 area. This condition is achieved on a practical basis by use of Class 100 benches located within a Class 100,000 clean room.

A clean room does not automatically result in clean hardware. The key to cleanliness is strict adherence by personnel to processes and procedures while operating within a clean area. Personnel motivation will do much to ensure success in this vital part of the program.

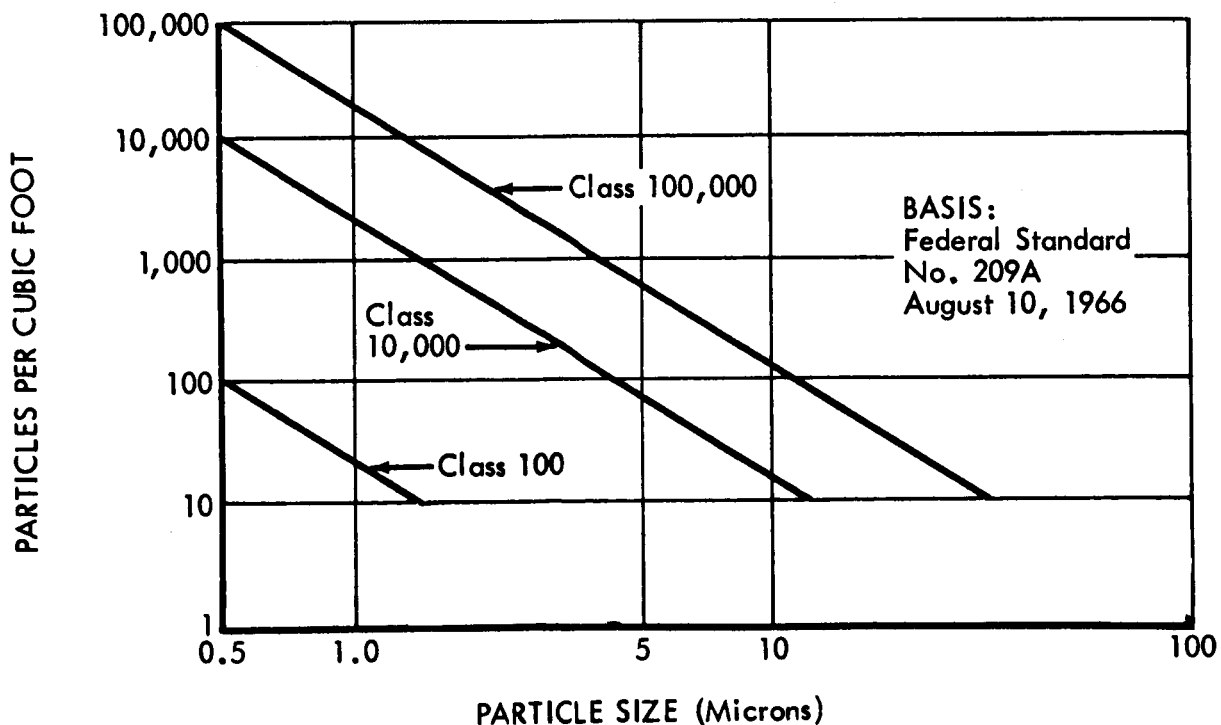


Figure 38: CLEAN ROOM CHARACTERISTICS

Costs of clean rooms vary with location and are dependent on type and size of clean room as shown in Figure 39. Here it is seen that costs vary from about \$40/square foot to \$120/square foot when considering horizontal or vertical laminar flow rooms. For the range of ceiling heights considered, doubling the height about doubles the cost per square foot. It is interesting to note that, for any particular type of clean room, (i.e., vertical flow), the initial construction cost is the same regardless of the class of clean room desired. The cleanliness level achieved is dependent on the number of personnel in the room, the activities performed, and the frequency of cleaning the items being processed.

Clean room operating cost predictions were based on an examination of the manufacture and assembly history on Lunar Orbiter. Examination of costs on that program showed that an overall increase of about 14% in manhours/vehicle resulted from cleaning operations.

Based on this task, the following conclusions were reached:

- 1) A practical level of particulate contamination control for Voyager requires a Class 100,000 clean room. Limited operation using Class 100 benches within the clean room are required.

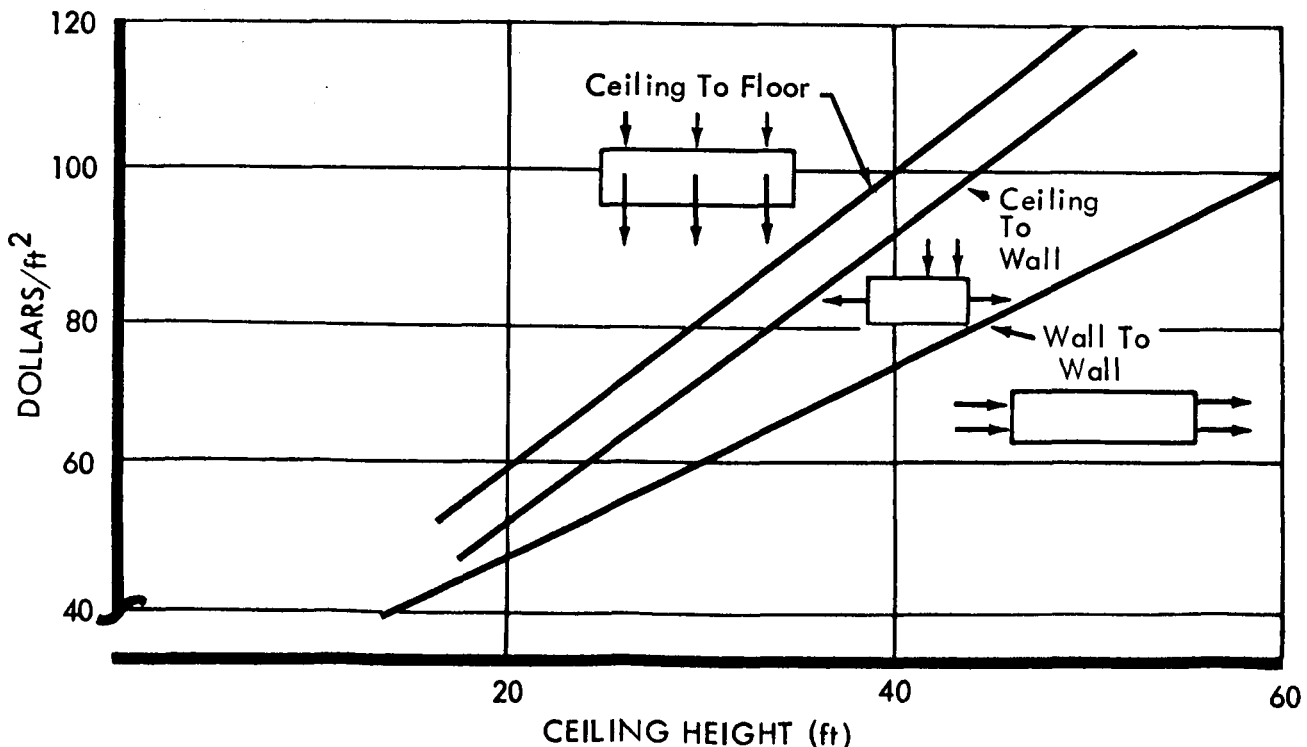


Figure 39: CLEAN ROOM FACILITY UNIT COST

- 2) A wall-to-wall laminar flow clean room is recommended for minimum cost.
- 3) Personnel motivation has been proven to be a key factor in maintaining hardware cleanliness based on Lunar Orbiter experience.

5.5 PHOTOIMAGING CONSIDERATIONS

The objectives of this task were to:

- 1) Compare vidicon, electrostatic tape, and film photoimaging systems for performing the Voyager imaging experiment.
- 2) Determine the impact of the photoimaging system on the flight spacecraft.
- 3) Determine the highest achievable resolution as a function of weight for each of the three systems.

The requirements for photoimaging are derived from the basic scientific objectives for planetary exploration. These objectives include the understanding of the evolution of the solar system and the search for extraterrestrial life. The science payload evolution task, discussed earlier in this report, traced the path from these top-level objectives down to the conduct of imaging experiments for Voyager Mars. The imaging experiment contributes to meeting the above objectives during Mars orbital operations by:

- Locating evidence of favorable environment for life.
- Studying planet surface composition.
- Studying cloud patterns and storm dynamics.
- Providing a map for geologically-oriented data from other science experiments.
- Studying effects of seasonal variations.

The requirements for resolution and coverage to satisfy the above objectives were developed considering the science payload evolution task and are summarized below:

Coverage at Medium Resolution

- 75% of the planet, 60°S to 40°N - 1973;
- Entire planet - 1975-1977-1979;
- 100 to 300 meters resolution - 1973-1975-1977-1979

Coverage at High Resolution

- 0.1% of planet surface - 1973
- 1% of surface - 1975-1977-1979
- 1 to 10 meters resolution - 1973-1975-1977-1979

These requirements are similar to those of the Lunar Orbiter for the Moon missions. Resolution on the order of 1 meter will enable the detection of objects on the surface of Mars comparable to the size of large Earth mammals. Ground resolution less than 0.5 meter probably cannot be achieved because of atmospheric scattering, turbulence, and haze.

Three types of imaging systems were evaluated in detail in this study. They are the vidicon (TV) system, the electrostatic tape system, and a silver halide film camera system. The advantages and disadvantages of these three concepts are shown in Figure 40.

Film systems have been used for both Earth orbital and the Lunar Orbiter photographic missions. The key feature of this type of system is that it can store a very large quantity of data at an extremely high packing density. It is classified as a high resolution system.

The vidicon has seen more use than the other types in Earth orbital, lunar, and planetary missions. The system is very simple and has demonstrated reliable performance under planetary space environment conditions. Due to the low persistence of the image on the sensor, the vidicon system incorporates a tape recorder to store the imaging data for later transmission to Earth.

The electrostatic tape camera (ESTC), developed for Nimbus, combines the function of the vidicon and tape recorder into one system. Photoconductive and electric conductive materials are deposited on a tape. Exposure consists of simultaneously exposing the tape to an optical input and to a raster scan of electrons. Readout is accomplished by scanning the tape with a focused electron beam.

The three imaging systems concepts were evaluated on the basis of ground coverage and resolution, weight, size, smear and image motion compensation, format, focal length, aperture, exposure time, and optics design. Parametric data were developed for all of these considerations which allows imaging system as well as spacecraft system-level tradeoffs to be conducted. As an example, Figure 41 shows the expected improvement trends for the three systems plotted in terms of surface resolution and the width of surface coverage. For a selected ground resolution, surface coverage improvements can be obtained by increasing the sensor size or resolution. The vidicon appears to have the most growth capability; however, even by 1975, its predicted growth does not match the resolution/coverage capability of current film systems.

Another example of parametric data comparing the three types of imaging systems is shown in Figure 42 where the ground resolution attainable for various concepts is plotted versus weight. Beyond 300 pounds, the curves are estimates of potential growth. The dashed lines for the vidicon and electrostatic tape represent the theoretical maximum resolution attainable for these systems. Practical systems will operate at 30% to 50% of these values as represented by the shaded areas. This figure indicates that the vidicon is lighter in weight for the medium or low resolutions and that the film camera is lighter in weight for resolutions higher than about 30 meters.

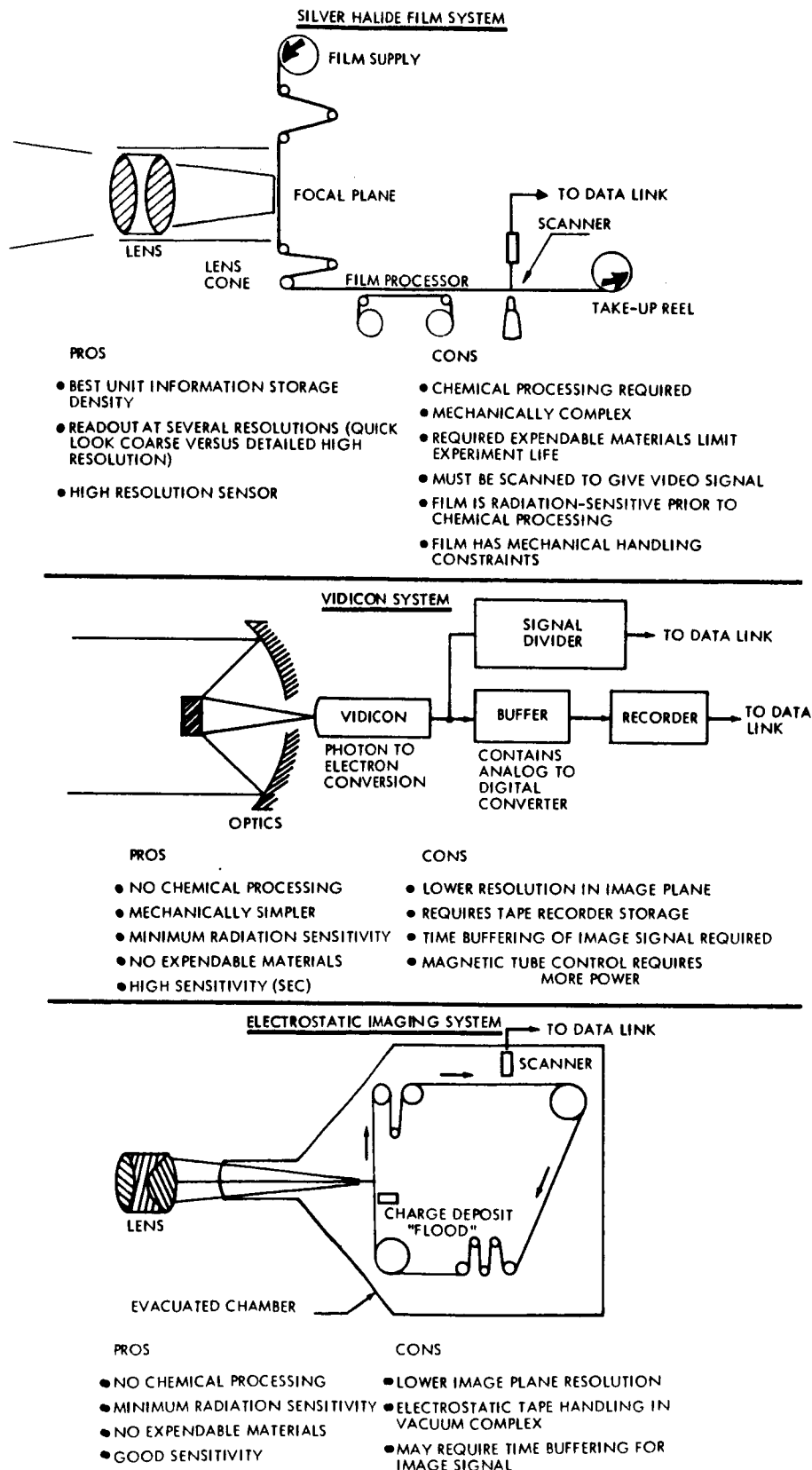
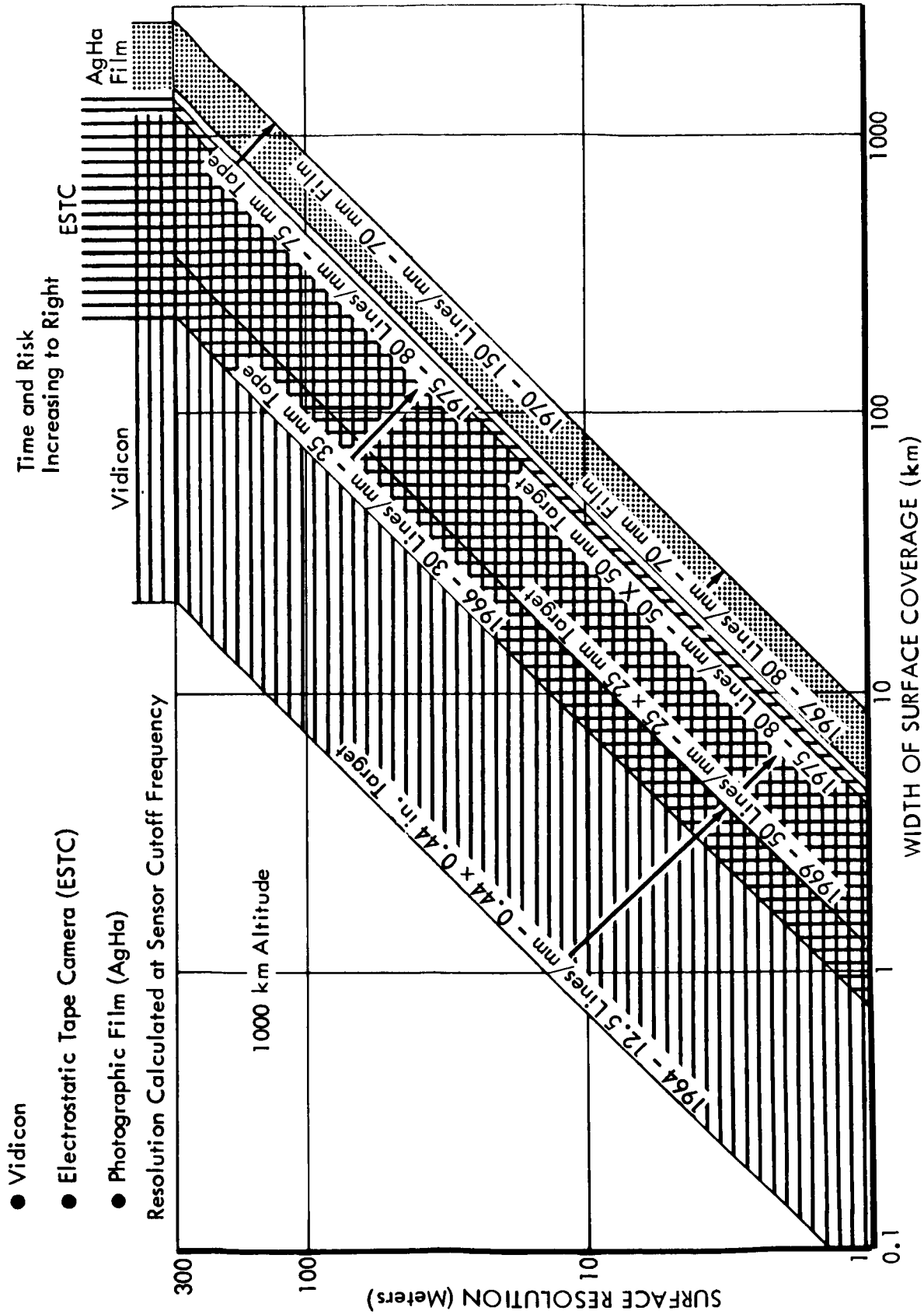


Figure 40: CANDIDATE PHOTOIMAGING SYSTEMS FOR THE VOYAGER SPACECRAFT



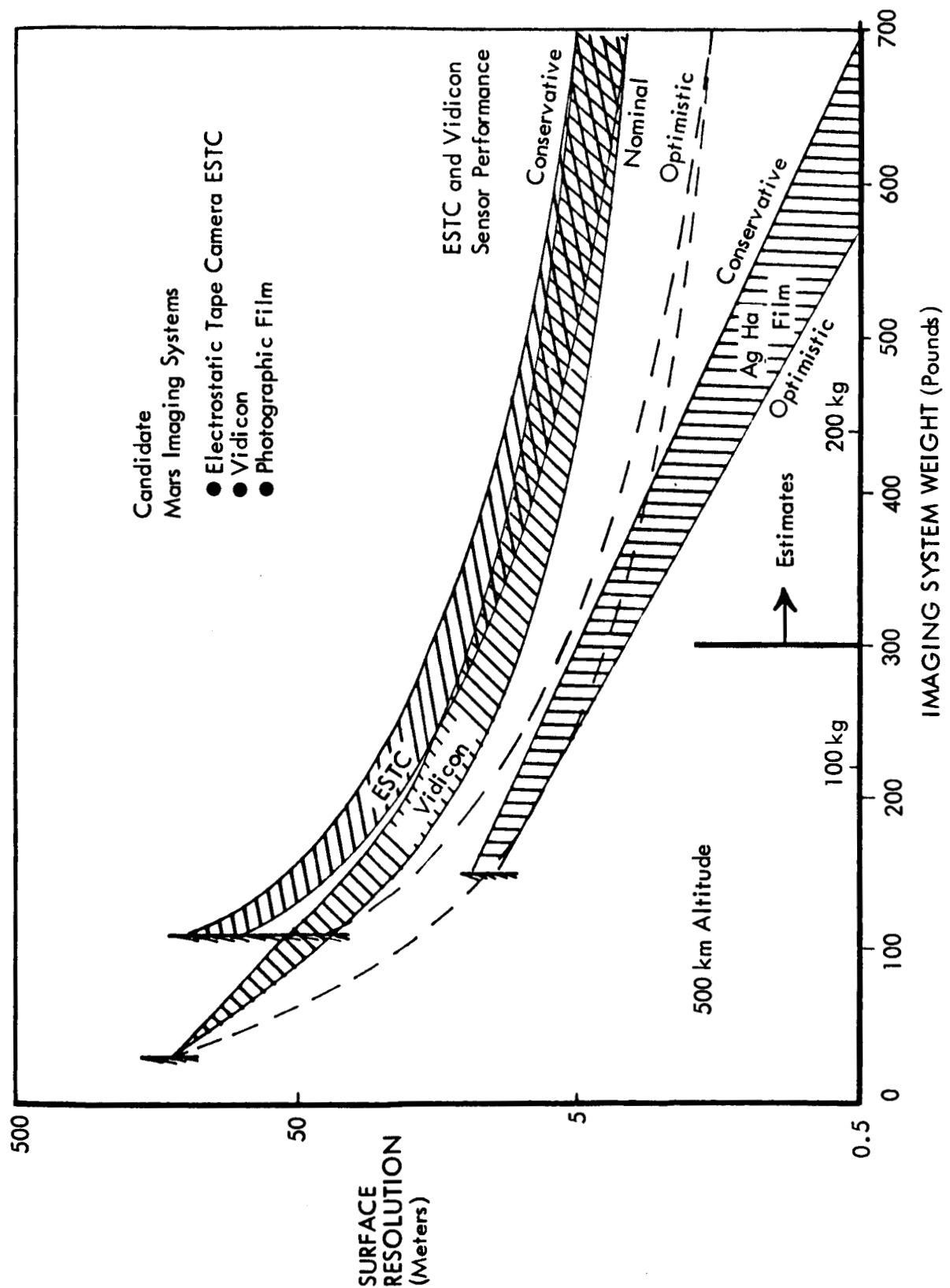


Figure 42: WEIGHT/RESOLUTION TREND FOR CANDIDATE IMAGING SYSTEMS

This discussion has compared the three systems on the basis of imaging system weight only. The total impact on the spacecraft must be taken into account before final conclusions can be drawn. The spacecraft must support the imaging system by providing mechanisms, support structure, control functions, and electric power. Any addition of weight must also be reflected in terms of increased propellant to carry the added weight into orbit. It takes about 1 pound of propellant for each added pound of spacecraft weight. These trends are shown in Figure 43. The upper left curve (43a) shows how provision requirements will vary with desired surface resolution. The impact on the power subsystem versus resolution as a function of the average data return rate is shown on the upper right (43b) of this figure. The amount of data depends upon both the system resolution and ground coverage.

Combining these two curves results in a payload versus resolution curve in the lower left (43c) of this figure. Now, adding the propellant (Figure 43d) to handle the added payload results in the total spacecraft penalty as a function of surface resolution.

The requirements for imaging, stated earlier, were in terms of both ground coverage and surface resolution. Coverage can be traded for resolution, i.e., for the same total data a small area can be covered at high resolution, or a larger area can be covered at medium resolution. This is indicated on the left side of Figure 44 where the percentage of the Martian surface covered in a 180-day orbital mission is plotted versus surface resolution for three different data rates. For example, entering the ordinate at 10-meters resolution and reading to the left, it is seen that surface coverage goes down as data rate (and therefore total data bits) is decreased. For a constant surface coverage of 0.1% of the planet, decreasing total data bits results in degradation in resolution. The right side of this figure shows how the weight of the imaging system and its required support varies with resolution requirements. Again, various data rates are given as parameters. The total figure may be read as follows: For a desired surface resolution of 4 meters and ground coverage of 1%, a data rate of 320,000 bps will result in a system weight of about 1200 pounds. At the same 4-meters resolution, but requiring only 0.1% coverage of the planet, a data rate of 50,000 bps results in a system weight of about 1000 pounds.

Significant conclusions resulting from this task are:

- 1) High resolution (1 to 10 meters) photoimaging coverage of a small portion of Mars surface (0.1 to 1%) and medium resolution (150 to 300 meters) coverage of most of the planet's surface will satisfy the currently understood scientific objectives.
- 2) The vidicon, electrostatic tape, and film camera can each satisfy the medium resolution and coverage requirements.
- 3) The film camera provides the highest resolution of any of the three systems and at the least impact on spacecraft weight for an orbit of 500-km periapsis altitude.

Based upon this study, a vidicon system is recommended for the 1973 and 1975 missions and a large film camera system is recommended for the 1977 and 1979 missions.

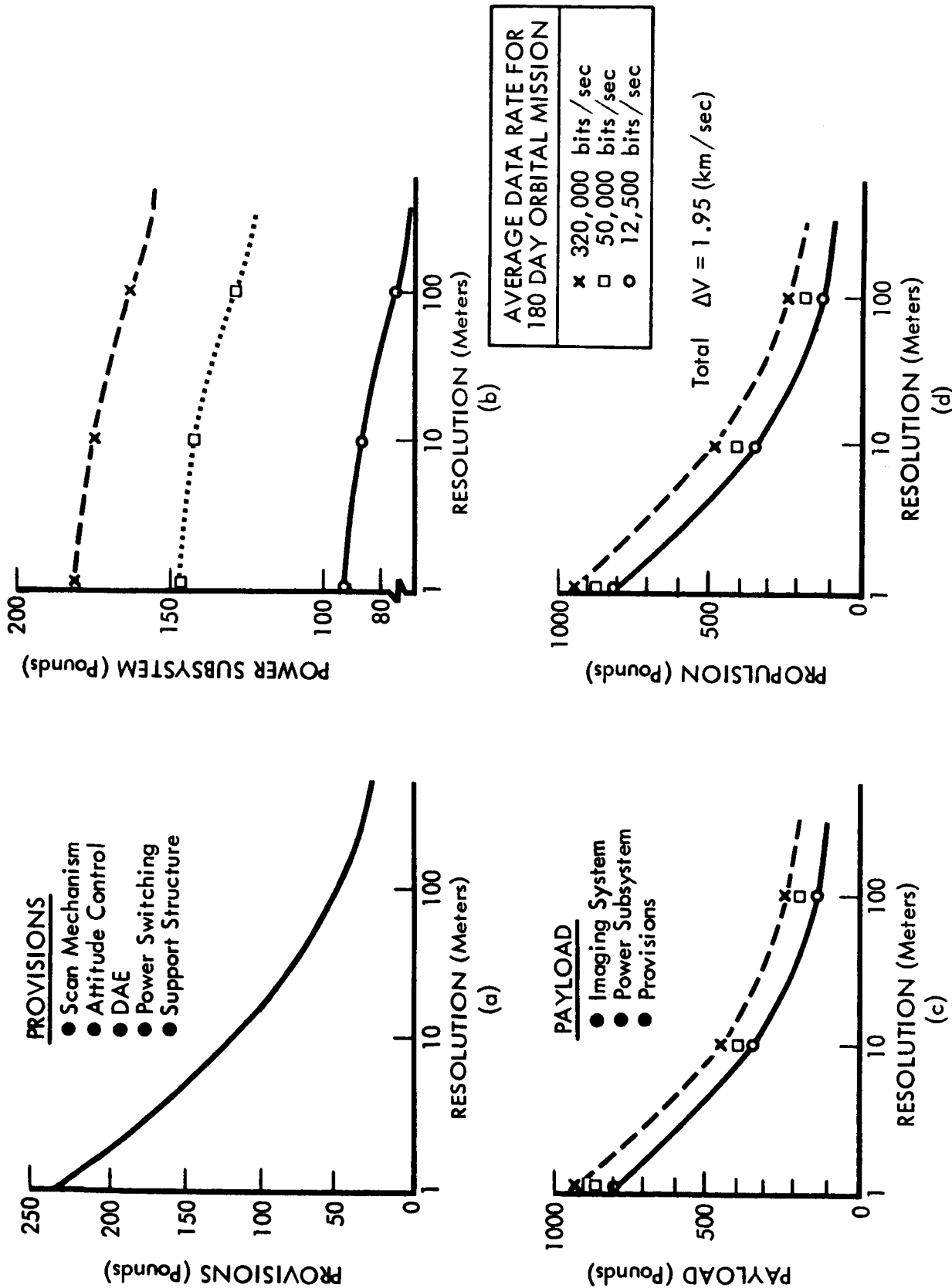


Figure 43 : ESTIMATED IMPACT OF IMAGING SYSTEM RESOLUTION ON SPACECRAFT SUBSYSTEM WEIGHTS

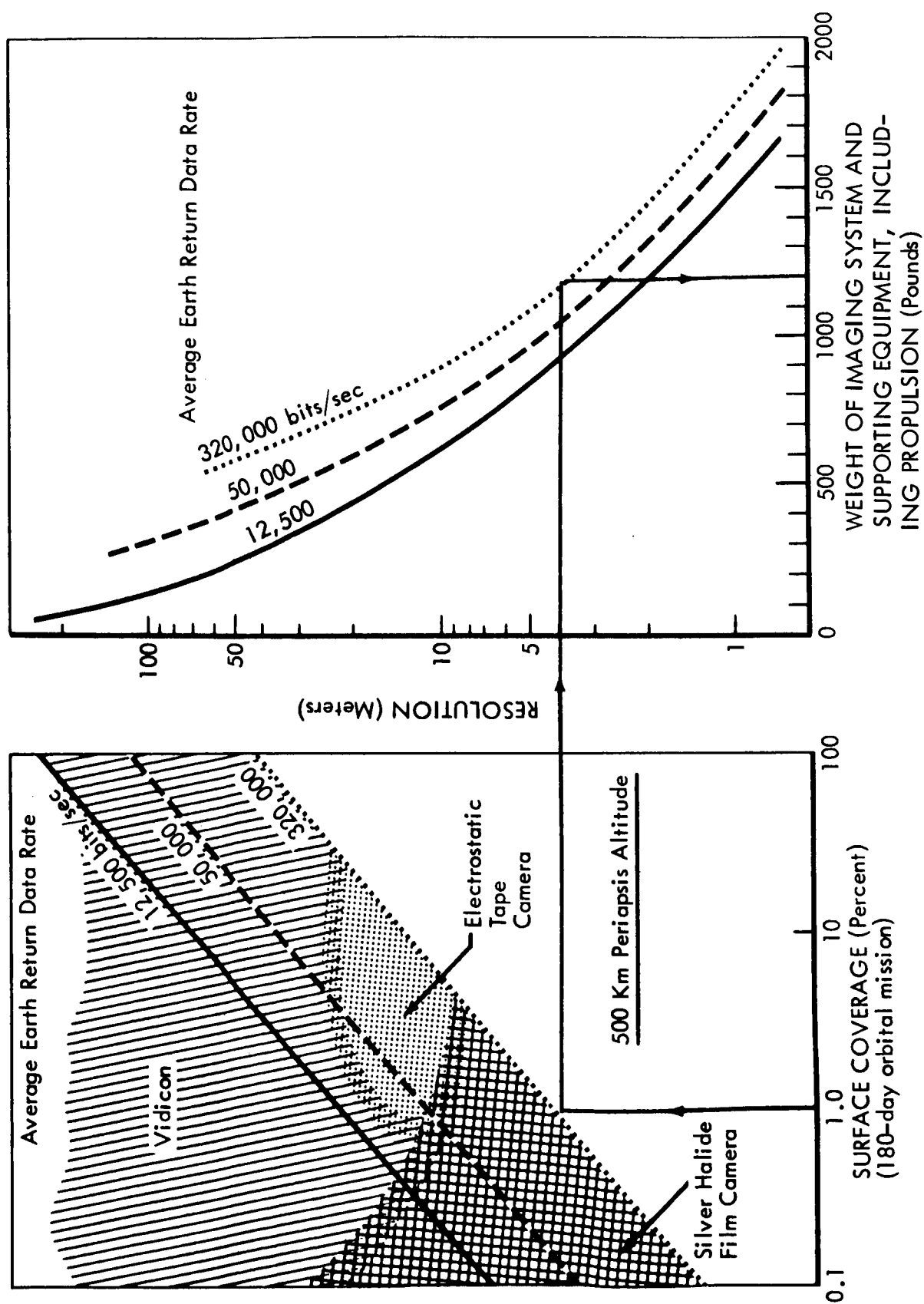


Figure 44: ESTIMATED IMPACT OF IMAGING SYSTEM HIGH RESOLUTION AND COVERAGE ON FLIGHT SPACECRAFT WEIGHT