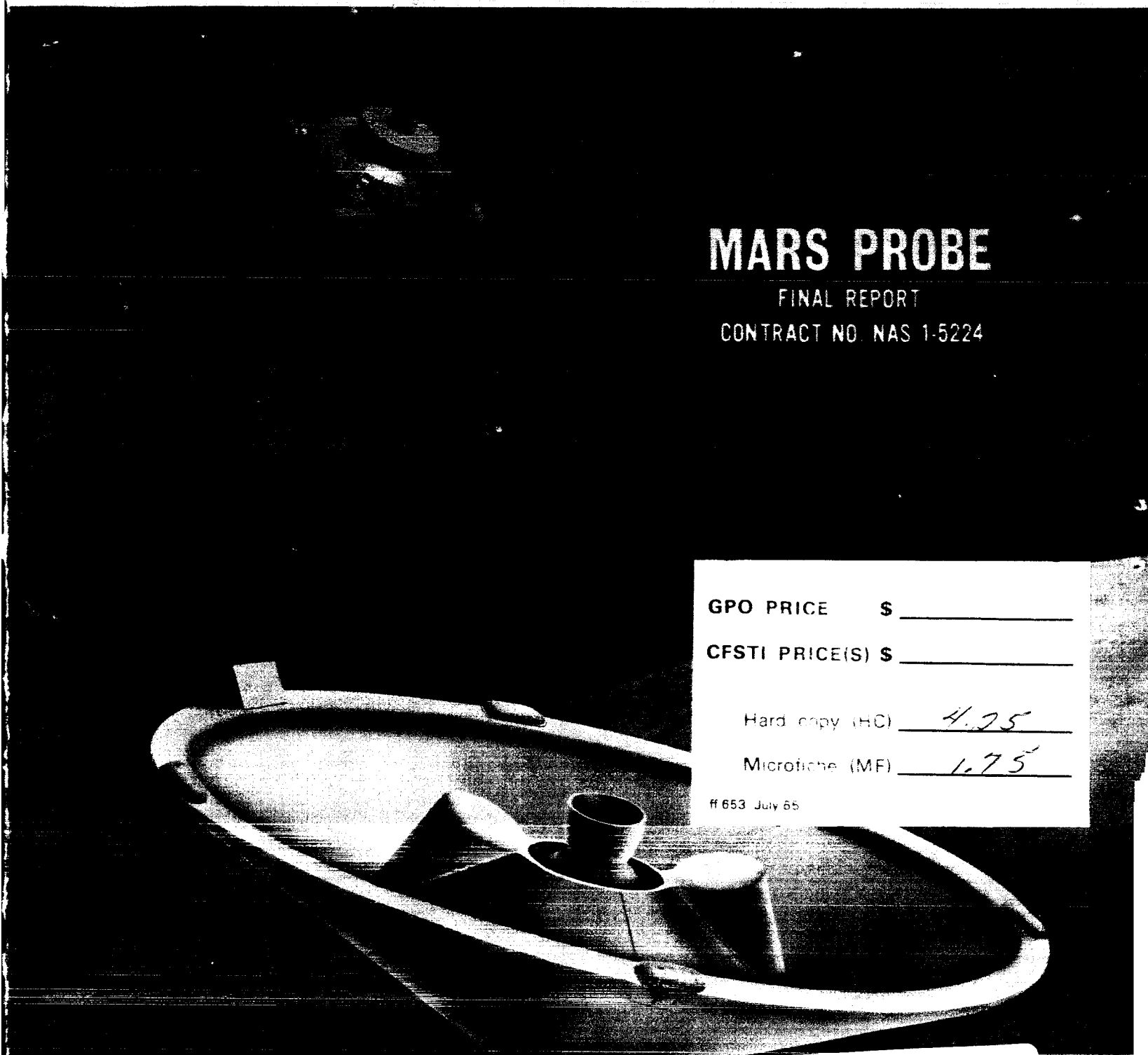
VOLUME II

BOOK 1

PROBE/LANDER, ENTRY FROM THE APPROACH TRAJECTORY

SYSTEM DESIGN

NASA CR - 66129



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BOOK INDEX

VOLUME	I	SUMMARY
VOLUME	II	PROBE/LANDER, ENTRY FROM THE APPROACH TRAJECTORY
Book	1	System Design
Book	2	Mission and System Specifications
VOLUME	III	PROBE, ENTRY FROM ORBIT
Book	1	System Design
Book	2	Mission, System and Component Specifications
Book	3	Development Test Programs
VOLUME	IV	STERILIZATION
VOLUME	V	SUBSYSTEM AND TECHNICAL ANALYSES
Book	1	Trajectory Analysis
Book	2	Aeromechanics and Thermal Control
Book	3	Telecommunications, Radar Systems and Power
Book	4	Instrumentation
Book	5	Attitude Control and Propulsion
Book	6	Mechanical Subsystems

NASA CR-66129

COMPARATIVE STUDIES OF CONCEPTUAL DESIGN AND QUALIFICATION PROCEDURES FOR A MARS PROBE/LANDER

FINAL REPORT

VOLUME II PROBE/LANDER, ENTRY FROM THE APPROACH TRAJECTORY

Book 1 SYSTEM DESIGN

Prepared by

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Prepared for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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PREFACE

The results of Mars Probe/Lander studies, conducted over a 10-month period for Langley Research Center, NASA, are presented in detail in this report. Under the original contract work statement, studies were directed toward a direct entry mission concept, consistent with the use of the Saturn IB-Centaur Launch Vehicle, wherein the landing capsule is separated from the spacecraft on the interplanetary approach trajectory, some 10 to 12 days before planet encounter. The primary objectives of this mission were atmospheric sampling by the probe/lander during entry and terrain and atmosphere physical composition measurement for a period of about 1 day after landing.

Studies for this mission were predicated on the assumption that the atmosphere of Mars could be described as being within the range specified by, NASA Mars Model Atmospheres 1, 2, 3 and a Terminal Descent Atmosphere of the document NASA TM-D2525. These models describe the surface pressure as being between 10 and 40 mb. For this surface pressure range a payload of moderate size can be landed on the planet's surface if the entry angle is restricted to be less than about 45 degrees.

Midway during the course of the study, it was discovered by Mariner IV that the pressure at the surface of the planet is in the 4 to 10 mb range, a range much lower than previously thought to be the case. The results of the study were re-examined at this point. It was found that retention of the direct entry mission mode would require much shallower entry angles to achieve the same payloads previously attained at the higher entry angles of the higher surface pressure model atmospheres. The achievement of shallow entry angles (on the order of 20 degrees), in turn, required sophisticated capsule terminal guidance, and a sizeable capsule propulsion system to apply a velocity correction close to the planet, after the final terminal navigation measurements.

Faced with these facts, NASA/LRC decided that the direct entry from the approach trajectory mission mode should be compared with the entry from orbit mode under the assumption that the Saturn 5 Launch Vehicle would be available. Entry of the flight capsule from orbit allows the shallow angle entry (together with low entry velocity) necessary to permit higher values of M/C_DA, and hence entry weight in the attenuated atmosphere.

It was also decided by LRC to eliminate the landing portion of the mission in favor of a descent payload having greater data-gathering capacity, including television and penetrometers. In both the direct entry and the entry from orbit cases, ballistic atmospheric retardation was the only retardation means considered as apecifically required by the contract work statement.

Four months had elapsed at the time the study ground rules were changed. After this point the study continued for an additional five months, during which

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period a new design for the substantially changed conditions was evolved. For this design, qualification test programs for selected subsystems were studied. Sterilization studies were included in the program from the start and, based on the development of a fundamental approach to the sterilization problem, these efforts were expanded in the second half of the study.

The organization of this report reflects the circumstance that two essentially different mission modes were studied -- the first being the entry from the approach trajectory mission mode and the other being the entry from orbit mission mode -- from which two designs were evolved. The report organization is as follows:

Volume I, Summary, summarizes the entire study for both mission modes.

Volume II reports on the results of the first part of the study. This volume is titled Probe/Lander, Entry from the Approach Trajectory. It is divided into two books, Book 1 and Book 2. Book 1 is titled System Design and presents a discursive summary of the entry from the approach trajectory system as it had evolved up to the point where the mission mode was changed. Book 2, titled Mission and System Specifications, presents, in formal fashion, specifications for the system. It should be understood, however, that the study for this mission mode was not carried through to completion and many of the design selections are subject to further tradeoff analysis.

Volume III is composed of three books which summarize the results of the entry from orbit studies. Books 1 and 2 are organized in the same fashion as the books of Volume II, except that Book 2 of Volume III presents component specifications as well. Book 3 is titled Development Test Programs and presents, for selected subsystems, a discussion of technology status, test requirements and plans. This Book is intended to satisfy the study and reporting requirements concerning qualification studies, but the selected title is believed to describe more accurately the study emphasis desired by LRC.

Volume IV presents Sterilization results. This information is presented separately because of its potential utilization as a more fundamental reference document.

Volume V presents, in six separate books, Subsystem and Technical Analyses. In order (from Book 1 to Book 6) they are:

Trajectory Analysis
Aeromechanics and Thermal Control
Telecommunications, Radar Systems and Power
Instrumentation
Attitude Control and Propulsion
Mechanical Subsystems

Most of the books of Volume V are divided into separate discussions of the two mission modes. Table of Contents for each book clearly shows its organization.

CONTENTS

1.0 Introduction 1

2.0 Mission Objectives and Constraints 2

2.1 Mission Objectives 2

2.1.1 Preseparation 2
2.1.2 Separation 2
2.1.3 Flight Capsule Separation to Entry 2
2.1.4 Entry to Parachute Deployment 2
2.1.5 Parachute Deployment to Impact 3
2.1.6 Surface Operations 3

2.2 Study Ground Rules 4

2.2.1 Technology Cutoff Date 4
2.2.2 Launch Vehicle 4
2.2.3 Atmospheric Models 4
2.2.4 Aerodynamic Shapes 10
2.2.5 Impact Attenuation System 10
2.2.6 Descent Subsystem 10
2.2.7 Sterilization 15
2.2.8 Deep Space Network 15
2.2.9 Launch Opportunities and Flight Capsule Separation 15
2.2.10 Flight Spacecraft Characteristics 16
2.2.11 Guidance 16
2.2.12 Landed Capsule Surface Mission Lifetime 16
2.2.13 Landing Site 16
2.2.14 Structures 17

3.0 System Characteristics 18

3.1 System Description 18
3.2 Design Description 19

3.2.1 Reference Design 19
3.2.2 Alternate Design - Flotation Sphere Landed Capsule 33

3.3 Operational Sequence 38
3.4 Weight and Balance 42

3.4.1 Oblate Spheroid Landed Capsule Design 42
3.4.2 Flotation Sphere Landed Capsule Design 48

CONTENTS (Cont'd)

4.0 Subsystem Characteristics 50

4.1 Command and Programming 50

4.1.1 Objectives and Requirements 50
4.1.2 Subsystem Mechanization 50

4.2 Scientific and Engineering Instrumentation 54

4.2.1 Objectives and Requirements 54
4.2.2 Subsystem Design 56
4.2.3 Subsystem Performance 61
4.2.4 Subsystem Mechanization 64

4.3 Telecommunication 65

4.3.1 Requirements and Constraints 65
4.3.2 Subsystem Design 68
4.3.3 Subsystem Operation 72
4.3.4 Subsystem Mechanization 74

4.4 Power Subsystem 74

4.4.1 Design Requirements 74
4.4.2 Subsystem Design 76
4.4.3 Subsystem Mechanization 80

4.5 Propulsion 82

4.5.1 Objectives and Requirements 82
4.5.2 Subsystem Design 82
4.5.3 Subsystem Performance 83

4.6 Attitude Control and Thrust Vector Control 83

4.6.1 Objectives and Requirements 83
4.6.2 Subsystem Design 84
4.6.3 Subsystem Performance 85
4.6.4 Subsystem Mechanization 86

4.7 Parachute 89

4.7.1 Objective and Requirements 89
4.7.2 Subsystem Design 89
4.7.3 Subsystem Mechanization 95

CONTENTS (Cont'd)

4.8 Impact Attenuation Subsystem 98

4.8.1 Objectives and Requirements 98
4.8.2 Subsystem Design 98
4.8.3 Subsystem Performance 99
4.8.4 Subsystem Mechanization 99

4.9 Entry Shell 100

4.9.1 Design Requirements 100
4.9.2 Structure 102
4.9.3 Heat Shield 103

4.10 Thermal Control 105

4.10.1 Objectives and Requirements 105
4.10.2 Design and Performance 108

5.0 Entry Vehicle Shell Selection 114

5.1 Shell Mission Utilization Concepts 114
5.2 Entry Shell Configurations 115

5.2.1 Modified Apollo 115
5.2.2 Tension Shell 121

5.3 Basis for Shell Shape Comparison 124
5.4 Method for Determination of Residual Weights 127

5.4.1 M/C_DA Determination 133
5.4.2 Heat Shield and Structure Design Criteria 135
5.4.3 Heat Shield and Structure Weight Determination 138
5.4.4 Residual Weight - Vehicle Diameter Comparison 138
5.4.5 Entry Vehicle Diameter Selection 138

5.5 Comparison of Residual Weights 150

5.5.1 Comparison at 197-inch Diameter 150
5.5.2 Comparison of Residual Weights for Three Shell Concepts 150

CONTENTS (Cont'd)

5.6 Comparison of Shell Shapes for Other Selection Criteria 161

5.6.1 Residual Weight 161
5.6.2 Aerodynamic Stability 162
5.6.3 Center of Gravity and Packaging Versatility 162
5.6.4 Design Confidence 162
5.6.5 Test Requirements 162
5.6.6 Manufacturing Ease 162
5.6.7 Payload Integration 162
5.6.8 External Interfaces 163

5.7 Available Landed Capsule Weight 163

6.0 Landed Capsule Tradeoffs 166

6.1 Introduction 166
6.2 Landed Capsule Configuration Comparison 166

6.2.1 Review of Landed Capsule Concepts 166
6.2.2 Candidate Landed Capsule Evaluation 171

6.3 Payload Synthesis 179

6.3.1 Design Guidelines 179
6.3.2 Entry and Descent Payload Synthesis 180
6.3.3 Post Impact Payload Synthesis 196
6.3.4 Oblate Spheroid Landed Capsule Synthesis 213
6.3.5 Flotation Sphere Landed Capsule Synthesis 228

6.4 Landed Capsule Selection 231

7.0 Attitude Control System Tradeoffs 235

7.1 System Requirements/Design Guidelines 235

7.1.1 Communication Lead Time 235
7.1.2 Landing Point and Entry Angle Dispersion 235
7.1.3 Trajectory Analysis 236
7.1.4 Summary 236

CONTENTS (Concl'd)

7.2 Candidate Techniques by Mission Phase 238

7.2.1 Orient Separated Vehicle to Thrust Attitude 238
7.2.2 Maintain Attitude During Thrust 238
7.2.3 Reorient for Proper Entry Attitude 238
7.2.4 Maintain Attitude During Cruise 238

7.3 Comparison of Candidate Systems 238

7.3.1 Spin Only System 238
7.3.2 Active ACS with Spin 239
7.3.3 Active ACS 239
7.3.4 Active ACS with Engine Gimbal and Spin 239

7.4 Selected Approach 239
7.5 Terminal Guidance 239
7.6 Flight Spacecraft Separation Maneuver 240
7.7 Despin 241

8.0 Failure Mode Analysis 247

8.1 Failure Mode Summary 247
8.2 Failure Before Separation 249
8.3 Separation Failure 249
8.4 Attitude Control System Failure 249
8.5 Propulsion Failure 250
8.6 Entry Shell Failure 250
8.7 Parachute Failure 251
8.8 Communication Failure 251
8.9 Instrumentation Failure 252
8.10 Impact/Re-Erection Failure 252

9.0 Launch Period Analysis 253

9.1 Constraining Parameters 253
9.2 1971 Launch Opportunity 255
9.3 1973 Launch Opportunity 255
9.4 1975 Launch Opportunity 260
9.5 Launch Period Analysis Summary and Conclusions 265

ILLUSTRATIONS

Figure 1 Flight Capsule Operational Diagram and Terminology..... xxiv

2 Capsule System Functional Diagram Terminology xxv

3 Voyager Design Studies Spacecraft Envelope 5

4 Gross Injected Payload for Saturn IB/Centaur versus C_3 Typical Parking Orbit (100 - nm) Flight Mode and Launch Azimuth of 90 Degrees 6

5 Comparison of Terminal Descent Atmosphere with Model 3 7

6 Tension Shell Configuration 11

7 Apollo Configuration 12

8 Modified Apollo Configuration - Sharp Corner 13

9 Blunted Cone 14

10 Blunted Cone - Oblate Spheroid-Launch Configuration 22

11 Blunted Cone - Oblate Spheroid-Entry Configuration 25

12 Blunted Cone - Entry Shell Structure 28

13 Oblate Spheroid Landed Capsule 32

14 Blunted Cone - Flotation Sphere-Launch Configuration.... 35

15 Flotation Sphere Landed Capsule 37

16 Flight Operational Sequences 40

17 Flight Capsule External Central Computer and Sequencer... 53

18 Internal Central Computer and Sequencer 55

19 Telecommunication Subsystem 69

20 Telecommunications System Block Diagram 75

PRECEDING PAGE BLANK NOT FILMED.

ILLUSTRATIONS (Cont'd)

Figure 21 Flight Capsule Power Profile 77

22 Power System Configuration-Before Separation 78

23 Attitude Control Subsystem Schematic 87

24 Mathematical Model of Attitude Control Subsystem 88

25 Cold Gas Reaction Subsystem 90

26 Single Parachute System-Impact Velocity versus Main Parachute Area/Suspended Weight 92

27 Single Parachute System Main Parachute Nominal Diameter versus Impact Velocity 93

28 Main Parachute Weight versus Dynamic Pressure 94

29 Descent Time Main Parachute versus Area/Suspended Weight 96

30 Time from Peak Acceleration to Mach 1.3 versus Peak -g 97

31 Blunted Cone 180-Inch Diameter Flight Capsule with Oblate Spheroid Landed Capsule, Cruise Near-Earth and Near-Mars Temperature History 109

32 Typical Postimpact Payload Temperature History 110

33 Thermal Control System 111

34 Modified Apollo-Launch Configuration 117

35 Modified Apollo-Entry Configuration 118

36 Tension Shell - Flotation Sphere Landed Capsule 122

37 Tension Shell Structural Details 125

38 Parametric Weight Terminology 129

39 Parametric Weight Analysis of Descent Subsystem 130

ILLUSTRATIONS (Cont'd)

Figure 40 Parametric Weight Analysis of Flotation Sphere Landed Capsule 131

41 Parametric Weight Analysis of Oblate Spheroid Landed Capsule 132

42 Maximum M/C_DA Allowable versus Vehicle Diameter 133

43 Effects of Main Parachute Reefing on M/C_DA 136

44 Reefed Deployment at Mach 1.3 137

45 Heat Shield and Structure Weight versus Diameter-Modified Apollo Configuration 139

46 Heat Shield and Structure Weight versus Diameter-Blunted Cone 140

47 Residual Weight Model 3 Atmosphere 141

48 Residual Weight Model 2 Atmosphere 142

49 Residual Weight Model 1 Atmosphere 143

50 Shell - Launch Shroud Constraint 146

51 Shell - Launch Shroud Constraint With Afterbody 147

52 Entry Weight Limitation - 1971 Mission Weight versus Vehicle Diameter 148

53 Entry Weight versus Entry Angle 164

54 Landed Concepts - Advanced Mariner 167

55 Landed Concepts - Mars Probe/Lander 169

56 Tension Shell - Air Bag Concept 172

57 Candidate 1971 Landed Systems 173

58 Flotation Sphere - Landed Payload - Center of Mass versus Available Volume 174

ILLUSTRATIONS (Cont'd)

Figure 59 Oblate Spheroid Landed Payload - Center of Mass versus Available Volume 175

60 Communications Geometry at Entry 181

61 Relay Frequency Selection 186

62 Parachute Dynamics 187

63 Relay Link Flight Capsule Look Angle Requirements 189

64 Effect of Shape on Antenna Pattern 190

65 Planar Spiral Antenna Pattern 191

66 Orbit Contact Geometry 199

67 Communication Range - 1971 201

68 Efficiency Estimates for S-Band Power Amplifiers 203

69 Direct - Link Telemetry 204

70 Direct - Link Look Angle Requirements 206

71 Gain Distribution Function S-Band "V" Antenna 208

72 Gain Distribution Function S-Band "V" Antenna 209

73 Gain Distribution Function S-Band "V" Antenna 210

74 Gain Distribution Function S-Band "V" Antenna 211

75 Direct Link Antenna Gain for Uniform and Gaussian Distributed Landed Attitudes 212

76 Flight Capsule Look Angle 214

77 Allowable Thrust Application Angle Error 237

78 Effect of Spin Rate on Integrated Heating 242

79 Configuration Comparison - Altitude at Mach 1.3 243

ILLUSTRATIONS (Concl'd)

Figure 80 Configuration Comparison - Altitude at Mach 1.3 244

81 Configuration Comparison - Maximum Dynamic Pressure 245

82 Configuration Comparison - Maximum Dynamic Pressure 246

83 Pertinent Trajectory Parameters 1971 Orbiter Launch Period 256

84 Pertinent Trajectory Parameters 1973 Orbiter Launch Period 257

85 Pertinent Trajectory Parameters 1975 Flyby Launch Period 261

86 Pertinent Trajectory Parameters 1975 Fastest Launch Period 262

87 Summary of Trajectory Parameters, 1971 Launch Period 266

88 Summary of Trajectory Parameters, 1971 Launch Period 267

89 Summary of Trajectory Parameters, 1975 Launch Period 268

90 Summary of Trajectory Parameters, 1975 Launch Period 269

TABLES

Table I Summary of Standard Model Atmosphere Parameters for Mars 8

II An Atmosphere Model for Terminal Descent Calculations 9

III Comparison of Reliability Goals and Predictions 20

IV Flight Capsule Weight Summary-Oblate Spheroid Landed Capsule 44

V Entry Shell Weight Breakdown 45

VI Instrumentation Weight Breakdown 46

VII Telecommunications and Power Weight Breakdown 47

VIII Flight Capsule Weight Summary-Flotation Sphere Landed Capsule 49

IX Descent Science and Engineering Instrumentation 58

X Landed Science and Engineering Instrumentation 59

XI Relationship of Instruments and Objectives 60

XII Data Transmission Objectives 66

XIII Telecommunications Characteristics 71

XIV Flight Capsule Instrumentation Power Requirements 79

XV Salient Characteristics of the Electrical Power Subsystem - 1971 81

XVI Load Summary - Blunted Cone - Multi-Mission Structure 101

TABLES (Cont'd)

Table	XVII	Heating Summary - Blunted Cone - 1971 Heat Shield	102
	XVIII	Summary of Heat Shield Design Data	104
	XIX	Comparison of Candidate Materials	106
	XX	Thermal Properties and Ablative Characteristics of Materials	106
	XXI	Blunted Cone-Heat Shield Requirements Entry Weight 1390 Pounds	107
	XXII	Temperature Limitations for Structural Members and Components	112
	XXIII	Blunted Cone 180-Inch Diameter Flight Capsule with Oblate Spheroid Landed Capsule	113
	XXIV	Modified Apollo - Weight Summary	116
	XXV	Modified Apollo - Entry Shell Weight Breakdown	120
	XXVI	Tension Shell - Weight Summary	123
	XXVII	Tension Shell Structural Weight Breakdown	126
	XXVIII	Single Mission Shell Design Criteria	144
	XXIX	Diameter Selection	149
	XXX	Weight Summary 1971 Mission - Multi-Mission Shell-197-inch Diameter	151
	XXXI	Weight Summary 1971 Mission - Multi-Mission Structure - 1971 Heat Shield - 197-inch Diameter	152
	XXXII	Weight Summary 1971 Mission-Heat Shield and Structure Designed for 1971 Mission - 197-inch Diameter	153

TABLES (Cont'd)

Table	XXXIII	Weight Comparison of Model 3 Designed Shell Diameter = 197-inches	154
	XXXIV	Weight Comparison of Model 2 Designed Shell Diameter = 183-inches	155
	XXXV	Weight Comparison of Model 1 Designed Shell Diameter = 140-inches	156
	XXXVI	Design Conditions for Entry Shells-180-inch Diameter	157
	XXXVII	Shell Synthesis - Multi-Mission Structure - 1971 Heat Shield	158
	XXXVIII	Shell Synthesis - Concept Comparison	159
	XXXIX	Shell Synthesis - Residual Weights	160
	XL	Mars Flight Capsule - Shell Comparison	161
	XLI	Weight Summary - Blunted Cone	163
	XLII	Shell Synthesis - Residual Weights	165
	XLIII	Weight Summary - Blunted Cone	165
	XLIV	Landed Capsule Concept Comparison	176
	XLV	Significant Spacecraft Characteristics	183
	XLVI	Relay-Link Data Rates	194
	XLVII	Science and Engineering Data List	195
	XLVIII	Entry and Descent Payload Weight Volume and Power Breakdown	197
	XLIX	Comparison of Direct-Link Performance	215
	L	Postimpact Data - Recommended Payload	217
	LI	Weight, Volume, and Power Requirements of Science Payload	218

TABLES (Concl'd)

Tables

LII Weight, Volume, and Power Requirements of Telecommunications 219

LIII Recommended Payload Salient Characteristics 221

LIV Oblate Spheroid Landed Capsule - Weight Summary 222

LV Revised Entry and Descent Data Requirements 223

LVI Revised Postimpact Data Requirements 225

LVII Weight for Different Science Payloads 226

LVIII Oblate Spheroid Landed Capsule - Weight Summary 229

LIX Oblate Spheroid Landed Capsule - Salient Characteristics 230

LX Flotation Sphere Landed Capsule - Weight Summary 231

LXI Flotation Sphere Landed Capsule - Salient Characteristics 232

LXII Failure Mode Summary 248

LXIII Pertinent Trajectory Parameters 1971 Orbiter Launch Period 258

LXIV Pertinent Trajectory Parameters 1973 Orbiter Launch Period 259

LXV Pertinent Trajectory Parameters 1975 Flyby Launch Period 263

LXVI Pertinent Trajectory Parameters 1975 Fastest Launch Period 264

GLOSSARY OF TERMS

Entry Shell: The entry shell is defined as the primary load-carrying structure including the ablative heat shield.

Multi-Mission Shell: Entry shell designed to survive entry into any of the Martian atmospheres considered at entry velocities up to 23,500 ft/sec., entry angles from -20 degrees to -90 degrees and entry weights up to 4500 pounds.

Single Mission Shell: Entry shell designed specifically for one launch opportunity or one Martian atmospheric model (e.g., 1971 design, Model 3 design).

Entry Weight: The weight of the entry vehicle at atmospheric entry.

Residual Weight: Entry weight less the entry shell weight.

Launch Vehicle System: The Launch Vehicle System includes the three stages of the Saturn IB/Centaur, with its guidance subsystems and the ascent fairing which shrouds the Planetary Vehicle, to make up the Launch Vehicle (LV) as the flight hardware; plus the supporting ground equipment, software, and associated manpower.

Spacecraft System: The Spacecraft System includes the Flight Spacecraft (FS) as its flight hardware; plus flight hardware spare parts, development models, associated operational support equipment (hardware and software), and the management and engineering teams.

Capsule System: The Capsule System includes the Flight Capsule (FC) as the flight hardware; plus flight hardware spare parts (or spare Flight Capsules depending on the time of spares replacement), development and sterilization assay models, control documentation and associated software, operational support equipment, and the management and engineering teams.

Planetary Vehicle: The Planetary Vehicle (PV) is defined as the composite Flight Spacecraft and Flight Capsule integrally attached and operated up to separation in the vicinity of the selected planet.

Space Vehicle: The Space Vehicle (SV) is the combined Launch Vehicle and Planetary Vehicle or Vehicles which physically leave the launch pad in conduct of the mission.

Mission Operations System (MOS): The MOS includes that portion of the Project which plans, directs, controls and executes (with support provided by the Deep Space Network) the space flight operation after injection of the Planetary Vehicle on its trajectory, the Mission-Dependent Equipment required at the Deep Space Network, and the operational teams.

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Deep Space Network (DSN): The DSN is comprised of the Deep Space Instrumentation Facility (DSIF), the Space Flight Operations Facility (SFOF), the Ground Communications System (GCS) connecting the two facilities, and the personnel who regularly operate these facilities.

Launch Operations System (LOS): The LOS includes those elements of the Project responsible for planning and executing the preflight and launch-to-injection phases of the mission.

Operational Support Equipment (OSE): The OSE includes the equipment and facilities required for the assembly, servicing, checkout, sterilization, and testing of the subsystems of the Flight Capsule.

Launch opportunity: a re-occuring duration of time, every 25.6 months, when favorable Earth - Mars spacial positions allow for practical interplanetary transfer trajectories.

Launch period: the number of days within the launch opportunity when practical Earth - Mars transfer trajectories are selected depending on mission objectives and Launch Vehicle constraints.

Launch window: the duration of time each Earth day when Space Vehicle launch is practical to achieve desired Planetary Vehicle transfer orbit orientation and characteristics depending on mission objectives and Launch Vehicle constraints.

Flight Capsule Terminology: Figures 1 and 2 present a further breakdown of the Flight Capsule, identifying the terminology at the operational stages of separation and/or deployment. In summary, the Flight Capsule is attached to the Flight Spacecraft by the forward and aft sections of the FC to FS adapter. Operation of the sterilization canister lid separation mechanism followed by the operation of the separation system on the FC to FS adapter, results in the Separated Vehicle. Attitude control and propulsion maneuvers are performed to place the Separated Vehicle on a preselected planetary impact trajectory. After these maneuvers, the propulsion and ACS (Attitude Control System) electronics assembly is separated and the resultant Entry Vehicle cruises to and enters the planet atmosphere. After entry, the entry shell (including the ACS reaction subsystem and spin/despin rockets) is separated and the Suspended Capsule descends through the atmosphere with the parachute. At a preselected time during descent, a separation mechanism operation extends the Landed Capsule from the parachute by use of a tether. At impact the Landed Capsule is separated from the tether for landed operations.

Note that the instrument packages are shown on the Suspended Capsule actually mounted to the entry shell and Landed Capsule support structure external to the Landed Capsule. In addition to the selected portions of the science instrumentation mounted externally and internally, appropriate portions of the

programming and sequencing, thermal control, telecommunications, electrical power and control, and the signal and power interconnection subsystems hardware are also individually attached and operationally integrated both internally and externally, to accomplish the data acquisition and transmission functions of the Capsule System mission.

Additional terminology used in this Book are as follows:

CC&S: Central Computer and Sequencer
TVC: Thrust Vector Control
ΔV: Velocity increment imparted to the Separated Vehicle

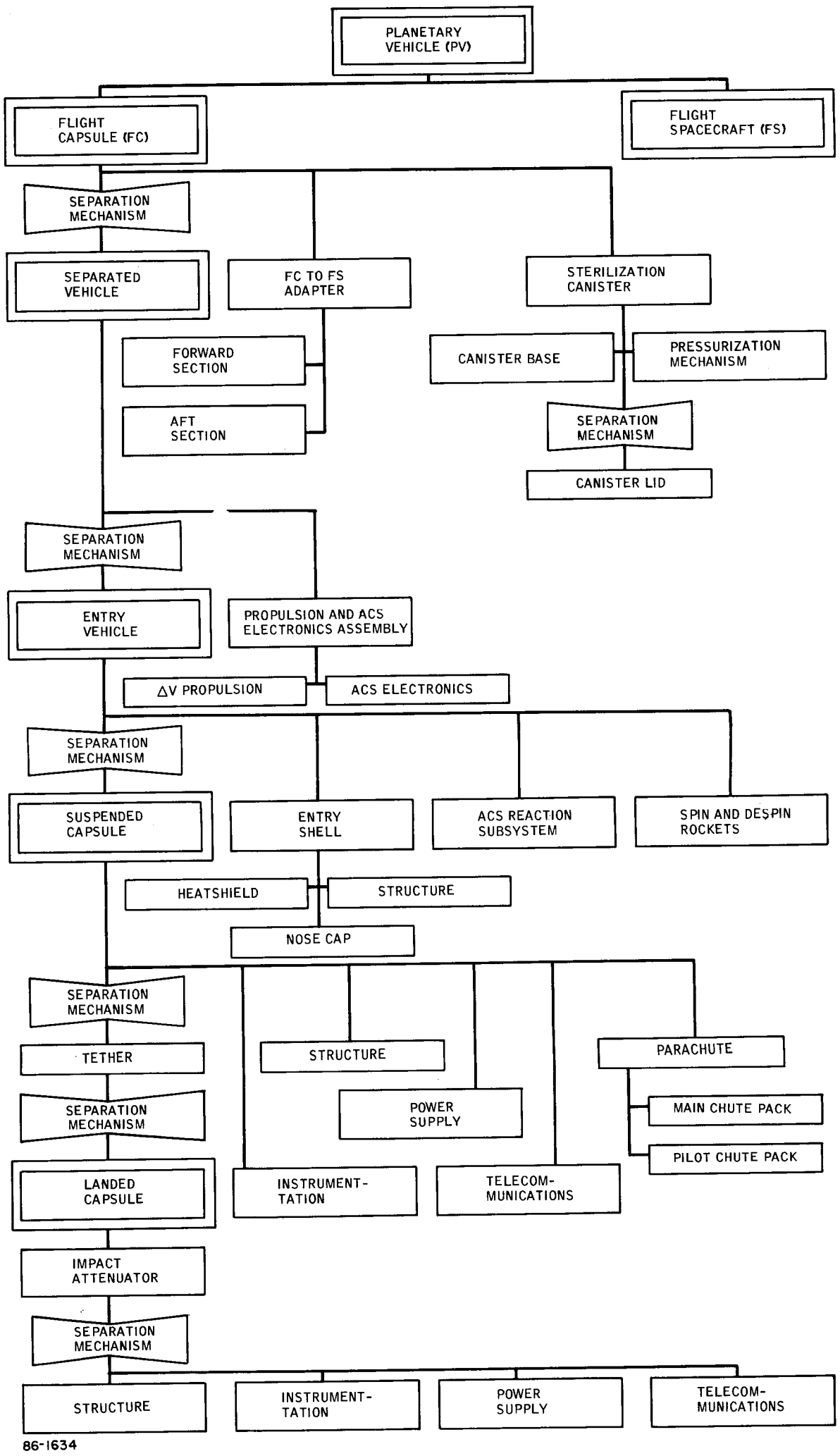

Figure 1 FLIGHT CAPSULE OPERATIONAL DIAGRAM AND TERMINOLOGY

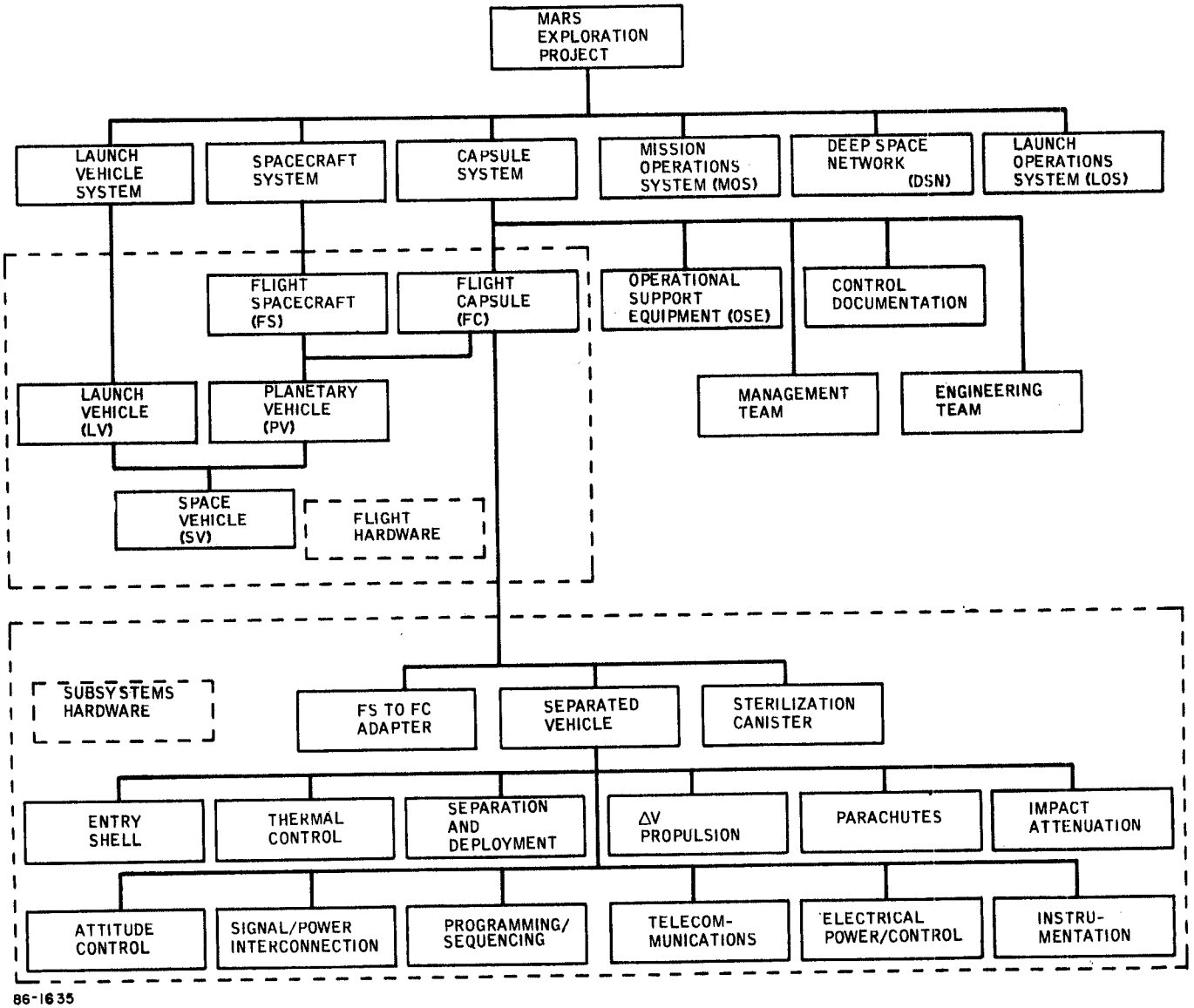

Figure 2 CAPSULE SYSTEM FUNCTIONAL DIAGRAM AND TERMINOLOGY

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The conduct of the study and technical preparation of this report involved the participation and close coordination of many people, all of whose contributions were important to the end results.

It is impractical to single out each individual, but the major contributors to the study, reflected in the material presented in this Book, are as follows:

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1.0 INTRODUCTION

This book describes the design for a Mars Probe/Lander (Flight Capsule) for the 1971 Launch Opportunity. The mission includes hard-landing on the surface and a surface mission duration of about one day. Several features of the design have been selected to accommodate the anticipated requirements of missions beyond 1971 so that some hardware commonality, from mission to mission, can be achieved. The mission concept discussed herein involves separation of the Flight Capsule from the Flight Spacecraft on the approach trajectory a few million kilometers from Mars. The trajectory of the Flight Capsule is then diverted by application of a small velocity increment to impact the planet while the Flight Spacecraft remains on its original trajectory to ultimately achieve orbit about Mars. During deceleration of the Flight Capsule within the Martian atmosphere, and subsequent parachute descent to the surface, a series of instruments determine the atmospheric properties. Once on the surface, several additional instruments are employed to determine certain of the surface properties of Mars. The surface mission lasts approximately 24 hours.

This Flight Capsule design is based upon three NASA model atmospheres synthesized before the successful Mars flyby of Mariner IV; the range of surface pressures considered in the design is 10 to 40 millibars. The results of the Mariner IV flyby which occurred late in this phase of the study, lowered the range of predicted surface pressure on Mars. The performance of a Flight Capsule utilizing the entry from the approach trajectory concept was re-examined in the light of this reduced range of surface pressures; the results showed this concept to be impractical for all but the simplest of atmospheric probes.

During the study several Flight Capsule entry shell design concepts of single and multiple-mission capability were evaluated. Alternate landing system approaches were considered and their capabilities compared. While these concepts are no longer appropriate for an entry from approach trajectory mission, they may prove useful if a hard lander design is considered for the newly established entry from orbit mission concept. The tradeoff studies presented therefore may still be meaningful when appropriately reexamined.

2.0 MISSION OBJECTIVES AND CONSTRAINTS

The entry from approach trajectory portion of this study was conducted under a specified set of mission objectives and constraints. The more significant of these ground rules are summarized below.

2.1 MISSION OBJECTIVES

The mission objectives are divided according to six mission phases: (1) preseparation, (2) separation, (3) separation to entry, (4) entry to parachute deployment, (5) parachute deployment to impact, and (6) surface operations. Engineering diagnostic data will be included for all mission phases. The scientific mission objectives include experiments to obtain the following data:

2.1.1 Preseparation

None

2.1.2 Separation

None

2.1.3 Flight Capsule Separation to Entry

Trapped radiation within Mars magnetosphere.

2.1.4 Entry to Parachute Deployment

Prime emphasis will be placed on obtaining atmospheric data; redundant instrumentation should be utilized where possible. Data as a function of altitude is to be obtained for the following:

a) Atmospheric density,

b) Atmospheric pressure,

c) Atmospheric temperature,

d) Atmospheric composition,

e) Trapped radiation within the atmosphere, and

f) Ionosphere electron density.

2.1.5 Parachute Deployment to Impact

Prime emphasis will be placed on obtaining atmospheric data; redundant instrumentation should be utilized. Data as a function of altitude will be obtained for the following:

a) Parachute deployment conditions,

b) Atmospheric density,

c) Atmospheric pressure,

d) Atmospheric temperature,

e) Atmospheric composition,

f) Trapped radiation in the lower atmosphere and,

g) Wind speed.

The following experiments will provide surface data:

1) Surface roughness

2) Impact loads.

2.1.6 Surface Operations

Samples from the immediately adjacent area will be obtained for the following:

a) Atmospheric density,

b) Atmospheric temperature,

c) Atmospheric composition,

d) Surface wind speed,

e) Surface hardness,

f) Soil composition,

g) Dust particle concentration,

h) Solar constant, and

i) Water vapor concentration.

2.2 STUDY GROUND RULES

The following is a list of basic systems ground rules and constraints that are to be utilized throughout the study. Unless otherwise specified, information applies to the 1971 through 1975 launch opportunities.

2.2.1 Technology Cutoff Date

September 1966 is considered the technology cutoff date for 1971 mission, and February 1968 for the 1973 mission.

2.2.2 Launch Vehicle

The reference launch vehicle for this program is the Saturn IB/Centaur. The dynamic envelope is to be 3 inches inside the static envelope, shown in Figure 3. The payload capability of the Saturn IB/Centaur configuration versus the energy parameter C_3, is presented in Figure 4. The Flight Capsule weight shall not exceed 2500 pounds in 1971 and 1973, and shall not exceed 5200 pounds in 1975 and 1977. The Flight Capsule envelope (nominal interface and hardpoints) is shown in Figure 3.

2.2.3 Atmospheric Models

The model atmospheres to be used are contained in Reference 1. The following data from Reference 1 are presented:

Table I -- Summary of Standard Model Atmosphere Parameters for Mars.

Table II -- An Atmosphere Model for Terminal Descent Calculation.

Figure 5 -- Comparison of Terminal Descent Atmosphere with Model 3.

Models 1, 2, and 3 atmospheres will be used for entry vehicle design and Models 1, 2, 3, and the terminal descent atmospheres will be used for parachute design. An altitude of 800,000 feet will be used for reference entry angle and entry velocity. (An altitude of 2.5×10^6 feet will be used for heating and dynamic motion studies in the Model 1 atmosphere.) A surface wind profile of 100 ft/sec shall be assumed and gusts of 50 ft/sec for a 10-second duration during parachute descent shall be used.

Reference 1 NASA Engineering Models of the Mars Atmosphere for Entry Vehicle Design, Edited by George M. Levin, Dallas E. Evans, and Victor Stevens, NASA TN D-2525.

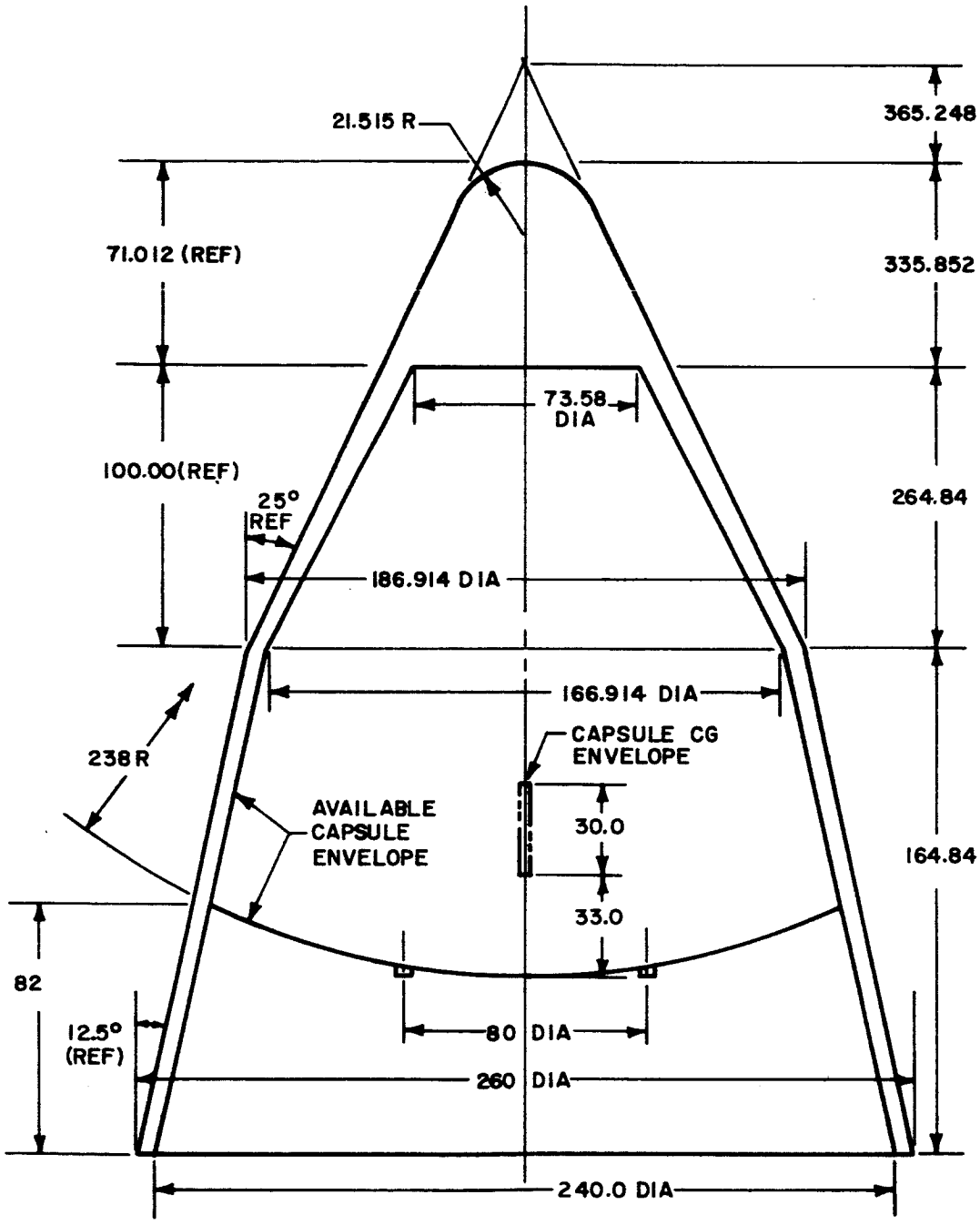

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Figure 3 VOYAGER DESIGN STUDIES SPACECRAFT ENVELOPE

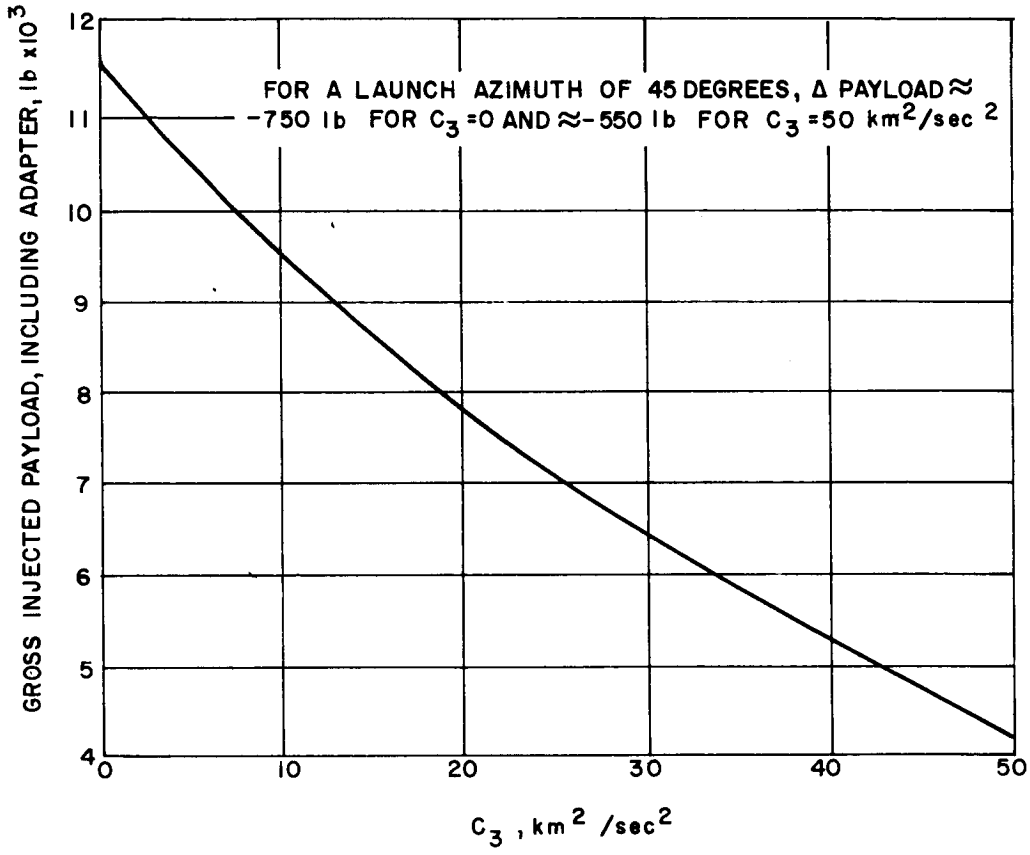

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Figure 4 GROSS INJECTED PAYLOAD FOR SATURN IB/CENTAUR VERSUS C_3 TYPICAL PARKING ORBIT (100 NM) FLIGHT MODE AND LAUNCH AZIMUTH OF 90 DEGREES

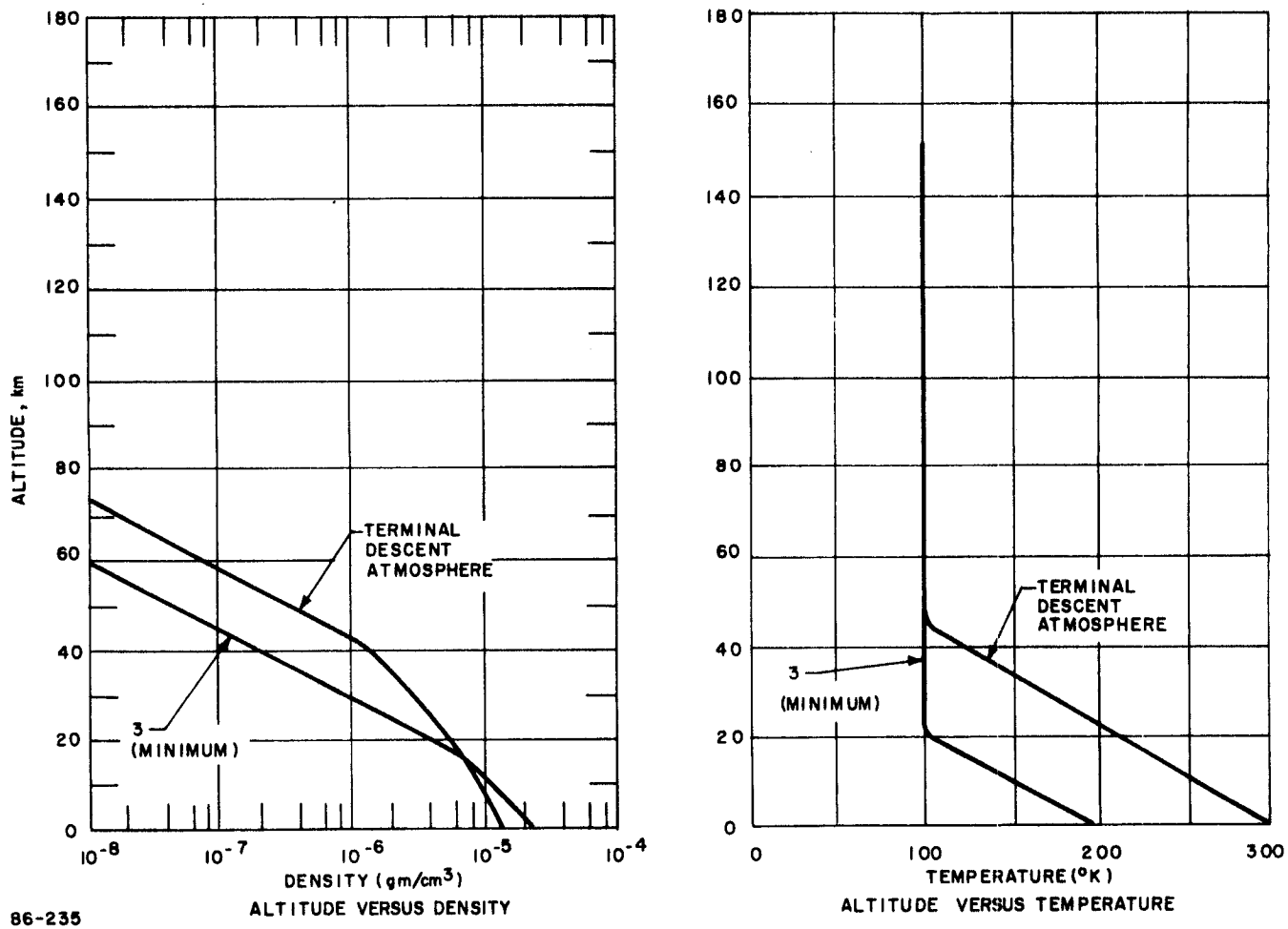


Figure 5 COMPARISON OF TERMINAL DESCENT ATMOSPHERE WITH MODEL 3

TABLE I

SUMMARY OF STANDARD MODEL ATMOSPHERE PARAMETERS FOR MARS

Parameter	Units	Maximum (Model 1)	Mean (Model 2)	Minimum (Model 3)
Surface Pressure	mb	40	25	10
Pressure	lb/in^2	0.58	0.363	0.145
Composition	CO_2 percent by Mass	7.5	16	60
	N_2 by Mass	92.5	84	40
	CO_2 by Volume	4.9	10.8	48.8
	N_2 percent by Volume	95.1	89.2	51.2
Molecular Weight	--	28.8	29.7	35.85
Acceleration of gravity at surface	cm/sec^2	375	375	375
	ft/sec^2	12.3	12.3	12.3
Surface Temperature	°K	300	250	200
	°R	540	450	360
Troposphere lapse rate	°K/km	-3.636	-3.89	-4.55
	$°R/10^3$ ft	-1.995	-2.134	-2.496
Tropopause altitude	km	11	18	22
	ft	36,100	59,100	72,200
Stratosphere temp.	°K	260	180	100
	°R	468	324	180
Top of stratosphere	km	150	150	150
	ft	492,100	492,100	492,100
Thermosphere lapse rate	°K/km	2	2	--
	$°R/10^3$ ft	1.097	1.097	--
Surface density	gm/cm^3	4.62×10^{-5}	3.57×10^{-5}	2.16×10^{-5}
	$slugs/ft^3$	8.97×10^{-5}	6.94×10^{-5}	4.19×10^{-5}

TABLE II

AN ATMOSPHERE MODEL FOR TERMINAL DESCENT CALCULATIONS

An alternate model atmosphere for use in the design of a retardation system for terminal descent may also be postulated. This model is generated by substituting a surface temperature of 300°K in place of the 200°K surface temperature in model 3 and keeping the same surface pressure and the same composition. The result is a model that has a higher atmosphere density at altitudes above 15 km, but at altitudes below 15 km the atmosphere density is lower than that of model 3. Equations are given below for the density in the troposphere and stratosphere regions in both metric and english units.

Metric Units

Troposphere region (below 46 km)

$$\rho = 1.44 \times 10^{-5} (1 - 0.0145\, Z)^{2.50} \quad \text{gm/cm}^3$$

Stratosphere region (above 46 km)

$$\rho = 8.64 \times 10^{-4}\, e^{-\frac{Z}{6.5}} \quad \text{gm/cm}^3$$

(note Z is in km)

English Units

Troposphere region (below 150, 900 ft)

$$\rho = 2.79 \times 10^{-5} (1 - 4.41 \times 10^{-6}\, Z)^{2.50} \quad \text{slugs/ft}^3$$

Stratosphere region (above 150, 900 ft)

$$\rho = 2.0 \times 10^{-3}\, e^{\frac{-Z}{21,000}} \quad \text{slugs/ft}^3$$

(note Z is in ft)

2.2.4 Aerodynamic Shapes

Three basic entry shapes* with various modifications will be considered as follows:

Tension shell -R_N/R_B = 0 or 0.1 -- Figure 6

Apollo with minimum afterbody -- Figures 7 and 8

Blunted cone -- Figure 9

Alterate afterbody configurations are shown. The hypersonic drag coefficients in air for the various shapes are as follows:

Tension Shell (R_N/R_B = 0.1 or sharp)	1.60
Apollo	1.44
Modified Apollo	1.58
Blunted Cone (120 degrees)	1.59
Blunted Cone (110 degrees)	1.48

2.2.5 Impact Attenuation System

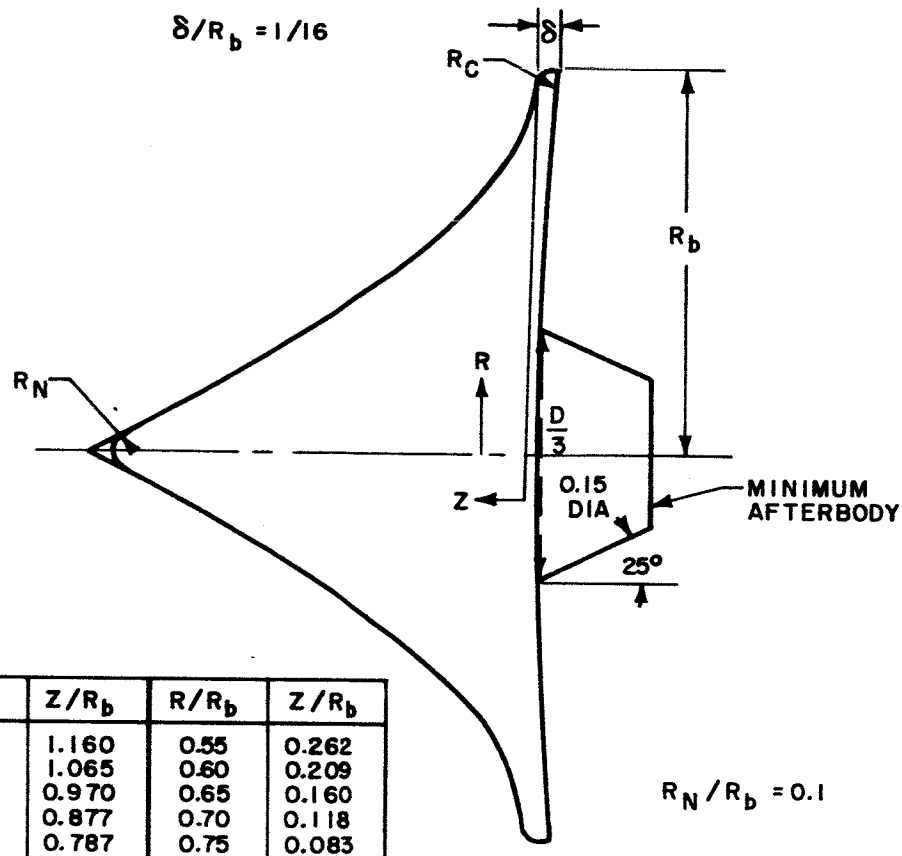
An omnidirectional passive impact attenuator (crushup material or deployable stroke concept) shall be used. The maximum g level to be considered is 1000 Earth g. However a higher level, 2500 Earth g, is allowable for failure mode. Impact will be on a nonyielding surface. The Martian terrain model to be considered is specified in Reference 2. A spherical landed capsule will be considered for all three entry vehicle shapes. A conical landed capsule will be considered for the tension shell and blunted cone. A lenticular landed capsule will be considered for Apollo and blunted cone.

2.2.6 Descent Subsystem

The main parachute is to be fully deployed subsonically at a minimum altitude of 15,000 feet. The maximum vertical impact velocity will be 80 ft/sec. Sizing and design of the parachute system will be based upon deceleration in the terminal descent atmosphere. For flight in the Model 3 atmosphere, the deployment of the parachute will be assumed to begin in the Model 3 atmosphere and be fully deployed in the terminal descent atmosphere with altitude,

*Weight parametric studies will consider only the three basic entry shapes. Each shape will be characterized by its drag coefficient. The highest C_D of each basic shape will be used.

Reference 2 Mars Engineering Atmosphere and Surface Models, Dwain F. Spencer, JPL Document, 21 July 1965.



R/R_b	Z/R_b	R/R_b	Z/R_b
0	1.160	0.55	0.262
0.05	1.065	0.60	0.209
0.10	0.970	0.65	0.160
0.15	0.877	0.70	0.118
0.20	0.787	0.75	0.083
0.25	0.699	0.80	0.053
0.30	0.615	0.85	0.030
0.35	0.534	0.90	0.014
0.40	0.459	0.95	0.003
0.45	0.388	1.00	0
0.50	0.322		

86-236

Figure 6 TENSION SHELL CONFIGURATION

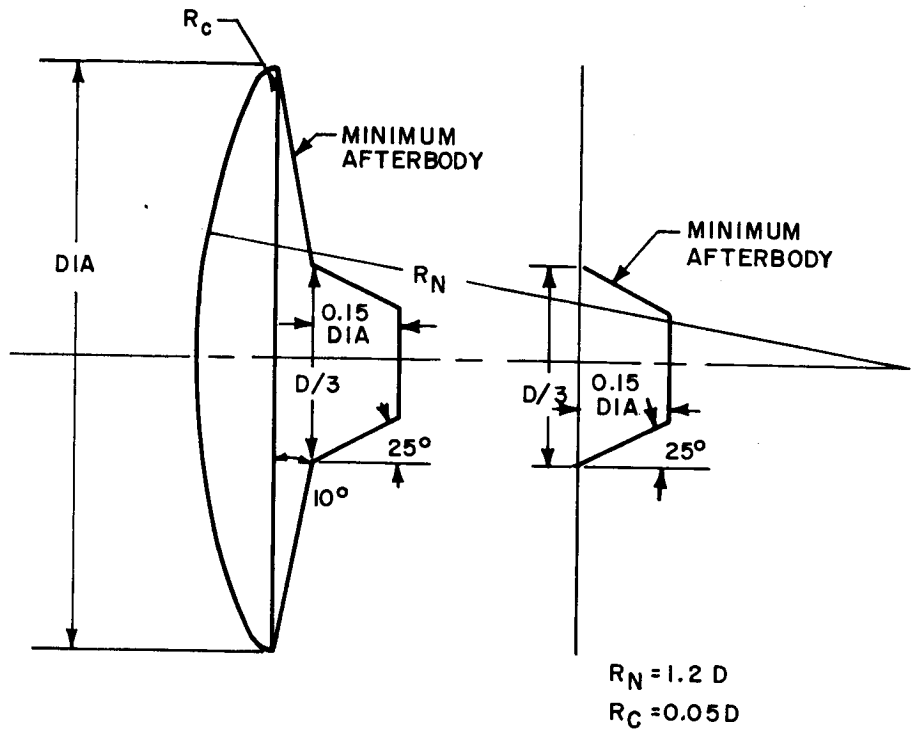


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Figure 7 APOLLO CONFIGURATION

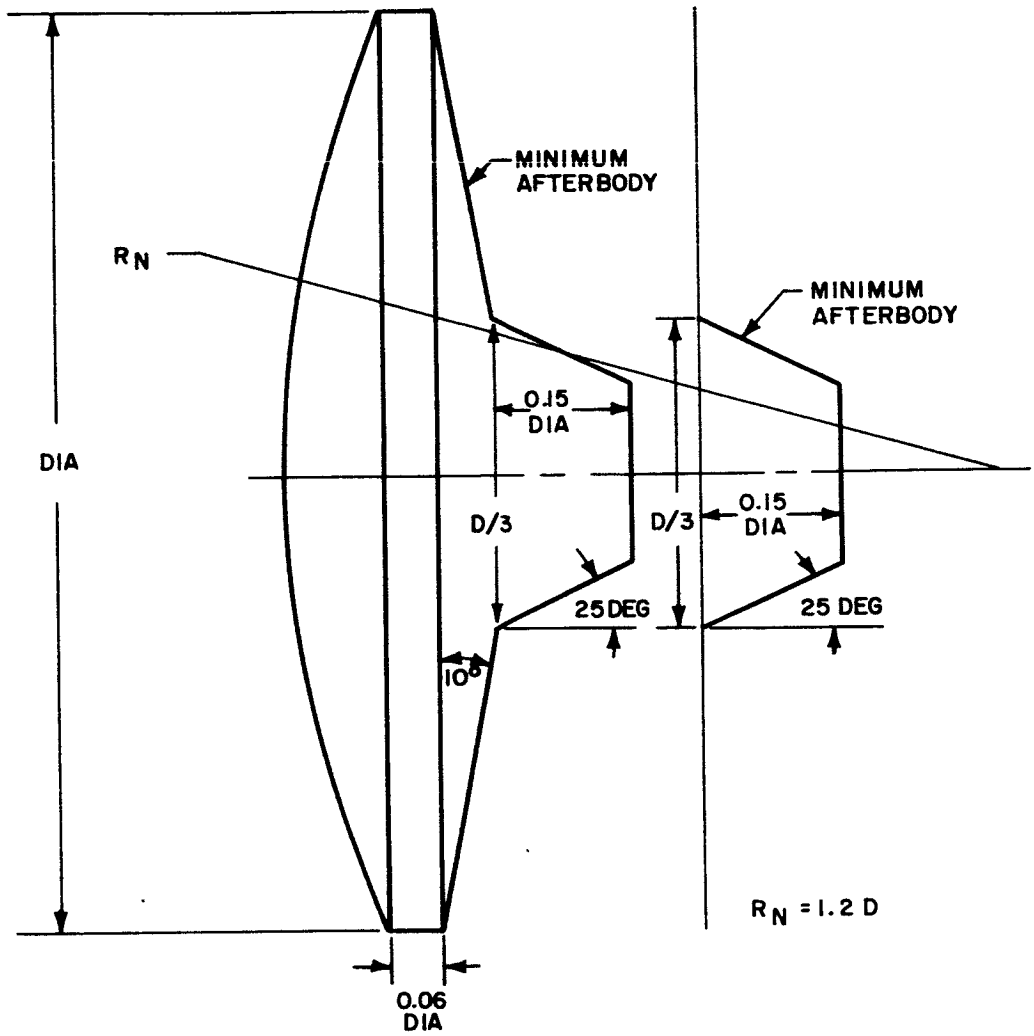

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Figure 8 MODIFIED APOLLO CONFIGURATION - SHARP CORNER

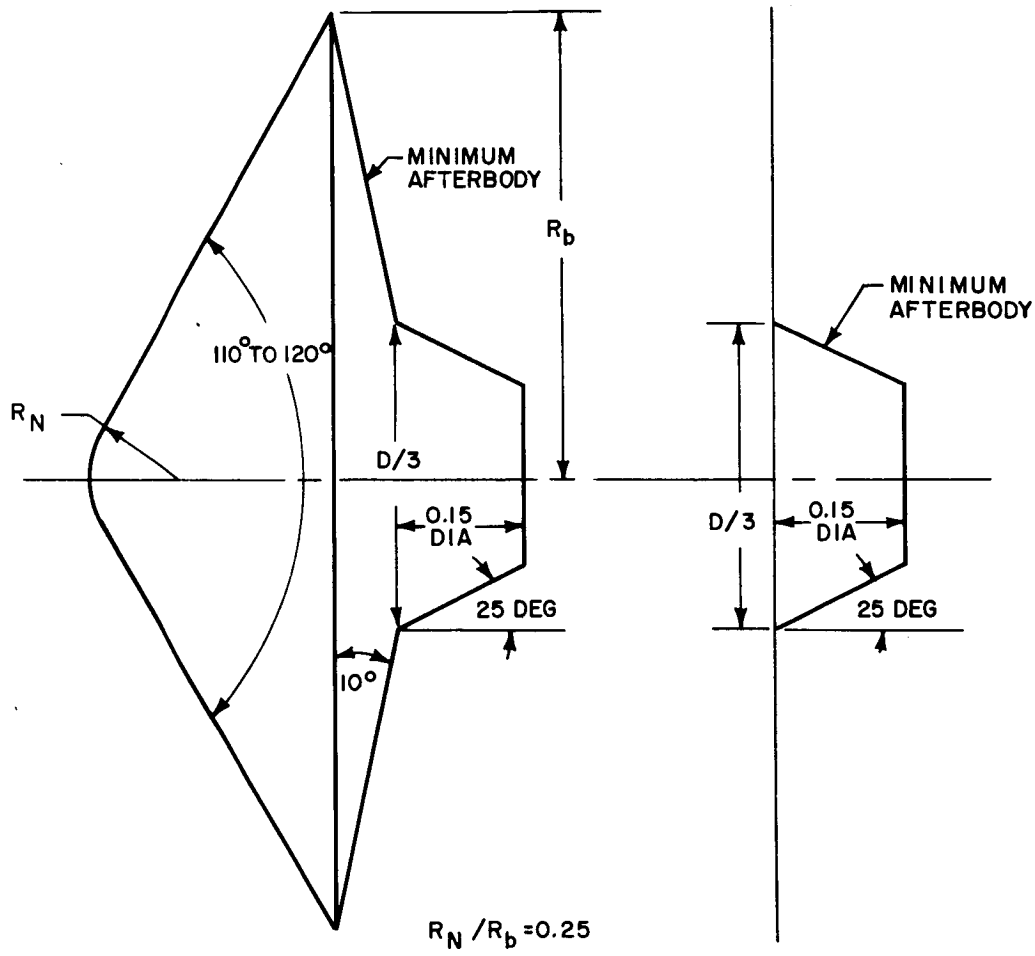

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Figure 9 BLUNTED CONE

velocity, and time being continuous and dynamic pressure being discontinuous between the two atmospheres. No change of atmospheric conditions will be used for Models 2 and 1 atmospheres.

2.2.7 Sterilization

The Flight Capsule system shall be capable of sustaining qualification testing of 3 cycles of heat in a dry atmosphere at 145°C for 36 hours per cycle with sufficient time between cycles to return the Flight Capsule temperature to ambient and terminal sterilization of one 24-hour cycle of heating at 135°C in dry nitrogen. The probability of Mars contamination shall be less than 10^{-4} for the Flight Capsule. The total internal microbial content of the Flight Capsule immediately prior to terminal heat sterilization shall be less than 10^{8} viable organisms. The suspended capsule will be assembled in a clean room. Major subsystems other than suspended capsule do not require clean-room assembly. Final assembly of all major subsystems and terminal sterilization of the entire Flight Capsule will be performed at Cape Kennedy.

The Flight Spacecraft is not to be sterile.

2.2.8 Deep Space Network

The Deep Space Network (DSN) including the Deep Space Instrumentation Facility (DSIF) and the Space Flight Operations Facility (SFOF) shall be utilized as defined in Reference 3.

2.2.9 Launch Opportunities and Flight Capsule Separation

The launch opportunities and Flight Spececraft trajectories to be considered are:

1971 -- orbiter spacecraft (flyby spacecraft as failure mode)

1973 -- orbiter spacecraft (flyby spacecraft as failure mode)

1975 -- flyby spacecraft

Considerations for Flight Capsule separation shall be:

On the planetary vehicle approach trajectory, and

After establishment of Flight Spacecraft orbit.

Reference 3 System Capabilities and Development Schedule of the Deep Space Instrumentation Facility, 1964-68, Technical Memorandum 33-83, Revision 1, 24 April 1964.

2.2.10 Flight Spacecraft Characteristics

The Flight Spacecraft will serve as a transport vehicle for the Flight Capsule and provide required services until Flight Capsule separation. The Flight Spacecraft may serve as a communication relay station from Flight Capsule separation throughout the lifetime of the Flight Capsule. Consideration should be given to direct communication. A reasonable degree of flexibility will be considered during the Flight Spacecraft - Flight Capsule interface design. Separation shall occur during the period between 2 and 20 days prior to encounter. Consideration shall be given to Flight Spacecraft - Flight Capsule separation, both with and without a Flight Spacecraft attitude maneuver prior to separation. The Flight Capsule shall be mounted along the Flight Spacecraft center line (no cocked mounting). Both Flight Capsule speedup and Flight Spacecraft slowdown shall be considered to obtain communication lead time. The reference Flight Spacecraft orbits considered shall be 4000 x 10,000 km and 4000 x 20,000 km. The periapsis altitude for the 1971 orbiter window will be:

h_p = 21,000 ± 12,800 km - 1 mid-course correction

h_p = 4000 ± 1000 km - 2 mid-course corrections

h_p = 4000 ± 500 km - 3 mid-course corrections

2.2.11 Guidance

The rms error in the measurement of the position and velocity of the Flight Spacecraft with respect to Mars at planetary distances will be taken as 150 km and 0.003 m/sec (1σ). The planetary radius will be taken as 3,385 km, and the Mars gravitational parameter as 42,977.8 km^3/sec^2.

2.2.12 Landed Capsule Surface Mission Lifetime

The surface mission lifetime for the various launch opportunities are as follows:

1971 mission -- at least one day

1973 mission -- 1 to 6 months

1975 mission -- approximately 6 months

2.2.13 Landing Site

The Landed Capsule should be landed (including dispersions) at a Mars latitude which is within 30 degrees of the sub-Earth point for preliminary

consideration. Nominally, landing should occur to allow direct link communications in an area which is a circle of 500-km radius (3σ).

2.2.14 Structures

A shell concept capable of carrying variable size payloads shall be used in the design. Weight estimates shall be confirmed by a detailed layout of the design. Flight Capsule exterior profile shall be defined as the structural shape.

3.0 SYSTEM CHARACTERISTICS

3.1 SYSTEM DESCRIPTION

The 1971 flight capsule is capable of completely independent space flight from the point of separation from the flight spacecraft, on the approach trajectory, through ballistic entry and parachute descent in the Martian atmosphere to survival on the planet surface for at least one day. The scientific payload includes instruments to determine the character of the Martian atmosphere and to grossly define the planetary surface.

The 2500-pound flight capsule, as part of the planetary vehicle, is launched on a Saturn S1B/Centaur Launch Vehicle and transported in a semi-passive state to the vicinity of Mars. It is controlled and monitored by the flight spacecraft which provides all the necessary housekeeping services during interplanetary transfer.

The flight capsule separates from the flight spacecraft 12 days prior to flight spacecraft orbital injection. A small velocity increment is applied to the flight capsule after separation to achieve a Mars impact trajectory and to provide sufficient lead time prior to flight spacecraft orbital injection to accommodate relay communications during entry and descent. The timing of the separation event provides flight capsule entry, parachute descent, and impact in view of the DSIF station at Goldstone, California. Active attitude control is employed after separation to maneuver the flight capsule to the proper thrusting attitude and to provide thrust vector control during engine firing. Immediately after the application of velocity increment the flight capsule is spun up to 10 rpm for attitude control through entry, and the ACS Electronics and Propulsion Assembly is jettisoned.

The 1850-pound blunted-cone entry vehicle has an M/C_DA of 0.20 slug/ft^2 and is constructed with beryllium face sheet-stainless steel core honeycomb structure and cork silicate heat shield.

The entry vehicle structure is of a multi-mission design capable of entry velocities up to 23,800 ft/sec and entry angles from -20 to -90 degrees in the model 2 or model 3 atmospheres at M/C_DA's from 0.15 to 0.48 slug/ft^2. However, the actual range of entry angles for the 1971 mission is more restricted, shallower than -52 degrees, because an M/C_DA of at least 0.20 slugs/per square foot was required to realize significant Landed Capsule payload. An M/C_DA of less than 0.15 slug/ft^2 is necessary to achieve the desired parachute deployment altitude and Mach number conditions in the Model 3 atmosphere at an entry angle of -90 degrees. The increased M/C_DA results in a reduction in the allowable range of entry angles (see Section 5.7). The heat shield is accordingly designed for the reduced requirements of the 1971 mission. A

single subsonic parachute is deployed in a reefed condition at Mach 1.3 about 19,000-feet altitude and disreefed at 16,000 feet in the terminal descent model atmosphere; higher deployment altitudes obtained in the more dense atmospheres. An oblate spheroid landed capsule, protected by a foam-filled fiberglass honeycomb crushable material impact attenuator, lands on the planet surface. Direct and relay link communications are used to provide functional redundancy for data return. Relay communications is prime during entry and parachute descent. Direct communications to Earth is used after landing. The data from entry is transmitted over the direct link after impact. The system is powered by sterilizable nickel cadmium batteries.

Analysis of the flight capsule design shows a predicted reliability of 0.278. Reliability goals for each subsystem were determined to increase the flight capsule reliability to 0.51, consistent with overall mission goals. These predictions and goals are shown in Table III.

3.2 DESIGN DESCRIPTION

Two designs are presented in this section. The reference design is the Blunted-Cone Entry Shell - Oblate Spheroid Landed Capsule. An alternate design is the Blunted-Cone Entry Shell - Flotation Sphere Landed Capsule.

3.2.1 Reference Design

The subsystem characteristics are summarized in detail in Section 4.0. This section presents the design integration of these subsystems in terms of detailed inboard profiles of the reference design in both the launch and entry configuration. An inboard profile is also presented for the Oblate Spheroid landed capsule. These inboard profiles are presented in Figures 10 through 13.

The design consists of several major assemblies; sterilization canister, ΔV propulsion and ACS electronics, Flight Capsule/Flight Spacecraft Adapter, Suspended Capsule Structure, Entry Shell, and Landed Capsule. This grouping of assemblies is dictated by the flight operational sequence of the flight capsule from prespacecraft separation to post-impact surface operation. This is presented in Section 3.3, Figure 16.

3.2.1.1 Sterilization Canister

The sterilization canister performs two vital functions. The first is to isolate all items to be landed on Mars, from the presence of viable strains of Earth organisms (a biological barrier). The second function is to help control the internal temperature of the flight capsule during transfer from Earth to near encounter with Mars.

TABLE III

COMPARISON OF RELIABILITY GOALS AND PREDICTIONS

Subsystems/Subassemblies	Design Goal	Prediction (presterilization)	Prediction (poststerilization)
1. Sterilization shroud assembly	0.982	0.958	0.957
2. Entry shell assembly	0.978	0.992	0.987
3. Descent payload assembly	0.952	0.978	0.599
4. Landed payload assembly	0.903	0.986	0.967
5. Electronic assembly	0.687	0.801	0.740
6. Internal payload assembly	0.900	0.991	0.688
Overall System Reliability	0.51	0.728	0.278

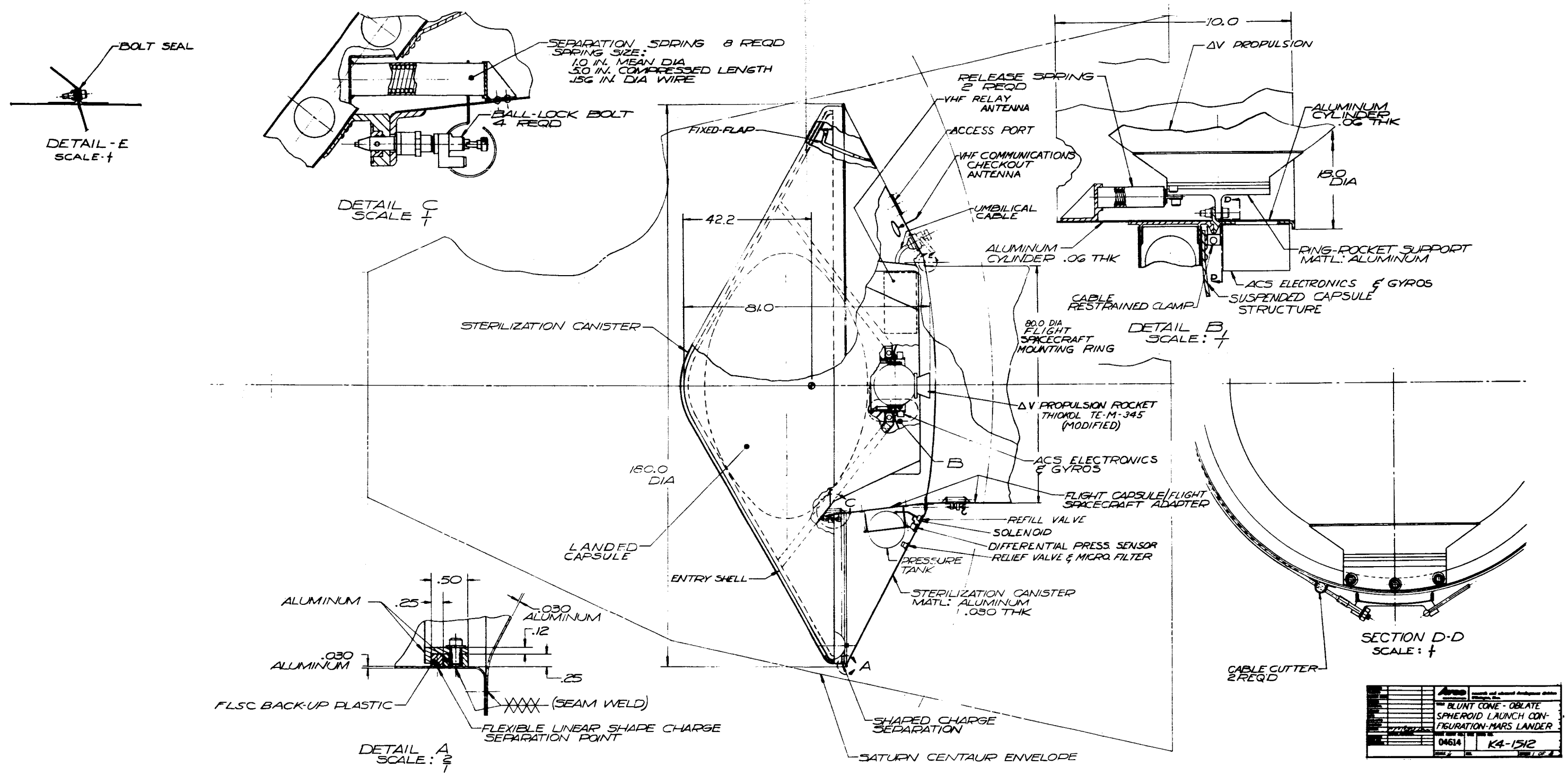


Figure 10 BLUNTED CONE - OBLATE SPHEROID-LAUNCH CONFIGURATION

To provide the first function it was necessary to construct the canister of welded aluminum shells consisting of two main sections, the base and lid. The base section is a weldment consisting of two basic pieces; a center spherical cap and an outer truncated cone. These two sections are welded to a ring at the intersection with the flight spacecraft/flight capsule adapter. Because of the canister size (~183 inches in diameter) each of these pieces will be constructed by welding together gore sections from standard size sheets.

The flight spacecraft/flight capsule adapter sections (forward and aft) are bolted to the canister base at the ring joining the two pieces of the base section (Reference detail E, Figure 10). Detail design of the seal of these holes is not complete at this time, but is recognized as a problem area because of the severe sterilization requirements.

The lid is composed of a flat cone (similar in shape to the entry shell), again made from standard size sheets welded together. The lid separation system is mounted at the outer rim. It consists of two rings. One ring is welded to the canister, while the second ring is bolted to the first, clamping the flexible linear shaped charge, FLSC, and its plastic backup material against the canister (see detail A of Figure 10). Because of the available lengths of FLSC it will consist of four equal length sections, butted together, thereby requiring each length to be approximately 142 inches long.

Two problems have been recognized; one is that even 142 inches may make handling of the shaped charge, without damaging difficult. The second is the problem associated with the expansion and contraction of the sterilization canister with respect to the shaped charge during temperature changes. Further investigation and test are required in these areas.

A detonator with dual ignitor circuits is used behind each of the shaped charge butt joints. The electrical lines from each of these detonators lead to one electrical connector which interfaces with a connector on the base section.

Near encounter with Mars, an electric current is introduced to the ignitor circuit. This ignites the detonators which, in turn, detonate the flexible linear-shaped charge. Since the butted ends of the shaped charge allow propagation of the explosion from one length of explosive to the next, the firing of a length will be initiated by any combination of two detonators and two adjacent charges.

The lid and the base have turned rims at their outermost diameters. These rims are welded together as the final step of assembly just prior to exposing the flight capsule to the sterilization heat cycle. Also at this time, the electrical connector to the ignitor circuit for the flexible linear-shaped charge, is mated with that from the safing and ignition circuit in the base.

Pads are bonded to the lid section to provide separation between the canister lid and the entry shell. Investigation into the pad composition is required to assure compatibility with the requirements of non-adherence and non-damage to the heat shield surface during launch vibration.

A pressurization subsystem is included in the sterilization canister and is attached to the base section. The purpose of this subsystem is to maintain a higher internal pressure than the external ambient pressure ($\approx$ 1 psi), so an outflow of gases at minute leaks, will preclude the entrance of micro-organisms.

The subsystem consists of a 12-inch diameter tank and regulator valve that senses the differential between the internal canister pressure and the ambient pressure. Maintenance of the necessary internal differential pressure is accomplished by the pressure regulator feed from the storage tank.

In addition to the pressure tank and regulator, a fill line and shutoff valve, and a pressure relief valve with a microfilter, are incorporated in the system. The fill line is only used after the sterilization cycle to refill the internal tank from another tank that also experiences the same cycle. The relief valve is required to bleed off overpressure during the launch phase of the mission.

3.2.1.2 ΔV Propulsion and ACS Electronics

In order to increase the usable entry vehicle weight it became desirable to jettison as much of the flight capsule weight as possible prior to atmospheric entry. One subsystem that could be jettisoned was the ΔV rocket case and the ACS electronics-gyro package. These two subsystems are mounted in one package, since both are utilized at approximately the same time and have completed their function soon after flight dpacecraft separation. As presented in Figure 10, the ΔV rocket is a spherical, solid propellant (Thiokol No. TE-M-345 modified) with thrust termination by nozzle deployment. For reasons of available space, separation simplicity and gyro requirements, it is desirable to make the jettisoned package as compact as possible. Therefore the ACS electronics and gyros are mounted around the ΔV

rocket as shown in detail B, Figure 10. Also shown in this detail is the separation mechanism. It consists of clamps (similar to the Marmon type) and a restraining cable (Reference detail D). The restraining cable holds the clamps (located in four places equally spaced) in place and is severed by cable cutters to release the clamps. The cable cutters are located in two places; either one can release the system. Separation of the package is accomplished by two springs diametrically opposite each other. These springs will push the package away from the separated vehicles at 1.5 ft/sec.

3.2.1.3 Flight Capsule/Flight Spacecraft Adapter

This assembly forms the major load path, mounting and separation technique for the complete flight capsule. The assembly is divided by the sterilization canister forming two sections; an aft section, where the mounting ring to the spacecraft is located, as well as the backup flight capsule separation system, and a forward section where the main separated vehicle separation system and mounting interface is located. Both sections are conical monocoque structures with simple end rings. A detail of the ball-lock separation mechanism is shown in detail C of Figure 10. Four of these mechanisms (all manifolded together) are used to release the separated vehicle from the flight spacecraft. Eight coil springs, paired at each ball lock, push the separated vehicle away from the spacecraft at 1.5 ft/sec. The coil springs are matched at assembly, such that they are of equal force to reduce possible tipoff errors. Each spring is completely encased in a cylinder to stabilize the spring and to provide a controlled guide during separation.

3.2.1.4 Suspended Capsule Structure

This portion of the flight capsule forms the primary structural element, in that it, encapsulates and supports the landed capsule during entry and during parachute descent, provides the major load path system for the entry shell and landed capsule during launch, provides the mounting surface for most of the external payload, parachute package and ΔV propulsion and ACS electronics package, and provides the separation system interface between the entry shell, landed capsule, tether and ΔV propulsion packages. The details of this structure are illustrated in Figure 11. It consists of eight straight radial beams (aluminum channels) running out from the ΔV propulsion interface to the entry shell interface. The entry shell separation mechanism, consisting of ball lock release mechanisms manifolded together with two gas generators (either of which will activate the system), is located at this interface. This system is initiated by a microswitch triggered by a load cell mounted in the parachute riser line. When the load in the parachute (in the reefed condition) riser line just

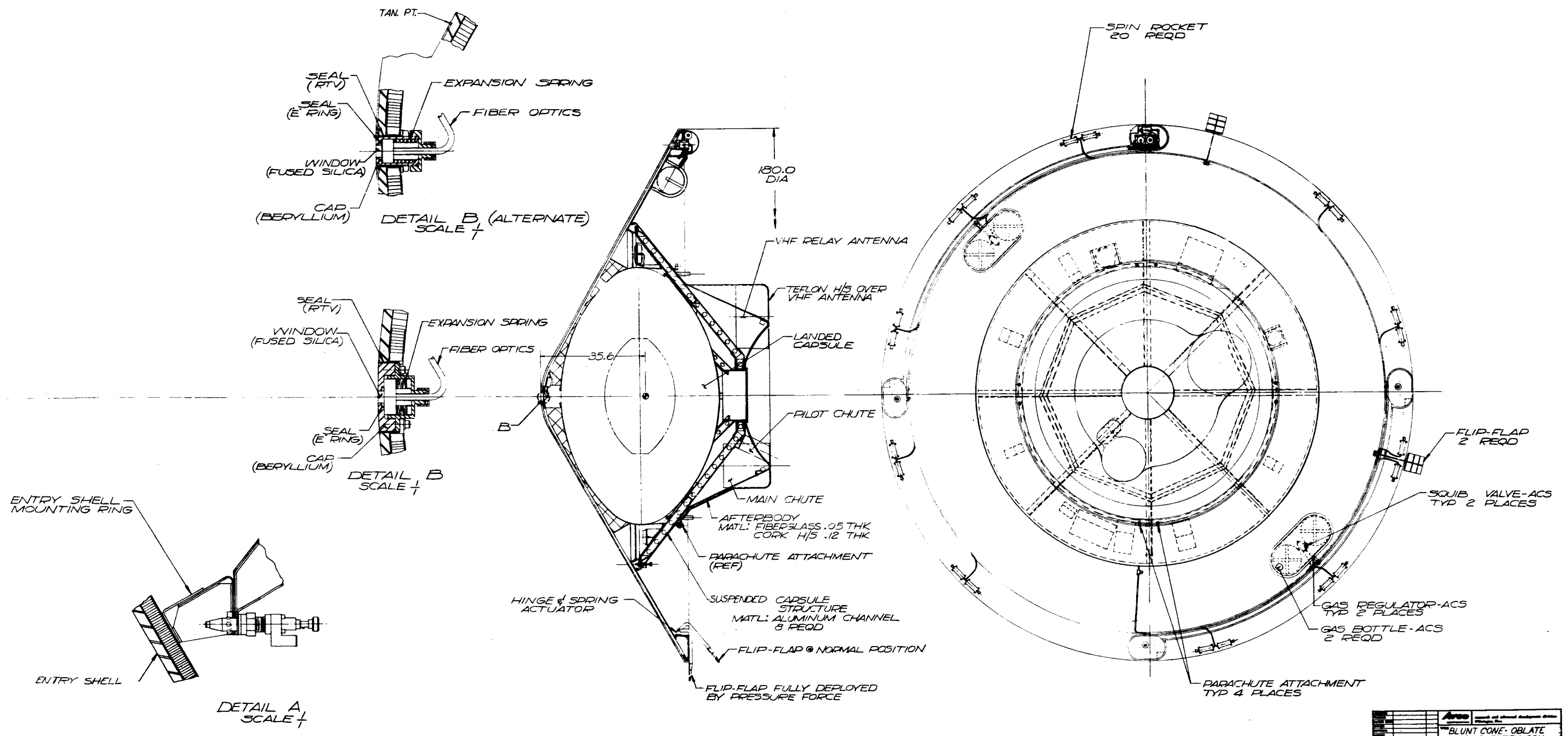


Figure 11 BLUNTED CONE - OBLATE SPHEROID - ENTRY CONFIGURATION

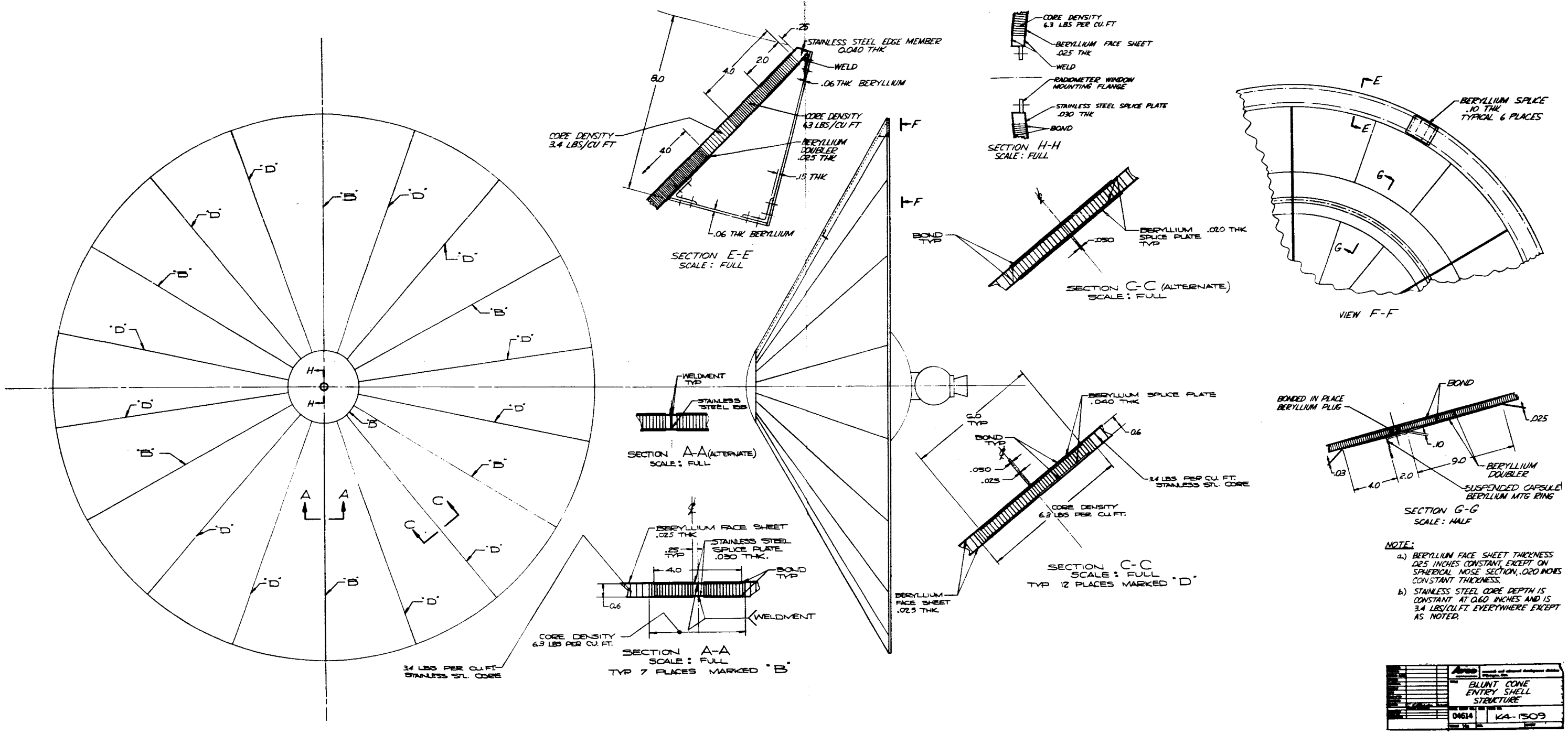
passes its peak, the microswitch is initiated. This technique reduces the dynamic motion of the entry shell at separation such that its final motion will not damage the landed capsule as it falls away. Details of the separation system are shown in Figure 11 detail A. This system is the same as that utilized in the separated vehicle-flight spacecraft separation.

In the equipment cavity between the landed capsule and the suspended capsule structure, are mounted the science and engineering instrumentation, telecommunications and power equipment for the suspended capsule. The various components of the telecommunications electronics and the science instrumentation are located around the periphery of the cavity and mounted to the ring between the radial beams. This is clearly illustrated by the end view in Figure 11. The equipment is shielded during entry from wake heating by a fiberglass-cork insulated afterbody extending from the entry shell interface to the aft end of the Suspended Capsule Structure. The afterbody as defined in Figure 11 serves two basic purposes; it provides a thermal protection shield for the landed capsule and separated capsule equipment, and it provides the righting moment if rearward entry should occur.* The afterbody shape was defined, for aerodynamic reasons, to provide a rearward center of pressure location behind the center of gravity of the entry vehicle.

The parachute package is also housed in the afterbody. The package consists of a pilot chute and mortar, and a main chute. Each chute is enclosed in a container in which a low pressure is maintained during the interplanetary transfer portion of the mission. The container is opened by the mortar during pilot-chute ejection. The pilot chute is ejected at an angle to the entry vehicle axis to force it out of the wake of the vehicle. It, in turn, pulls the main chute bag from the container and deploys the main chute. The main chute deploys in an 18-percent reefed condition and remains in that condition until the vehicle reaches 16,000-feet altitude. Reefing cutters are then actuated and the main chute is fully deployed.

The main chute is attached to four of the eight radial beams, through a four-line bridle. The bridle and the attachment points are designed to withstand the loads of the chute opening on two points. During the initial opening of the chute to a reefed condition, the loads of the entry shell and the landed capsule are taken out at the beam and entry shell interface. On release of the entry shell the landed capsule loads are distributed to the parachute attachment points by straps

*This afterbody would only work efficiently for rearward entry instability at entry angles of attack less than 150 degrees. A flap design is employed between 180 and 150 degrees to guarantee the needed righting moment at these angles. This design is presented in Figure 11.



NOTE:

a.) BERYLLIUM FACE SHEET THICKNESS 0.25 INCHES CONSTANT EXCEPT ON SPHERICAL NOSE SECTION, .020 INCHES CONSTANT THICKNESS.

b.) STAINLESS STEEL CORE DEPTH IS CONSTANT AT 0.60 INCHES AND IS 3.4 LBS/CU.FT EVERYWHERE EXCEPT AS NOTED.

Figure 12 BLUNTED CONE - ENTRY SHELL STRUCTURE

around the forward face of the capsule. These straps are secured at the forward end by a single cable. On reaching 500 feet of altitude, this cable is severed by cable cutters and the landed capsule is released to the end of a tether.

A VHF (270 MH_z) antenna is mounted in the aft end of the afterbody. The antenna consists of a surface spiral on a cavity one-half wave length in diameter by one-quarter wave length in depth. This antenna is used for the communication link from flight spacecraft separation to impact (through descent). All objects in the antenna pattern field of view must be dielectric. For this reason a non-charring Teflon low temperature ablator was place over the surface of the antenna.

At the end of blackout during Mars atmospheric entry, the radar altimeter is turned on. This system utilizes two antennas, a high-altitude antenna, and a parachute descent antenna. The first antenna is derived by energizing the outer closeout ring (stainless steel) of the entry shell ($\approx$180 inches in diameter). The second antenna is a 6-inch diameter flat-slot array mounted in the Suspended Capsule Equipment Cavity. After entry shell jettisoning (parachute deployment) this antenna has a clear field of view to the planet's surface.

3.2.1.5 Entry Shell

The entry shell assembly consists of the four major components: the thermal protection system, the entry structure, a mounting ring, and the landed capsule bearing pad.

The thermal protection system used in the generation of this design is chopped cork with a silicon binder compressed to the desired bulk density ($\approx$ 30 lb/ft^3). Panels, approximately 3 ft^2, are applied to the entry structure by a thin layer of adhesive. The required thickness of the cork is 0.47 inch at the nose to 0.37 inch at the outer rim of the shell.

During entry, wake heating is experienced on the aft side of the entry structure.* To maintain the structural temperature at the design allowable ($\approx$ 500°F) some insulation must be supplied over the aft portion. A thin layer of cork is employed here as well. Analysis indicated that minimum gage ($\approx$ 0.125 inch) would suffice.

*In case of postseparation flight sequence failure, the entry vehicle may enter rearward. In this case, the afterbody will experience an initial heating pulse before the normal wake heating pulse.

The entry structure employed in this design is a bonded beryllium face sheet-stainless steel core (honeycomb) sandwich shell. The selection resulted from a parametric tradeoff study which indicated that significant weight savings (~ 30 to 40 percent) could be obtained by using this type of construction over all stainless steel or aluminum honeycomb. The details of the shell design are illustrated in Figure 12.

Due to the limitation of beryllium sheet size (~36 inches by 100 inches) the structure is made from 18 equal sized radial gores (indicated as B and D in Figure 12). Panels consisting of three sections are fabricated at one time. These panels consist of six (three on each side) beryllium face sheet gores, four beryllium splice plates (see Section A-A), the stainless steel core and two tapered beryllium doublers (see Section G-G). This complete panel assembly is then bonded together to form one-sixth of the completed structure. All of the panels are welded together, as shown in Section A-A, to form the conical section of the shell (i. e., without spherical cap). A spherical cap section is then bonded together with two stainless steel ring weldments (i. e., inside and outside similar to Section A-A). This cap is then welded to the cone section along with a stainless steel closeout ring (reference Section E-E) to form the basic structure. The final operation is the attachment of the beryllium ring sections, as shown in Section E-E and Section G-G. The first set of rings is made up of extruded angles and flat web sections riveted together. This set of rings is then bonded to the honeycomb shell and the stainless steel closeout ring. The second ring is the suspended capsule structure mounting ring. This ring is an extruded angle and is bolted to the shell at preselected locations where beryllium plugs are bonded in place. *

The face sheet thickness is held constant over the complete structure (0.025 in the cone section and 0.020 in the spherical cap) thus eliminating the necessity of chem-milling tapered face sheets and local doublers (all doublers are bonded in place) which simplifies fabrication and reduces cost. The core of 0.60 inch in depth is held constant throughout the complete shell. However, the density of the core is varied to strengthen local stress areas such as splice and weld joints.

*The fabrication techniques present in this paragraph are very similar to the techniques that would be employed in the fabrication of aluminum honeycomb, except that aluminum weldments would be utilized. Stainless steel honeycomb is also similar except that brazing would be used instead of bonding.

The final component that is included as part of the entry shell is the landed capsule bearing pad. This pad is the major load path of the landed capsule to the entry shell during the high entry decelerations. The large area is required due to the low bearing strength of the landed capsule impact attenuator which is made of foam plastic (most likely polyurathane) formed to the oblate spheroid contour.

The reaction control system, the spin and despin systems and some of the external science instrumentation are mounted to the entry shell. All of the reaction control systems (nozzles, tanks, etc.) are mounted on the entry shell in the area of the end closeout ring assembly. The base ring supports and forms the mounting pad for these systems as illustrated in Figure 11 in the end view.

The reaction control system consists of four sets of three-axis nozzles (one set in each quadrant) and two sets of cold gas tanks and regulators, one set for each two sets of nozzles, forming two complete systems. The system has the capability of pure couple reactions but is redundant, in that there are two completely independent systems, such that if one set is inoperative, the other set will provide the reaction capability. The nozzles in the pitch and yaw direction protrude through the entry shell because the available space in the dynamic ascent shroud envelope would not allow the nozzles to be mounted on the outside rim of the entry shell.

The spin and despin rocket motors are also mounted in the same area as the reaction control system. Ten spin rockets are provided in two groups, one group is used under normal operational sequence (i.e., 10 rpm spin stabilization after ΔV thrusting) and the other group to spin the Separated Vehicle up to 50 rpm in case of an ACS failure before separation. In the second case a third group of eight despin rocket motors are required to despin back down to 10 rpm early in entry.

Located in the nose section of the blunt cone entry shell is the radiometer quartz window which is utilized during the peak entry heating pulse. Around the quartz window is a beryllium heat sink plug to keep products of ablation away from the window. Details of this window along with an alternate approach are shown in Figure 11.

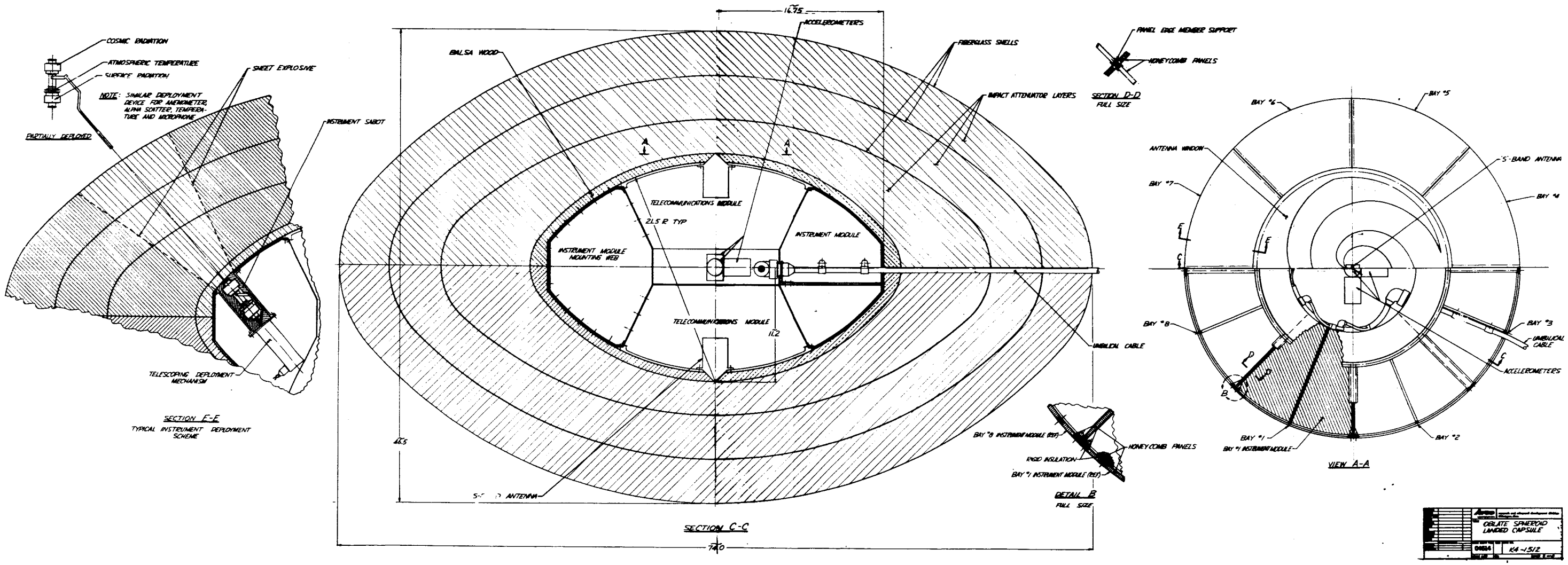


Figure 13 OBLATE SPHEROID LANDED CAPSULE

3.2.1.6 Oblate Spheriod Landed Capsule

The mission of the landed capsule is to absorb the impact velocity and, after coming to rest, deploy or expose the payload instruments to record the surface conditions and environments. Several landed capsule concepts were examined to select a landed capsule design. This evaluation (as presented in Section 6.0) resulted in two basic approaches; the oblate spheriod and the flotation sphere. Both approaches employ a crushable material impact attenuator. Of the two approaches, the oblate spheriod was selected as the reference for conceptual design evaluation in a complete system. The flotation sphere was considered as an alternate approach and is presented in the following section (3.2.2). An inboard profile of the reference design is shown in Figure 13.

An impact attenuation material is required to absorb the loads produced by impact velocities up to 130 ft/sec (80 ft/sec descent and a 100 ft/sec wind). In the following design, laminate fiberglass/honeycomb was chosen as the crushable impact attenuation material because of its high efficiency in absorbing these velocities at low g-level decelerations (500 to 1000 g). It is filled with polyurethane foam to give it better anisotropic properties. More detailed discussions of impact materials and problems are given in Section 3.0, Book 6, Volume V.

To absorb the impact at a deceleration of 500 g, the required attenuator is 15 inches thick on each of the flatter sides and 23 inches thick at the large diameter (reference Volume V, Book 6 Appendix A). The attenuation material is constructed with the honeycomb core cells aligned radially. It is built up to the required thickness by bonding several thicknesses together, the total thickness being bonded to the internal structure of the oblate spheroid. Separation of the attenuator is accomplished by an explosive sheet around the area of instrument deployment as shown in Section E-E.

The oblate spheroid was chosen, among other reasons, since the package should come to rest on one of two sides after landing on the planet. However, this requires instrument deployment in two directions. With two directions, instruments can be swiveled to the correct position or duplicated so that at least one of each may be deployed from the package. In the design shown in Figure 13, this deployment mechanism combines three atmospheric-type instruments into one unit. A similar mechanism would be used for other atmospheric and surface deployable equipment such as the anemometer, microphone, atmospheric temperature probe, surface radiation detector acoustics densetometer surface temperature probe and alpha scatter detecter. The last two instruments require deployment to the surface. All mechanisms for

deployment utilize telescoping tubes that are gas-pressure actuated. Other mechanisms such as springs, and electric motors, have been investigated and found inadequate. Each of the telescoping mechanisms uses a plastic foam sabot around the instruments to protect them from the high g-loads at impact. The determination of which side of the capsule is up after it has come to rest for instrument deployment is made by a gravity-sensing mercury switch. The internal structure is 24 inches high by 40.8 inches in diameter. It is constructed of aluminum honeycomb panels with stiffeners at jointing points and at instrument mounting locations (reference details D and E). The unit is of modular construction with the instruments mounted in radial bay sections that are connected to a main umbilical. The units are then bolted together making one assembly.

This type of construction was developed to facilitate assembly, in that certain modules would contain a selected grouping of equipment such that it could be checked out as an entity. Hence, checkout time during manufacturing assembly is kept to a minimum.

Mounted in the center of each face of the landed capsule S-band V-horn antennas. These antennas are fed by a coaxial cable in the center.

The main umbilical line from the landed capsule support structure is in the outer periphery of the impact attenuator. The cable is fed through one bay into the central section of the landed capsule where it picks up the various bays of equipment modules. At separation from the support structure, this cable is cut by a cable cutter to terminate electrical contact with the external payload while on the tether.

In addition to the required crushable impact material as discussed previously a thin layer ($\sim$1 inch) of balsa wood is used around the internal structure to protect the landed payload from sharp rocks or surface hazards.

3.2.2 Alternate Design - Flotation Sphere Landed Capsule

Due to the particular advantages of the flotation sphere landed capsule design, primarily its greater efficiency in delivering a given payload, an alternate conceptual design was developed. This design utilizes the blunted-cone entry shell configuration and the flotation sphere landed capsule.

3.2.2.1 Modifications to Entry Shell Design

The only modifications in the blunted-cone-flotation sphere other than the landed capsule itself, are the suspended capsule structure and the VHF relay antenna location. The significant difference between the oblate spheroid and spherical landed capsule is the relative size. As can be seen in Figure 14, the sphere is somewhat smaller and fits much more compactly in the entry shell (i.e., it sets forward producing a more favorable c.g. location). This then moves the entry shell mounting ring into a smaller diameter, thus reducing the landed capsule support structure weight and the afterbody size and weight.

The modification to the suspended capsule structure is shown in Figure 14 details D and E. The modification consists of straightening out the load path system from the entry shell to the flight spacecraft adapter intersection to form a cylindrical shell section, instead of the eight radial beams presented in the previous design. It is not clear which is the better approach at present and only rather detailed structural analysis will verify the correct approach. This modification also affected the location of the mounting ring. The mounting ring location on this design is a 58-inch diameter whereas the mounting ring in the reference design is a 120-inch diameter. As noted previously, this smaller diameter reduces the mounting ring weight, however, the shell weight probably increases due to the smaller area on which the landed capsule loads are reacted by the entry shell (i.e., a higher concentration of loads). This is particularly true of the multi-mission structure for which both shells are designed to hold a 4500-pound entry vehicle. For that mission the loads are reacted as a line load at the mounting ring. It is obvious from the shell standpoint that the further out the mounting ring is, for the 4500-pound entry vehicle loading condition, the better off the shell will be. If the shell were designed only for the 1971 mission it may prove more advantageous to move the mounting ring in as far possible. More tradeoff studies are required to substantiate this conclusion.

The other significant difference in this design approach over the reference design is the use of the VHF antenna within the landed capsule for all communication links (i.e., from post-separation to post-impact). This implies that all structural and miscellaneous components aft of the antenna should be dielectric material, such that they will not interfere with the antenna pattern. Hence, all structures, such as the ΔV propulsion support, afterbody (including heat shield), etc. must be fabricated from fiberglass or similar materials. Preliminary analysis of the heat shield requirements for the afterbody indicated that Teflon will be required, and will weigh

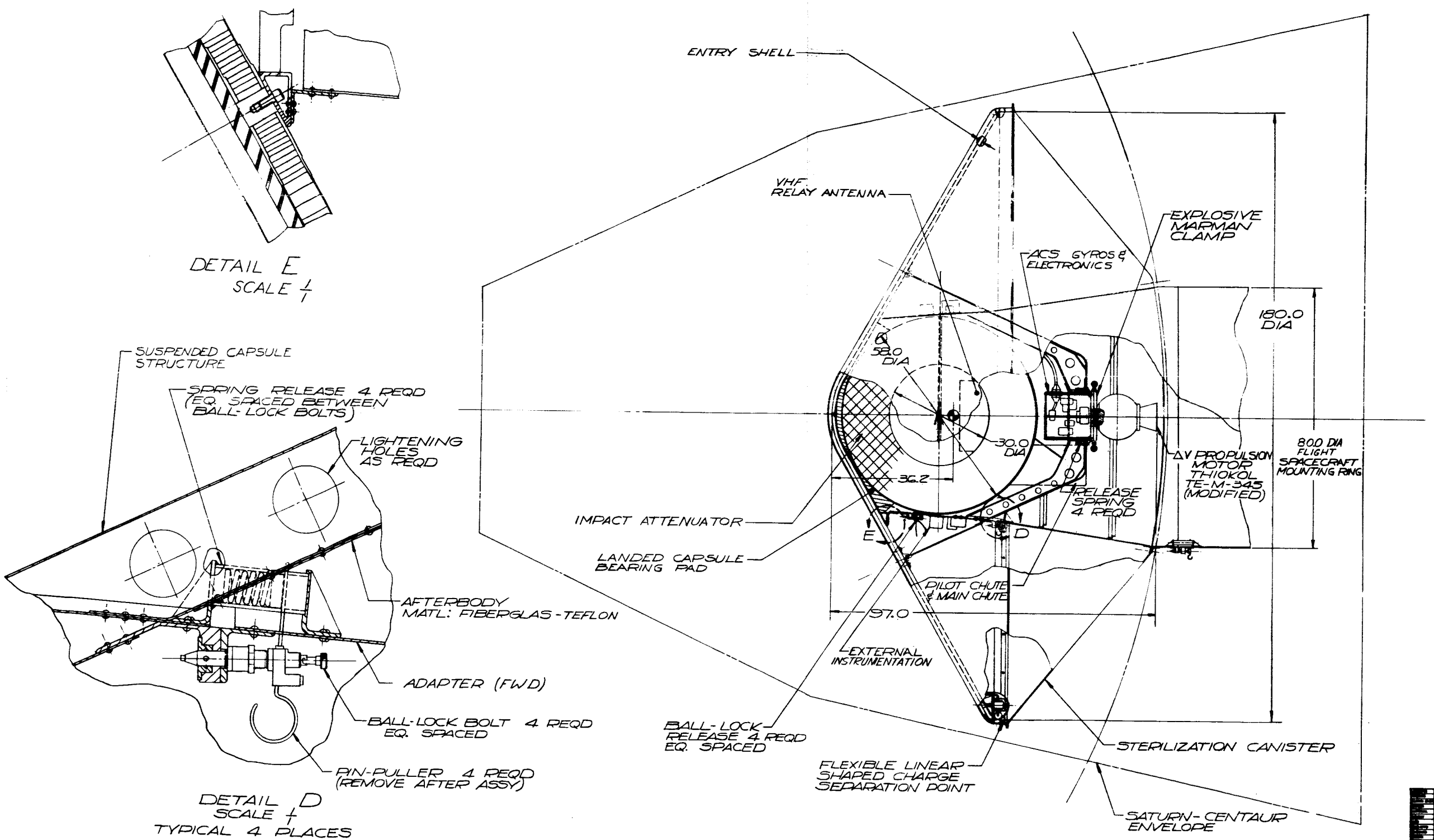


Figure 14 BLUNTED CONE - FLOTATION SPHERE - LAUNCH CONFIGURATION

approximately 100 pounds more than the cork heat shield used on the blunted cone-oblate spheroid design. This was the main reason for utilizing a separate VHF antenna on the outside (in the afterbody) on the reference design, along with the ease in fabrication of the metal landed capsule support structure instead of fiberglass as required in this design approach.

3.2.2.2 Landed Capsule - Flotation Sphere

A spherical shape was chosen as an alternate design because it is the most compact method of packaging a given mass. It also is the most effective shape for protecting the payload from a random-direction shock load. The details of the flotation sphere landed capsule are shown in Figure 15.

Unlike the oblate spheroid, the spherical impact attenuator has a uniform thickness of 14.5 inches. It is removed from the flotation shell by cutting it with explosive charges and ejecting it by explosive forces and supplementary springs.

The internal structure is made of two spheres, one floating in a fluid within the other, so that in a random attitude after impact the inner sphere will rotate to align a particular axis with the local vertical. The instruments are packaged in the inner sphere with an offset center of gravity position below the center of buoyancy ($\sim$2.0 inches) to rotate the inner sphere to the vertical position. This ability to orient itself with the local vertical eliminates the need for duplicate deployable instruments as in the oblate spheroid. The outer sphere is constructed from two fiberglass hemispheres ($\sim$0.30 inch thick) with precision molded interior surfaces.

The inner sphere is constructed in the same manner as oblate spheroid, i.e., it is made up of a series of equipment module bays (total of six) that form the structural assembly. Each of the six bays are fabricated of a thin membrane outer layer (forming the sphere contour) and a radial web. This web is a honeycomb panel utilized in mounting equipment.

The complete bay, with certain selected equipment (forming an independent module), is mounted to membrane shear webs extending from a cylindrical section in the center of the sphere. This center section houses the battery power unit, accelerometers, part of the telecommunication system and the main electrical cabling network (including the umbilical caging mechanism). The structural breakdown is illustrated in view A-A and Section P of Figure 15.

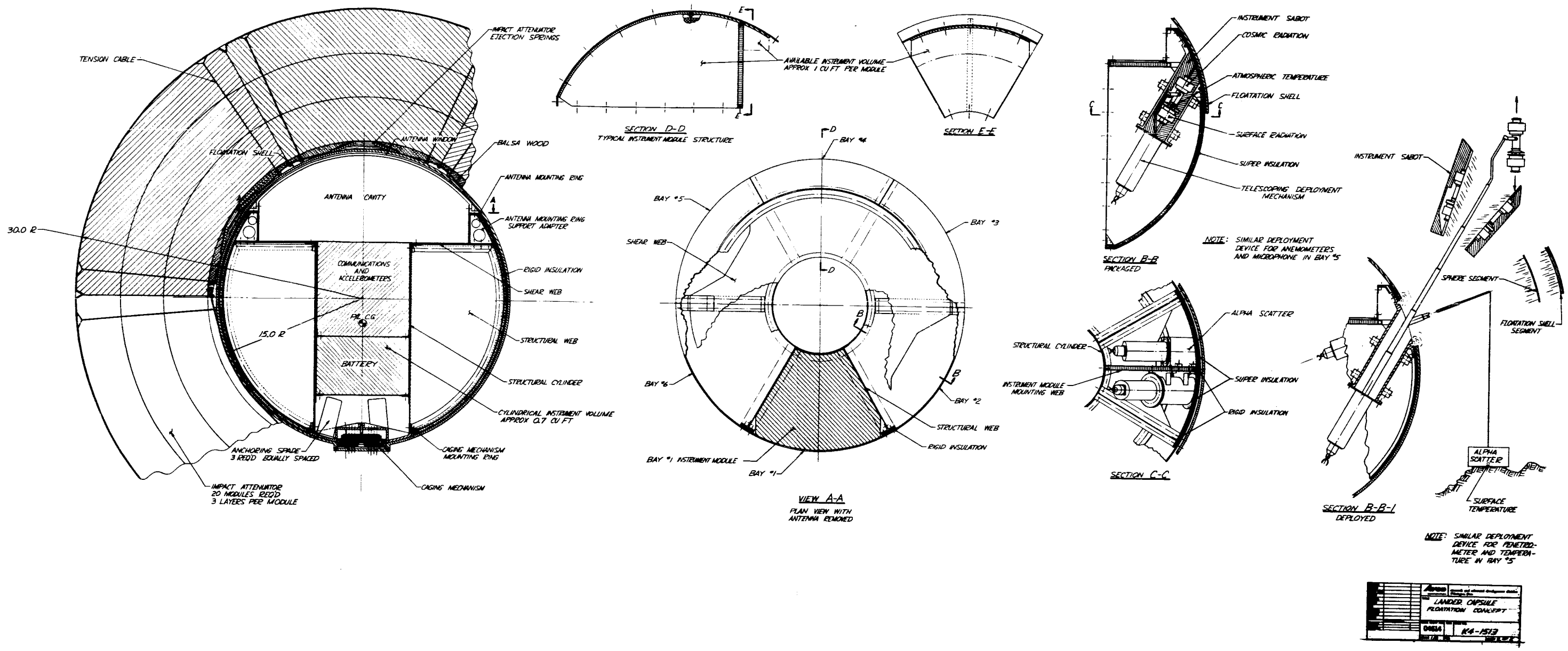


Figure 15 FLOTATION SPHERE LANDED CAPSULE

37-1

Until impact, the two spheres are united by the caging umbilical. After the landed capsule has come to rest, the impact attenuation material is ejected and the caging mechanism is released. The inner sphere then orients itself to the local vertical at which time three anchoring devices are fired through the flotation shell and impact attenuator (if it is still there) into the Martian surface. These devices have an uncoiling wire mechanism that is attached between the spade (that portion fired) and the inner sphere which holds the sphere firm to the surface. Three are used for redundancy and stability. These anchoring devices also recage the two spheres together.

The deployment of scientific instrumentation is accomplished by cutting small ports through both spheres with linear-shape charges mounted in the inner sphere around the instrument deployment opening. Here, again, a sabot is utilized around the science instrument to absorb the shock loads during impact. The deployment system is exactly the same as that presented for the oblate spheroid.

The selection of a flotation fluid is predicated on meeting several design requirements: it must be RF transparent; it must have a high boiling point of 300°F, due to sterilization; it must have a low freezing point of 0°F, due to space environment; it must have a low viscosity and finally, it must have the exact density of the inner sphere. For the conceptual design Freon-L3 (a flourocarbon-developed by DuPont) was used. It satisfies all of these requirements except possibly the density requirement. However, the density can be easily varied in this fluid to meet the required value. Further development would be necessary, by taking into account its compatibility with adjacent materials, before final decisions as to the flotation fluid are made.

Thermal control of the landed capsule could become a critical problem in this design. During the Martian night the landed capsule will lose a significant portion of its internal heat because of the highly emissive surface, brought about by the dielectric requirements of the relay (VHF) and direct (S-band) antenna systems. That is, the surface must be a nonmetallic and hence a high emitting surface, whereas it would be desirable to have a poor emitting metallic surface. This problem area could be overcome to a certain extent by utilizing a radiation barrier on the inside of the structure such as super insulation (i.e., aluminized Mylar).

3.3 OPERATIONAL SEQUENCE

In the process of the flight capsule preparing for landing, and in the landing itself, several mechanical separations occur. Electronic warmup and

communications activity are discussed further in other sections of the report. The mechanical activity results in approximately 16 major functional operations between the cruise mode and the instrument deployed condition on the planet surface. This is clearly illustrated in Figure 16 and in the following operation descriptions:

1. The first step, after the cruise mode, is taken from 20 to 2 days prior to encounter with Mars. An electric current is applied to the ignition circuit of the canister cutting charge. This ignites the detonators which in turn detonate the shaped charge.

   The explosion of the shaped charge makes a circumferential cut around the forward part of the lid and the impulse from the explosion forces the lid away from the canister and the separated vehicle.

2. The second step is initiated by an electric signal to the cable cutters on the separated vehicle forward adapter separation system. With this signal the cable-cutter ordnance is detonated and the cables are cut. The cables release the four clamps, and the six springs eject the separated vehicle from the spacecraft adapter. All clamps and cables are tethered to the forward adapter and thus stay with the canister base.

3. Immediately after separation the attitude control system assumes control of the separated vehicle. It stabilizes the vehicle, removing perturbations induced in the separation, and then reorients the vehicle so the thrust axis is parallel to the desired direction of thrust application.

4. As the separated vehicle stabilizes in its new attitude, the ΔV rocket motor is initiated and induces the required incremental velocity with a resultant vector to cause the vehicle to impact the planet. With this velocity correction at the thrust application angle the vehicle will have a 3 hour lead time over the flight spacecraft at planet impact. The attitude control system holds the correct separated vehicle orientation during the firing of the ΔV rocket motor.

5. After thrust termination the control system reorients the separated vehicle to the correct attitude for a zero-entry angle of attack.

6. Rockets on the outer rim of the entry shell are then ignited imparting a spin to the separated vehicle of 10 rpm to stabilize its attitude for the remainder of its space travel.

7. The attitude control has completed its mission at this point. The cable cutters are actuated, severing the restraining cable on the

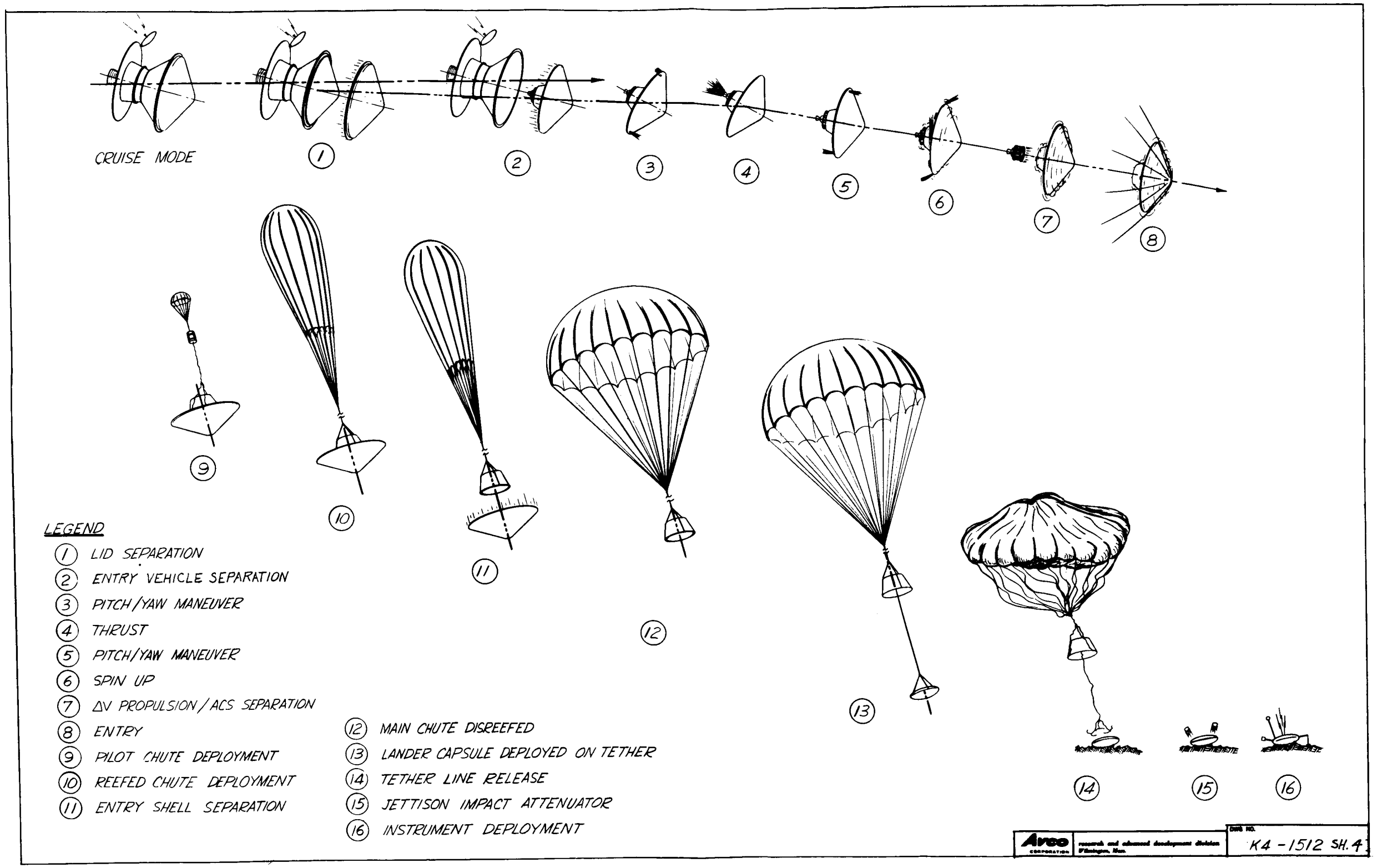


Figure 16 FLIGHT OPERATIONAL SEQUENCES

40-2

clamps securing the attitude control electronics ΔV rocket case and support. The support, with the electronics and the rocket case are then ejected by a spring force. With the discarding of the ACS ΔV rocket support, the separated vehicle is trimmed of all excess weight and becomes an entry vehicle committed to a set ballistic entry trajectory.

8. The entry vehicle enters the planet atmosphere.

9. At approximately 21, 000 feet the cover of the parachute container is ejected and the pilot chute mortar is fired. The pilot chute is propelled at an angle rearward to take it outside the vehicle wake.

10. The pilot parachute pulls the parachute cover and the parachute out of the container and into the airstream. The cover is released from the parachute as the end of the riser lines are reached allowing the cover and the pilot parachute to be discarded. At this time the main chute is in a reefed position.

11. The shock as the reefed parachute reaches the end of the riser line produces a high loading pulse that peaks suddenly then reduces to a nominal value and finally to the terminal velocity value. As the loading passes the peak value, any vehicle perturbations produced by the opening tend to vanish. At this time the entry shell ball locks are activated and the entry shell is separated from the suspended capsule.

12. At approximately 15, 000 feet the reefing lines are cut and the main parachute opens to its full size.

13. Atmosphere sampling will be continued throughout the entire entry phase. However, at about 500-feet altitude the radar altimeter will signal for the electrical disconnect of the landed capsule and also for the severence of the cable holding the capsule sling. This will allow the capsule to drop to the end of a 20-foot tether.

14. Upon striking the surface, the internal accelerometers signal the tethering explosive bolt disconnect, thus propelling the tether assembly from the vicinity of the landed capsule.

15. The accelerometers sense the static condition of the landed capsule when it comes to rest, and the initiation of the explosives discard selected segments of the impact attenuator material from the capsule.

16. The final action is the deployment of the instrumentation. On the oblate spheroid the holes through the structure are opened on ejection of the impact attenuation material by explosive sheets and the signal is immediately initiated to deploy the instruments. Three groups of instruments are deployed by telescoping deployment mechanisms actuated by a gas generator.

In case of an indication of attitude control failure prior to separation of the separated vehicle from the flight spacecraft, the procedure can be changed as follows:

1. From the cruise mode attitude, the sterilization canister lid is separated from the canister.

2. The flight spacecraft is reoriented to the proper thrust application angle.

3. The separated vehicle is separated from the flight spacecraft.

4. Immediately, the separated vehicle is spun up to 50 rpm.

5. The ΔV rocket is fired.

6. The ΔV rocket/attitude control electronics is ejected.

7. The separated vehicle is de-spun to approximately 10 rpm.

The remaining steps from entry to instrument deployment, are the same as the nonfailure mode steps.

3.4 WEIGHT AND BALANCE

Included in this section are the weight and balance summaries of the reference and alternate Mars Probe/Lander designs. The weights presented are the outcome of detail design layout described in the previous sections, tradeoff studies and landed capsule synthesis derived in Section 6.0 of this book.

3.4.1 Oblate Spheroid Landed Capsule Design

Presented in Table IV is the weight summary of the reference design. The breakdown starts with the complete flight capsule weight of 2500 pounds (limit defined by study ground rules) and proceeds to remove the various major subsystem weights to arrive at the next system weight category. This method of weight summary is in concurrence with the operational sequence discussed previously. The 2500-pound weight limit on the flight capsule limits the entry vehicle weight to 1850 pounds due to the fixed weights of the sterilization canister and propulsion units. An M/C_DA of

0.20 slug/ft^2 is required to obtain the entry and landing mission objectives.

Most of the weights presented on Table IV are calculated from given subsystem characteristics generated from preliminary analysis. Only the electrical connectors, thermal control coating weights were estimated since they depend on rather detailed designs.

Table V gives the detail weight breakdown for the entry shell primary structure, heat shield, and afterbody.

In the primary structure weight breakdown the landed capsule mounting ring and bearing pad weights are included. These weights are included to indicate the total entry shell structural weight that is separated at parachute deployment. A contingency factor of 15 percent has been included in the entry shell primary structure to account for miscellaneous brackets and fittings, that cannot be determined in a conceptual design. Included in the heat shield weight breakdown is the heat shield on the aft side of the entry shell from the base ring to the suspended capsule mounting ring (secondary heat shield). Afterbody refers to the enclosure (heat shield and structure) over the suspended capsule in this design the short truncated cone running between the suspended capsule mounting ring to the ΔV propulsion tiedown.

Tables VI and VII give the weight breakdown for the instrumentation, telecommunications and power respectively for the weight summary presented in Table IV. The instrumentation weight breakdown as presented in Table VI is the recommended instrumentation list to accomplish the required mission objective.

Those instruments marked with a double asterisk on this list were removed from the reference design, along with their portion of the cabling and bracketing weight in order to meet the required weight allocation of the landed capsule as indicated in the weight summary (see Section 6.3). These instruments, however, were included in the alternate design where the flotation sphere landed concept was used.

Similarly that equipment marked with an asterisk on Table VII which represents the relay link equipment for the landed capsule telecommunications, were also removed from the reference design in order to meet the weight restrictions. Again, these equipment were included in the alternate design.

A contingency factor of 25 percent was applied to the suspended and landed capsule instrumentation, telecommunication, and power weights to account for unknown factors that could not be predicted in a conceptual design. The structure of the landed capsule was estimated as 35 percent of the internal weight less the thermal control weight which is an estimated value.

TABLE IV

FLIGHT CAPSULE WEIGHT SUMMARY
OBLATE SPHEROID LANDED CAPSULE

	Weight (pounds)		c. g.* (inches)	I_{xx} (slug/ft^2)	I_{yy}
Flight Capsule		2500.0	42.2	1259.1	861.3
FC-FS adapter	100.0				
Sterile canister	366.9				
Electrical and mechanical connectors	50.0				
Separated Vehicle		1983.1	38.0	809.5	561.6
ΔV propulsion	98.5				
ACS electronics	10.0				
Spin propellant	2.1				
Propulsion supports	10.0				
Miscellaneous	12.5				
Entry Vehicle		1850.0	35.6	808.9	526.3
Entry shell heat shield	290.0				
Entry shell structure	451.2				
Thermal control	25.0				
ACS nozzles, tanks, etc.	69.3				
Spin rockets and supports	10.0				
Electrical and mechanical connectors	55.5				
Contingency	23.0				
Suspended Capsule		926.0	41.6	139.3	140.7
Instrumentation	35.3				
Telecommunications	20.6				
Power	33.0				
Miscellaneous	4.6				
Contingency (25 percent on above)	23.1				
Parachute	74.0				
Structure	120.0				
Afterbody heat shield	21.0				
Landed Capsule		595.0	32.0	39.3	39.3
Impact attenuator	215.0				
Electrical and mechanical connectors	15.5				
Internal Weight		364.5			
Instrumentation	48.0				
Telecommunications	98.7				
Power	70.1				
Miscellaneous	2.0				
Contigency (25 percent on above)	54.7				
Thermal control	15.0				
Structure	76.0				

*Note: Center of gravity (c. g.) is from entry shell nose

TABLE V

ENTRY SHELL WEIGHT BREAKDOWN

		Weights (pounds)
Primary Structure		
Base ring		61.4
Beryllium face sheets		102.5
Stainless steel core		63.5
Stainless steel weldments		25.3
Beryllium splice plates		35.5
Epoxy bond		26.0
Contingency (at 15 percent)		50.0
Beryllium mounting rings		47.0
Foam bearing pad - Landed Capsule		40.0
	total	451.2
Heat Shield		
Cork primary heat shield		213.0
Epoxy bond		27.0
Cork secondary heat shield (t = 0.125 inch)		41.5
Epoxy bond		8.5
	total	290.0
Afterbody		
Cork heat shield (t = 0.125 inch)		16.5
Bond		4.5
	total	21.0

TABLE VI

INSTRUMENTATION WEIGHT BREAKDOWN

		Weight (pounds)
Science Payload External		
**Trapped radiation (1)		2.2
Pressure (2)		0.6
Temperature (2)		0.6
**RF probe		1.3
Acoustic densitometer		3.0
Mass spectrometer		8.0
Gas chromatograph		4.8
Radar altimeter		8.0
Radiometer		2.25
Beta scatter		0.8
Cables		5.0
Brackets		3.85
	total	40.4
Science Payload Internal		
*Accelerometer (3)		1.8
**Impact accelerometer (3)		1.0
Gas chromatograph		4.8
Water detector		0.5
Pressure (2)		0.6
Atmospheric temperature (2)		0.6
Acoustic densitometer (2)		3.0
Cosmic radiation (2)		4.4
Surface radiation (2)		4.0
**Hot wire anemometer (2)		1.0
Force anemometer (2)		4.0
Microphone (2)		1.0
**Penetrometer (2)		2.0
Surface temperature (4)		1.2
Alpha scatter (2)		1.4
Cables		19.0
Brackets		7.4
	total	57.7

*Accelerometers are part of the descent and entry instrumentation. However, they are contained inside the Landed Capsule since they must be near the Entry Vehicle center of gravity.

**These instruments are removed for reference design but are included in alternate design.

TABLE VII

TELECOMMUNICATIONS AND POWER WEIGHT BREAKDOWN

	Weight (pounds)
SUSPENDED CAPSULE	
Telecommunications	
Data handling and telemetry	5.0
Cabling	5.0
Power conditioning	2.0
Brackets and mounting hardware	1.0
Relay antenna	2.5
Relay transmitter (1)	3.0
Miscellaneous RF relay	2.0
Command receiver antenna	0.1
total	20.6
Power	
Batteries	33.6
LANDED CAPSULE	
Telecommunications	
Direct antenna (2)	2.5
*Relay antenna (2)	5.0
Direct transmitter (2)	12.0
Miscellaneous RF hardware direct	5.0
RF power supply direct	3.5
*Relay transmitter (1)	3.0
*Miscellaneous RF components relay	2.0
Command receiver/decoder (2)	5.0
Central control and sequencer	9.6
Data automation equipment	5.5
Telemetry system	3.0
Storage	2.0
Cables	22.0
Power conditioning	12.0
Brackets and mounting hardware	9.7
Diagnostic monitors	4.0
Engineering data transducers	4.0
total	108.7
Power	
Batteries	70.1

*These items are removed for the reference design but are included in the alternate design

3.4.2 Flotation Sphere Landed Capsule Design

As discussed in Section 3.2, the only changes in the alternate design from the reference design are the landed capsule configuration, the suspended capsule structure, and the VHF relay antenna location. The weight breakdown of the alternate design is exactly the same as the reference design down to the suspended capsule weight category as presented in Table IV. Table VIII presents the alternate design flight capsule weight breakdown. The basic changes in weight occur primarily in the instrumentation, power and impact attenuator of the landed capsule. The instrumentation weight breakdown as presented in Table VI was used except that only one each of the following deployable instruments is required in the flotation sphere concept where vertical direction is assured:

- Acoustic Densitometer
- Cosmic Radiation
- Surface Radiation
- Hot-Wire Anemometer
- Force Anemometer
- Microphone
- Penetrometer
- Surface Temperature
- Alpha Scatter

The reduction in the number of instruments and the number of communications antennas, along with a higher gain antenna permit a reduction in battery weight. This, together with the higher efficiency of the flotation sphere landed capsule shape, resulted in a large reduction in attenuator weight. The total landed capsule weight is reduced from 595 pounds to 572 pounds, thus allowing more instrumentation and power capability on the suspended capsule as indicated in the weight summary. The contingency on the entry vehicle weight was also increased by 10 pounds.

TABLE VIII

FLIGHT CAPSULE WEIGHT SUMMARY
FLOTATION SPHERE LANDED CAPSULE

Flight Capsule		2500.0
FC-FS Adapter	100.0	
Elect. and Mech. Connectors	50.0	
Sterilize Canister	366.9	
Separated Vehicle		1983.1
ΔV Propulsion	98.5	
ACS Electronics	10.0	
Spin Propellant	2.1	
Propulsion Supports	10.0	
Misc.	12.5	
Entry Vehicle		1850.0
Entry Shell Heat Shield	290.0	
Entry Shell Structure	451.2	
Thermal Control	25.0	
ACS Nozzle, Tanks, etc.	69.3	
Spin Rockets and Supports	10.0	
Elect. and Mech. Connectors	55.5	
Contingency	33.0	
Suspended Capsule		916.0
Instrumentation	40.4	
Telecommunications	20.6	
Power	37.5	
Miscellaneous	4.0	
Contingency (25% on above)	26.0	
Parachute	74.0	
Structure	120.0	
Afterbody Heat Shield	21.0	
Landed Capsule		572.5
Impact Attenuator	186.9	
Elect. and Mech. Connectors	15.5	
Internal Weight		370.1
Instrumentation	46.7	
Telecommunications	98.7	
Power	63.0	
Miscellaneous	2.0	
Contingency (25% on above)	52.6	
Thermal Control	15.0	
Structure	92.1	

4.0 SUBSYSTEM CHARACTERISTICS

4.1 COMMAND AND PROGRAMMING

4.1.1 Objectives and Requirements

The Central Computer and Sequencer (CC&S) Subsystem performs timing, sequencing and associated computational functions for the flight capsule. Computations are performed to solve equations involving time, acceleration and altitude. A master time base is provided for use in all computations and sequences. The initiation of all events in appropriate order is provided by properly timed outputs to other subsystems.

There are four sequences to be controlled. These may be characterized by the mission phases associated with them. They are the separation, cruise, entry-descent, and landed sequences. These sequences are initiated in one of two ways: by a time-determined signal (for example, a signal from the cruise sequence) or by the occurrence of a non-timed event (for example, the indication by the radar altimeter that a certain altitude has been reached). The cruise sequence is initiated by MOS command through the flight spacecraft CC&S. The separation and entry-descent sequences are initiated by the cruise sequence. The landed sequence is initiated by an accelerometer at impact.

4.1.2 Subsystem Mechanization

Two separate CC&S subsystems comprise the overall Central Computer and Sequencer. The first subsystem, used from preseparation to impact, is located outside the flight capsule. The other subsystem, inside the flight capsule, is employed from impact through the end of the mission.

4.1.2.1 External CC&S Functional Description

The external CC&S provides timing and sequencing services for the flight capsule subsystems from before separation to impact. This CC&S initiates flight capsule events in three different sequences as follows:

1. Separation Sequence -- The separation sequence controls all flight capsule events from 240 minutes before separation to 15 minutes after separation. During this time, the external CC&S initiates preseparation checkout (external mode 1) and separation. The events initiated are as follows:

Apply test voltage to ACS and gyros

Turn FC systems on

Turn all FC systems off (except ACS and gyros)

Transmit FC direct link via FS

Turn off FC direct link via FS

Separate

Actuate external mode 1 (preseparation checkout)

Start postseparation timers No. 1 and No. 2

Initiate FC ΔV maneuver

Terminate ΔV propulsion

Initiate FC attitude control maneuver

Jettison propulsion structure

2. Cruise Sequence -- The cruise sequence controls all flight capsule events during the cruise phase after separation. During this time, external mode 2 (postseparation checkout) is initiated once each day. Except for those times during which it is operating in external mode 2, the CC&S operates the flight capsule in external mode 5, a standby mode.

3. Entry-Descent Sequence -- The entry-descent sequence controls all flight capsule events from 5 minutes before entry to impact. During this time the following major events are initiated by the external CC&S:

Start External Mode 2 (pre-entry checkout)

Start External Mode 3 (entry data storage)

Start External Mode 4 (entry-descent data transmission)

Arm entry and landed pyrotechnics

Deploy reefed parachute

Disreef parachute

Deploy science probes

Tether Lander Capsule

Figure 17 shows a functional block diagram of the external CC&S. This figure shows both primary and backup sources of initiation signals, and the major events initiated for each phase of the mission controlled by the external CC&S. The external CC&S contains the following major items:

Master timer

Cruise sequencer matrix

Separation sequencer matrix

Entry-descent comparison circuits

4.1.2.2 Internal CC&S Functional Description

The internal CC&S provides timing and sequencing services for the flight capsule subsystems from impact through the end of the mission. It initiates flight capsule events in a single sequence, called the landed sequence, which starts when the payload is tethered and ends with the end of the mission. During this period, the following events are initiated by the internal CC&S.

Energize internal mode 1 (impact data storage)

Jettison chute and external payload

Terminate direct link transmission

Jettison impact attenuator

Deploy cosmic radiation detector, and atmospheric temperature probe

Surface radiation detector

Deploy alpha scatter and surface temperature detectors

Deploy anemometer and microphone

Energize internal mode 2 (direct link transmission)

Terminate direct link transmission

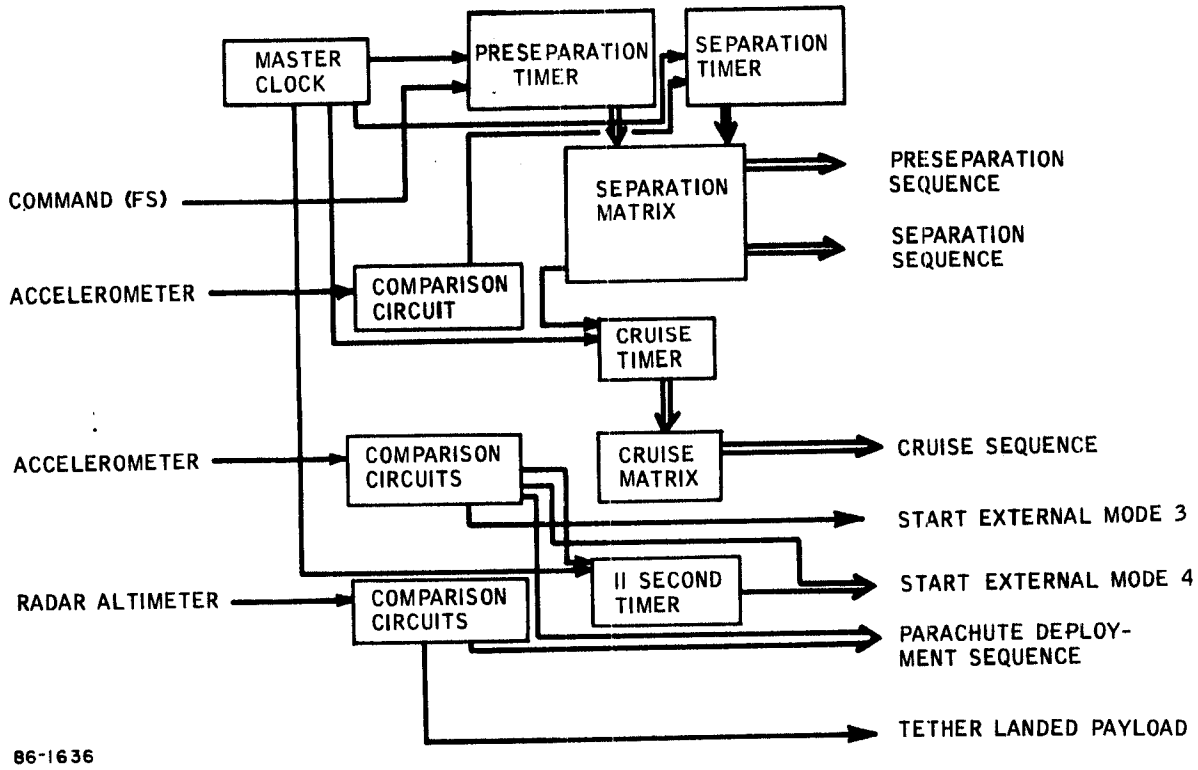


Figure 17 FC EXTERNAL CENTRAL COMPUTER & SEQUENCER

Figure 18 is a functional block diagram of the internal CC&S subsystem. This figure shows both primary and backup initiation signal sources and the major events initiated by the internal CC&S. The internal CC&S contains a master timer and a post impact sequencer matrix.

It is desirable for all sequences to be initiated by functionally redundant signals. However, during certain times, functionally redundant initiation is not possible (such as during portions of the post-impact science mission). In such cases, redundancy is obtained through direct link command from Earth whenever possible; this capability exists only during the portion of the Martian day when there is mutual visibility. Further redundancy is obtained by using block-redundant master timers in both CC&S subsystems.

4.2 SCIENTIFIC AND ENGINEERING INSTRUMENTATION

4.2.1 Objectives and Requirements

The mission objectives specified by the Langley Research Center (LRC) were to provide data necessary for the design of future flight capsules (engineering objectives) and to provide the data necessary for the design of future experiments which would definitively elucidate the nature of the planet Mars, including its biological, geological, and meteorological phenomena both past and present (scientific objectives). The scientific and engineering instruments were selected to meet these objectives. The primary goal of the payload selection task was to approximate as closely as possible the actual payload which the National Aeronautics and Space Administration in concert with the scientific community would choose. The actual method of payload selection and the experiments which were considered are given in Volume V, Book 4, Sections 2.3 and 2.4.

The most important objective of the scientific and engineering instrumentation is the measurement of the Martian atmospheric density profile. Knowledge of this profile will allow the design of future entry vehicles which do not include large design contingencies for atmospheric density uncertainties. A second important objective is the determination of the wind velocity spectrum near the surface. This information is essential for the design of hard landers which utilize parachutes for their terminal descent phase. Although knowledge of the density profile will allow the design of parachute systems which will reduce the vertical velocity component to acceptable levels, unless the horizontal (wind induced) component is known very conservative design of the impact attenuation system will be necessary. The need for wind velocities is less critical for the design of soft landers which utilize retropropulsion systems for the terminal descent phase. The basic capabilities of these systems must be so large that the added cost of the propulsion to counter even the most severe winds is not

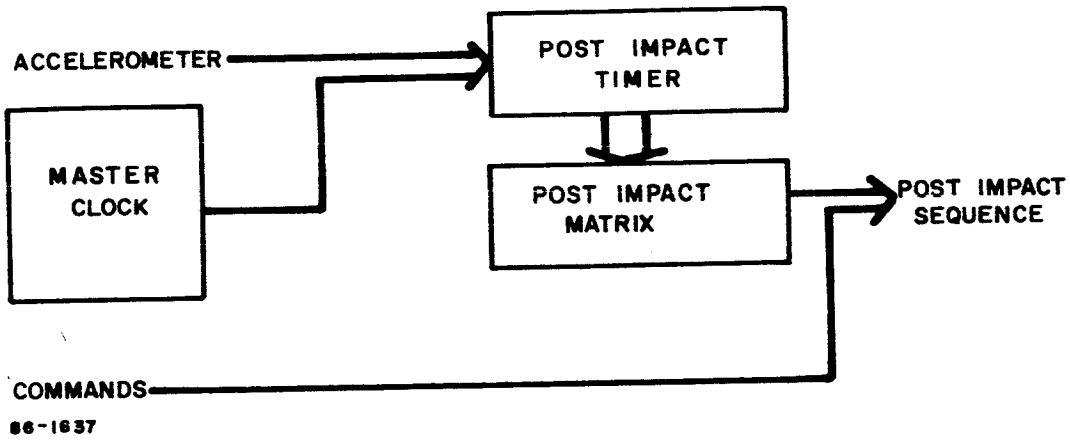


Figure 18 FC INTERNAL CENTRAL COMPUTER AND SEQUENCER

very significant. An additional requirement for wind velocity data comes from the landed capsule and its projected lifetime. If landed mission durations of the order of weeks or months are planned, then the probability of damage to sensitive components or even burial of the entire capsule by windblown sand must be evaluated, and possible countermeasures planned. The third most important objective is the definition of the nature of the Martian surface, i.e., its bearing strength, hardness, and large- and small-scale contours. These data are essential for the design of landed capsules. A fourth objective is to obtain information which will allow the formulation of definitive life detection and characterization experiments. This may be accomplished by describing the near-surface environment in which such organisms are most likely to be found. The aspects of this environment which are of the most interest include surface and atmospheric chemical composition and temperature, radiation (both electromagnetic and particulate), and surface physical properties. Almost all of these aspects are also useful in the engineering design of future missions. Additional lower ranked objectives of geophysical interest include the characterization of the ionosphere, the magnetosphere, and the planetary interior.

Certain basic ground rules were imposed on the payload studies by LRC. Television, either during the entry, parachute descent, or landed phases was not to be considered for the 1971 mission. Similarly, active biological experiments, i.e., experiments seeking to detect Martian life forms by measuring growth or processes associated with metabolism were to be excluded from consideration. Chemical composition experiments were not, however, to be excluded. A mobile lander was not to be considered. This was understood to mean that, while roving vehicles which might leave the lander to obtain samples were beyond the scope of the study, simple sample collecting devices were allowable. The payload instrumentation was, of course, also required to withstand the various environments to be encountered during the mission, e.g., the sterilization qualification test, the 500- to 1000-g impact shock, and the lower amplitude but longer duration entry deceleration. Finally, the selected instrumentation had to have a development status such that it would be available for the mission. This was defined as being available for test in prototype form by 1 September 1966, based on accelerated development programs initiated in late 1965.

4.2.2 Subsystem Design

From the objectives and requirements discussed in the previous section and from studies of possible experiments, their interface requirements, and their performance, a scientific and engineering instrumentation payload has been developed. The payload actually consists of two physically separate systems, one system which is (with the exception of the accelerometers) located external to the landed capsule, for entry and parachute

descent measurements, and a second system which is located wholly within the landed capsule, for postlanding measurements. It should be noted that, although the experiments themselves are fairly rigidly separated, many of the experiment support functions, particularly communications and power, are much less sharply divided by the walls of the landed capsule. The selected instruments and some of their requirements are listed in Tables IX and X. The instruments in these tables are listed in order of decreasing priority. As the flight capsule design evolved (particularly as the oblate spheroidal landed capsule was selected), it became apparent that all of the experiments could not be carried because of weight limitations. Thus those experiments marked with asterisks in the tables were not included in the reference design. These deletions have eliminated from the payload experiments which will provide data on the physical nature of the Martian surface. Recent reevaluations of the need for such data have indicated a greater requirement than was originally recognized. In any further work on this mission concept it is recommended that serious consideration be given to the inclusion of the penetrometer experiment at the expense, if necessary, of the landed acoustic densitometer.

These tables show that certain instruments are duplicated in the two systems. That is, there are two gas chromatographs, one in the suspended payload and one within the landed capsule. This situation also prevails with the acoustic densitometer, the atmospheric temperature probes and the atmospheric pressure probes. The selection of the oblate spheroidal landed capsule introduces an additional multiplicity into the instrumentation, since those experiments which require orientation relative to the local vertical must be duplicated to allow for the two possible rest attitudes of the capsule. Instruments which are in this category include: cosmic radiation, surface radiation, hot-wire anemometer, force anemometer, particle microphone, penetrometer, surface temperature, and alpha scatter.

Table XI shows how the candidate scientific instrumentation payload meets the objectives which were discussed in Section 4.2.1. In this table an "X" is placed under an objective if the instrument in question provides information which may satisfy the objective either alone or in a simple combination with other instruments. Certain higher level combinations have been excluded. It may be seen that all of the more important objectives are supported by information from at least three instruments, and in one case by as many as ten instruments. This actually understates the redundancy obtained, since some of the instruments are duplicated either within a given portion of the mission or between the descent and landed missions. Questions of excessive redundancy might well be raised, but much of this redundancy is functional rather than block redundancy. Thus many of the experiments which appear redundant for a given objective are in fact complementary. Under atmospheric composition, the radiometer, the

TABLE IX

DESCENT SCIENCE AND ENGINEERING INSTRUMENTATION

Instrument	Weight (pounds)	Volume (in^3)	Power (watts)	Outputs per Measure	Bits per Measure	Sampling Rate
Accelerometer (3)	1.8	15	2.0	3	30	1/second
Radar altimeter	8.0	100	10.0	1	7	1/second
Atmospheric pressure (2)	0.6	6	0.3	2	14	1/2 second
Atmospheric temperature (2)	0.6	6	0.2	2	14	1/2 second
Gas chromatograph	4.8	200	4.0	8	56	1/8 second
Acoustic densitometer	3.0	49	4.0	3	21	1/8 second
Radiometer	2.2	7	1.5	6	42	1/second
Mass spectrometer	8.0	400	10.0	6	42	1/8 second
Beta scatter	0.8	15	0.2	1	7	1/4 second
*RF probe	1.3	2	0.3	1	5	1/0.5 second
*Trapped radiation	2.2	100	0.8	1	24	1/5 minute
Descent total	29.8	798	32.2			

*Not included in final payload selection or in column totals.

TABLE X

LANDED SCIENCE AND ENGINEERING INSTRUMENTATION

Instrument	Weight (pounds)	Volume (in^3)	Power (watts)	Outputs per Measure	Bits per Measure	Sampling Rate
Force anemometer (2)	4.0	28	1.0	6	42	5 times
Particle microphone (2)	1.0	66	0.4	2	14	5 times
Water detector	0.5	16	1.0	1	7	1/hour
Atmospheric temperature (2)	0.6	6	0.2	2	14	2/hour
Atmospheric pressure (2)	0.6	6	0.3	2	14	2/hour
Gas chromatograph	4.8	200	4.0	8	56	5 times
Surface temperature (2)	1.2	12	0.2	2	14	1/hour
Cosmic radiation (2)	4.4	200	0.4	1	7	1/hour
Surface radiation	4.0	200	0.4	1	7	1/hour
Alpha scatter (2)	1.4	23	1.4	200	1000	5 times
Acoustic densitometer	3.0	49	4.0	3	21	1/hour
*Penetrometer (3)	2.0	42	0.9	300	2100	1 time
*Impact accelerometer (3)	1.0	15	2.0	300	2100	1 time
*Hot-wire anemometer	1.0	8	0.5	6	42	5 times
Landed total	25.5	806	13.3			
Mission total	55.3	1604	45.5			

*Not included in final payload selection or in column totals.

TABLE XI

RELATIONSHIP OF INSTRUMENTS TO OBJECTIVES

Instruments	Objectives	Atm. Density Profile	Wind Velocity	Surf. Phys. Charact.	Atm. Comp. Profile	Atm. Temp. Profile	Surf. Comp.	Surf. Temp.	Radiation Env.	Geophysics
Accelerometer		X			X					
Radiometer					X					
Atmospheric temperature		X			X	X				
Atmospheric Pressure		X			X	X				
Mass spectrograph		X			X					
Gas chromatograph		X			X					
Radar altimeter		X			X	X				
Acoustic densitometer		X			X	X				
Beta scatter		X			X	X				
Impact accelerometer				X						
Penetrometer				X						
Water detector					X					
Cosmic radiation									X	X
Surface radiation									X	X
Force anemometer			X	X						
Hot-wire anemometer			X	X						
Particle microphone			X	X						
Surface temperature								X		
Alpha scatter							X			

mass spectrometer, the gas chromatograph, the acoustic densitometer, and the water detector each performs a composition analysis task uniquely well, but no single instrument provides all of the desired information. The radiometer provides data on the major components of the upper atmosphere (carbon dioxide, nitrogen, and probably argon) very early in the mission. This is desirable both because few other instruments may be effective at this altitude on an entering vehicle, and because of the possibility of a catastrophic failure later on in the mission. The mass spectrometer will analyze for a series of specific atmospheric components and do it very well, but instruments appropriate for missions of this type cannot detect small amounts of carbon monoxide in either nitrogen or carbon dioxide (a biologically important analysis). The gas chromatograph on the other hand will handle the carbon monoxide well but has trouble separating oxygen and argon, a trivial analysis on the mass spectrometer.

The acoustic densitometer is present primarily to determine density, but it also allows the measurement of mean molecular weight and the heat capacity ratio, C_p/C_v. These latter data will provide a check on the element-specific detectors so that if an unexpected major component is present, some estimates of its identity will be available. The water detector provides an estimate of the water content, something none of the other instruments do well at the low concentrations predicted for Mars. In addition, the accelerometer, the atmospheric temperature probes, the atmospheric pressure probes, the beta scatter instrument, and the radar altimeter are credited to the atmospheric composition objective. The first four of these serve primarily to allow the reconstruction of the density profile, but knowing density, pressure, and temperature, the mean molecular weight can be calculated to provide, as noted above, a check on the element-specific sensors. The radar altimeter provides altitude reference points for the profile. Thus although an apparent ten-fold redundancy exists, none of the instruments are present solely for the purpose of building redundancy. Each serves a well defined and independent role. Similar situations exist with other objectives having a high redundancy in instruments, although there actually are a few cases of almost pure block redundancy, e. g., the acoustic densitometer and the beta scatter instruments under the density profile objective.

4.2.3 Subsystem Performance

Some information on the performance of the science and engineering instrumentation has been given in the previous section. However, it is the purpose of this section to describe in detail the functions of the various instruments. The three accelerometers will be mounted as close to the center of gravity of the entry vehicle as possible with each accelerometer input axis parallel to the pitch, roll, and yaw axes of the vehicle respectively. By sampling the decelerations experienced by vehicle between

entry and Mach 1, data will be obtained which, by the application of suitable reduction methods, will allow the reconstruction of the atmospheric density profile in this region. Backup for this calculation will be provided by the data from the radar altimeter, the pressure probes, and the temperature probes. A failure of almost every other experiment and even the loss of communications during the later phases of the mission could not compromise the achievement of the critical parts of the most important mission objective as long as the accelerometers function.

The radar altimeter will provide altitude data from the end of blackout until impact. Use of the entry shell structure as the high altitude antenna will allow this broad range of operation without excessive power requirements. After separation of the entry shell, the altimeter will shift from the low frequency (18 MHz) to the higher frequency (324 MHz) system which includes its own antenna. The data obtained will provide reference points for the data taken by the other instruments. The time versus altitude information will allow the reconstruction of a portion of the density profile independent of the accelerometer experiment. In addition, with a single altitude reference point, the accelerometer experiment can be used to construct a large portion of the density profile even though low altitude data are not obtained.

The atmospheric pressure probes will be so placed on the capsule that one will read stagnation pressure at the nose, and the other will read a pressure which may be related to the free-stream pressure. From these data the velocity of the vehicle may be calculated. This velocity will be used in the reconstruction of the atmospheric density profile. The pressure measurement is, of course, also of great interest for its own sake. The lander pressure measurements are useful only in this latter regard. The pressure instrumentation to be used will be either vibrating plate transducers or diaphragms with solid-state force sensors.

The atmospheric temperature probes will be identical total temperature probes which will provide data from which static temperature may be calculated. At lower velocities the measured total temperature will approach the static temperature. The experiment will give information of use in the reconstruction of the density profile, of meteorological interest and also of use in the design of thermal control systems for future missions. The landed instrument will provide data applicable only for the last two uses. The surface temperature probe will, in addition to providing meteorological and thermal control data, also provide some insight into the nature of the surface materials from the variation of their temperature over the diurnal cycle.

The gas chromatographs will detect the concentrations of atmospheric argon, carbon dioxide, carbon monoxide, krypton, neon, nitrogen, oxygen,

and xenon, employing a miniature multiple column, multiple detector system. These are the major gases whose presence is anticipated in the Martian atmosphere. The presence of major amounts of other gases, as perhaps methane or nitrogen oxides, will either not be detected or will degrade the planned analyses. However, several other experiments will allow the calculation of mean molecular weight or will measure individual components by other means.

The acoustic densitometers will transmit measurements of the acoustic velocity, acoustic impedance, and temperature of contained atmospheric samples. From these data, the atmospheric density, mean molecular weight, and heat capacity ratio (C_p/C_v) will be calculated. The experiment will provide low altitude atmospheric density data to complement the higher altitude accelerometer data and will allow an independent check of the analyses of the major components of the atmosphere.

The radiometer is actually a group of six ratio-measuring radiometers which will measure the intensities of six spectral lines in the emissions from the shock-excited atmospheric gases in front of the stagnation point. From the intensities of these lines the amounts of nitrogen, carbon dioxide, and argon may be determined. The main advantage of this experiment is that it obtains data during the peak heating portion of the entry mission in which it is difficult to obtain valid data from most composition measuring instruments. Thus this experiment, and the accelerometer experiment, provide excellent insurance in the event of failures during the later part of the mission.

The mass spectrometer will utilize a quadrupole mass spectrometer to measure the amounts of five preselected atmospheric components as well as to obtain an estimate of total pressure. The instrument is stabilized to assure that the five selected mass peaks will be correctly focused. The elements to be determined are argon, carbon dioxide, neon, nitrogen, and oxygen. As noted in the previous section, the mass spectrometer and the gas chromatograph rather nicely complement each other to provide with high reliability a broad range of analyses.

The beta scattering experiment will utilize the back scattering of beta particles from atmospheric molecules to obtain estimates of atmospheric density. The beta particles are generated by a radioisotopic source and detected by counters which are shielded from the direct emissions from the source.

In the landed capsule payload the most critical experiment is probably the wind velocity measurement. A drag force anemometer has been specified as the most desirable instrument, although if the development of sonic anemometers proceeds at a fast enough pace, it is possible that a sonic

device might have some merits over the drag force instrument. The drag force unit measures the dynamic pressure of the wind on a spheric sensor. This device requires knowledge of atmospheric density for accurate interpretation, while the sonic instrument measures along three perpendicular axes the wind-induced change in the transit time of an acoustic pulse between two points. This unit can be made more nearly independent of temperature and compositional uncertainties. Two separate units are required to provide reliable vertical erection from the oblate sphere configuration.

The particle detector microphone will detect the number of particle impacts against an erected surface. The number of impacts above and below a given threshhold energy will be counted and the accrued counts played out five times during the landed mission. The counts will be logarithmically encoded so that a wide dynamic range will be obtained.

The water detector is included on the landed payload rather than the descent payload to provide longer equilibration times for the sensor. It is expected that at low Martian temperatures and water concentrations the response time of the sensor and the time required for the capsule to come to equilibrium with its environment will place serious doubts on descent data. The water detector will utilize the capacitance changes which absorbed water induces in thin alumina films as a measurement technique.

The cosmic and high energy radiation instruments will measure the radiation environment at the Martian surface. To reduce the possibility of confusing surface and cosmic radiation sources, the detectors will be appropriately shielded and directionally gated.

The alpha scattering instrument will provide data on the elemental composition of the surface of the planet. The energy distribution of alpha particles propelled from a radioisotopic source and then back-scattered from the surface will be measured with a 200-channel pulse height analyzer. The energy spectrum of the scattered particles is a function of the atomic number of the target material. The technique allows the resolution of individual elements in the range of atomic numbers from 5 to 17 (boron to chlorine in the periodic table) and the resolution of pairs of elements from 18 to 26 (argon to iron). At higher atomic numbers, the resolution degrades further. The sensitivity decreases with increasing atomic number in about the same way that resolution decreases. The sensitivity for carbon is about 0.1 percent.

4.2.4 Subsystem Mechanization

In spite of the very tight development timetable allowed for this mission, the actual mechanization of the scientific and engineering instrumentation

has not proceeded very far. All of the instruments can be assembled from state-of-the-art hardware, but only the mass spectrometer has actually been constructed to meet the various environmental conditions to be anticipated on this mission. Breadboard versions of several of the instruments have been built for most of the experiments to test the basic feasibility of the measurement concepts. Because of this situation, further discussion of the mechanization of the instrumentation is not meaningful.

4.3 TELECOMMUNICATION

4.3.1 Requirements and Constraints

The basic requirements (data preparation and communication), which must be satisfied by the telecommunication subsystem can be briefly stated as follows. The subsystem must be able to:

1. Prepare scientific and engineering data for efficient transmission to Earth by performing such functions as sampling, encoding, data compression and storage.

2. Establish communication contact with Earth and transmit cruise, entry, landing, and system status data. Of principal importance are the phases of entry, descent, and post landing. During the entry and descent phases sufficient data must be processed and transmitted to adequately define the vertical structure and composition of the atmosphere and wind velocity near the surface. During postlanding operation, climatic information must be collected over one diurnal cycle and the character of the Martian surface must be determined. The data transmission objectives for each mission phase are summarized in Table XII.

4.3.1.1 Environmental Constraints

The considerable difficulty associated with attitude control of the suspended capsule during the parachute descent phase of the mission makes the use of directional antennas impractical, thus imposing severe limitations on the effective radiated power which can be obtained. Similarly, the lack of orientation capability in the landed capsule makes wide antenna beamwidths necessary during postlanding operations. Reflection from the planet's surface further limits communication capabilities because of fading due to multi-path effects. Plasma attenuation created by aerodynamic heating during entry limits the communication time and makes storage of critical entry data necessary. Finally, the high deceleration during entry makes the normally efficient coherent modulation systems less attractive due to the high carrier power required to avoid loss of phase lock.

TABLE XII

DATA TRANSMISSION OBJECTIVES

Mission Phase	Data Objective
Preseparation	Determine integrity of FC subsystem
Separation	Verify execution of ACS, separation and ΔV events
Cruise	Periodic status checks
Preentry	Calibrate instruments in zero g
*Entry	Define upper atmosphere
Terminal descent	Define lower atmosphere
Impact	Determine survival status and local environmental conditions
Landed	Collect surface environmental and science data

*Approximate 7 to 14 seconds of communications blackout occurs during this phase.

All of these factors contribute to create a situation which severely constrains the data transmission capability of the flight capsule.

4.3.1.2 Total Energy Constraint

Thermal control difficulties created by the requirement for omnidirectional impact attenuation, coupled with the severe impact shock, make the use of high energy, high density power sources such as RTG's and solar cells infeasible. The degrading effects of the sterilization environment on lightweight batteries disqualifies them from consideration. The only power source remaining which presents acceptable low design risk is the highweight Nickel Cadmium battery. The mission thus becomes severely energy limited, making transmission of large quantities of data over the Mars-Earth distance impossible.

4.3.1.3 Flight Spacecraft Constraints

Since the execution of the flight spacecraft orbit injection maneuver requires a number of time consuming preparatory operations it is required to maintain a lead time of about 3 hours between flight capsule entry and flight spacecraft injection. Conformance to this requirement results in large ranges (approximately 35,000 km) between the two vehicles during flight capsule entry. It is desirable to minimize flight capsule requirements which introduce complexity into the flight spacecraft design. Therefore, large steerable antennas on flight spacecraft should be avoided. These factors tend to reduce the potential capability of the relay mode of data transmission.

4.3.1.4 Design Guidelines

The preceeding paragraphs clearly indicate a bit-limited mission. If, however, modest objectives are accepted, the probability of achieving these objectives can be made sufficiently high if a conservative approach is followed in the telecommunication system design. The following guidelines reflect such an approach.

a. Utilize a low-gain body-fixed antenna on the flight spacecraft and the flight capsule.

b. Employ functional redundancy in the telecommunication system design where possible.

c. Utilize low-power solid-state equipment where current day technology permits.

d. Avoid reliance on the flight spacecraft after orbit injection.

e. Select a data rate which allows adequate margin for fading conditions.

f. Minimize mode changes during critical mission phases.

The following paragraphs describe a design concept which satisfies the minimum requirements within the imposed constraints.

4.3.2 Subsystem Design

4.3.2.1 General Approach

Figure 19 presents the overall telecommunications system block diagram. Two independent data handling and data transmission systems are employed. The external system, located outside the landed capsule, is used during the period between flight capsule separation and landed capsule impact on the planet's surface and utilizes a VHF relay (to the flight spacecraft) mode of data transmission. The internal system is used exclusively during the landed phase of the mission and employs a direct (to Earth) mode of data transmission. The required interfaces between the two systems are minimized, being limited by the following constraints:

1. Accelerometers internal to the landed capsule (near the center of gravity) must provide information during the entry phase and therefore must be connected to the external system.

2. The use of the direct link to provide redundant transmission of entry data after landed capsule impact dictates connection between the internal and external telemetry systems.

   The handling function for each system is split into two data handling sections and one storage section. The data handling sections are the data-automation subsystem and the telemetry subsystem. The data automation equipment handles all scientific instrumentation, while the telemetry subsystem handles all engineering data. The reasons for doing this are:

   a. A failure in either subsystem will not compromise operation of the other.

   b. The data acquisition requirements of the two subsystems are dissimilar.

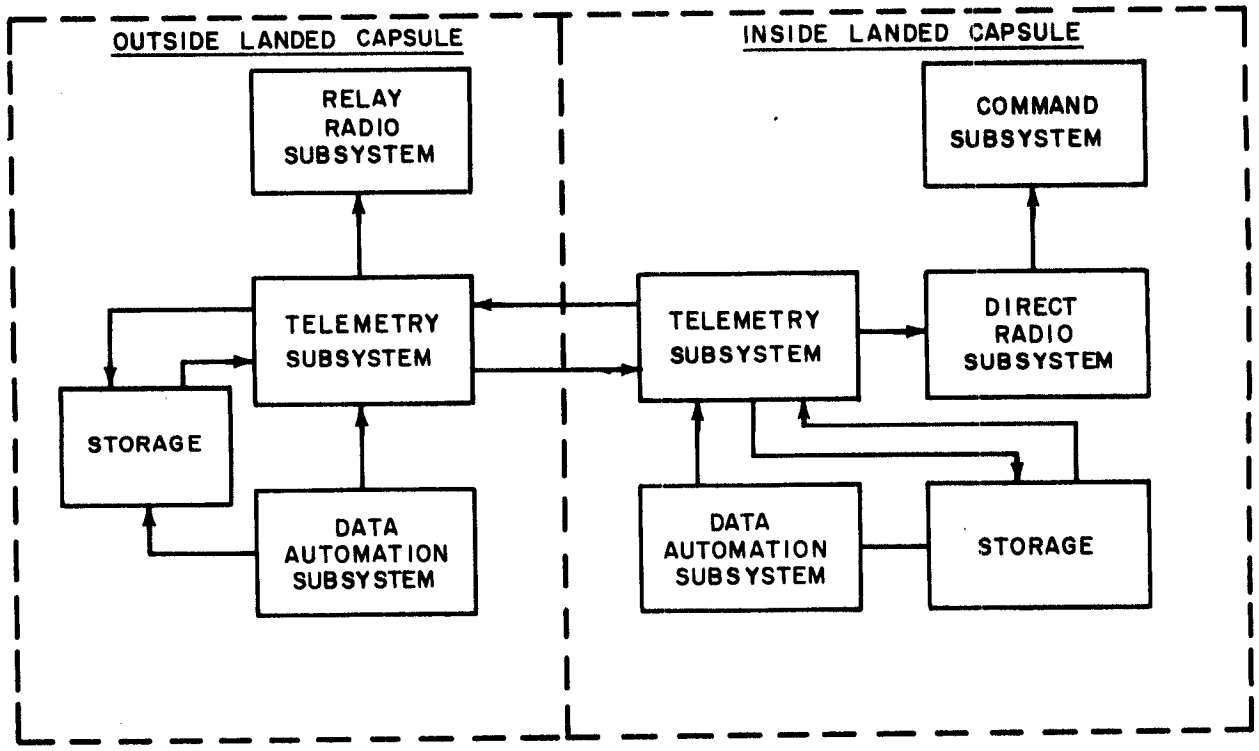


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Figure 19 TELECOMMUNICATION SUBSYSTEM

c. The science instrumentation package is more susceptible to change than engineering requirements, therefore, a less complex interface results from use of two subsystems. The radio subsystem for the external and internal telemeters are quite dissimilar. The external (relay system) operates at approximately 270 MHz, employs a 30-watt solid-state transmitter and utilizes the Frequency Shift Keying (FSK) modulation technique. The internal direct link system transmits at 2295 MHz, utilizes a 20-watt traveling wave tube (TWT) and employs a multiple frequency shift (MFS) or linear chirp modulation (LCM). A command receiver is incorporated in the direct link radio subsystem to allow landed capsule control flexibility in the postlanding operation. Table XIII presents a brief resume of the telecommunications system salient design and performance characteristics.

4.3.2.2 Relay Versus Direct Link

The proposed approach utilizing relay transmission before impact and direct link transmission after impact was selected after the following considerations:

1. The difficulty associated with pointing a directional antenna toward Earth or toward the flight spacecraft during the entry and parachute descent phase of the mission made a wide-beam antenna necessary.

2. The performance (data rate capability) of a relay link far exceeds that of a direct link when each is constrained to use a wide-beam antenna.

3. The probability of multiple line of sight contacts between the flight spacecraft and the landed capsule within the mission duration is not sufficiently high to make postlanding relay transmission alone acceptable.

4. The optimum relay transmission frequency for prelanding operation is in the VHF band and results in antennas of large volume. The added impact attenuator weight required to protect such antennas for the post-impact mission proved prohibitive for this design. The increase in operating frequency required to alleviate the weight problem would degrade the performance of the link to an unacceptable level.

TABLE XIII

TELECOMMUNICATIONS CHARACTERISTICS

External System	
Mode of transmission	Relay via FS
Transmitter power	30 watts
Frequency	270 MHz (approximately)
Modulation	FSK
FC antenna type	Spiral
FS antenna type	Helix (body fixed)
FS receiver noise figure	5 db
Data rate (at 35,000 kilometers)	64 bits per second
Internal System	
Mode of transmission/reception	Direct with Earth
Transmitter power	20 watts
Frequency	
Transmit	2295 MHz
Receive	2115 MHz
Modulation	MFS or LCM
Antenna type	V Horn
Data Rate	2 bits per second

4.3.2.3 Transmitter Power and Modulation Selection

Since the technology cutoff date for the study was established at September 1966, selection of a direct link transmitter power at S-band in excess of 20 watts would have been highly speculative. Powers much less than 20 watts would result in data rates so low that the value of the mission would be questionable. As it stands, the moderately high risk associated with even a 20-watt unit when one considers the impact-shock requirement may well put the landed mission below the threshold of feasibility. Thirty watts at 270 MHz represents the maximum power achievable using a solid-state design. The potential problems associated with gaseous breakdown, and possibly multipacting makes avoidance of high voltages (as required by vacuum tubes) desirable. In the absence of hard constraints regarding the flight spacecraft performance as relay receiver, the 30-watt point appears to provide a reasonable compromise between system performance and equipment complexity.

Both the relay and direct link systems proposed the use of noncoherent modulation techniques. These selections came about because of the large fraction of total power which would be required to provide a coherent reference. In the relay case, large-loop bandwidths would be necessary to ensure rapid acquisition and to allow tracking during the high acceleration periods of entry. In the direct link case, the very low effective radiated power resulting from the poor antenna performance on the planet's surface would hardly be adequate to maintain lock with realizable-loop bandwidths in the DSIF receiver.

For these reasons, the normally less efficient, but more easily mechanized noncoherent FSK system, proved better suited to the unusual environments during entry and descent and the 32-level MFS system or LCM system proved to be the best choices for the post-landing mission.

4.3.3 Subsystem Operation

The telecommunications subsystem operates in a variety of data acquisition and transmission modes as described below:

4.3.3.1 External Mode I -- Preseparation/Separation Checkout

The flight capsule functional checkout mode is used 1) during automatic checkout prior to launch, 2) periodically during the in-transit phase prior to separation, and 3) during the separation phase. Data is available prior to separation either via hard line from both internal and external telemetry subsystems, or via radio link from any transmitter by means of parasitic antennas on the sterilization canister.

4.3.3.2 External Mode 2 -- Postseparation Checkout

The status of the flight capsule is checked periodically via relay link during the cruise phase between separation and entry. Five minutes before entry (t_e-5) a final status check starts. During this check of the flight capsule, all entry and descent science instruments, with the exception of the radiometer, are also monitored. The external system remains in this mode and continues to transmit data via the relay link until impact unless external Mode 4 operation is initiated.

4.3.3.3 External Mode 3 -- Entry Data Storage

During this mode, initiated at entry when 0.1 g is sensed, critical entry data (e.g., acceleration) is stored. This is done because communications blackout occurs shortly after sensing 0.1g.

4.3.3.4 External Mode 4 -- Entry/Descent Data Transmission

During this mode, initiated at 11 seconds after sensing 10 g ascending (10 g + 11), the format of external Mode 2 is changed. The data stored in external Mode 3 is substituted for much of the engineering and other science data no longer required. The system is required to complete transmission of all data stored during communication blackout (external Mode 3) before impact, even if the parachute should fail to open. Also, in this mode, the first 15 frames of data played out via the relay link are stored in internal storage for subsequent transmission directly to Earth after landing. This is done to provide a functionally redundant path for transmission of critical entry/descent data in the event of relay link failure.

4.3.3.5 External Mode 5 -- Standby

During this mode, all subsystems requiring battery power are turned off, except for the CC&S master-timer sequencer which initiates external Mode 2.

4.3.3.6 Internal Mode 1 -- Impact Data Storage

During this mode, initiated when the payload is tethered, impact science, post-impact engineering, and post-impact science data are stored.

4.3.3.7 Internal Mode 2 -- Direct Link Transmission

During this mode, first initiated one hour after sensing 0.1g, the 15 frames of data stored during external mode 4 are transmitted to

Earth at a rate of 2 bps. This mode is initiated again 24 hours later, and the data collected since the first direct-link transmission are transmitted to Earth.

4.3.4 Subsystem Mechanization

Figure 20 shows the overall telecommunications subsystems in detailed block diagram. As previously noted, two functionally independent systems are used. As the names imply, the external system is physically outside of the landed capsule and the internal system is located within it.

The external and relay systems feature a radio subsystem containing redundant 30-watt solid-state transmitters which are capable of being switched by on-board failure sensing equipment. The transmitter is protected against the effects of high VSWR during entry by a three-port circulator in the antenna feed-line. Assuming load is utilized to dissipate the reflected power under this condition, the relay antenna, mounted on the exterior of the suspended capsule, is a planar spiral providing the desired circular polarization. Near 0-db gain is obtained at a look angle of 90 degrees. The data handling equipment includes an 8,000-bit core storage, a 105-channel PCM telemetry for diagnostic measurements and a 32-channel telemetry for landing experiments. The internal or direct link system contains redundant S-band 20-watt traveling wave tubes feeding redundant antenna systems. Linearly polarized V-horn antennas mounted on opposite sides of the landed capsule allow communications regardless of capsule landing attitude. Sensing of reflected power allows the system pointing away from the planet's surface to be selected for use following a brief turn on of both systems after impact. An S-band command receiver is diplexed to the antenna system using redundant three-port circulators. The internal data handling, like the external system, utilizes separate equipments for diagnostic and experimental measurements and contains a 12,000-bit core storage system.

4.4 POWER SUBSYSTEM

4.4.1 Design Requirements

The electrical power subsystem is required to supply all flight capsule power from preseparation checkout to the end of the mission. The power subsystem must be capable of complete recharge from the flight spacecraft power supply whenever it is partially discharged during inflight and preseparation flight capsule checkout. The equipment must be capable of sustaining the dry heat sterilization cycle with complete electrical checkout both before and after sterilization.

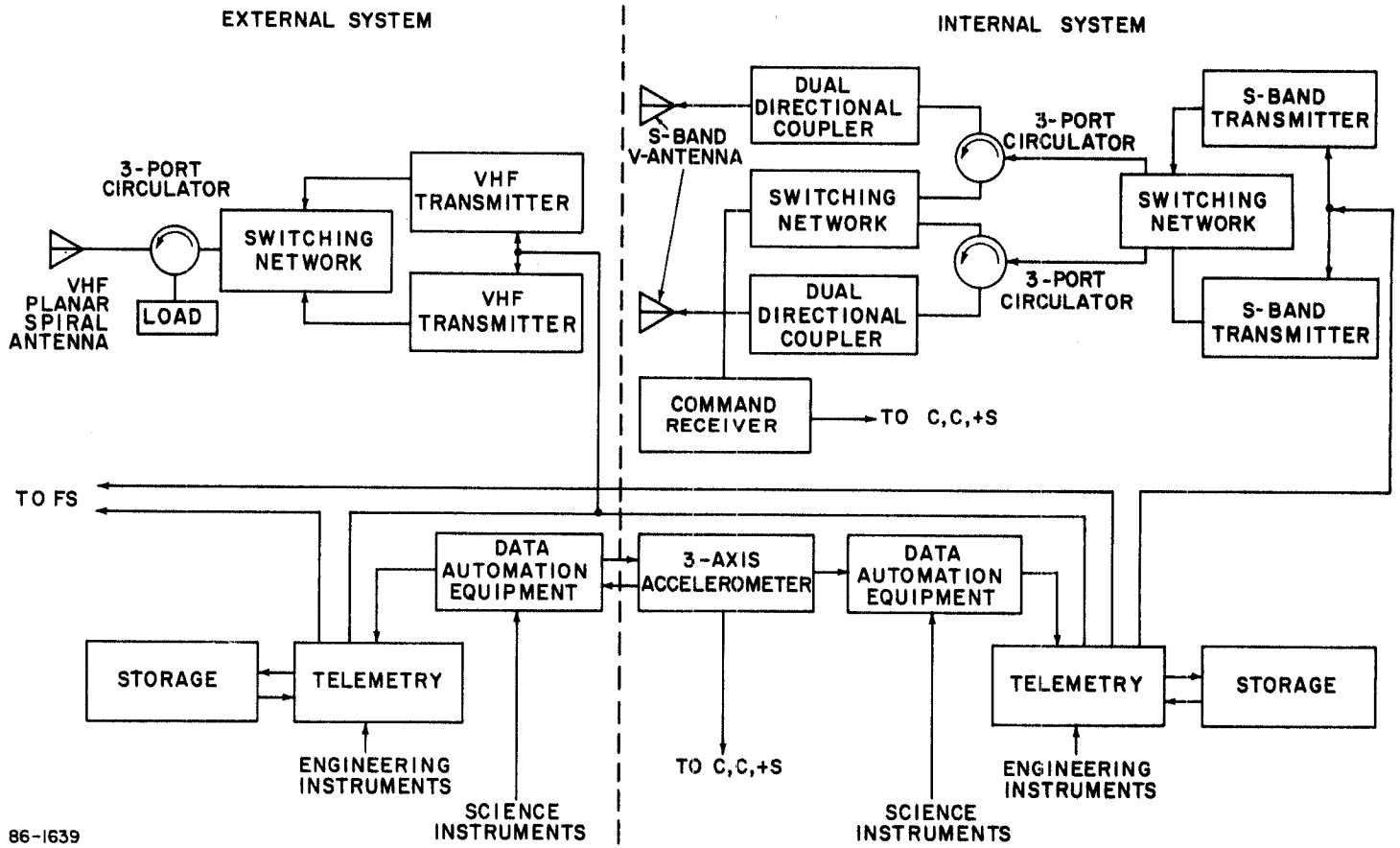

Figure 20 TELECOMMUNICATIONS SYSTEM BLOCK DIAGRAM

The primary environments which influence the power subsystem design are sterilization qualification, three 36-hour cycles of dry heat at 145°C, cruise temperatures from -15° to +60°C, and impact shock levels as high as 1000G during planetary impact. The system must be capable of retaining its charge during the 2 to 20 (nominally 12) days from separation to impact.

The subsystem power profile is shown in Figure 21. In general only total power levels are shown rather than the requirements for separate components. For example, the "checkout" block represents the total power and energy used by transmitters and instruments intermittently over the 12-day cruise period; the only continuous power user is the Central Computer and Sequencer which requires 0.1 watt. The transmitters use raw dc power. All other users, however, require regulated power and condition it to their particular needs. Table XIV shows a complete list of the power requirements by mission phase of each component, and total energy requirement. The transmitters represent the major power user. The power levels shown include the power dissipated in the regulator.

An additional 25 percent of battery capacity is required to provide energy to keep the battery at operating temperature.

4.4.2 Subsystem Design

The power requirements and the environmental condition which must be accommodated lead to only one power source, the nickel cadmium battery. Practical alternate power sources are silver-zinc batteries and the Lithium-chlorine fuel cell. The silver-zinc batteries cannot be considered at this time since they have not been shown to survive sterilization. The fuel cell is not yet adequately developed.

A schematic of the power system configuration before separation is shown in Figure 22. Weight restrictions do not allow for a fully redundant system. However, some measure of redundancy can be achieved by using two separate battery-regulator circuits and allowing battery No. 2 in the landed capsule to act as backup to battery No. 1 located in the external payload. The maximum energy drain on battery No. 2, as a backup during entry and descent, would still allow it to perform its function fully under any but the most extreme temperature environment. Under extremely adverse conditions it would provide 75 percent of the post-impact requirement.

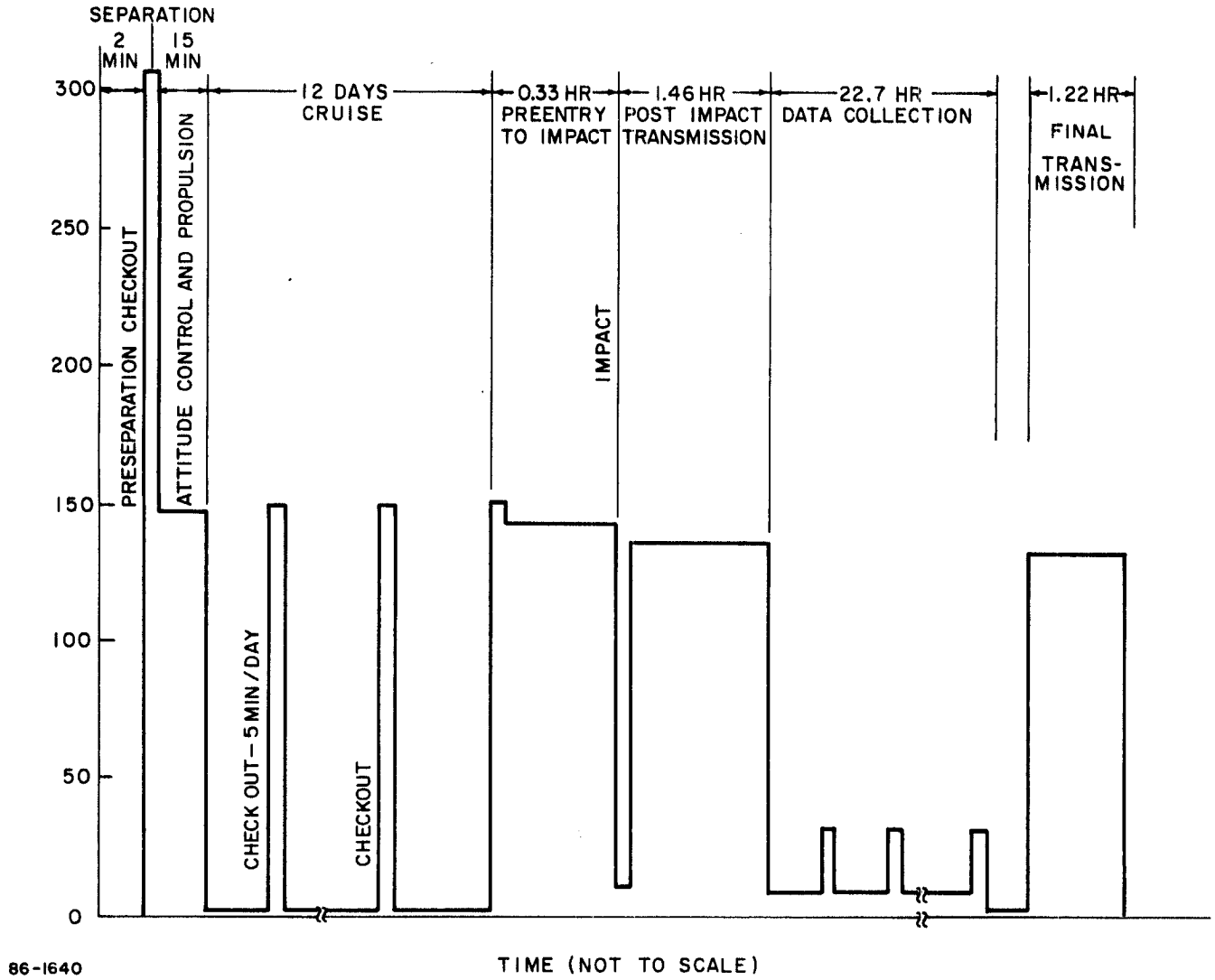

Figure 21 FLIGHT CAPSULE POWER PROFILE

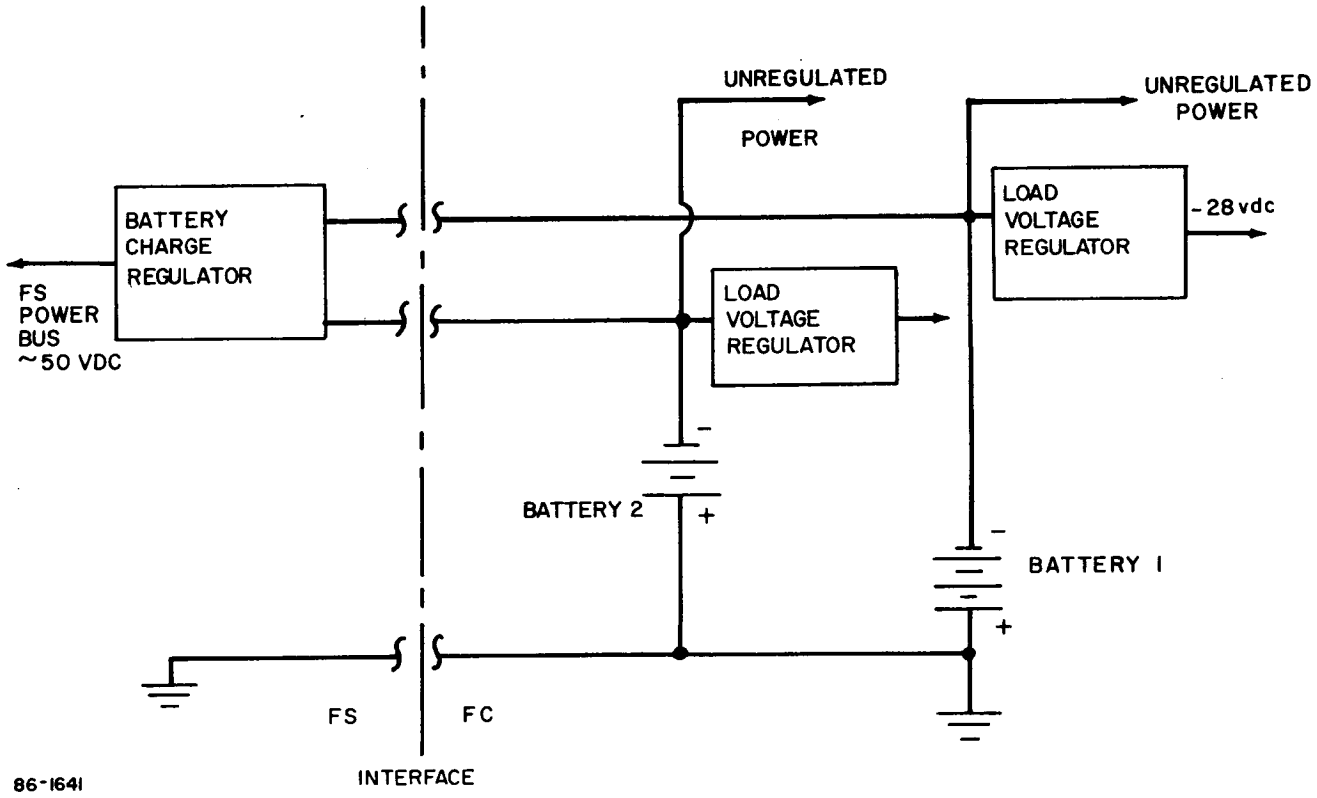

Figure 22 POWER SYSTEM CONFIGURATION - BEFORE SEPARATION

TABLE XIV

FLIGHT CAPSULE INSTRUMENTATION POWER REQUIREMENTS

	Command Receiver	Central Control and Sequencer	Data Automation Equipment	Telemetry System	Relay Transmitter	Direct Transmitter and Amplifier	Attitude Control	Radiometer	Accelerometer	Pressure Transducer	Temp. and Mass Spectrometer	Gas Chromatograph	Radar Altimeter	Beta Scatter	Atmospheric Density	Anemometer	Dust Particle Detector	Temp. and Pressure	Cosmic and Surface Radiation	Alpha Scatter and Water Detector		Total Power (watts)	Time of Phase (hours)	Energy (watt hours)
Preseparation	3	3	10	5	75.5	111	50	2.1	2.9	0.4	14.6	5.7	14.3	0.4	5.7	1.4	0.6	1	1.1	3.4		311	0.03	10.4
Attitude control and propulsion	3	3	10	5	75.5	-	50	-	-	-	-	-	-	-	-	-	-	-	-	-		146.5	0.25	36.6
Cruise	-	0.1	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-	-		0.1	288	28.8
Status check	3	3	10	5	75.5	-	-	2.1	2.9	0.4	14.6	5.7	14.3	0.4	5.7	1.4	0.6	1	1.1	3.4		150	0.4	60
Preentry to impact	3	3	10	5	75.5	-	-	2.1	2.9	0.4	14.6	5.7	14.3	0.4	5.7	-	-	-	-	-		142.6	0.33	48.2
Postimpact transmission	3	3	10	5	-	111	-	-	-	-	-	-	-	-	-	1.4	0.6	1	-	-		135	1.46	197
Landed data collection-continuity	3	0.1	1	-	-	-	-	-	-	-	-	-	-	-	-	1.4	0.6	1	-	-		7.1	20.8	147
Landed data collection 5 min/hr	3	3	10	5	-	-	-	-	-	-	-	-	-	-	5.7	-	-	-	1.1	3.4		31.2	1.9	60
Post-mission transmission	3	3	10	5	-	111	-	-	-	-	-	-	-	-	-	-	-	-	-	-		132	1.22	161

79-1

The total power subsystem weight is approximately 103 pounds and the volume is 0.93 cubic feet.

4.4.3 Subsystem Mechanization

4.4.3.1 Power Source

The recommended power source consists of hermetically sealed nickel cadmium cells connected in series to form a battery. The energy provided by battery No. 1 for descent and entry is 184 watt hours. The energy provided by battery No. 2 after impact is 565 watt hours. The cells are assembled together in an unsealed case for the purpose of mounting and heat dissipation.

4.4.3.2 Power Conditioning

Power to all components requiring voltage regulated dc is supplied through two "buck-boost" regulators. These regulators are able to accept input voltages above or below the required output level. As a result, most of the battery capacity can be used and overall system weight is conserved. Each load voltage regulator weighs 3.5 pounds and occupies an estimated volume of 86 cubic inches.

The battery charge regulator is located on the flight spacecraft and provides for a continuous trickle charge to each battery at a rate of 1 percent of the nominal ampere-hours/per hour (c/100) or more.

Allowance is made for fast charging at up to 3 amperes. The charge regulator weight is 1.3 pounds and volume estimated at 40 cubic inches. A protection charge cutoff is provided should the battery temperature rise too high during charging.

4.4.3.3 Subsystem Characteristics

Table XV shows the salient characteristics of the power subsystem. Projected weights for alternate silver-zinc and lithium-chlorine power sources are also shown for comparison. The attractiveness of the alternates, particularly the lithium-chlorine fuel cell for future missions, is clearly indicated.

TABLE XV

SALIENT CHARACTERISTICS OF THE ELECTRICAL POWER SUBSYSTEM -- 1971

Source	Prime Nickel-Cadmium Battery	Back-Up Silver-Zinc Battery	Back-Up Lithium Chlorine Fuel Cell System
Delivered energy-separation to impact	184 watt hours	184 watt hours	184 watt hours
Weight-separation to impact	33 pounds	14 pounds	see post-impact weight
Delivered energy post-impact	565 watt hours	565 watt hours	565 watt hours
Weight-postimpact	71 pounds	28 pounds	30 pounds (supplied energy from separation to end of mission)
Output voltage limits before regulation	25 to 33 vdc	25 to 37 vdc	3.0 to 3.5 vdc*
Charge regulator	Resistor and diode in series for unlimited trickle charge	Current controller with full capacity cut-off and temperature compensation	None required, extra fuel for extra discharge

*dc-to-dc converter required to raise voltage

4.5 PROPULSION

4.5.1 Objectives and Requirements

The primary objective of the propulsion system is to provide a small incremental velocity to place the separated vehicle on an impact trajectory and to accelerate the separated vehicle to provide three hours lead time for relay link communications during entry and parachute descent before the flight spacecraft begins its orbit injection attitude maneuver. The velocity increment is applied shortly after flight capsule - flight spacecraft separation 12 days before planet encounter. A maximum velocity increment of 100 ft/sec is required adjustable downward to accommodate dispersion in the planetary vehicle approach trajectory.

The requirements which have the largest impact upon the selection of the propulsion unit are the necessity for only one firing cycle (no restart capability) and the dry heat sterilization technique. Only equipment which represents the expected state of the art in September 1966 was considered. A thrust cutoff capability is required. Both solid and liquid engines were considered with respect to reliability, sterilizability, space storability, total impulse accuracy, and the total impulse needed (4,850 pounds seconds). A solid propulsion rocket motor with thrust cutoff provided by nozzle deployment was selected.

4.5.2 Subsystem Design

The propulsion subsystem consists of a solid propellant rocket motor. The rocket firing is controlled by the flight capsule CC&S which stores the start time and duration of fire, updated as needed through the DSIF-to-planetary vehicle-to-flight capsule communication link. After the attitude control system has positioned the separated vehicle in the correct firing attitude, at the prescribed time, the rocket is ignited by an electrical signal originated in the flight capsule CC&S. Thrust termination is controlled by the flight capsule accelerometers which measure the ΔV, a backup termination signal is based on a burn time stored in the flight capsule CC&S. The thrust termination is achieved by nozzle deployment which is followed by the jettisoning of the expended motor.

The rocket motor is a modified Titan vernier motor (TE-M-345). The primary modification consists of replacing the present propellant with a sterilizable propellant (TP-H-3105). The motor is spherical in shape, 13.5 inches in diameter and 18.6 inches long, having a TH-1050 stainless steel case. The exhaust nozzle is partially submerged, has an area ratio of 18.7, and is made of vitreous silica phenolic. The nozzle is retained in the motor case by a split flange which is held together by two explosive bolts so that on receipt of an electrical signal the bolts are released, the flange separates, and the nozzle is blown free of the case, resulting in a sudden drop in chamber pressure which terminates thrust.

The motor is mounted in the flight capsule using mounting flanges presently existing on the Titan vernier motor. The total loaded weight of the propulsion subsystem is 81.0 pounds yielding a propellant mass fraction of 0.788.

To contribute to the reliability design goal of 0.990 the dual ignitors of the Titan motor are retained; each has a minimum firing current of 5 amperes.

4.5.3 Subsystem Performance

The Titan vernier motor using sterilizable propellant has a total impulse capability between 255 lb-sec. minimum and 16,320 lb-sec. maximum due to its thrust termination feature. The required total impulse of 4850 lb-sec. nominal results in a ΔV of 100 ft/sec, while the total impulse available results in a ΔV capability of 290 ft/sec for the separated vehicle. The total impulse capability is reduced by off-loading propellant which would lower the rocket motor total weight by approximately 30 pounds. The resulting ΔV capability would be 145 ft/sec.

To accurately control the total impulse it is necessary to have a highly repeatable shutdown impulse. The Titan motor has a total impulse repeatability for any one burn time of less than 1 percent (3-sigma). When thrust termination is used, the repeatability is improved because of the faster shutdown. A large pressure spike for a fraction of a second (40 msec) accompanies the shutdown, but it will dissipate through the mounting joint sufficiently, due to joint design such that there is no detrimental effect on the vehicle or subsystems.

The propellant specific impulse is 255 seconds which does not impose any important weight problem because of the small propellant quantities. The requirement that only gaseous exhaust products shall result from the combustion process and the fact that the propellant is to be sterilizable prevents, at this time, improving the specific impulse of the propellant by using metal additives.

The average thrust level is 768 pounds and offers no problems for the separated vehicle due to acceleration loads since the thrust to weight ratio is less than one-half.

4.6 ATTITUDE CONTROL AND THRUST VECTOR CONTROL

4.6.1 Objectives and Requirements

The attitude control system (ACS) is required to orient the separated vehicle to the proper attitude for application of the velocity increment following separation from the flight spacecraft, to maintain it in this attitude while

thrusting, and then to place the separated vehicle in the proper attitude for entry and maintain this attitude. More specifically, the systems must perform the following functions:

a. Nullify the tipoff rates due to separation from the flight spacecraft and realign the separated vehicle to the flight spacecraft reference attitude.

b. Maneuver the separated vehicle thrust axis into a preselected attitude, with respect to the flight spacecraft reference attitude, for ΔV thrust application.

c. Provide thrust vector control (TVC) during the operation of the ΔV propulsion system.

d. Reorient the separated vehicle from the thrust application attitude to the attitude required to achieve a zero-angle of attack at entry.

e. By means of spin rockets, provide spin stabilization for the balance of the trajectory.

In the event of failure of the ACS to perform in its normal operating mode, it shall be capable of providing spin stabilization immediately after separation to maintain the attitude attained by flight spacecraft maneuvering and to provide TVC during the ΔV propulsion operation.

4.6.2 Subsystem Design

The design selected makes use of a combination of active attitude control using a gyro controlled cold-gas reaction system together with spin stabilization which is accomplished by solid propellant spin rockets. The angular rates of the separated vehicle are measured by three body mounted rate gyros; the outputs of these gyros are electronically integrated so that angular position as well as angular rate is available. This information is used by the control logic to operate the valves of the cold-gas reaction control system. The system provides three-axis control in couples by means of twelve nozzles. Spin stabilization is provided by two redundant groups of solid propellant rockets. Normally only one group is required for spin stabilization, but if the primary operational mode of the ACS fails, both sets of spin rockets will be used in the backup mode. (In this case it is necessary to despin prior to entry, and a third set of rockets is provided for that purpose).

The gyros and electronics will be turned on and allowed to warm up while the separated vehicle is attached to the flight spacecraft. During this time the ACS will be checked out; this also provides an opportunity to trim the drift of the gyros and integrators. Next the separated vehicle is separated from the flight spacecraft, the reaction control system is activated and realigns the separated vehicle to the flight spacecraft reference attitude, correcting for any disturbances which occured during separation. The separated vehicle is then oriented so that its thrust axis is in the desired ΔV direction and maintained in that direction during thrusting. The separated vehicle is then reoriented to have zero-angle of attack at entry and is spun up by one set of spin rockets to maintain this attitude for the remainder of its trajectory until entry. Finally the propulsion and ACS electronics assembly are jettisoned to reduce the entry weight of the entry vehicle.

Several alternatives were available in the design of the ACS and TVC. These are discussed in greater detail in later sections, but deserve brief mention here. Particular attention was given to a spin-only system. This approach is particularly attractive because of its simplicity, but has two serious drawbacks. One is the requirement that the flight spacecraft maneuver to place the separated vehicle in the proper attitude for thrusting; the other is the poorer accuracy of the spin-only system, resulting in unacceptable entry angle dispersions. An active closed loop system was therefore selected for initial stabilization, orientation, TVC, and reorientation. The active system could have been designed to continue to operate until entry, but spin stabilization was the preferred choice because of the small additional weight required and the high reliability of a spin system compared to an active system which must operate for several days from the time of separation until entry. Because the entry vehicle is aerodynamically stable, no attitude control is required during entry. However, it is not desirable for the entry vehicle to be tumbling at entry due to the longer time required for stabilization and the attendant effect on heating, loads, dispersion, and parachute deployment. Consequently spin stabilization is used to ensure a stable orientation at entry with no angle of attack.

4.6.3 Subsystem Performance

The most stringent performance requirement on the ACS is the control of the direction of the imparted velocity vector. It will be seen in Section 7.1 that the requirement for communications lead time (which is obtained by accelerating the separated vehicle) together with rather narrow limits on allowable entry angle dispersion -- these two factors dictate the accuracy requirements of the ACS. The system overall accuracy is 0.23 degree (one sigma) compared to 0.25 degree required for an entry angle of -40 degrees. It operates for a total of 15 minutes and weighs 90 pounds. The total stored impulse is 764-lb-sec, compared to 232 lb-sec

required. It has a limit cycle amplitude of less than 0.1 degree, and a maximum rate of 1.0 deg/sec. The limit cycle and rate limit were selected to minimize impulse requirements within the accuracy constraints on the system. In addition to the limit cycle itself, other contributors to the pointing error are drift and scale factor in the gyros and integrators, errors introduced by the electronics, and alignment errors.

More than three times the required total impulse is stored in the system to provide for leakage and possible failure. There are two completely redundant systems with sufficient extra impulse in each system to compensate for a valve stuck open in either system.

Under normal operation the spin rockets are required to achieve a spin rate high enough to provide stabilization in the zero-angle of attack attitude until entry, and this is accomplished with two rockets each having a total impulse of 45 lb-sec, giving a spin rate of 10 rpm. In the backup mode a higher spin rate of 50 rpm is required to achieve stabilization during thrusting; and this is accomplished by using the redundant set of spin rockets together with the primary set. An additional set of eight rockets are provided to despin 10 rpm at entry in the backup mode.

4.6.4 Subsystem Mechanization

A subsystem schematic is shown in Figure 23. Three-axis control is provided by the cold-gas reaction system until the final orientation after ΔV thrusting. The separated vehicle is then spin stabilized with solid propellant rockets. A mathematical model showing the logic for stabilization is presented in Figure 24.

4.6.4.1 Gyros

The gyros are the Kearfott Alpha Series. Modifications will be made to features such as the gyro gain, characteristic time, and gyro heater voltage to be more compatible with operational requirements.

4.6.4.2 Integrator

An electronic analog integrator integrates the gyro output to provide an indication of angular position. It is instrumented by using a dc-operational amplifier with a capacitor in the feedback loop.

4.6.4.3 Control Logic

A Schmitt trigger circuit appropriately biased for the correct firing level receives the error signal (proportional to attitude and attitude rate). The hysteresis of the Schmitt trigger is set to provide stability.

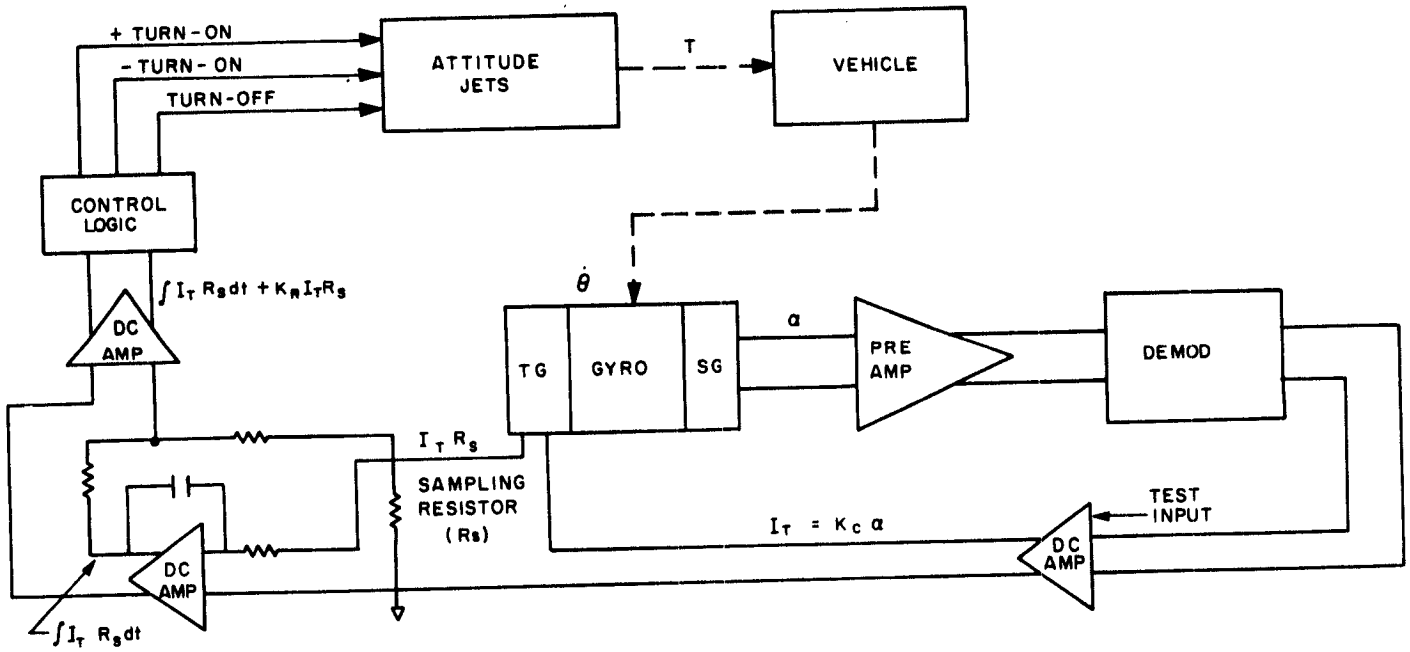

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Figure 23 ATTITUDE CONTROL SUBSYSTEM SCHEMATIC

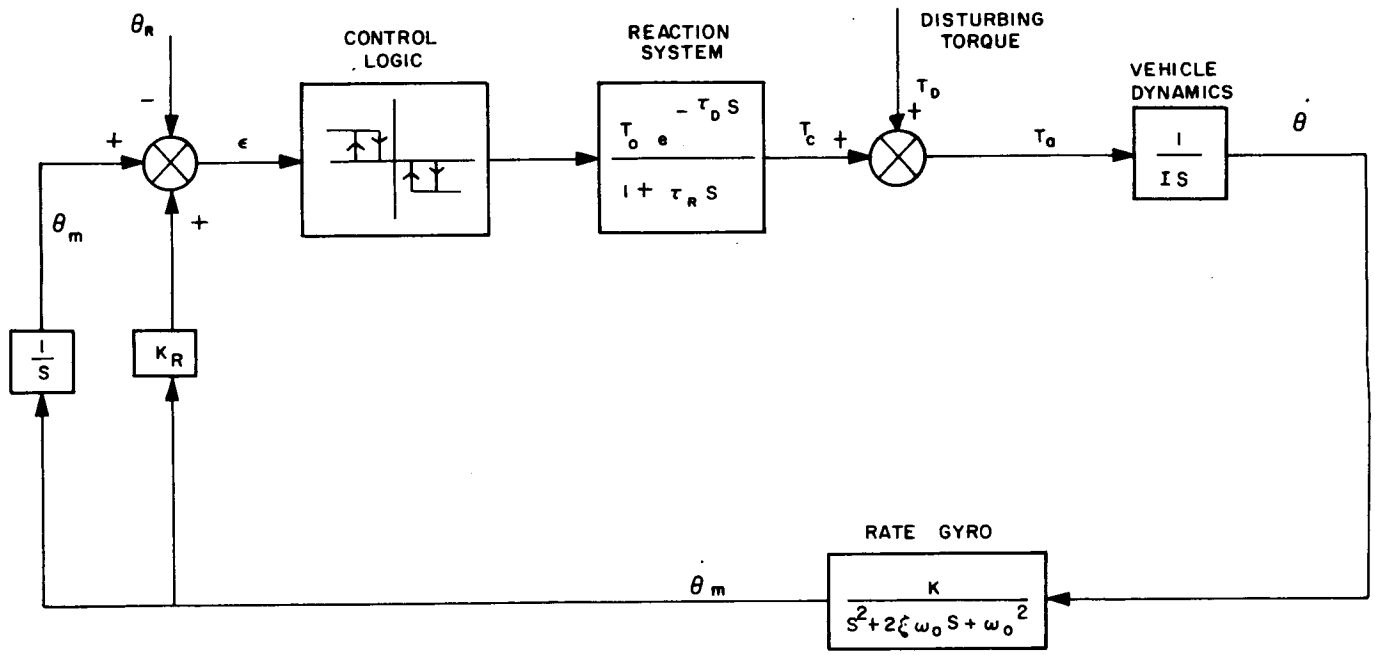

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Figure 24 MATHEMATICAL MODEL OF ATTITUDE CONTROL SUBSYSTEM

4.6.4.4 Reaction Subsystem

A schematic of the cold-gas nozzle and plumbing portion of the reaction subsystem is shown in Figure 25 spin rockets are not included. These twelve nozzles provide three-axis control in couples. Spin stabilization is provided by solid propellant rockets. The spin rockets will be arranged in two groups. In the case of failure of one group, the other will be used for spinup. Both sets of spin rockets will be used if the primary operational mode of the ACS fails and the backup mode is selected.

4.6.4.5 Spin Rockets

The spin rocket is a solid propellant scout spin motor MARC -4B2 (Atlantic Research), modified. The primary modification is the replacement of the presently used propellant with a sterilizable propellant.

4.7 PARACHUTE

4.7.1 Objective and Requirements

The primary objective of the parachute descent system is to decelerate the landed capsule to a reasonably low verticle impact velocity while rendering adequate communications time during the terminal phase of the flight. The parachute system and allowable attendant deployment conditions, dictate to a large degree the maximum M/C_DA that can be allowed for a vehicle entering a given atmosphere with particular entry conditions. The final selection of such a retardation system was based first on reliability, second on weight, and finally on performance and developmental risk.

The selected descent system must contain a terminal descent parachute which is fully inflated and subsonic at 15,000-feet altitude. The system must also ensure a vertical impact velocity of no greater than 80 ft/sec in the worst case (terminal descent atmosphere). A reasonably low impact velocity was chosen to allow for the design of an impact attenuation system which would minimize the impact g on the landed scientific instrumentation.

4.7.2 Subsystem Design

The model atmospheres which must be accommodated were the Models 1, 2, and 3 combined with the terminal descent atmosphere. The entry vehicle must be designed to sustain the environments of Models 1, 2, and 3 and at commencement of descent retardation with the model switched to the terminal descent atmosphere. Switching to the terminal descent atmosphere at

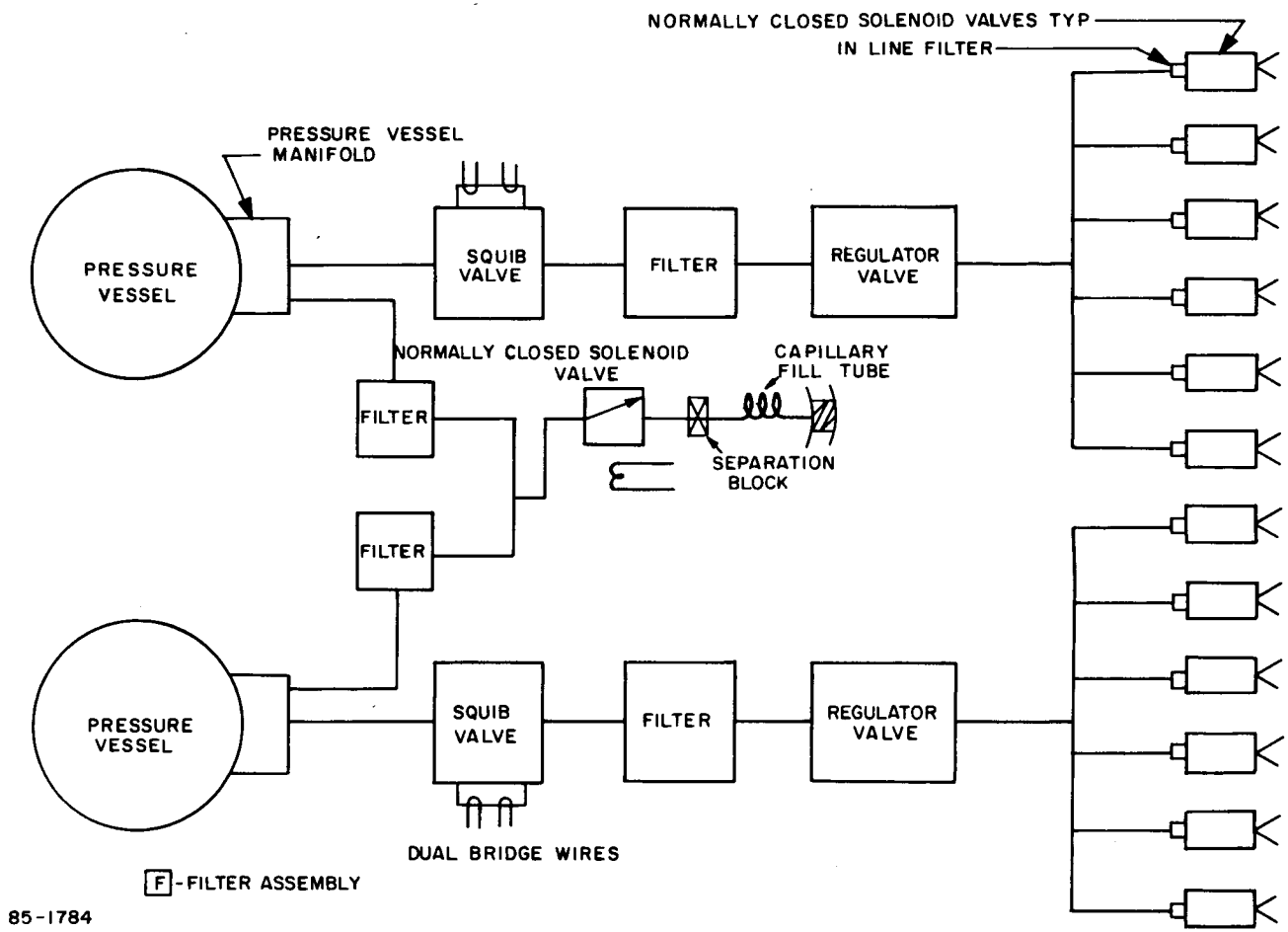


Figure 25 COLD GAS REACTION SUBSYSTEM

parachute deployment has the effect of changing the Mach number while maintaining the same flight velocity. It was concluded that Model 3 atmosphere dictated the allowable M/C_DA based on the design deployment conditions of the descent system and that the terminal descent atmosphere, sized the parachute required to satisfy time and impact velocity criterion.

The selected retardation system utilizes a single main parachute which is pulled out of its canister by a pilot parachute. The main parachute is deployed in a reefed condition at Mach 1.3 and is disreefed at 16,000 feet after having received a signal from the radar altimeter. Selection of the canopy configuration for this mission was based on four factors, performance characteristics, reliability, weight (volume) penalty, and development risk. An optimum parachute configuration requires simultaneous evaluation of the above mentioned factors, coupled with an appraisal of the effect of parachute configuration on payload characteristics and payload subsystem requirements. It was found that the ring-sail canopy best satisfied the design requirement.

Utilizing the terminal descent velocity expression, a useful parameter involving parachute area and suspended weight can be evolved. (i.e. $A_{mc}/W_{susp} \propto \frac{1}{V_v^2}$). Figure 26 depicts this parameter as a function of impact velocity for each of the atmospheres under consideration. Note that the area (A_{mc}) is based on a nominal parachute diameter and that the system weight scales approximately linearly with required parachute area. For a vertical sea-level descent velocity of 80 ft/sec the required ratio of A_{mc}/W_{susp} is approximately 6.1 ft^2/Earth pounds in the terminal descent atmosphere. Based on the required A_{mc}/W_{susp} values in the terminal descent atmosphere, Figure 27 presents the nominal main parachute diameter (ring-sail canopy) as a function of impact velocity for a range of suspended weights. The reference capsule design has a suspended weight of 926 pounds. Based on this weight and an impact velocity of 80 ft/sec, a 85-foot nominal diameter parachute is required. Trajectory results for the reference M/C_DA of 0.15 slugs/ft^2 indicate that for deployment of the main parachute at Mach 1.3, the maximum opening dynamic pressure is approximately 10 lb/ft^2. Figure 28 shows parachute weight versus opening dynamic pressure for a range of nominal parachute diameters. The indicated parachute weight reflects only the canopy and shroud lines. The weight of the riser line, pilot parachute, canisters, etc. must also be included. For the reference 85-foot nominal diameter parachute and an opening dynamic pressure of 10 lb/ft, the required canopy and shroud line weight is 70 pounds. However, reefing of the parachute (18 percent of projected parachute area) equalizes opening shock loads and lowers the extremely high descent times in the model 1 and two atmospheres, and

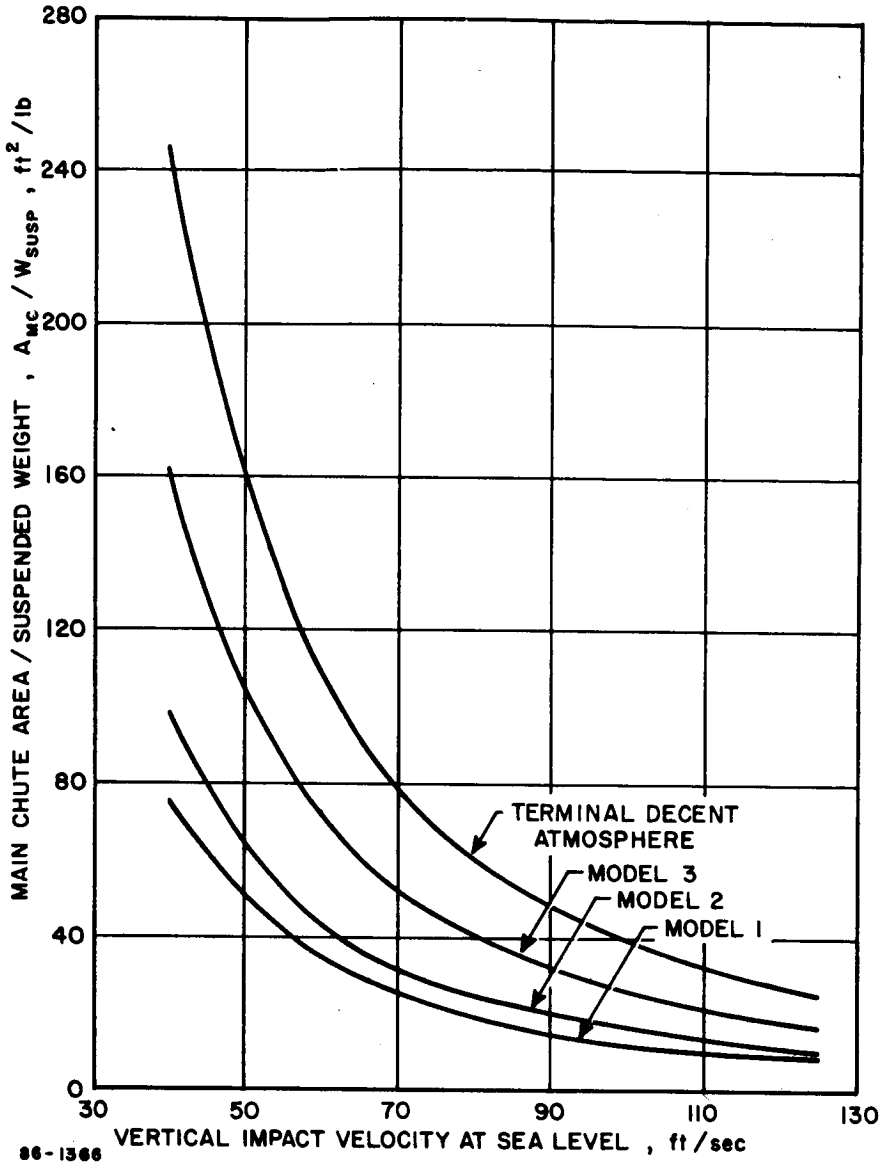


Figure 26 SINGLE PARACHUTE SYSTEM-IMPACT VELOCITY VERSUS MAIN PARACHUTE AREA/ SUSPENDED WEIGHT

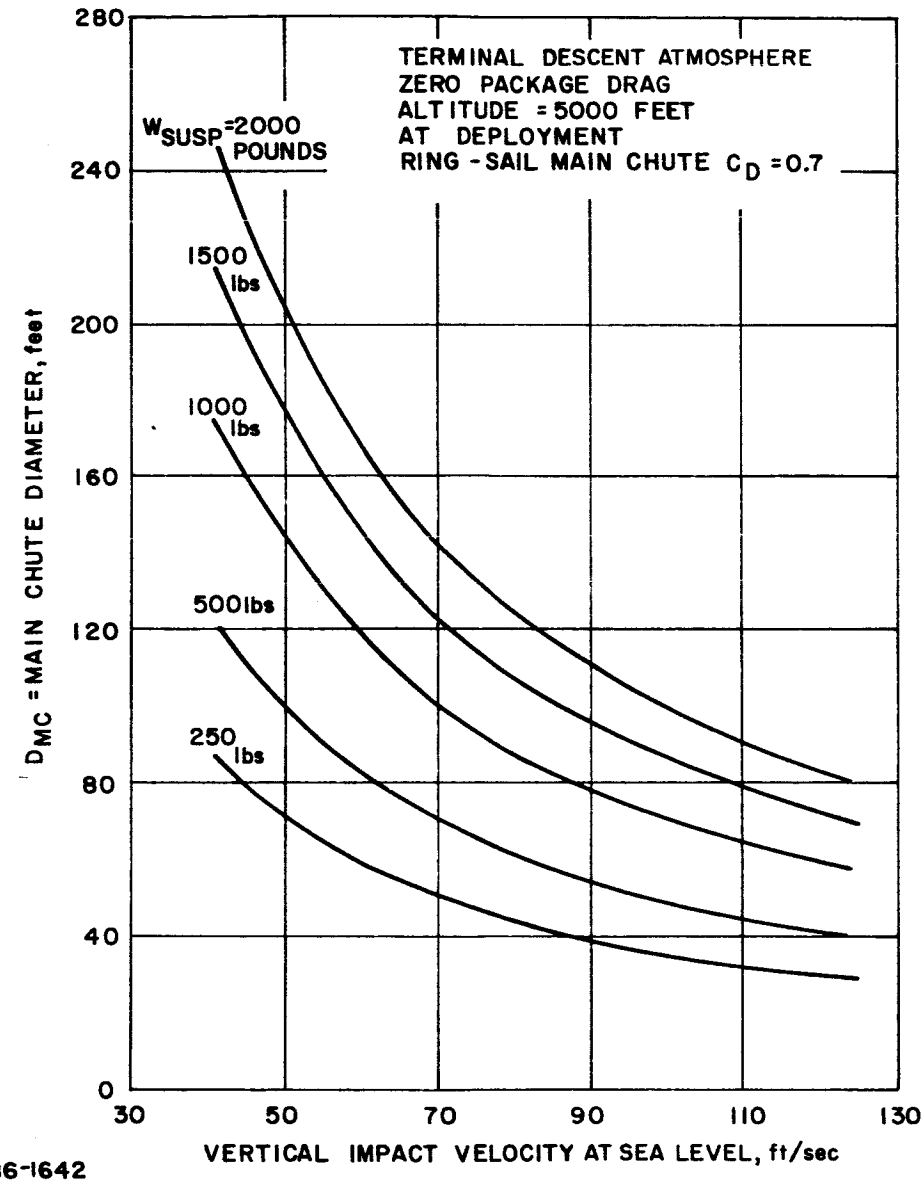


Figure 27 SINGLE PARACHUTE SYSTEM MAIN PARACHUTE NOMINAL DIAMETER VERSUS IMPACT VELOCITY

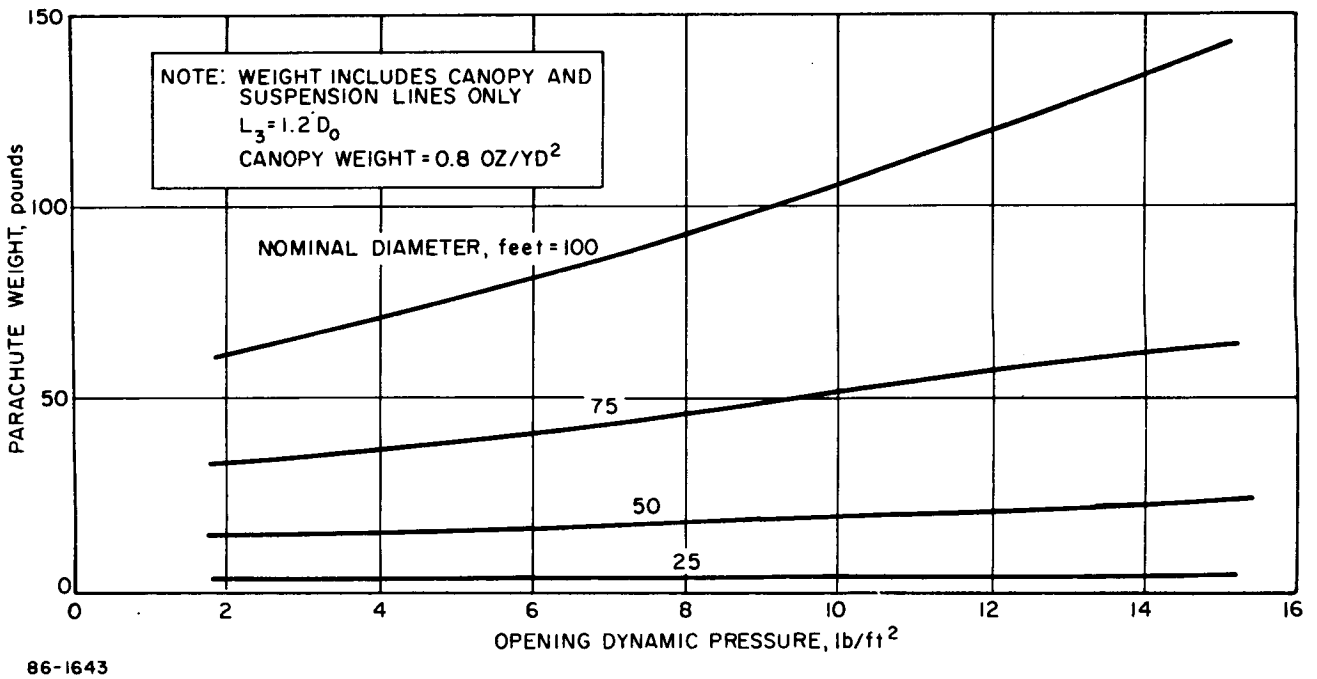


Figure 28 MAIN PARACHUTE WEIGHT VERSUS DYNAMIC PRESSURE

reduces the weight by some 20 percent to 56 pounds. Including the suspension line, harness assembly, pilot parachute, and deployment canister, the final total system weight is 74 pounds.

It is of prime importance to maintain parachute descent time within a reasonable minimum and maximum range in order to provide adequate but not excessive communications time. Figure 29 plots descent time versus A_{mc}/W_{susp} for parachute full open at 16,000 feet. For the four atmospheres under consideration and the design A_{mc}/W_{susp} of 6.1 lb/ft^2, the descent time ranges from 175 seconds to a maximum of 332 seconds. This range of descent time is within the overall system operational requirements.

4.7.3 Subsystem Mechanization

The pilot parachute is deployed nominally at Mach 1.3. The pilot parachute in turn pulls the main parachute out of its canister in a reefed condition. Satisfactory operation of the descent system is dependent upon initiation and upon the separation mechanisms used to accomplish the deployment sequence.

Initiation of the system is based on a peak acceleration - time correlation. Figure 30 is a plot of time from peak acceleration to Mach 1.3 versus peak acceleration. The data points are based on the range of trajectories and atmospheres considered. A curve fit of the data yields a correlation between peak entry g and time. The curve fit shown ensures initiation at Mach 1.3 or less for all cases and results in deployment altitudes above 16,000 feet where disreefing of the parachute occurs. Figure 30 presents data for only one entry velocity, (Volume III, Book 2, Section 6.7.4, presents a summary curve utilizing this type of initiation system where entry velocity is expressed as part of the curve-fit equation).

Implementation of the reference initiation system requires sensing vehicle axial acceleration. The magnitude of the peak value is read into an analog circuit which correlates it to a given Δt as shown on Figure 30. At peak acceleration a timer is started and at the end of its Δt excursion, initiation takes place.

At the end of Δt excursion a signal is sent to an electrical squib which in turn ignites the mortar charge. The mortar fires the pilot parachute out of its canister at 100 ft/sec. The pilot parachute in turn pulls the main parachute out of its canister. In the event the mortar fails to fire the the pilot parachute, a gas generator is used as a backup ejecting the main parachute directly out at 30 ft/sec. Once the main parachute is out and inflated, it descends in a reefed condition to 16,000 feet at which point disreefing of the parachute takes place. The disreefing altitude is sensed by a radar altimeter which sends a signal to the reefing line cutters.

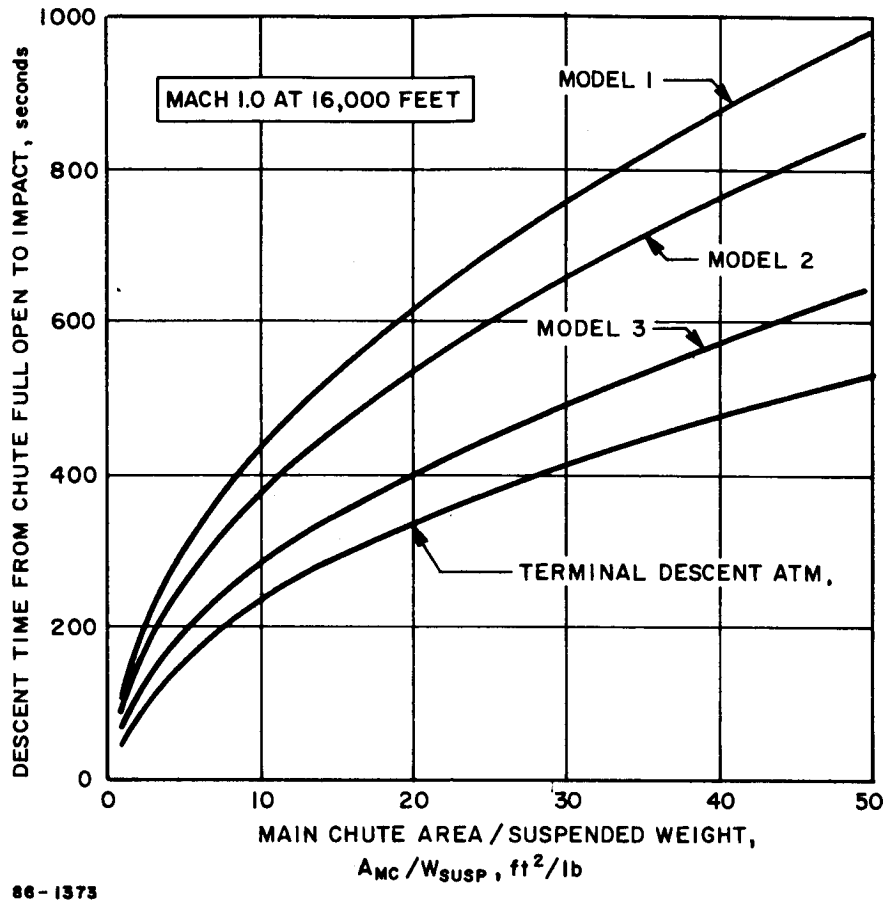


Figure 29 DESCENT TIME VERSUS MAIN PARACHUTE AREA/SUSPENDED WEIGHT

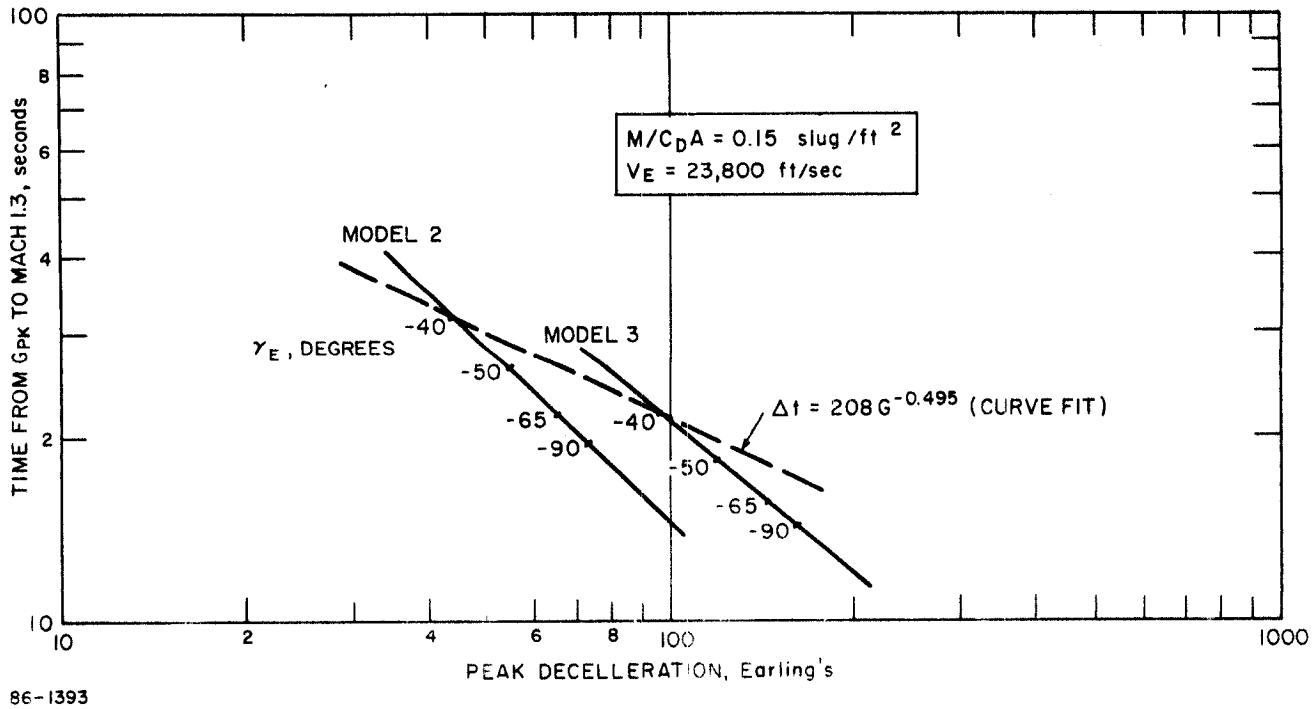


Figure 30 TIME FROM PEAK ACCELERATION TO MACH 1.3 VERSUS PEAK-g

4.8 IMPACT ATTENUATION SUBSYSTEM

4.8.1 Objectives and Requirements

An impact attenuation is employed to absorb the initial landing impact kinetic energy and any subsequent impacts as the landed capsule comes to rest on the Martian surface. The landed capsule should be capable of surviving the resultant velocity of 100 ft/sec horizontal from steady winds during descent, and 80 ft/sec vertical velocity* without exceeding 1000g in the instrumentation packages. It is desirable to have the impact attenuators made of RF transparent material in order to eliminate duplicate antenna systems for the communication link during descent and to eliminate the need for removal of the attenuator after impact.

Some of the important terrain features of the Martian surface which affect designs of the impact attenuation subsystem are: the surface bearing strength, which defines the depth of penetration into the surface; the slope of local terrain, which affects the landing orientation and final erection; and the roughness of the terrain, which defines the depth of impact material required to avoid local hazards such as rocks.

Other constraints on the impact attenuator are the configurational constraints imposed by the entry vehicle shape. These are further discussed in Section 6.2. An oblate spheroid landed capsule configuration was selected as the reference design and flotation sphere - landed capsule as an alternative approach. All environmental and mission constraints imposed on the overall flight capsule must also be considered in the design of the impact attenuator. Such environments as sterilization, space vacuum, and cold temperature soak are of particular concern.

4.8.2 Subsystem Design

Design of an impact attenuation system primarily evolves around the selection of the type of attenuator and materials available to meet the above objectives. The high horizontal wind and the necessity for omnidirectional protection eliminated many impact attenuator schemes such as penetration spikes, liquid shock absorbers, etc. The design was directed toward crushable materials and air bags because of their relatively high-impact energy absorbing-efficiency. It was found however, that air bags were considerably heavier then the crushable material approach.

*Optimization analysis between the impact attenuator and the parachute subsystem proved that 80 ft/sec vertical descent velocity was optimum in terms of maximizing the available internal weight.

The first step in the design of the impact attenuator was to select the proper crushable material based on presently available data. A survey of material properties indicated that fiberglass honeycomb with polyurethane foam-filled cells best suited the energy absorption characteristics (including good anisotropic behavior) required for the design. Plastic foam materials prove too inefficient (large stroke). The design resulted in using 3/16-inch cell size fiberglass honeycomb-filled with 2 lb/ft^3 of polyurethane foam, which represents an overall density of 4 lb/ft^3. On the oblate spheroid the required thickness is 15 inches on the minor axis, and 23 inches on the major axis. For the alternate design, the flotation sphere, the required thickness is a constant 15 inches. On both designs, a thin layer ($\sim$1 inch) of balsa wood is provided on the inner surface to protect the internal package from the penetration hazards of sharp rocks.

4.8.3 Subsystem Performance

The thickness of impact attenuation used in these designs was determined on the basis of 500-g peak deceleration resulting from impact in any direction. This deceleration load is based on landing on a flat, hard surface at a resultant velocity of 130 ft/sec. A small weight savings ($\sim$5 to 10 percent) could be realized by increasing the peak g level to 1000 g, however, the 500-g level was used to reduce the instrumentation development problems.

One of the more significant design problems is the landed capsule penetration into the planet's surface. Preliminary analysis for the oblate spheroid design and the most conservative surface model showed a maximum penetration of 0.61 foot.

4.8.4 Subsystem Mechanization

In order to preserve the crushing stress characteristics, in particular the good anisotropic properties, of the attenuator throughout the thickness, the honeycomb material is uniformly oriented along the radial direction. To accomplish this the honeycomb material is constructed of three layers with a thin sheet of fiberglass between each layer and on the cover. This composite is then bonded to the inner layer of balsa wood which is in turn bonded to the internal structural shell. On the oblate spheroid design, deployment of instruments is accomplished through holes made in the crushable material by sheet explosives. On the flotation sphere, the attenuator is made up of 14 segments, each segment is constructed of three layers of honeycomb bonded together as for the oblate spheroid. Each segment cover sheet overlaps the adjacent segment, thus creating a locked joint between segment from separating during impact. All of the segments are bonded to a continuous inner spherical shell which houses a

separation system for jettisoning the attenuator after impact. While it is desirable not to jettison the impact attenuator, the operation of the flotation sphere makes it necessary to ensure that the damaged part of the attenuator is not impeding data transmission and to allow instrument deployment.

4.9 ENTRY SHELL

Three generic entry vehicle shapes were considered for the flight capsule, the blunted cone, modified Apollo, and tension shell. The blunted cone shape was selected on the basis of a slightly better entry shell weight fraction and higher confidence in the aerodynamic performance prediction.

A major design goal of the study was the development of an entry shell with multi-mission capability. The multi-mission entry shell would have the capability of entering any model of the Martian atmosphere at any entry velocity up to 23,800 ft/sec, any entry angle from -20 to -90 degrees, and any M/C_DA up to the 4500-pound entry weight limit for future mission. The multi-mission shell, however, imposes too large a weight penalty on the weight-limited 1971 mission.

The selected multi-mission structure - 1971 heat-shield entry-shell concept retains most of the development advantages of the multi-mission shell, requiring only that a new heat shield be designed for each launch opportunity.

4.9.1 Design Requirements

The primary design requirements upon the entry shell result from the environment during atmospheric entry. A summary of the loading and heating parameters which designed the entry shell structure and heat shield is shown in Tables XVI and XVII.

4.9.1.1 Loads

The entry-load parameters are presented for normal (zero-angle of attack) entry and for the failure-mode rearward entry. For rearward entry (179-degree angle of attack), the entry vehicle is righted with the aid of a flap, however, the loads shown in Table XVI assume a flap failure. Table XVI also indicates the maximum normal loads. A 25-percent increase in loads generally occurs in rearward entry.

4.9.1.2 Heating

Pertinent heating data is presented in Table XVII. Angle of attack causes increased total integrated heating for the blunted cone; in general, high spin rates decrease heating.

TABLE XVI

LOAD SUMMARY - BLUNTED CONE - MULTI-MISSION STRUCTURE

Entry Weight = up to 4500 pounds
Entry Velocity = 23,800 ft/sec
Entry Angle = -90 degrees
Diameter = 15 feet
Model 2 Atmosphere

	Entry Weight (pounds)	Peak Dynamic Pressure (lb/ft^2)	Peak Axial Acceleration (g)	Peak Normal Acceleration (g)	Angle of Attack at Peak Loads (degrees)
Normal Entry $\alpha_E = 0°$	4500 (M/C_DA - 0.42)	1152	73.3	--	--
	1390 (M/C_DA - 0.15)	777	161	--	--
Failure Mode Entry $\alpha_E = 179°$	4500 (M/C_DA - 0.42	1426	91	7	16
	1390 (M/C_DA - 0.15)	1009	209	19	20

TABLE XVII

HEATING SUMMARY - BLUNTED CONE - 1971 HEAT SHIELD

Entry Weight = 1390 pounds
Entry Velocity = 23,800 ft/sec
Entry Angle = -20 degrees
Diameter = 15 feet
Entry Angle of Attack = 0 degree
Model 2 Atmosphere

Peak Heating Rate ($Btu/ft^2/sec$)		Total Integrated Heating (Btu/ft^2)		
		Stagnation Point		Sonic Point
Convective	Radiative	Convective	Radiative	Convective
70	1	2798	19	803

4.9.2 Structure

4.9.2.1 Objectives and Requirements

The primary shell and internal structure is required to maintain its integrity throughout its operating sequence. This sequence begins at the factory and includes: 1) sterilization cycle temperatures and loads, 2) handling and transportation loads, 3) ascent loads, 4) spaceflight temperatures 5) maneuvering and separation forces, 6) entry temperatures and loads, and 7) parachute-opening shock loads.

The most important constraint on the structural design is the desire for minimum entry weight. To achieve this objective, the most efficient practical structural configuration and materials were considered.

A summary of the design conditions for the flight capsule is given in Table XV.

For the multi-mission structure entry shell, the inertia of the payload was assumed to be distributed over the rear face of the shell for the early mission and concentrated on a circular line for the future

missions. The inertial force of the future mission payload was larger than the inertial force of the 1971 payload although the deceleration of the capsule was greater for the 1971 mission. This was because the mass of the 1971 payload was considerably smaller than that of the future mission payloads. The structural requirement of the nose portion of the multi-mission shell was governed by the 1971 requirements, while the outer portion of the shell was designed for the higher external pressure associated with the large payload of future missions.

4.9.2.2 Subsystem Design

A honeycomb sandwich structural concept was selected for the blunted-cone entry shell. For the lightly loaded large shell structures considered, this type of construction is the most efficient of the various state-of-the-art types of construction. Because of the requirements for very low weight, beryllium was used for the face-sheet material. The honeycomb core selected was stainless steel due to its thermal expansion compatibility with beryllium and its lower costs and greater availability. In order to achieve an optimized structure, the face-sheet thicknesses and core depths were selected so that the stresses in the face sheet approached the yield strength of the material, while at the same time the structure was at the point of incipient buckling. In order to accomplish this, it was necessary to increase face-sheet thicknesses and core depths locally at points of concentrated-force application and at cutouts.

4.9.3 Heat Shield

4.9.3.1 Objectives and Requirements

The requirements imposed on the heat shield parallel those for the structure through the mission sequence from the factory to parachute deployment. During the spaceflight phase, the heat shield is aided by the thermal control system and together they assure integrity of the structure and of the payload.

The assurance of the integrity of the structures and of the payload through attenuation of the external thermal environment during entry at minimum weight expenditure are then the objectives of the heat shield design.

In achieving these objectives the main constraint upon the heat shield is to accommodate the critical environments created at the boundaries during the entry phase while providing protection for the structure to perform its function. The design conditions for the heat shield stem from the aerodynamic environments and from the structural design criteria.

The satifaction of the design conditions is predicted on the availability, selection, and understanding of the behavior of appropriate heat shield materials, i. e., materials displaying a proper combination of thermal, optical and ablative characteristics. As a result, complex interactions have to be considered in establishing the heat shield design and material specifications.

On the other hand, the weight of the heat shield is sensitive to the initial conditions (temperatures) existing at the onset of entry. These temperatures depend on the flight capsule thermal control exercised prior to entry, while attached to the flight spacecraft and during cruise. Thus, in addition to the environmental, structural and material requirements present in any entry vehicle design, a set of thermal-control constraints may exist for the heat shield (or vice versa).

Finally, the requirement for decontamination and sterilization imposes a constraint on the selection of heat shield materials from the beginning of the design process, limiting the choice to only such materials that can satisfy this initial requirement. A summary of design conditions used for the reference design is given in Table XVII and in Table XVIII.

TABLE XVIII

SUMMARY OF HEAT SHIELD DESIGN DATA
(BLUNTED CONE)

M/C_DA	0.15
α_e	11.5 degrees
$\alpha_{peak\ heating}$	< 1 degree
$Q_{stag\ tot}$	2798 Btu/ft^2
$\dot{q}_{max\ diam}$	70 Btu/ft^2
Pulse duration	130 seconds
Material heat shield	Cork Silicone
Material structures	Beryllium 0.020 inch
$T_{structure}$	300 to 500°F
T at entry	100 to 300°F
S.F.	1.0
End of pulse	Impact

The design of the heat shield presented is for the normal entry mode only, and is valid for an M/C_DA of 0.15 slug/ft^2.

4.9.3.2 Subsystem Design and Performance

The thermal protection concept applied to the blunt cone was an ablative heat shield which appeared to be most efficient for at least the forebody of the entry vehicle for the anticipated environmental conditions. Low-density and low-conductivity materials with moderately good ablative characteristics were desirable for this application. A number of materials with these characteristics were investigated and are shown in Table XIX.

Of these material candidates, only four were selected for further investigation either on the basis of more acceptable thermal properties (since most of them exhibited similar gross ablative behavior) or minimum degradation in mechanical (as well as thermal) properties when subjected to simulated decontamination and dry heat sterilization cycles. The preliminary thermal properties and ablative characteristics used in the design studies with these materials are shown in Table XX.

In the parametric studies preceding the selection of aerodynamic shape and base diameter, the relationships between the heat shield thickness (for the preceding four materials) total aerodynamic heating and allowable structural temperature were established which were used then together with the relationship between aerodynamic heating and base diameter in systems tradeoff. The angle-of-attack effect on heat shield weight was also investigated and was found to be small. Once the reference designs were established, appropriate local heat shield thickness and weight were calculated and are shown for the reference designs and concepts in Table XXI for the forebody heat shield only. The cork silicone material displayed the minimum requirement for the heat shield weight and thus was selected for reference purposes. The heat shield weight fraction is shown in Table XXI.

4.10 THERMAL CONTROL

4.10.1 Objectives and Requirements

The main objective of the thermal control system is to maintain electronic components, batteries, structural members, and the heat shield within specified temperature limits during the various phases of flight. It must be compatible with the flight spacecraft within the available power and the general weight allocation. It must be compatible with other flight capsule systems allowing for nominal performance (failure modes) during all the phases of the mission. The system should be passive if at all possible. Its passive elements should not degrade the performance of other materials,

TABLE XIX

COMPARISON OF CANDIDATE MATERIALS

Material	Insulative Heat of Ablation (Btu/lb)
Purple Blend Mod 5	15,000
Cork silicone	12,000
Avcoat 5026-99	11,000
DC2048	10,000
Avcoat 5026-39M	11,000
Avcoat 3008	11,000
Armstrong 2755 cork	11,000

TABLE XX

THERMAL PROPERTIES AND ABLATIVE CHARACTERISTICS OF MATERIALS

Property \ Material		P. B. Mod. 5	Cork Silicone	5026-99	Low Density Nylon Phenolic
K	Btu/ft-hr-°F	0.067	0.045	0.045	0.058
ρ	lb/ft^3	37.5	25.6	24.0	36.0
C_p	Btu/lb-°R	0.387	0.52	0.44	0.40
ϵ		0.60	0.59	0.59	0.80
η_L		0.51	1.26	0.81	0.995
η_T		0.17	0.42	0.27	0.332
H_V	Btu/lb	-330	-2390	-1850	-2075
T_A	°F	3000	3000	3000	3000

TABLE XXI

BLUNTED-CONE HEAT SHIELD REQUIREMENTS

Entry Weight = 1390 pounds
M/C_DA = 0.15 slug/ft
Atmosphere 2
V_e = 23,800 ft/sec
γ_e = 20 degrees
Diameter = 15 feet

Station (S/R_B)	Q_{conv} (Btu/ft^2)	Q_{rad} (Btu/ft^2)	Nylon Phenolic (inches)	Nylon Phenolic (lb/ft^2)	Cork Silicone (inches)	Cork Silicone (lb/ft^2)	Avcoat 5026-99 (inches)	Avcoat 5026-99 (lb/ft^2)	Purple Blend (inches)	Purple Blend (lb/ft^2)
0.0	2800	19	0.52	1.56	0.45	0.96	0.50	1.00	0.54	1.69
0.2	1875	30	0.48	1.44	0.42	0.90	0.47	0.94	0.51	1.59
0.6	900	76	0.43	1.29	0.38	0.81	0.43	0.86	0.47	1.47
1.0	800	108	0.41	1.23	0.37	0.79	0.42	0.84	0.46	1.44

$$\text{Weight Fraction} = \frac{\text{Forebody H/S}}{\text{Entry Vehicle}} = 14 \text{ percent-cork silicone}$$

or themselves be degraded by other materials. Flight capsule temperature control must be maintained during surface operations as well. The thermal control system must minimize the thermal disturbance to the flight spacecraft at flight capsule separation. Alternatively, the thermal interface between the flight spacecraft and the flight capsule must be defined to determine, for the particular configuration, the relationships between the various allowable temperatures, flight spacecraft power available for heating and the optical property requirements for the flight capsule thermal control coatings. It has been assumed that the base of the flight capsule sterilization canister was isothermal at -20°F near Mars (Ref. 4). The temperature limitations of the system components, and other design conditions and assumptions used in the design and performance analysis are shown in Table XXII.

4.10.2 Design and Performance

The scope of thermal control analysis and design was largely limited to the investigation of the critical conditions which might arise during the mission to establish the limiting thermal control system requirements for the conceptual flight capsule design. The performance analysis was devoted to the various failure modes which might occur due to unfavorable flight capsule - sun orientation during various phases of the mission, and to the determination of the sensitivity of the flight spacecraft power requirement to the temperature-emissivity (α/ϵ) relationship of the flight capsule thermal control coatings.

The performance requirements for the blunted-cone 1971 mission, based on the temperature limitations shown in Table XXII and the parametric results shown in Table XXIII and Figures 31 and 32 lead to recommendation of the thermal control system shown in Figure 33. This system consists of a low ϵ (0.05) thermal control coatings on the primary and secondary heat shield faces and a moderately low ϵ (α/ϵ = 1 to 3) coating on the afterbody to maintain the critical components within the allowable temperature range during the postseparation phase. Since the external battery tends to run at low temperatures, some heat will have to be supplied by the flight spacecraft during interplanetary cruise, and it will have to be additionally prior to separation. Superinsulation will be required to maintain proper operation temperatures (40 to 160°F) after separation. The internal battery may not require heat during cruise; however, it also must be insulated and warmed up prior to separation to maintain a sufficiently high operating temperature for post-impact use. The above design will accommodate the anticipated post separation flight capsule - sun orientation angles (53 to 90 degrees from the rearward axial direction) and will not cause overheating for direct sun impingement. The oblate-spheroid landed capsule with its metallic surface will operate within prescribed limits. The post-separation phase presents the most difficult thermal control problem in absence of a heat source in the flight capsule.

[4]Conceptual Design Studies of an Advanced Mariner Spacecraft IV, Flyby Bus Design RAD-TR-64-36 Section 9, Thermal Control (28 October 1964).

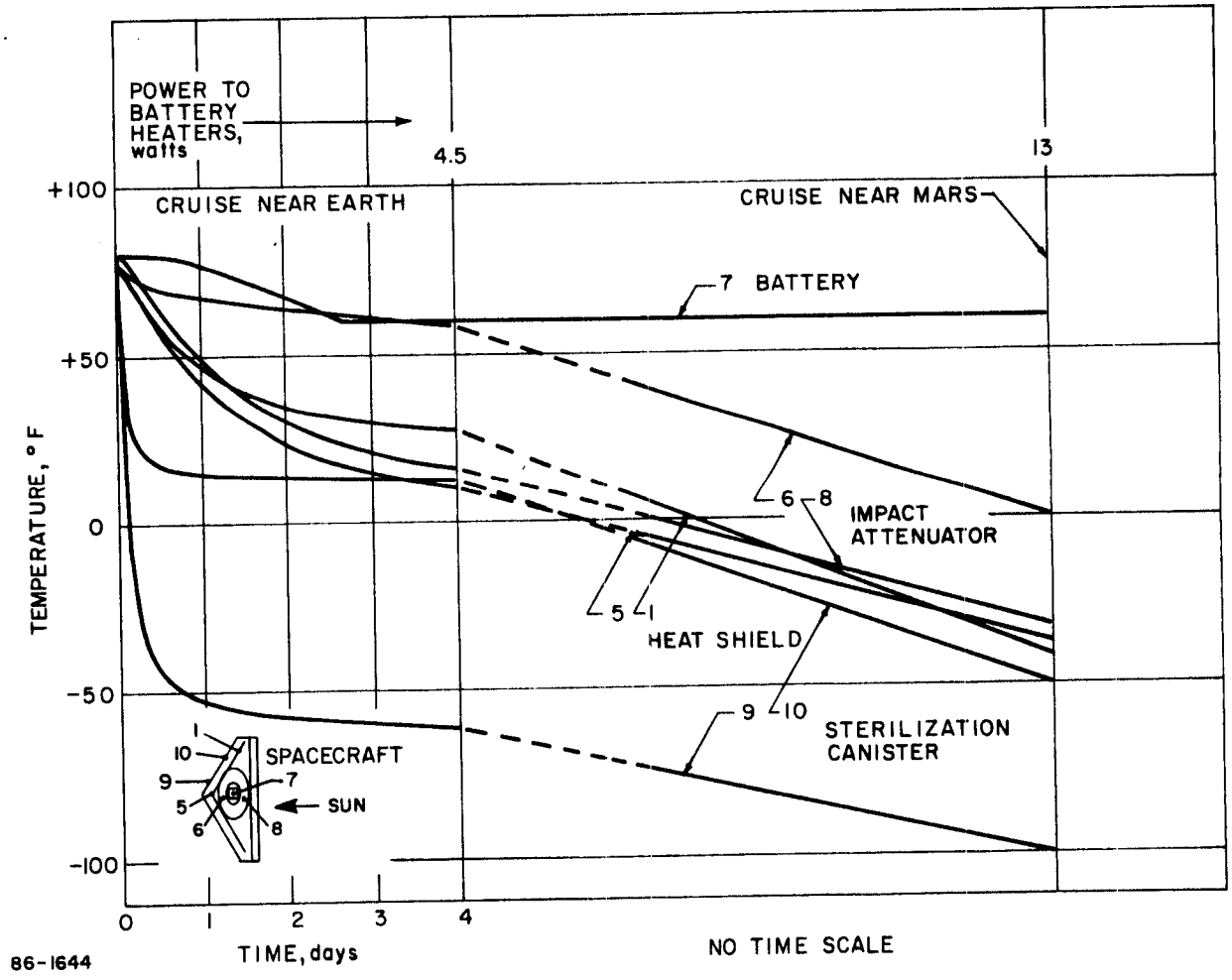


Figure 31 BLUNTED CONE 180-INCH DIAMETER FLIGHT CAPSULE WITH OBLATE SPHEROID LANDED CAPSULE, CRUISE NEAR-EARTH AND NEAR-MARS TEMPERATURE HISTORY

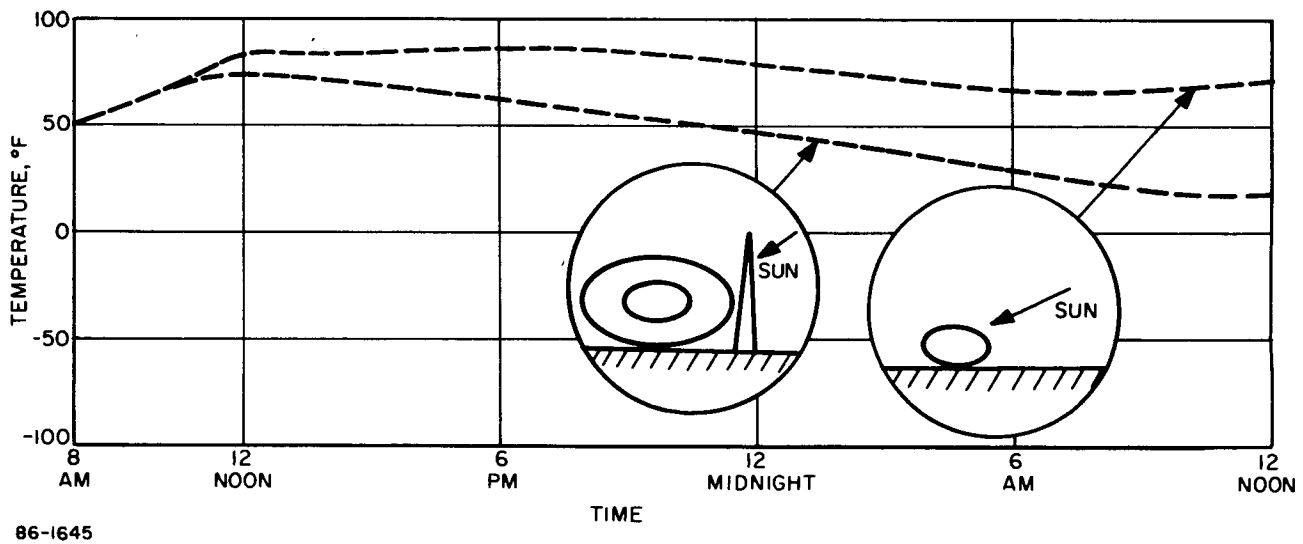


Figure 32 TYPICAL POSTIMPACT PAYLOAD TEMPERATURE HISTORY

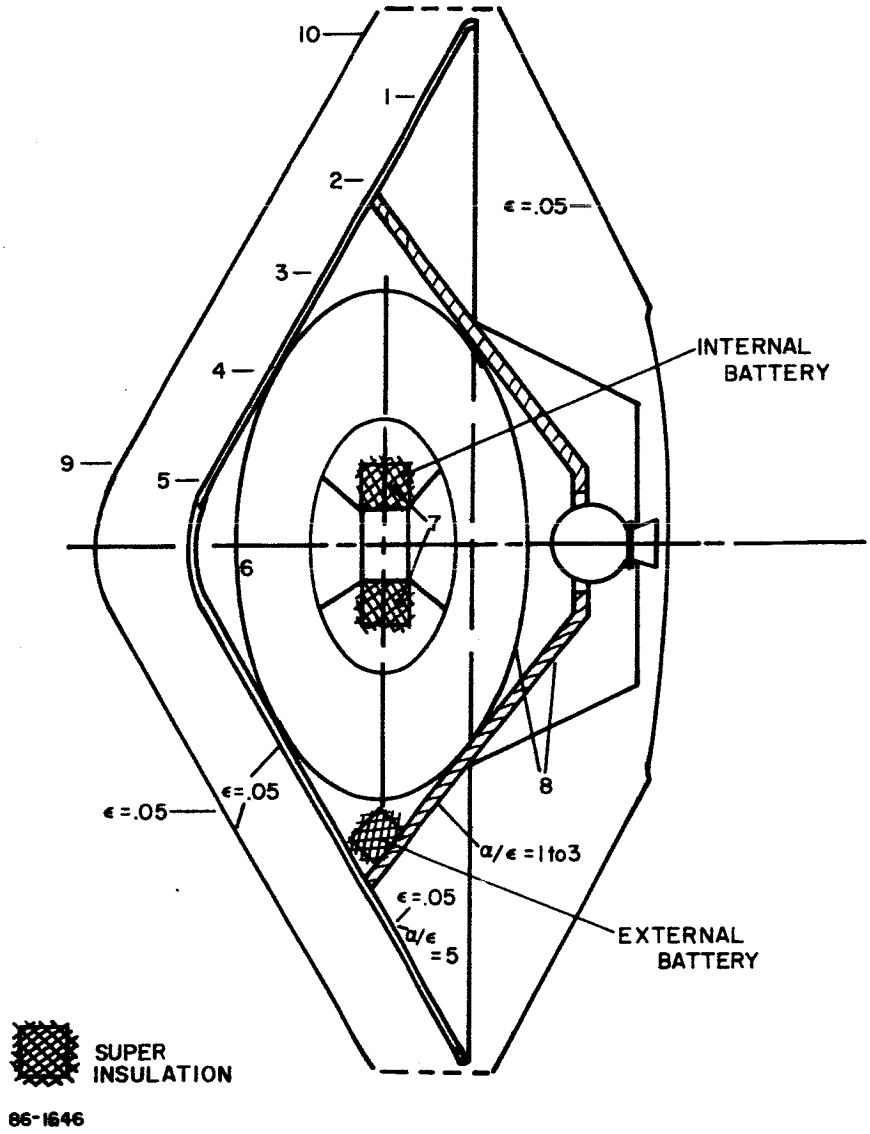


Figure 33 THERMAL CONTROL SYSTEM

TABLE XXII

TEMPERATURE LIMITATIONS FOR STRUCTURAL MEMBERS AND COMPONENTS

Critical Component or Subsystem	Typical Temperature Specification	Critical Phase of Operation	Thermal Design Approach
Batteries	-65 to 160°F nonoperative +40 to + 160 °F operative	a) Cruise near Mars (low temperature) b) Postseparation dependency on entry vehicle orientation relative to the sun c) Postimpact (diurnal cycle)	1. Insulation + heaters using spacecraft power. 2. Low-ϵ coatings on external surface of heat shield and afterbody. Insulation and warm-up prior to separation.
Propulsion	-60 to 300°F nonoperative 0 to 140°F operative	Just prior to separation and post-separation.	electrical heaters and insulation
Instrumentation	-35 to 300°F nonoperative 0 to 140°F operative	Postimpact	Insulation + heat sink
Electronics	-35 to 300°F nonoperative 0 to 140°F operative	Postimpact	Insulation + heat sink
Parachute system	-35 to 300°F	Entry	Insulation from entry heating
Explosive systems	-160 to 300°F	Space flight, cruise mode	Overall system design
Heat shield and structure	To be determined (low entry temperatures desired)	Entry Entry	Ablation shield insulation (if required)
Impact attenuator	-160 to 300°F nonoperative	Impact	Overall systems design. Low ϵ on heat shield.
Sterilization canister	-160 to 300°F	Space flight, cruise mode and midcourse maneuver	High/α/ϵ
Flight spacecraft/flight Capsule interface	+ 60°F -20°F	Cruise near Earth Cruise near Mars (x)	If lower temperatures obtained due to F/S-F/C interface, heaters to be installed in the H/S

(x) Assumption based on Ref. 1.

TABLE XXIII

BLUNTED CONE 180" DIA. FLIGHT CAPSULE WITH OBLATE SPHEROID LANDED CAPSULE

(Postseparation steady-state temperatures)

Structure or Component	Node Designated	No Afterbody			Afterbody Metallic Coating $\alpha/\epsilon = 5.0$			Afterbody Nonmetallic Coating $\alpha/\epsilon = 0.6$			Temperature at Separation (°F)
		α/ϵ	Case A	Case B	α/ϵ	Case A	Case B	α/ϵ	Case A	Case B	
Heat shield	1	5.0	+7	+229	5.0	+8	+229	5.0	+8	+229	-44
Heat shield	2	5.0	+8	+224	5.0	+9	+224	5.0	+8	+224	-48
Heat shield	3	5.0	+14	+212	5.0	+16	+213	5.0	+13	+211	-50
Heat shield	4	5.0	+28	+109	5.0	+33	+116	5.0	+25	+105	-54
Heat shield	5	5.0	+30	+81	5.0	+36	+89	5.0	+27	+76	-54
Crushable material	6	1.0	+20	+112	1.0	+30	+126	1.0	+15	+103	-52
Battery	7	---	-48	+120	---	+4	+208	---	-78	+60	-41
Crushable material (1)	8	1.0	-139	+131	5.0	-31	+320	0.6	-202	+7	-28 (2)

(1) Or afterbody
(2) For metallic afterbody

Case A - Normal sun impingment

Case B - Axial sun impingment from rear

5.0 ENTRY VEHICLE SHELL SELECTION

A primary objective of this study has been the comparison of various approaches to the conceptual design of a flight capsule to land on the Martian surface from approach trajectory. During the study, parametric analyses of the entry vehicle shell were performed to provide tradeoff data from which a recommended shell could be selected. This section presents those tradeoff analyses.

5.1 SHELL MISSION UTILIZATION CONCEPTS

The entry shell for the Mars flight capsule represents a major design and development task. Although the scientific mission of the flight capsule may change from one launch opportunity to the next, it would be desirable to utilize the same entry shell for each opportunity. The term entry shell, as used throughout this section, will refer to the basic load carring entry vehicle structure and the heat shield required for thermal protection during entry.

Several entry vehicle shell concepts were formulated for this study. Multi-mission shell refers to a shell that can be utilized over several launch opportunities. This concept would eliminate the necessity for expensive redevelopment of a new entry shell for each mission. The multi-mission shell, therefore, has been designed for entry velocities up to 23, 800 ft/sec, the maximum entry velocity anticipated through the 1975 launch opportunity; entry angles from -20 to -90 degrees to accommodate the full range of anticipated landing site requirements through the 1975 launch opportunity; model atmospheres 1, 2, and 3, and entry weights up to 4, 500 pounds, limited by the launch vehicle capability.

From the parametric weight analysis performed, it was determined that the multi-mission shell design concept placed a severe limitation on the payload weight available for the 1971 mission. To evaluate these weight penalties associated with the multi-mission shell concept, several single mission shell concepts have been postulated.

The first approach utilizes a multi-mission structure designed to withstand the variety of entry conditions postulated for the multi-mission shell. The heat shield is designed for a 1971 mission only. This concept has the advantage of providing a decrease in heat shield weight and yet having a partial multi-mission shell capability.

The second concept utilizes an entry shell designed specifically for the 1971 mission at an M/C_DA of 0. 15 slug/ft^2. The weight of both heat shield and structure can be decreased for the 1971 mission; however with this approach the entire multi-mission shell concept is discarded, requiring a new entry-shell development for each mission.

The remaining design approaches were formulated to determine the benefit of designing entry shells for the specific known atmosphere for future missions. Entry shells were synthesized for the model 3, 2, and 1 atmospheres to evaluate the penalty associated with the multi-mission shell for future missions in the event that the atmosphere of Mars is determined to be either model 3, 2, or 1.

5.2 ENTRY SHELL CONFIGURATIONS

This study compared three basic entry configurations. The design details of the blunted cone are discussed in Section 3.2. The designs of the modified Apollo and tension shell are discussed in this section insofar as they differ from the blunted-cone design.

5.2.1 Modified Apollo

5.2.1.1 Design Description

The design details of the modified Apollo* configuration follow very closely the design of the blunted cone configuration. Only a few minor differences are noted in the design as shown in Figures 34 and 35 for the launch and entry configurations, respectively. These differences are primarily in the entry shell and mounting arrangement, separation systems, and suspended capsule structure. The entry shell is a spherical cap with the same type of construction as the blunted cone (i.e., beryllium face sheets with stainless-steel honeycomb core). The details of this design are the same as the details of the blunted cone, except the face sheets are doubly contoured instead of singly contoured. This makes forming of the face sheets and bonding of the structure somewhat more difficult.

At the time this design was completed the thickness of the core was 2 inches and the face sheets were 0.030 inch thick. Further structural analysis has indicated that a tapered core varying from 0.50 inch at the center to 1.63 at the mounting ring to 0.27 at the outer rim can reduce the structural weight. Face-sheet thicknesses are 0.035 inch with doublers added at the mounting ring to bring the thickness to 0.065. These changes in the core and face-sheet thicknesses were not available in time to alter the design layouts but are reflected in the weight breakdown presented in Table XXIV.

The heat shield system for this vehicle is also cork-silicone, a constant 0.37-inch thickness over the entire spherical cap.

*The modification in the original Apollo configuration is the elimination of the 33-degree afterbody and the toroidal corner, thus leaving only a spherical cap section.

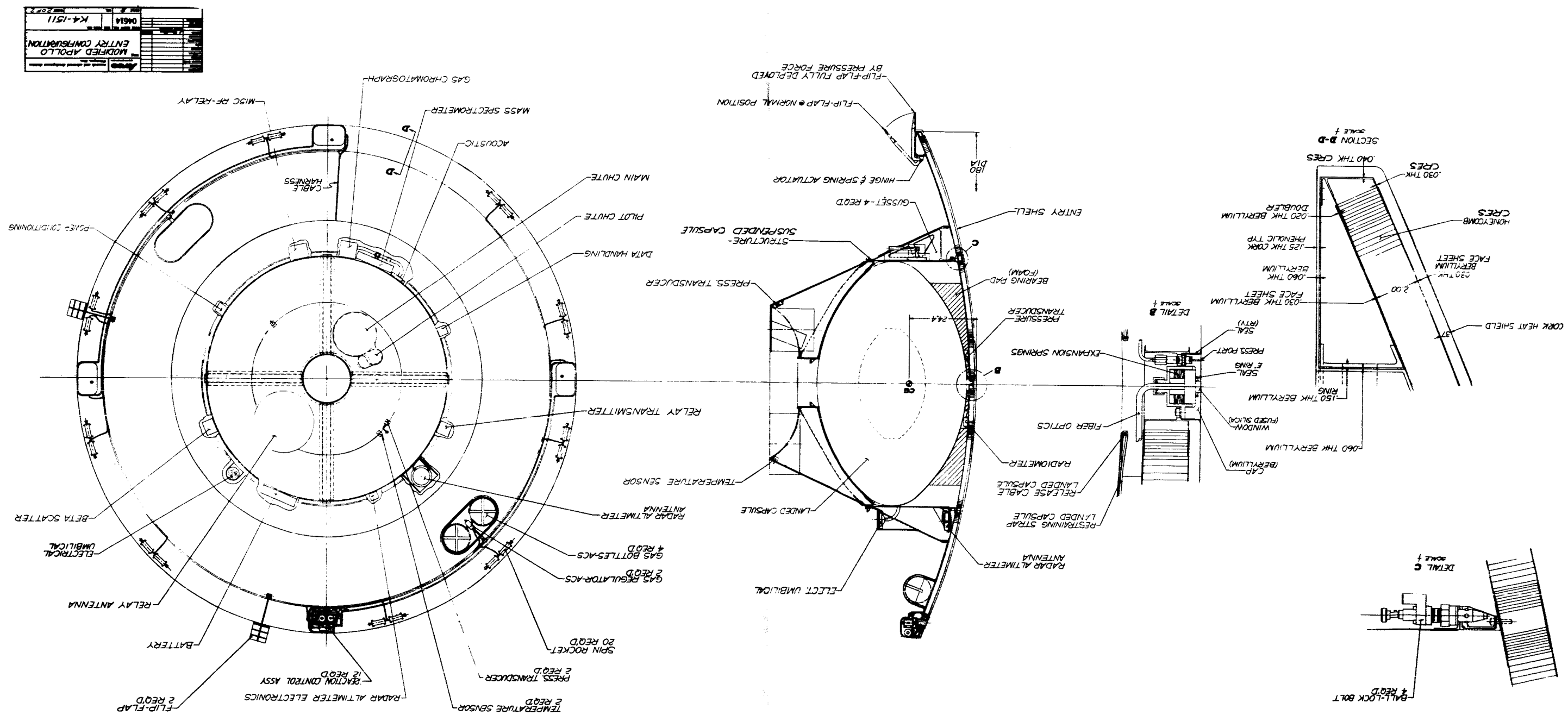
TABLE XXIV

MODIFIED APOLLO -- WEIGHT SUMMARY

	Weight (pounds)		c. g.* (inches)	Ixx (slug/ft^2)	Iyy (slug/ft^2)
Flight Capsule		2500.0	33.0	1210.7	880.8
FC/FS adapter	100.0				
Sterile canister	366.9				
Elec. and mech. connectors	50.0				
Separated Vehicle		1983.1	27.3	774.2	542.8
ΔV Propulsion	98.5				
ACS electronics	10.0				
Spin motors propellant	2.1				
ΔV propulsion structure	10.0				
Miscellaneous	12.5				
Entry Vehicle		1850.0	24.4	769.6	483.8
Entry shell-heat shield	249.5				
Entry shell structure	524.3				
Thermal control	25.0				
ACS nozzles, tanks, etc.	69.3				
Spin motors and supports	10.0				
Elec. and mech. connector	55.5				
Contingency	25.0				
Suspended Capsule		891.4	32.8	142.0	113.0
Instrumentation	40.4				
Telecommunications	20.6				
Power	37.5				
Miscellaneous	4.0				
Contingency (25 percent on above)	25.6				
Main chute, pilot, etc.	71.3				
Structure	143.0				
Afterbody	24.5				
Landed Capsule (available)		514.5	30.1	52.2	33.8

*Note: Center of gravity from entry shell forward nose location

Figure 35 MODIFIED APOLLO - ENTRY CONFIGURATION



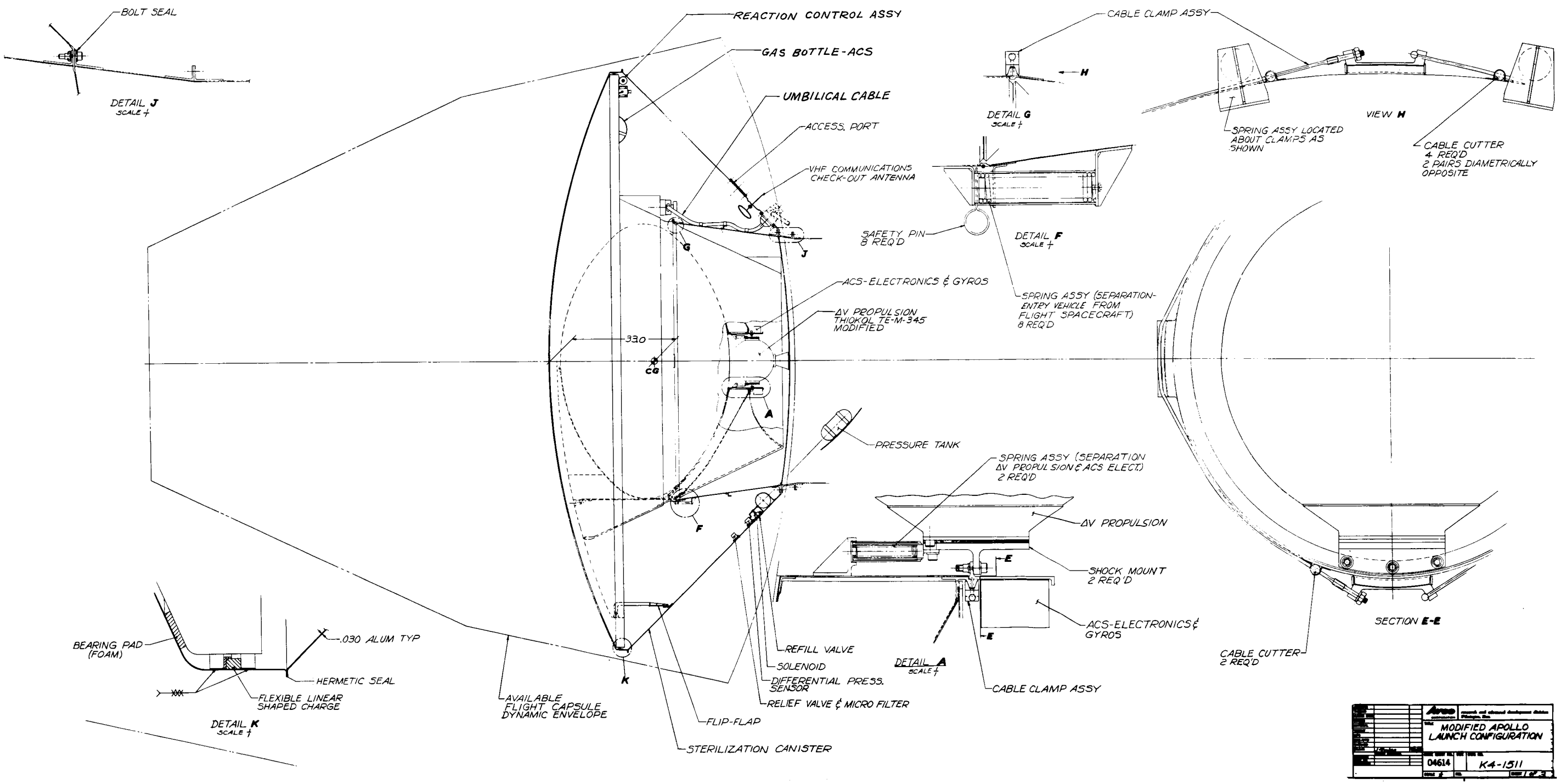


Figure 34 MODIFIED APOLLO - LAUNCH CONFIGURATION

117-1

An alternate separated vehicle/flight spacecraft separation system is presented for this design. This separation system is very similar to the blunted-cone ΔV propulsion package separation, i.e., clamp-cable mechanism (reference Section E-E, Figure 34).

The suspended capsule structure was also changed, due to the limited space behind the landed capsule and the restrictive ascent shroud-dynamic envelope. The modified Apollo must set further aft in the shroud than the blunted cone (to maintain 4-inch clearance around the 180-inch diameter vehicle), restricting the allowable space for the ΔV propulsion.

In this design, the suspended capsule structure, which consists of four radial beams and a cylindrical shell, fully cradles the landed capsule. The external equipment is again mounted on this structure, around the cylindrical shell section. Web gussets are located radially around the cylindrical section to support the forward conical portion of the afterbody, between the entry shell and the flight spacecraft adapter mounting ring. The aft cover of the suspended capsule is constructed of fiberglass as in the blunted cone and is mounted to the radial web structure. The concave aft end is provided to enhance the rearward entry righting moment and to provide clearance for the ΔV propulsion package separation.

All other features of the design are the same as the blunted-cone design described in Section 3.2. The flight sequence is also the same.

5.2.1.2 Weight Summary

A summary weight similar to that of the blunted cone is presented in Table XXIV. The significant weight difference is in the primary structure. This is illustrated in Table XXV where a detailed weight breakdown of the entry shell is given. The major difference between the blunted cone and modified Apollo structural design is the thickness of the honeycomb core. In the blunted cone, the core is 0.60 inch thick, while in the modified Apollo the core varies from 0.5 inch to 1.63 inches thick over most of the shell. The face-sheet thickness is also slightly greater on the modified Apollo (0.035 inch compared to 0.025 inch on the blunted cone).

In the first column of Table XXIV, external instrumentation and telecommunications is shown as in the blunted-cone design. The available landed capsule weight is again much too small to accommodate the desired payload. An adjusted weight synthesis similar to that of the blunted cone could be accomplished for this design to arrive at a weight balance between the suspended capsule and landed capsule.

TABLE XXV

MODIFIED APOLLO - ENTRY SHELL WEIGHT BREAKDOWN

PRIMARY STRUCTURE

	Weight (pounds)
Base ring	65.6
Beryllium face sheets	127.6
Stainless steel core	101.0
Stainless steel weldments	42.0
Beryllium splice plates	44.3
Epoxy bond	22.7
Contingency (at 15 percent)	60.5
Beryllium mounting rings	24.0
Foam bearing pad	36.6
Total	524.3

HEAT SHIELD

	Weight (pounds)
Cork primary heat shield	184.4
Epoxy bond	12.6
Cork secondary heat	43.5
Epoxy bond	9.0
Total	249.5

AFTERBODY

	Weight (pounds)
Cork heat shield	20.5
Epoxy bond	4.0
Total	24.5

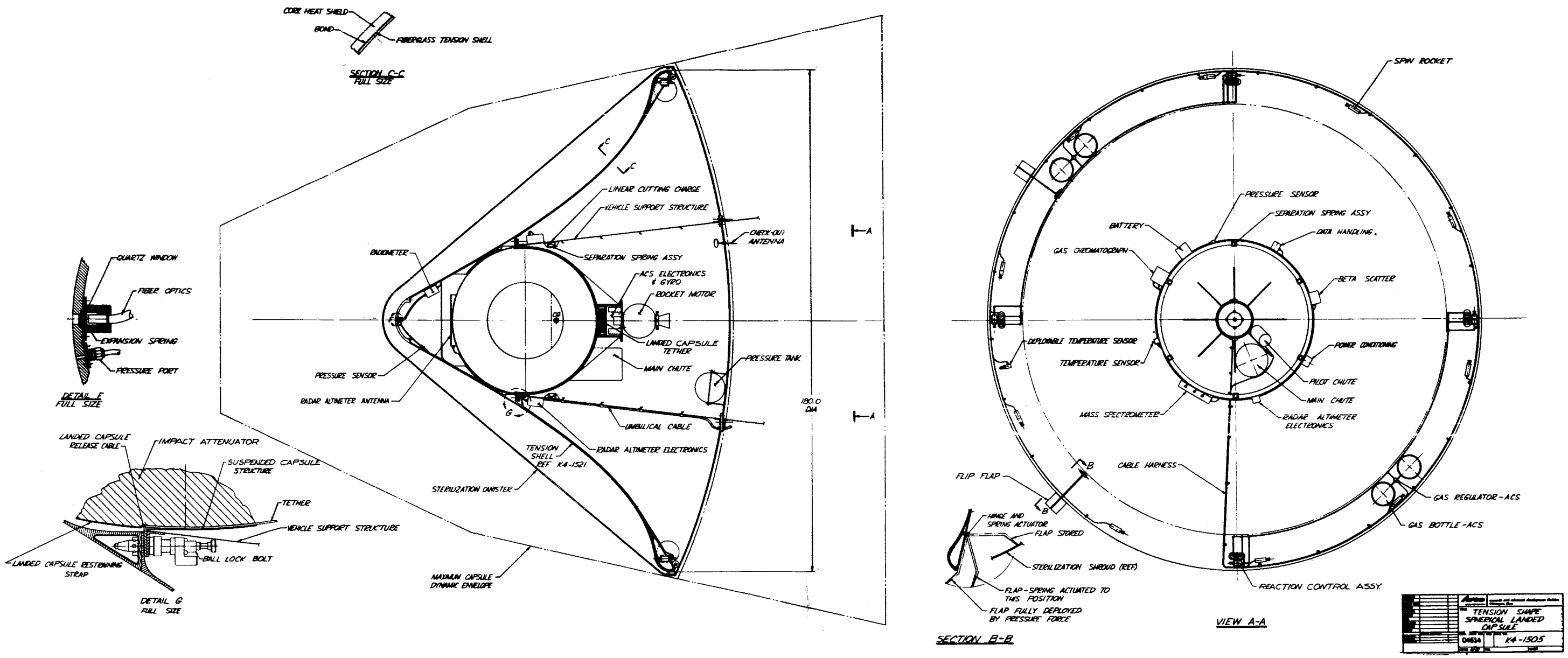


Figure 36 TENSION SHELL - FLOTATION SPHERE LANDED CAPSULE

5.2.2 Tension Shell

5.2.2.1 Design Description

The design presented in Figure 36 shows a blunted-nose ($0.1R_B$) tension shell (tangent angle of 30 degrees) with a flotation-sphere landed capsule. The landed capsule is located as far forward in the entry shell as possible and preloaded against the shell. Adjustment is provided at the interface ring to attain the proper preload. The flight capsule/flight spacecraft adapter is also attached at this same location (see detail G). The separated vehicle is separated from the flight spacecraft by first cutting the adapter with a linear-shaped charge, and then using a set of four coil springs to push it away. A short section of this adapter remains with the separated vehicle. This section is used to mount the external equipment.

Separation of the landed capsule from the entry shell is accomplished by releasing a set of ball-lock bolts, manifolded together for reliability at the peak load during parachute deployment (see detail G).

The entry shell for this design is made up of several major sections; a nosecap, a nose cone, a landed capsule mounting ring, an aft shell, and a compression ring. The nosecap is a beryllium heat sink similar to the other designs. A short nose cone sections runs from this nosecap to the landed capsule mounting ring. Two designs are shown for this section, a beryllium shell and a fiberglass shell. The beryllium is used in the weight statement of Table XXVI because of the significant weight savings.

The mounting ring for the landed capsule is designed for two loading conditions, that associated with the 1971 landed capsule and that associated with a future capsule of over 3000 pounds. Under the first condition, the ring only reacts the preload between the landed capsule and the bearing pad which is attached to the nose cone. All of the inertial load during entry is transmitted through the bearing pad. However, the total inertia weight of the future landed capsule is transmitted through the mounting ring since the landed capsule is of unknown character.

The portion of the entry shell aft of the landed capsule mounting ring is referred to as the aft shell and is the only portion of the entry shell that can be considered a tension shell. This portion of the shell is a tapered reinforced fiberglass-laminated monocoque structure which is integrally fastened to the mounting-ring flange and the compression ring. Near the compression ring, the aft shell is divided and wrapped around a honeycomb spacer and the main box section of the compression ring. Details of this section

TABLE XXVI

TENSION SHELL - WEIGHT SUMMARY

	Weight- (pounds)		c. g. (inches)*
Flight Capsule		2500.0	74.1
FC/FS adapter	100.0		
Sterile canister	366.9		
Elec. and mech. connectors	50.0		
Separated Vehicle		1983.1	72.6
ΔV propulsion	98.5		
ACS electronics	10.0		
Spin propellant	2.1		
Propulsion supports	22.5		
Entry Vehicle		1850.0	71.5
Entry shell heat shield	415.6		
Entry shell structure	359.0		
Thermal control	25.0		
ACS nozzles, tanks, etc.	69.3		
Spin motors and supports	10.0		
Elec. and mech. connectors	55.5		
Contingency	25.0		
Suspended Capsule		890.6	67.3
Instrumentation	40.4		
Telecommunications	20.6		
Power	37.5		
Miscellaneous	4.0		
Contingency (25 percent on above)	25.6		
Main chute pilot, etc.	71.0		
Structure	96.0		
Afterbody	25.0		
Landed Capsule (available)		570.5	52.0

*Note: Center of gravity from entry shell forward nose location

are shown in Figure 37, detail A. The compression ring design is composed of extruded beryllium angles, channels and flat sections riveted together to form a box-like section. Local beads are chem-milled into the flat sections to increase the local buckling strength. An alternate design is shown for detail A consisting of a square box section designed to more efficiently utilize the section in buckling. However, this increase in structural efficiency was gained at the cost of added spacer material and an increased depth of the sections which could impose difficult design and fabrication problems. The detail A design was used as shown in the main cross section. On this design, the ring is elongated, reducing the spacer thickness and weight. This ring is also used to mount all of the reaction control equipment as in the blunted cone design.

All remaining design features are described in detail in the blunted-cone and modified-Apollo designs.

5.2.2.2 Weight Summary

Presented in Table XXVI is a weight summary for the tension shell design. Here again, the only basic change in the weight summary from that of the blunted cone is the entry-shell weight (i.e., the primary structure and heat shield) which is given in more detail in Table XXVII. The heat shield weight is significantly increased over the other designs while the structural weight has decreased giving a total entry-shell weight approximately the same as the blunted cone. This is more evident in the available landed capsule weight which is approximately the same as for the blunted-cone design. This design was not pursued to the same depth as the blunted-cone design (except in structural analysis) and the design could change significantly if pursued further.

Table XXVII illustrates the wide variation in entry shell weight encountered when various alternate approaches are considered. The design shown is one of the lightest considered, at the possible expense of additional fabrication and development problems.

5.3 BASIS FOR SHELL SHAPE COMPARISON

The selection of an entry shape can be based on entry shell efficiency if a definite trend can be established to show an optimum shape for a specific purpose. However, the comparison is not clear cut and therefore other factors must be considered. Factors such as testing requirements, manufacturing and fabrication problems and systems compatibility must be investigated to perform a thorough shape - selection analysis. Careful consideration was given to all phases of the design, development, and manufacture. The following is a list of the major

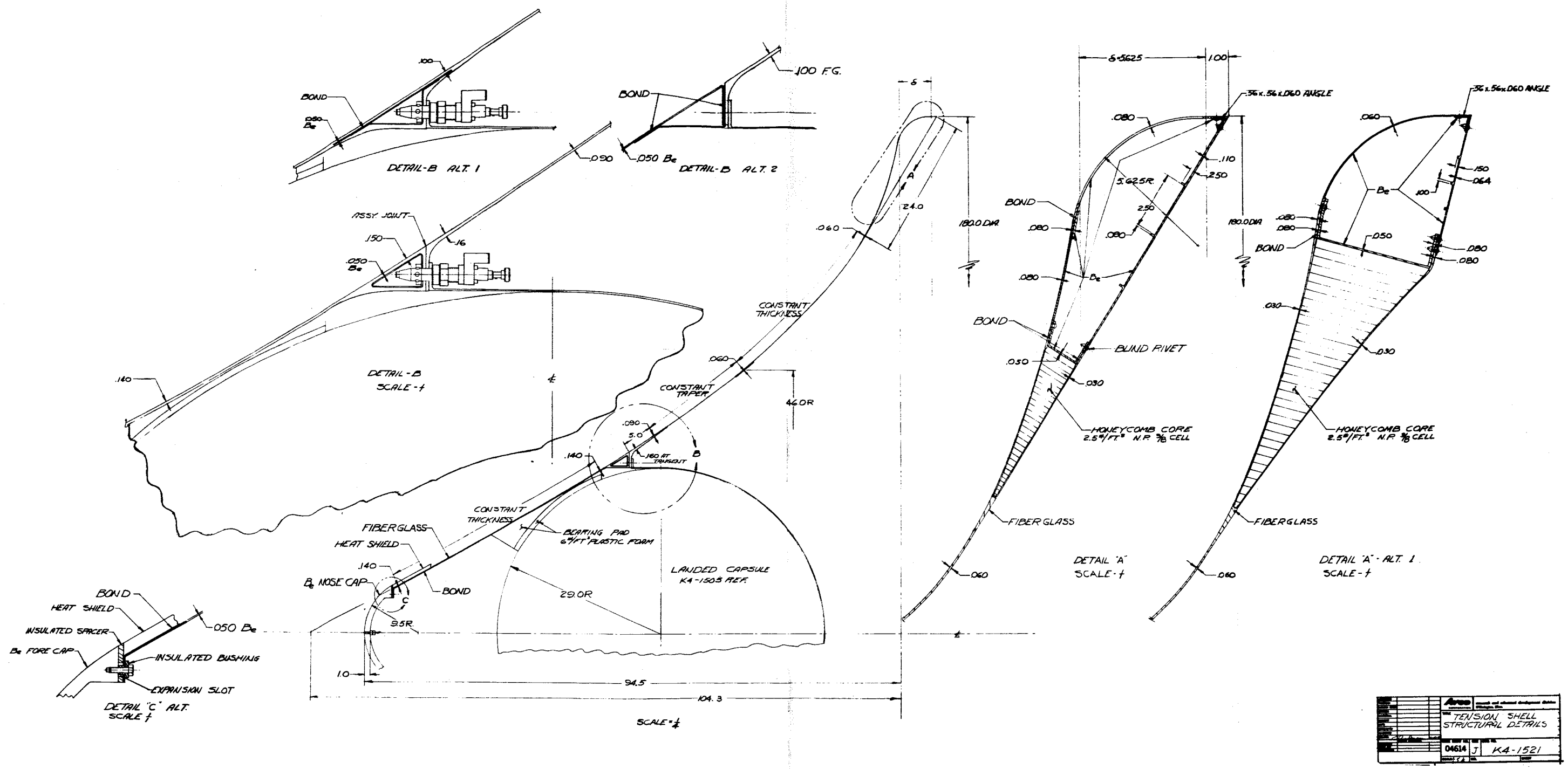


Figure 37 TENSION SHELL STRUCTURAL DETAILS

125-1

TABLE XXVII

TENSION SHELL STRUCTURAL WEIGHT BREAKDOWN

	REFERENCE*	ALT. RING*
Compression ring	161	149
Fiberglass aft shell	74	74
Beryllium nose cone	16	16
Beryllium nose cap	20	20
Mounting ring (detail "B" alt. 2)	21	21
Contingency (15 percent)	47	45
Bearing pad	20	20
total (pounds)	359	345
Nose section alternate 1*		
Fiberglass nose cone	45	45
Mounting ring (detail "B")	26	29
Contingency (15 percent)	52	50
total (pounds)	398	384
Nose section alternate 2*		
Fiberglass Nose Cone	45	45
Mounting ring (detail "B" alt. 1)	27	29
Contingency (15 percent)	52	50
total (pounds)	399	385

*Reference Figure 37

criteria that were used to establish the optimum shape for the conceptual design:

1. Residual weight (a measure of entry shell efficiency)
2. Aerodynamic stability
3. Center of gravity and packaging versatility
4. Design confidence
5. Test requirements
6. Manufacturing ease
7. Payload integration
8. External interfaces.

5.4 METHOD FOR DETERMINATION OF RESIDUAL WEIGHTS

The primary and most powerful method of selecting the optimum entry vehicle shell is that of determining the residual weight associated with each of the various configurations. In addition to determining residual weights, a weight analysis of the landed capsule was performed for comparative purposes.

During the ensuing discussion, the following definition of terms are used extensively. Figure 38 presents a pictorial representation of the various terms that are considered in the parametric synthesis of the Entry Vehicle.

The following is a further definition of the relationships used in the analytical synthesis of the vehicle and its subsystems:

W_E = Entry weight

$W_{H\&S}$ = Heat shield and structure weight (entry shell)

W_R = Residual weight = $W_E - W_{H\&S}$

W_{MC} = Main parachute weight

W_{ES} = External structure weight = $0.10\ W_R$

W_{LC} = Landed capsule weight = $W_R - W_{MC} - W_{ES}$

W_{CU} = Crush-up material weight

W_I = Internal weight = $W_{LC} - W_{CU}$

W_{IS} = Internal structure weight = $0.35\ W_{LC}$

W_{IP} = Internal payload weight = $W_I - W_{IS}$

The weight analysis of the Landed Capsule includes determination of the parachute, impact attenuator, and various support structure weights. While these weight studies do not directly relate to a comparison of entry shell configurations and concepts, they aid in evaluating the overall impact of the shell selection on system design.

The main parachute used in this system synthesis is of ring sail design which develops a supersonic drag coefficient of 0.70. It is designed to perform under the most severe atmospheric conditions that is, deployment in the Model 3 atmosphere, and descent and impact in the terminal descent atmosphere. It is designed to allow the landed capsule to impact with a vertical velocity of 80 ft/sec. Figure 39 indicates the weight and diameter of the main parachute as a function of suspended weight. Since test data is severly limited above this diameter, the largest diameter parachute utilized was 100 feet in diameter. For suspended weights greater than 1300 pounds, three-chute clusters were employed resulting in a weight penalty in excess of 30 pounds. An additional weight of 4 to 6 pounds, dependent on main parachute size, is required to account for the pilot parachute and deployment mortar to completely synthesize the descent system parametric weight.

For comparison purposes, two configurations of landed capsules were analyzed; the flotation sphere and the oblate spheroid. The crushup material used for the parametric analysis was foam-filled fiberglass honeycomb based upon the required total impact velocity of 130 ft/sec. In addition to the vertical impact velocity, a horizontal wind velocity of 100 ft/sec was assumed, giving a resultant velocity at impact of approximately 130 ft/sec. Figures 40 and 41 indicate the crush-up weight required to prevent the impact loads from exceeding 500 or 1000 g as a function of R_2. R_2 is defined as the principle radius of curvature of landed capsule. An internal packaging density of 2.0 slug/ft^3 was assumed for the flotation sphere, which represents a presently realizable density. However, due to the unique packaging requirements associated with the oblate spheroid, a packaging density of 1.0 slug/ft^3 was assumed for that configuration. The crushable material density is a variable along the curves; the optimum material density is assumed at each point.

In addition to the heat shield, primary structure, parachute, and crushup weights, a parametric approach to system synthesis must consider supporting structures to arrive at a realistic internal payload weight. The external support structure is defined as the brackets and attachments that are required to support the landed capsule, parachutes and other external equipment within the entry shell. From past experience, the external support structure weight has proven to be about 10 percent of the residual weight.

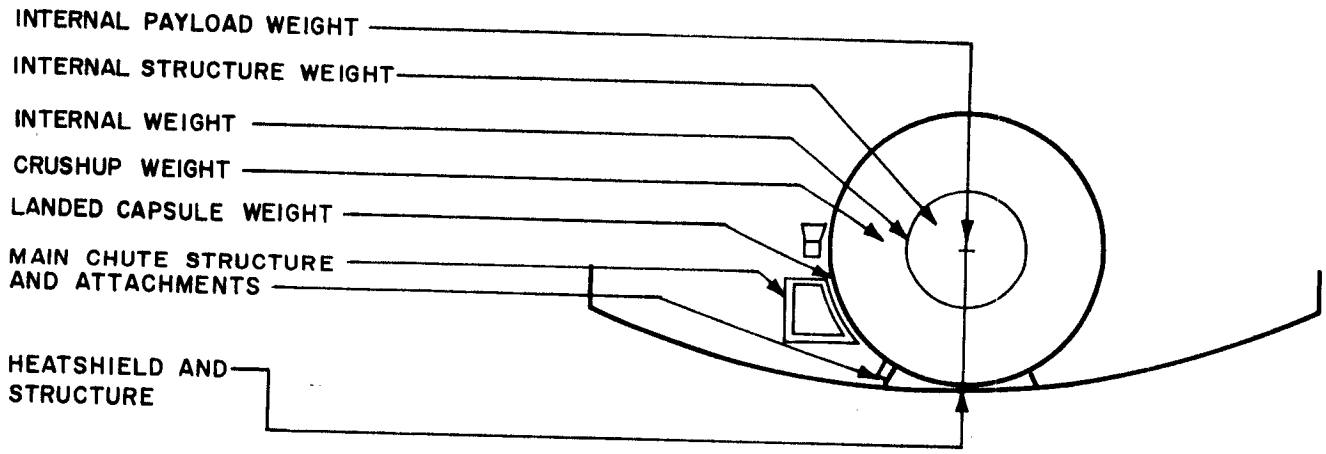


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Figure 38 PARAMETRIC WEIGHT TERMINOLOGY

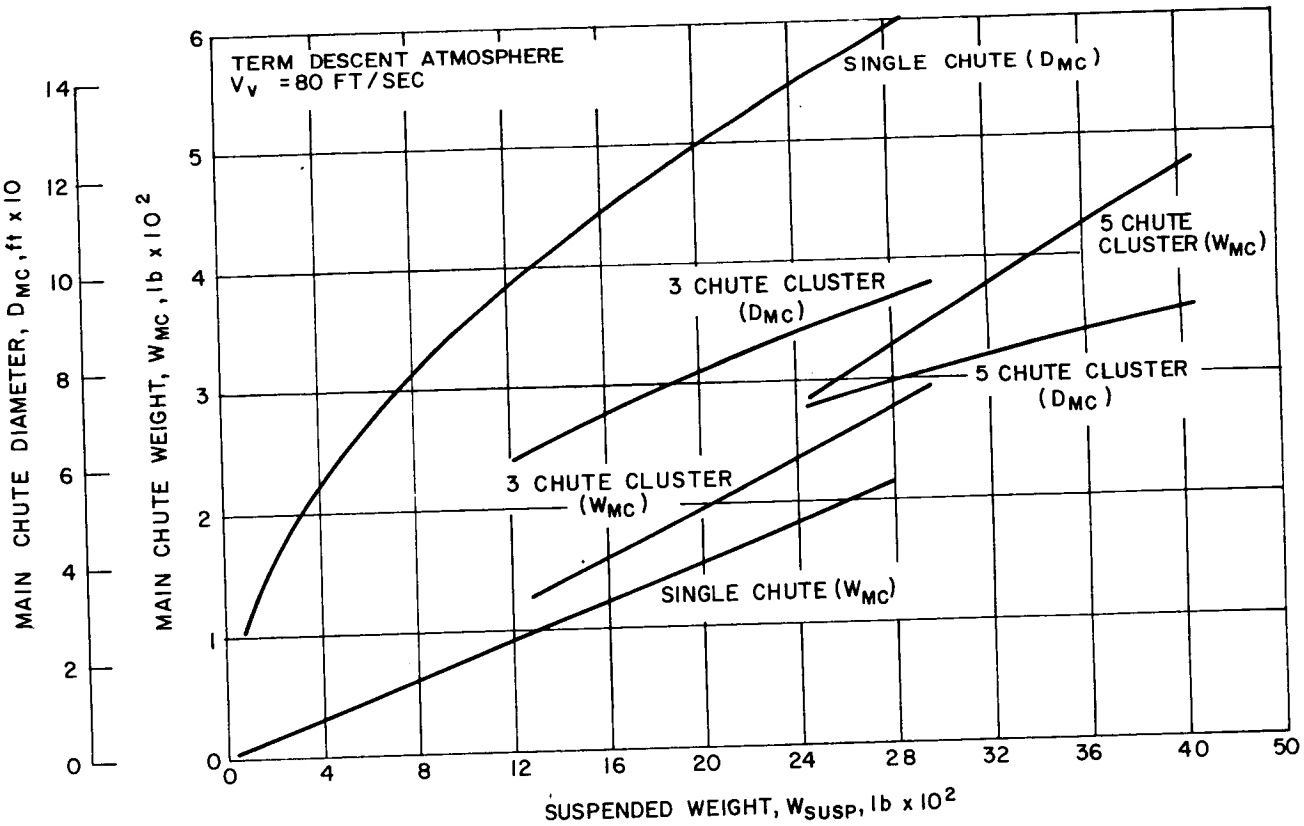

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Figure 39 PARAMETRIC WEIGHT ANALYSIS OF DESCENT SUBSYSTEM

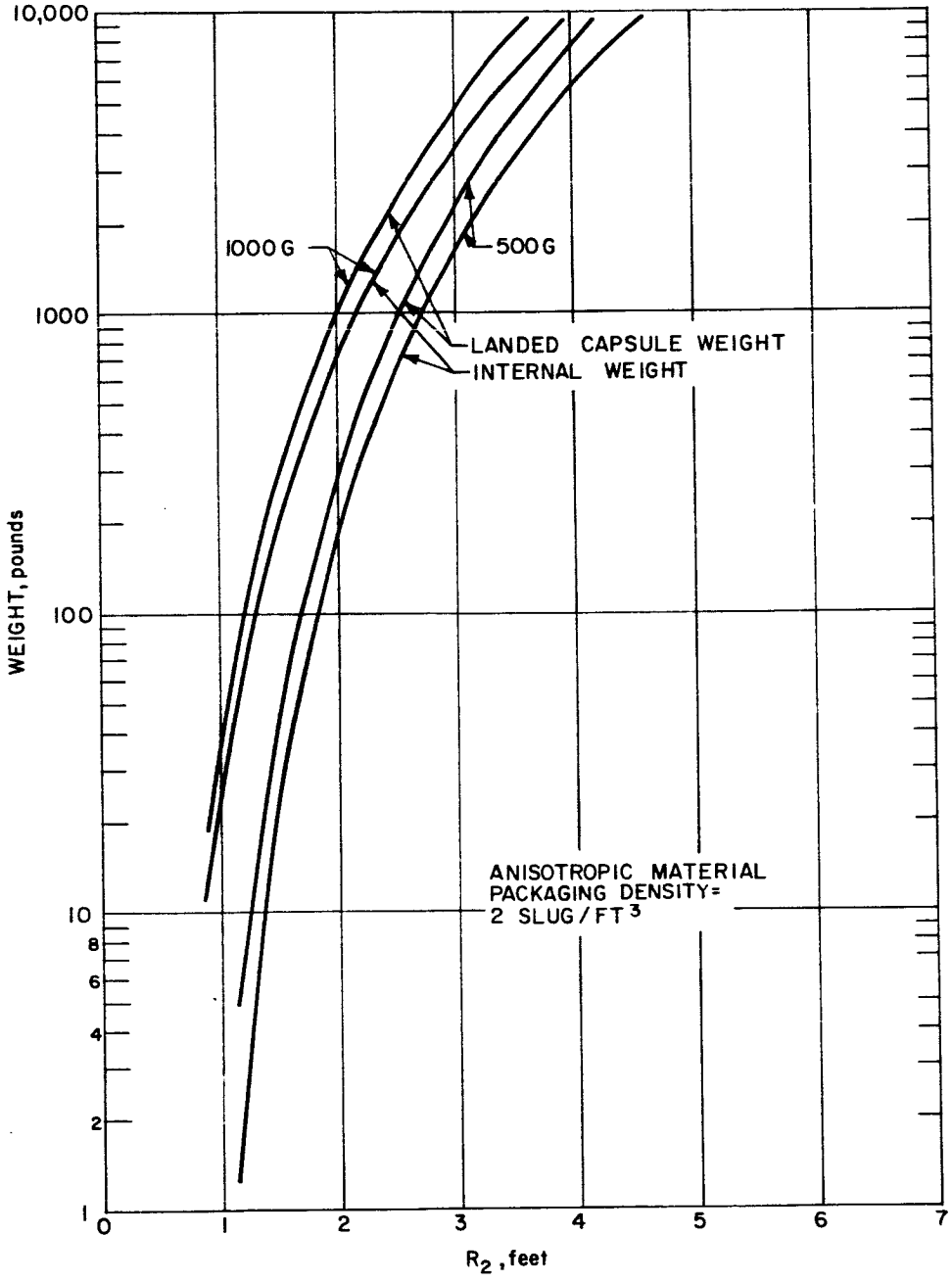

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Figure 40 PARAMETRIC WEIGHT ANALYSIS OF FLOTATION SPHERE LANDED CAPSULE

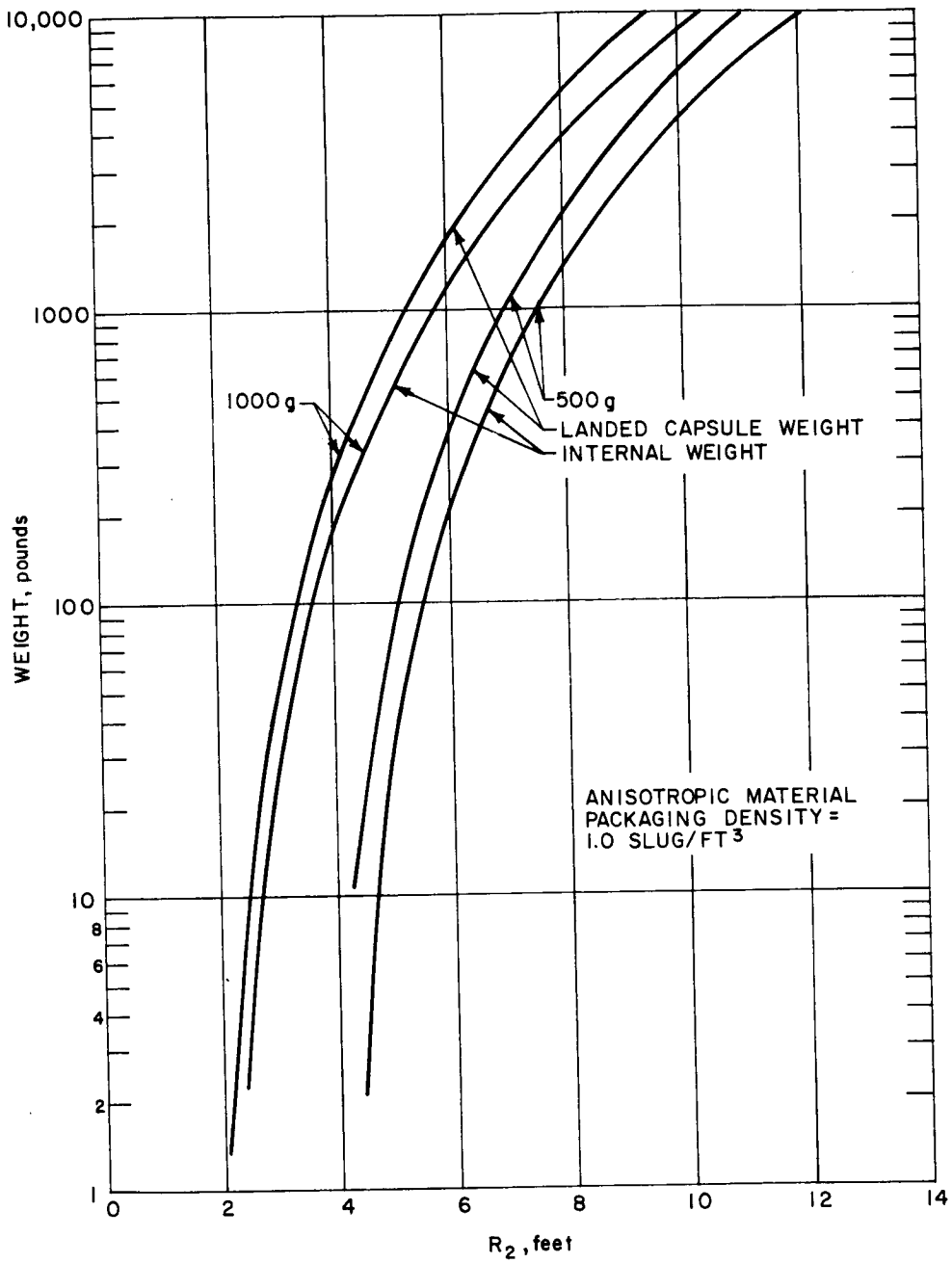

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Figure 41 PARAMETRIC WEIGHT ANALYSIS OF OBLATE SPHEROID LANDED CAPSULE

Another major weight fraction considered is the internal support structure necessary to support the internal payload within the landed capsule. Again, past experience and other studies have shown that this weight is approximately 35 percent of the landed capsule weight.

5.4.1 M/C_DA Determination

The study ground rules stipulated that the main parachute is to be fully opened subsonically at a minimum altitude of 15,000 feet. The parachute must be initially deployed at an altitude of 16,000 feet allowing 1000 feet of altitude for the parachute to completely open and decelerate the vehicle to subsonic speeds; a parachute deployment Mach number of 1.0 was selected as a maximum practical limit to insure subsonic velocities when the parachute was fully open. These altitude and Mach number contraints together with the model atmospheres considered in the study and the entry conditions specified define the entry vehicle hypersonic M/C_DA.

The atmospheres, Model 1 (40 millibar surface pressure), Model 2 (25 millibar surface pressure), and Model 3 (10 millibar surface pressure) are as specified in NASA TND-2525. The Model 3 atmosphere results in the lowest density profile throughout the entry altitude range and as such is the constraining atmosphere in the determination of the hypersonic M/C_DA.

For the 1971 opportunity an entry velocity range from 18,000 ft/sec to 23,800 ft/sec and an entry angle range from -20 degrees (near skip-out) to -90° must be accommodated.

Figure 42 presents M/C_DA as a function of model atmosphere and entry angle for an entry velocity of 23,800 ft/sec. This figure shows the maximum allowable M/C_DA to accommodate the entire range of entry velocities, entry angles and model atmospheres to be 0.15 slug/ft^2. This figure also illustrates as a function of diameter, the M/C_DA that corresponds to an entry weight of 4500 pounds; the maximum entry vehicle weight available for future missions.

It is possible to increase the M/C_DA by staging (reefing) the main parachute such that partial deployment occurs at some higher Mach number and altitude. Full parachute deployment (disreefing) would still take place at Mach 1.0 at 16,000 feet. Mach 1.3 was chosen as the reefed parachute deployment Mach number since experimental data indicates serious inflation and stability problems above this Mach number. The application of reefing generally enhances the over-all performance of the system, if the opening shock load is limited to a level which optimizes the required canopy and shroud line strength for both reefed deployment and disreefing. This requires matching the shock loads at each of the two events. There exists a maximum reefed area which matches the opening shock load and minimizes

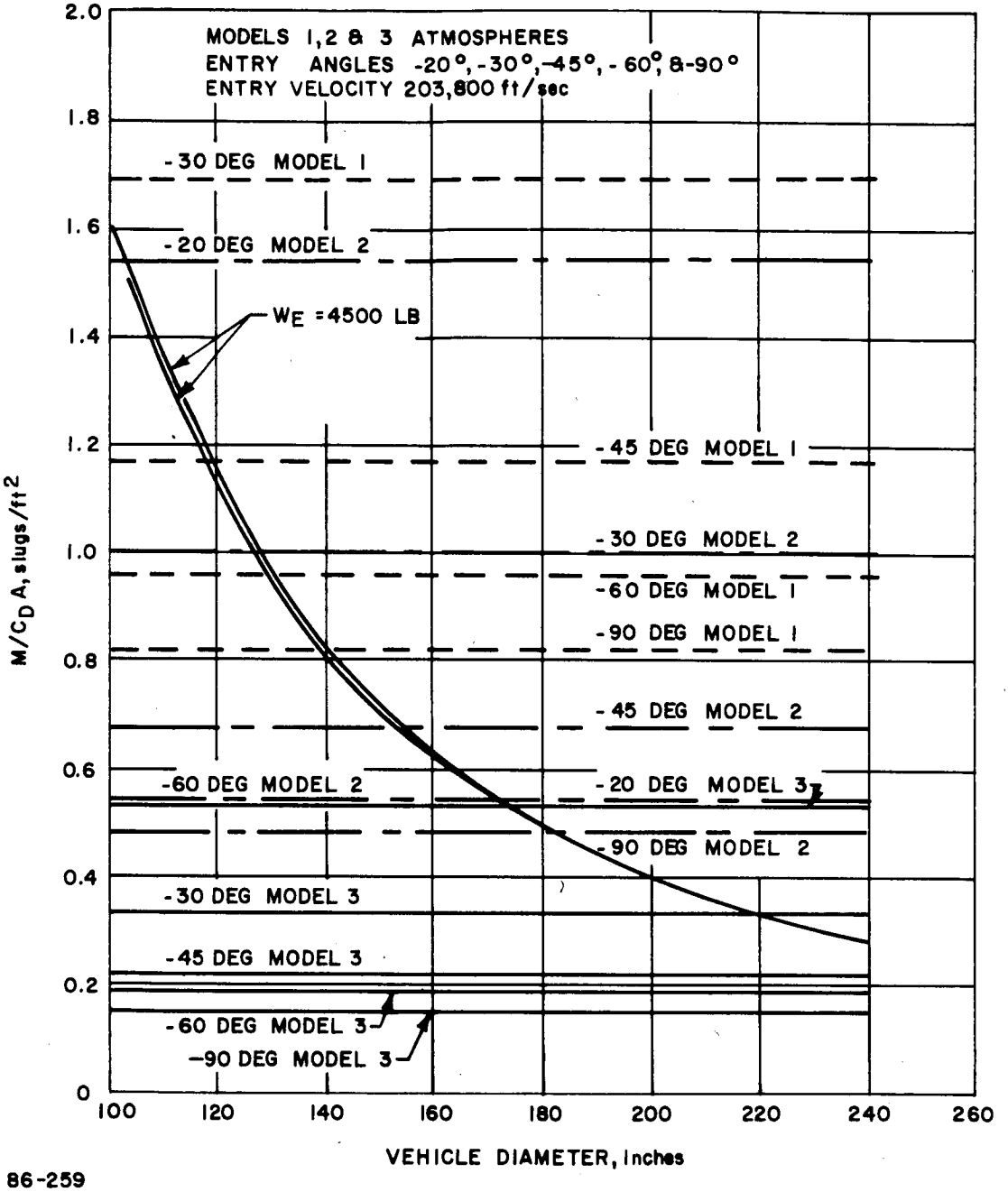


Figure 42 MAXIMUM M/C_DA ALLOWABLE VERSUS VEHICLE DIAMETER

the parachute system weight. Figure 43 shows the allowable M/C_DA as a function of reefed parachute drag area. Figure 44 indicates that the deployment and disreef shock loads are equal at 18 percent of the full parachute area. At this point it is possible to obtain an M/C_DA of 0.153 slug/ft^2. The above discussion and conclusions do not make any allowance for parachute actuation system dispersions and drag coefficient degradation due to angle of attack. Hence, the M/C_DA should be reduced slightly to 0.15 slug/ft^2 in order to allow for these effects. The remaining analysis utilizes an M/C_DA of 0.15 slug/ft^2, deploying an 18 percent reefed parachute at Mach 1.3 and disreefing it at 16,000 feet.

5.4.2 Heat Shield and Structure Design Criteria

Having determined the M/C_DA to be 0.15 slug/ft^2, the entry vehicle heat shield and structure design conditions can be established for each of the entry vehicle shell concepts. These design conditions are defined to allow heat shield and structure weights to be calculated as a function of diameter. The structure design condition is that combination of entry velocity, entry angle, model atmosphere, and M/C_DA (future missions may use a multi-mission shell at an M/C_DA higher than 0.15 slug/ft^2) which results in the highest value of stagnation pressure. For the multi-mission structure, this condition occurs in the Model 1 atmosphere at an entry velocity of 23,800 ft/sec, an entry angle of -90 degrees, and an M/C_DA of 0.82 slug/ft^2 (corresponding to the -90 degree Model 1 line shown in Figure 42. At a diameter of about 140 inches, an entry vehicle with an M/C_DA of 0.82 slug/ft^2 weights 4500 pounds; the future mission limit on entry weight. At diameters larger than 140 inches, the M/C_DA is 0.48 slug ft^2 (corresponding to the -90 degree Model 2 line shown in Figure 42).

The heat shield design condition is that combination of entry velocity, entry angle, model atmosphere, and M/C_DA which results in the highest value of total integrated heating. For the multi-mission shell, this condition occurs in the Model 1 atmosphere at an entry velocity of 23,800 ft/sec, an entry angle of -20 degrees and an M/C_DA corresponding to the 4500 pound weight limit, the value of M/C_DA to be varied as a function of diameter.

As previously discussed, several single mission entry shells were synthesized to determine the weight advantage associated with the multi-mission shell concept. The other combinations investigated were:

1. Multi-mission structure with 1971 heat shield
2. 1971 designed entry shell
3. Model 3 atmosphere designed entry shell

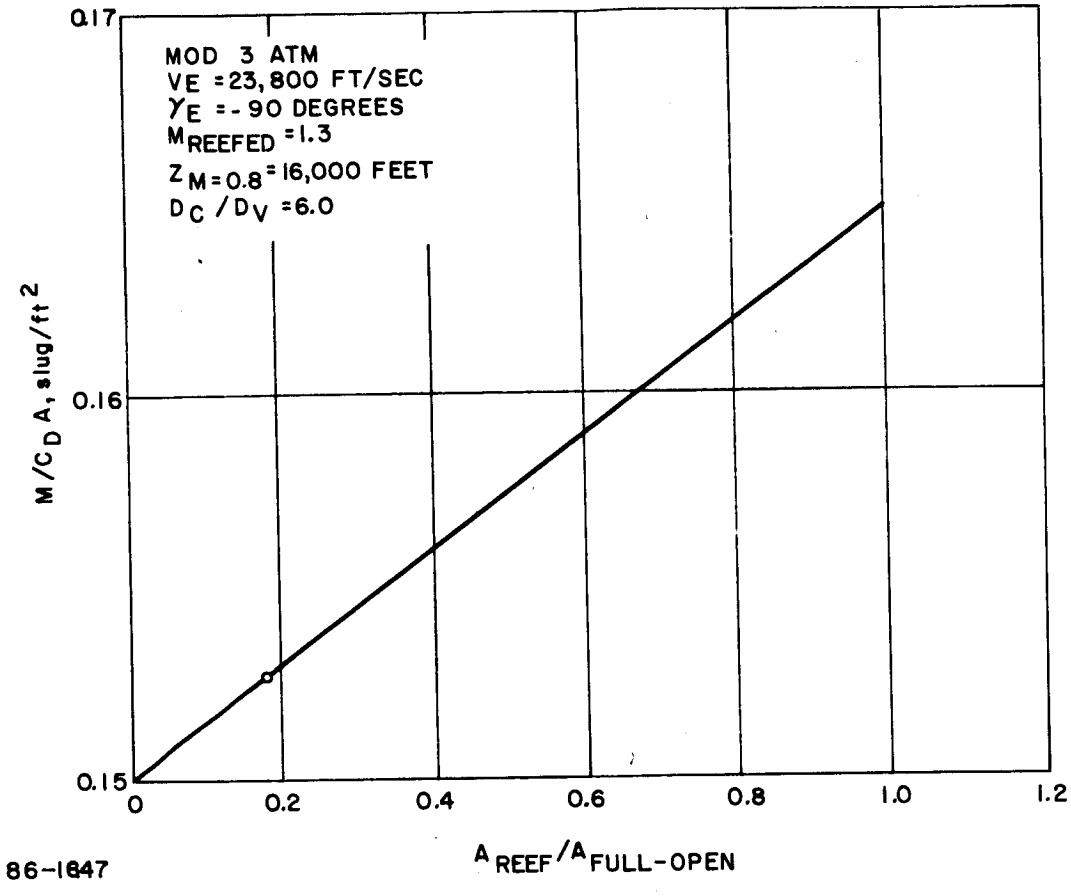
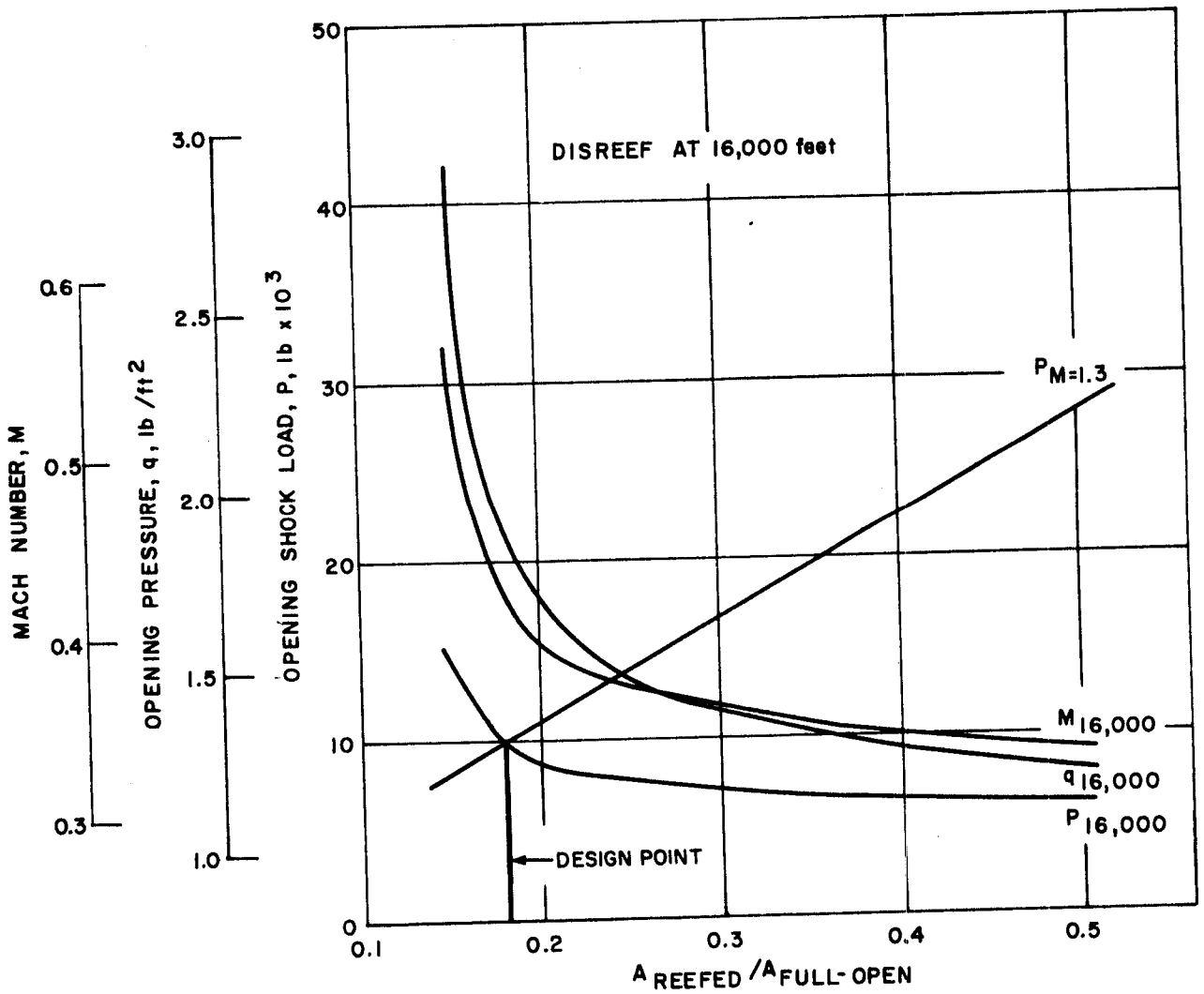


Figure 43 EFFECTS OF MAIN PARACHUTE REEFING ON M/C_DA



86-261

Figure 44 REEFED DEPLOYMENT AT MACH 1.3

4. Model 2 atmosphere designed entry shell

5. Model 1 atmosphere designed entry shell

The first two are to be used specifically for the 1971 opportunity and the remaining three are for future missions dependent upon final determination of the Martian atmosphere. Table XXVIII presents the design conditions for these five entry shells.

5.4.3 Heat Shield and Structure Weight Determination

Figures 45 and 46 show the resulting heat shield and structure weights as a function of diameter derived from the design criteria for the multi-mission shell concept. The information is presented for several candidate materials for both the heat shield and the structure. Only the modified Apollo and blunted cone configurations are included, since for the tension shell at these design conditions, the heat shield and structure weight required proved to be larger than the entry weight available. Various combinations of heat shield and structural materials yield a range of available residual weights for each shape.

5.4.4 Residual Weight -- Vehicle Diameter Comparison

The entry weight (W_E) is a direct function of the M/C_DA for a given entry shell configuration and for a given diameter. Using the M/C_DA required for a -90 degrees entry into the Model 3 atmosphere, 0.15 slug/ft^2, and the lightest heat shield and structural materials, the residual weight for each candidate entry shape is shown in Figure 47. This figure also shows the resulting weight for the landed capsule and internal weight determined as previously indicated. In order to maximize the weight available for the landed capsule for the 1971 mission, for which the entire range of atmospheric models must be considered, the entry vehicle diameter should be as large as possible.

However, the multi-mission shell design must consider future missions, at a time when the atmosphere may be better defined. If the atmosphere is determined to be Model 2, the resulting entry, residual, landed capsule, and internal weights are shown in Figure 48. Under these conditions, the optimum diameter is that diameter at which the entry weight is equal to the maximum allowable weight for future missions, 183 inches. Figure 49 shows the same weight profile for a future mission if Model 1 is the correct atmosphere. The optimum diameter now becomes 140 inches.

5.4.5 Entry Vehicle Diameter Selection

The establishment of the entry vehicle shell diameter is determined by tradeoff analysis of the constraints placed on it by the mission, launch

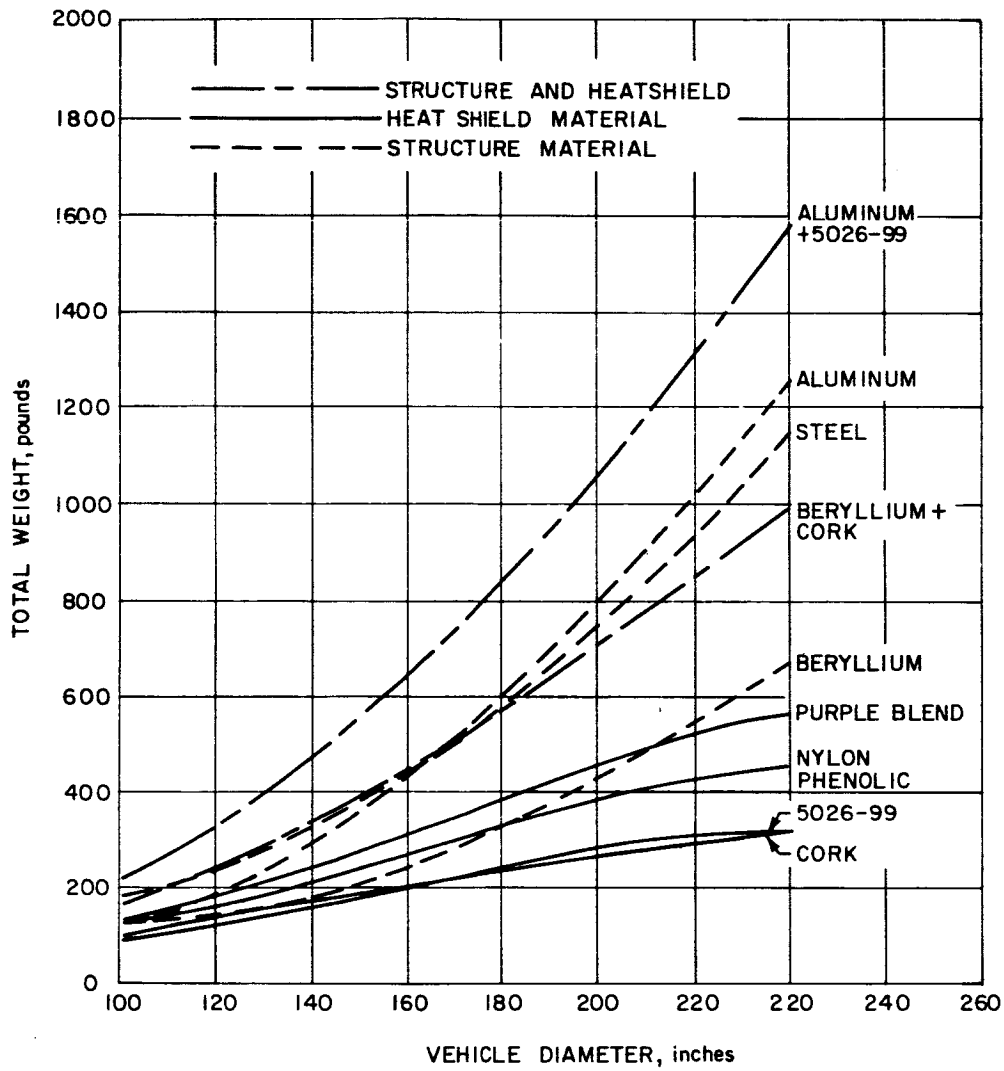


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Figure 45 HEAT SHIELD AND STRUCTURE WEIGHT VERSUS DIAMETER-MODIFIED APOLLO CONFIGURATION

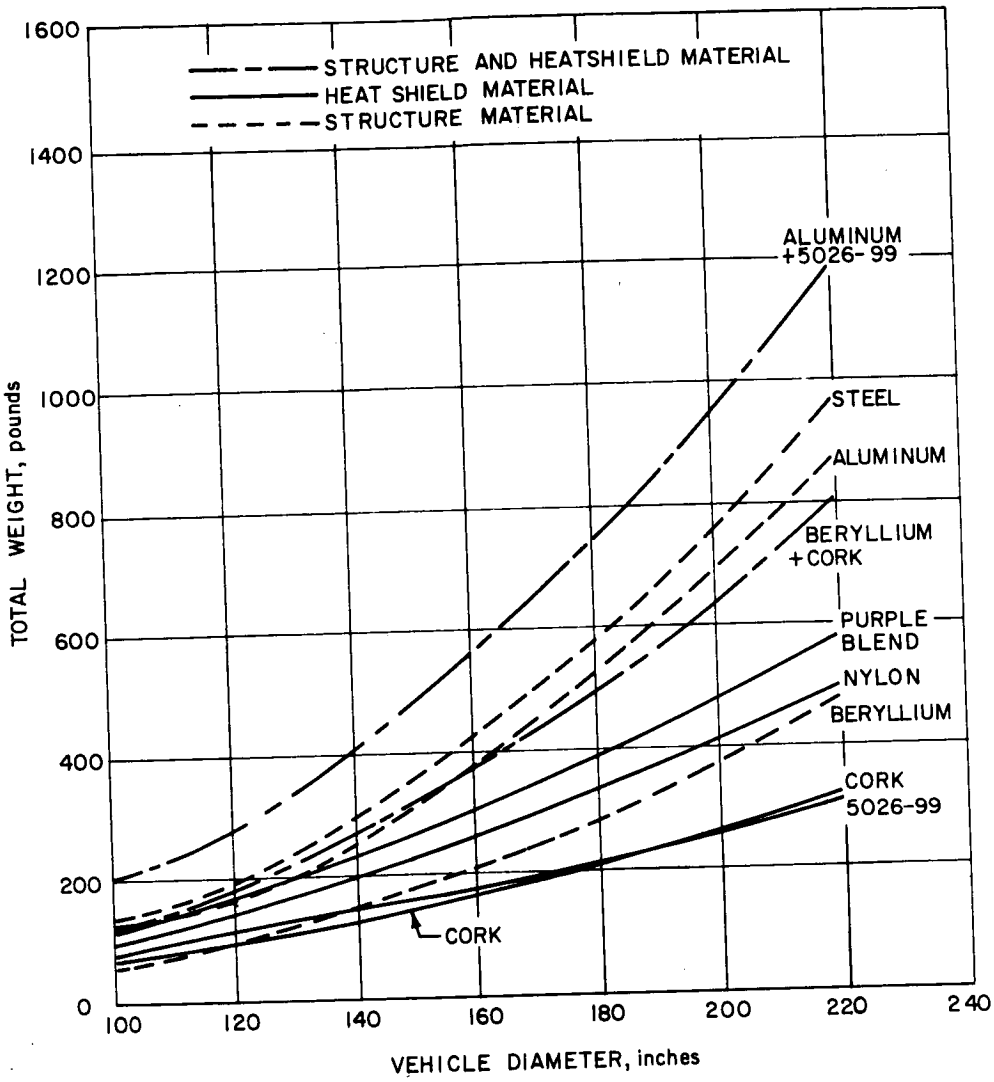

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Figure 46 HEATSHIELD AND STRUCTURE WEIGHT VERSUS DIAMETER-BLUNTED CONE

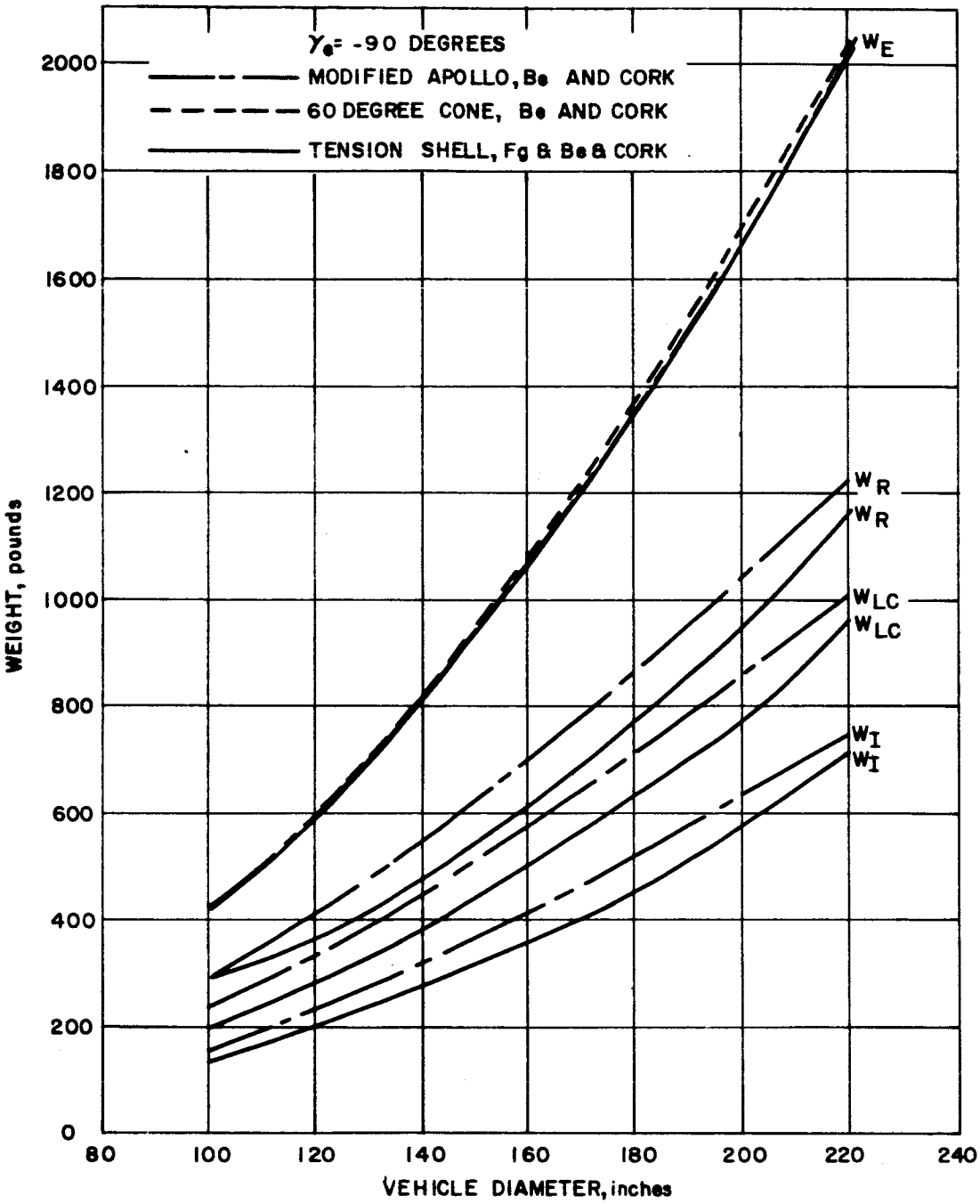

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Figure 47 RESIDUAL WEIGHT MODEL 3 ATMOSPHERE

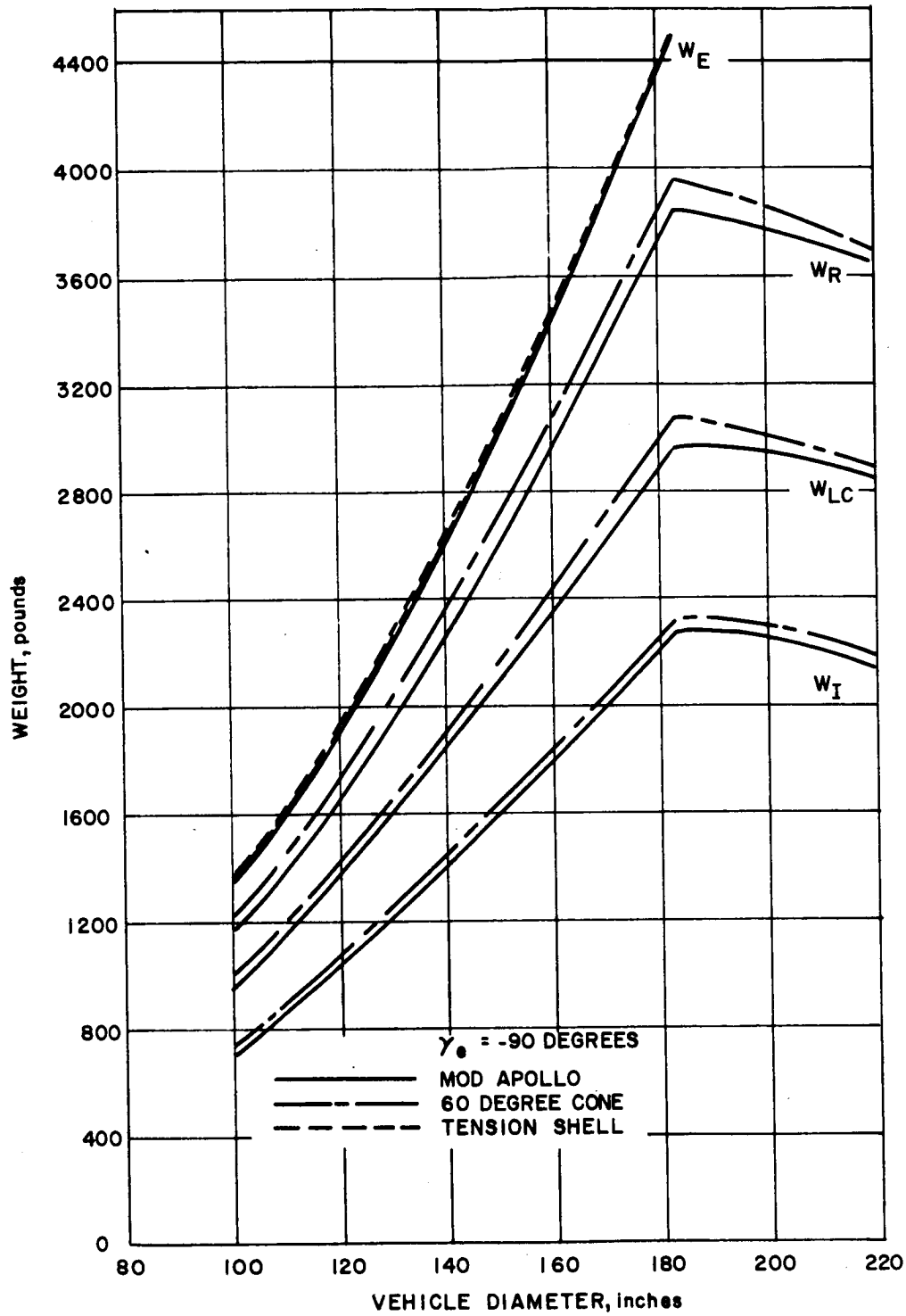

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Figure 48 RESIDUAL WEIGHT MODEL 2 ATMOSPHERE

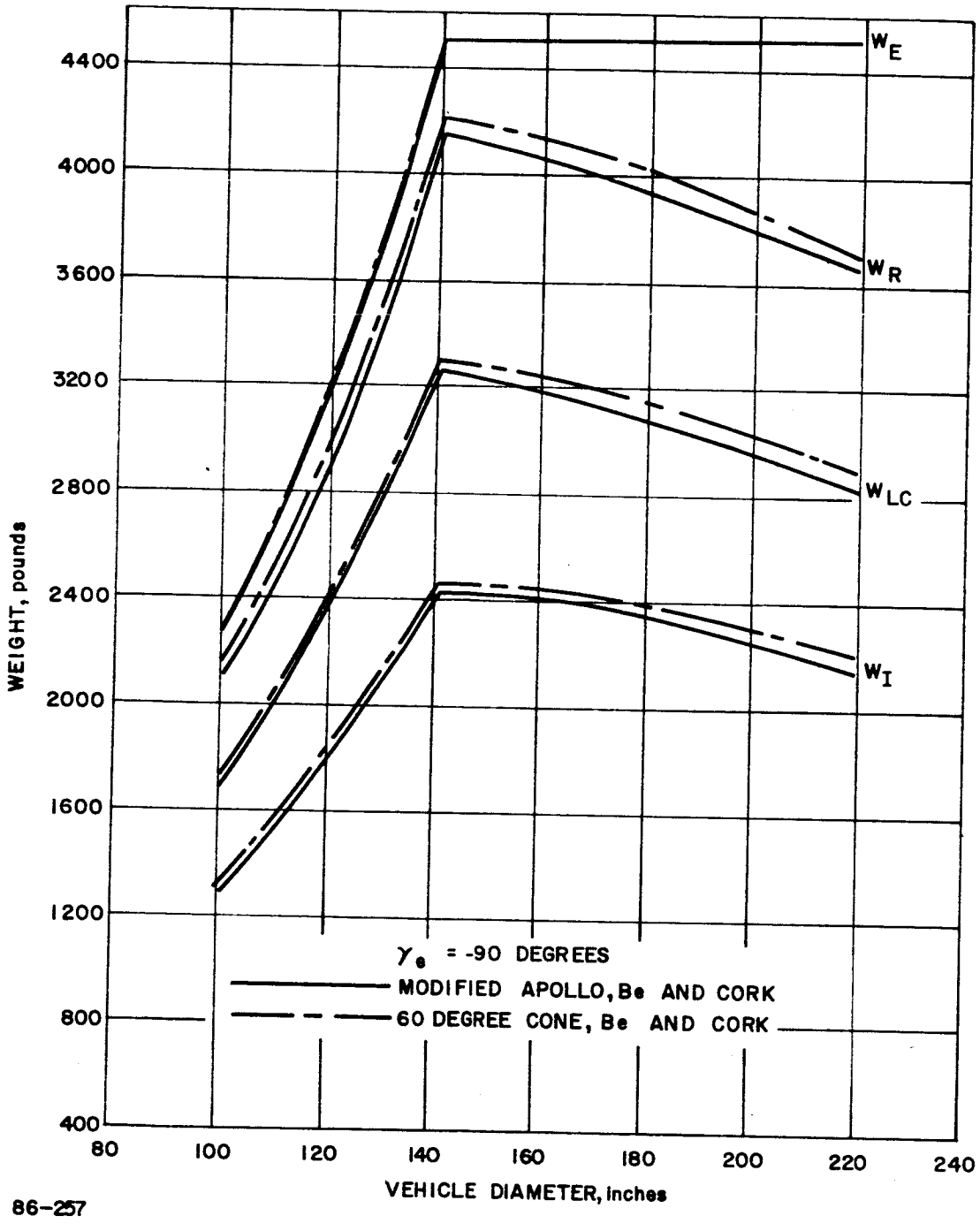

Figure 49 RESIDUAL WEIGHT MODEL 1 ATMOSPHERE

TABLE XXVIII

SINGLE MISSION SHELL DESIGN CRITERIA

Design Approach	Heat Shield	Structure
1. Multi-Mission Structure plus 1971 designed heat shield	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -20 degrees Model 1	Same as multi-mission design
2. 1971 designed shell	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -20 degrees Model 1	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -90 degrees Model 3
3. Model 3 shell	M/C_DA for γ_e = -20 degrees Model 3, evaluate at γ_e = -20 degrees Model 3	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -90 degrees Model 3
4. Model 2 shell	M/C_DA for γ_e = -20 degrees Model 2, evaluate at γ_e = -20 degrees Model 2	M/C_DA for γ_e = -90 degrees Model 2, evaluate at γ_e = -90 degree Model 2
5. Model 1 shell	M/C_DA for γ_e = -20 degrees Model 1, evaluate at γ_e = -20 degrees Model 1	M/C_DA for γ_e = -90 degrees Model 1, evaluate at γ_e = -90 degrees Model 1

vehicle, flight spacecraft and the desired landed capsule weight. To conform with the mission requirements specified previously, the ascent shroud for the Saturn IB-Centaur Launch Vehicle and the available flight capsule envelope shown in Figure 3 were employed to determine the maximum shell diameter that could be used for each of the three shapes. Practical clearances were assumed between the flight capsule and the sterilization canister; and between the sterilization canister and the launch vehicle ascent shroud. The candidate shapes were examined both with and without afterbodies. If an afterbody was not used, flaps were employed to reorient the entry vehicle to accommodate the failure mode of rearward entry. Figure 50 presents the three candidate entry shapes in the launch configuration. Each candidate shape has been packaged in the orientation which tends to maximize its diameter without an afterbody. The tension shell and blunted cone are in a nose up position, whereas the modified Apollo is nose down. The nose up orientation is preferred, since the flight capsule support structure need not pass through the primary heat shield and structure of the flight capsule. In addition, the capsule ΔV rocket is oriented to accelerate the capsule away from the spacecraft. Figure 51 illustrates the three candidate entry shapes packaged with afterbodies showing the maximum diameter for each case. The tension shell is shown nose up with the modified Apollo and blunted cone nose down. In each case, the diameter available with the alternate orientation is shown. If no afterbodies are employed, the largest common diameter which can be used to compare the candidate entry shapes is 189 inches. With the use of minimum afterbodies, as shown, the maximum common diameter is 180 inches.

The Saturn IB-Centaur payload capability is shown in Figure 4 as a function of the energy parameter C_3 (the hyperbolic excess velocity squared). Consideration of Figure 4 with the projected weights of the flight spacecraft and flight capsule over the launch opportunities from 1971 through 1975 indicates that a practical value of C_3 to be utilized for design purposes is 18 km^2/sec^2. This will provide a maximum weight capability of the planetary vehicle of 7500 pounds. Using a weight for the flight spacecraft including its propulsion system of 5000 pounds for the 1971 and 1973 orbiter missions will provide a total flight capsule weight allocation of 2500 pounds. This allocation includes the weight of entry vehicle as well as sterilization canister, flight capsule propulsion system and the support structure between the flight capsule and the spacecraft. In 1975, when the flight spacecraft is a fly-by vehicle, the propulsion requirement is reduced allowing the flight capsule weight allocation to be increased to 5200 pounds. To determine the allowable entry weight for the capsule as a function of diameter, the total capsule weight allocation must be reduced by the weight of its sterilization canister, propulsion system, and support structure. The weight of these items will vary as a function of capsule diameter. Therefore, the maximum allowable entry weight will decrease with increasing diameter. This relationship, for the 1971 mission, is shown in Figure 52 as is the entry weight associated with an M/C_DA of 0.15 $slug/ft^2$. From this figure, it can be seen that

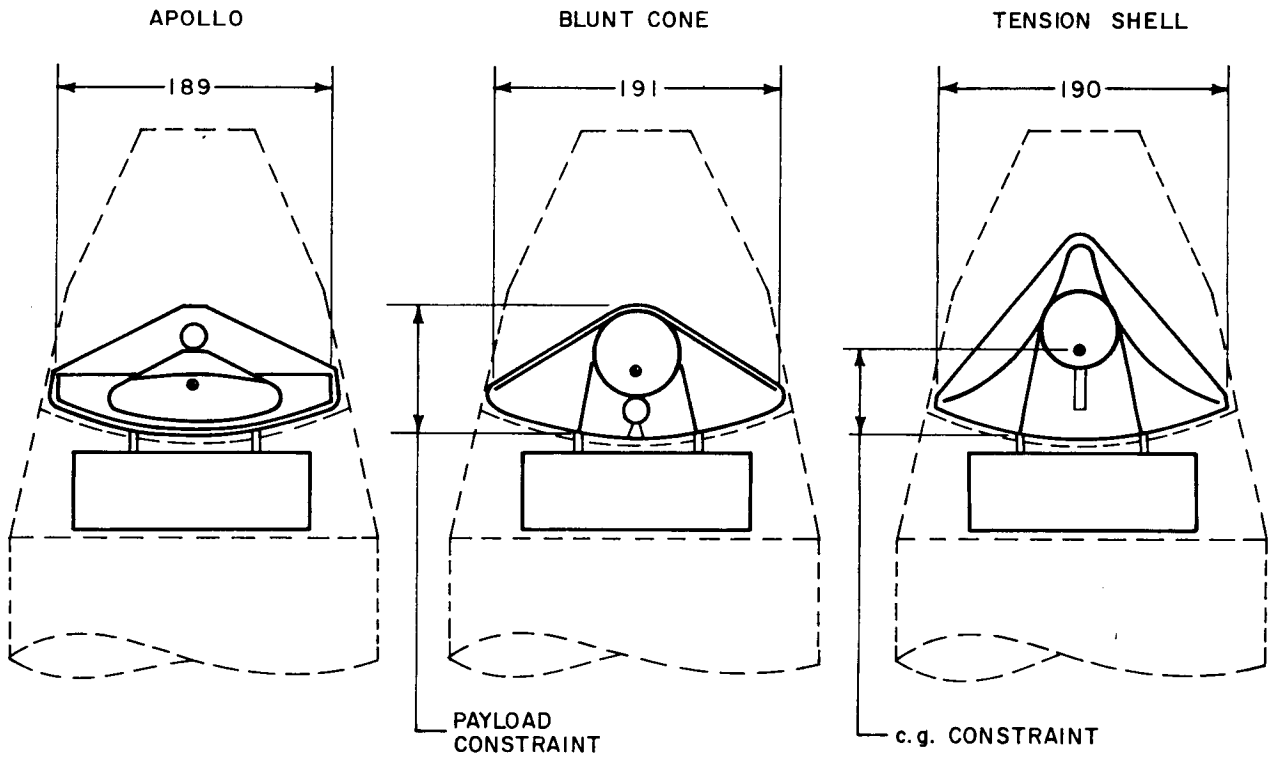

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Figure 50 SHELL - LAUNCH SHROUD CONSTRAINT

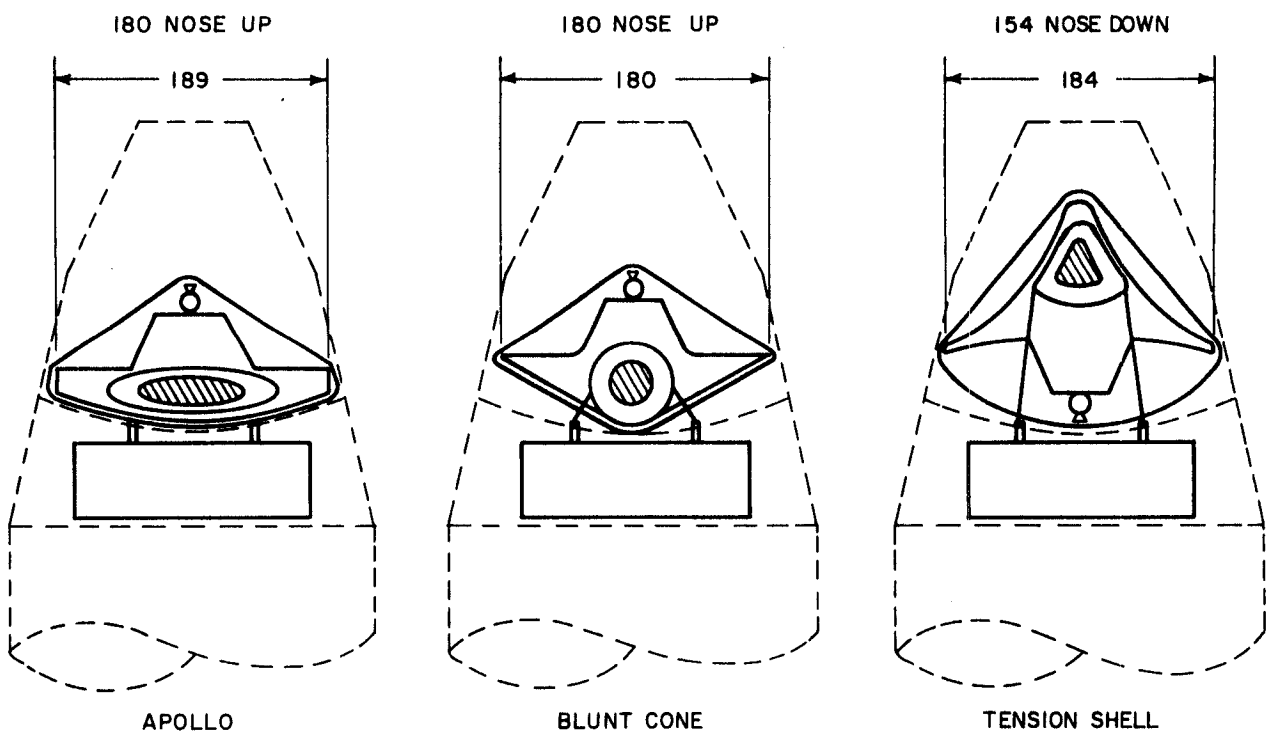

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Figure 51 SHELL - LAUNCH SHROUD CONSTRAINT WITH AFTERBODY

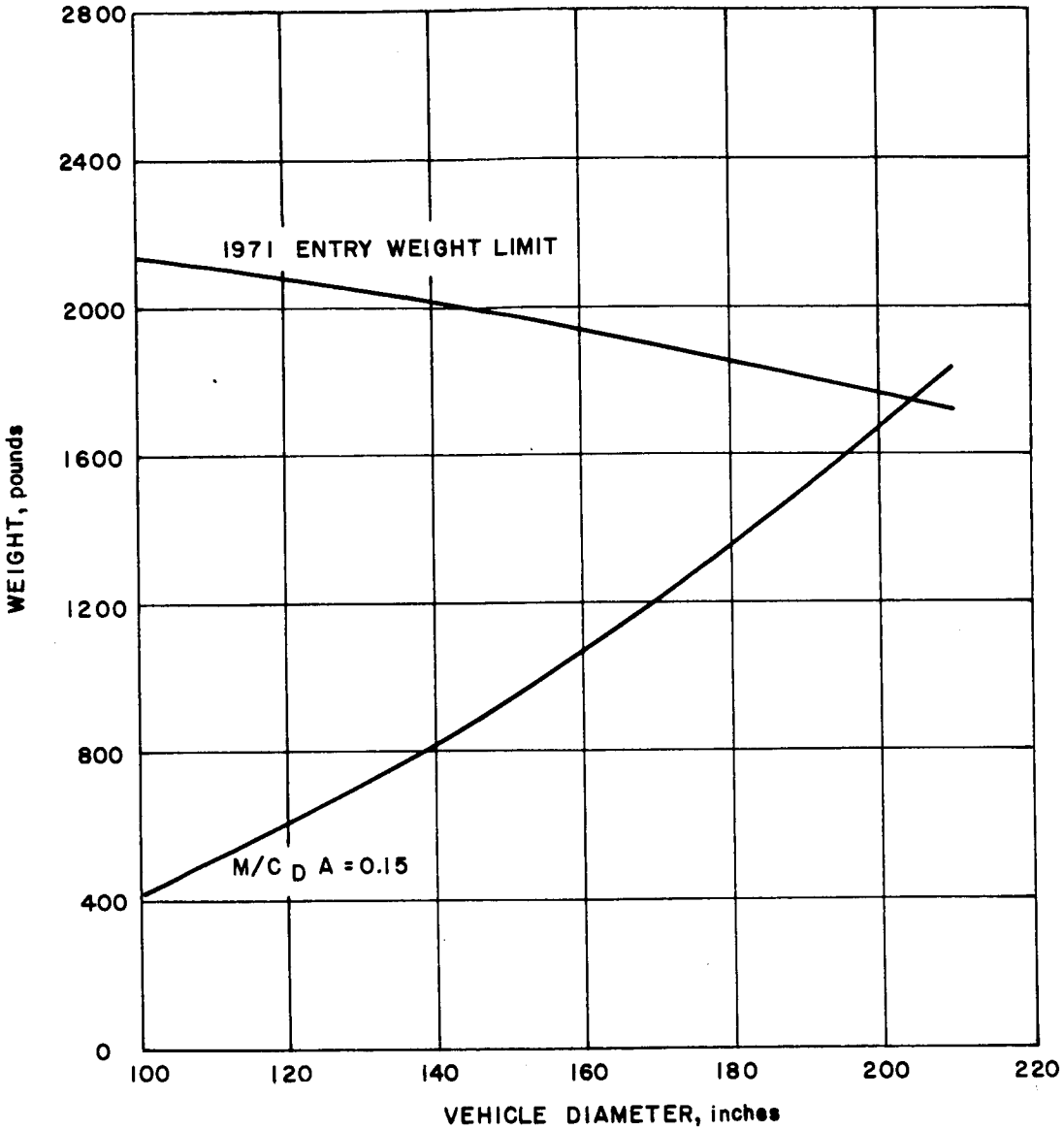

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Figure 52 ENTRY WEIGHT LIMITATION – 1971 MISSION WEIGHT VERSUS VEHICLE DIAMETER

the constraint on diameter imposed by the available launch weight, at 204 inches, is not limiting. Since the use of afterbodies rather than flaps to accommodate the rearward entry failure mode is preferred, the maximum common diameter which can be used for all three shapes is 180 inches.

The diameter selected for comparison of the entry vehicle shell concepts and configurations could be the maximum diameter, 180 inches, to maximize the future mission residual weight for the Model 2 and Model 3 atmospheres or 140 inches to maximize the future mission residual weight in the Model 1 atmosphere. A diameter selection of 180 inches maximizes the residual weight for the 1971 mission which must consider all three model atmospheres.

Table XXIX presents the weight penalties associated with each of these diameter selections. In the 1971 design and the future mission (Model 2 atmosphere) design, the comparison shows a severe penalty in residual weight for the 140-inch entry vehicle instead of the 180-inch entry vehicle. The penalties are less severe for the future missions (Model 1 atmosphere) design if the 180-inch vehicle is used. Additionally, if the latter incremental weight loss is compared to the total residual weight available in the denser atmosphere, the percentage loss is extremely small. The initial vehicle synthesis was, therefore, based on an 180-inch vehicle diameter.

TABLE XXIX

DIAMETER SELECTION

Mission	Diameter	Residual Weight	
		Modified Apollo	60-Degree Blunt Cone
1971 Mission	180	777	870
(All Model Atmospheres)	140	484	555
Future Mission	180	3730	3840
(Model 2 Atmosphere)	140	2260	2362
Future Mission	140	4140	4200
(Model 1 Atmosphere)	180	3930	4016

5.5 COMPARISON OF RESIDUAL WEIGHTS

5.5.1 Comparison at 197-inch Diameter

The initial comparison of the residual weights from the six entry vehicle shell concepts was made at a diameter of 197 inches before the final diameter selection was completed. The conclusion of this comparison; that the multi-mission shell concept imposes too large a weight penalty over single mission designs; is even more valid for the selected diameter of 180 inches. Table XXX presents the weight analysis developed for the multi-mission concept; the tension shell weights are shown for completeness. Tables XXXI through XXXV show the comparison of the multi-mission shell and each of the single mission shells in turn. For the 1971 mission, a residual weight penalty of 237 pounds for the modified Apollo and 197 pounds for the blunted cone results if the multi-mission shell is used rather than an entry shell specifically designed for the 1971 mission. The penalty for future missions is most severe if the atmosphere is determined to be Model 3 when the residual weight penalty associated with the use of multi-mission shell is 159 pounds for the modified Apollo and 196 pounds for the blunted cone.

On the basis of the penalty in residual weight associated with the multi-mission concept for this extremely weight limited design, the multi-mission structure-1971 heat shield design. This retains most of the desirable cost saving features of the multi-mission shell while elminating about one half of the weight penalty.

5.5.2 Comparison of Residual Weights for Three Shell Concepts

At this point in the study, the Mariner IV results were returned from Mars indicating the atmosphere to be far less dense than anticipated. The Model 1 atmosphere was eliminated from further consideration, reducing the structural penalty associated with the multi-mission structure - 1971 heat shield concept still further. The remainder of the study was concerned with the comparison of three entry shell concepts, the multi-mission structure - 1971 heat shield, the 1971 shell, and the future missions Model 3 atmosphere shell.

Each of the three entry shell concepts was re-evaluated for entry into the Model 2 and Model 3 atmospheres only. New design conditions were established for each shell concept as presented in Table XXXVI derived in the same manner as the original design conditions of Table XXVIII.

The analysis at 180-inch diameter also included the real gas effects of the Martian atmosphere upon the entry vehicle drag coefficients. Real gas effects increase the drag coefficient of the modified Apollo from 1.58 to

TABLE XXX

WEIGHT SUMMARY 1971 MISSION

MULTI-MISSION SHELL 197-INCH DIAMETER

Configuration	Modified Apollo		Blunted Cone		Tension Shell
Materials	Alum and 5026-99	Be and Cork	Alum and 5026-99	Be and Cork	Fiberglass Be and Cork
Entry Weight	1620	1620	1630	1630	1640
Heat Shield and Structure Weight	1029	692	915	610	>1640
Residual Weight	591	928	715	1020	0
Landed Capsule Weight	488	758	595	840	
Internal Weight	340	559	420	620	
Internal Payload Weight	221	363	274	403	

TABLE XXXI

WEIGHT SUMMARY 1971 MISSION

MULTI-MISSION STRUCTURE -- 1971 HEAT SHIELD

197-INCH DIAMETER

Configuration	Modified Apollo		Blunted Cone	
Materials	Be and Cork		Be and Cork	
Design	Multi-Mission	Design Condition	Multi-Mission	Design Condition
Entry Weight	1620	1620	1630	1630
Heat Shield and Structure Weight	692	572	610	499
Residual Weight	928	1048	1020	1131
Landed Capsule Weight	758	863	840	932
Internal Weight	559	640	620	700
Internal Payload Weight	363	416	403	455

TABLE XXXII

WEIGHT SUMMARY 1971 MISSION

HEAT SHIELD AND STRUCTURE

DESIGNED FOR 1971 MISSION -- 197-INCH DIAMETER

Configuration	Apollo		Blunted Cone	
Material	Be and Cork		Be and Cork	
Design	Multi-Mission	Design Condition	Multi-Mission	Design Condition
Entry Weight	1620	1620	1630	1630
Heat Shield and Structure Weight	692	455	610	413
Residual Weight	928	1165	1020	1217
Landed Capsule Weight	758	959	840	1002
Internal Weight	559	720	620	750
Internal Payload Weight	363	468	403	487

TABLE XXXIII

WEIGHT COMPARISON OF MODEL 3 DESIGNED SHELL

197-INCH DIAMETER

Configuration	Modified Apollo		Blunted Cone	
Material	Be and Cork		Be and Cork	
Design	Model 3	Multi-Mission	Model 3	Multi-Mission
Entry Weight	1620	1620	1630	1630
Heat Shield and Structure Weight	533	692	414	610
Residual Weight	1087	928	1216	1020

TABLE XXXIV

WEIGHT COMPARISON OF MODEL 2 DESIGNED SHELL

183-INCH DIAMETER

Configuration	Modified Apollo		Blunted Cone	
Material	Be and Cork		Be and Cork	
Design	Model 2	Multimission	Model 2	Multimission
Vehicle Weight	4500	4500	4500	4500
Heat Shield and Structure Weight	592	655	427	540
Residual Weight	3908	3845	4173	3960

TABLE XXXV

WEIGHT COMPARISON OF MODEL 1 DESIGNED SHELL

190-INCH DIAMETER

Configuration	Modified Apollo		Blunted Cone	
Material	Be and Cork		Be and Cork	
Design	Model 1	Multimission	Model 1	Multimission
Entry Weight	4500	4500	4500	4500
Heat Shield and Structure Weight	360	360	300	300
Residual Weight	4140	4140	4200	4200

TABLE XXXVI

DESIGN CONDITIONS FOR ENTRY SHELLS

180-INCH DIAMETER

Design Approach	Heat Shield	Structure
1. Multimission Structure plus 1971 Designed Heat Shield	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -20 degrees Model 2	M/C_DA for γ_e = -90 degrees Model 2, evaluate at γ_e = -90 degrees Model 2
2. 1971 Designed Shell	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -20 degrees Model 2	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -90 degrees Model 3
3. Model 3 Shell	M/C_DA for γ_e = -90 degrees, evaluate at γ_e = -20 degrees Model 3	M/C_DA for γ_e = -90 degrees Model 3, evaluate at γ_e = -90 degrees Model 3

TABLE XXXVII

SHELL SYNTHESIS -- CONCEPT COMPARISON

	Blunted Cone (pounds)				Modified Apollo (pounds)			
	Structure	Heat Shield		Total	Structure	Heat Shield		Total
	Be/SS	Cork			Be/SS	Cork		
Multimission Structure 1971 Heat Shield	451.2	290.0		741.2	524.3	249.5		773.8
		pri	sec			pri	sec	
		240.0	50.0			197.0	52.5	
1971 Shell	423.1	290.0		713.1	501.3	249.5		750.8
		pri	sec			pri	sec	
		240.0	50.0			197.0	52.5	
Model 3 Shell	423.1	314.7		737.8	501.3	274.4		775.7
		pri	sec			pri	sec	
		264.7	50.0			221.9	52.5	

TABLE XXXVIII

SHELL SYNTHESIS - MULTI-MISSION STRUCTURE-1971 HEAT SHIELD

Blunted Cone (pounds)				Modified Apollo (pounds)				Tension Shell (pounds)			
Structure Be/SS	Heat Shield Cork		Total (pounds)	Structure Be/SS	Heat Shield Cork		Total (pounds)	Structure Fg/Be	Heat Shield Cork		Total (pounds)
451.2	290.0		741.2	524.3	249.5		773.8	359.0	415.6		774.6
	primary	secondary			primary	secondary			primary	secondary	
	240.0	50.0			197.0	52.5			340.7	74.9	

to 1.70, the blunted cone from 1.59 to 1.63 the tension shell from 1.60 to 1.70. The modified Apollo configuration was further modified by eliminating the cylindrical skirt at the outer diameter. The weight summary shown in Table XXXVII illustrates the resulting heat shield, structure and total entry shell weights for each of the entry shell shapes for the multi-mission structure - 1971 heat shield entry shell concept. The tension shell was also included in this analysis to determine the effect of eliminating the Model 1 atmosphere upon the tension shell weights. For the blunted cone and modified Apollo shapes beryllium face sheets with a stainless steel core were considered for the honeycomb structural material. The tension shell structure was fiberglass with a beryllium closeout ring. The heat shield material is subdivided into the primary heat shield on the outside of the entry shell, and the secondary heat shield on the backface of the entry-shell skirt to protect the structure in the event of rearward entry. All three entry shells are approximately the same total weight.

Design analyses were pursued for the modified Apollo and blunted cone entry shells for the other entry shell concepts. Table XXXVIII shows the entry shell weights which result. Work on the tension shell for the 1971 entry shell and the model 3 atmosphere entry shell was not complete when the entry from approach trajectory study phase was terminated. The structural weights shown in Tables XXXVII and XXXVIII include the landed capsule support structure mounting ring and capsule bearing pad. These structural weights represent the portion of the entry shell which is jettisoned at parachute deployment and are not strictly comparable with the weight shown in the previous parametric analysis.

Table XXXIX shows the residual weight which results from the use of the multi-mission structure - 1971 heat shield entry shells at an M/C_DA of 0.15 slug/ft^2. The effect of the higher drag coefficient of the modified Apollo (C_D = 1.70) as opposed to the blunted cone (C_D = 1.63) can be noted.

TABLE XXXIX

SHELL SYNTHESIS -- RESIDUAL WEIGHTS
MULTI-MISSION STRUCTURE 1971 HEAT SHIELD
M/C_DA= 0.15 slug/ft^2

Blunted Cone			Modified Apollo		
Entry Weight	Heat Shield and Structure Weight	Residual Weight	Entry Weight	Heat Shield and Structure Weight	Residual Weight
1391	741.2	649.8	1450	773.8	676.2

5.6 COMPARISON OF SHELL SHAPES FOR OTHER SELECTION CRITERIA

Table XL presents the shape selection criteria to be considered as a function of vehicle configuration. The ratings indicated are on a relative basis and are used for comparison between these shapes.

TABLE XL

MARS FLIGHT CAPSULE SHELL COMPARISON

Factor	Modified Apollo	Cone	Tension Shell
Residual weight ($W_E - W_{H/S + S}$)	Good	Best	Poor
Stability	Probably good	Probably best	Fair
c. g. and packaging versatility	Best	Good	Poor
Design confidence	Good	Best	Poor
Test requirements	High	High	Highest
Manufacturing ease	Good	Best	Fair to poor
Payload integration and access	Best	Good	Fair
External interfaces	Good	Good	Poor

5.6.1 Residual Weight

The term residual weight applies to the weight that is remaining after the weight of the entry shell (heat shield and structure) has been subtracted from the entry weight. For a given set of parachute deployment conditions, a specific entry M/C_DA is required, independent of the entry vehicle shape selected. If only one entry vehicle diameter is considered, the entry weight is dependent upon only the hypersonic drag coefficient, which for the vehicles of interest varies only slightly (C_d = 1.63 for blunted cone and 1.70 for modified Apollo and tension shell). Therefore, considering these constraints and specifying the entry conditions for particular missions, the residual weight is considered to be a measure of the efficiency of the entry shape and shell to perform its function.

The blunted cone proves to be the most efficient configuration requiring the least weight for the entry shell under equal conditions of entry velocity and angle, model atmosphere and entry M/C_DA.

5.6.2 Aerodynamic Stability

Based on the primary mode of entry, a slowly spinning vehicle, the blunted cone, tends to provide the best dynamic stability characteristics. However, as indicated in Table XL, precise analytical data is limited in this area; the blunted cone and modified Apollo appear to be quite similar.

5.6.3 Center of Gravity and Packaging Versatility

From a center of gravity and packaging point of view, the modified Apollo is the best configuration. The allowable center of gravity margin for the modified Apollo shape is the largest, allowing a wider variety of payload packages. When afterbodies are incorporated into the designs, the size of the afterbody could restrict the payload package size, negating the advantage of the modified Apollo configuration over that of the blunted cone.

5.6.4 Design Confidence

Design confidence, the amount of previous experience which exists in the design and fabrication of a particular shape, places the blunted cone in a preferred position. This is true primarily due to the vast experience that exists in Earth reentry vehicle design and fabrication for this shape.

5.6.5 Test Requirements

In the area of test requirements, all shapes will require an extensive development program with the tension shell requiring the most extensive testing. This is principally due to the fact that little test data is available for this shape during entry. Testing of other system elements such as parachutes, is common to all configurations.

5.6.6 Manufacturing Ease

The best configuration as far as ease of manufacture is concerned is again the blunted cone chiefly because of simplicity of the geometric shape.

5.6.7 Payload Integration

The modified Apollo shape possesses the greatest degree of accessibility and thus provides ease in payload integration and assembly of components into the entry shell. The other shapes also require relatively complex payload support structures.

5.6.8 External Interfaces

The final consideration is that of external interfaces between the flight capsule and the flight spacecraft. This is a function of the packaging arrangement within the launch vehicle ascent shroud and the position of the spacecraft relative to the capsule. When one considers all configurations in a nose up position mounted on top of the spacecraft, the modified Apollo and blunted cone present approximately the same degree of complexity as to interface design.

In conclusion, both the blunted cone and the modified Apollo appear to be of about equal capability as a Mars entry shell configuration. The blunted cone was selected for the sample design on the basis of its slightly more efficient entry shell and on the basis of the extensive experience with this shape in Earth re-entry vehicle design.

The multi-mission structure - 1971 heat shield entry shell concept was selected over the 1971 shell since the weight penalty is nominal, 28 pounds, and the potential savings in development for future missions could be large in both money and development time which can be concentrated instead on the more complex features of the future flight capsules.

5.7 AVAILABLE LANDED CAPSULE WEIGHT

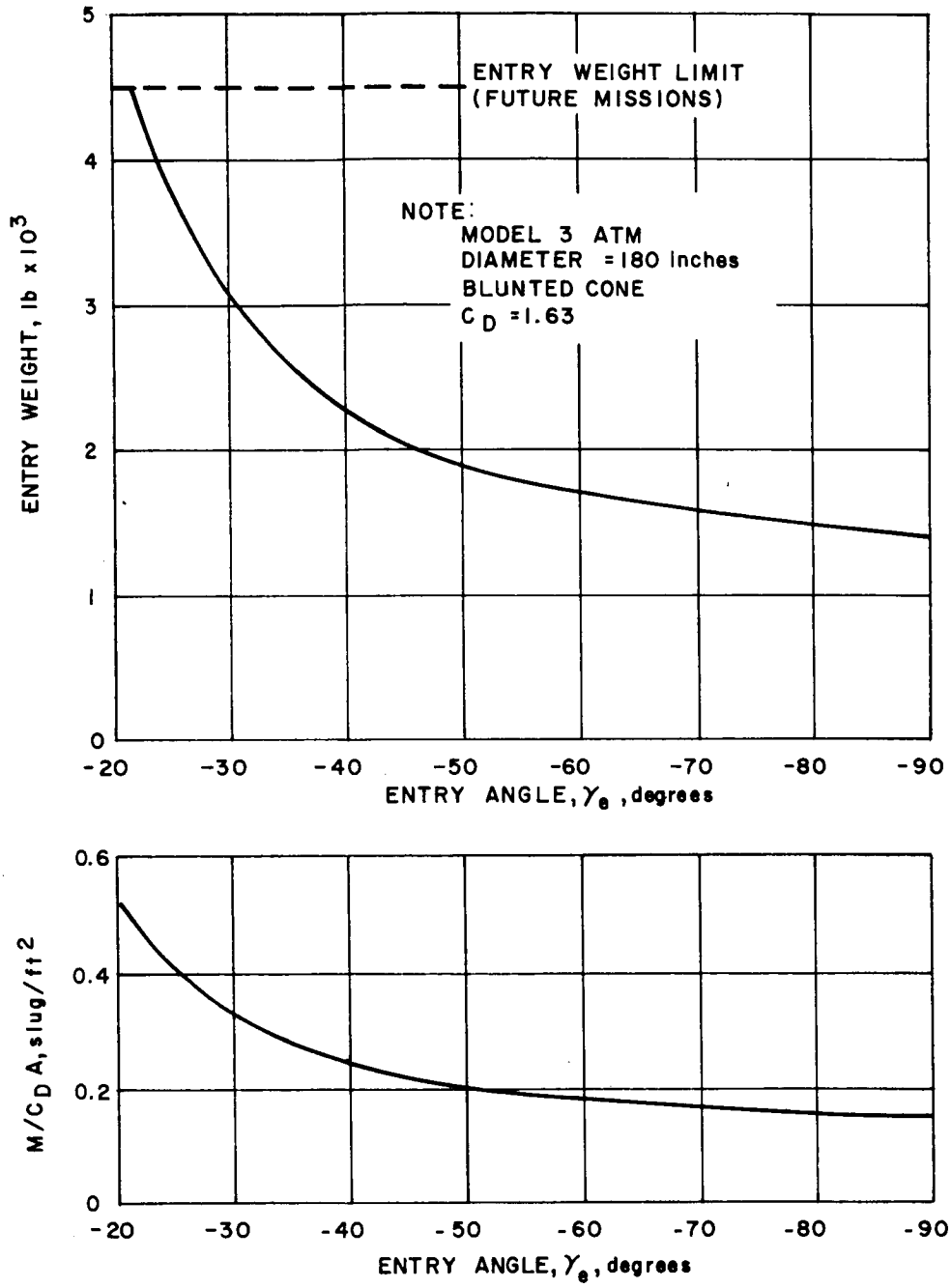
Using the blunted cone entry shell weight at an M/C_DA of 0.15 slug/ft^2, the available landed capsule weight is shown in Table XLI. Preliminary analysis of the required weight of the landed capsule shows that 162 pounds is clearly inadequate and must be increased to provide a reasonable scientific payload for the 1971 misssion. As shown in Figure 52, the maximum entry weight available for the 1971 opportunity at a vehicle diameter of 180 inches is 1850 pounds constrained by the total flight capsule weight limit for the 1971 mission. The entry weight can be increased by increasing the entry vehicle M/C_DA and limiting the allowable entry angle while maintaining the main parachute deployment altitude and Mach number. The allowable entry weight versus entry angle is shown in Figure 53; for an entry weight of 1850 pounds (M/C_DA of 0.2 slug/ft^2) the maximum allowable entry angle is approximately -52 degrees.

TABLE XLI

WEIGHT SUMMARY - BLUNTED CONE

($M/C_DA = 0.20$ slug /ft^2)

Entry weight	1391 pounds
Heat shield and structure weight	741 pounds
Residual weight	650 pounds
Suspended weight	465 pounds
Available landed capsule weight	162 pounds



86-262

Figure 53 ENTRY WEIGHT VERSUS ENTRY ANGLE

Table XLII shows the residual weight for the modified Apollo and the blunted cone when the M/C_DA has increased to the point where the flight capsule total weight allocation becomes limiting. As can be seen, the difference in drag coefficient becomes unimportant. In this case, the lighter heat shield and structural weight of the blunted cone provide a margin in residual weight of 33 pounds over the modified Apollo

TABLE XLII

SHELL SYNTHESIS -- RESIDUAL
MULTI-MISSION STRUCTURE 1971 HEAT SHIELD

Blunted Cone			Modified Apollo		
Entry Weight	Heat Shield and Structure Weight	Residual Weight	Entry Weight	Heat Shield and Structure Weight	Residual Weight
1850	741.2	1108.8	1850	773.8	1076.2

An entry weight consistent with the maximum allowable for the 1971 opportunity was selected. To achieve this entry weight (1850 pounds) the entry angle was restricted to a nominal value of -44 degrees. The weight summary for blunted cone configuration is presented in Table XLIII.

TABLE XLIII

WEIGHT SUMMARY - BLUNTED CONE

($M/C_DA = 0.20$ slug/ft^2)

Entry weight	1850 pounds
Heat shield and structure weight	741 pounds
Residual weight	1109 pounds
Suspended weight	924 pounds
Landed capsule weight	581 pounds

6.0 LANDED CAPSULE TRADEOFFS

6.1 INTRODUCTION

This section presents the major elements of design synthesis for the landed capsule. The synthesis primarily includes the selection of a communication subsystem, power subsystem, and scientific instrumentation. The synthesis is presented for the entry and descent and for the surface mission phases. Several landed capsule configurations are presented, the most applicable of which are examined to determine the impact of capsule configuration on the various subsystem designs. The resulting designs are then compared to select the capsule configuration for conceptual design.

An incompatibility exists between the residual weight necessary to accommodate the desired instrumentation payload and the communications and power equipment to support it, and the residual weight available for this equipment after consideration of the entry shell design. Various approaches can be taken to provide a landed capsule design within available residual weight constraints.

6.2 LANDED CAPSULE CONFIGURATION COMPARISON

6.2.1 Review of Landed Capsule Concepts

In the initial design study the primary effort was devoted to selection of a landed capsule concept (that portion of the flight capsule ultimately landing on the planet). Since the selection of the landed capsule concept is greatly influenced by the mission objectives and more particularly the entry configuration, it was felt that substantial effort must be placed on its design generation. Presented in Figure 54 are some typical landed concepts that were developed in previous studies[5]. Several of these concepts have some particular design features that warrant discussion.

The first concept "sphere-in-sphere" possesses several unique properties. One of the most significant is that it does not depend on the planet terrain for antenna erection and science deployment, i.e., the erection process is within the outer sphere, and since the inner sphere has no physical connection to the outer sphere (except by a flotation fluid) it is capable of erecting by gravity force regardless of the external sphere orientation. Other features are, low g-level impact, compact science and electronic packaging characteristics, low-impact attenuator weight and the compatibility with most entry shell configurations. These features are somewhat over-shadowed by the design complexity the concept would have under the overall mission environments. Other systems presented in Figure 54 that

[5]Conceptual Design Studies of an Advanced Mariner Spacecraft, Avco/RAD TR-64-36, III. (28 October 1964)

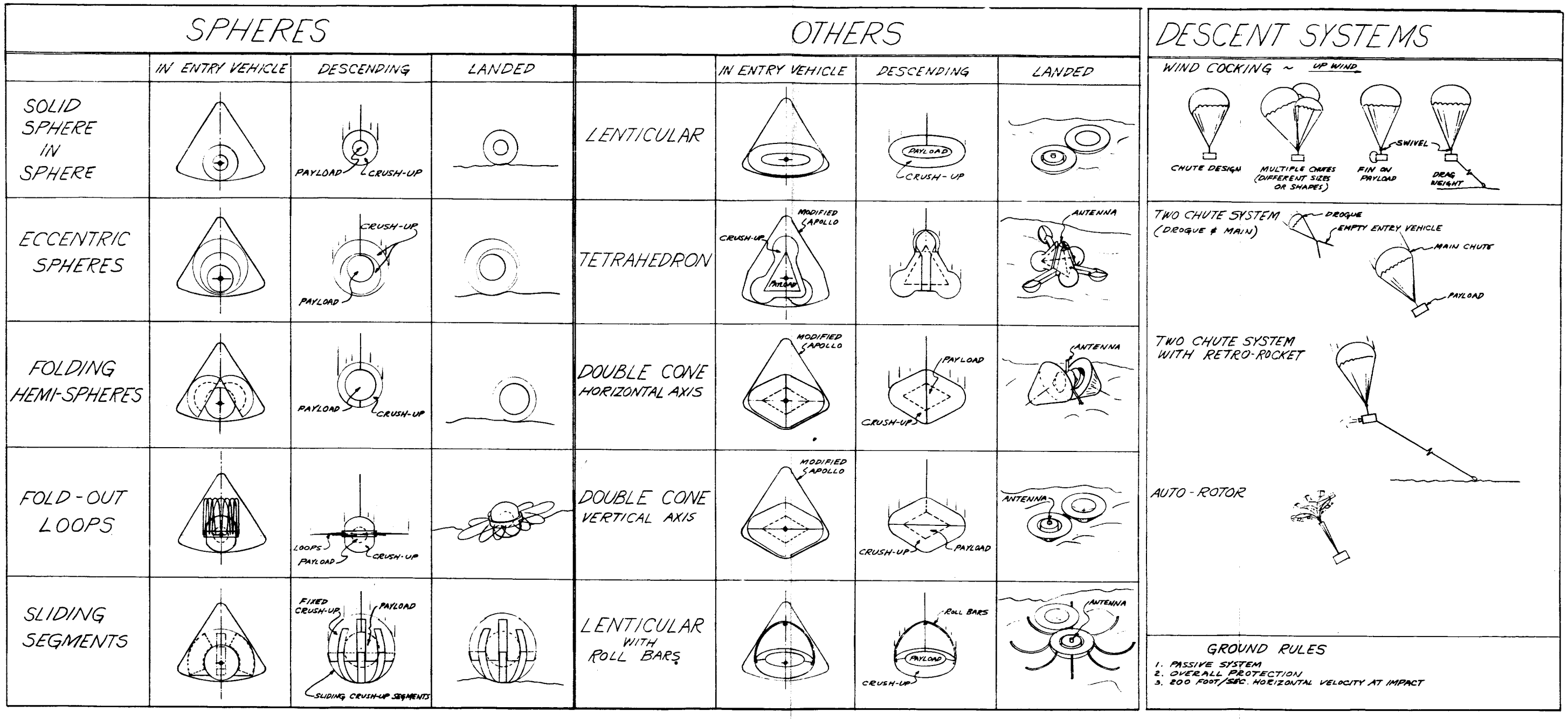


Figure 54 LANDED CONCEPTS - ADVANCED MARINER

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[redacted]

indicate promise are the tetrahedren and lenticular (or oblate spheroid) concepts. These concepts have one thing in common, they have a high probability of resting on one of the preferred sides after impact. For instance, the lenticular concept would most likely land on one of two sides, hence two communication antennas would be required. The same reasoning applies to the tetrahedren concept except that an erectable antenna would be employed at one corner and the attenuator deployed to erect it to a vertical position, since the tetrahedren could land on one of four sides. The tetrahedron, however, is not compatible with entry configurations of the class considered for Mars entry (modified Apollo and blunted cone). The lenticular shape on the other hand is very compatible with these entry configurations but is undesirable with the tension shell, due to the sharp nose angle (30 degrees).

One other landed concept that is similar to the lenticular is the double cone (vertical axis). It is compatible with the blunt cone entry shell and possibly the tension shell. The "clam shell" concept is highly desirable from the science deployment and antenna erection standpoints, but requires complex mechanisms to achieve deployment. The concepts presented in Figure 54 either are too complex or are incompatible with design requirements and entry shell configurations.

After extensive evaluation of the above concepts, several other approaches were investigated to simplify erection and deployment techniques of the antenna and science, as well as to maximize compatibility with the entry configuration (i.e., center of gravity locations and volume). These new concepts are presented in Figure 55. The concepts are placed in three groups: topple, topple and erect, and land erect. They represent the degree of simplicity in achieving the mission objective.

The first concept is again the lenticular shape, however the deployment system jettisons a portion of the impact attenuator, uncovering the science equipment for deployment, in lieu of the "clam shell" concept. This concept reduces the design complexity, but requires multiple deployment systems. The more complex deployment system and the dual antenna system requirement cause an increase in packaging volume and consequently an increase in attenuator weight (due to increased size) which ultimately reduces the payload weight availability. The second concept presented in the toppled category is a spherical air-bag approach. In this concept the payload is suspended inside the air bag giving it complete omnidirectional impact protection. This concept creates several critical design problems of which payload accessability is perhaps the most difficult. The payload is jettisoned by inflating an inner tube and ejecting the spherical payload through it; clear of the deflated air bag. Air-bag attenuation systems are not as efficient as crushable material impact attenuators, however one significant advantage of an air-bag system is that it could be utilized also as descent retardation system in place of a parachute.

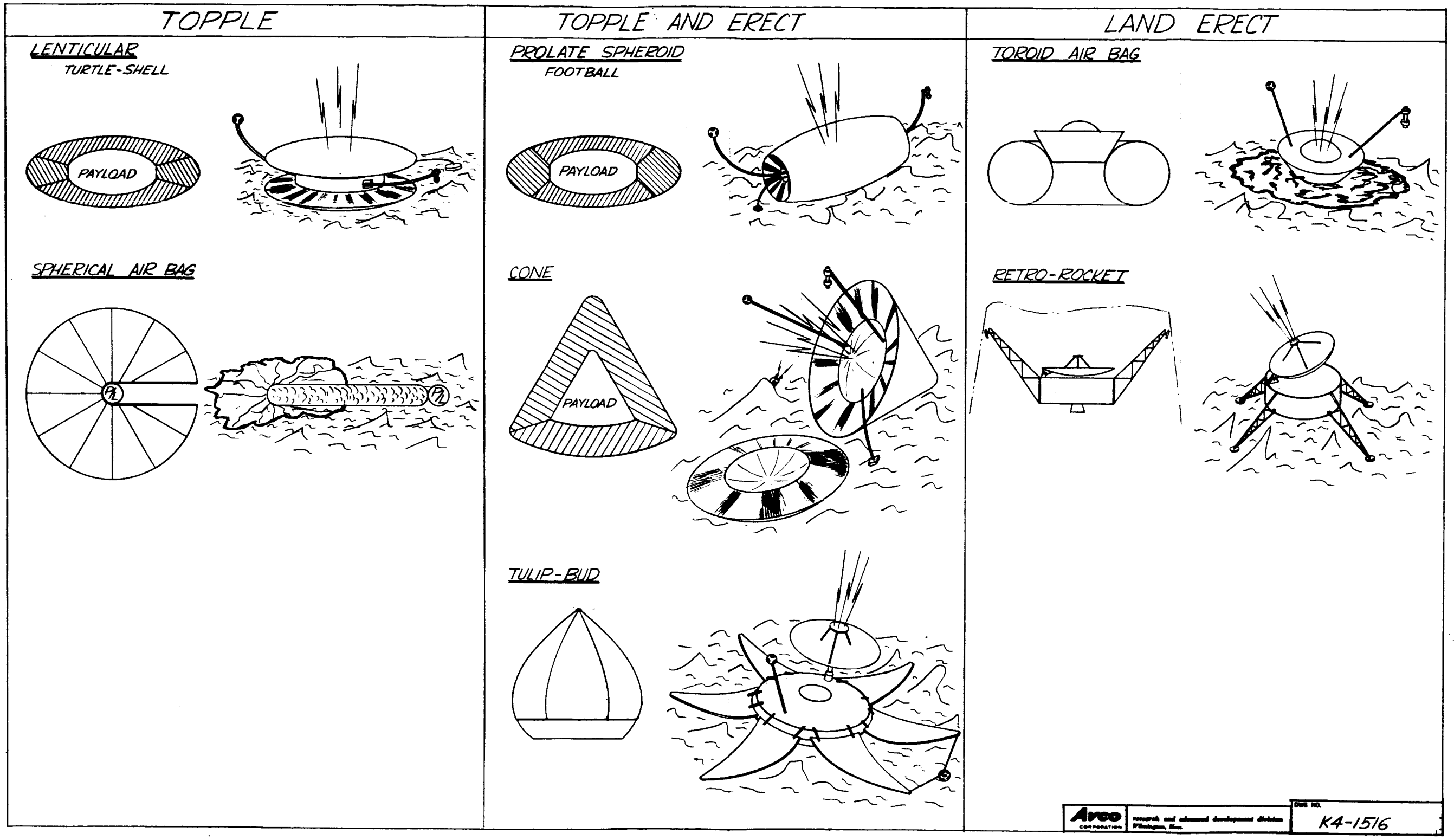

Figure 55 LANDED CONCEPTS - MARS PROBE/LANDER

In the topple and erect category two design features received primary emphasis throughout the design effort, a) the use of a concept that would have the principle feature of the flotation sphere (i.e., erection independent of terrain) and b) the maximum utilization of the packaging volume available in the entry shell configuration. In the first landed capsule concept, a prolate spheroid (football), a cylindrical payload packaging section, is mounted on end bearings allowing the payload to rotate in at least one plane. By designing the antenna mount to gimbal in the other plane, complete upper hemisphere antenna coverage is achieved. Science deployment is accomplished out the ends of the cylindrical section after jettisoning local sections of the impact attenuator. This concept was particularly developed to fit into the tension shell entry configuration, however, after careful analysis it was found that a sphere or cone utilized more of the available volume.

The second landed capsule concept, a solid cone, is similar to the football concept (i.e., it uses a rotating section for erection) except that is uses a conical packaging section rather than a cylindrical section. This concept was developed particularly for the tension shell configuration. Other approaches to this concept have been examined, including a nonerectable antenna system and nonerectable instrument deployment, however, these approaches did not appear promising. One of the significant disadvantages of this landed capsule concept is that at impact a line contact is possible on the side of the cone, causing very high initial g levels. By giving the cone a slight spherical contour, similar to a half prolate spheroid, this problem can be largely overcome.

The last landed capsule concept presented in the topple and erect category is the "Tulip Bud" concept developed in an early Voyager study[6]. For erection, this concept utilizes actuation of petals forming the upper portion of the capsule. It must be noted that this approach was developed for vertical landing velocities of 40 ft/sec with no horizontal wind effect on impact, hence only a minimum of crushable material was needed on the petals. Under the present study ground rules the L capsule must be capable of surviving impact at a 130 ft/sec impact velocity (100 ft/sec horizontal and 80 ft/sec vertical velocity components). This requires complete omnidirectional protection. It is questionable whether the petal concept would work since 10 to 15 inches of crushable materials is required on each petal.

In the land erect category, two concepts are presented: toroidal air bags and retrorocket. The first concept utilizes five spherical air bags (spherical air bags present a more efficient design than a true toroidal design) mounted under the landed payload. At impact the bags rupture and deflate attenuating

[6]Voyager Design Studies, Avco/RAD TR-63-34, V (15 October 1963)

the impact loads. After complete deflation, the landed payload remains erect tilted only by the local terrain slope. Initial analysis of the air bags indicated that they would weigh twice as much as crushable material impact attenuators for the same design conditions. The impact dynamics of this system for a terrain model of 30-degree slope including rocks is also questionable. This concept did offer some significant features such as good communication antenna patterns, ease of science deployment, and good thermal control characteristics. An air bag landed concept is presented in Figure 56 as mounted in the tension shell entry configuration.

The retrorocket concept is felt to be too sophisticated for early emissions, as good terminal guidance and rocket thrust vector control under high wind velocities are necessary requirements for this type of landing. This concept would also require previous knowledge of the terrain to insure adequate impact stability and post-impact erection. Retrorocket firing into the planet's soil prior to landing would contaminate the soil for scientific measurement, hence retrothrusting at high altitudes with free fall (under active control) to impact would be required. These are a few of the problem areas foreseen in this approach which are felt to be beyond the state-of-the-art for the 1971 mission but which must be solved for 1975 and possible 1973 missions.

6.2.2 Candidate Landed Capsule Evaluation

After a preliminary evaluation of the preceding landed concepts, taking into account all design and functional requirements including 1967 state-of-the-art techniques in design and fabrication, it was concluded that the flotation sphere, oblate spheroid (lenticular), solid cone, and the toroidal air bag concepts should be evaluated in more detail during the system selection phase (these concepts are presented in Figure 57).

To properly evaluate these concepts, they must be compared in detail including as many of the design requirements as possible. A comparison matrix is presented in Table XLIV. The comparison represents the pertinent criteria for concept selections.

6.2.2.1 Entry Shell-Landed Capsule Interface

The effect of entry shell configuration upon the landed capsule concepts is illustrated in Figure 58 and 59. The volume available utilizing the flotation sphere concept in the three candidate entry configurations and volume available utilizing the oblate spheroid concept in the three candidate entry configurations are indicated. On each of these curves the center of mass location of the landed capsule within the entry shell is plotted against the available volume of the landed capsule. These curves also show the limiting center of gravity

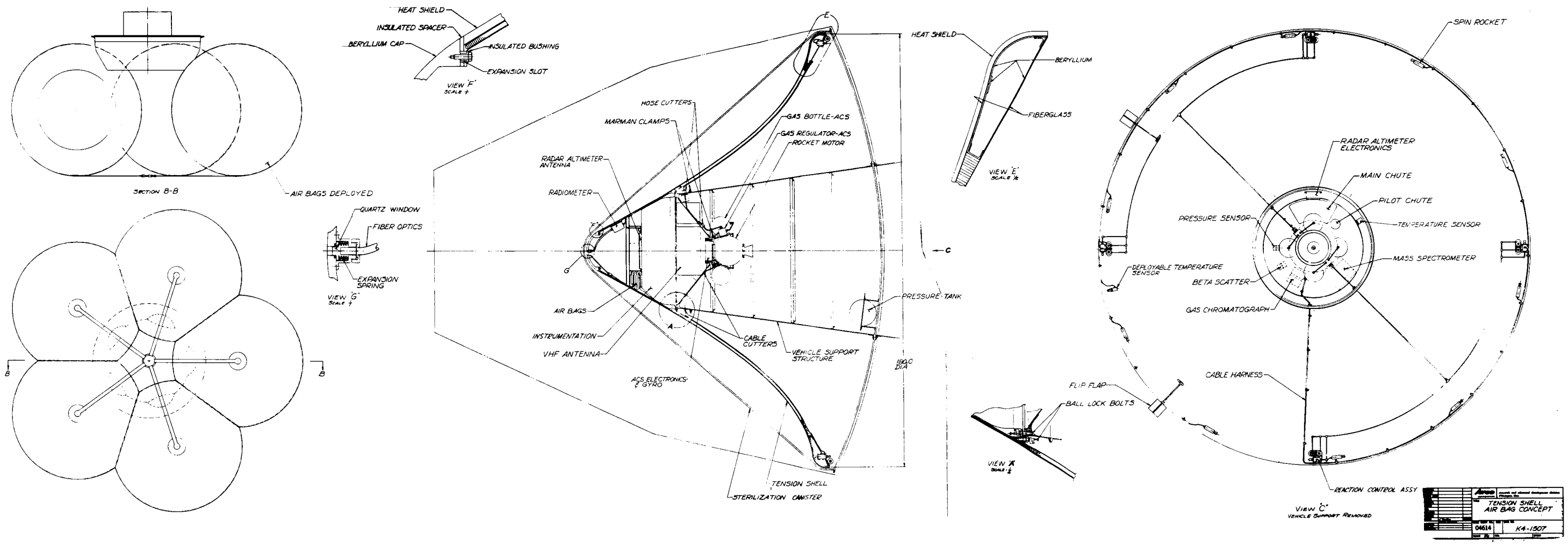

Figure 56 TENSION SHELL - AIR BAG CONCEPT

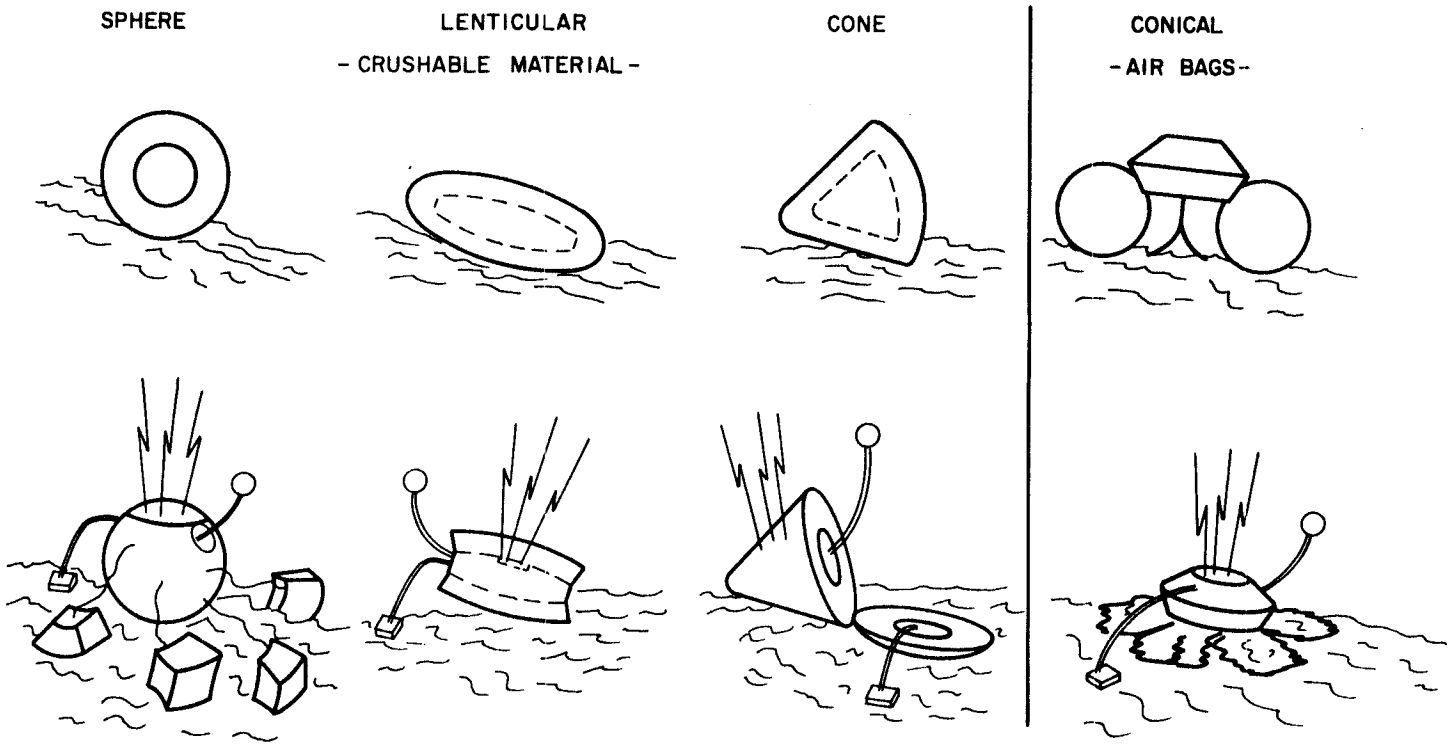

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Figure 57 CANDIDATE 1971 LANDED SYSTEMS

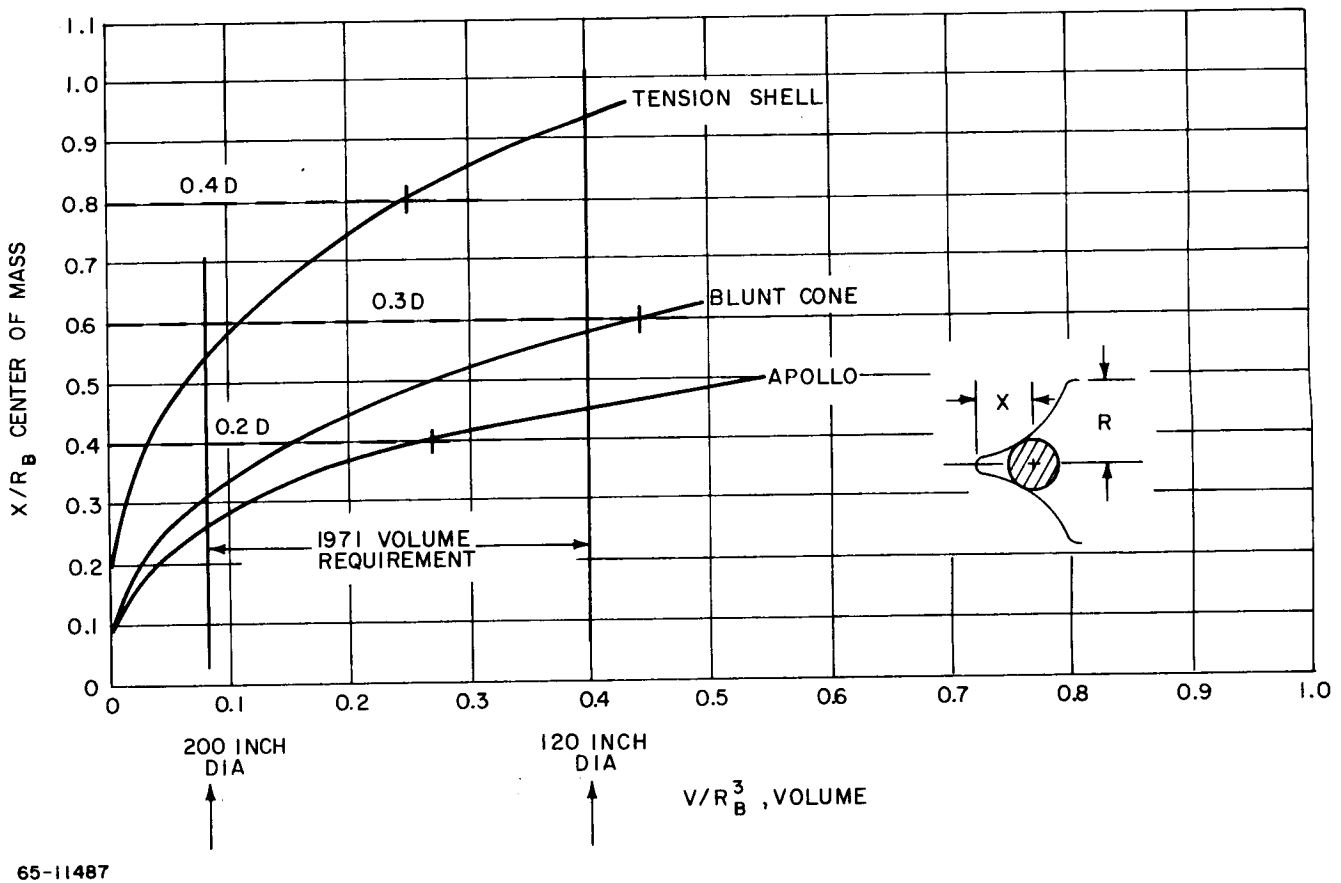


Figure 58 FLOTATION SPHERE - LANDED PAYLOAD - CENTER OF MASS VERSUS AVAILABLE VOLUME

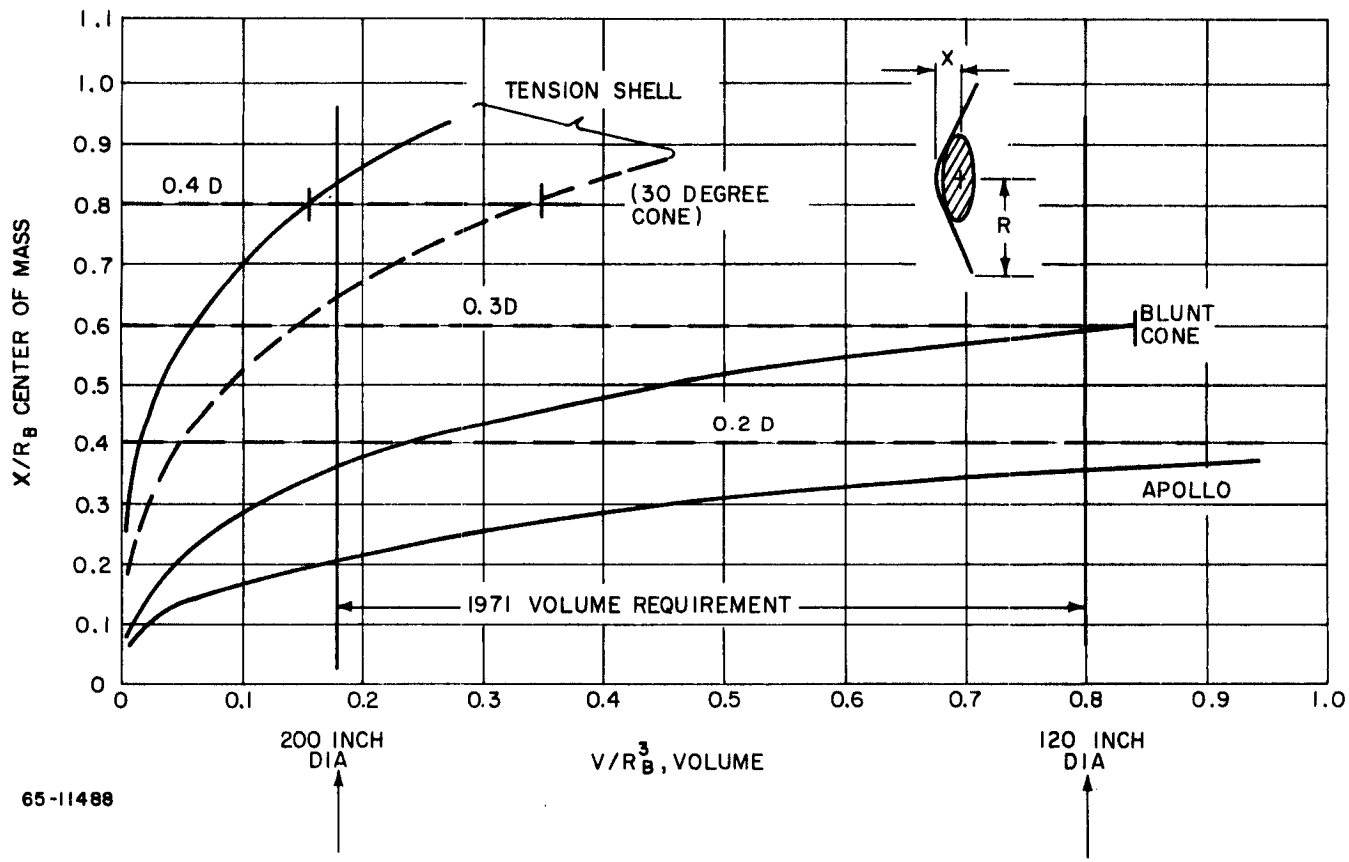
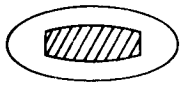
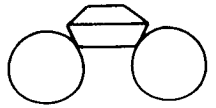


Figure 59 OBLATE SPHEROID LANDED PAYLOAD - CENTER OF MASS VERSUS AVAILABLE VOLUME

TABLE XLIV

LANDED CONCEPT COMPARISON

Comparison \ Concept				
Shell Configuration	Cone Tension	Apollo Cone	Tension	All
Communication and Antenna Requirement	Best	Fair	Fair - Poor	Good
Instrumentation:				
A. Deployment	Poor	Fair	Fair	Best
B. Orientation	Best	Poor	Poor	Fair
Impact Attenuator:				
A. G-Level	Low	Highest	High	Lowest
B. Weight	Lowest	High	High	Highest
Thermal Control	Poor	Fair	Fair	Good
Complexity:				
A. Engineering	Greatest	Low	Low	High
B. Fabrication	Greatest	Low	Low	Least
Reliability:				
A. Engineering	Low	High	High	Lowest
B. Mission	Highest	Lowest	Lowest	Low
Available Weight	Highest	Low	Low	Lowest

position for each of the entry configurations (dashed lines) and the probable capsule volume requirements for the 1971 mission (solid lines). For the oblate spheroid, the volume requirements are approximately twice those of the flotation sphere. It is evident from Figure 58 that the flotation sphere is compatible with all three entry shell configurations, and is extremely well suited for use in the blunt cone.

The oblate spheroid will not meet the volume requirement when mounted in the tension shell, but easily satisfies the volume requirements when used with the Apollo and blunt cone entry shells as shown in Figure 59. The 30-degree solid cone landed capsule concept was developed to more fully utilize the tension shell packaging volume. It can be seen from Figure 59 that the 30-degree solid cone in the tension shell will easily meet the 1971 volume requirements. The air-bag system is quite compatible with any of the entry configurations since the packaging shape of the air bag is not dependent upon the landed capsule shape and hence can conform to almost any configuration.

6.2.2.2 Antenna Requirements

The flotation sphere concept provides orientation; antenna directivity can be obtained, enabling use of a narrow-angle antenna.

Another concept, that of the oblate spheroid and conical shapes require use of wide-angle antennas since the orientation is not well controlled. Of these two concepts, the oblate spheroid allows a somewhat simpler antenna design.

If the air bag impact attenuator concept can keep the landed payload erect after impact, antenna directivity will be more favorable than for the oblate spheroid, but not as efficient as for the flotation sphere.

6.2.2.3 Instrumentation - Deployment, and Orientation

If a crushable type of impact attenuator material is used, the material must be removed after impact to allow deployment of the instrumentation. Use of an air bag concept for impact attenuation allows ready deployment of the instrumentation after impact. The flotation sphere has an additional penalty -- the complex problem of jettisoning the flotation shell surrounding the payload.

From the orientation standpoint, a flotation sphere is most efficient. The other shapes suffering from the same deficiencies are noted in the antenna discussion.

6.2.2.4 Impact Attenuator

Two basic factors must be considered in evaluating the impact attenuator:

1. The g-level reduction which can be achieved.

2. The weight of the attenuator.

The g-level reduction which can be achieved is basically a problem of providing sufficient stroke. The air-bag concept will yield the lowest g-level since it has the greatest stroke capability. However, the air-bag concept is the heaviest. It would weigh about twice as much as a crushable material type impact attenuator for a given flotation sphere payload weight (65 percent of the payload weight versus 31 percent).

The flotation sphere represents the most efficient system because of the compactness of payload packaging which it allows. This means that less impact attenuator volume is required to enable the payload to be encapsulated.

6.2.2.5 Thermal Control

To minimize post-impact loss of heat from the payload to the cold environment, it is desirable to provide the landed package with an outer surface with low radiation characteristics. However, the surface of a crushable material type of impact attenuator may not retain its low radiation characteristics after impact. This would make it desirable to jettison the impact attenuator. There would probably be no need to jettison an air-bag type impact attenuator because of thermal control considerations. It should be noted that instrumentation deployment requirements may require jettisoning of the impact attenuator regardless of thermal control problems.

6.2.2.6 Complexity

Design and fabrication complexities are frequently more dominant considerations than other criteria in selection of a concept. It is clear that the flotation sphere is the most complex and the air-bag concept is the simplest to fabricate. The weight advantages of the flotation sphere landed capsule must be weighed against its design and fabrication difficulties.

6.2.2.7 Reliability

Due to its design and fabrication complexity, the inherent reliability of the flotation sphere landed capsule is actually lower than that of the other concepts. However, assuming that the reliability of the flotation sphere landed capsule can be made equal to that of the other concepts by sufficient development and testing, then the probability of achieving a successful mission with the flotation sphere

is greatest because of its independence of the configuration of the Martian terrain.

6.2.2.8 Available Weight

As noted previously, the flotation sphere concept for the landed capsule provides the greatest available payload weight and the air-bag concept for the impact attenuator allows the least available payload weight. However the results of a study of a typical air-bag design indicated, that adequate weight allowance for this design could not be provided in the payload.

The choice between the flotation sphere and oblate spheroid landed capsule concepts is not yet clearly defined. If weight is the most critical factor, the flotation sphere should be employed. However, a conservative approach would dictate use of the oblate spheroid. Both concepts are further analyzed in the remainder of this section.

6.3 PAYLOAD SYNTHESIS

This section presents, in moderate detail, the results of the more significant tradeoffs within the communications power and instrumentation systems. First, the communications system is data rate limited, restricting the amount of information which can be transmitted to Earth. Second, the flight capsule is severely weight limited, as previously discussed. The more important constraint was the former, however, which results from any attempt to communicate over large distances as that between Earth and Mars without the aid of large directional antennas on both ends of the communications link.

The synthesis presented results in a recommended oblate spheroid landed capsule design which meets all the objectives of the mission but which is significantly overweight. Several compromises in the complement of the scientific instruments were made to reduce the weight of the landed capsule. In addition, post-impact relay communication were eliminated from the design. The alternate flotation sphere landed capsule design synthesis results in an acceptable weight; no compromises were required to accommodate it.

6.3.1 Design Guidelines

The central theme of the communications synthesis was conservatism. That is , when alternatives existed, the approach involving the least technical risk was taken. Functionally redundant means of data transmission were used whenever possible. If either of the above approaches had to be sacrificed because of weight restrictions, the approach was identified as technically risky and alternative approaches were investigated.

6.3.2 Entry and Descent Payload Synthesis

Since the flight capsule communications system was a bit limited, the available data capacity constrained, to some degree, the selection of the engineering and scientific instrumentation to be carried. Only the relay link communications system was considered for the entry and descent portion of the mission.

In analyzing the performance capability of the relay link, the following critical features formed the basis for the evaluation:

1. Communication range between the flight capsule and the flight spacecraft.
2. Flight spacecraft receiving system characteristics
3. Flight capsule transmitter power
4. Operating frequency
5. Flight capsule antenna gain
6. Modulation - detection techniques

6.3.2.1 Communication Range Between the Flight Spacecraft and The Flight Capsule

Figure 60 shows the geometrical relationship between the terminal points of the relay link during the entry phase of operation. In this phase, the communications range is longer than in any other mission phase and is determined by the desired lead time between the flight capsule and the flight spacecraft. Selection of the lead time is based on the requirement that the relay communications mission be accomplished and that transmission of data to Earth be completed before the spacecraft is maneuvered into its orbit injection attitude. The exact lead time chosen is a function of the degree of conservatism with which it is selected. In this study, three lead times, reflecting three different degrees of conservatism, were considered, as follows:

1. Two Hours -- This represents the absolute minimum time which could be assigned on the basis of the following assumptions: 1) the entry and descent period is a maximum of 15 minutes, 2) the postlanding relay operations are 10 minutes, 3) the flight spacecraft-to-Earth data capability is far greater than the flight capsule-to-flight spacecraft data capability and hence, data transmission to Earth consumes no additional time, 4) the command to the flight spacecraft to assume its maneuver attitude is received immediately after the last data is sent to Earth via relay link, 5) the transit delay for the command verification

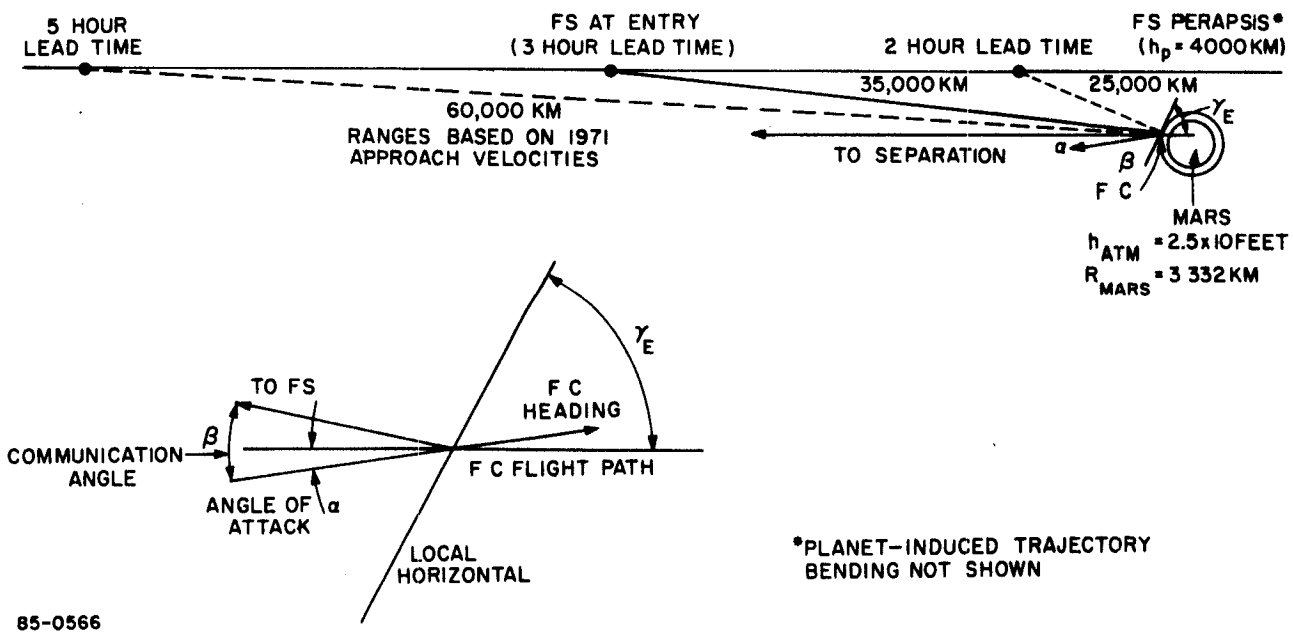


Figure 60 COMMUNICATIONS GEOMETRY AT ENTRY

signal from the flight spacecraft to Earth is 11 minutes, 6) the transit delay for an execute command from Earth is 11 minutes, 7) 30 minutes is taken to execute the attitude change maneuver, 8) the transit delay for telemetering engineering data to Earth is 11 minutes, 9) the transit delay for an orbit injection command from Earth is 11 minutes, and 10) approximately 20 minutes is used on Earth to cogitate.

2. Three Hours -- This represents the nominal lead time with additional time for decision making on Earth and, possibly, time for multiple attempts to attain the proper injection attitude.

3. Five Hours -- This represents the maximum lead time with a time allowance for multiple attempts to attain the proper injection attitude, as well as considerable time to think between events.

Figure 60 illustrates the approximate communications ranges associated with the three lead times are 25,000 km, 35,000 km, and 60,000 km.

6.3.2.2 Flight Spacecraft Receiving System Characteristics

As above, several assumptions were made regarding the flight spacecraft. The characteristics listed in Table XLV reflect these assumptions. The rationale for their selection follows:

1. Receiving Antenna

a. Minimum Performance -- It was assumed that size and view-angle constraints would inhibit freedom in selecting the receiving antenna system. The highest frequency below L-band, where an allocation can be obtained without difficulty is 400 MHz. Since system performance at the above L-band frequencies would be greatly degraded, a 400 MHz turnstile (5.5 db) antenna was selected to meet minimum performance requirements.

b. Nominal Performance -- It was assumed that the optimum relay frequency, from a flight capsule viewpoint, could be chosen and that the only major constraint was that the antenna be a body-fixed type. A 10-db helix-type antenna was selected to meet nominal performance requirements.

c. Maximum Performance -- It was assumed that the antenna could be mounted on the Planetary Science Platform (PSP) and pointed toward the center of the planet both during entry and after orbit injection. Such an arrangement would simplify the problem of relay link communications after orbit injection.

2. Receiver Noise Temperature -- No particular difficulty was expected in achieving a system noise temperature of 1450°K, but any

TABLE XLV

SIGNIFICANT SPACECRAFT CHARACTERISTICS

	A. (Minimum Performance)	B. (Nominal Performance)	C. (Maximum Performance)
Communications range at entry	60,000 kilometers	35,000 kilometers	25,000 kilometers
Lead time	5 hours	3 hours	2 hours
Receiving antenna gain and type	Body fixed 5.5 db maximum gain	Body fixed 10 db maximum gain	Steerable on PHP 10 db maximum gain
Receiver noise temperature	1450°K	1450°K	1450°K
Operating frequency*	400 mc/s	272 mc/s	272 mc/s
Down link data rate**	<100 bit/sec	>1000 bit/sec	>1000 bit/sec
Command capability	None	None	
. Function	N/A	N/A	Lander turn-on capability
. Frequency	N/A	N/A	VHF
. Power	N/A	N/A	25w
Orbital operation	No contact after orbit injection	No contact after orbit injection	Contact by command during period between 20-30 hours after entry
Orbit geometry			
. Periapsis	N/A	N/A	4000 kilometers
. Apoapsis	N/A	N/A	14,000 kilometers
. Inclination	N/A	N/A	40 degrees
Heading	N/A	N/A	South

*The operating frequencies were selected after analysis of the overall relay problem and should not be considered as inferred constraints.

**Data rate from the spacecraft to earth.

appreciable reduction in the figure was considered impractical. Hence, a maximum noise temperature of 1450°K was assigned for all models.

6.3.2.3 Flight Capsule Transmitter Power

The 1971 flight capsule will very likely employ an energy storage device such as a battery for the primary power supply. It might therefore be assumed that transmitted power is not a particularly significant parameter. This assumption is based upon the following logic. If the energy required to transmit a single bit of information is constant, and if the mission data bit content is constant, then for the same battery weight, the desired information can be transmitted at any power level desired. Unfortunately, the constant energy per bit and hence constant battery weight for a given total bit content assumption is not quite true. The following practical considerations determine the transmitter power used:

1. Potential Antenna Breakdown Problems -- The problem of antenna breakdown becomes serious at low atmospheric pressures (Volume V Book 3 Sect. 4.4) and must be considered in selecting the power level. Although antennas can be designed to operate in any atmosphere at any power level, experience indicates that for simple antenna configurations, and, in general, for antennas with a single radiating element, extreme care must be taken in handling power levels as low as 10 watts to insure freedom from voltage breakdown in low pressure atmospheres.

Because of the present uncertainty in estimates of the Martian atmospheric composition, it is not wise at this time to consider transmitter power levels greater than 30 to 40 watts.

2. State-Of-The-Art Limitations On Device Power Handling Capabilities -- Solid-state devices (transistors) are currently available with power output capabilities in the 10 and 15 watt range for carrier frequencies in the 270-400 MHz band of interest. It is predicted, however, that by mid-1966 this capability will be increased to 15 to 30 watts in the same frequency band. Power levels above the previously stated values can be achieved using vacuum tube amplifiers.

3. Avoidance Of High Voltages -- It is desirable to avoid high voltage potentials when operating at the low atmospheric pressures expected during entry and descent because high voltage arcing is possible unless adequate precautions, such as pressurizing the high voltage components, are taken.

In summary, transmitter powers above 40 watts are unattractive because of potential antenna breakdown problems, devices requiring high voltages are unattractive from the point of view of design risk in the event of pressurization failure, and solid-state devices are not currently available and will not be available in the near future with output powers above 30 watts.

Since high voltage arcing will always constitute a design risk, 30 watts was chosen as the transmitter power level with the recommendation that a solid-plate amplifier be developed for its provision.

6.3.2.4 Relay Operating Frequency

Figure 61 summarizes the results of a frequency selection analysis which is presented in detail in Volume V Book 3 Section 4.4. The criteria used in selection process were the following:

1. Received signal power, assuming a fixed aperture constraints upon the flight spacecraft antenna.
2. Received signal power, assuming a fixed beamwidth constraint upon the flight spacecraft antenna.
3. Transmitter power output, assuming a completely solid-state design.
4. Ability to obtain a frequency allocation.
5. Weight of the transmission system and battery for a fixed data content.

All relevant factors point to selection of frequencies in the VHF region. Allocations can be obtained without difficulty at approximately 137, 272, and 400 MHz. Payload volume limitations made a frequency of 137 the least attractive of the three possibilities. 400 MHz was selected for the minimum performance model, while 272 MHz was selected for the nominal and maximum performance models.

6.3.2.5 Flight Capsule Antenna Gain

Look angles - Since the most critical phase of relay operation occurs during the period between the end of communications blackout and planetary impact, the first step in defining antenna performance requirements is examination of the flight capsule-to-flight spacecraft look angles during that period. The factors which determine the look angles are:

1. Entry Angle -- The entry angle, i.e., the angle between the velocity vector of the vehicle and the local horizontal. This angle is determined by the location of the desired impact point and arrival date.

2. Angle of Attack -- The angle of attack, that is, the angle between the flight path and the roll axis of the flight capsule. During the latter part of entry (after blackout but before parachute deployment) this angle is relatively small due to convergence caused by aerodynamic forces.

3. Parachute Dynamics -- The possibility of high winds being present near the planet's surface make it necessary to consider the effect of parachute dynamics. Figure 62 shows the response of the vehicle to a 50 ft/sec step in wind velocity. It should be noted that the net effect is a transient oscillation of the suspended payload about a mean angle of attack.

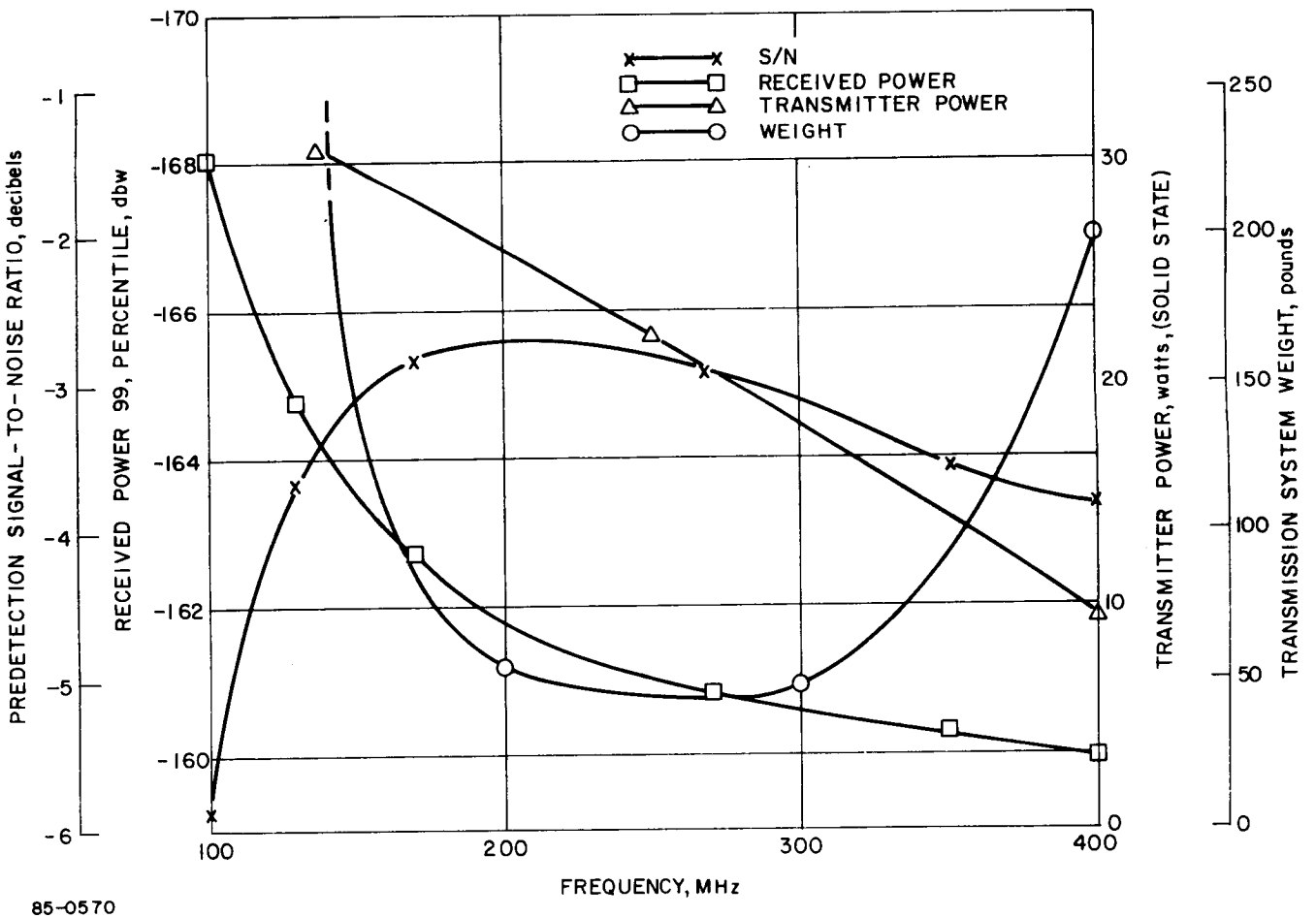
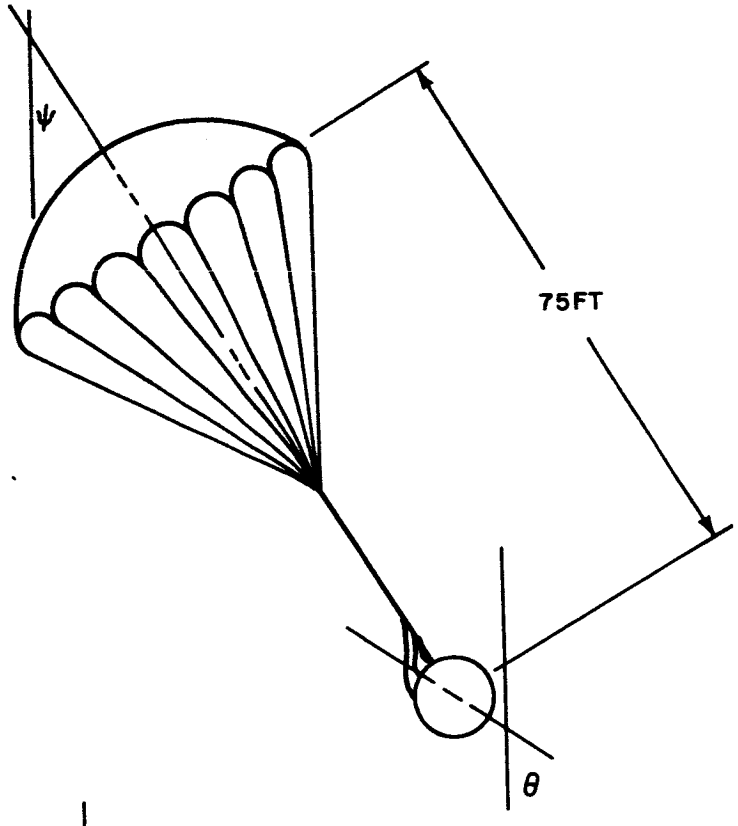


Figure 61 RELAY FREQUENCY SELECTION



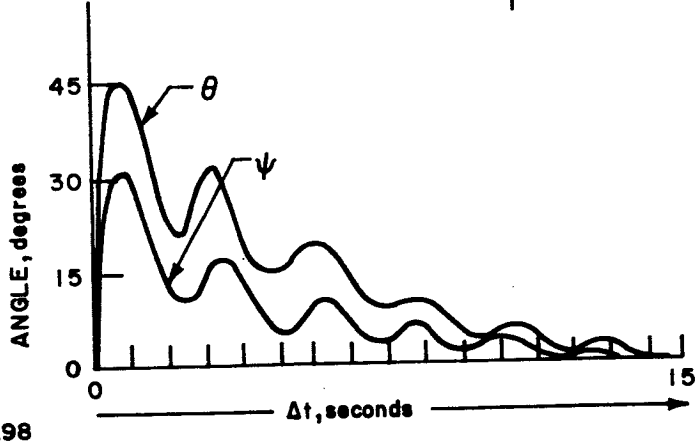


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Figure 62 PARACHUTE DYNAMICS

4. Flight Capsule to Flight Spacecraft Range and Approach Asymptote -- The range between the relay link terminals during entry and descent, and the flight spacecraft trajectory effect the value of the flight capsule-to-flight spacecraft look angles.

A composite representation of the look-angle time history for the case of 44 degree entry angle and a 50 ft/sec wind gust during terminal descent is given in Figure 63. It can be seen that a broad antenna beamwidth is required for operation during this phase.

Multipath transmission - The second step in defining antenna performance requirements is examination of the various aspects of multipath transmission. Figure 64 shows the various concepts under consideration is designing the suspended package and the radiation patterns during entry and during parachute descent. It is apparent from the figure that the antenna must be designed for a number of configurations, including the following configurations.

1. Spherical Package without an afterbody -- Broad beam-width antennas mounted on this shape direct considerable energy toward the planet.

2. Lenticular Package without an Afterbody -- Broadbeam antennas on this shape direct little energy toward the planet.

3. Any Package with an afterbody -- Antennas on this type of vehicle are similar to those on the lenticular package in that they confine the radiation to the desired hemisphere.

The problem created when energy is directed toward the planet results from the fact that a certain fraction of this energy is reflected in the direction of the flight spacecraft. Since the suspended package is in motion, the flight spacecraft receives a fluctuating signal caused by the periodic cancellation and reinforcement of the direct and reflected signals. If the signal drops below the receiver threshold during a null, detection is impossible. For a complete treatment of this subject, see Volume V Book 3 Paragraph 4.4.

In summary, the desired antenna is essentially hemi-omni. That is, it confines all of the energy within the desired hemisphere and with approximately uniform distribution within that hemisphere. Figure 65 indicates the free space pattern of a candidate antenna for use on the suspended package during terminal descent.

6.3.2.6 Modulation/Detection Techniques

In choosing a suitable modulation technique for the flight capsule relay

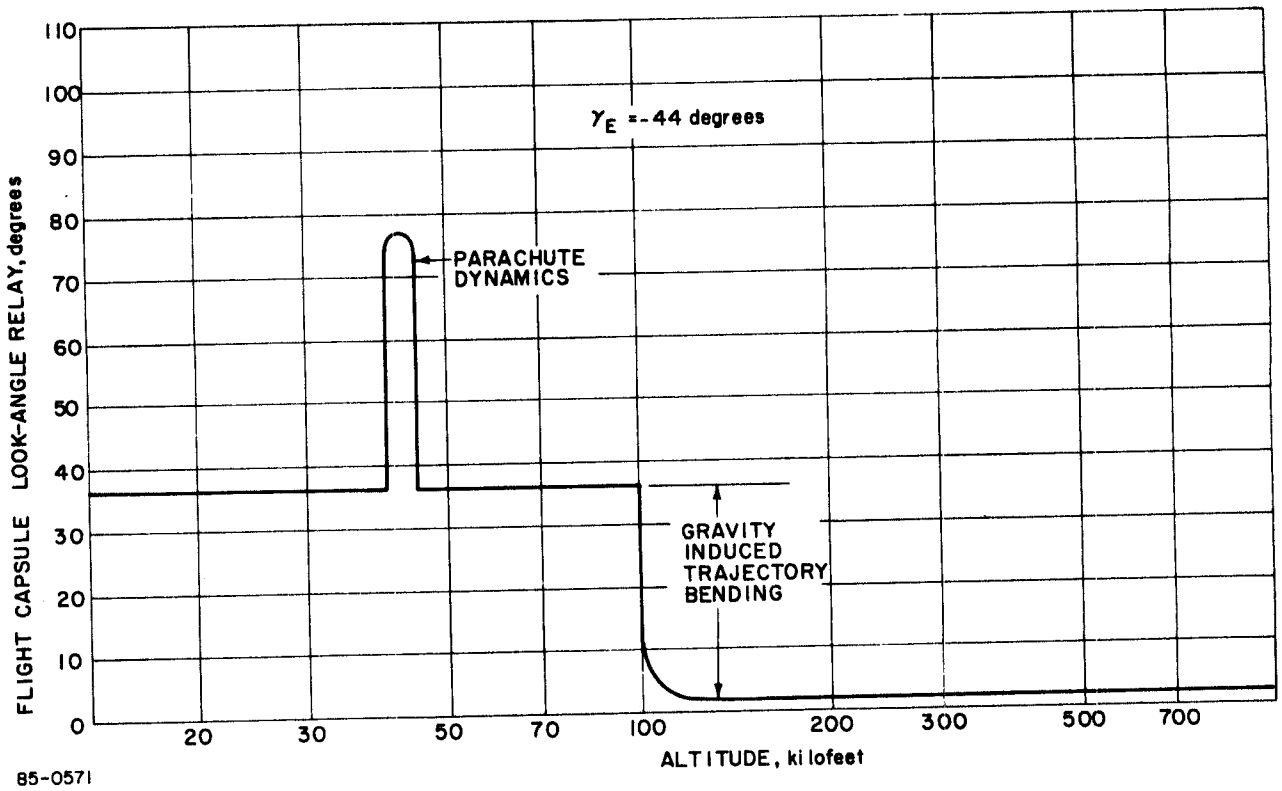


Figure 63 RELAY-LINK FLIGHT CAPSULE LOOK ANGLE REQUIREMENTS

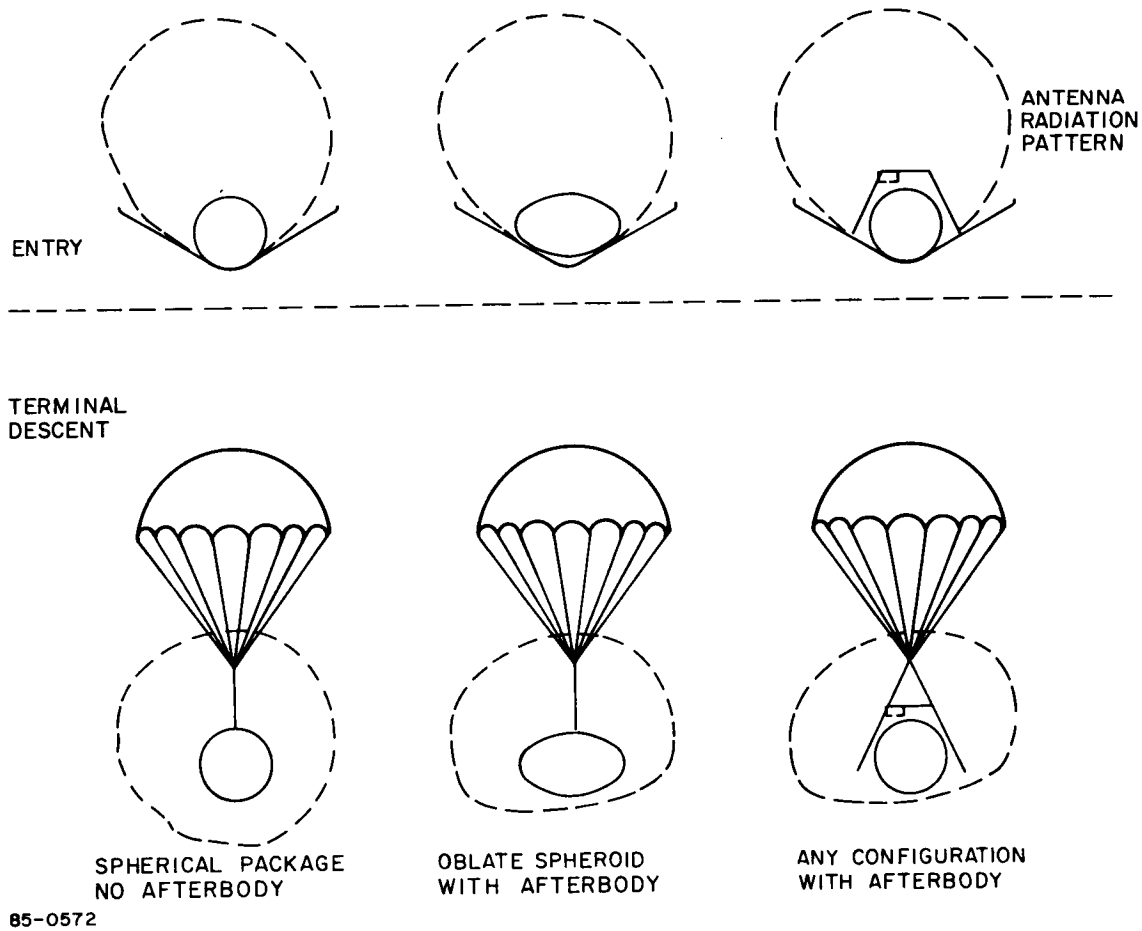


Figure 64 EFFECT OF SHAPE ON ANTENNA PATTERN

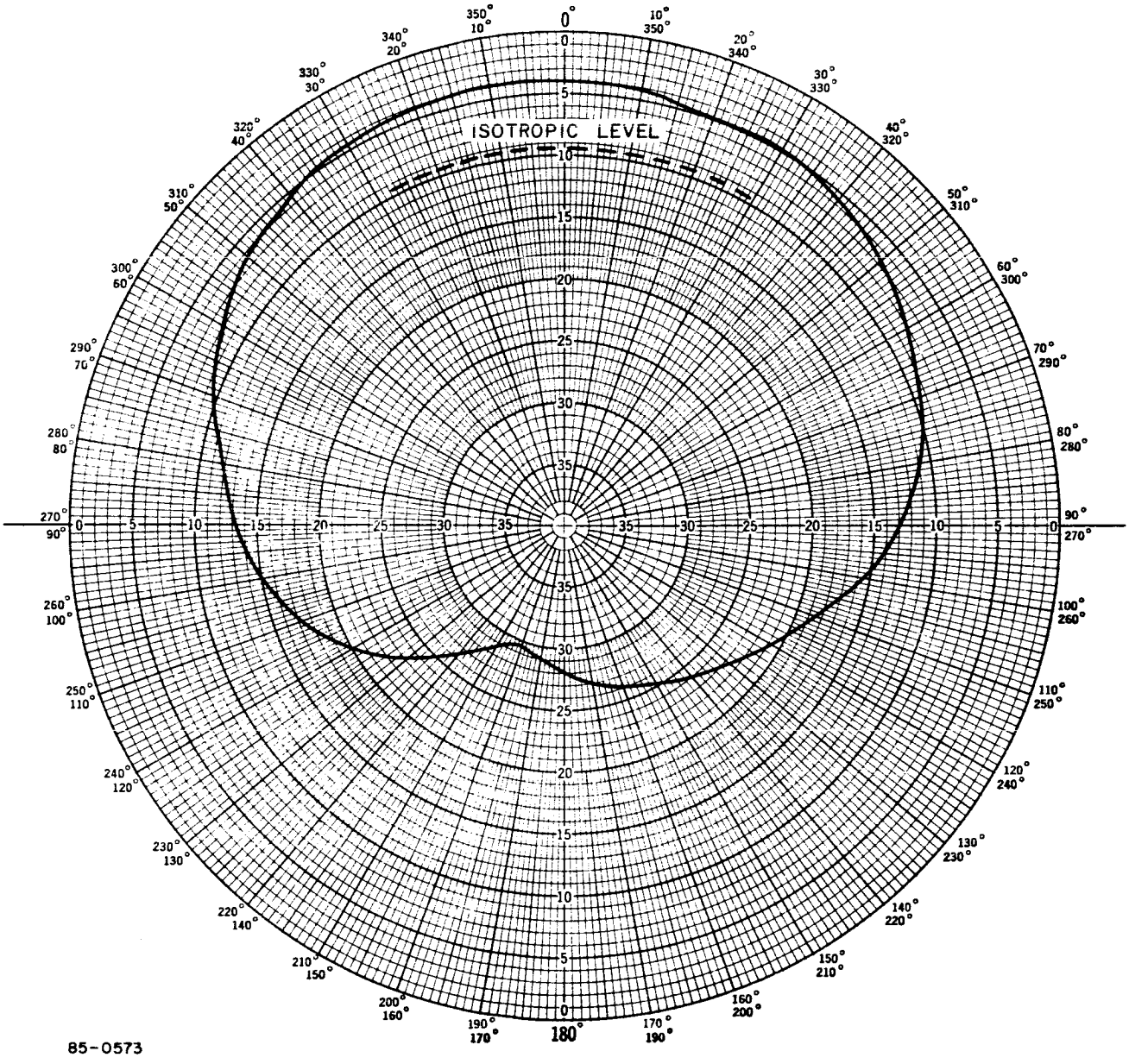


Figure 65 PLANAR SPIRAL ANTENNA PATTERN

link, a number of criteria must be considered. The optimum choice necessitates a critical analysis of, and subsequent compromise among various conflicting selection criteria. Such an analysis was conducted during the study (see Volume V Book 3 Paragraph 4.4). A variety of communication systems were considered including systems for coherent and for noncoherent detection applications.

The major criteria upon which selection of the relay link modulation technique was based were:

Relative communication efficiency

Compatibility with mission requirements

Compatibility with environment

Equipment complexity

Potential for growth

Evaluation of the results of the analysis resulted in the conclusions summarized below:

1. From an overall communication efficiency view point, the the coherent (PCM/PSK/PM) pulse code modulation/phase shift keying phase modulation technique appears to be the best choice for a relay data link.

2. From a compatibility with mission requirements viewpoint, either coherent PCM/PSK/PM or noncoherent (FSK) frequency-shift-keyed modulation will satisfy a minimum 1971 mission without TV. The coherent PCM/PSK/PM system must be used as prime relay link communication mode in 1973 if descent TV is to be provided.

3. From a compatability with environment viewpoint a noncoherent rather than a coherent modulation technique is more attractive since, during entry the radio link will experience blackout for a short period of time and the coherent system will most likely experience loss of carrier phase lock and will need to require this lock after blackout. As the amount of time allocated for reacquisition is decreased, the communication efficiency of the coherent system will degrade since larger amounts of carrier power will have to be allocated to the carrier phase lock loop. A similar argument applies for the descent phase when exposure to large wind gusts could cause conditions during which the relay link performance margin would drop below threshold and cause the coherent link to again lose carrier phase lock. This situation will not exist with a noncoherent modulation scheme such as noncoherent FSK.

4. From an equipment complexity viewpoint a noncoherent modulation technique is slightly favored over a coherent one since the requirement for carrier lock is eliminated in the noncoherent scheme.

The additional complexity of the coherent scheme is minor, however, since the phase lock loop does not contain complex circuitry.

5. From a growth viewpoint the coherent PCM/PSK/PM appears most attractive since it would be required for missions in which TV data is collected.

In summary, both coherent PCM/PSK/PM and noncoherent PCM/FSK are attractive modulation technique candidates. Coherent PCM/PSK/PM is desirable from viewpoints of overall communication efficiency, compatability with mission requirements and growth potential. Noncoherent FSK is desirable from compatability with environment and slightly reduced equipment complexity. From the viewpoint of overall mission success, the following recommendations are made:

1. A single transmitter capable of either phase modulation (PM) or frequency shift keying (FSK) operation should be developed for the 1971 mission.

2. Payloads should be developed for the 1971 mission with FSK data rate capabilities.

3. An experimental program designed to verify the performance of the wideband PCM/FSK system at low signal-to-noise rations (S/N) should be initiated.

4. An experimental program designed to verify the performance of the coherent PCM/PSK/PM system in the areas of acquisition and loss-of-lock characteristics in high velocity and/or deceleration regimes should be initiated.

5. Use of some form of feed-through system to avoid the necessity for bit-by-bit detection onboard the Flight Spacecraft appears feasible. However, considerably more work must be done in this area to arrive at a suitable resolution for the various tradeoffs required.

In developing the relay-link performance capability during the entry descent mission phases, the noncoherent FSK modulation technique was used. Table XLVI presents the data rates achievable for the various Flight Spacecraft options. The link calculations from which these rates were determined are shown in Volume V Book 3 paragraph 4.4.

TABLE XLVI

RELAY LINK DATA RATES

(bits/second)

Flight Spacecraft Model	Minimum	Nominal	Maximum
Data Rate	2	64	128

The mission achievable with the minimum relay link would be clearly unacceptable and the choice at this time of the maximum Flight Spacecraft capability would be unwise since the small lead time allowed for this Flight Spacecraft model would represent a deviation from the principle of conservatism. Accordingly, the prime communication system for the entry and descent phase is considered to the the nominal relay link with its 64 bits per second data rate.

Table IX indicates the scientific experiments which must be included to meet the mission objectives. In addition to the scientific experiments noted there, a number of critical engineering and diagnostic measurements must be made during entry. These include temperature and critical voltage measurements, as well as various event monitors.

Since communications may be blacked out for several seconds early in the entry phase, data taken during that time must be stored and retransmitted prior to impact. If the parachute should fail to open, the descent phase will be too short to transmit other than data of critical importance. Accordingly, during the entry and descent phases of the mission only high priority data will be transmitted in the period between entry and impact unless there is an indication that the parachute has opened. Upon sensing successful parachute deployment and the resultant reduction in the rate of descent, the communication system will switch modes, and the data acquired during blackout interspersed with data taken during parachute descent will be transmitted.

The critical entry and descent data must also be stored for post impact transmission in the event that the descent relay communications system fails.

On the basis of a maximum transmission rate of 64 bits/second the total data transmission capability is 3276 bits for the parachute failure mode, 14,000 bits for a nominal 90-degree entry into the Model 3 atmosphere, and 57,000 bits for the shallower angle of 42 degrees into the Model 2 atmosphere. The science and engineering data list was synthesized on the basis of this limited total data capability. The data list, shown on Table XLVII also indicates the

TABLE XLVII

SCIENCE AND ENGINEERING DATA LIST

Model 3 Atmosphere V_e = 23840'/s γ_e = -90 Degrees

Model 2 Atmosphere V_e = 18400'/s γ_e = -42°

Instrument	Sampling Rate	Bits/ Sample	Acquisition Phase	Shortest Trajectory			Failure Mode Trajectory			Longest Trajectory		
				Time for Phase	Bits Transmitted During Descent	Bits Transmitted Post Impact	Time for Phase	Bits Acquired	Bits Transmitted	Time for Phase	Bits Transmitted During Descent	Bits Transmitted Post Impact
Trapped Radiation Detector	1/5 min	20	200 min prior to entry	200 min	800	200	200 min	800	200	200 min	800	200
Accelerometer (3)	1/sec	21	0.1 g to chute open	24. sec	504	126	58 sec	1218	1218	65 sec	1365	341
	1/20 sec		chute open to impact	205 sec	210	0		0	0	847 sec	882	0
Pressure Transducer (2)	1/sec (tot)	7 each	0.1 g to impact	229 sec	1603	401		406	101	912 sec	6384	1596
Temperature Sensor (2)	1/sec (tot)	7 each	0.1 g to impact	229 sec	1603	401		406		912 sec	6384	1596
Gas Chromatograph	1/20 sec	56	0.1 g to impact	229 sec	616	163		112	56	912 sec	2520	630
Mass Spectrometer	1/5 sec	42	0.1 g to impact	229 sec	1840	462		462	42	912 sec	7644	1911
Radar Altimeter	1/sec	7	0.1 g to impact	229 sec	1603	401		406	101	912 sec	6384	1596
Atmospheric Density (Acoustic)	1/20 sec	21	0.1 g to impact	229 sec	231	63		42	21	912 sec	1845	462
Beta Scatter	1/5 per sec	7	chute open to impact	205 sec	287	72		0	0	847 sec	5929	1482
Radiometer	1/sec	42	0.1 g to chute open	24 sec	1008	252		2436	609	65 sec	2730	682
RF Probe (1)	1 per 2 sec	7	0.1 g to chute open	24 sec	168	42		203	50	65 sec	228	57
Engineering and diagnostic	1/20 sec	24	See Engineering	229 sec	264	0		48	12	912 sec	1080	0
Engineering and diagnostic	1/60 sec	37	and Diagnostic Data	229 sec	111	0		0	0	912 sec	555	0
Engineering and diagnostic	1/300 sec	14	List Sec. 3		0	0		0	0	912 sec	42	0
Engineering and diagnostic	1/5 sec	32	Volume I	229 sec	1504		58	352	88	912 sec	5824	0
Synchronization, Identification & Spares					1812	387		1033	375		7589	1583
Total					14164	2970		7926	2873		58185	12136
Max bits which can be transmitted from 0.1 g to impact less blackout time				(220.3 sec)	14099		51.2sec		3276	(890 sec)	56960	

sampling rates, the quantity of data to be transmitted during descent and, correspondingly, the quantity of data to be stored for post-impact transmission. The total data transmitted is illustrated for the three cases previously discussed: 1) nominal - 90-degree entry into the Model 3 atmosphere, 2) the parachute failure mode, and 3) a -42 degree entry into the Model 2 atmosphere. In the case of the parachute failure mode, only a portion of the data collected during entry is actually transmitted and some instruments, which normally operate only during parachute descent, are never sampled.

With the communications and instrumentation requirements and the maximum available transmission time defined, the descent and entry payload weight can then be determined.

The weight, volume and power consumption of each element was based on use of conservative, state-of-the-art equipment. For a detailed discussion of the individual elements of these systems, see Volume V Book 3 Section 4.0.

A summary of the entry and descent payload is shown in Table XLVIII.

6.3.3 Post-Impact Payload Synthesis

One of the Probe/Lander objectives for the 1971 mission is performance of post-landing functions for a minimum of 24 hours. An effort has been made to define a system not only capable of meeting this goal but also capable of meeting it with minimum reliance of the flight spacecraft.

Both the maneuver of injecting the flight spacecraft into a Martian orbit and the collection and forwarding of data from scientific experiments on the Martian surface are themselves spectacularly difficult accomplishments. It was considered imprudent, therefore, to rely on the relay link as the sole means for accomplishment of the Martian surface data acquisition and transmittal tasks since this would require successful completion of the orbit injection as necessary condition for successful completion of the surface tasks. In view of this, development of a system for direct Mars-Earth communication is required as the prime post-impact communication link or, alternatively, a high-probability of failure to meet the 24-hour requirements for the 1971 mission must be accepted. In a backup role, it would be desirable to have the capability for communications contact between the flight capsule and the orbiting flight spacecraft after the Martian night. This would provide an alternate means of obtaining nighttime scientific data which would be of definite value as insurance against failure of direct link equipment at or after impact. The price paid for this capability must, however, be consistent with the expected return. After careful consideration of the many intricate operations which the flight spacecraft must execute before it can be used as an in-orbit relay link terminal, it was decided that in-orbit contact capabilities after the Martian night would be considered for incorporation in the minimum or nominal performance models discussed earlier only if:

TABLE XLVIII

ENTRY AND DESCENT PAYLOAD WEIGHT VOLUME AND POWER BREAKDOWN

	Weight (pounds)	Volume (in.^{3})	Power Consumption (watts)
Science Payload External			
*Accelerometer (3)	1.8	15	2.0
Trapped radiation (1)	2.2	100	0.8
Pressure (2)	0.6	6	0.3
Temperature (2)	0.6	6	0.2
RF probe (2)	1.3	2	0.3
Acoustic	3.0	49	4.0
Mass spectrometer	8.0	400	10.0
Gas chromatograph	4.8	200	4.0
Radar altimeter	8.0	100	10.0
Radiometer	2.2	7	1.5
Beta scatter	0.8	15	0.2
Cables	5.0	---	---
Brackets	3.9	---	---
Total	40.4	900	33.3
Telecommunications External			
Battery	27.0	420	N/A
Data handling and telemetry	5.0	160	1
Cabling	5.0	---	0
Power conditioning	2.0	30	5
Brackets and mounting hardware	1.0	---	0
Relay antenna	2.5	4800	---
Relay transmitter (1)	3.0	70	75.5
Miscellaneous RF relay	2.0	80	---
Command receiver antenna	0.1	---	---
Total	47.6	5560	81.5

Total descent payload 88.0 pounds

*Weight in postimpact payload

1. The flight spacecraft antenna system used during entry and descent could also be used for in-orbit operations without repointing (see Figure 66).

2. The mutual flight capsule - flight spacecraft visibility time is sufficiently long to be useful and sufficiently predictable to allow contact to be made without command assistance from the spacecraft.

Essentially, in-orbit contact capability would be considered for the minimum or nominal performance models if little or no change had to be made in the flight spacecraft receiving system, a system necessary to support communications prior to orbit injection.

An exhaustive analysis (summarized in Volume V, Book 3 paragraph 4.5) was conducted to select the orbit or orbits capable of satisfying constraints 1) and 2) above. Although several orbits provided adequate mutual visibility, the dispersion in visibility times caused by the various error sources present make the required relay transmitter operating time prohibitive,* unless controlled by command from the flight spacecraft. Incorporation of this capability into the spacecraft was considered so difficult that command capability was eliminated from consideration for the minimum and nominal performance models.

However, this capability was sufficiently attractive, in view of the requirements for a 30-day mission in 1973 and the potential support which could be provided by an in-orbit relay, that it was included in consideration of the maximum performance model.

As a result of this analysis of the orbit geometry, it was concluded that communication after the Martian night would be accomplished by a direct link to Earth and that in-orbit relay capability would be considered only for the maximum performance model.

It presently appears that the 1971 Flight Capsule will be relatively simple in concept and will be incapable of using a high gain antenna. As a result, its direct-link communications capabilities will be limited. Furthermore, there is considerable uncertainty regarding terrain characteristics in the vicinity of the impact point. Both of these considerations dictated adoption of a conservative approach in selecting data rates. In analyzing the performance capability of the direct link, the following critical features formed the basis for the evaluations.

1. Communication range to Earth
2. Receiving terminal characteristics
3. Flight capsule transmitter power
4. Modulation detection technique
5. Antenna performance

*This is true principally because of the low energy-to-weight ratio attainable with Nickel-Cadmium batteries. Selection of a more efficient power source could lead to reconsideration and possible revision of this conclusion.

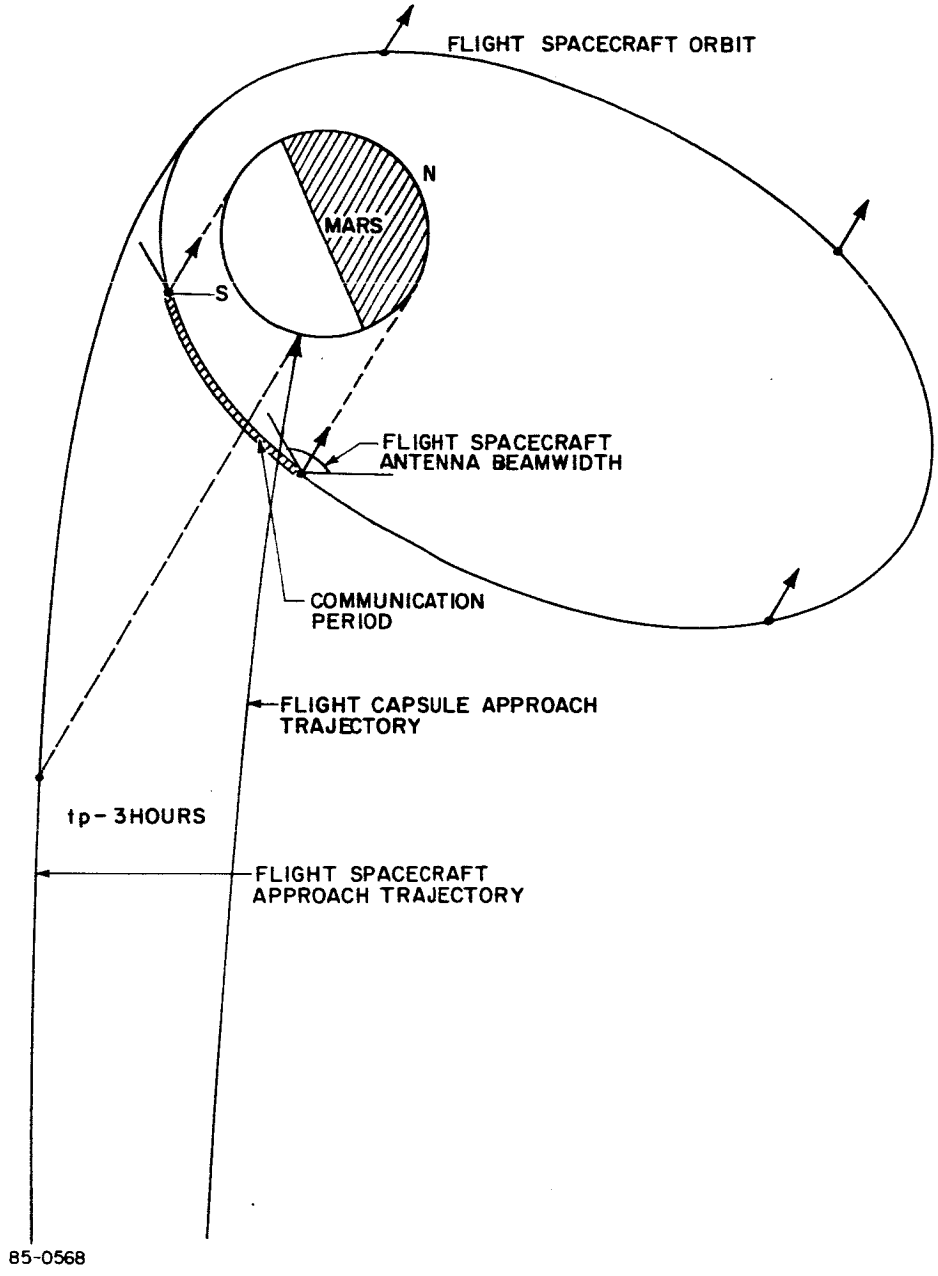


Figure 66 ORBIT CONTACT GEOMETRY

6.3.3.1 Communications Range to Earth

The communications range to Earth is presented as a function of arrival date for the 1971 opportunity in Figure 67. It is assumed that the maximum range will be that which exists on the last day of the orbiter payload optimized window. Consideration has also been given to flying constant arrival date trajectories, but even in such cases the most probable arrival dates are late in the window and, hence, consistent with the last day assumption.

6.3.3.2 Receiving Terminal Characteristics

The deep space network (DSN), including the deep space instrumentation facility (DSIF) and the space flight operations facility (SFOF), will be utilized as defined in Jet Propulsion Laboratory (JPL) document TM No. 33-83, Rev., 1, The Deep Space Network-Space-Flight Operations Facility Capability and Deep Space Network Ground Communication System Development Plan.

6.3.3.3 Flight Capsule Transmitter Power

The following practical considerations determine the transmitter power used:

1. Potential antenna breakdown problems.

2. State-of-the art limitations on power generating device.

Potential antenna breakdown problems were discussed previously (entry and descent) and it was concluded there that due to present uncertainties in estimates of the Martian atmospheric composition, it is not wise at this time to consider transmitter power levels greater than 40 watts.

For direct-link communications a carrier frequency at S-band is required to be compatable with the receiving terminal (DSN) characteristics. Solid state devices for power generation at S-band are currently limited to power output capabilities in the 1 to 2 watt range and the prospects for substantial increases (20 to 30 watts) in the near future are not promising. Although it is desirable to avoid high voltage potentials, as was done in selecting the entry and descent relay-link transmitter power to remain within solid state capabilities, the same option is not available for selection of the direct-link transmitter power if the approach taken is to operate near the antenna breakdown limit of 40 watts. It will be shown later that avoidance of high voltages by limiting the direct-link transmitter power to the solid state range

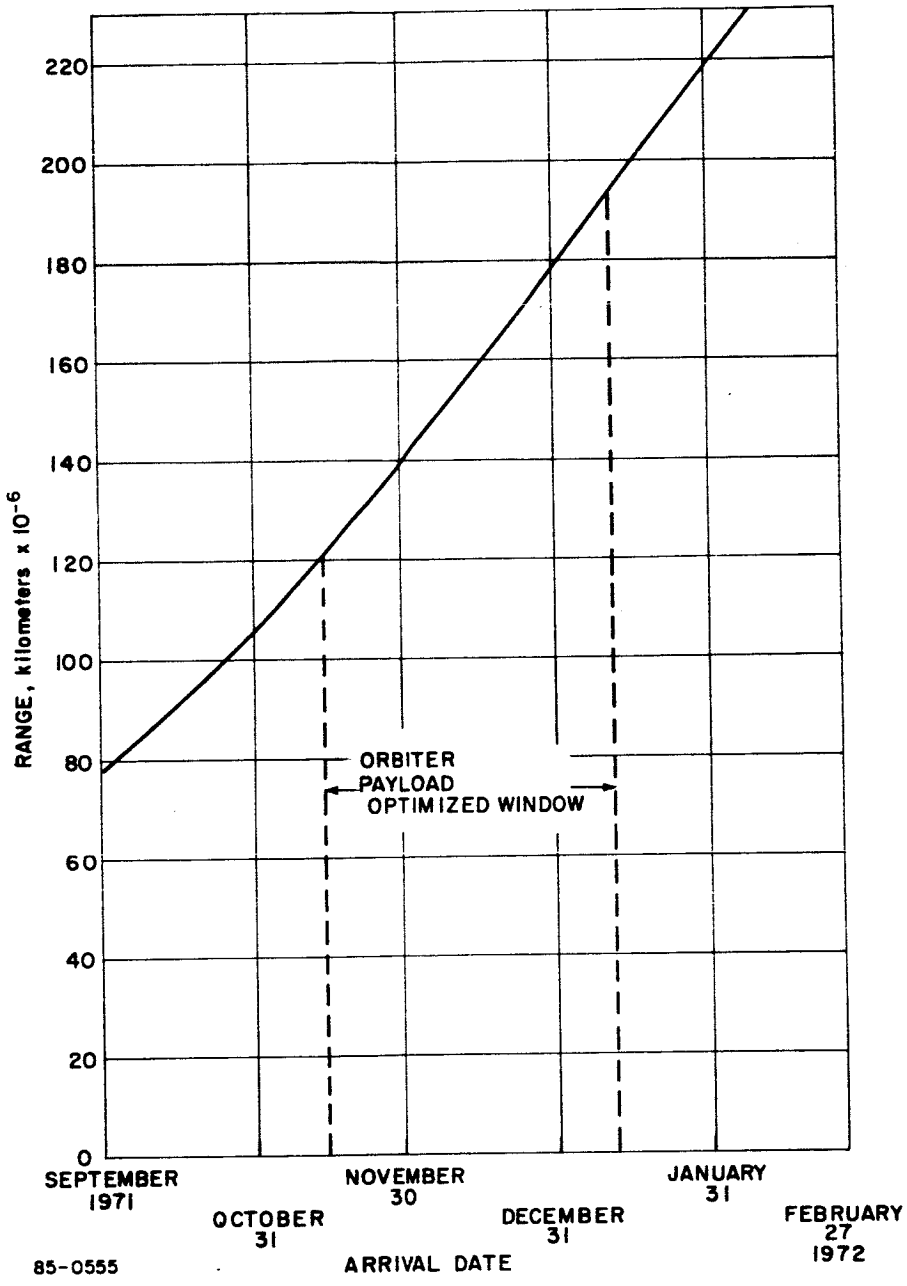


Figure 67 COMMUNICATION RANGE - 1971

of 1 to 2 watts would result in unacceptable link performance. For this reason, the direct link will require the acceptance of the high voltage design risk which is avoided in the relay link.

Figure 68 presents the efficiency achievable, as a function of power, for several candidate tubes operating at S-band frequencies. Two significant conclusions can be drawn from this data:

a. The efficiency improves somewhat with increasing power. Hence, the energy per-bit transmitted decreases with increasing power.

b. Existing hardware for space use is available in the 20-watt or lower class in the case of the more efficient tube types.

In summary, 20 watts was chosen as the direct-link transmitter power level since unacceptable link performance would be attained if the solid state limitation of 1 to 2 watts was used as the criterion to avoid the high voltage arcing design risk.

6.3.3.4 Modulation/Detection Techniques

Because previous deep space programs have used a coherent PCM/PSK/PM approach, it is natural to consider use of the same approach for the flight capsule direct link. However, for antenna gain-transmitter power products of 8.5 dbw or less, the efficiency of coherent PCM/PSK/PM is reduced to zero as the result of the carrier power requirements, as indicated in Figure 69. To a great extent, this reduction results from the fact that oscillator instability thwarts attempts to reduce the receiver noise bandwidth. The PSK data presented in Figure 69 is based on carrier loop bandwidths of 12 and 5 cycles. If smaller bandwidths were used, some relief would result. However, the restrictions resulting from use of smaller bandwidths were so intolerable that a study of other modulation techniques was initiated. Volume V Book 3, Section 4.6 presents the results of this study. In this paragraph the performance of binary coherent systems is compared with the N-ary noncoherent systems (Figure 69 shows the efficiency of 32-ary MFS system assuming optimum detection). Results of the study indicate that if the power-gain product is less than 10 db, there is little choice but to use the more efficient N-ary system. If, however, the power-gain product is between 10 and 15 db, either the N-ary or the phase modulation system could be used, although there is less risk associated with use of the N-ary system. At power-gain levels above 15 db, the phase modulation system would probably be chosen because of its proven performance in other deep space programs.

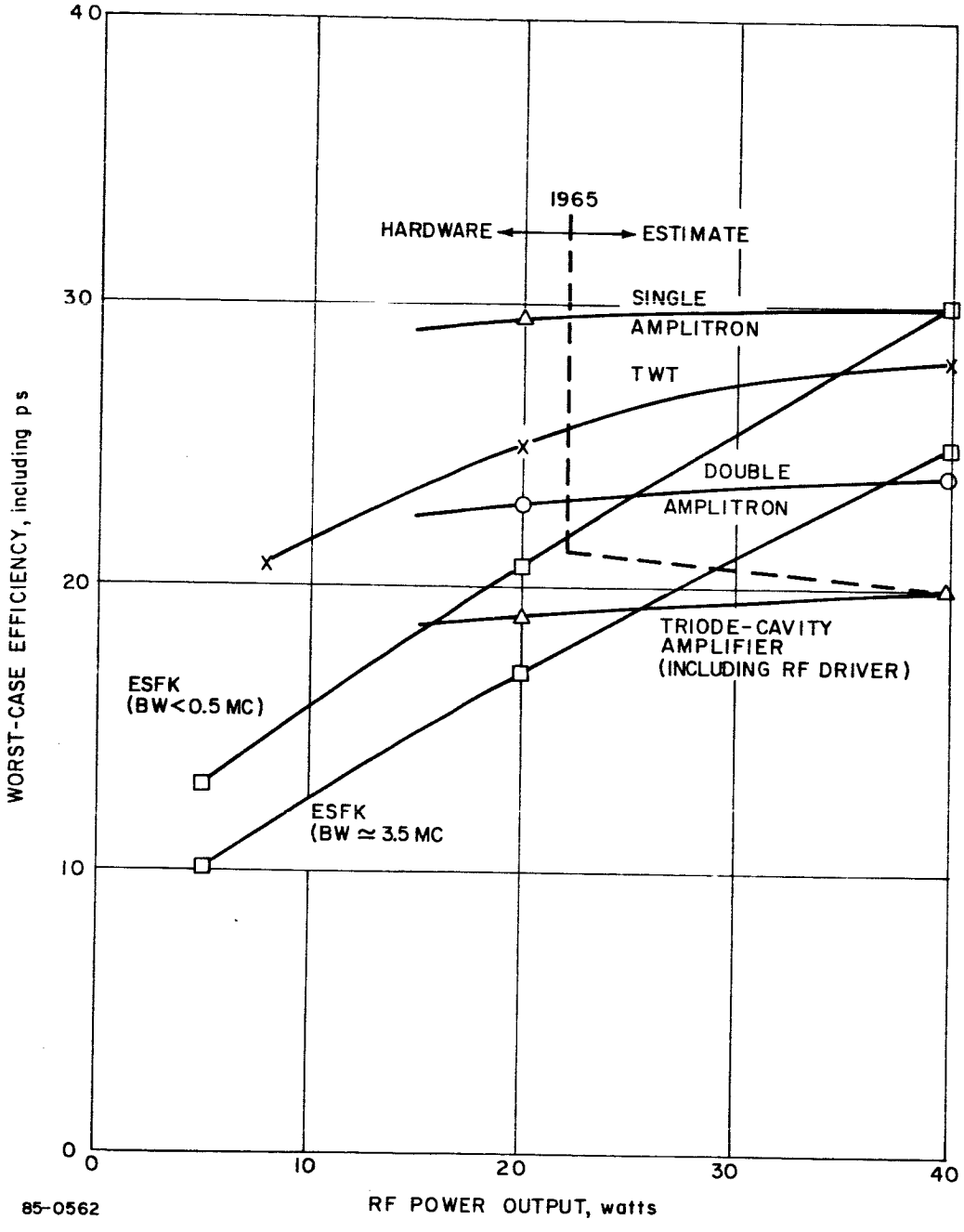


Figure 68 EFFICIENCY ESTIMATES FOR S-BAND POWER AMPLIFIERS

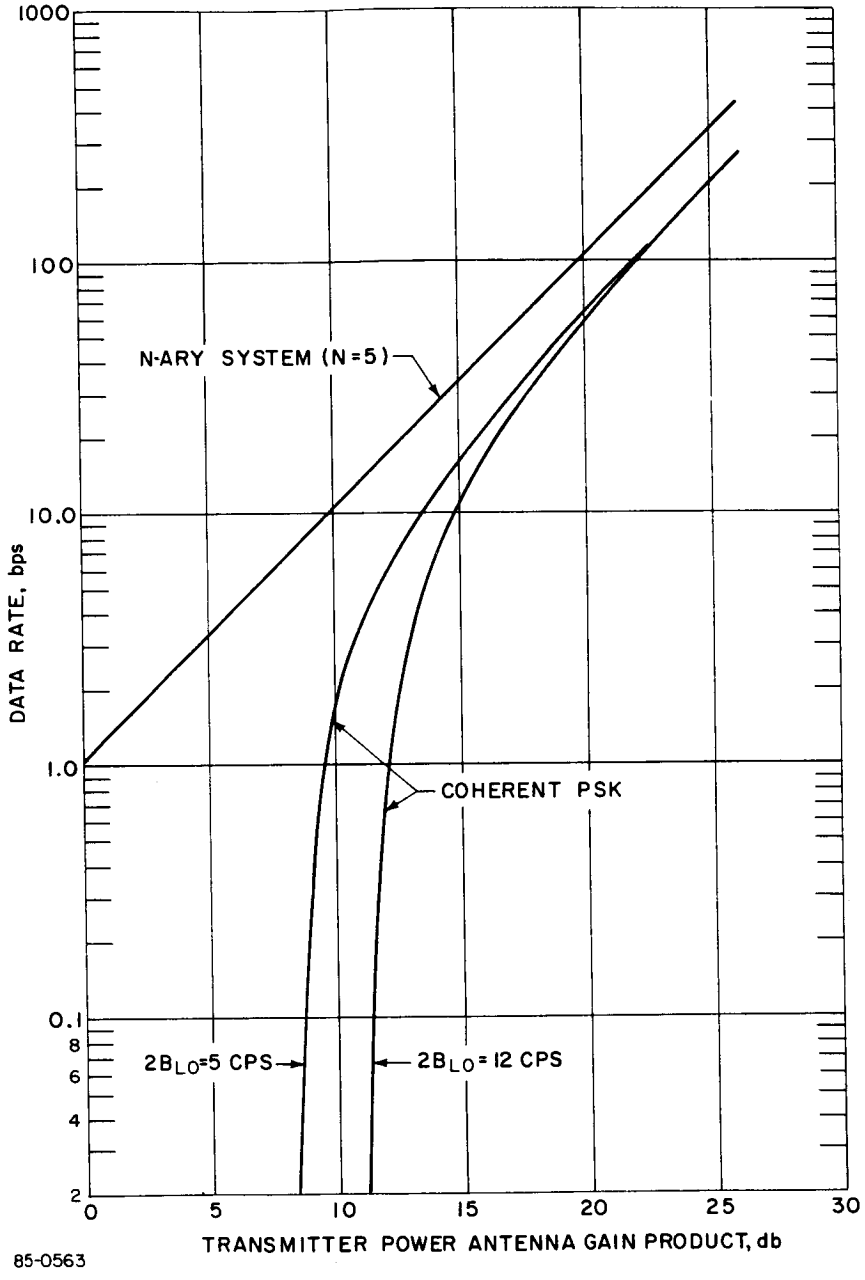


Figure 69 DIRECT-LINK TELEMETRY

Since the transmitter power for the direct link has previously been selected as 20 watts (13 dbw), the antenna performance must be examined before a decision can be reached as to which modulation technique is recommended. Knowledge of antenna performance will determine the power-gain product required for this decision.

Two types of landed payloads were considered during the study, namely: (1) a gimballed payload (flotation sphere configuration) in which the major direction of radiation from the antenna can be at a predetermined angle relative to the local vertical, thus making the antenna design quite straightforward and its performance predictable; and (2) a non-gimbaled payload (oblate spheroid configuration) in which the major direction of radiation from the antenna is a random variable dependent upon the landed attitude of the vehicle.

6.3.3.5 Flotation Sphere Configuration

For this type of payload the antenna beamwidth and, hence, gain is determined principally by Earth-Mars geometrical considerations, including:

a. Landing-site latitude relative to the sub-Earth point

b. The desired transmission time to Earth

c. Errors in knowledge of the arrival time.

For a treatment of direct-link communication system look angles, see Volume V Book 3, Paragraph 4.6.

Figure 70 shows the required look angles as a function of transmission time, for a landing site at the sub-Earth latitude, on the last day of the orbiter window. For a ± 5 degree antenna axis alignment error relative to the vertical and 3-hour transmission time, the required beamwidth is 55 degrees. This corresponds to gains of from 5.0 to 10.0 db at the 27.5 degree points, depending upon the type of antenna used and whether or not beam shaping techniques are employed. In determining the communication capabilities described herein a conservative gain value 5.5 db was used.

This gain results in a power-gain product of 18.5 db and leads to selection of coherent PCM/PSK/PM as the recommended modulation technique.

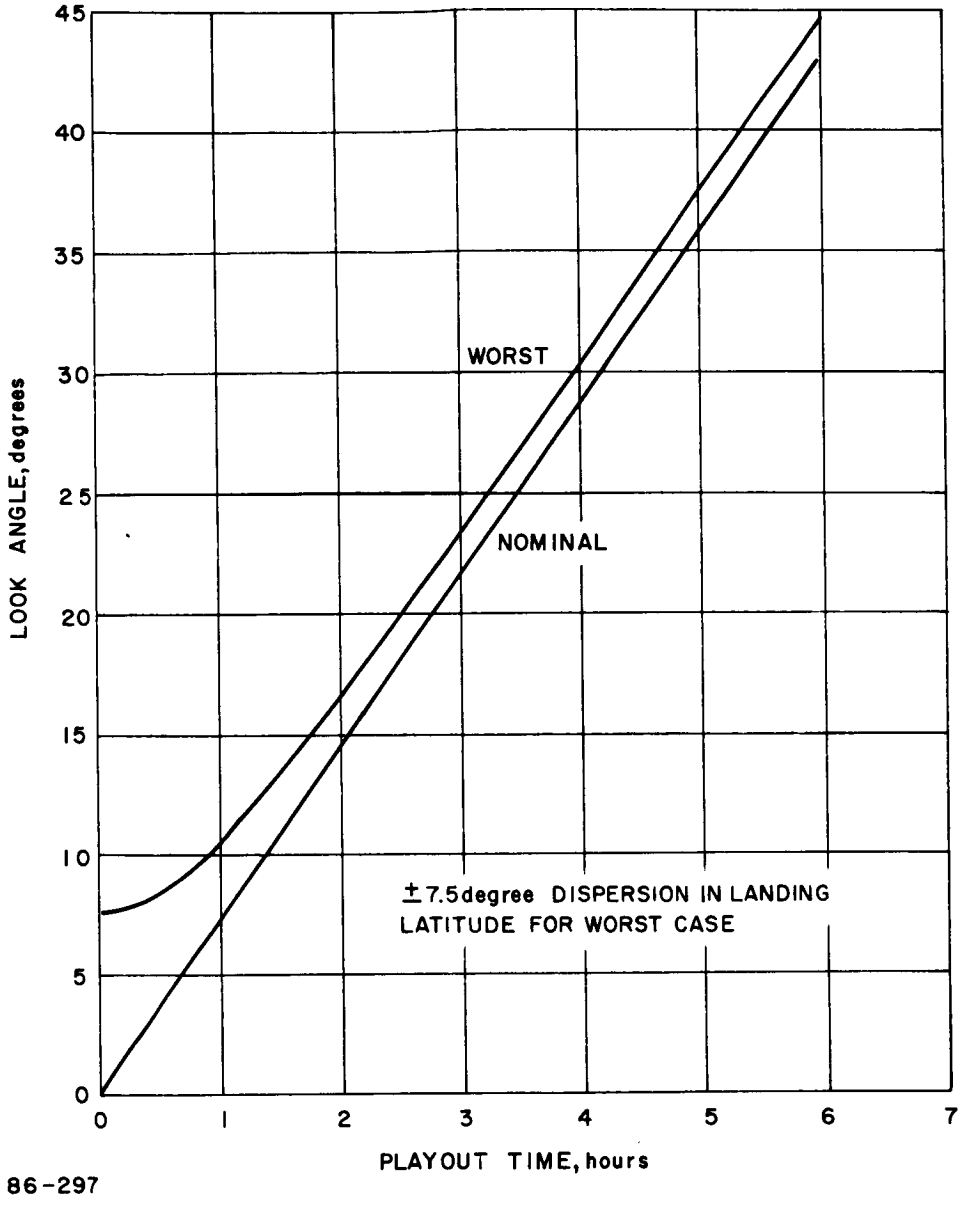


Figure 70 DIRECT-LINK LOOK ANGLE REQUIREMENTS

6.3.3.6 Oblate Spheroid Configuration

A brief analysis was sufficient to show that major problems were associated with use of mechanically steerable antennas on the 1971 mission hard lander. The problems were so formidable that use of such antennas should be avoided unless their absence imposes unacceptable limits on the mission. Elimination of mechanically steerable antennas from further consideration left the following types of antennas as candidates for mission use on the oblate spheroid configuration.

1. Electronically Steerable Arrays--Although a thorough analysis of the feasibility of using this type of antenna was not completed, preliminary results indicate that severe equipment complexity would result from use of electronically steerable antenna arrays on a non-gimballed payload. Steering information would have to be provided either by onboard equipment or by suitable command from Earth. Onboard steering control capable of crude pointing of the antenna could be obtained by using a system based on a combination of time data and knowledge of the local vertical and azimuth. The gains which could be realized by application of this technique would be in the order of 10 to 13 db if a large number of radiating elements were used.

2. Multiple Individually Selectable Antennas--This concept is based on use of a number of moderately high gain antennas, such as horns, and selection of the antenna or antennas which point most nearly toward Earth at a given time. Again, as noted in regard to the electrically steerable antennas, command or onboard vertical and azimuth sensing and timing capabilities would be required. Integration problems and competition between the direct-link antenna and the relay-link antenna for space and viewing angles (if the latter link were used after landing) make selection of this approach unattractive.

3. Fixed Low-Gain Antenna -- For the oblate spheroid configuration, this approach represents the minimum technical risk but requires acceptance of gross penalties in performance capability. Despite its limitations, this type of antenna is receiving the greatest consideration for use on payloads based on a non-gimballed design. In such applications, any orientation must be assumed to be possible, with a particular probability of occurrence assigned to each orientation. This situation for an oblate spheroid configuration is treated in depth in Volume V Book 3, Paragraph 4.6. This paragraph includes data based on pattern measurements made on particular antenna types, which allows presentation of the gains achievable in a probabilistic manner. Figures 71 through 74 present the cumulative distributions of antenna gain for different landed orientations of the payload after landing*. Figure 75

*These distributions are confined to a cone of radiation of 60-degrees half-angle centered about the local vertical. This limitation minimizes the effects of multipath which is prevalent in directions which make small angles with the surface.

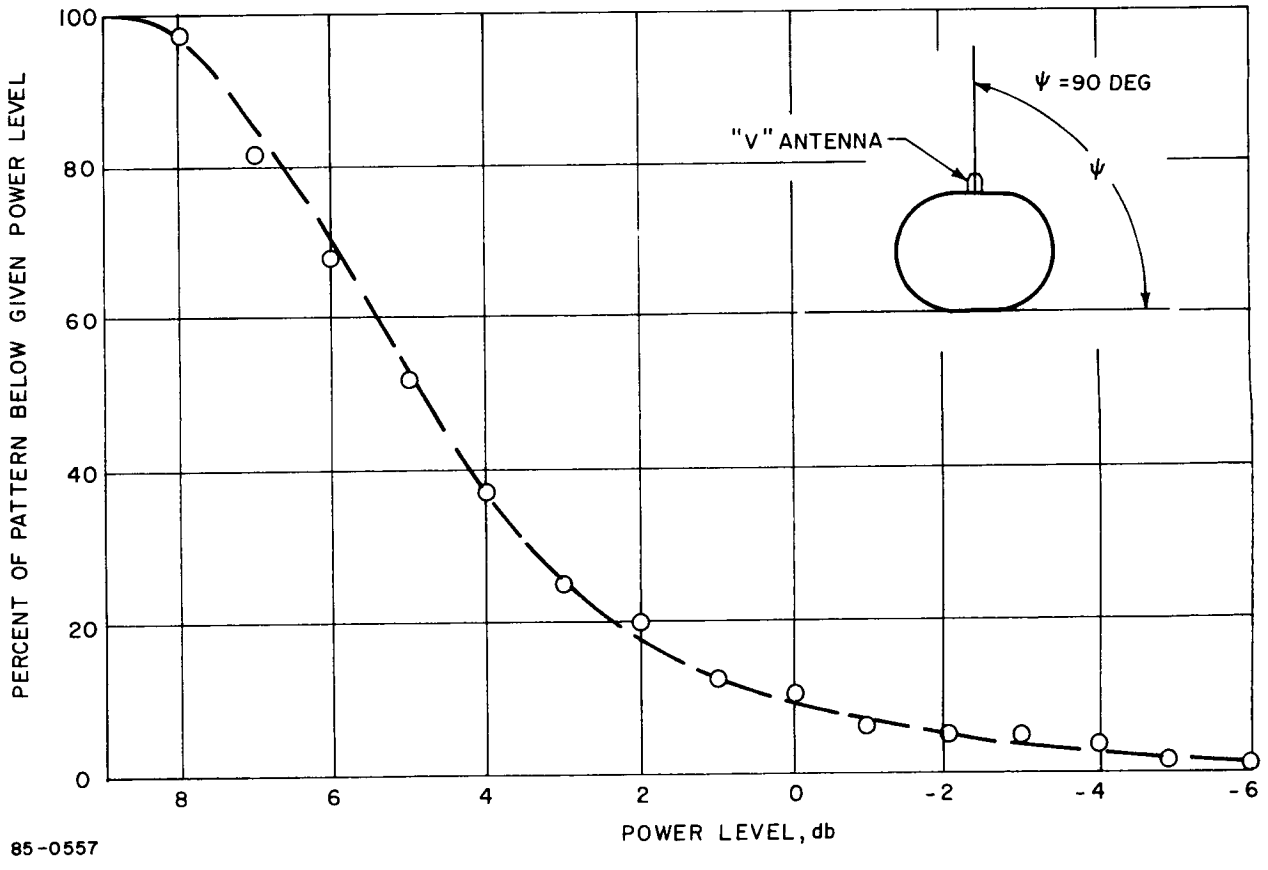


Figure 71 GAIN DISTRIBUTION FUNCTION S-BAND "V" ANTENNA

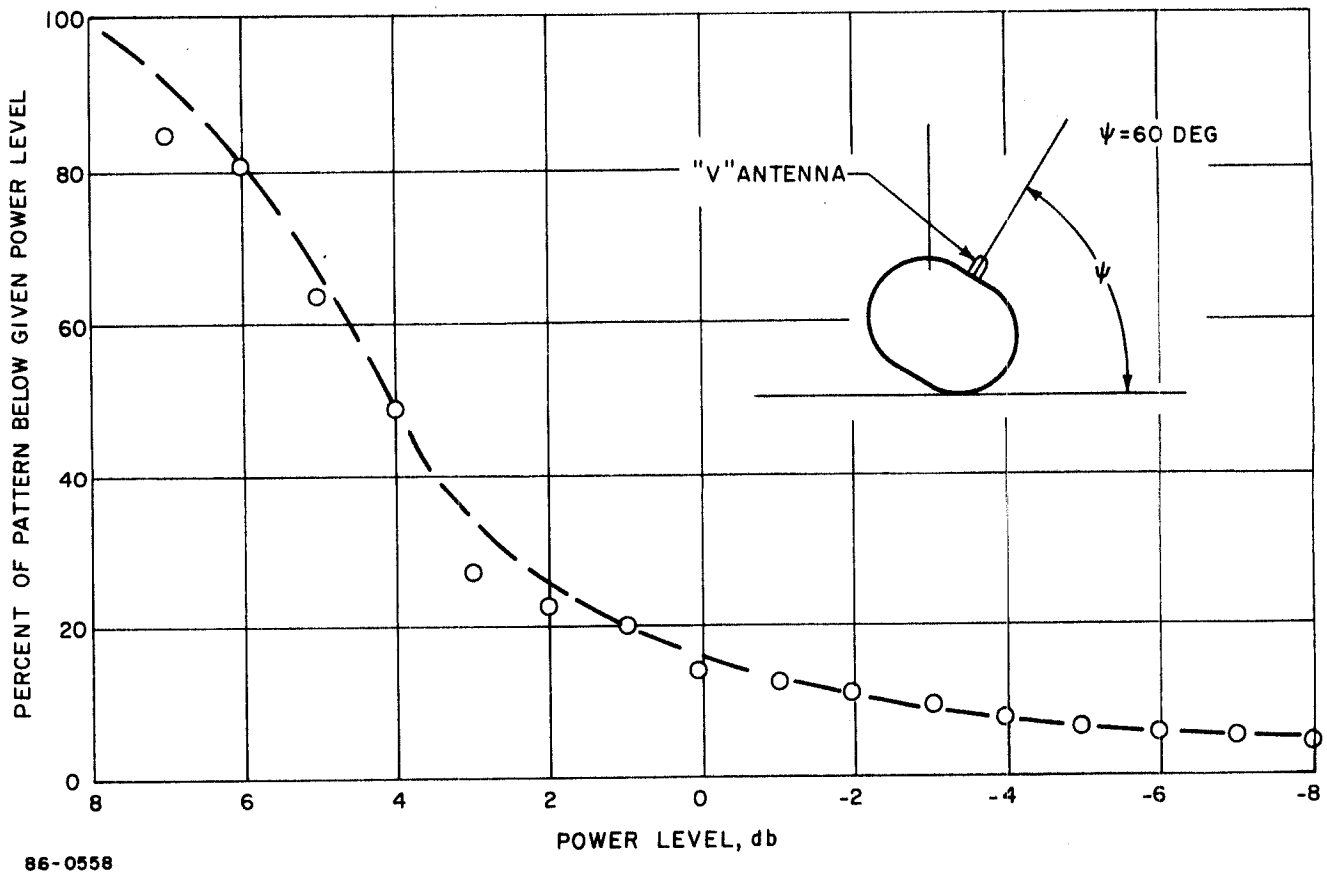


Figure 72 GAIN DISTRIBUTION FUNCTION S-BAND "V" ANTENNA

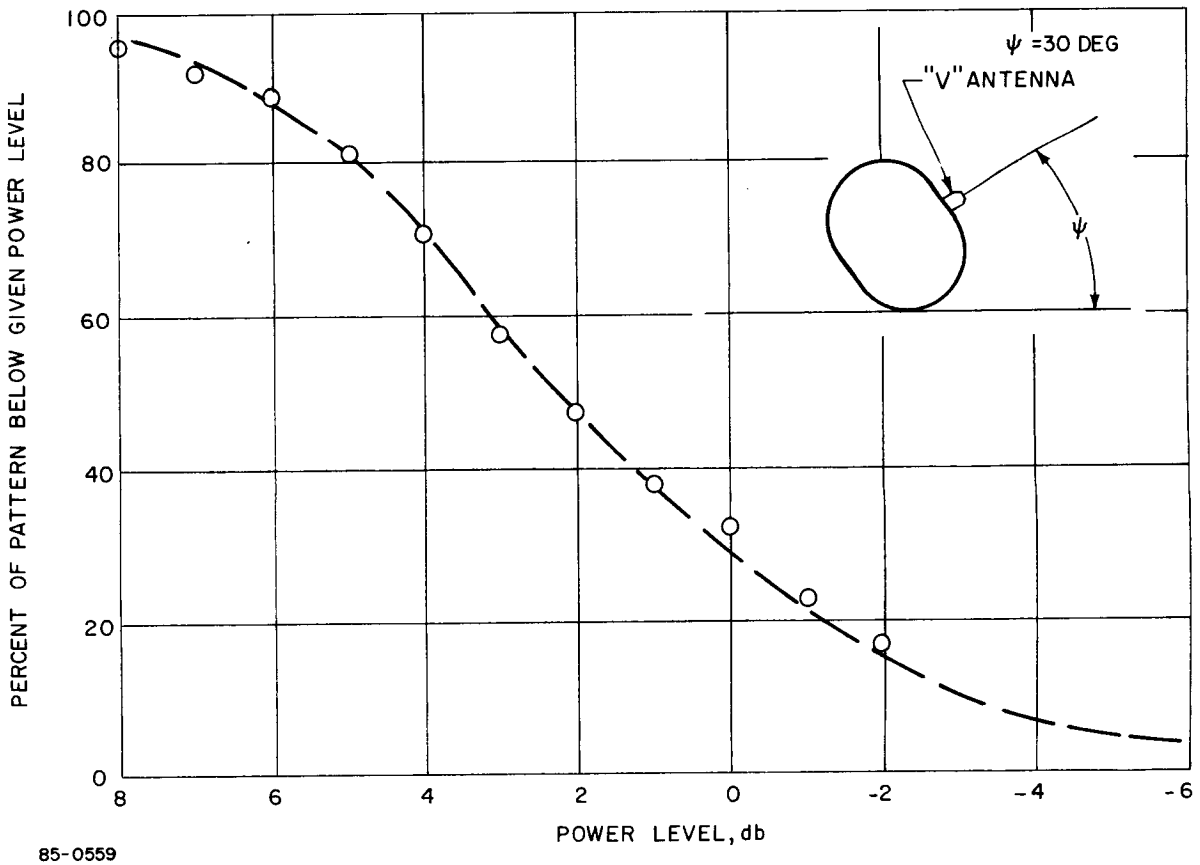
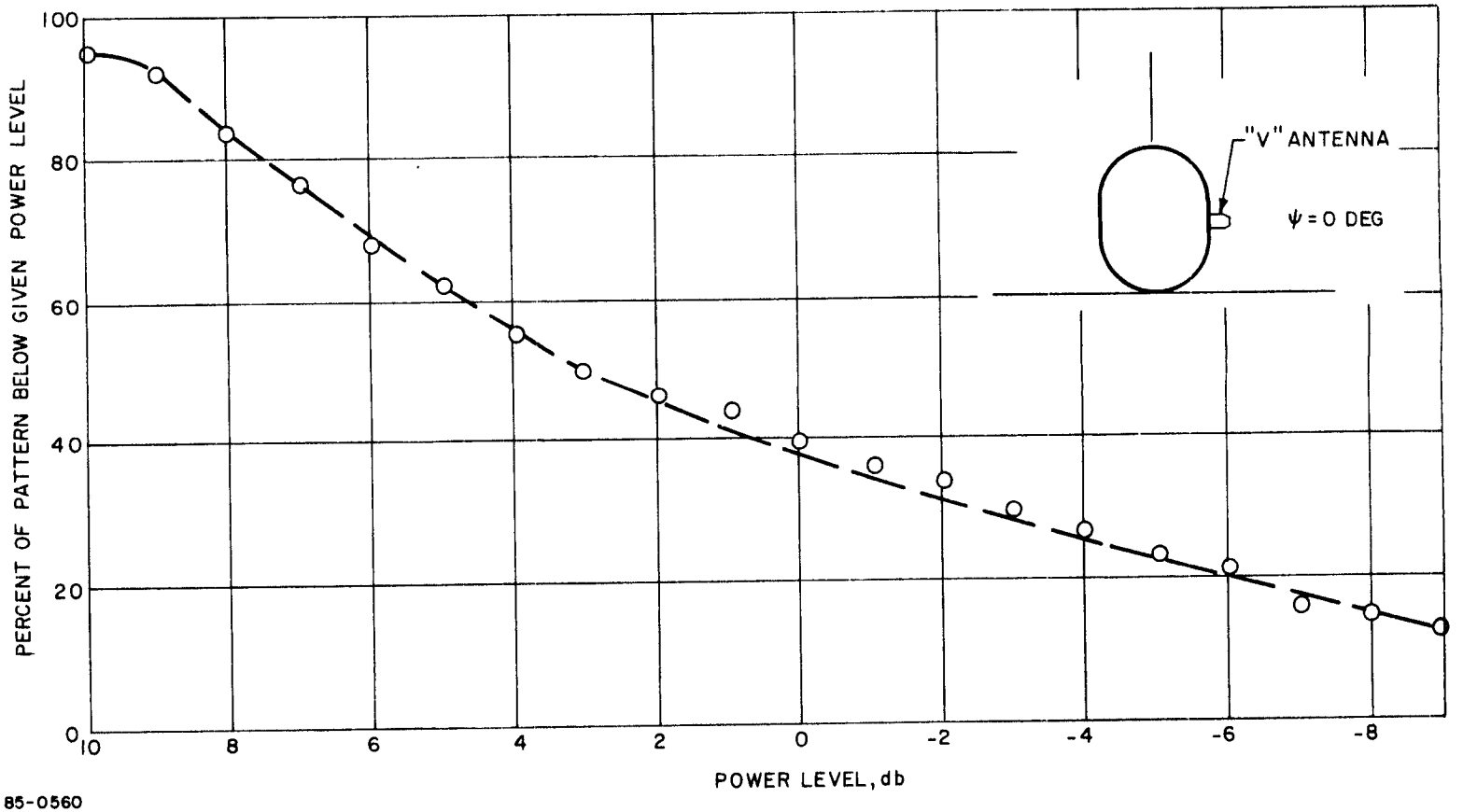


Figure 73 GAIN DISTRIBUTION FUNCTION S-BAND "V" ANTENNA



85-0560

Figure 74 GAIN DISTRIBUTION FUNCTION S-BAND "V" ANTENNA

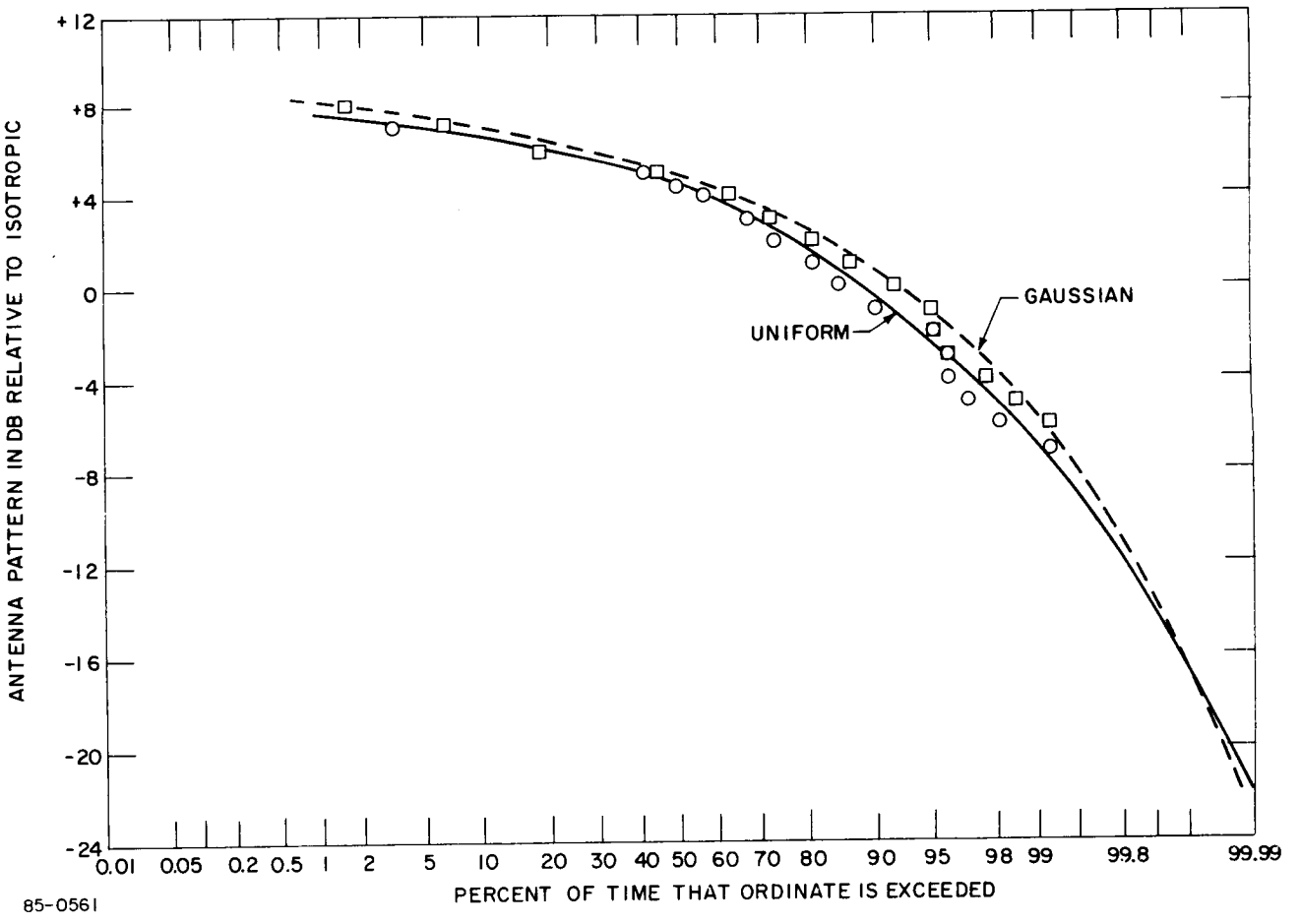


Figure 75 DIRECT-LINK ANTENNA GAIN FOR UNIFORM AND GAUSSIAN DISTRIBUTED LANDED CAPSULE ATTITUDES

presents the resulting gain distribution, based on the assumption that the landing attitude is a random variable with a uniform distribution (a very pessimistic assumption), as well as similar gain information based on the assumption that the distribution is Gaussian with a mean angle of 0 degree relative to the horizontal and a standard deviation of 30 degrees.

If above-threshold performance during more than 99 percent of the time is assumed to constitute an acceptable operational condition, then a worst-case antenna gain of -7 db can be assigned for a uniform distribution of -5.9 db for Gaussian distribution.

This gain results in a worst case power-gain product of +6 db and leads to selection of the N-ary modulation as the recommended technique.

6.3.3.7 Comparison of Alternatives

Table XLIX, compiled by combining the results discussed in the previous paragraphs provides a matrix of alternatives from which a clear picture of the interaction of the various factors which influence performance capability of the direct link in the postlanding phase can be obtained.

6.3.4 Oblate Spheroid Landed Capsule Synthesis

6.3.4.1 Recommended Landed Capsule Synthesis

The most conservative approach to the synthesis of the oblate spheroid landed capsule was the selection of the proven modulation technique (PSK/PM) and the lower transmitter power (20 watts). Unfortunately these selections result in a data rate of zero. Therefore, the newer modulation technique (Non coherent N-ary) was selected as the lowest risk alternative. This selection results in a data rate of 2 bits per second. Figure 76 presents the direct link Mars to Earth look angles as a function of time from landing. Since it is desirable, as a means of minimizing potential multi-path transmission problems, to limit communications to periods when the Earth is high in the Martian sky (for example above 60 degrees, which yields look angles of 30 degrees), communication is possible for a total of 3 hours per day. At the data rate selected, this represents a landed capsule mission bit-content of 43000 bits for a 24-hour mission during which two Earth contacts, one just after impact and one a day later, would be made.

To be consistent with the general guideline that a functionally redundant means of satisfying the mission objectives be provided, it would be

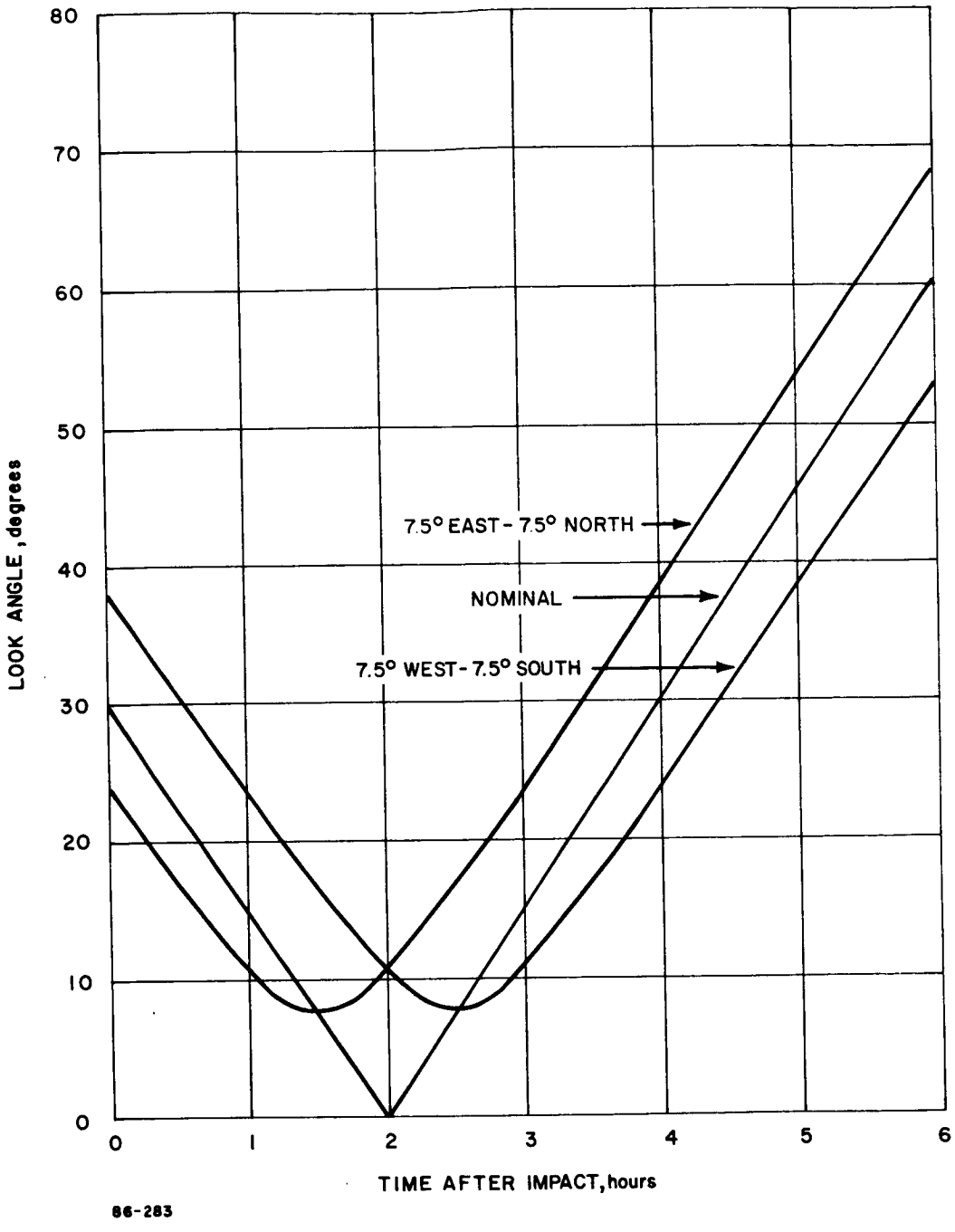


Figure 76 FLIGHT CAPSULE LOOK ANGLE

TABLE IV. XLIX

COMPARISON OF DIRECT LINK PERFORMANCE

Payload	Binary Coherent PSK-PM Modulation		Nary (32 Level)		Remarks
	Transmitter Power 20 watts	Transmitter Power 40 watts	Transmitter Power 20 watts	Transmitter Power 40 watts	
Flotation Sphere	8*	16*	32	64	Superior from communications viewpoint
Oblate Spheroid uniform distribution	0	0	2	4	Landed slope distribution pessimistic very conservative
Oblate Spheroid gaussian distribution	0	0	4	8	Landed slope distribution probably realistic

*A margin of 3.5 db has been applied to the system to permit operation in the linear portion of the curve.

desirable to include the following backup capability:

1. Critical entry and descent data should be selected and stored for transmission over the direct link after impact.

2. The relay link should be included to provide backup means of communication in support of the direct link during the early party of the postlanding mission. The relay link can be used during that period to transmit impact and surface data to the Flight Spacecraft before the orbit injection maneuver is initiated.

Between 3000 and 12000 bits are needed to satisfy the requirement for transmitting critical entry data to Earth. This leaves between 31,000 and 40,000 bits for the post-impact mission. The exact make up of this critical entry data has not been established; a detailed evaluation of the requirements of each experiment must be made to establish the proper balance and to assure a maximum return from the entry experiments in the event of failure of the relay link.

The scientific instrumentation necessary to satisfy the post-impact scientific mission objectives is shown in Table X. In addition to the scientific data to be compiled after impact, numerous diagnostic and engineering measurements must be made. These include a complete status check of all subsystems after impact, as well as verification that the impact attenuator was jettisoned and that the various articulated experiments were properly deployed. Periodic measurements of voltages, component temperatures, and voltage-standing wave ratios are also required. The data list for the post-impact phase of the mission is shown in Table L.

The size, weight, and power requirements of the post-impact science experiments and the associated communications and power source (internal payload) are shown on tables LI and LII. The size, weight, and power requirements of the entry and descent science experiments and the associated communications and power source (external payload) are included on that table for completeness.

Two sets of antennas and two units of each scientific instrument which requires a particular direction of deployment have been included in the landed package. The concept of block redundancy has been implemented in the area of communication equipment which is vital to the success of the mission. This equipment includes such items as the command receivers and decoders, and the direct-link radio frequency (RF) exciters and power amplifiers.

TABLE I

POST IMPACT DATA - RECOMMENDED PAYLOAD

Measurement	Sampling Rate	Bits Per Sample	Total Bits Acquired
Alpha scatter	5 samples	1000	5000
Force anemometer	20 per hour	7	3360
Particle detector	20 per hour	7	3360
Gas chromatograph	5 samples	56	280
Cosmic radiation	1 per hour	14	350
Surface radiation	1 per hour	28	700
Pressure transducer (2)	6 per hour	7	2016
Temperature transducer (2)	1 per hour	7	2016
Atmospheric density	1 per hour	21	525
Water detector	1 per hour	7	175
Atmospheric temperature (2)	6 per hour	7	1008
Impact accelerometer (3)	1 time	4000	4000
Penetrometer	1 time	4000	4000
Hot-wire anemometer	20 per hour	7	3360
Engineering and diagnostic and diagnostic	79 per hour of data acquisition 96 per hour of transmission		5431
Identification synch. and spare	15 percent		5311
Total			40,892

TABLE LI

WEIGHT, VOLUME AND POWER REQUIREMENTS OF SCIENCE PAYLOAD

	Weight (pounds)	Volume (in^3)	Power Consumption (watts)
Science Payload External			
Trapped radiation (1)	2.2	100	0.8
Pressure (2)	0.6	6	0.3
Temperature (2)	0.6	6	0.2
RF probe (2)	1.3	2	0.3
Acoustic	3.0	49	4.0
Mass spectrometer	8.0	400	10.0
Gas chromatograph	4.8	200	4.0
Radar altimeter	8.0	100	10.0
Radiometer	2.2	7	1.5
Beta scatter	0.8	15	0.2
Cables	5.0	---	---
Brackets	3.9	---	---
total	40.4	900	33.3
Science Payload Internal			
*Accelerometer (3)	1.8	15	2.0
Impact accelerometer (3)	1.0	15	2.0
Gas chromatograph	4.8	200	4.0
Water detector	0.5	16	1.0
Pressure (2)	0.6	6	0.3
Atmospheric temperature (2)	0.6	6	0.2
Acoustic	3.0	49	4.0
Cosmic radiation	4.4	200	0.4
Surface radiation	4.0	200	0.4
Hot-wire anemometer	1.0	8	0.5
Force anemometer	4.0	28	1.0
Microphone	1.0	66	0.4
Penetrometer	2.0	42	0.9
Surface temperature (2)	1.2	12	0.2
Alpha scatter	1.4	23	1.4
Cables	19.0	---	---
Brackets	7.4	---	---
total	57.7	886	18.4
Total Science	98.1	1731	

*Accelerometers are part of the descent and entry payload. However, they are contained inside the landed capsule since they must be near the entry vehicle center of gravity.

TABLE LII

WEIGHT, VOLUME AND POWER REQUIREMENTS OF TELECOMMUNICATIONS

	Weight (pounds)	Volume in^3	Power Consumption (watts)
Telecommunications External			
Battery**	37.5	500	N/A
Data handling and telemetry	5.0	160	1
Cabling	5.0	---	0
Power conditioning	2.0	30	5
Brackets and mounting hardware	1.0	---	0
Relay antenna	2.5	4800	---
Relay transmitter (1)	3.0	70	75.5
Miscellaneous RF relay	2.0	80	---
Command receiver antenna	0.1	---	---
total	58.1	5640	81.5
Telecommunications Internal			
Battery	130	2170	N/A
Direct antenna (2)	2.5	45	0
Relay antenna (2)	5.0	9600	0
Direct transmitter (2)	12.0	300	100*
Miscellaneous RF hardware direct	5.0	260	0
RF power supply direct	3.5	60	11*
Relay transmitter (1)	3.0	70	755*
Miscellaneous RF components relay	2.0	180	0
Command revr/decoder	5.0	115	3
Central control and sequencer	9.6	285	3
Data automation equipment	5.5	180	10
Telemetry system	3.0	80	5
Storage	2.0	100	2
Cables	22.0	100	0
Power conditioning	10.9	---	12
Brackets and mounting hardware	9.7	170	0
Diagnostic monitors	4.0	120	0
Engineering data transducers	4.0	200	0.5
total	238.7	14035	101/146.5*

*Both transmitters not on at same time except for preseparation check.

**The battery weight shown includes 10.5 pounds for status and calibration of the over-all payload prior to impact.

An additional weight of battery has been added to the external payload. This has been incorporated to allow a status check and to provide means for calibration of the entire flight capsule payload during the period from separation until impact, a period when the system is powered by the external battery.

The salient features of the recommended payload, including the required payload weights, are summarized in Table LIII. The resulting oblate spheroid landed capsule weight is shown in Table LIV to be 928.1 pounds as compared to the availabe capsule weight of 581 pounds.

6.3.4.2 Landed Capsule Weight Adjustment

This incompatibility between the required and available payload weight can also be resolved, at least in part, by reducing the required payload weight. Various approaches are available to reduce the payload weight. The tradeoffs between these approaches must be carefully considered to establish the proper balance.

1. Decrease Total Data Requirement -- The data lists shown on Tables XLVII and L in paragraphs 6.3.2 and 6.3.3 require that a maximum of 58,000 bits of data be taken during entry and descent, and that 41,000 bits be taken after impact. Approximately 20 percent of the descent data (or 12,000 bits) must be transmitted after impact. This brings the total post impact data transmission to 53,000 bits. This data requirement is incompatible with the direct link capability of 43,000 bits for a one day mission. A fairly sizeable reduction in the amount of data to be actually transmitted may be accomplished be the following means:

A more sophisticated central computer and sequencer (CC&S) which will control selected instruments at specific altitudes or times rather than operating them for a large portion of the entry and descent phase, and

The liberal use of threshold detectors, event counters and on board data reduction techniques in both the pre-and post-impact phases.

Table LV shows the reductions in sampling rates and times which are possible with careful programming. The table also shows the total bits acquired during the shortest duration trajectory (γ_e = 90 degrees, V_e = 23,480 ft/sec., Model 3 atmosphere), the longest duration entry (γ_e = 42 degrees, V_e = 18,400 ft/sec., Model 2 atmosphere) and the parachute failure mode which allows minimal post-blackout data acquisition and transmission time.

TABLE LIII

RECOMMENDED PAYLOAD
SALIENT CHARACTERISTICS

	Descent and Entry	Post Impact
Communications		
Mode of operation	Relay (64 bits/sec)	Direct (2 bits/sec) Relay Backup (64 bits/sec)
Total data capability	14 to 57,000 bits	43,000 bits
Duration of transmission	3-1/2 to 15 minutes	6 hours
Transmitter power	20 watts	20 watts
Frequency	272 mc	2295 mc data 2115 mc command
Modulation	FSK	Navy MFS
Antenna	Spiral	V-horns
Power		
Type	Ni Cad Battery	Ni Cad Battery
Energy	221 watt-hours	1282 watt-hours
Science	11 instruments	14 instruments
Weight		
Science	40.4 pounds	57.7 pounds
Telecommunications	20.6 pounds	108.7 pounds
Power	37.5 pounds	130.0 pounds
Miscellaneous	4.0 pounds	2.0 pounds
Contingency	25.6 pounds	74.6 pounds
Total	128.1 pounds	373.0 pounds

TABLE LIV

OBLATE SPHEROID LANDED CAPSULE - WEIGHT SUMMARY

	Recommended (weight-pounds)
Landed Capsule	(928.1)
Impact attenuator	394.0
Flight connector and miscellaneous	15.5
Internal Weight	(518.6)
Science	57.7
Telecommunication	108.7
Power	130.0
Miscellaneous	2.0
Contingency	74.6
Thermal control	15.0
Internal structure	130.6

TABLE LV

REVISED ENTRY AND DESCENT DATA REQUIREMENTS

Instrument	Sampling Rate	Bits/ Sample	Acquisition Phase	Model 3 Atmosphere $V_e = 23840'/s$ $\gamma_E = -90°$ → Shortest Trajectory			Failure Mode Trajectory →			Model 2 Atmosphere $V_e = 18400'/s$ $\gamma_E = -42°$ Longest Trajectory		
				Time for Phase	Bits Transmitted During Descent	Bits Transmitted Post Impact	Time for Phase	Bits Acquired	Bits Transmitted	Time for Phase	Bits Transmitted During Descent	Bits Transmitted Post Impact
Trapped Radiation Detector	1/5 min	20	200 min prior to entry	200 min.	800	200	200 min	800	200	200 min	800	200
Accelerometer (3)	1/sec	30	0.1 g to chute open	24.1 sec	723	181	58 sec	1740	1740	65 sec	1950	488
	1/20 sec	30	chute open to impact	205 sec	300	0		0	0	847 sec	1260	0
Pressure Transducer (2)	1/sec (tot)	7 each	0.1 g to impact	229.1 sec	1603	401		406	101	912 sec	6384	1596
Temperature Sensor (2)	1/sec (tot)	7 each	0.1 g to impact	229.1 sec	1603	401		406		912 sec	6384	1596
Gas Chromatograph	5 Times	56	5 Timespost blackout	-	280	280		56	56	-	280	280
Mass Spectrometer	5 Times	42	5 Timespost blackout	-	210	210		42	42	-	210	210
Radar Altimeter	1/sec	7	0.1 g to impact	229.1 sec	1603	401		406	101	912 sec	6384	1596
Atmospheric Density (Acoustic)	5 Times	21	5 times post blackout	-	105	105		21	21	-	105	105
Beta Scatter	1/4 per sec	7	chute open to impact	205 sec	287	72		0	0	847 sec	5929	1482
Radiometer	1/sec	42	0.1 g to chute open	24.1 sec	1008	252		2436	609	65 sec	2730	682
RF Probe (1)	1/sec	7	0.1 g to chute open	24.1 sec	168	42		203	50	65 sec	228	57
Engineering & diagnostic -	1/20 sec	24		229.1 sec	264	0		48	12	912 sec	1080	0
"	1/60 sec	37		229.1 sec	111	0		0	0	912 sec	555	0
"	1/300 sec	14			0	0		0	0	912 sec	42	0
"	1/5 sec	32		229.1 sec	1472	0	58	352	88	912 sec	5824	0
Identification Synchronization and Spares					1521	365		959	375		5869	1213
Total					12058	2910		7875	3395		46014	9505
Max bits which can be transmitted from 0.1 g to impact less blackout time				220.3 sec	14099		51.2 sec		3276	890 sec	56960	

The number of bits contributed by each instrument throughout the post landing phase is shown in Table LVI.

In summary, the total data requirement previously used for system synthesis of this volume may be reduced from 58,000 to 45,000 bits during entry and descent and from 41,000 to 23,800 bits post landing. Any further reduction in sampling rate would not produce a significant gain in weight reduction and would begin to seriously curtail the scientific return of the 1971 mission. Such reductions would require detailed evaluation of the individual experiment concerned to determine the impact on the overall mission.

All subsequent methods of reducing payload weight discussed in this section are based on the assumption that the total post impact data requirement has been decreased to that shown on Table LVI.

By reducing the quantity of data which must be transmitted to earth via the direct link, the weight of the battery carried in the landed capsule can be reduced from 130 pounds to 84 pounds.

2. Decrease Scientific Payload -- The payload weight can be reduced if some of the scientific instruments are deleted from the design. The weight savings include the weight of the instrument itself, as well as its cabling, brackets and mounting hardware, the weight of battery required to power it and the weight of battery necessary to transmit the data which it generates.

Table LVII shows the required payload weight as a function of the number of instruments carried. If all the instruments are flown and the decreased total data requirement discussed in the previous section is used, the total weight of the payload, not including the contingency factor is 354.9 pounds.

If the trapped radiation detector,its brackets, cables and the weight of battery to power it and transmit the data generated by it are removed, the resulting payload weight is 347.2 pounds.

If the RF probe and associated equipment is also deleted, the weight will be 345.0 pounds. This reduction is carried out in accordance with the priority of the instrument until all of the instruments have been deleted except for the accelerometer and force anemometer. This results in a payload weight of 186.8 pounds.

This table does not account for the fact that each item of internal payload required some structure and a volume of impact attenuator material. A pound added to the internal payload results in an increase of 0.44 pounds of structure and over 1.2 pound of impact attenuator.

TABLE LVI

DATA BITS - SCIENCE INSTRUMENTS

POST IMPACT PHASE

Transducer	Outputs	Sampling Rate	Bits Sample	Total Bits Acquired
Impact Accelerometer (3)	30	1 sample	7	2100
Penetrometer (3)	30	1 sample	7	2100
Alpha Scatter	200	5 samples	5	5000
Cosmic Radiation	1	1/hr	7	168
Gas Chromatograph	8	5 samples	7	280
Hot Wire Anemometer	6	5 samples	7	210
Force Anemometer	6	5 samples	7	210
Microphone	2	5 samples	7	70
Surface Radiation	1	1/hr	7	168
Pressure Transducer (2)	2	2/hr	7	672
Temperature Transducer (2)	2	1/hr	7	336
Acoustic Transducer	3	1/hr	7	501
Water Detector	1	1/hr	7	168
Atmospheric Temperature	2	2/hr	7	672
Engineering Diognostics				2303
Synchronization, Identification and Spaces				2252
		Total Bits		17,210

TABLE LVII

WEIGHT FOR DIFFERENT SCIENCE PAYLOADS

1. Telecommunications with accelerometer and anemometer	186.8 lbs	13. Gas chromatograph	270.2 lbs
2. Radar altimeter	201.7 lbs	14. Surface temperature	275.3 lbs
3. Pressure	207.8 lbs	15. Cosmic radiation	282.3 lbs
4. Temperature	213.8 lbs	16. Surface radiation	288.9 lbs
5. Gas chromatograph	220.6 lbs	17. Alpha scatter	307.0 lbs
6. Atmospheric density	225.3 lbs	18. Beta scatter	312.9 lbs
7. Particle detector	230.6 lbs	19. Atmospheric density (acoustic)	322.6 lbs
8. Water detector	234.0 lbs	20. Penetrometer	331.4 lbs
9. Radiometer	239.3 lbs	21. Impact accelerometer	339.7 lbs
10. Mass spectrometer	250.2 lbs	22. Hot Wire Anemometer	345.0 lbs
11. Atmospheric temperature	255.7 lbs	23. Radio frequency probe	347.2 lbs
12. Atmospheric pressure	261.8 lbs	24. Trapped radiation detector	354.9 lbs

3. External versus Internal Payload -- To minimize the size and weight of landed capsule impact attenuator and structure, all components which are not required after impact, can be carried in the external payload. This includes the battery which will power the entire system from preseparation to impact. Also, the data conditioning equipment and power booster regulator can be carried in the external payload. Each pound of instrumentation removed from the internal payload results in a 2.97 pound reduction in landed capsule weight. This factor includes the weight of structure and impact attenuator no longer required.

4. Delete Landed Relay Link -- Additional savings of internal payload can be accomplished by the deletion of the post-impact relay transmission link; this is associated with the minimum spacecraft support option discussed in paragraph 6.3.2. Entry and descent data would be handled by the external relay link; the resulting penalty is the loss of a redundant transmission mode for entry and landed science data. The savings would be 10 pounds of internal telecommunications payload. However, since the resonant cavity for the VHF antenna is 10.5 inches deep and has an average diamter of almost 18 inches, the relay antenna represents a significant portion of the payload volume. Two VHF antennas are carried each of which require 27 percent of the total volume within the landed capsule. If the VHF antennas were removed, the size of the landed capsule including its impact attenuator would be significantly reduced resulting in a savings in entry weight of approximately 130 pounds.

In summary, a reduced total data requirement shown in Tables LV and LVI, achieved by a combination of more complex programming and the liberal use of onboard data reduction equipment, results in a savings of 46 pounds of internal battery weight. The reduction in battery weight is further amplified by the reduction in the impact attenuator and support structure which result. The net weight savings is 137 pounds.

The elimination of the following instruments would provide a weight savings of 70 pounds in the required landed capsule weight:

a) Penetrometer

b) Impact Accelerometer

c) Trapped Radiation detector

d) Hot-wire anemometer

e) RF probe

Since two of these instruments are included in the external payload, the available landed capsule weight is increased by 10 pounds.

The elimination of the post-impact relay communications subsystem allows a total weight savings of 130 pounds. The weight of the oblate spheroid landed capsule before and after these modifications is shown in Table LVIII. The salient characteristics of the resulting payload are shown in Table LIX.

6.3.5 Flotation Sphere Landed Capsule Synthesis

If the flotation sphere landed capsule is used rather than the oblate spheroid landed capsule, a single direct 11 db antenna with a -1 db gain could be used instead of the -7 db spiral antenna. This would make possible a direct-link communication bit rate of 8 bits/second and reduce the energy required to transmit the reduced data list of paragraph 6.3.4.2 from 378 to 92 watt-hours. This reduction would correspond to a 67 pounds reduction in battery weight.

Articulation of those experiments requiring deployment would also be simplified. Only one of each instrument need be carried. This results in a saving of 17 pounds of instruments.

Approximately 10 pounds of telecommunications equipment may also be deleted since only one antenna with its associated RF equipment is required instead of two. Considerably less impact attenuation material is now required for the flotation sphere landed capsule than for the oblate spheroid landed capsule. This weight saving results from the reduction in payload previously mentioned in addition to the increase in internal packaging density realizable. Two hundred and seven pounds less of impact attenuator material is required for the flotation sphere.

A weight summary of the flotation sphere landed capsule is shown in Table LX and a summary of the salient features is shown in Table LXI.

TABLE LVIII

OBLATE SPHEROID LANDED CAPSULE - WEIGHT SUMMARY

	Recommended (weight-pounds)	Adjusted (weight-pounds)
Landed Capsule	(928.1)	(595.0)
Impact Attenuator	394.0	215.0
Flight Connector and Misc.	15.5	15.5
Internal Weight	(518.6)	(364.5)
Science	57.7	48.0
Telecommunication	108.7	98.7
Power	130.0	70.1
Miscellaneous	2.0	2.0
Contingency	74.6	54.7
Thermal Control	15.0	15.0
Internal Structure	130.6	76.0

TABLE LIX

OBLATE SPHEROID LANDED CAPSULE SALIENT CHARACTERISTICS

Communications	Descent and Entry	Postimpact
Mode of operation	Relay (64 bits/sec)	Direct (2 bits/sec)
Total data requirement	37, 115 bits	19, 295 bits
Duration of transmission	9.7 minutes	2.68 hours
Transmitter power	20 watts	20 watts
Frequency	272 mc	2295 mc data 2115 mc command
Modulation	FSK	Nary MFS
Antenna	Spiral	V horns
Power		
Type	Ni Cad Battery	Ni Cad Battery
Energy	184 watt-hours	565 watt-hours
Science	9 instruments	11 instruments
Weight (pounds)		
Science	35.3	48.0
Telecommunications	20.6	98.7
Power	33.0	70.1
Miscellaneous	4.0	2.0
Contingency	23.2	54.7
	Total 116.1 pounds	Total 273.5 pounds

TABLE LX

FLOTATION SPHERE LANDED CAPSULE WEIGHT SUMMARY

	wt - lbs
Landed Capsule	(572.5)
Impact attenuator	186.9
Flight connectors and miscellaneous	15.5
Internal Weight	(370.1)
Science	46.7
Telecommunication	98.7
Power	63.0
Miscellaneous	2.0
Contingency	52.6
Thermal control	15.0
Internal structure	92.1

6.4 LANDED CAPSULE SELECTION

Two landed capsules have been synthesized, conceptual design drawings have been prepared, and the performance capability of these designs has been determined. Each of the landed capsule designs has significant advantages and disadvantages over the other. The primary advantage of the flotation sphere is its markedly superior performance capability, higher bit rate, and an instrument complex that more completely satisfies the mission objectives. However, the oblate spheroid represents the more conservative selection since its operation on the surface is nearly passive, requiring no re-erection prior to its operation. A summary of the major tradeoffs between these two landed capsules is presented below.

Both landed capsule concepts are compatible with the selected entry shell (blunted cone), however, the more compact nature of the flotation sphere requires less suspended capsule structure to support it within the entry shell

TABLE LXI

FLOTATION SPHERE LANDED CAPSULE

SALIENT CHARACTERISTICS

Communications	Descent and Entry	Postimpact
Mode of operation	Relay (64 bits/sec)	Direct (8 bits/sec)
Total data requirement	37,115 bits	19,295 bits
Duration of transmission	9.7 minutes	0.67 hours
Transmitter power	20 watts	20 watts
Frequency	272 mc	2295 mc data 2115 mc command
Modulation	FSK	PSK
Antenna	Spiral	slot
Power		
Type	Ni Cad Battery	Ni Cad Battery
Energy	184 watt-hours	421 watt-hours
Science	11 instruments	14 instruments
Weight (pounds)		
Science	35.3	46.7
Telecommunications	20.6	98.7
Power	33.0	63.0
Miscellaneous	4.0	2.0
Contingency	23.2	52.6
	116.1 pounds	263.0 pounds

and during parachute descent. The required center of mass position within the blunted cone is easily satisfied by either landed capsule.

The primary advantage of the flotation sphere lies in its ability to achieve vertical orientation after impact independent of the surface terrain slope. This allows the use of a narrower beam antenna since the pointing direction is well known. A log spiral antenna is used for the relay link and a slot antenna, built into the log spiral antenna, is used for direct-link communications. The relatively narrower beamwidth provides a -1 db direct-link gain over look angles of interest. The resulting power gain product, with a 20 watt transmitter is sufficiently high to allow use of the proven PSK/PM modulation technique at a data rate of 8 bits per second direct to Earth. The knowledge of pointing direction also provides for better VHF relay link performance.

The oblate spheroid provides only general knowledge of post-impact orientation. Either of the two flat sides may be facing up after the capsule comes to rest. Two S-Band direct-link antennas are therefore required with the one facing upward selected for use once the capsule attitude is determined. This capsule is sensitive to the terrain slope, its final orientation could be as much as 45 degrees from the vertical, and requires a broader antenna beamwidth. A -7 db antenna gain is realizable over the look angles of interest. The resulting power gain product with a 20 watt transmitter forces the use of a noncoherent N-ary modulation technique. A data rate of only 2 bits per second by direct link to Earth can be achieved. No post-impact relay link is provided in the oblate spheroid design since there is insufficient weight available for its inclusion. The obvious advantage of such a link if the direct link should fail cannot be provided in this design.

The flotation sphere, being more compact and not so severely weight limited as the oblate spheroid, includes a more complete complement of instruments. Five additional instruments have been included in the blunted cone flotation sphere design. Two external instruments, the trapped radiation detector and the RF probe for use during the preentry, and entry mission phases and three internal instruments, the penetrometer, impact accelerometer, and the hot-wire anemometer for surface measurements have been included.

The deployment of surface instruments which must sample the atmosphere is difficult for both designs, since at least a section of the protective crushable material must be deployed. The oblate spheroid deploys a section of crushable material around its maximum diameter and deploys instruments through the edge. The flotation sphere, since the relative orientation of the inner and outer spheres is completely random, must deploy the entire shell of crushable material. In addition, the outer flotation shell around the payload must be jettisoned to allow instrument deployment. The orientation of the deployed instruments is well known for the flotation sphere. However, the oblate spheroid must carry two of each deployable instrument to insure that

one such instrument is properly oriented. The weight of the post-impact scientific instrument package for the oblate sphere is higher even though it contains three fewer instrument types.

Thermal control for the flotation sphere is a more complex problem than for the oblate spheroid. The oblate spheroid can be provided with a low emmissivity thermal-control coating over the entire capsule except the region directly over the antennas to limit the heat loss during the Martian night. The flotation sphere, however, cannot be so coated, since the region of the outer flotation sphere which will be over the antennas is not a known apriori. The metallic thermal control coating could inhibit communications if it happened to cover the antennas. For the flotation sphere, additional batteries must be provided to heat the capsule during the Martian night.

Both capsules employ crushable material impact attenuators which limit the impact loads to 500 Earth g. The oblate spheroid, because of its larger surface area to internal volume ratio, and the lower internal packaging density which can be achieved (2 slug/ft^3 as opposed to 3 slug/ft^3 for the flotation sphere) requires a significantly heavier impact attenuator. The optimum impact attenuator material density is also lower, in fact so much lower that the practicality of such materials is questionable. This may further increase the weight of the oblate spheroid impact attenuator.

The principle disadvantage of the flotation sphere is its total dependence upon re-erection after impact for successful operation. The oblate spheroid is semi-passive after impact and is therefore a much more attractive operational concept.

Selection between these two landed capsule configurations is therefore not clear cut. The performance advantages of the flotation sphere are attractive for the weight limited design. However, the less complex oblate spheroid was selected as the reference concept as dictated by the overall ground rule of conservatism. Both landed capsule designs have been shown as alternates in Section 3 of this book.

7.0 ATTITUDE CONTROL SYSTEM TRADE-OFFS

7.1 SYSTEM REQUIREMENTS/DESIGN GUIDELINES

The simple requirement of the attitude control system to stabilize the separated vehicle during thrusting is complicated by the rather stringent demands placed upon the control which must be maintained over the thrust application angle, or more precisely, over the direction of the resultant velocity vector. Three factors combine to produce the overall accuracy requirements on the direction of the velocity vector. The first is the need to accelerate the separated vehicle so that it will arrive at the entry point sufficiently in advance of the flight spacecraft to permit communication between the two during the time allocated for playout of entry and descent scientific data. This requirement results from the need to complete this transmission of data before the flight spacecraft begins its orbit injection attitude maneuver. The second factor is the necessity to control the landing point to within a 500 kilometer radius of the desired impact point. The third factor is the requirement to control the entry angle so that it is within limits which are acceptable to the flight capsule mission. Each of these factors will be discussed in turn to develop the requirements which then formed the basis for the tradeoff studies of the ACS.

7.1.1 Communication Lead Time

With the flight spacecraft on a fly-by trajectory, it is necessary to impart a velocity change to the separated vehicle to place it on an impact course. It is also necessary to provide an additional separation velocity between spacecraft and separated vehicle to obtain the necessary communication lead time. This can be done either by speeding up the separated vehicle or by decelerating the spacecraft. If the separated vehicle speedup is selected, this maneuver can be combined with the velocity change required to place it on an impact trajectory.

The acceleration of the separated vehicle to achieve lead time not only increases the propulsion required but also has a profound influence on the dispersion in entry angle and impact point which result and consequently on the pointing accuracy required of the attitude control system.

7.1.2 Landing Point and Entry Angle Dispersion

It is a design requirement of the system that the dispersion in impact point be no more than 500 km (3 sigma). This corresponds to a variation in entry angle of approximately 7.5 degrees (3 sigma); this tolerance is essentially constant over the range of trajectory parameters considered. Furthermore, the maximum entry angle must be less than -52 degrees to

satisfy the parachute deployment conditions with the selected M/C_DA of 0.2 slug/ft^2. In brief, the attitude control system is required to control the direction of the velocity increment to maintain a tolerance of 7.5 degrees (3 sigma) in entry angle for a nominal entry angle of 44 degrees, while at the same time providing the lead time necessary to meet communication requirements.

7.1.3 Trajectory Analysis

Detailed analyses have been carried out to show which trajectory parameters are important in determining the thrust pointing accuracy required, and the results are presented in Sections 3.0 and 4.0, Volume V, Book 1. These analyses show that entry angle dispersion is virtually independent of separation range and periapsis altitude, but the dispersion increases rapidly with increasing lead time, with increasing approach velocities, and with shallower entry angles. The hyperbolic approach velocity can be minimized by proper choice of launch period. The effect of lead time and entry angle on ACS pointing accuracy requirements can be seen in Figure 77. This figure shows the allowable error in thrust application angle as a function of lead time for a range of entry angles. The nominal separation conditions are also stated. The data presented is based on an allowable entry angle dispersion of 2.5 degrees (1 sigma), on a position uncertainty at the time of separation of 150 km (1 sigma) and an uncertainty in velocity increment of one percent (1 sigma). (The position uncertainty is due principally to uncertainty in the ephemeris of Mars. This is a very significant contributor to entry angle dispersion and if reduced by improved ephemeris data can greatly ease the ACS pointing accuracy requirements.) These are the only important error sources affecting entry dispersion other than pointing accuracy and are the expected nominal values.

From this figure it can be seen that to achieve a lead time of three hours the allowable error in thrust application angle is 0.38 degrees (1 sigma) for an entry angle of -50 degrees and 0.25 degrees (1 sigma) for an entry angle of -40 degrees, and at -30 degrees entry angle the desired dispersion is unachievable even with a perfect ACS. On the other hand if lead time is reduced, or achieved by flight spacecraft slowdown, the pointing accuracy requirement of the ACS is drastically relaxed.

7.1.4 Summary

In summary the following conclusions can be stated:

1. The thin atmosphere requires shallow entry angles which in turn give rise to large entry angle dispersion.

2. The lead time requirement together with the shallow entry angles impose severe constraints on the ACS accuracy.

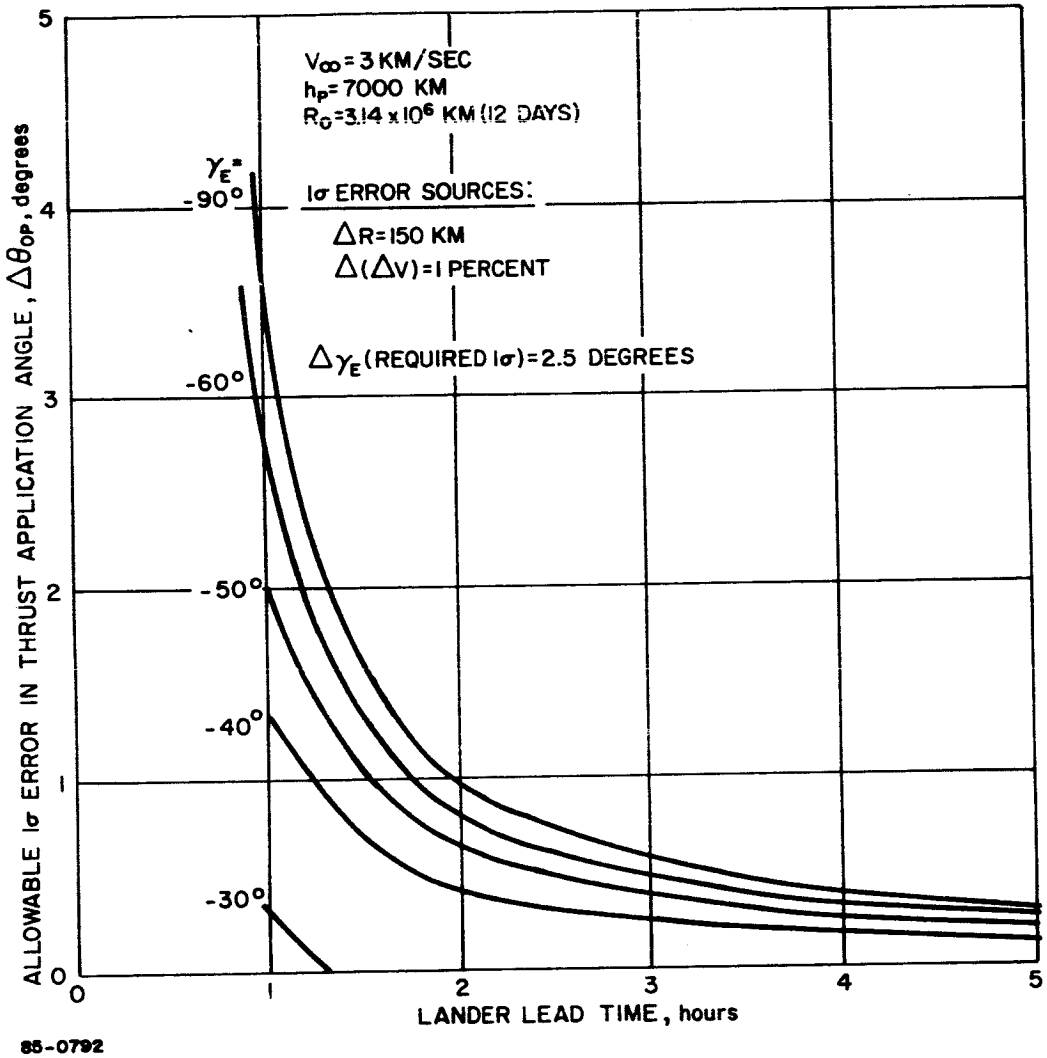


Figure 77 ALLOWABLE THRUST APPLICATION ANGLE ERROR

3. The ACS requirements can be relaxed by the use of flight spacecraft slowdown and also by improved ephemeris data.

7.2 CANDIDATE TECHNIQUES BY MISSION PHASE

The possible approaches for the design of the ACS can be conveniently discussed by examining the operation required during each phase of the mission and considering the candidate techniques which are suitable.

7.2.1 Orient Separated Vehicle to Thrust Attitude

The separated vehicle can be placed in the proper orientation for thrusting either by maneuvering the flight spacecraft prior to separation or by maneuvering the flight spacecraft prior to separation or by maneuvering the separated vehicle after separation by means of an active control system.

7.2.2 Maintain Attitude During Thrust

The attitude of the separated vehicle can be maintained during thrusting by spin stabilization, by means of an active ACS, or by gimballing the engine.

7.2.3 Reorient for Proper Entry Attitude

The entry vehicle dynamic performance during entry will be improved if the angle of attack at entry is zero. If an active ACS is used, it can perform the reorientation. If spin stabilization alone is used, no reorientation is possible, but at least the spin rate should be reduced before entry so that angle-of-attack convergence is not hindered.

7.2.4 Maintain Attitude During Cruise

The attitude of the entry vehicle can be maintained until entry by spin stabilization or by an active ACS.

7.3 COMPARISON OF CANDIDATE SYSTEMS

From the above techniques, possible candidate systems can be synthesized, then compared and evaluated in terms of their performance, reliability and complexity, weight, flexibility, growth potential, and their effect on other systems.

7.3.1 Spin Only System

This system is oriented to the thrusting attitude by the flight spacecraft, is spin stabilized after separation for thrusting and cruise; the spin rate is decreased prior to entry. This is the simplest and lightest system but

requires a maneuver of the flight spacecraft which may be a disadvantage. It does not have the required pointing accuracy if a three hour lead time must be obtained by capsule bus speedup. However its performance is more than adequate with reduced lead times.

7.3.2 Active ACS with Spin

This system uses an active ACS for orientation, to maintain attitude during thrusting, and to orient for entry attitude. Spin stabilization is then employed until entry. Adequate pointing accuracy can be achieved for lead times up to three hours and nominal entry angles of -40 degrees or steeper. It does not require a flight spacecraft maneuver, is flexible and has growth potential, but it is heavier, more complex and less reliable than a spin-only system. It also presents a difficult problem to achieve proper alignment between the spacecraft attitude reference and the gyro reference on the separated vehicle.

7.3.3 Active ACS

This system is the same as that previously described except that it uses no spin stabilization and maintains cruise attitude by reaction control. Due to the long time during cruise, the total impulse required becomes excessive and attitude accuracy due to gyro drift becomes intolerable.

7.3.4 Active ACS with Engine Gimbal and Spin

This is the same as the system described in paragraph 7.3.2 except that engine gimbaling is used during thrusting. Since the thrust level and total impulse of the ΔV engine are moderate, the reaction control system can readily be sized to provide thrust vector control. Consequently, no advantage is obtained by gimbaling or other auxiliary means of thrust vector control.

7.4 SELECTED APPROACH

The approach which has been selected is the active ACS with spin, and is described in paragraph 4.6 of this book and in greater detail in Volume V Book 5. This choice was dictated by the stringent pointing accuracy requirements imposed by a lead time of three hours together with the desire to avoid the complication of flight space orientation and slowdown maneuvers. If these maneuvers should be permitted, the spin only system would be the preferred choice.

7.5 TERMINAL GUIDANCE

From the discussion in paragraph 7.1 it should be clear that even the performance of the closed-loop active ACS is marginal in meeting the accuracy

requirements. If a communication lead time of at least three hours must be obtained by separated vehicle speed up and if no improvement in Mars ephemeris data is obtained, then the active ACS will not permit entry angles shallower than about -40 degrees in order to meet dispersion requirements. If shallower angles become a requirement, it may be necessary to employ terminal guidance on the entry vehicle. Such a system would include, in addition to the ACS, a precision planet tracker, sun tracker, computer, and possibly a star tracker to make a navigation fix at a range of about 30,000 km from Mars. A velocity correction would then be performed using the ACS and propulsion system to correct to the desired trajectory. The additional weight of such a system would be about 100 pounds plus the weight of additional propellant required which amounts to 100-200 pounds, depending on the range from the planet at which the velocity correction is made. A terminal guidance system would greatly ease the demands on ACS accuracy for thrusting after separation, and may in fact be strongly desirable for missions in which better control of impact point location is required. In this sense it is an attractive approach since it provides growth potential for more ambitious future missions. However, it must be regarded as a less attractive choice for the present mission than reducing the entry angle dispersions at the time of separation. These dispersions will be reduced by updating the Martian position data from Mariner IV information and reducing the separation event errors. The most effective method of reducing the separation event errors is to provide the required lead time by slowing down the flight spacecraft instead of speeding up the separated vehicle. This allows a thrust application angle of nearly ninety degrees and the required accuracy of the vehicle attitude during thrust application is dramatically reduced. By reducing the separation event errors in this manner and updating the Martian orbit data, the entry angle dispersions can be reduced to less than 1 degree. The need for a terminal maneuver is then eliminated. Furthermore, the reduced accuracy requirement for the separated vehicle attitude control allows the use of the simple spin system and eliminates the alignment mechanization problem.

7.6 FLIGHT SPACECRAFT SEPARATION MANEUVER

An important option in the flight sequence which has a direct impact on the design of the ACS is whether or not to perform a flight spacecraft preseparation maneuver designed to provide the required separated vehicle thrust application angle. Such a maneuver is required if a simple spin system is used for stabilization while thrusting. If the flight spacecraft maneuver is not used, the ACS must perform a vehicle orientation maneuver after separation. In this case an active attitude control system is required. It has been shown previously that a spin system cannot meet the dispersion requirements if lead time must be obtained by separated vehicle speed up. Consequently the spacecraft preseparation maneuver is only beneficial to the design of the ACS if it is also possible to obtain communications lead time by slowing down the spacecraft. In this case the simple spin system can be used.

7.7 DESPIN

The influence of spin rate upon the vehicle performance and subsequently the environment (loads and heating) is reflected through the higher angle of attack history associated with increasing spin rate. Since all shapes considered are characterized by reduced drag at angle of attack, the result is an increase in the effective M/C_DA as spin is increased. It is to be expected then, that as the spin rate increases the heating and loads will also increase. In addition, the altitude at which parachute deployment can be initiated will decrease. These trends apply independent of the atmosphere, or entry shell shape.

Typical results for the integrated heating at the stagnation point for the blunted cone are shown illustratively in Figure 78 where, initial angle of attack and model atmosphere are presented as parameters. The integrated heating increases at higher spin rates, as expected. The initial angle of attack is seen to be a significant parameter. Local aggravations associated with angle of attack which would result in a higher sensitivity to variation of initial angle of attack and spin rate are not included. Similar results were obtained for the modified Apollo and tension shell. For the tension shell, however, test data indicates a fovorable effect on the distribution with a possible reduction in heating over some portions of the body.

Also of significance is the variation of parachute deployment altitude with spin rate which is shown in Figure 79 and 80. The effect of the damping derivative, C_{m_q}, is seen to be critical. These results are representative of the supersonic or late entry phase of the trajectories. In this case the tension shell which exhibits a limit cycle motion, results in lower altitudes for parachute deployment and is also most sensitive to the spin rate.

The variation in the loads are as expected and shown in Figures 81 and 82. The tension shell is most sensitive to spin rate. The effect of C_{m_q} is not significant here as is to be expected since the initial angle of attack convergence is associated with the dynamic pressure variation C_{m_q} being critical during the supersonic and transonic regime where the dynamic pressure is decreasing.

A despin maneuver is therefore desired during early entry to reduce these effects. The despin should be initiated at the onset of aerodynamic loads to avoid tumbling. The effect of despin is primarily reflected through the superior angle-of-attack convergence.

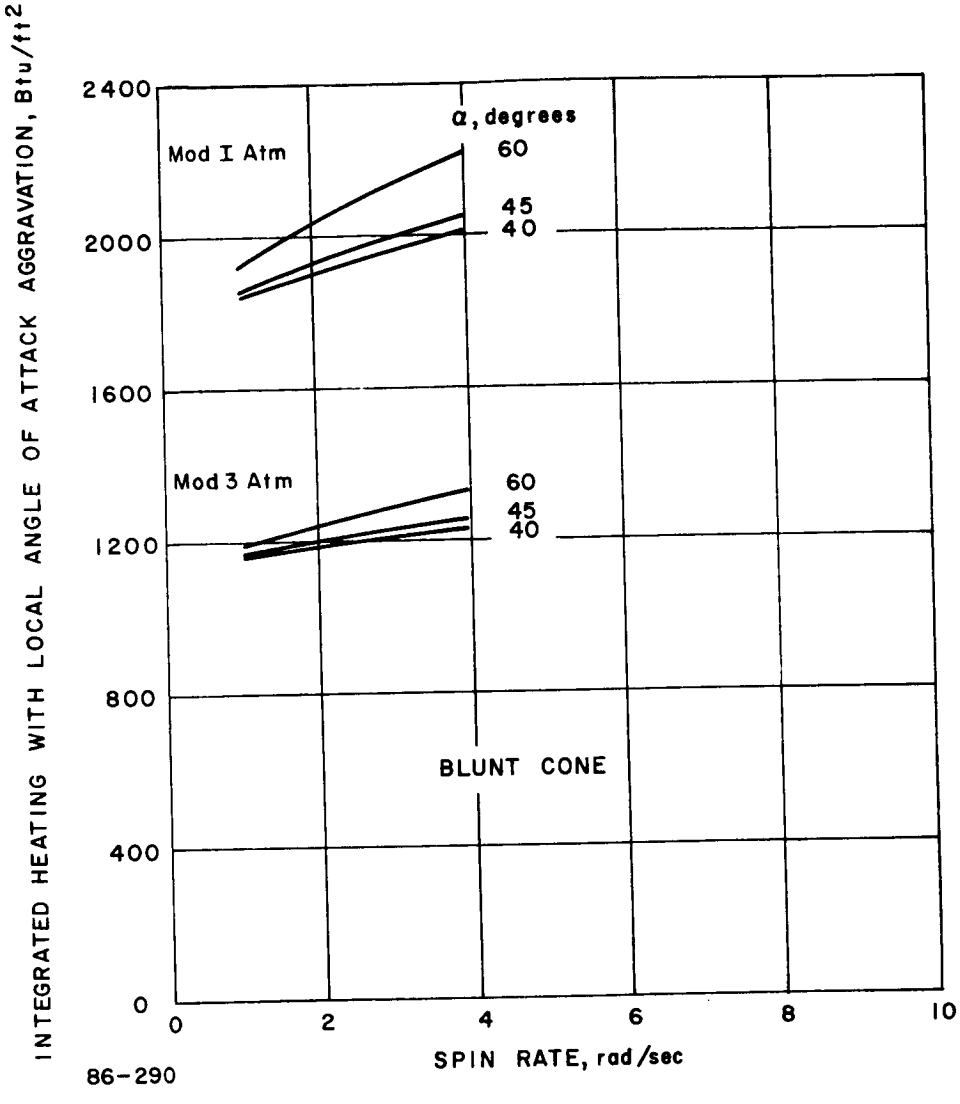


Figure 78 EFFECT OF SPIN RATE ON INTEGRATED HEATING

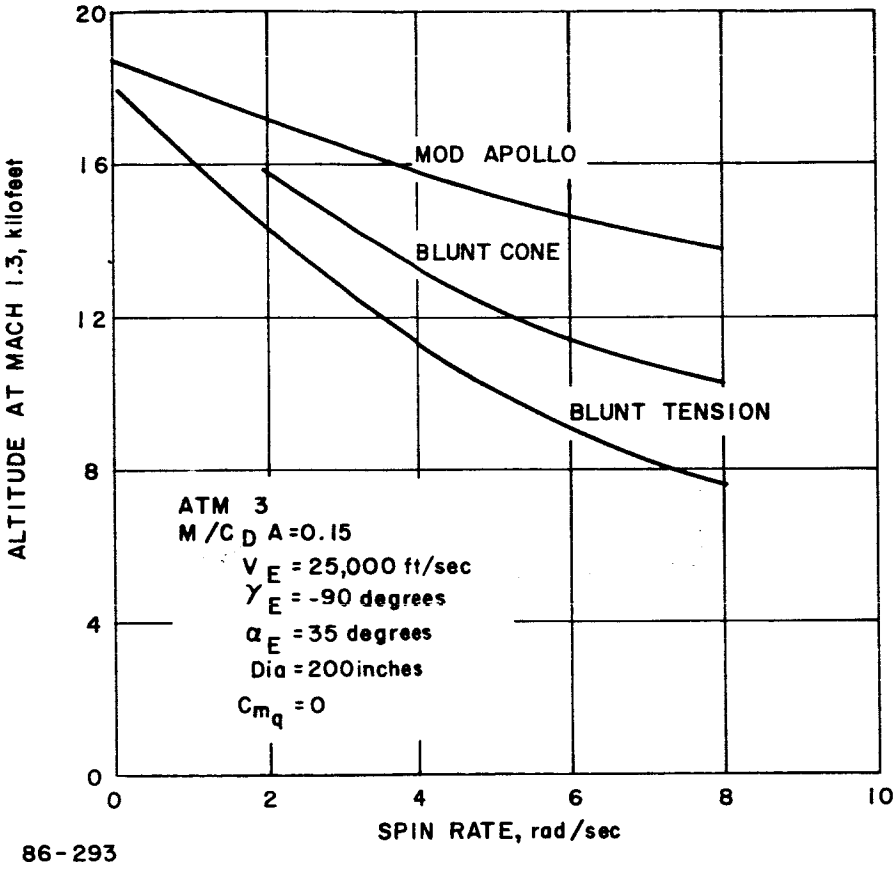


Figure 79 CONFIGURATION COMPARISON-ALTITUDE AT MACH 1.3

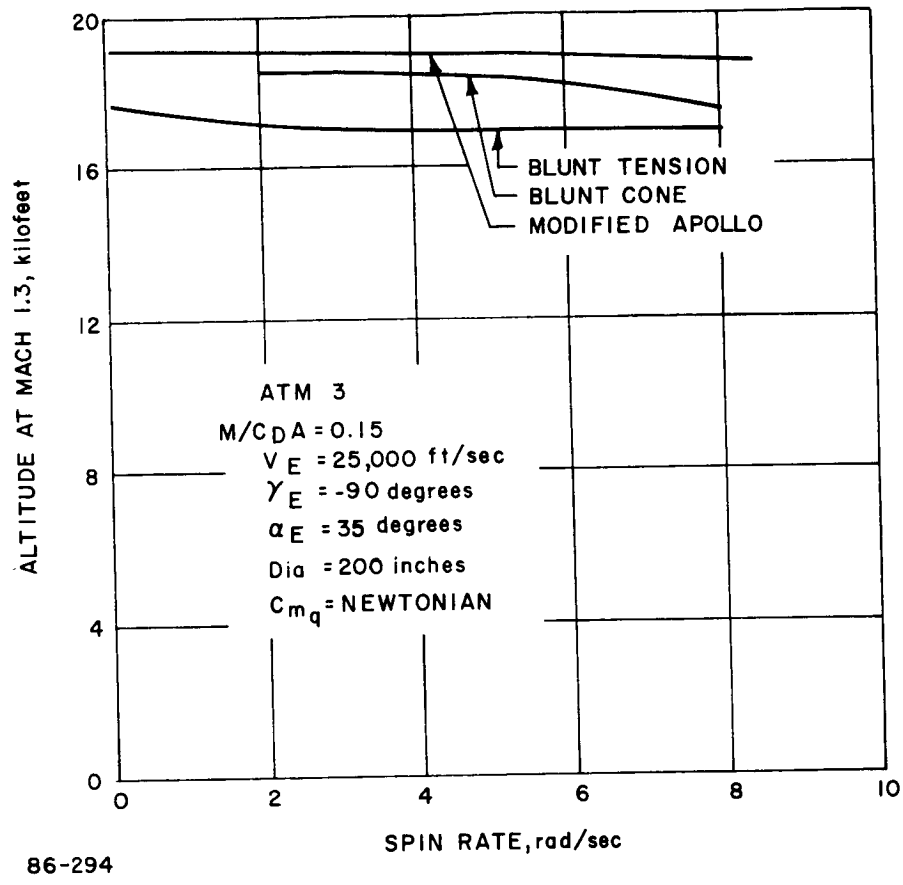


Figure 80 CONFIGURATION COMPARISON-ALTITUDE AT MACH 1.3

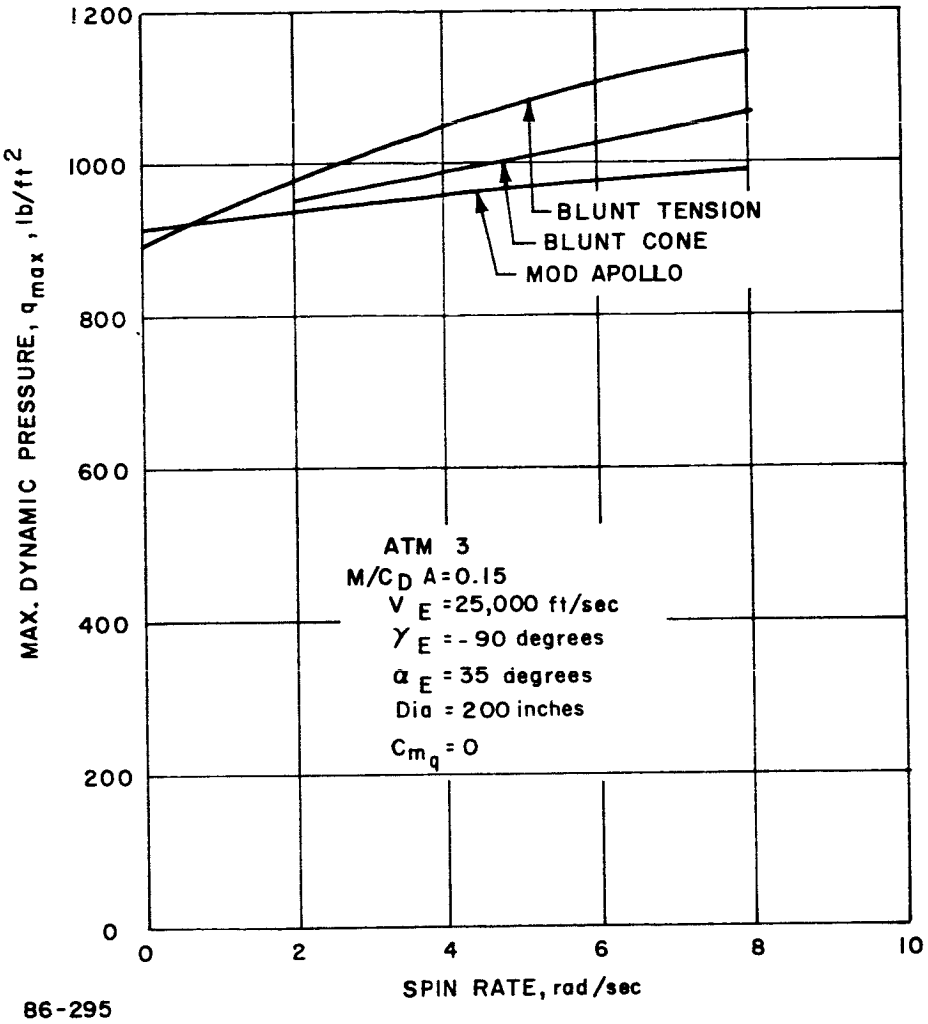


Figure 81 CONFIGURATION COMPARISON-MAXIMUM DYNAMIC PRESSURE

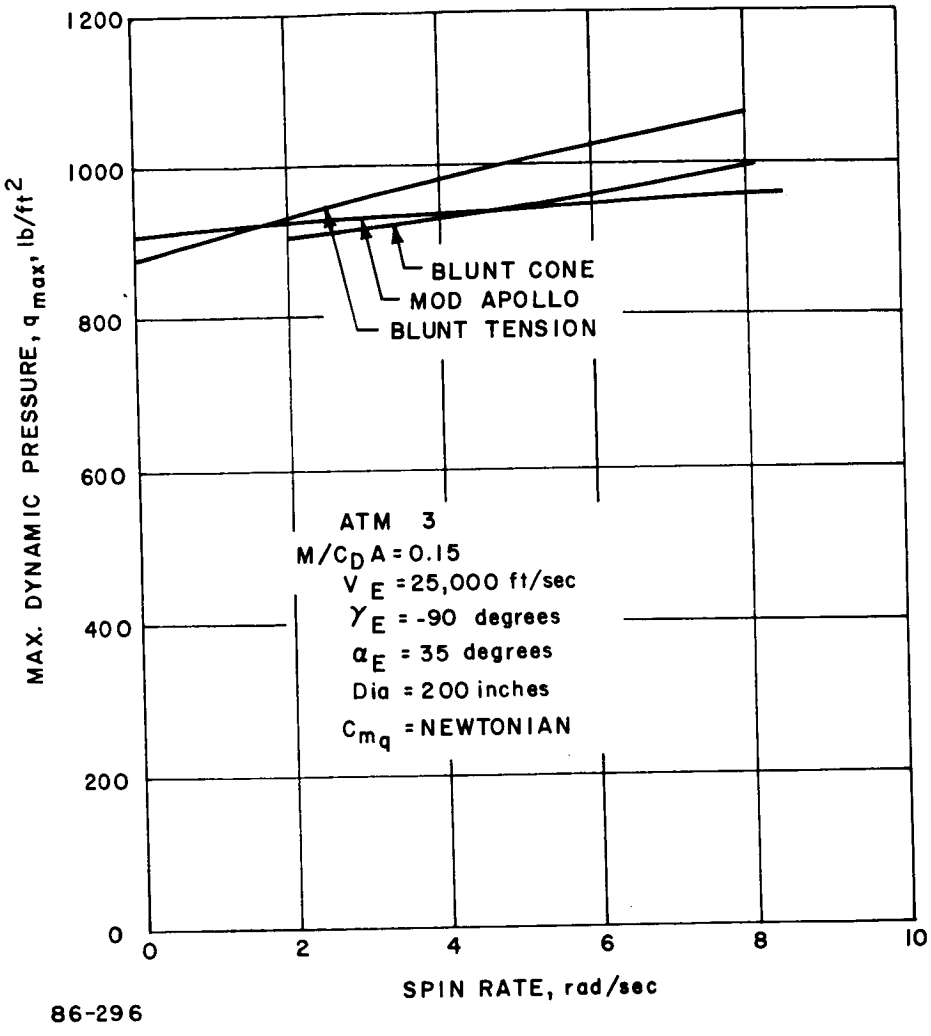


Figure 82 CONFIGURATION COMPARISON-MAXIMUM DYNAMIC PRESSURE

8.0 FAILURE MODE ANALYSIS

Once the typical mission flight sequence was established (Volume II, Book 2) subsystem operational requirements were investigated to determine the effects of subsystem failures on the over-all mission. In addition to this investigation, redundant features designed to compensate for a variety of possible flight capsule subsystem failures which could occur between launch from Earth and the end of post-landing operations were also studied.

The flight capsule design, as finally defined, included several of the recommended redundant features or suitable backup modes of operation. The design incorporated both the diagnostic instrumentation needed to identify failures and means to allow selection of appropriate failure mode operation. The selection is made either on the basis of MOS judgement or, automatically, by the capsule central computer and sequencer, (CC & S) .

The analysis of each failure mode was based on:

1. Definition of the failure mode.

2. Identification of a diagnostic method to determine whether a failure had occurred.

3. Identification of the backup command necessary either to reactivate the appropriate mode or to select an alternative mode of operation.

4. Identification of an alternative or redundant mode of operation capable of compensating for the original failure.

5. Analysis of the effect on the over-all mission if:

   a. The backup command activated the original mode of operation.

   b. The alternative or redundant mode of operation was successful.

   c. No alternative or redundant mode of operation was available or successful.

8.1 FAILURE MODE SUMMARY

Table LXII presents an abbreviated flight sequence with the related failure modes considered and their relative effects on the mission.

TABLE LXII

FAILURE MODE SUMMARY

Event	Time	Failure Mode Provision	Effect on Mission
Jettison sterilization canister lid	Approach trajectory	Jettison flight capsule	Flight capsule mission failure
Activate and checkout flight capsule systems	Preseparation	Redundant systems	No effect
Separate flight capsule	Postcheckout	Back-up separation joint	Flight capsule mission failure
Orient flight capsule for ΔV application	Postseparation	Redundant ACS capability	No effect
Thrust-solid engine	Postorientation	None	Flight capsule mission failure
Orient flight capsule for $\alpha_E = 0$ degrees	Postthrust	Redundant ACS capability	No effect
Spin flight capsule	Postorientation	Design shell for failure mode	No effect→minor
Jettison engine and ACS electronics	Postspin	Design shell for failure mode	Decreased descent time increased impact velocity and g
Deploy reefed parachute, jettison entry shell	Mach no = 1.3	Priority data transmission in the event of chute failure	Reduced data quantity
Deploy full open chute	16,000 foot altitude	Priority data transmission in the event of disreefing failure	Reduced data quantity
Land	Impact	Designed with factor of safety for impact g	Reduction → loss of surface data
Jettison chute and impact attenuator material	Postimpact	Redundant pyro devices	None
Deploy surface instrumentation	Posterection or stabilization	Redundant deployment devices	None

8.2 FAILURE BEFORE SEPARATION

System failures prior to separation are considered to deal mainly with those failures detected during preseparation checkout and the jettisoning of the sterilization canister lid. During checkout, the failure of a system to respond to checkout command is interpreted as a failure. After all backup and redundant excitation methods have been exhausted, dependent upon the seriousness of the loss, the flight capsule would be discarded and the entire capsule mission lost.

Another failure that would cause the loss of the entire flight capsule mission, would be the failure of the sterilization canister lid to jettison, thus preventing the capsule itself from separating from the flight spacecraft. In the event of this failure, the entire sterilization canister together with the capsule would be jettisoned at the field joint between the sterilization canister and the flight spacecraft by a command signal from the deep space instrumentation facility (DSIF).

8.3 SEPARATION FAILURE

The physical separation of the flight capsule from the flight spacecraft will be accomplished by explosive pen pullers; the signal for detonation will be provided by the flight spacecraft. Redundancy will be provided in the mechanization of the separation process, thus providing a high degree of probability of separation. In the event that separation does not occur, the base of the sterilization canister and the capsule will be jettisoned and the capsule mission will be lost. This backup separation will occur at the field joint between the flight capsule system and the flight spacecraft. The command required for this jettison is supplied by DSIF via the flight spacecraft.

8.4 ATTITUDE CONTROL SYSTEM FAILURE

The time of the detection or occurrence of the Attitude Control System (ACS) failure plays a significant role in the failure mode provision and effect of the failure on the mission. As previously mentioned, prior to separation a flight capsule system checkout is performed. If the checkout should detect an irreparable ACS malfunction (e.g., gas supply depleted, electronics malfunction, etc.) The mode of operation for the failure would be to orient the planetary vehicle to provide the flight spacecraft with the proper orientation, prior to separation, for ΔV application. After separation, the capsule would spin up to 50 rpm to provide stabilization during thrusting. However, since the ACS is inoperative, orientation for zero angle of attack at entry, in the normal mode, would not be possible. The exact extent of the mission degradation due to this operating mode is dependent upon the angle of attack at entry (entry shell design condition) and the look angle between the capsule and the spacecraft during transit for communication purposes.

Redundant ACS system capabilities have been incorporated into the system in the quantity of cold gas storage, thruster nozzles, and piping. In the event that the primary system should fail, sufficient backup capability will be available to provide normal operation and therefore no significant effect on the flight capsule mission will result.

8.5 PROPULSION FAILURE

The propulsion system aboard the flight capsule is required to provide sufficient ΔV capability to alter the capsule's trajectory from close approach to impact and to provide a lead time sufficient to allow communications during the descent portion of the flight capsule mission. Consequently, for failure mode analysis considerations, the relative effects on the mission of ΔV values larger and smaller than nominal were investigated as well as complete system failure.

In the event that the actual ΔV was larger than nominal, during the time of entry and descent of the flight capsule, the communication range between the capsule and the spacecraft would be large when compared to the design range. Also, the entry angle (γ_e) would be steeper than nominal. The failure mode provision for this event is compensated for by designing the communication system to handle ranges that would be produced in the event that the ΔV applied was equal to the maximum that the system was capable of producing (approximately 265 ft/sec). As a failure mode to provide for the steep entry angle, the entry shell has been designed to accommodate entry angles from -20 to -90 degrees.

The effect of an insufficient ΔV could be reflected in two ways (1) the ΔV could be too low to cause the space capsule to impact the planet or (2) the lead time between capsule impact and flight spacecraft orbital injection could be decreased. In the first case, it is obvious that the capsule mission is lost. In the second case, however, the lead time requirement used as a design basis was of sufficient magnitude to provide failure mode capability.

The failure of the rocket to ignite is an obvious total loss of the flight capsule mission. However, as previously stated, the rocket utilized is a solid type which possesses inherent ignition and operational reliability. In addition, redundant bridgewires are used for ignition.

8.6 ENTRY SHELL FAILURE

The failure of the entry shell during entry would naturally have catastrophic results on the flight capsule mission. Therefore, the early results of the failure-effects analysis were incorporated into the design of the shell. In addition to designing the entry shell heat shield and structure for the worst entry conditions that could be encountered (see Section 5.0 of this book), consideration

was given to off-nominal entry angle of attack spinning and nonspinning. As previously mentioned, the nominal angle of attack at entry (α_e) is 0 degrees. However, ACS failure after thrusting could result in a random attitude at entry, thus requiring the entry shell to be designed for any entry angle of attack from 0 to 180 degrees in a spinning or nonspinning mode.

8.7 CHUTE FAILURE

Early in the study, it became obvious that failure of the parachute system would result in loss of the surface operation experiments and severely restrict the amount of atmospheric data that could be obtained. With this fact in mind, the design of the canopy utilized a load factor of safety of 2.7 and the shroud and riser lines used a factor of 1.7. In addition to the added strength provided in the parachute materials themselves, redundant and backup devices were utilized for parachute deployment. In the normal mode, the parachute is deployed reefed at M = 1.3 by the accelerometer employing a time delay from peak g. As a backup to this operation, in the event the accelerometer malfunctions and the parachute is not deployed as programmed, a deployment signal is initiated by the altimeter at an altitude of 18,500 feet. The actual deployment is performed by a pilot parachute that is ejected by a mortar. A failure of the primary ejection system would automatically initiate the secondary system, a gas generator used to simultaneously deploy and inflate the main parachute. The parachute is then disreefed at an altitude of 16,000 feet, upon command from the altimeter by dual reefing line cutters electrically activated. The communication system has been designed to provide sufficient capability to relay a major portion of the atmospheric data sampled in the event that the main chute does not disreef (decreased descent time).

8.8 COMMUNICATION FAILURE

The communication subsystem has been designed to provide as high a degree of reliability as possible utilizing functional and block redundancy techniques consistent with the weight limitations of the total flight capsule system. Several of the redundant or backup features of the communications subsystem have been discussed as they pertain to the failure modes of other functional areas; e.g., increase communication range to account for an off-nominal incremental velocity from the propulsion system and rapid and selective playout of descent data to account for parachute failure. Another important redundancy feature of the communication system, is the use of both relay and direct communication links to provide the maximum probability of obtaining data from the flight capsule. The relay link is considered prime during entry and descent while the direct link is considered prime after landing. To utilize the direct link as backup to the relay capability, the entry and descent data is stored and played out, direct to DSIF, after landing.

8.9 INSTRUMENTATION FAILURE

The acquisition of data characterizing the Martian atmosphere has been considered as the primary mission objective in the 1971 lander design. Consequently, to improve the probability of obtaining data on the atmospheric characteristics, functional as well as block redundancy has been applied to the instrumentation, throughout the entry, descent and surface operation portion of the mission. As an example, consider the acquisition of atmospheric density, which can be read directly from the acoustical densitometer. However, density of the atmosphere can be derived from such quantities as: acceleration during entry, pressure, temperature and mean molecular weight. In addition, the same type of functional redundancy was applied to atmospheric pressure, temperature, composition and wind conditions.

8.10 IMPACT/RE-ERECTION FAILURE

Early in the study, the concept of a re-erecting flotation sphere was investigated, to be used as the method of packaging the instrumentation for surface operation. The flotation sphere would allow the landed capsule to come to rest in any position and then provide the means of orienting the antenna and instruments as required. However, the investigation showed that while the concept was good, the complexity of the re-erection system placed a severe penalty on the overall subsystem reliability. Using this analysis as a basis, the recommended design utilized an oblate spheroid as a landed capsule shape. The oblate spheroid can come to rest in only two possible positions, independent of the initial impact orientation. Therefore, redundant equipment requiring a given post-impact orientation was utilized in the design. Consequently, those instruments and equipment were properly oriented for deployment or use on both sides of the oblate spheroid.

9.0 LAUNCH PERIOD ANALYSIS

An integral segment of a preliminary design study for an interplanetary mission entails a comprehensive analysis of the various daily trajectory characteristics to select the optimum launch period based upon the mission design and scientific goals. Typical missions that have been postulated for Mariner-Voyager type vehicles include:

a. Flyby

b. Flyby/Lander

c. Orbiter

d. Orbiter/Lander

where each of the above designs could include multiple nonsurvivable atmosphere probes.

9.1 CONSTRAINING PARAMETERS

In theory, an unlimited number of possible interplanetary trajectories exist for a given target planet as there are, in general, at least four trajectory paths per given departure velocity per day. Within each launch opportunity there are two types of trajectories (Type I with heliocentric transfer angles less than 180 degrees and Type II with heliocentric transfer angles greater than 180 degrees) and within each type, the minimum energy point separates the Class I and Class II trajectories. Class I trajectories have shorter times of flight and smaller heliocentric transfer angles than the corresponding Class II trajectories. For Mars mission, the orbital rates of Earth and Mars about the sun are such that the oppositions and hence favorable launch opportunities occur approximately every 25.6 months (synodic period). In general, the launch opportunities precede opposition by 3 to 6 months, depending upon the transfer time. The metonic cycle for Mars is approximately 15 years or 7 synodic periods. In view of the fact that both the orbits of Earth and Mars are eccentric and that the orbit of Mars is inclined relative to the ecliptic plane, the energy requirements vary from opportunity to opportunity within a metonic cycle. The vast range of trajectory possibilities can be confined to tolerable limits by the employment of realistic engineering constraints. An evaluation of the payload characteristics of presently conceived boost vehicles, in conjunction with desirable mission payloads, places an upper bound on the departure velocity requirements and, in the case of an orbiter mission, on the approach velocity requirements. Additional engineering constraints which must be considered are:

1. Approach geometry that satisfies flight spacecraft mapping-mission requirements, and/or entry angle-impact site relationship for flight capsule.

2. Minimum Earth-Mars communications distance at encounter and/or at termination of the scientific mission.

3. Time of flight for system reliability considerations (flight times can exceed 400 days).

4. Declination of the launch asymptote (DLA) constraint which eliminates from consideration those trajectories where the declination of the hyperbolic departure trajectory is greater than the maximum orbital inclination achievable with launches from Cape Kennedy. However, the present range safety-launch-azimuth constraints of 114 degrees may be relaxed for missions in the mid-1970's thereby allowing declination greater than 33 degrees to be employed.

5. Scientific constraints on arrival date; it is desirable to land at a position and time which allow scientific measurements to be made during the wave of darkening.

6. Target-dispersion ellipse showing sensitivity of trajectory-orbit determination and velocity uncertainties associated with mid-course correction maneuvers.

With these and additional secondary trajectory constraints, the range of acceptable departure trajectories approaches reasonable proportions for each launch opportunity.

Since the cost per pound of scientific payload is extremely high for any Mariner-Voyager type interplanetary mission, every attempt must be made to maximize the useful payload. For a lander or flyby/lander mission, this is accomplished by employing the daily minimum departure velocity and selecting the launch period duration to obtain the desired maximum payload. In the case of an orbiter mission, the payload in the desired planetocentric orbit is a function of both the departure and approach velocities. This is maximized when the sum of the departure and approach velocities is minimized if the performance of the orbit establishment propulsion system is similar to that of the boost vehicle. Since, in general, the departure and approach velocities are not simultaneously minimized, the optimum launch period for an orbiter mission may be significantly different than the flyby period for the same launch opportunity. In the case of a lander/orbiter mission, the payload optimized period will be between the orbiter and flyby periods, and the proximity to either will depend upon the relative sizes of the lander and orbiter. Therefore, the selection of the optimum launch period with a given launch opportunity is directly

dependent upon the selected mission configuration and scientific objectives; numerous tradeoff studies exist to simultaneously achieve acceptable variations in all pertinent trajectory parameters. For the 1971 and 1973 launch opportunities the tentative mission configuration is an orbiter/lander, in 1975 a flyby/lander will be used. Therefore, the payload optimized periods afford an excellent starting point in design of the optimum launch period.

9.2 1971 LAUNCH OPPORTUNITY

The pertinent trajectory characteristics associated with the 1971 launch opportunity are presented in Table LXIII and Figure 83. The results of the launch period analysis for the 1971 opportunity indicate that:

1. This launch opportunity is one of the most favorable, in that both the departure and approach velocities are minimized near the same date at the middle of the period, thereby tending to maximize the payload for any mission configuration within the same period.

2. At opposition, Mars is near perihelion and the Earth-Mars distance is 56×10^6 km. The advantage of employing the shortest times of flight consistent with mission objectives is illustrated by the fact that the communication distance at encounter increases by 10^6 km for each day increase in the time of flight.

3. For the orbiter mission, the time of flight for Type I trajectories varies between 200 and 210 days.

4. The initial launch date in the payload optimized window is constrained by the DLA limitation of 33 degrees. However, this segment of the period can still be employed by utilizing off-optimum transfer trajectories.

5. The ZAP angle (angle between approach asymptote and Mars-sun line) varies from 109 to 69 degrees. The importance of this variation will be shown later to produce a similar variation in the capsule entry angle if a fixed impact location is assumed for the entire period.

9.3 1973 LAUNCH OPPORTUNITY

The pertinent trajectory characteristics associated with the 1973 launch opportunity are presented in Table LXIV and Figure 84. The results of the launch period analysis for the 1973 opportunity indicate that:

1. The departure velocity is approximately 30 percent larger than in 1971, and the approach velocity is reduced by approximately 10 percent. These variations, coupled with the fact that the approach velocity is

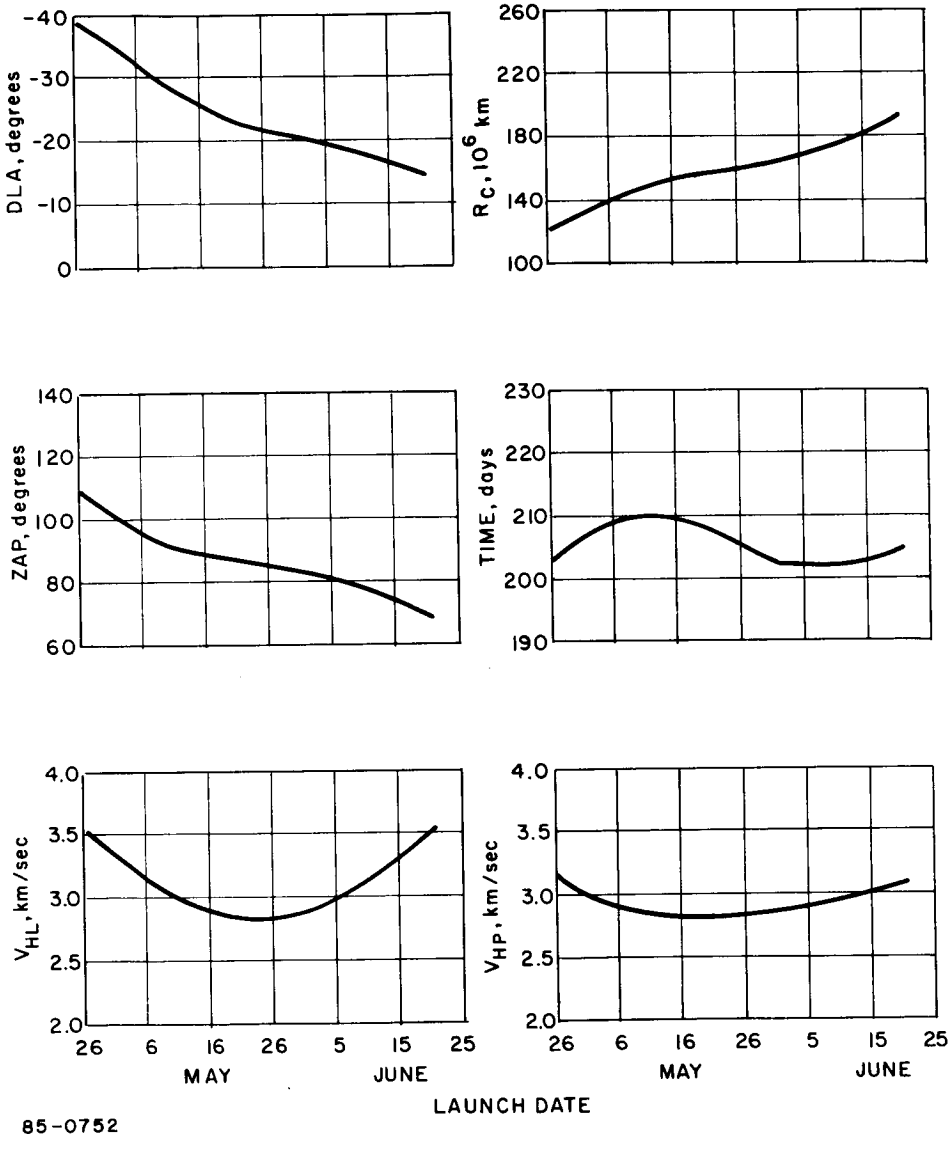


Figure 83 PERTINENT TRAJECTORY PARAMETERS 1971 ORBITER LAUNCH PERIOD

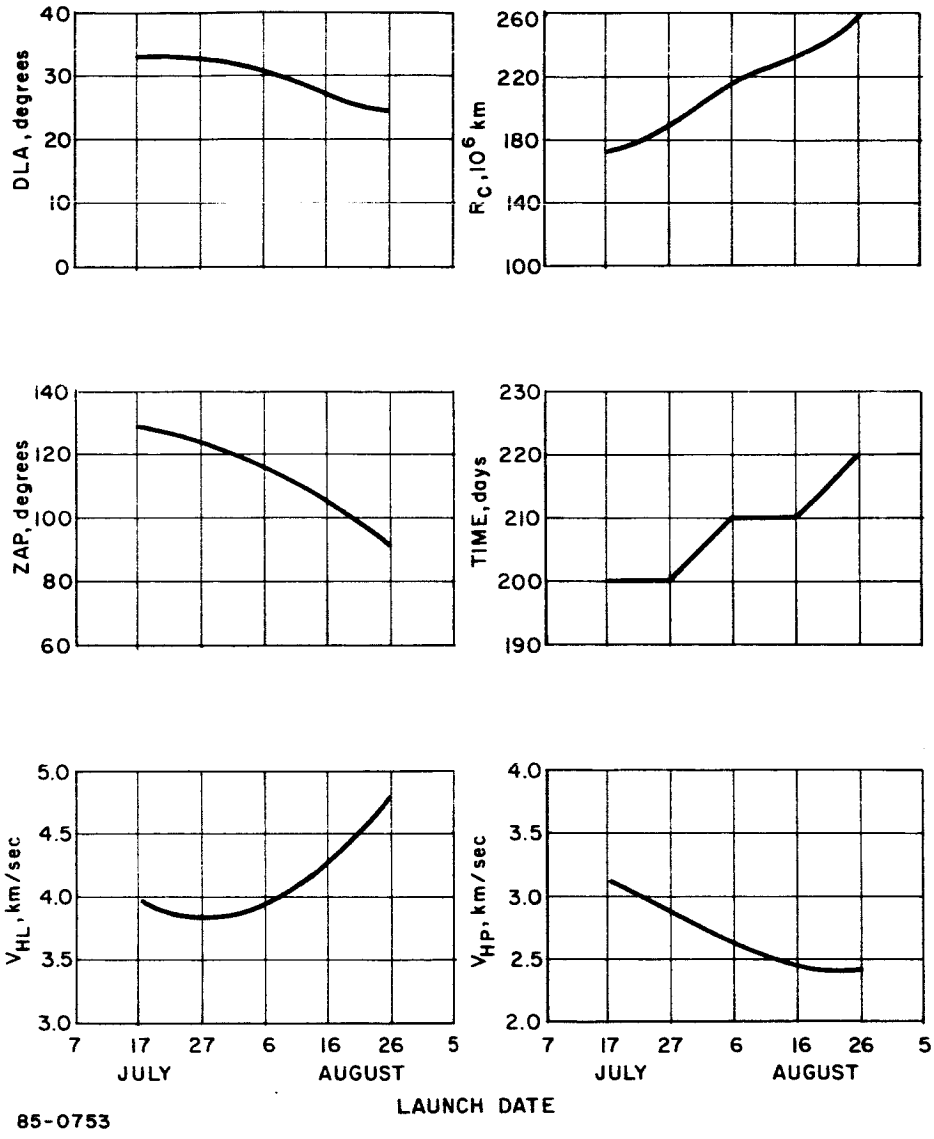


Figure 84 PERTINENT TRAJECTORY PARAMETERS 1973 ORBITER LAUNCH PERIOD

TABLE LXIII

PERTINENT TRAJECTORY PARAMETERS 1971 ORBITER LAUNCH PERIOD

Launch Date	Arrival Date	Time of Flight (days)	Departure Velocity (km/sec)	Approach Velocity (km/sec)	ZAP Angle (degrees)	Declination of Launch Asymptote (deg/sec)	Communication Range (10^6 km)
4/26/71	11/14/71	202	3.499	3.172	109	-37.9	121
5/4/71	11/28/71	208	3.197	2.925	97	-33.6	137
5/12/71	12/8/71	210	2.955	2.833	90	-27.8	149
5/20/71	12/14/71	208	2.833	2.818	86	-23.5	156
5/28/71	12/18/71	204	2.842	2.830	84	-21.3	161
6/5/71	12/24/71	202	2.973	2.870	80	-19.2	169
6/13/71	1/1/72	202	3.210	2.949	75	-16.9	179
6/21/71	1/11/72	204	3.533	3.075	69	-14.4	192

TABLE LXIV

PERTINENT TRAJECTORY PARAMETERS 1973 ORBITER LAUNCH PERIOD

Launch Date	Arrival Date	Time of Flight (days)	Departure Velocity (km/sec)	Approach Velocity (km/sec)	ZAP Angle (degrees)	Declination of Launch Asymptote (deg/sec)	Communication Range (10^6 km)
7/17/73	2/2/74	200	3.981	3.119	129	33.0	171
7/25/73	2/10/74	200	3.829	2.941	126	31.8	183
8/2/73	2/24/74	206	3.850	2.697	118	31.6	204
8/10/73	3/8/74	210	4.039	2.534	111	29.3	222
8/18/73	3/18/74	212	4.342	2.435	103	26.0	236
8/26/73	4/3/74	220	4.786	2.403	91	24.4	259

minimized 30 days after the minimum departure velocity, tend to degrade the payload capability that was available in 1971.

2. The times of flight and communication distance at encounter are slightly increased over the corresponding values associated with the 1971 opportunity.

3. The variation in the direction of the approach asymptote is again 40 degrees, however, the direction is rotated by 20 degrees so that the ZAP angle varies from 130 to 90 degrees.

9.4 1975 LAUNCH OPPORTUNITY

The pertinent trajectory characteristics associated with the normal 1975 launch opportunity are presented in Table LXV and Figure 85. An alternate, fast-approach period is proposed for this opportunity because of a communication blackout period due to superior conjunction; the characteristics of which are presented in Table LXVI and Figure 86. The results of the launch period analysis for the 1975 opportunity indicate that:

1. The major changes in the pertinent trajectory parameters associated with this period result from the fact that because of the DLA constraint, Type I trajectories cannot be employed; Type II trajectories are required for the entire launch period. Variations in the departure and approach velocities are minor when compared to the 1973 launch period, except that the minimum approach velocity occurs 35 to 40 days prior to the minimum departure velocity.

2. For a 60-day flyby period centered around 10 September 1975 (90 days before the 1975 opposition), the times of flight vary between 330 and 430 days. These times of flight, while being extremely long, might not impose a serious constraint on the mission were it not for the fact that the corresponding arrival dates are in the vicinity of superior conjunction (Mars and Earth on the opposite side of the sun). Therefore, in the vicinity of encounter there will be a period of 70 days where communication back to Earth is virtually impossible. For this reason alternate period is proposed; a constant departure velocity of 4.24 km/sec is employed to provide the fastest transfer trajectories (Class I). Such a 30-day launch period produces arrival dates which result in at least 70 days of surface operation before the angle subtended at Earth between the Mars line and sun line is less than 10 degrees, and 110 days of surface operation before superior conjunction.

3. The communication range for either window approaches 378×10^6 m for any extended surface mission.

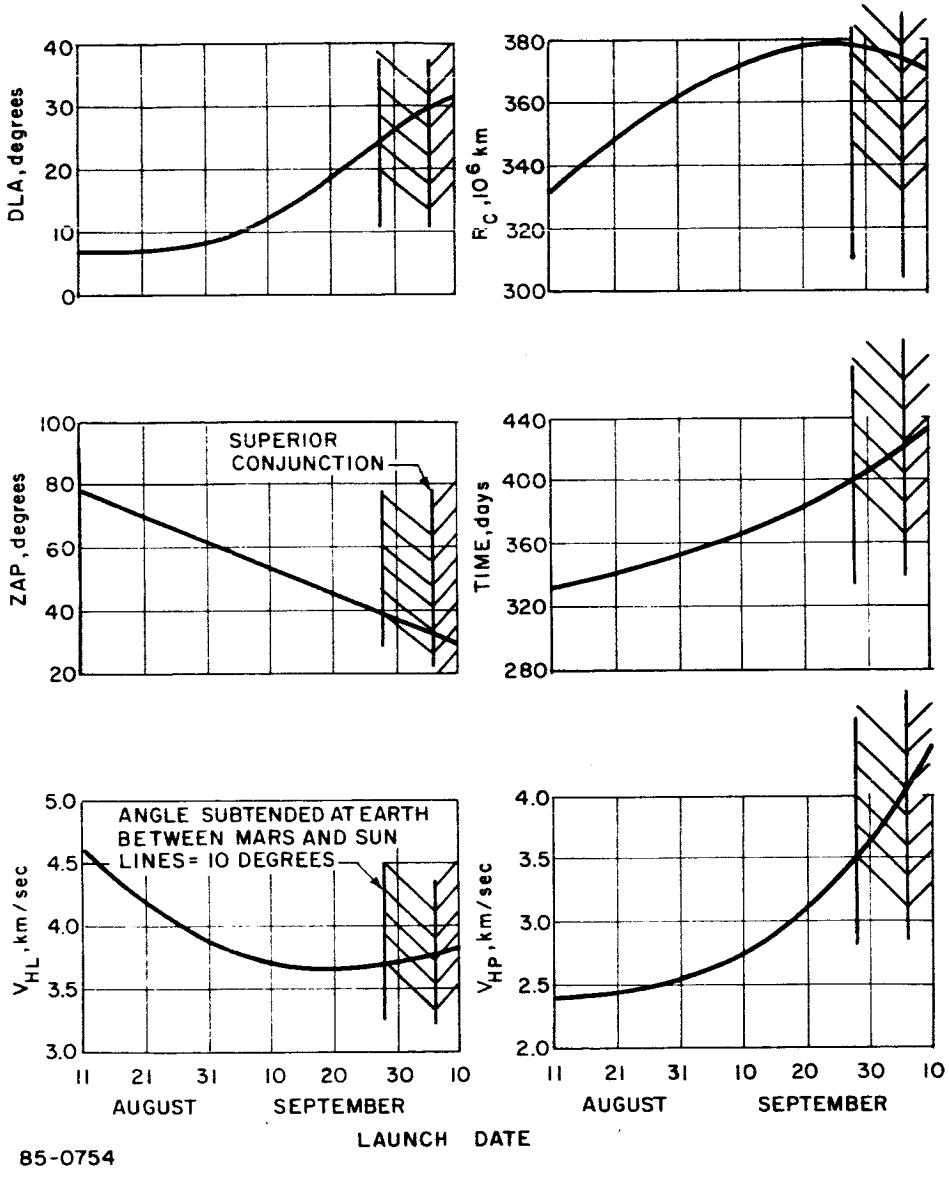


Figure 85 PERTINENT TRAJECTORY PARAMETERS 1975 FLYBY LAUNCH PERIOD

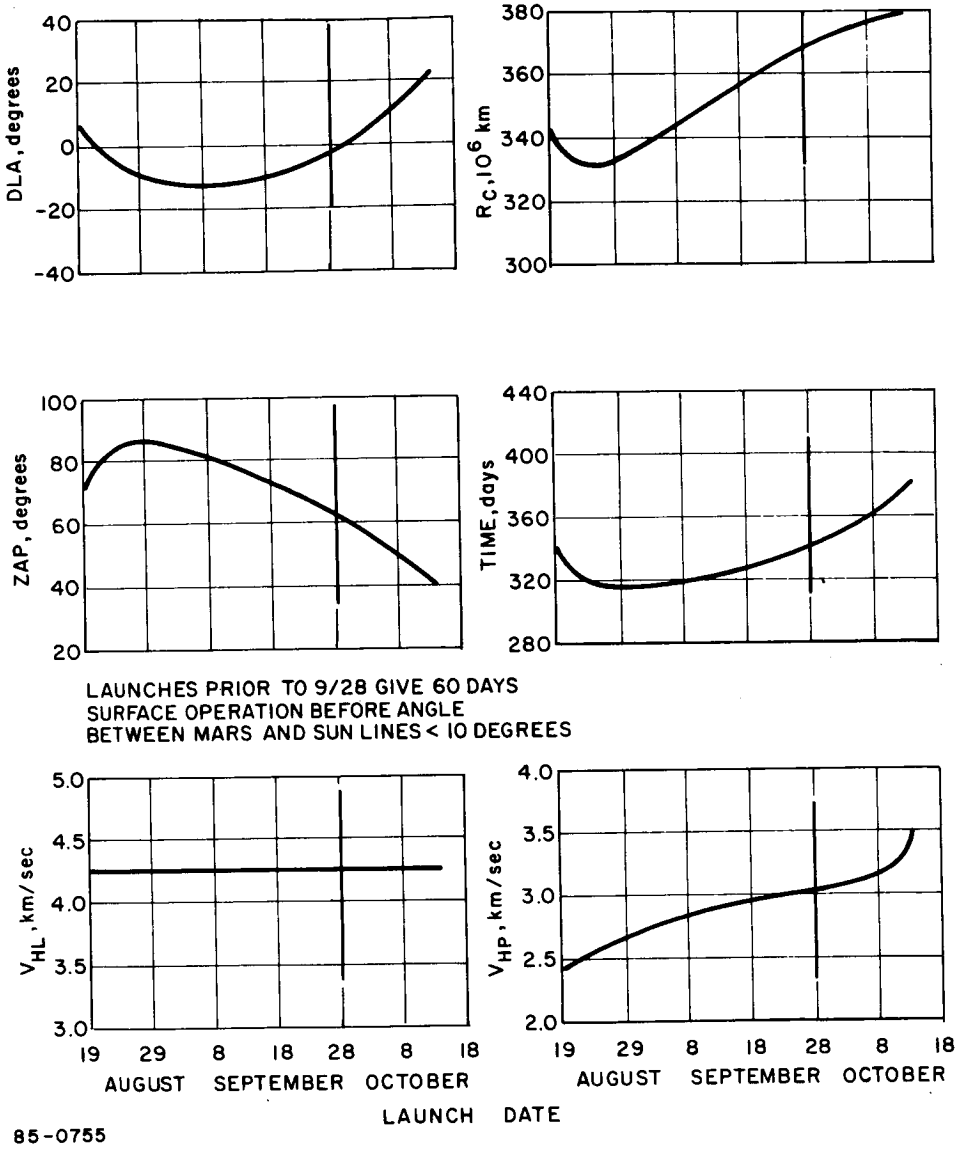


Figure 86 PERTINENT TRAJECTORY PARAMETERS 1975 FASTEST LAUNCH PERIOD

TABLE LXV

PERTINENT TRAJECTORY PARAMETERS 1975 FLYBY LAUNCH PERIOD

Launch Date	Arrival Date	Time of Flight (days)	Departure Velocity (km/sec)	Approach Velocity (km/sec)	ZAP Angle (degrees)	Declination of Launch Asymptote (degrees)	Communication Range (10^6 km)
8/11/75	7/8/76	332	4.597	2.385	78	6.2	332
8/19/75	7/24/76	340	4.260	2.428	72	6.4	345
8/27/75	8/8/76	347	3.985	2.496	67	7.0	355
9/4/75	8/26/76	357	3.786	2.616	60	9.5	365
9/12/75	9/13/76	367	3.675	2.780	53	12.8	371
9/20/75	10/5/76	381	3.653	3.042	46	17.9	377
9/28/75	11/2/76	401	3.701	3.462	38	24.1	378
10/6/75	12/1/76	422	3.782	4.000	32	29.1	373

TABLE LXVI

PERTINENT TRAJECTORY PARAMETERS 1975 FASTEST LAUNCH PERIOD (V_{HL} = 4.24 km/sec)

Launch Date	Arrival Date	Time of Flight (days)	Departure Velocity (km/sec)	Approach Velocity (km/sec)	ZAP Angle (degrees)	Declination of Launch Asymptote (degrees)	Communication Range (10^6 km)
8/19/75	7/24/76	340	4.260	2.428	72.0	6.5	345
8/27/75	7/8/76	316	4.250	2.610	86	-7.4	331
9/4/75	7/17/76	317	4.250	2.774	83	-11.8	339
9/12/75	7/30/76	322	4.246	2.884	78	-12.6	349
9/20/75	8/14/76	329	4.250	2.961	71	-9.8	359
9/28/75	8/31/76	338	4.257	3.011	63	-3.5	367
10/6/75	9/22/76	352	4.258	3.096	53	7.8	375
10/14/75	10/28/76	380	4.256	3.490	40	22.4	378

4. For the flyby period, the ZAP angle varies from 78 to 32 degrees, whereas for the 30-day fast period, the ZAP angle varies only from 86 to 71 degrees.

9.5 LAUNCH PERIOD ANALYSIS SUMMARY AND CONCLUSIONS

Summary parametric plots containing the pertinent trajectory parameters for 1971 and 1975 launch opportunities have been prepared from preliminary releases of JPL trajectory data, and are presented in Figures 87 through 90, respectively. These charts permit rapid tradeoff studies to be performed showing the variation in the parameters for various launch period selections. Reasonable engineering constraints for use with these figures include:

1. Departure velocity, $V_{HL} \leq 4.24$ km/sec

2. Approach velocity, $V_{HL} \leq 5.0$ km/sec

3. Declination of geocentric asymptote, DLA ≤ 33 degrees.

These constraints are based upon the launch and orbit injection capabilities of reasonable propulsion systems.

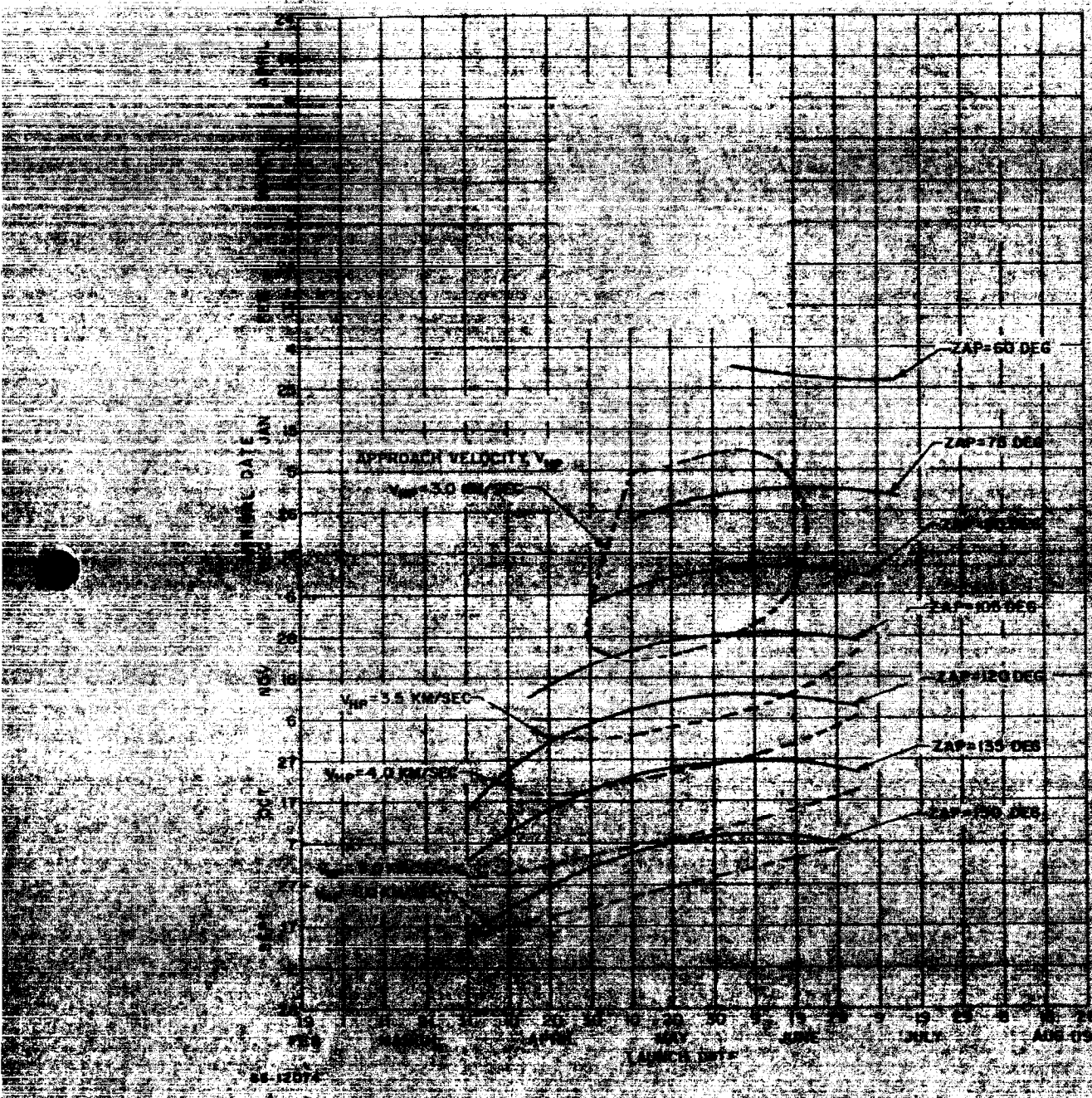
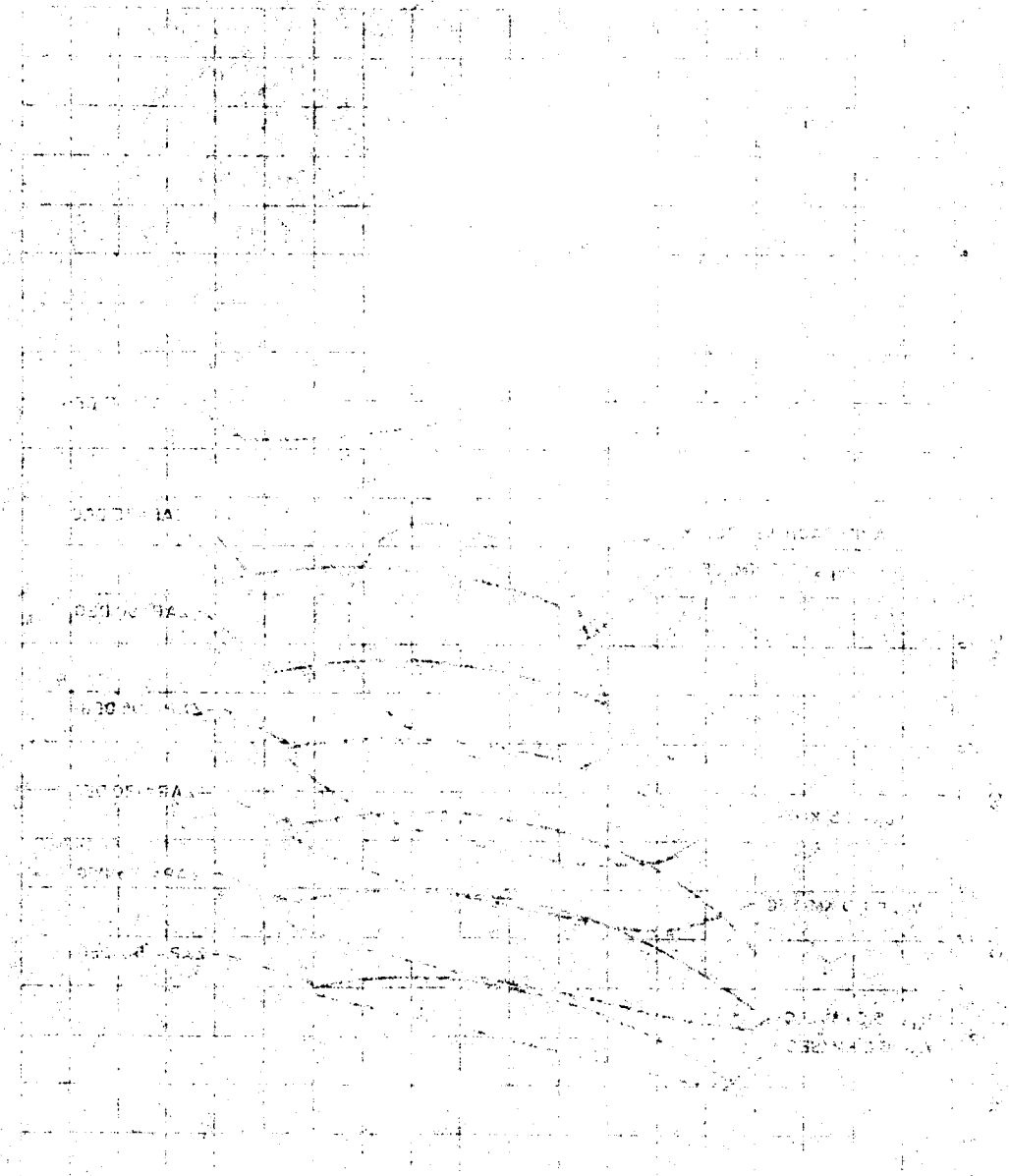


Figure 87. SUMMARY OF TRAJECTORY PARAMETERS, 1971 LAUNCH PERIOD



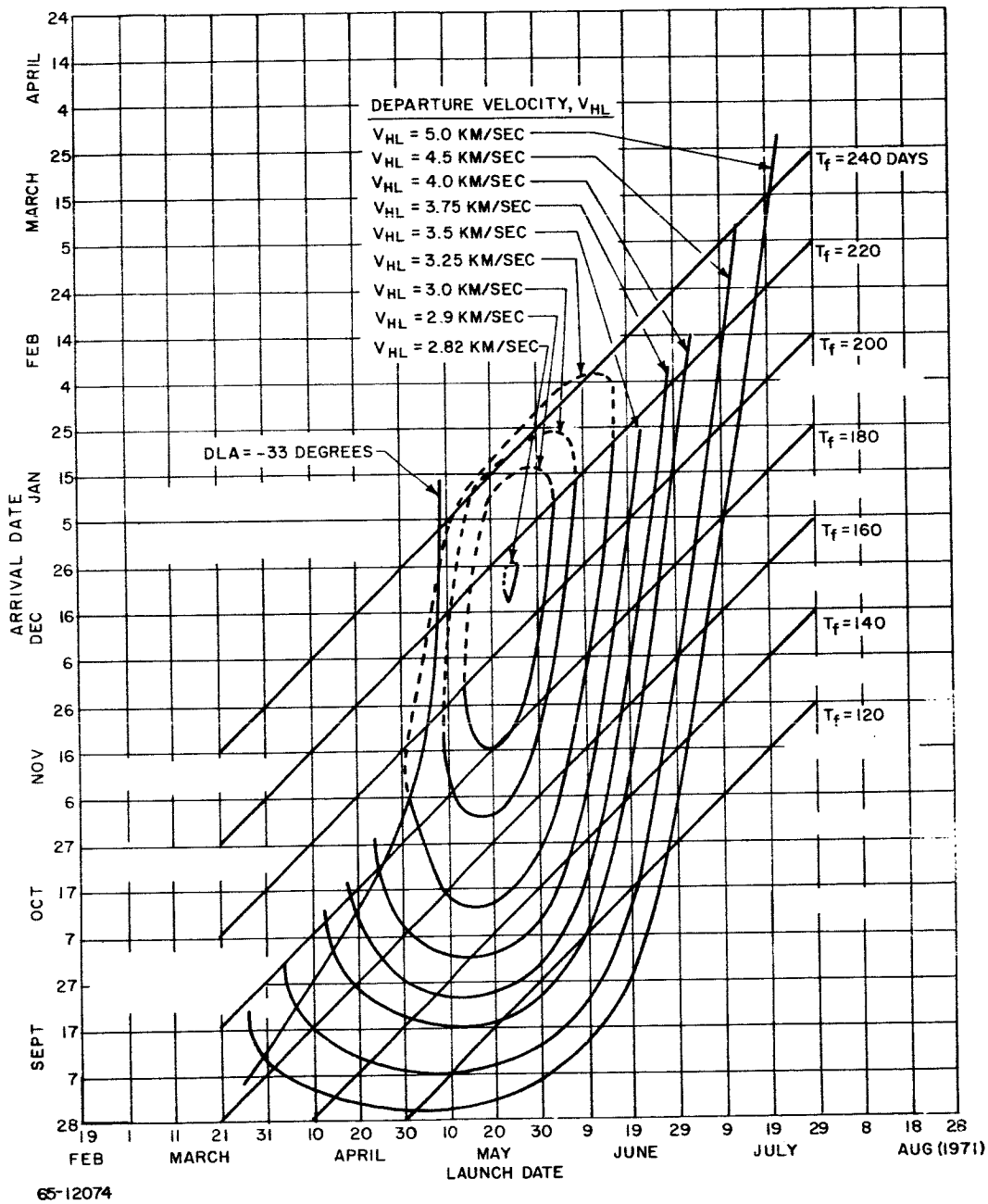


Figure 88 SUMMARY OF TRAJECTORY PARAMETERS, 1971 LAUNCH PERIOD

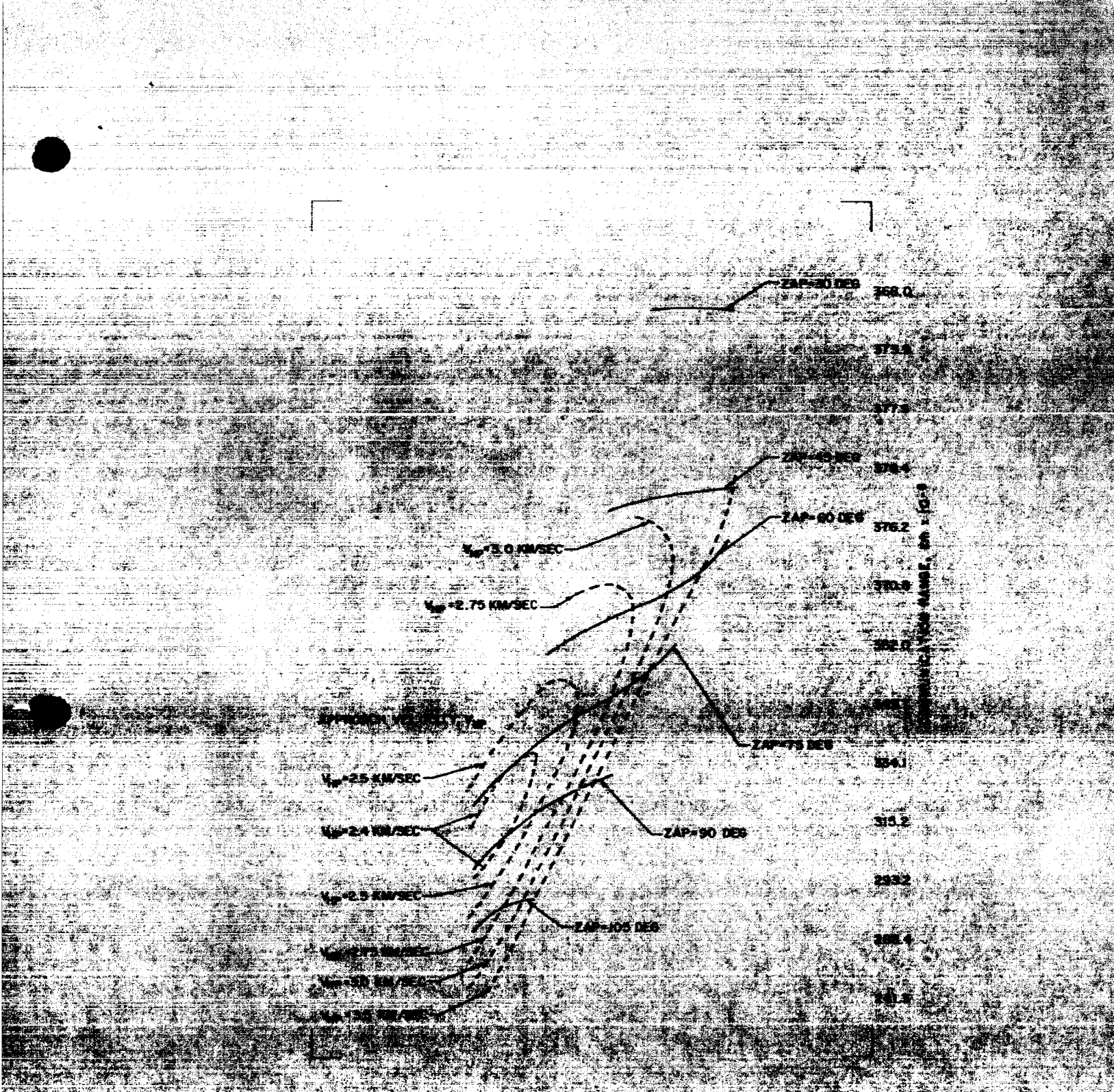
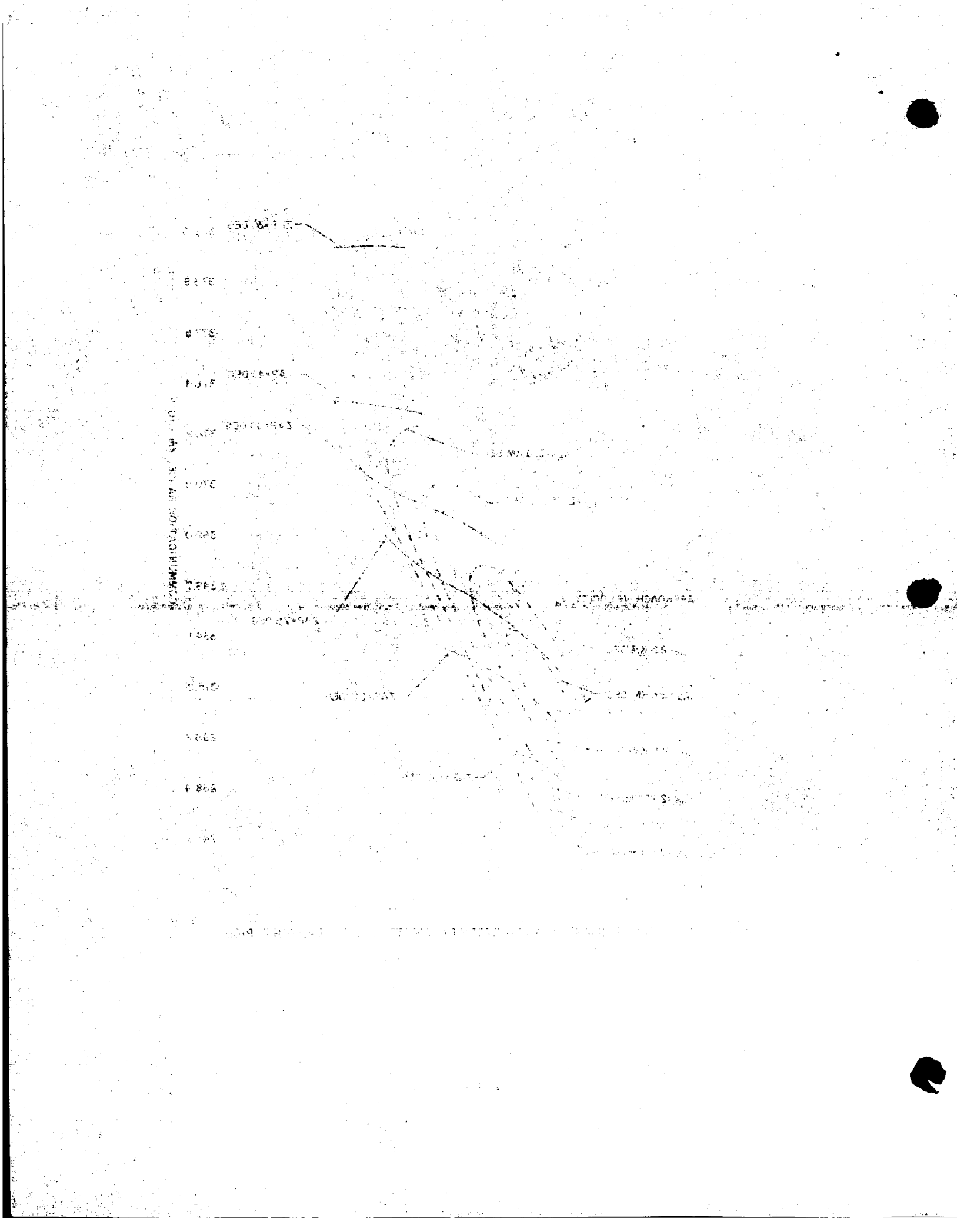


Figure 89: SUMMARY OF TRAJECTORY PARAMETERS, 1975 LAUNCH PERIOD



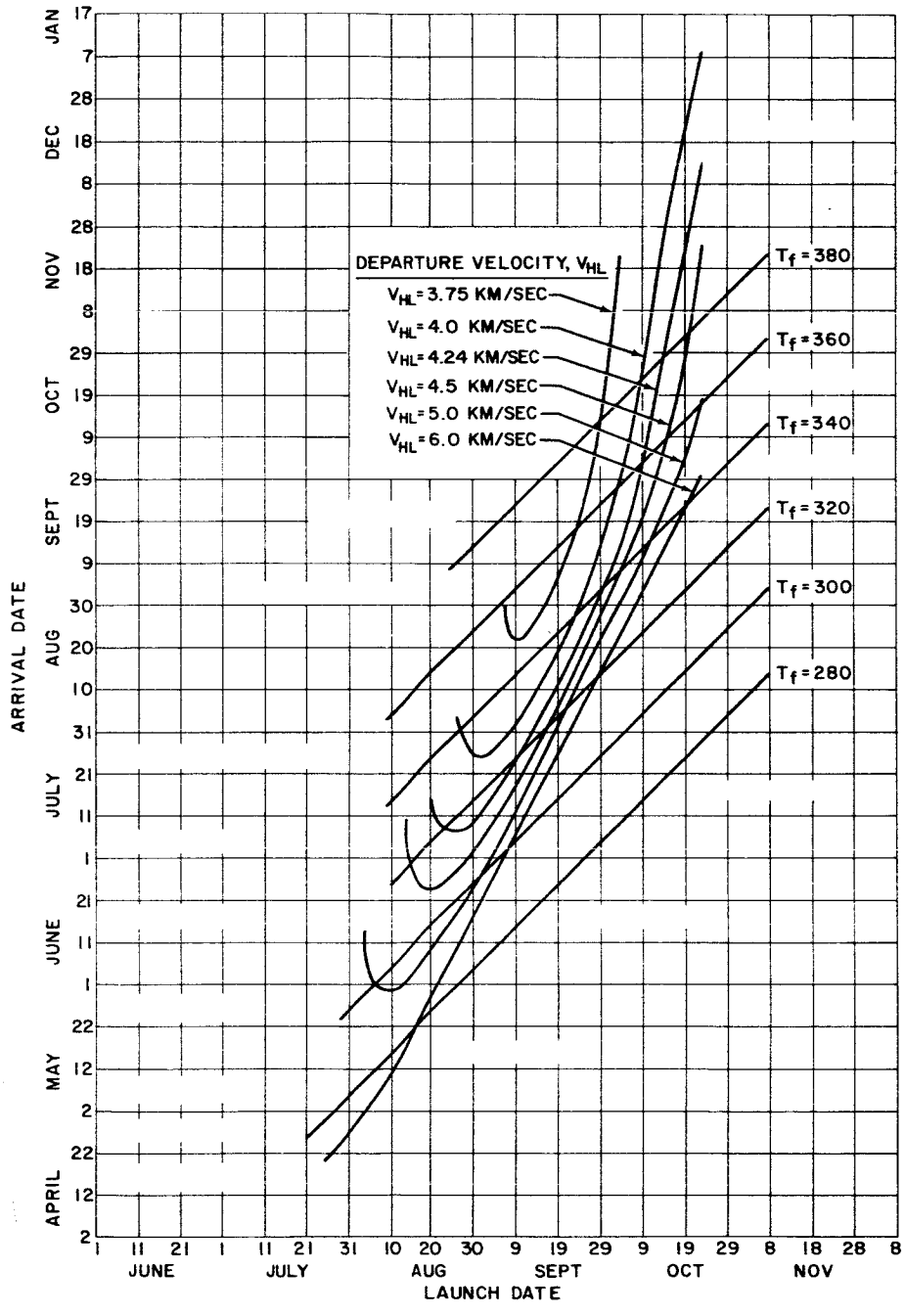


Figure 90 SUMMARY OF TRAJECTORY PARAMETERS, 1975 LAUNCH PERIOD