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ALTERNATE DESIGNS

SYSTEMS CONSIDERATIONS

APPENDIX

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PHASE 1A STUDY REPORT

VOYAGER SPACECRAFT

VOLUME 4
ALTERNATE DESIGNS
SYSTEMS CONSIDERATIONS
APPENDICES

30 July 1965

Prepared for
California Institute of Technology
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CONTENTS

CONTENTS (Continued)

CONTENTS (Continued)

APPENDIX A

RELIABILITY VERSUS WEIGHT TRADEOFF METHODS

1. INTRODUCTION

The Voyager 1971 mission specification allows spacecraft weight reserves to be allocated to extend launch periods, provide more favorable trajectories, increase performance margins, employ redundancy, and increase instrumentation, to improve the probability of mission success. This appendix is directed at reliability improvements to improve the probability of mission success. A meaningful approach to the allocation of weight reserves for this purpose requires:

- Recognizing and analyzing all potential areas in which weight reserve can be used to improve reliability
- Proposing detailed alternatives for implementation
- Applying a rational alternative selection procedure to allow the maximization of reliability within the weight reserve allocated for this purpose.

This Appendix provides a brief outline of the basic methods used for weight-reliability tradeoff in the Voyager Phase 1A study.

2. BASIC APPROACH

Before any weight allocation procedure can be applied, the quantitative relationship between probability of success and weight must be established for all items for which a feasible tradeoff exists. This constitutes the preponderance of effort required for the allocation procedure. It requires a thorough reliability analysis, weight analysis, trajectory analysis, and other system and subsystem analyses of the spacecraft design. The accuracy of any allocation procedure used is heavily depending upon the accuracy of the reliability and weight estimates of the nominal configuration and the alternate configurations being considered.

It must be recognized that weight is not the only constraint which exists. Many proposed usages of reserve weight (either individually or in certain combinations) will be rejected because of limitations on:

- Space or dimensional constraint within the nose fairing or spacecraft
- Available projected area for spacecraft electrical power. (In the process under discussion, an improvement alternative which demands more power is charged with the weight required to provide that power. It is desirable, however, to limit the total power to that which can be supplied by a fixed array.)
- Practical limits to implementing commands and switching
- Time available for development and test of complex options

Thus, it is seen at the outset that realistic options capable of utilizing weight to improve reliability must be invented within the other physical constraints of the Voyager design.

When it is assured that "probability of mission success" has been clearly defined and constraints other than weight have already been imposed, many optional weight allocations are valid provided they do not, in the aggregate, exceed the available weight reserve.

By definition mission success requires the successful performance of each subsystem required in each phase of the mission sequence, e.g., launch, cruise, lander separation, and retropropulsion. Thus the probability of mission success may be written

$$P = \prod_{i=1}^{n} P_i \qquad \text{(A-1)}$$

where P_i is the probability of success or reliability associated with the subsystems within their operational phases.

All subsystems and phases must be evaluated on an equitable basis. This basis is the measure of improvement in the individual reliability, P_i, corresponding to the utilization of a certain amount of the available weight reserve. Thus, there is some direct function expressing weight increase to implement the improvement. In some instances this functional relationship is essentially continuous (for example, the probability that the structure withstands the micrometeoroid environment versus the structural weight). In others it is composed of discrete options and as isolated inventions (viz., for the use of redundancy) which are subject to the ingenuity of the subsystem designer. In the latter case, some options are discarded when others exhibit greater reliability for less weight.

In order to compare all spacecraft subsystems and their elements in the same tradeoff they can be combined directly on the same plot when they are critical to the mission and are independent sources of failure potential. Independence (as expressed in Equation A-1) provides that a relative increase in reliability for any subsystem (expressed as the ratio of its improved reliability to the original reliability) achieves the same relative improvement for the combined total system. The successive analysis of subsystem reliability improvements, therefore, is tantamount to the successive analysis of system reliability improvements.

Generally confining weight limitations will force an optimization of reliability/weight conditions for all subsystems. Thus, there is a combination of subsystem reliability objectives for which the limited weight reserve is best utilized for total system reliability. For larger weight reserve conditions, however, it may be possible to invoke all manageable levels of equipment redundancy without the necessity for a more refined subsystem competition for weight.

In the general case, for each subsystem or phase

$$P_i = f(W_i) \tag{A-2}$$

where W_i is the amount of weight reserve devoted to the ith subsystem. The function is either continuous or quasi-continuous.

Subject to the constraint

$$\sum_i W_i \leq W_R \tag{A-3}$$

where W_R is the total available weight reserve, the allocation must determine the values of W_i so as to maximize P. This is done by maximizing the function

$$\ln P = \sum \ln P_i \qquad \text{(A-4)}$$

The derivatives of (A-4) are given by

$$\frac{\partial \ln P}{\partial W_i} = \frac{\partial \sum \ln P_i}{\partial W_i} = \frac{d \ln P_i}{\partial W_i} \qquad \text{(A-5)}$$

The condition for maximizing P is that

$$\delta P = \delta \ln P = \sum \frac{\partial \ln P}{\partial W_i} \delta W_i = \sum \frac{d \ln P_i}{dW_i} \delta W_i = 0 \qquad \text{(A-6)}$$

for any set of differential weights satisfying the constraint

$$\sum \delta W_i = 0 \qquad \text{(A-7)}$$

This condition is met if

$$\frac{d \ln P_1}{dW_i} = \frac{d \ln P_2}{dW_2} = \frac{d \ln P_3}{dW_3} = \cdots = \frac{d \ln P_n}{dW_n} = C \qquad \text{(A-8)}$$

3. WEIGHT ALLOCATION EXAMPLE

Figure A-1 illustrates the application of this allocation procedure to a hypothetical example in which a spacecraft mission is comprised of only three subsystems and one sensitive phase, i. e., launch period extension. An assumed reserve of 350 pounds is available.

The weight-probability data from the example cases of Table A-1 are plotted in the top of Figure A-1 subsystem against a logarithmic P_i scale. Straight-line segments connect the topmost points, ignoring, for the time, points which would cause a decrease in slope, if connected. These curves are the embodiment of Equation (A-2).

The upper curves are differentiated (as illustrated in the lower portion of Figure A-1), and the slopes, $d \ln P_i/dW_i$, are the step-like functions beneath. Equation (A-8) is indicated by the horizontal line (lower part of

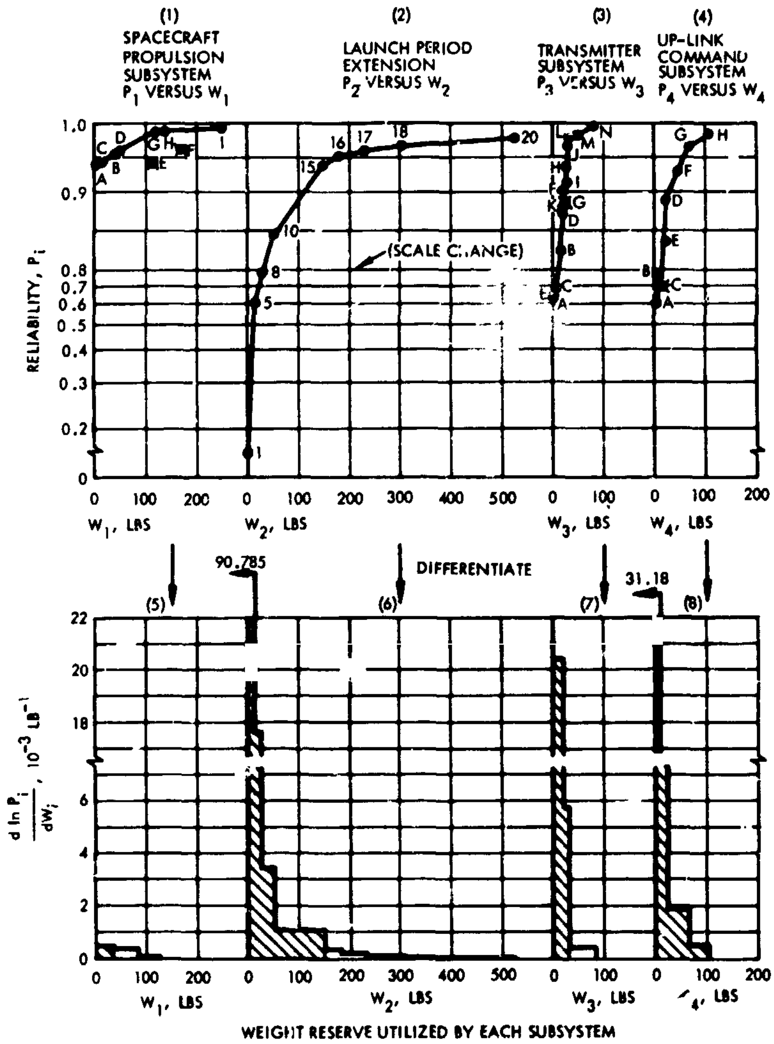
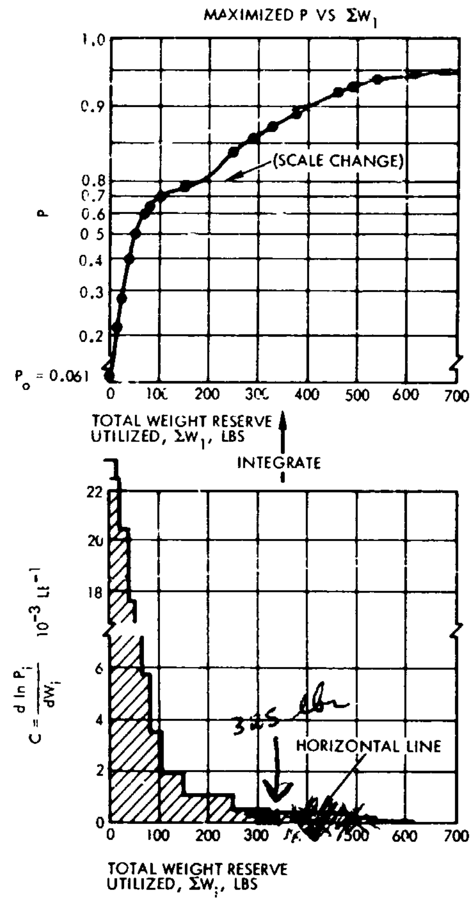


Figure A-1. Example of Subsystem and Phase Weight-Probability Functions

Figure A-2) showing the summation of subsystem weight increases, W_i, corresponding to C. The summing of d ln P_i/d W_i curves effects a ranking of the possible configuration changes. For, as C is reduced by lowering the horizontal line, each stepwise increase in the utilized weight reserve, $\sum W_i$, is identified with a specific change in a specific subsystem. And of all possible changes not yet implemented, this specific change is associated with the maximum Δln $P_i/\Delta W_i$; that is, it will increase mission reliability more (per pound) ther any other unrealized change in the entire spacecraft system.



SOLUTION OF EXAMPLE

USING MAJOR POINTS			
SUBSYSTEM	CONFIGURATION	W_i, LBS	P_i
(1)	B	36	0.956
(2)	15	150	0.940
(3)	L	32.2	0.978
(4)	H	106.8	0.990
		W_i = 325 LBS	0.870 = P

USING BY-PASSED POINT M			
SUBSYSTEM	CONFIGURATION	W_i, LBS	P_i
(1)	B	36	0.956
(2)	15	150	0.940
(3)	M	48.6	0.983
(4)	H	106.8	0.990
		W_i =341.4 LBS	0.875 = P

Figure A-2. Probability of Mission Success Versus Weight Reserve Utilized

The C versus $\sum W_i$ curve may be integrated to determine P, the maximized mission probability of success as shown in the upper part of Figure A-2. This integration satisfies the following:

$$\ln P = \ln P_o + A$$

$$= \ln P_o + \sum A_i$$

$$= \ln P_o + \sum \int_o^{W_i} \left(\frac{d \ln P_i}{dW_i}\right) dW_i$$

$$= \ln P_o + \sum \int_{P_{oi}}^{P_i} d \ln P_i$$

$$= \sum \ln P_{oi} + \sum \ln P_i - \sum \ln P_{oi}$$

$$= \ln P \qquad \text{(A-9)}$$

where

$P_o = \pi P_{oi}$ = Probability of mission success, no weight reserve utilized

P_{oi} = Reliability of ith subsystem for a weight increase, $W_i = 0$

$A = \sum A_i$ = Indicated area under C versus $\sum W_i$ curve (Figure A-2)

A_i = Indicated area under subsystem slope curve (Figure A-1)

The solution of the example is indicated by the tabular data of Figure A-2. Using the topmost points of each subsystem, 325 pounds of the available 350 pounds are used to achieve a 0.870 probability of mission success. The next major change would be to change transmitter configuration from L to N (see Table A-1). This would raise the probability from 0.870 to 0.888, but would exceed the permitted weight reserve. As the remaining weight reserve of 25 pounds is insufficient for any further changes of the topmost points, it is noted that changing the transmitter configuration from L to M, one of the previously ignored points, uses 16.4 pounds to raise the probability of mission success to 0.875, the maximum possible.

The previous discussion gives the results of four examples of weight-probability tradeoffs which have been illustrated in some detail. In each case, weight additions consist not only of redundant hardware but also additional retropropellant weight, electrical power supply weight, and other weight increases caused by each design change. In some cases, the base reliability—corresponding to a nonredundant design—is lower than that considered acceptable. This is intentional, as it serves to preclude starting from a design which might not be justified, and thereby removes any requirement to backtrack.

4. CONCLUSION

An orderly procedure is developed whereby spacecraft system alternates can be based upon the combined advantages of high velocity and low spacecraft weight. It is noted that this procedure requires input data which are the results of detailed engineering descriptions of subsystem designs for redundancy, reliability, and weight utilization. This is appropriate, since there are no other basic criteria on which a weight allocation procedure (for maximum probability of mission success) can rationally be implemented. By specific examples the applicability of the developed methods is demonstrated to be readily applied to weight-reliability tradeoff analyses. This is the process invoked for Voyager in Volume 4, Section III. 4.

Table A-1. Example Weight Utilization Options and Their Corresponding Reliability Improvements

(1) Spacecraft Propulsion System

	Configuration	Reliability P_1	Weight Increase W_3(lb)	
A	Nominal	0.9403	0	
B	Change 1	0.9564	36	
C	Change 2	0.9415	16	
D	Changes 1 and 2	0.9586	52	
E	Change 3	0.9430	113	(Discard)
F	Changes 1, 2 and 3	0.9614	165	(Discard)
G	Change 4	0.9877	121	
H	Changes 2 and 4	0.9891	137	
I	Changes 2, 3 and 4	0.9919	250	

Change 1 Quad solenoid valves on thrust chamber
Change 2 Redundant propellant isolation valves
Change 3 Redundant Pressurization system
Change 4 Redundant thrust chambers

(2) Launch Period Extension

Length of Launch Period (days)	Probability of Being Able to Launch Within Period	Weight Reserve Utilized W_2(lb)
1	0.170	0
5	0.606	14
8	0.775	27
10	0.845	52
15	0.940	150
16	0.949	180
17	0.958	230
18	0.965	305
20	0.976	525
30	0.996	1280

Table A-1. Example Weight Utilization Options and Their Corresponding Reliability Improvements (Continued)

(3) Transmitter Subsystem

	Configuration	Reliability P_3	Weight Increase W_3(lb)	
A	Nominal	0.636	0	
B	Change 1	0.825	12.2	
C	Changes 1 and 2	0.674	3.4	
D	Changes 1 and 2	0.873	15.6	
E	Change 3	0.658	2.6	
F	Changes 1, 2 and 3	0.903	18.2	
G	Change 4	0.884	23.3	(Discard)
H	Changes 2 and 4	0.936	26.7	
I	Changes 3 and 4	0.915	25.9	
J	Changes 2, 3 and 4	0.969	29.3	
K	Change 5	0.879	16.4	
L	Change 6	0.978	32.2	
M	Change 7	0.983	48.6	
N	Change 8	0.9975	81.4	

Change 1 One standby redundant power amplifier and converter
Change 2 One standby redundant modulator
Change 3 One standby mode selector and baseband assembly
Change 4 Two standby redundant power amplifiers and converters
Change 5 One standby redundant nominal circuit
Change 6 Two standby redundant nominal circuits
Change 7 Two configurations K in parallel
Change 8 Two configurations L in parallel

(4) Up Link Command Subsystem

	Configuration	Reliability P_4	Weight Increase W_3(lb)	
A	Nominal	0.598	0	
B	Change 1	0.758	7.6	
C	Change 2	0.705	14.6	(Discard)
D	Changes 1 and 2	0.893	22.2	
E	Change 3	0.837	21.6	
F	Change 4	0.932	42.6	
G	Change 5	0.969	64.8	
H	Change 6	0.990	106.8	

Change 1 One standby redundant decoder
Change 2 One standby redundant command receiver
Change 3 One standby redundant decoder/receiver circuit
Change 4 Two standby redundant decoder/receiver circuits
Change 5 Two configurations E in parallel
Change 6 Two configurations F in parallel

APPENDIX B

RELIABILITY ASSESSMENTS OF SUBSYSTEM ELEMENTS

This appendix presents the data established in the course of the analyses completed in support of the reliability conclusions presented in Volume 4, Section 4.

After listing the parts failure rates used in the reliability analyses, the appendix presents the reliability assessment work sheets for electrical equipment, the models and computations applied in the evaluation of redundancy options for electrical equipment, and finally the studies of mechanical equipment. For ease of reference the following list of the contents is provided:

* In support of the reliability data in the main text

4.10 Thermal Louvers

4.11 Strip Heaters

4.12 Magnetometer Booms

4.13 Stabilization and Control

4.14 Propellant Feed Configurations

Frequently in this appendix the convention is used whereby the number of 9's in a calculated probability is designated by an exponent. Thus

$$0.9^{(5)}42 \equiv 0.9999942$$

1. PART FAILURE RATES

	Part Generic Class and Type	Failure Rate (λ) Parts in 10^9 Hours
1.	Diode, Silicon, General Purpose	15
2.	Diode, Silicon, Digital	6
3.	Diode, Germanium	100
4.	Diode, Zener	40
5.	Varactor	50
6.	Microwave Mixer	1200
7.	Transistor, Silicon, General Purpose	50
8.	Transistor, Silicon, Digital	20
9.	Transistor, Silicon, Power	130
10.	Transistor, Germanium	300
11.	Capacitor, Fixed, Ceramic	15
12.	Capacitor, Fixed, Glass	3
13.	Capacitor, Fixed, Mica, Dipped	5
14.	Capacitor, Tantalum, Solid	20
15.	Capacitor, Paper, Mylar	30
16.	Capacitor, Variable	50
17.	Resistor, Fixed, Composition, General Purpose	8

	Part Generic Class and Type	Failure Rate (λ) Parts in 10^9 Hours
18.	Resistor, Fixed, Composition, Digital	2
19.	Resistor, Fixed, Metal Film	10
20.	Resistor, Fixed, Wire Wound, Precision	80
21.	Resistor, Fixed, Wire Wound Power	100
22.	Resistor, Variable, Composition	80
23.	Resistor, Variable, Wire Wound	120
24.	Transformer, One Winding	60
25.	Transformer, Two Winding	90
26.	Transformer, Three Winding	120
27.	Inductor	30
28.	Relay (2 contact sets)	480
29.	Relay, Magnetic Latching	680
30.	Connector Pins (normal usage)	10
31.	Connector Pins (quiescent ambient)	5
32.	Connection, Solder or Weld (normally with parts)	0.5
33.	Gyro	20,000
34.	Filter, R.F.I.	35
35.	Filter, Crystal	150
36.	Crystal, Quartz	75
37.	Attenuator	15
38.	Circulator Switch	250
39.	Diplexer	250
40.	Four-Port	250
41.	Traveling Wave Tube	10,000
42.	Cores, Memory (per cove)	0.01
43.	Thermistor	150
44.	Battery Cells (per cell)	400
45.	Integrated Circuits, Generic Digital Types	35
46.	Integrated Circuits, Operational Amplifier	80

	Part Generic Class and Type	Failure Rate (λ) Parts in 10^9 Hours
47.	Integrated Circuits, Digital Specific Types*	
47.1	Integrated Circuit (Type A) Dual Gate	16.3
47.2	Integrated Circuit (Type B) Flip-Flop	22.0
47.3	Integrated Circuit (Type C) Diode Gate	15.0
47.4	Integrated Circuit (Type D) Memory Isolation	15.0

2. RELIABILITY ASSESSMENT WORK SHEETS, ELECTRICAL EQUIPMENT

(Giving estimated equipment failure rates calculated from part populations and part estimated failure rates)

SUBSYSTEM: Telecommunications, S-Band Receiver

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Command Detector

Known (or Assumed) Environments: 50°C

Known (or Assumed) Packaging Technique: Cordwood & Integrated Circuits

Class of Usage: Digital, Analog

Parts Stress Derating Policy: 40% Analog, 10% Digital

*Integrated circuit failure-rate objectives are established by summing failure rates for a set of discrete parts (capable of achieving an identical circuit function) and dividing by 13, with no resulting value adopted below 15 parts in 10^9 hours. All part failure rates to be used subsequently for detailed design assessments will be subject to coordination with JPL to assure a maximum overall confidence in Voyager mission success predictions.

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	39	10	390
Transistor, Silicon	38	50	1900
Resistor, Comp.	85	5	425
Capacitor, Ceramic	30	15	450
Capacitor, Tantalum	69	20	1380
Transformers	4	120	480
Integrated Circuits (digital)	11	35	385
Total Failure Rate			5412 bits

SUBSYSTEM: Telecommunications, S-Band Receiver

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Receiver Selector

Known (or Assumed) Environments: 50°C

Known (or Assumed) Packaging Technique: Cordwood and Integrated Circuit

Class of Usage: Digital, Analog

Parts Stress Derating Policy: 40% Analog, 10% Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	7	15	105
Transistor, Silicon	6	50	300
Resistor, Comp.	18	8	124
Capacitor, Tantalum	1	20	20
Transformer (2 winding)	1	100	100
Integrated Circuits (digital)	16	35	560
Total Failure Rate			1209 bits

SUBSYSTEM: Telecommunications, S-Band Receiver

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Receiver Component

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	28	15	420
Diode, Zener	8	40	320
Transistor, Silicon	41	50	2050
Resistor, Comp.	111	8	888
Resistor, Metal Film	32	10	320
Capacitor, Ceramic	105	15	1575
Capacitor, Glass	80	3	240
Capacitor, Paper, Mylar	29	30	870
Capacitor, Tantalum	19	20	380
Inductors, RF (Transformers)	16	120	1920
Connector Pins	33	10	330
Crystal, Quartz	3	75	225
Mixer Diodes	6	1200	7200
Transformers, Power	2	120	240
Varactors	5	50	250
Total Failure Rate			17,228 bits

SUBSYSTEM: Telecommunications, Transmitter

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Modulator Exciter

Known (or Assumed) Environments: 50°C

Class of Usage: Analog

Parts Stress Derating Policy: 40% of Rated

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	25	15	375
Diode, Zener	7	40	280
Transistor, Silicon	16	50	800
Resistor, Comp.	50	8	400
Resistor, Metal Film	9	10	90
Resistor, W.W Prec.	1	80	80
Capacitor, Ceramic	33	15	495
Capacitor, Mica	7	5	35
Capacitor, Glass	17	3	51
Capacitor, Variable	14	50	700
Inductors, RF (Transformer)	3	120	360
Inductors, FF	25	30	750
Crystal, Quartz	1	75	75
Varactor	6	50	300
RFI Filter	7	35	245
Total Failure Rate			5034 bits

SUBSYSTEM: Telecommunications, Transmitter

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Power Amplifier

Known (or Assumed) Environments: 50°C

Class of Usage: Analog

Parts Stress Derating Policy: 40% of Rated

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	6	15	90
Transistor, Silicon	8	50	400
Resistor, Comp.	16	8	128
Resistor, Metal Film	20	10	200
Resistor, Variable	1	120	120
Resistor, Power	1	10	10
Capacitor, Ceramic	17	15	255
Capacitor, Variable	10	50	500
Capacitor, Paper, Mylar	5	30	150
Capacitor, Tantalum	23	20	460
Inductors, RF (Transformers)	4	120	480
Connector Pins (8 Connectors)	16	10	160
Tube, TWT	1	10,000	10,000
Isolator	1	2,000	2,000
Transformers, Power	2	120	240
Inductor, Power	1	120	120
Total Failure Rate			15,313 bits

SUBSYSTEM: Telecommunications, Transmitter Selector

Class of Usage: Digital, Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon, Analog	20	15	300
Transistor, Silicon, Analog	18	50	900
Resistor, Comp. Analog	32	8	256
Resistor, Metal Film, Dig.	6	10	60
Capacitor, Tantalum	8	20	160
Transformer, Power, 2 Winding	1	90	90
Integrated Flip/Flop	8	35	280
Integrated Dual Gate	25	35	875
Total Failure Rate			2921 bits

SUBSYSTEM: Telecommunications

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Bulk Data Storage

Known (or Assumed) Environments: 50°C

Known (or Assumed) Packaging Technique: Cordwood Electronics

Class of Usage: Digital

Parts Stress Derating Policy: 10% of Rated

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	32	6	192
Transistor, Silicon	132	20	264
Resistor, Metal Film	285	10	2850
Capacitor, Ceramic	32	15	480
Capacitor, Mica	43	5	215
Capacitor, Tantalum	22	20	440
Inductors	2	10	20
Relays (redundantly used)	7	50	350
Connector Pins	80	10	800
Transformers, Signal	4	25	100
		Electronics	5711
Mechanical Devices	24	100	2400
Total Failure Rate			8111 bits

SUBSYSTEM: Telecommunications

Equipment and/or Function: Data Handling Unit, Digital Telemetry Unit

Known (or Assumed) Environments: 50°C

Known (or Assumed) Packaging Technique: Integrated Circuits

Parts Population Analysis

Functional Element	Quantity	Integrated Circuits Per Element	Total Integrated Circuits
Command, Control & Logic	1	20	20
Bit Rate Selection	1	15	15
PN Generation	1	18	18
M. C. Word Selection	1	18	18

Functional Element	Quantity	Integrated Circuits Per Element	Total Integrated Circuits
Synch. Frame Mod	1	5	5
Capsule Buffer Memory	1	74	74
M.C. Bi-Level Multi	1	3	3
M.C. Analog Multi	1	54	54
SC Analog Multi	4	69	276
SC Bi-Level Multi	1	35	35
Elapsed Time	1	5	5
A/D Conversion	1	22	22
Combiner	1	6	6
Data Selection Control	1	7	7
Data PN Mod & Mixer	1	12	12
			570

λ /Integrated Circuit = 35×10^{-9} failures/hour

Series Failure Rate λ_t = 575×35 = $19,900 \times 10^{-9}$

R_{4280} = .91480 for no functional redundancy

An estimated threefold functional redundancy exists within the DTU functional complex. As a conservative (exponential) estimate of this upon equivalent failure rate (λ_e).

$\lambda_e = \frac{1}{3}\lambda_t$ = 6630×10^{-9} failures/hour for a single DTU component

R_{4280} = .97161 for effective functional redundancy

SUBSYSTEM: Telecommunication

Equipment and/or Function: VHF Capsule Preamp

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	4	15	60
Transistor, Silicon	2	50	100
Resistor, Comp.	10	8	80
Capacitor, Mica	8	5	40
Inductors	2	30	60
Total Failure Rate			340 bits

SUBSYSTEM: Telecommunications

Equipment and/or Function: VHF Capsule Receiver

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	6	15	90
Transistor, Silicon	11	50	550
Resistor, Comp.	60	8	480
Capacitor, Mica	30	5	150
Capacitor, Tantalum	2	20	40
Inductors	11	30	330
Transformer	1	120	120
Total Failure Rate			1760 bits

SUBSYSTEM: Telecommunications

Equipment and/or Function: VHF Capsule Demodulator

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	6	15	90
Transistor, Silicon	10	50	500
Resistor, Comp.	55	8	440
Capacitor, Mica	30	5	150
Inductors	10	30	300
Total Failure Rate			1480 bits

SUBSYSTEM: CS&C

Equipment and/or Function: Input Decoder (1) and (5)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	8	6	48
Transistor, Silicon	12	20	240
Resistor, Metal Film	12	10	120
Capacitor, Glass	4	3	12
I. C. Type A	40	16.3	652
I. C. Type B	25	22.0	550
Total Failure Rate			1622 bits

SUBSYSTEM: CS&C Centralized Memory Type

Equipment and/or Function: Command Decoder (2)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Transistor, Silicon	384	20	7680
Resistor, Metal Film	521	10	5210
I. C. Type A	16	16.3	260
I. C. Type B	10	22.0	220
Total Failure Rate			13,370 bits

SUBSYSTEM: CS&C (Centralized Memory Type)

Equipment and/or Function: Sequencer (3)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	16	6	96
Transistor, Silicon	210	20	4200
Resistor, Metal Film	100	10	1000
Capacitor, Glass	20	3	60
Inductors	1	30	30
Crystal	1	75	75
Cores	4148	.01	41
I. C. Type A	130	16.3	2119
I. C. Type B	150	22.0	3300
I. C. Type C	32	15.0	480
I. C. Type D	24	15.0	360
Total Failure Rate			11,761 bits

SUBSYSTEM: CS&C (Centralized Memory Type)

Equipment and/or Function: Power Converter (4)

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	50	6 (digital)	300
Diode, Zener	10	10	400
Transistor, Silicon	40	50	2000
Resistor, Metal Film	120	10	1200
Capacitor, Glass	55	3	165
Capacitor, Tantalum	29	20	580
Inductors	6	30	180
Transformers	16	90	1440
Total Failure Rate			6265 bits

SUBSYSTEM: CS&C (Distributed Memory Type)

Equipment and/or Function: Command Decoder (6)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Transistor, Silicon	384	20	7680
Resistor, Metal Film	512	10	5120
I. C. Type A	16	16.3	260
I. C. Type B	10	22.0	220
Total Failure Rate			13,280 bits

SUBSYSTEM: CS&C (Distributed Memory)

Equipment and/or Function: Sequence Decoder, I.C. Type (7X)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Transistor, Silicon	96	20	1920
Resistor, Metal Film	128	10	1280
I.C. Type A	440	16.3	7172
I.C. Type B	870	22.0	19140
Total Failure Rate			29,512 bits

SUBSYSTEM: CS&C (Distributed Memory)

Equipment and/or Function: Sequence Decoder (Core Type) (7Y)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	900	6	5400
Diode, Zener	60	40	2400
Transistor, Silicon	360	20	7200
Resistor, Metal Film	1250	10	12500
Resistor, W.W Prec.	50	80	4000
Capacitor, Tantalum	80	20	1600
Capacitor, Glass	270	3	810
Cores	700	.01	7
Transformers	4	120	480
I.C. Type A	42	16.3	684
I.C. Type B	40	22.0	880
Total Failure Rate			35,961

SUBSYSTEM: CS&C (Distributed Memory Type)

Equipment and/or Function: Clock (8)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	10	6	60
Transistor, Silicon	10	20	200
Resistor, Metal Film	20	10	200
Capacitor, Glass	4	3	12
Crystal	1	75	75
I. C. Type A	20	16.3	326
I. C. Type B	40	22.0	880
Total Failure Rate			1753 bits

SUBSYSTEM: CS&C (Distributed Memory Type)

Equipment and/or Function: Power Converter (9)

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	30	6 (digital)	180
Diode, Zener	6	40	240
Transistor, Silicon	25	50	1250
Resistor, Metal Film	80	10	800
Capacitor, Tantalum	20	20	400
Capacitor, Glass	40	3	120
Inductors	4	30	120
Transformer	10	90	900
Total Failure Rate			4010 bits

SUBSYSTEM: Power

Equipment and/or Function: Battery Regulator

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	7	15	105
Diode, Zener	8	40	320
Transistor, Silicon	18	50	900
Resistor, Comp.	15	8	120
Resistor, W.W Prec.	25	80	2000
Capacitor, Paper	2	30	60
Capacitor, Paper Tantalum	13	20	260
Inductors	2	30	60
Transformers	4	120	480
Thermistors	2	150	300
Total Failure Rate			4605 bits

SUBSYSTEM: Power

Equipment and/or Function: Battery

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Ag - CD Battery Cells	18	400	7200
Total Failure Rate			7200 bits

SUBSYSTEM: Power

Equipment and/or Function: Inverters (3)

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	6	15	90
Diode, Zener	1	40	40
Transistor, Silicon	15	50	750
Resistor, Comp.	36	8	288
Capacitor, Paper	9	30	270
Capacitor, Tantalum	13	20	260
Inductors	6	30	180
Connector Pins	4	10	40
Transformers	14	120	1680
Thermistors	3	150	450
Total Failure Rate			4048 bits

SUBSYSTEM: Power

Equipment and/or Function: Power Control Unit

Class of Usage: Analog

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	48	15	720
Diode, Zener	8	40	320
Transistor, Silicon	40	50	2000
Resistor, Comp.	35	8	280
Resistor, W.W Prec.	96	80	7680
Capacitor, Tantalum	15	20	300
Capacitor, Paper	49	30	1470
Transformer	5	120	600
Thermistor	1	150	150
Total Failur Rate			13,520 bits

SUBSYSTEM: Electrical Distribution

Tradeoff Study: Baseline to Reference, Option #2

Equipment and/or Function: Command Distribution Unit and Cables/Connectors

Known (or Assumed) Environments: 50°C

Class of Usage: Analog

Parts Stress Derating Policy: 25% CDU Criticality, 20% Connector Criticality

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	100	15	1500
Transistor, Silicon	100	50	5000
Resistor, Comp.	200	8	1600
Relays (contacts quad connected)	100	12.5	1250
	CDU for 100% critical commands		9350
Connector Pins*	2700	1	2700
	CDU for 25% critical commands		2340
Total Failure Rate			5040 bits

SUBSYSTEM: Electrical Distribution

Tradeoff Study: Baseline to Reference, Option #1

Equipment and/or Function: Command Distribution Unit and Cables/Connectors

Class of Usage: Analog

Parts Stress Derating Policy: 25% CDU Criticality, 20% Connection Criticality

*All critical pins paralleled at the connector.

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	100	15	1500
Transistor, Silicon	100	50	5000
Resistor, Comp.	200	8	1600
Relays (contacts quad connected)	100	12.5	1250
	CDU for 100% critical commands		9350
Connector Pins*	2700	5	13500
	CDU for 25% critical commands		2340
Total Failure Rate			15,840 bits

SUBSYSTEM: Electrical Distribution

Tradeoff Study: Baseline to Reference, Option 0

Equipment and/or Function: Command Distribution Unit and Cables/Connectors

Known (or Assumed) Environments: 50°C

Class of Usage: Analog

Parts Stress Derating Policy: 25% CDU Criticality, 20% Connection Criticality

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	100	15	1500
Transistor, Silicon	100	50	5000
Resistor, Comp.	200	8	1600
Relays	50	400	20000
	CDU for 100% critical commands		28100

*With an estimated 450 connectors with 30 pins each and 20% of them critical

	Quantity	Failure Rate/Part	Category Failure Rate
Connector Pins*	2700	5	13500
	~DU for 25% critical commands		7030
Total Failure Rate			20,530 bits

SUBSYSTEM: Science Support, Configurations A & B

Tradeoff Study: Baseline to Reference

Equipment ar o/or Function: Planet Oriented Package, 2 Gimbal

Known (or Assumed) Environments: 50^{o}C

Known (or Assumed) Packaging Technique: Cordwood Electronics

Class of Usage: Analog

Parts Stress Derating Policy: 40%

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	10	15	150
Transistor, Silicon	16	50	800
Resistor, Comp.	45	8	360
Capacitor, Ceramic	12	15	180
Capacitor, Paper	6	30	180
Relays	1	480	480
Connector Pins	40	10	400
Gimbal Assembly	2	500	1000
Pick-Off Assembly	2	50	100
Motor	2	240	480
Total Failure Rate:			4130 bits

*With an estimated 450 connectors with 30 pins each and 20% of them critical.

SUBSYSTEM: Science Support, Configuration C

Tradeoff Study: Baseline to Reference

Equipment and/or Function: Planet Oriented Package, 1 Gimbal

Known (or Assumed) Environments: 50°C

Known (or Assumed) Packaging Technique: Cordwood Electronics

Class of Usage: Analog

Parts Stress Derating Policy: 40% Rated

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	5	15	75
Transistor, Silicon	8	50	400
Resistor, Comp.	20	8	160
Capacitor, Ceramic	6	15	90
Capacitor, Paper	4	30	120
Relays	1	480	480
Connector Pins	30	10	300
Gimbal Assembly	1	500	500
Pick-Off Assembly	1	50	50
Motor	1	240	240
Total Failure Rate			2415 bits

SUBSYSTEM: OSE, MDE in the DSIF

Equipment and/or Function: Command Encoder

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	1856	6	11136
Transistor, Silicon	409	20	8180
Resistor, Comp.	1671	2	3342
Resistor, Metal Film	119	10	1190
Capacitor, Ceramic	5	15	75
Capacitor, Mica	211	5	1055
Capacitor, Paper, Metalized	8	30	240
Capacitor, Tantalum	213	20	4260
Relays	5	480	2400
Transformers	4	120	480
Potentiometer	2	80	160
Oscillator	1	250	250
Total Failure Rate			32, 768 bits
		MTF =	30, 517

SUBSYSTEM: OSE, MDE in the DSIF

Equipment and/or Function: Computer Buffer

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	1051	6	6306
Diode, Zener	18	40	720
Transistor, Silicon	277	20	5540
Resistor, Comp.	1093	2	2186
Resistor, Metal Film	9	10	90
Resistor, W. W Prec.	30	80	2400

	Quantity	Failure Rate/Part	Category Failure Rate
Capacitor, Mica	110	5	550
Capacitor, Paper Mylar	17	30	510
Capacitor, Tantalum	130	20	2600
Relays	3	480	1440
Connector Pins	2200	10	22000
Capacitor, Trimmer	6	50	300
Diode, Tunnel	6	100	600
Transformers	3	120	360
Total Failure Rate			45,602 bits
		MTF =	21,929

SUBSYSTEM: OSE, MDE in the DSIF

Equipment and/or Function: Telemetry Detector (works with Computer Buffer)

Class of Usage: Digital

Parts Population Analysis

	Quantity	Failure Rate/Part	Category Failure Rate
Diode, Silicon	1256	6	7536
Diode, Zener	6	40	240
Transistor, Silicon	268	20	5360
Resistor, Comp.	1097	2	2194
Resistor, Metal Film	103	10	1030
Resistor, W.W Prec.	46	80	3680
Capacitor, Ceramic	2	15	30
Capacitor, Mica	210	5	1050
Capacitor, Paper, Mylar	4	30	120
Capacitor, Tantalum	136	20	2720
Inductors, Transformers	13	90	1170

	Quantity	Failure Rate/Part	Category Failure Rate
Relays	5	480	2400
Connector Pins	2500	10	25000
Capacitor, Alu. Elect.	3	150	450
Capacitors, Glass	5	3	15
Resistors, Variable	6	120	720
Total Failure Rate			53,715 bits
		MTF =	18,617

SUBSYSTEM: Solar Array

The array is 36 x 116. module matrix. Essential spacecraft power requirements at encounter plus one month dictate that at least 24 (or the equivalent power of 24) strings remain operable. This level embraces all worst case conditions including that of a earth-equivalent radiation model.

A simplified analysis demonstrates the array has a negligibly small probability of not being able to support the spacecraft power demands at encounter plus one month. This analysis is worst case becuase it assumes that when the minimum number of cell failures occur, they occur in the worst failure pattern.

The estimates are as follows:

Number of Cell Failures	Probability of Occurrence
0	0.5072
1	0.3446
2	0.1171
3	0.0265
4	0.0045
10	0.0001

The probability of eight or ten cell failures is $\ll 0.0001$. This number of cells failing in the worst failure pattern would cause the loss of one string out of 36. Thus the probability of losing 12 strings, (36 - 24) is negligibly small. The probability of the array supporting essential spacecraft power demands at encounter plus one month is >>> 0.99.

Encounter plus six months requires operation of 34 of the 36 strings to satisfy essential spacecraft power requirements. If we assume no failures, as an extreme case, complete reliability is based upon the risk of all 38976 cells in series. The probability of the array satisfying this requirement for a six month period is 0.204, predicated on the temperature cyclic and non-cyclic failure rates of 1×10^{-8} failure per hour cell and 3×10^{-9} failure per hour per cell, respectively. On the basis of power reduction over a non-failed array, the power loss is less than 1%. Since the power margin is in excess of 5% the reliability will greatly exceed 0.204.

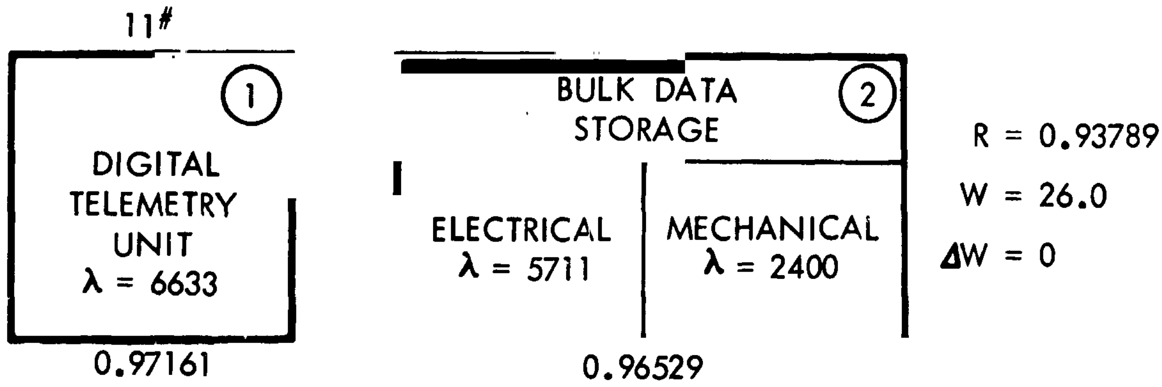
With an expected mean cell loss of less than 3 for the 36 string configuration, increasing the array string configuration to 39, still does not increase the expected mean cell loss to more than 3, since the nearest whole number of cells is 3 in both instances. Where the 36 string configuration may not hypothetically afford a single cell failure, the 33 string configuration could support the mission with as many cell failures as are in three strings. However, the discrete strings are not expected to fail. Instead, as few as 30 cell failures may in the worst failure pattern, have the same effect. Thus, determing the probability of not more than 30 cell failures occurring, which is more than ten times the mean, yields a preliminary estimate for 36 string array reliability.

The solar array reliability assessment has indicated a high probability, 99.9%, that the array will support spacecraft power demands at encounter plus one month. At six months after encounter, the array has a lower probability of supporting the spacecraft loads (99%) due to an increasing array peak power profile demand.

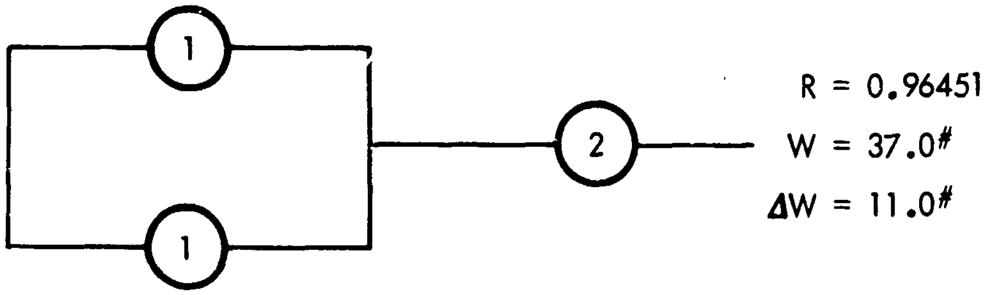
3. RELIABILITY MODELS AND COMPUTATIONS FOR REDUNDANCY OPTIONS, ELECTRICAL EQUIPMENT

Final Assessment (4280 Hours) Telecommunications, Data Handling Unit

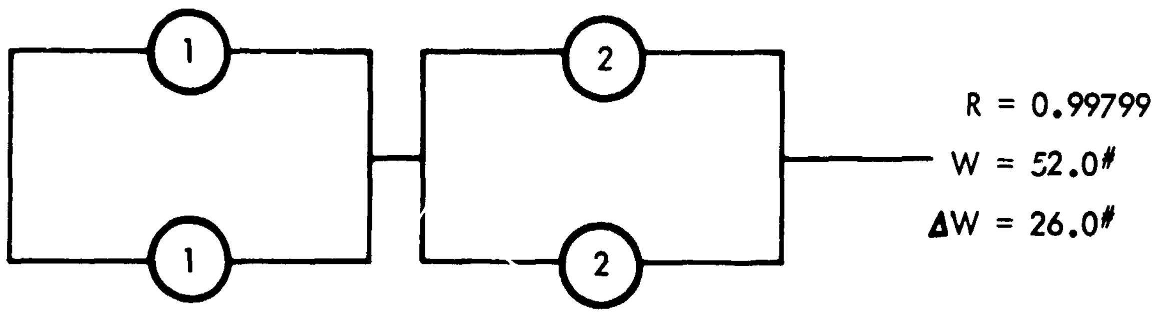
OPTION #0



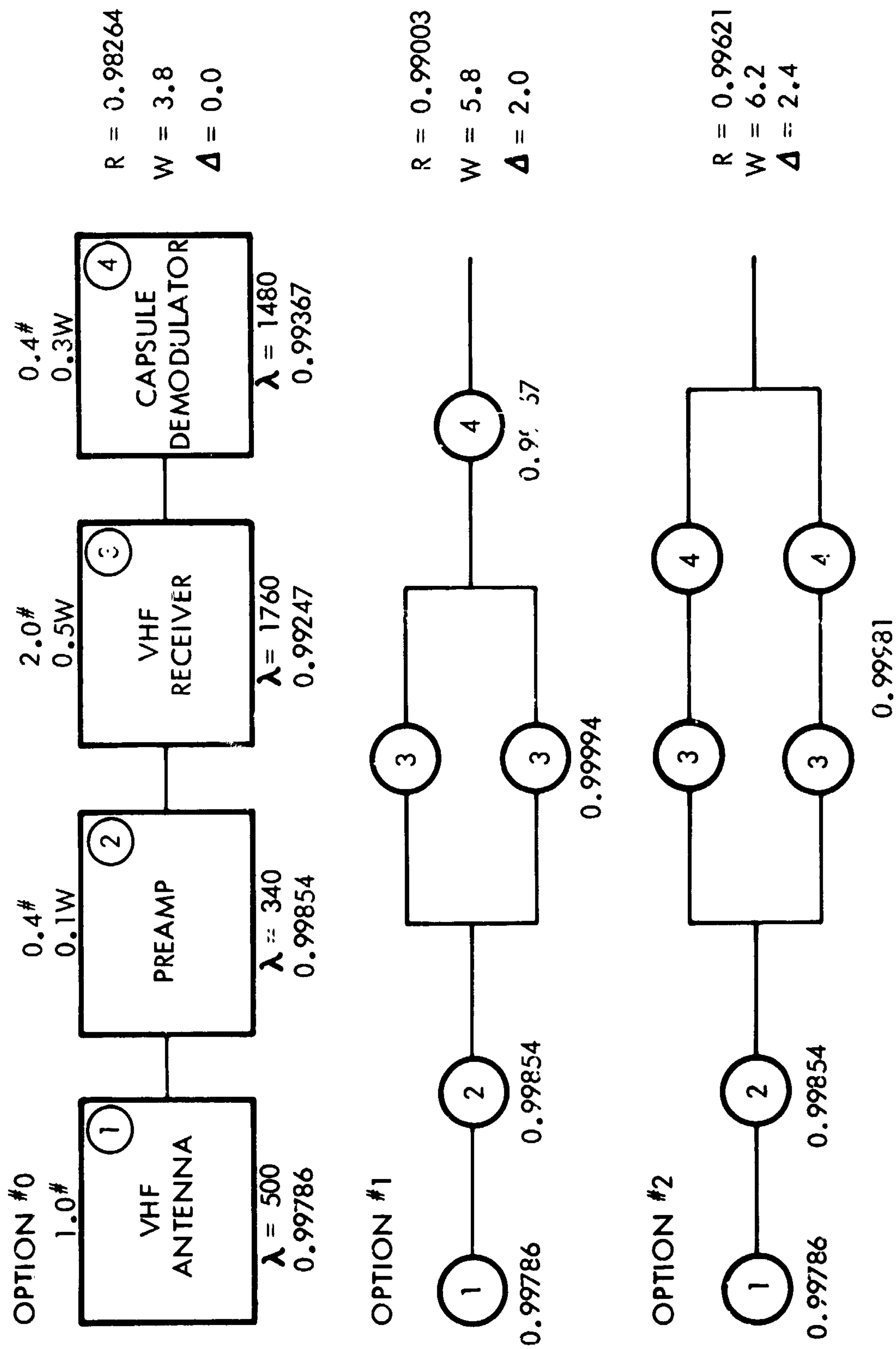
OPTION #1



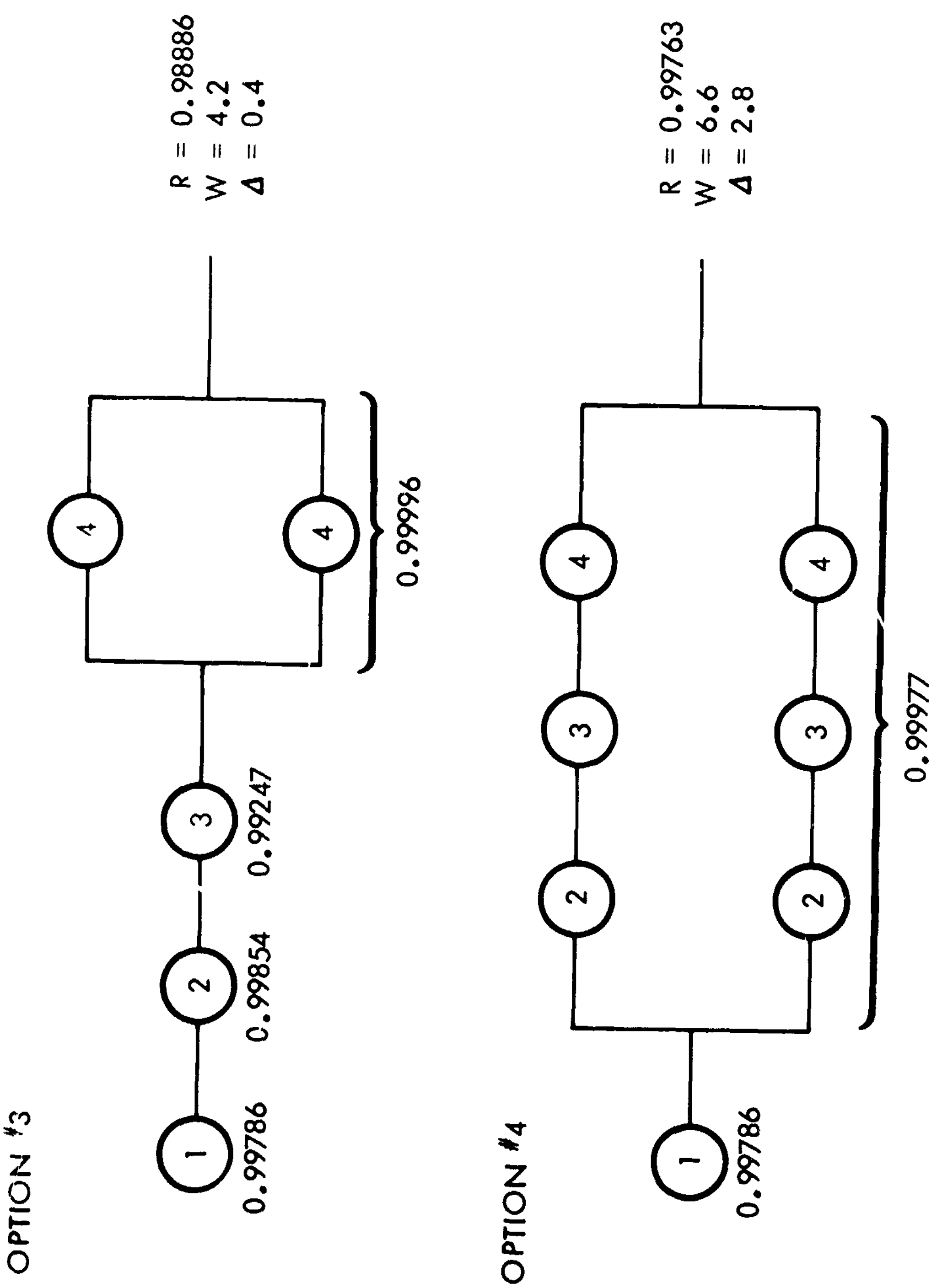
OPTION #2



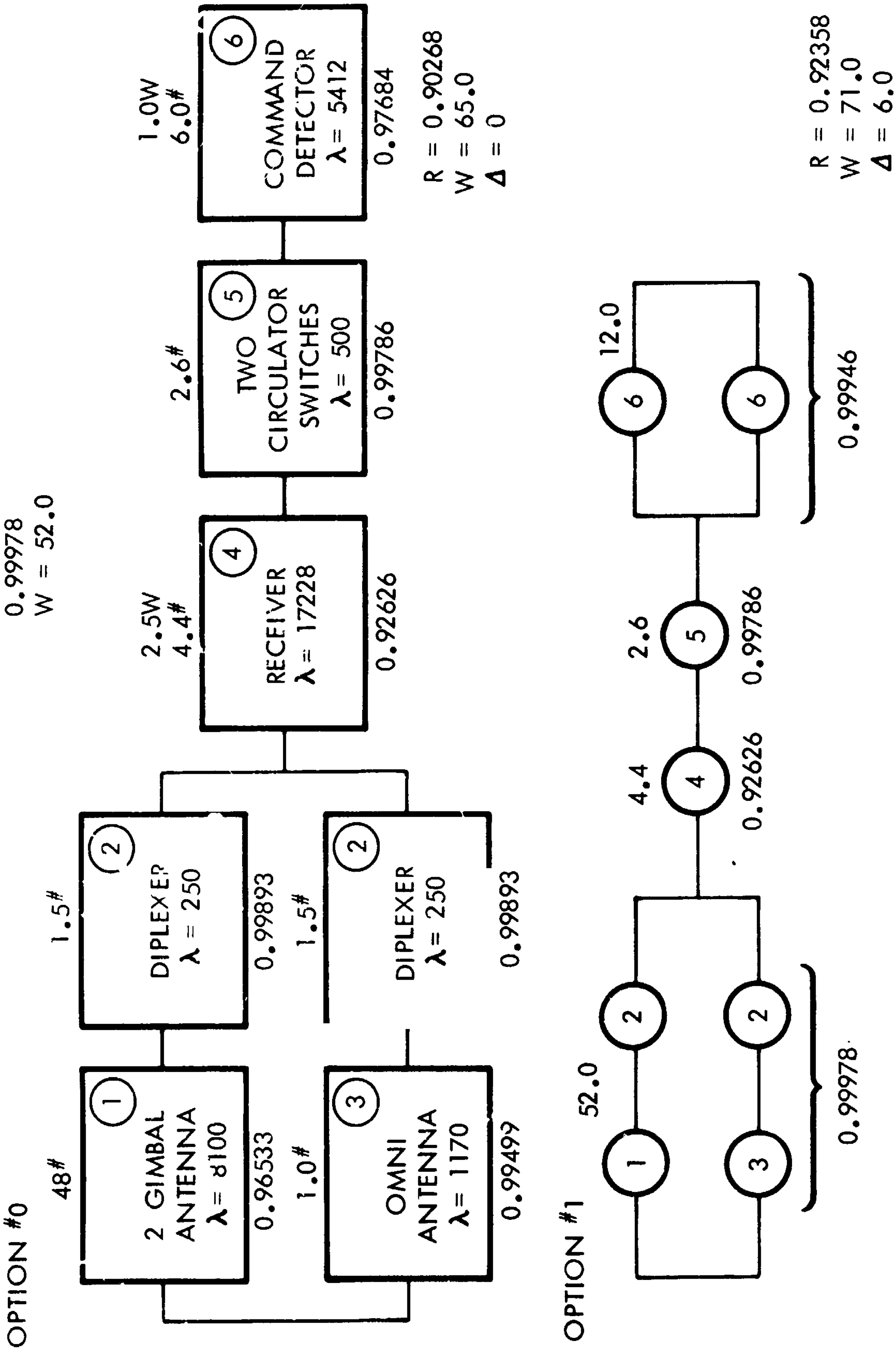
Final Assessments (4280 Hours) Telecommunications,
Capsume Receiver, VHF

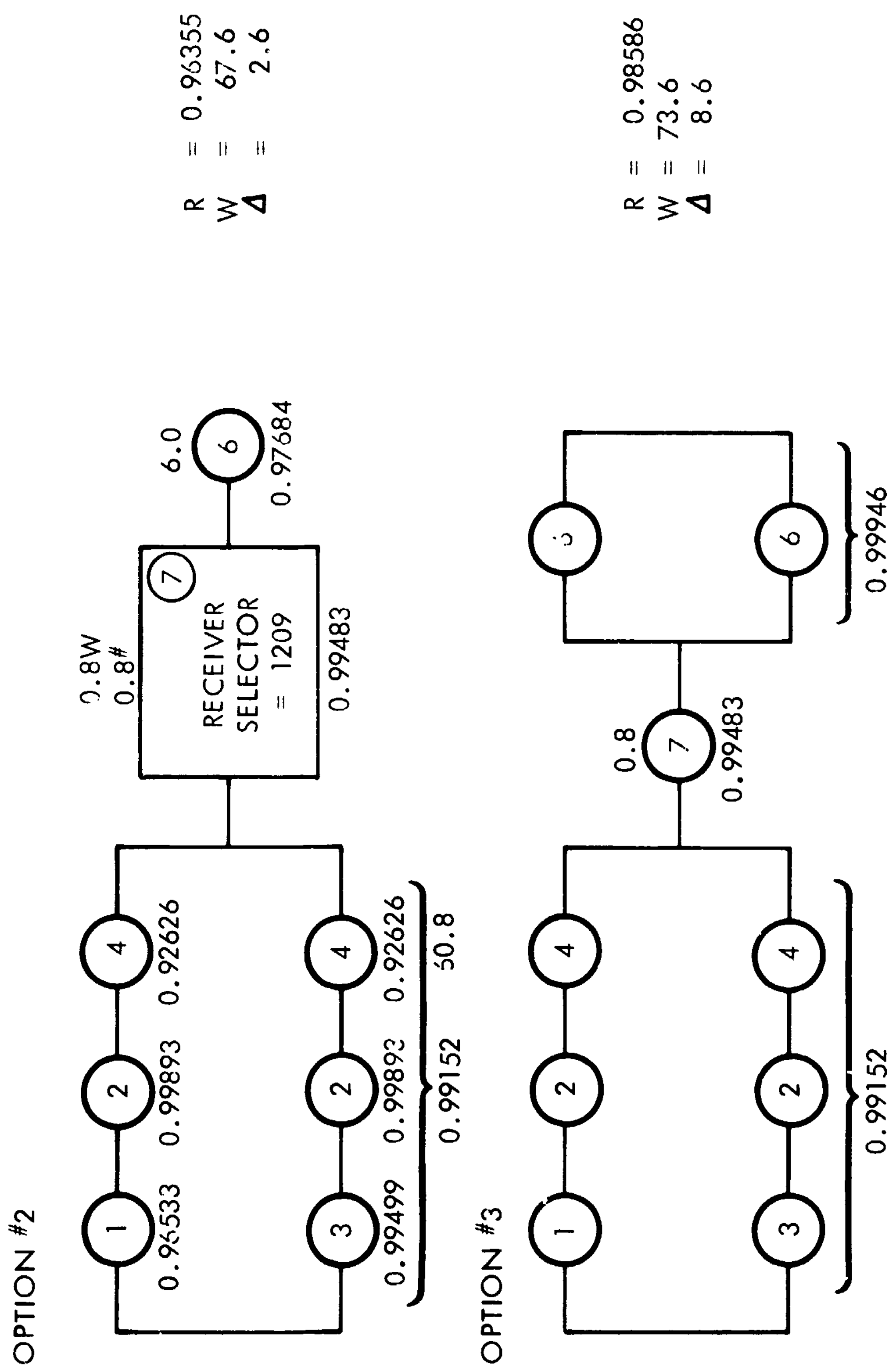


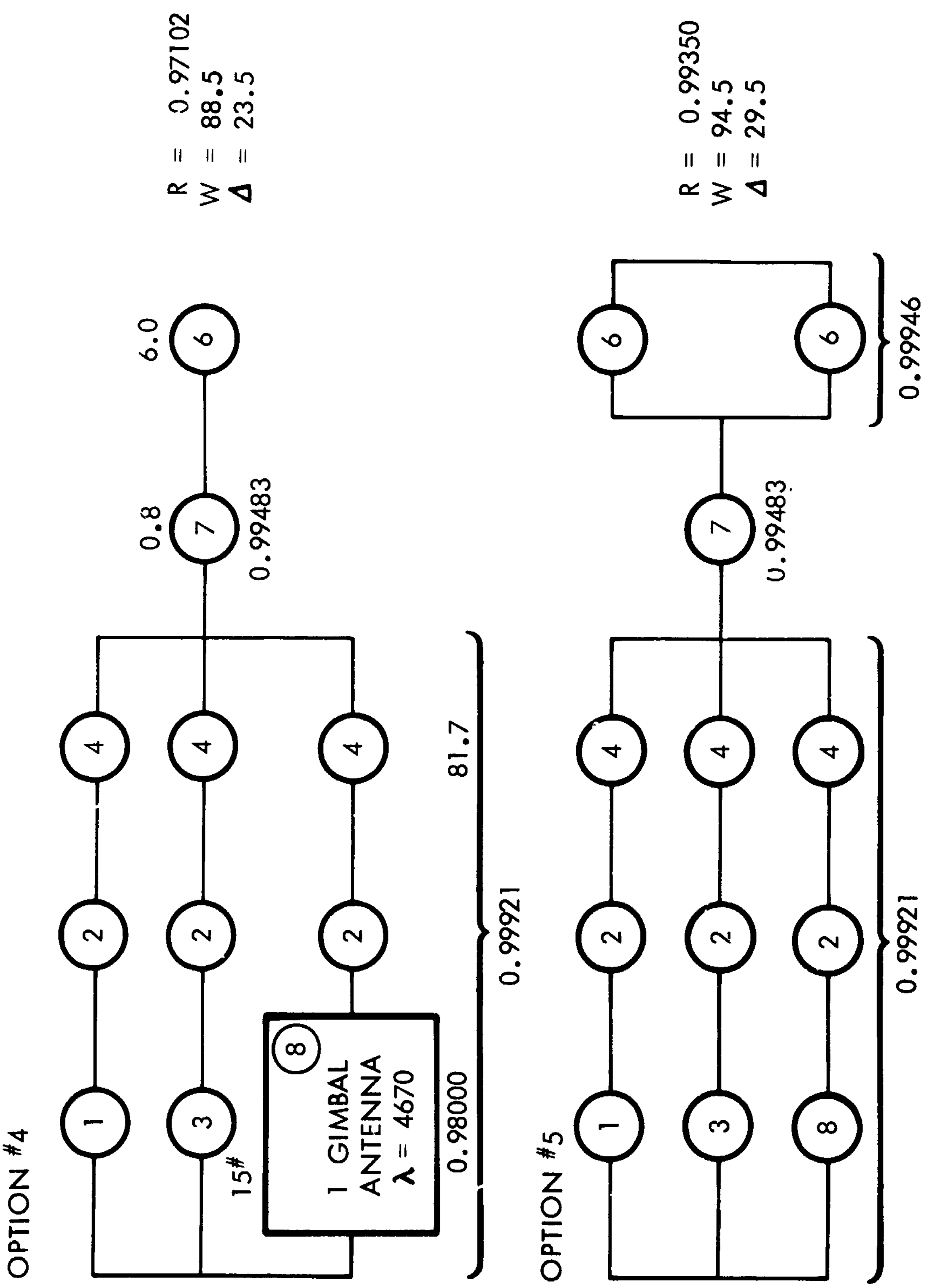
Telecommunications, Capsule Receiver (continued)



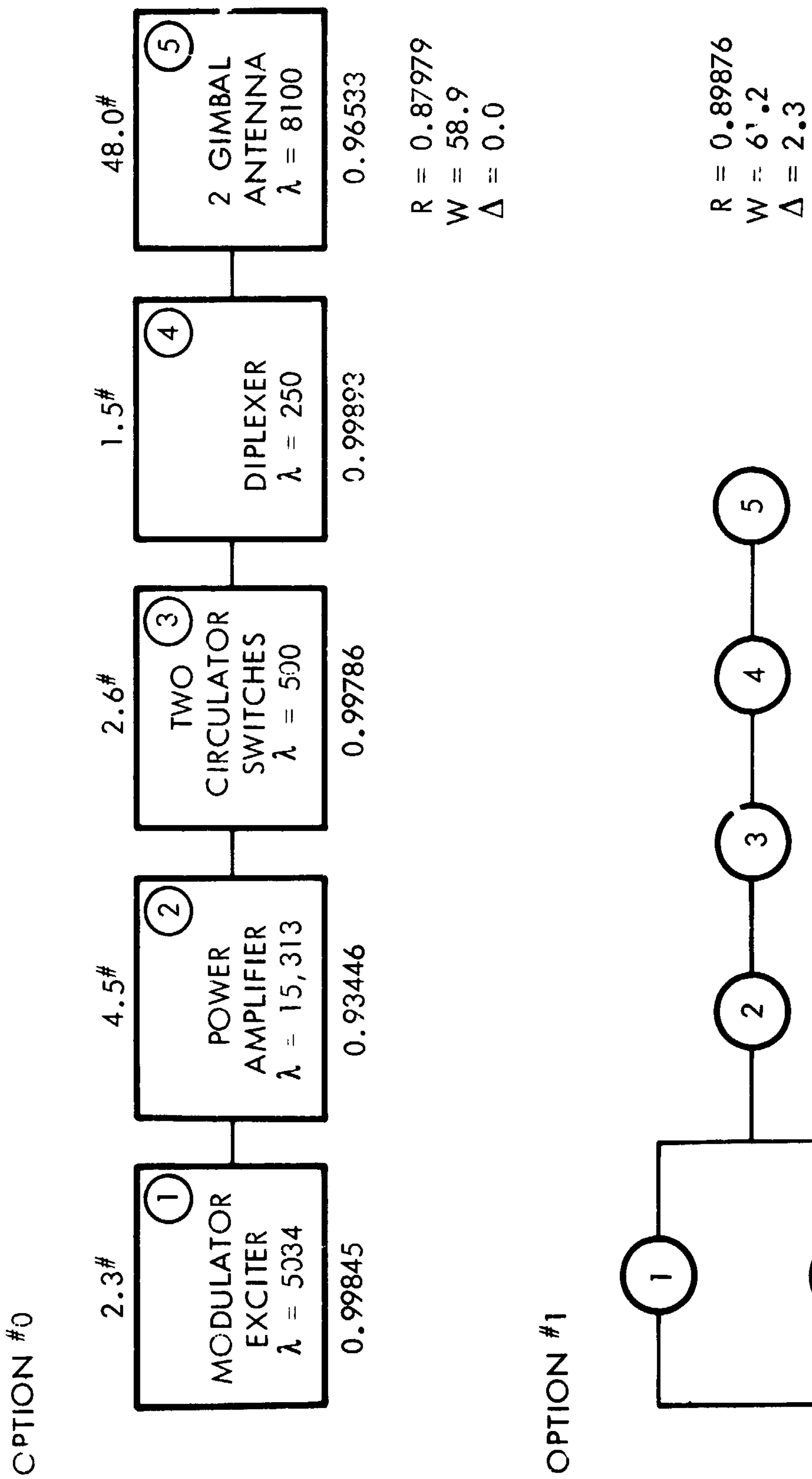
Final Assessment (4280 Hours) Telecommunications, Receiver - S-Band

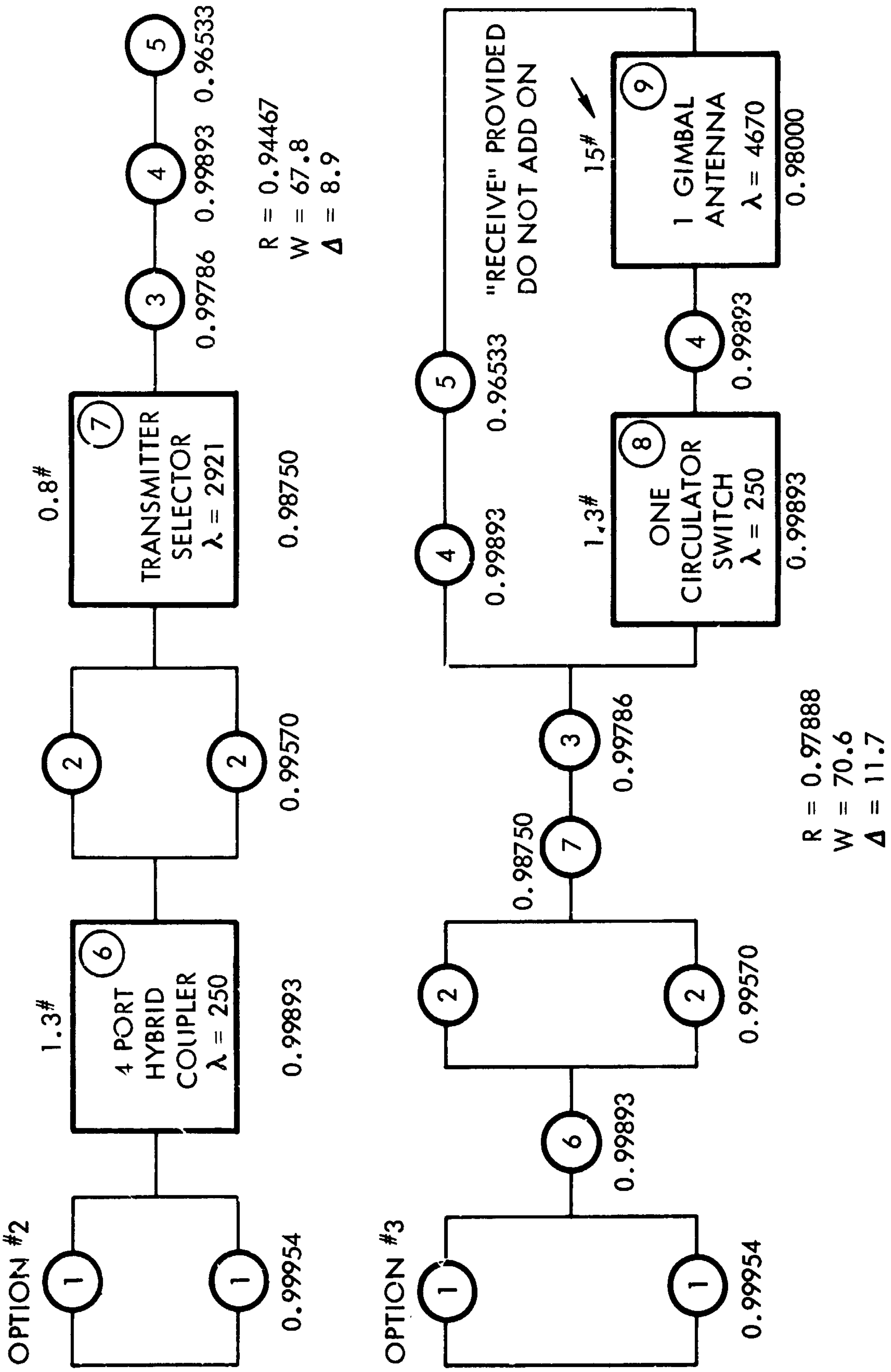


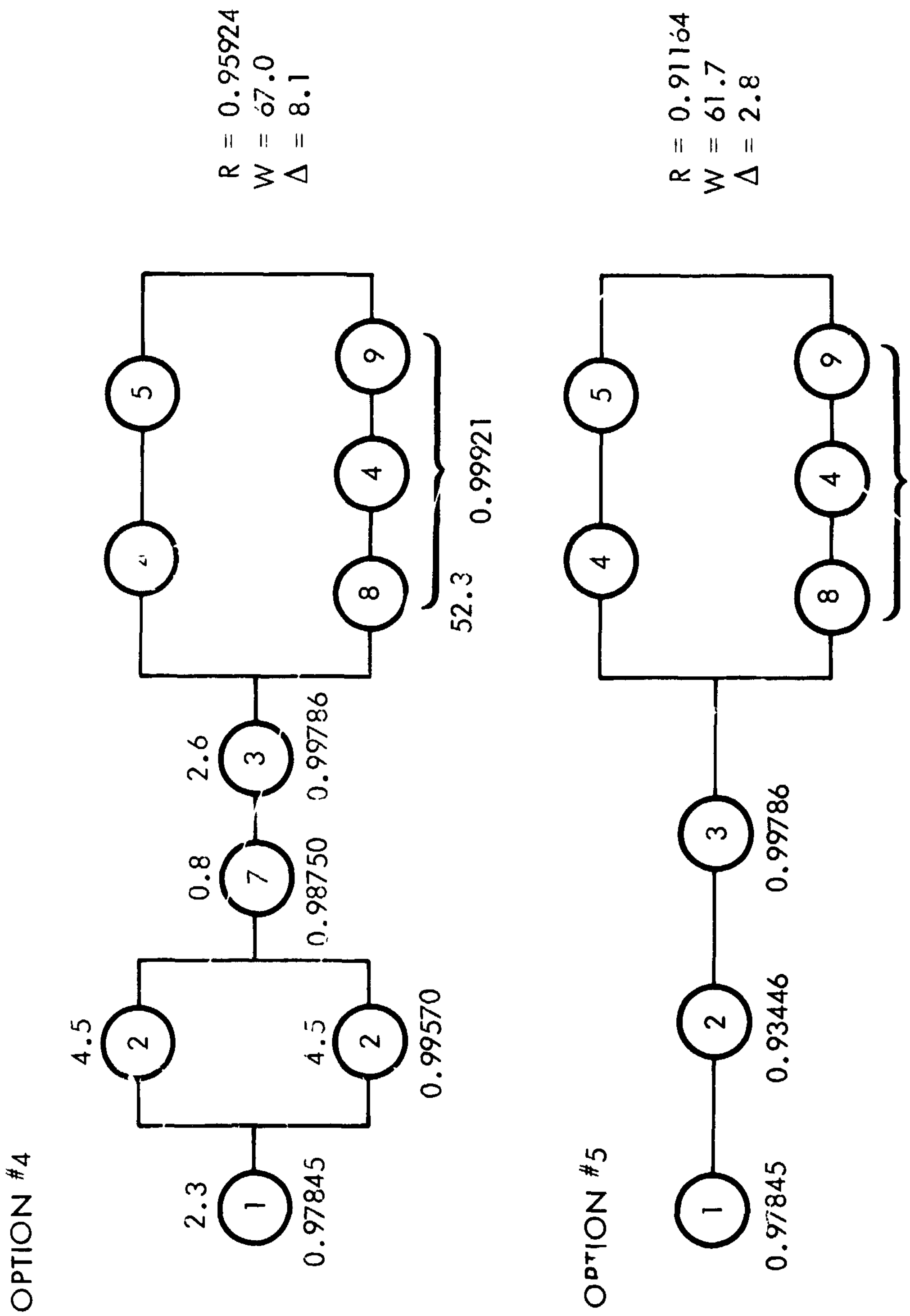

OPTION #2
1
0.95533
2
0.99893
4
0.92626
3
0.99499
2
0.99893
4
0.92626
0.99152
50.8
0.8W
0.8#
7
RECEIVER
SELECTOR
= 1209
0.99483
6.0
6
0.97684
R = 0.96355
W = 67.6
Δ = 2.6
OPTION #3
1
2
4
3
2
4
0.99152
0.8
7
0.99483
6
6
0.99946
R = 0.98586
W = 73.6
Δ = 8.6


OPTION #4
15#
1 GIMBAL ANTENNA
λ = 4670
0.98000
0.99921
81.7
0.8
0.99483
6.0
R = 0.97102
W = 88.5
Δ = 23.5
OPTION #5
0.99921
0.99483
0.99946
R = 0.99350
W = 94.5
Δ = 29.5

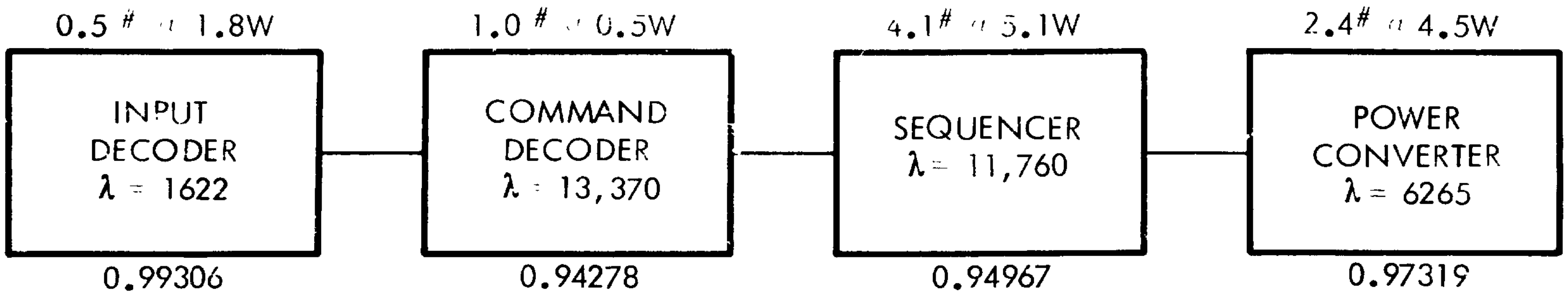
Final Assessment (4280 Hours) Telecommunications, Transmitter




OPTION #2
0.99954
1.3#
4 PORT HYBRID COUPLER λ = 250
0.99893
0.99570
0.8#
TRANSMITTER SELECTOR λ = 2921
0.98750
0.99786
0.99893
0.96533
R = 0.94467
W = 67.8
Δ = 8.9
OPTION #3
0.99954
0.99893
0.99570
0.98750
0.99786
0.99893
0.96533
"RECEIVE" PROVIDED DO NOT ADD ON
1.3#
ONE CIRCULATOR SWITCH λ = 250
0.99893
0.99893
15#
1 GIMBAL ANTENNA λ = 4670
0.98000
R = 0.97888
W = 70.6
Δ = 11.7

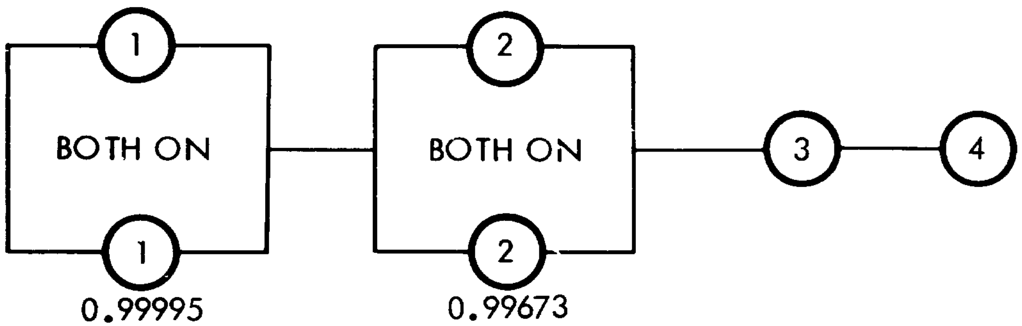

OPTION #4
2.3
0.97845
4.5
4.5
0.99570
0.8
0.98750
2.6
0.99786
52.3
0.99921
R = 0.95924
W = 67.0
Δ = 8.1
OPTION #5
0.97845
0.93446
0.99786
0.99921
R = 0.91164
W = 61.7
Δ = 2.8

OPTION #0 (CENTRALIZED MEMORY, TYPE)



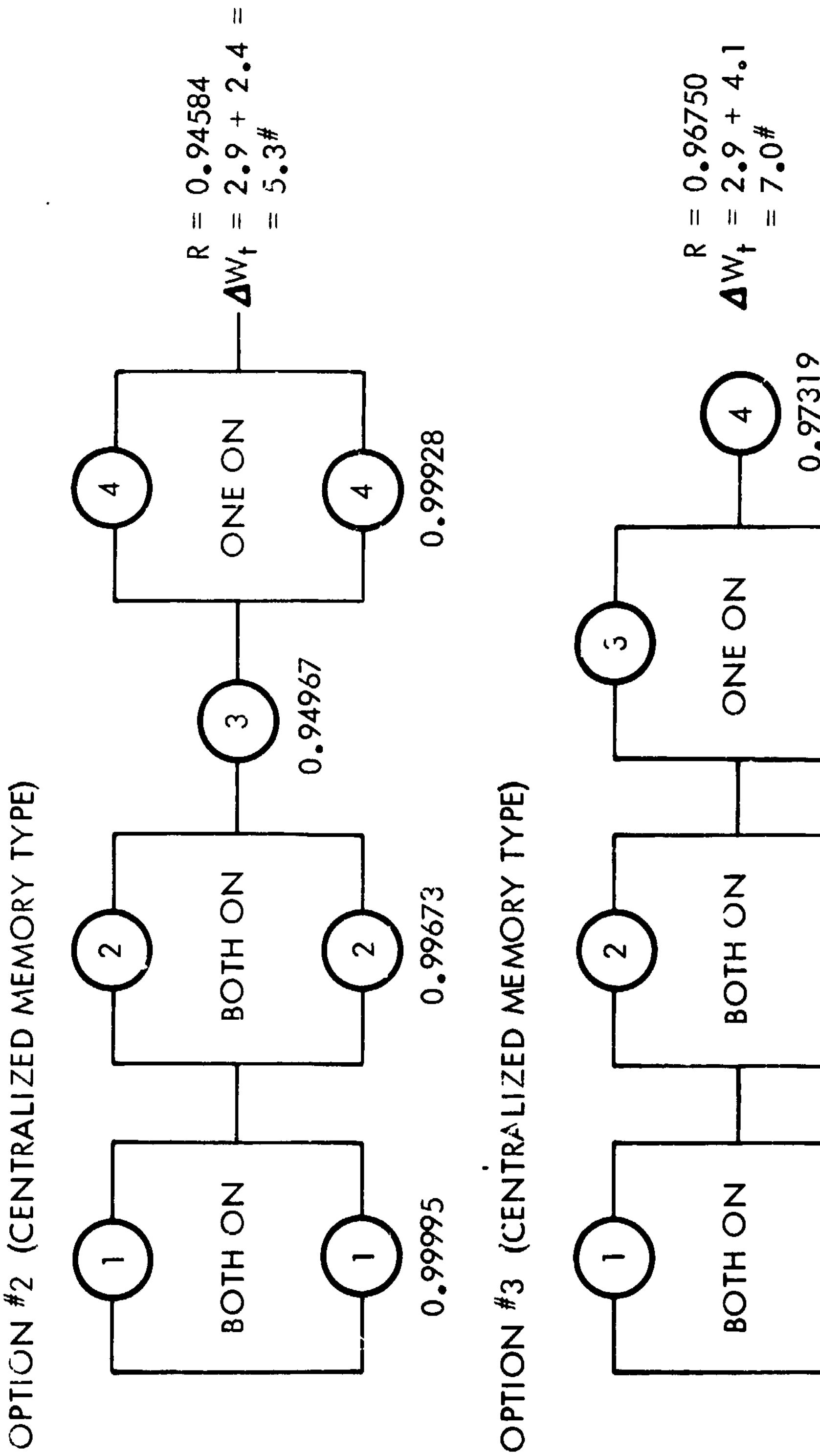
R = 0.86528
W = 8.0#
POWER 11.9 WATTS
ΔWt = 0

OPTION #1 (CENTRALIZED MEMORY TYPE)

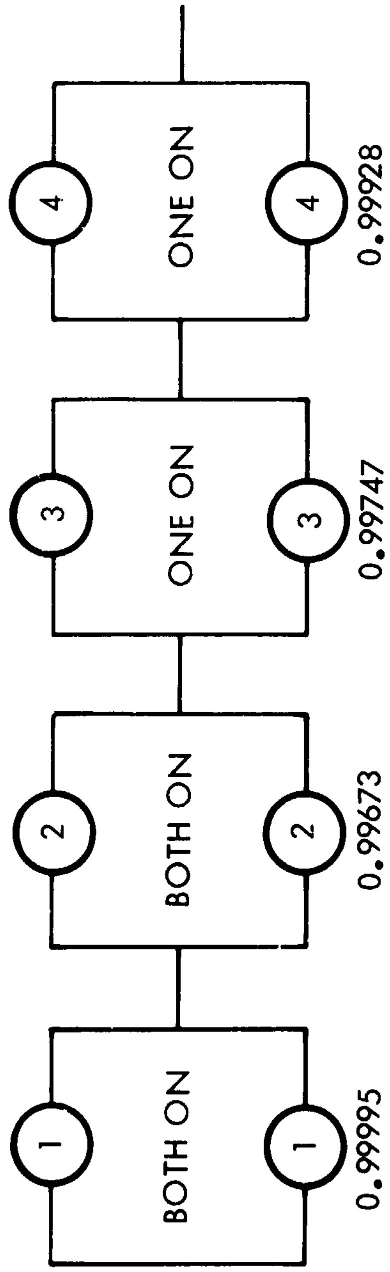


R = .92115
ΔPOWER = 2.3 W
ΔWt = 1.5 +
(2.3) (.6) = 2.9

TOTAL ΔWt = ΔWt + 0.6 (POWER - 11.9) AND PLOTTED AGAINST RELIABILITY FOR WEIGHT TRADE-OFF


OPTION #2 (CENTRALIZED MEMORY TYPE)
1
BOTH ON
1
0.99995
2
BOTH ON
2
0.99673
3
0.94967
4
ONE ON
4
0.99928
R = 0.94584
ΔWt = 2.9 + 2.4 =
= 5.3#
OPTION #3 (CENTRALIZED MEMORY TYPE)
1
BOTH ON
1
0.99995
2
BOTH ON
2
0.99673
3
ONE ON
3
0.99747
4
0.97319
R = 0.96750
ΔWt = 2.9 + 4.1
= 7.0#

OPTION #4 (CENTRALIZED MEMORY TYPE)



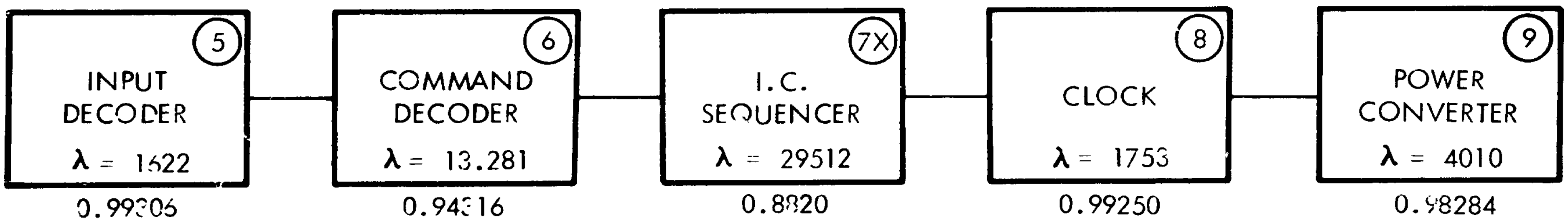
$R = 0.99343$

$\Delta W_t = 2.9 + 2.4 + 4.1$

$\Delta W_t = 9.4\#$

OPTION #5

DISTRIBUTED MEMORY TYPE



R = 0.8058
W(total) = 19.8#
Power = 50.1W

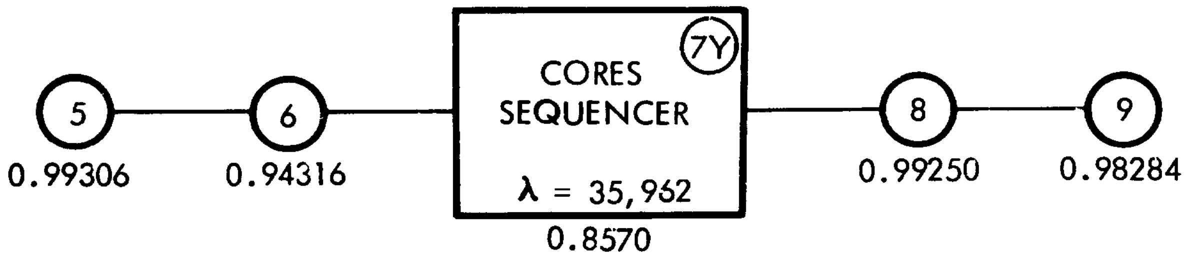
WEIGHT INCREASE ABOVE OPTION #0 (8.0#) ——— 11.8#
POWER INCREASE ABOVE OPTION #0 (11.9W) = 38.2W
EQUIVALENT WEIGHT INCREASE (0.6)(38.2) ——— 22.9

34.7#

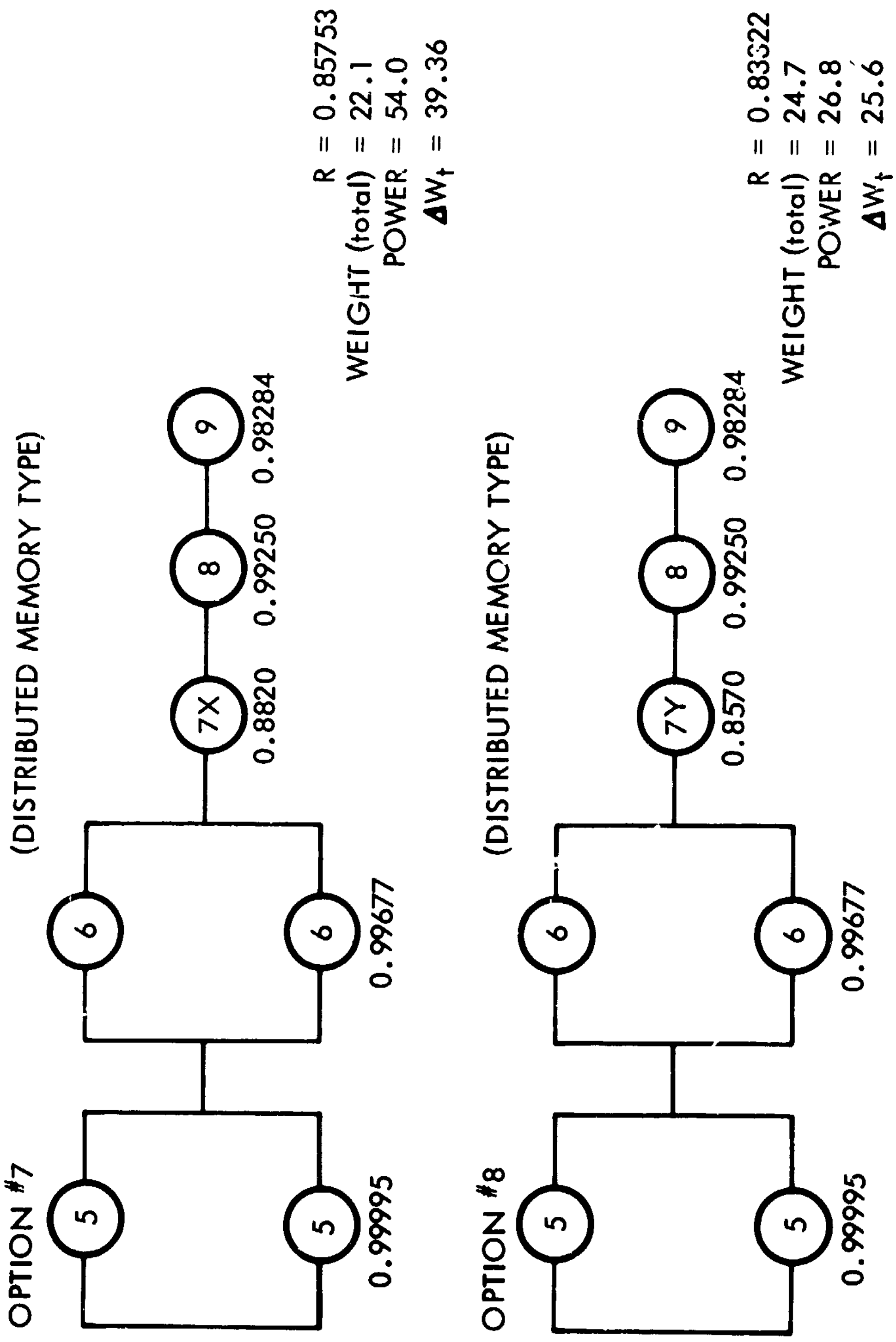
ΔW_t = 34.7

OPTION #6

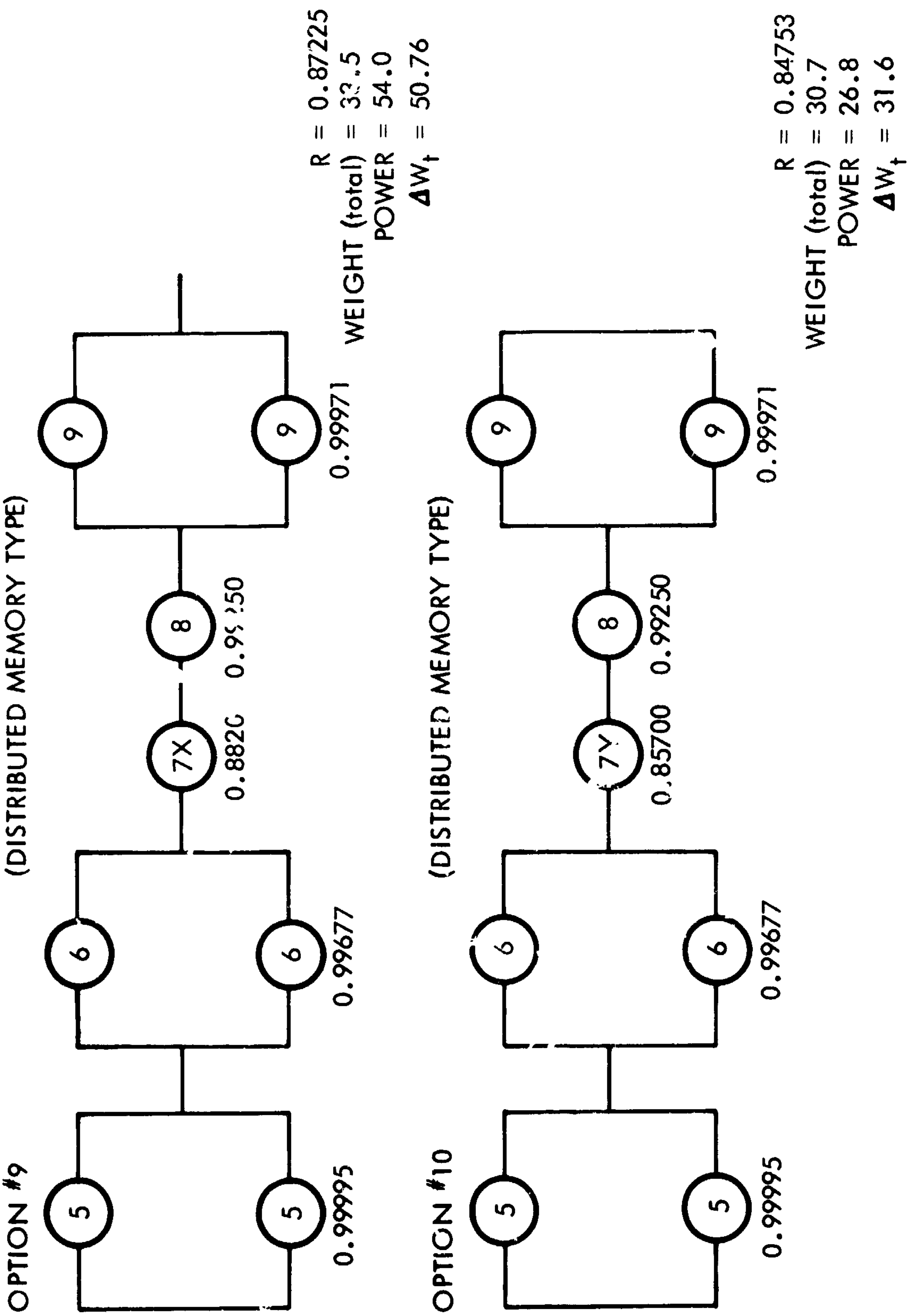
DISTRIBUTED MEMORY TYPE



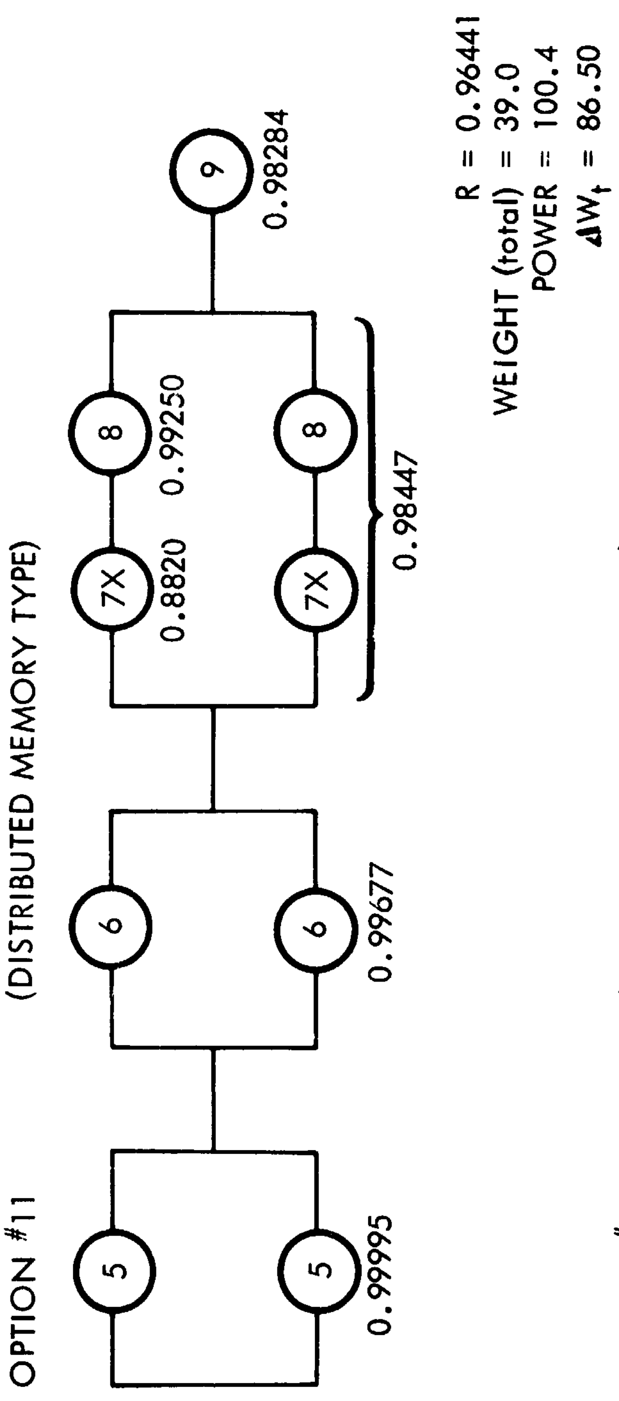
R = 0.78299
W(total) = 22.4#
POWER = 22.9W
ΔW_t = 21.0

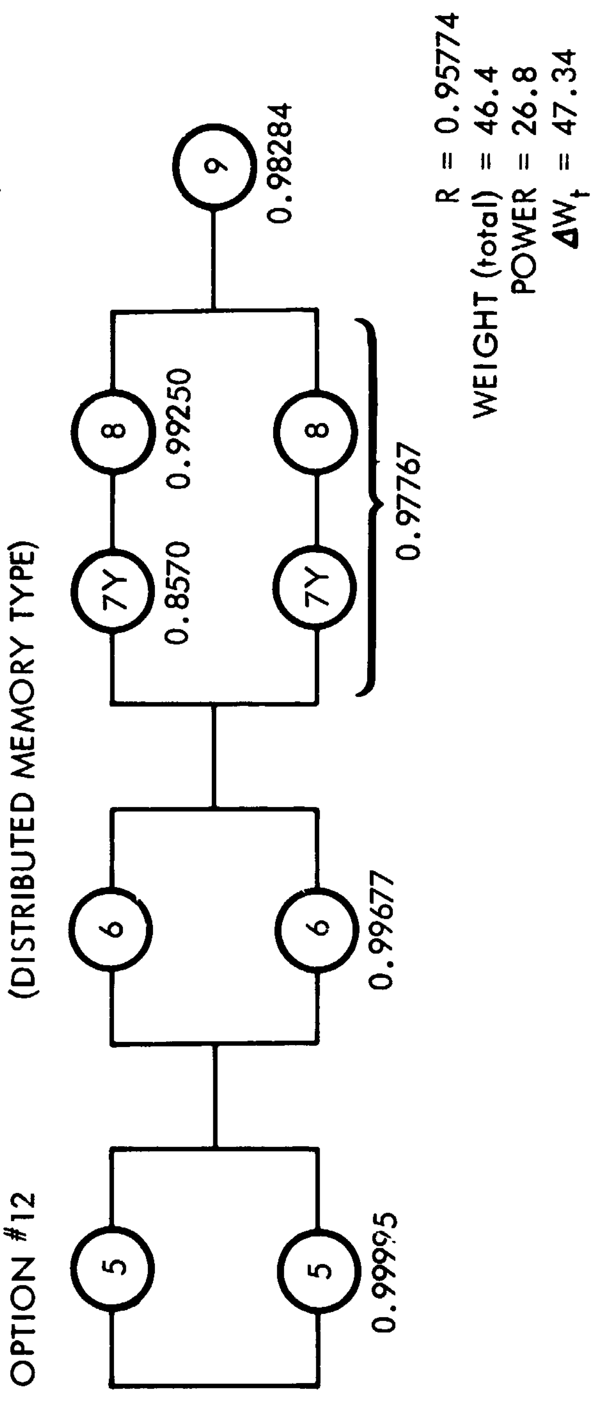

OPTION #7
(DISTRIBUTED MEMORY TYPE)
5
5
0.99995
6
6
0.99677
7X
0.8820
8
0.99250
9
0.98284
R = 0.85753
WEIGHT (total) = 22.1
POWER = 54.0
ΔWt = 39.36
OPTION #8
(DISTRIBUTED MEMORY TYPE)
5
5
0.99995
6
6
0.99677
7Y
0.8570
8
0.99250
9
0.98284
R = 0.83322
WEIGHT (total) = 24.7
POWER = 26.8
ΔWt = 25.6

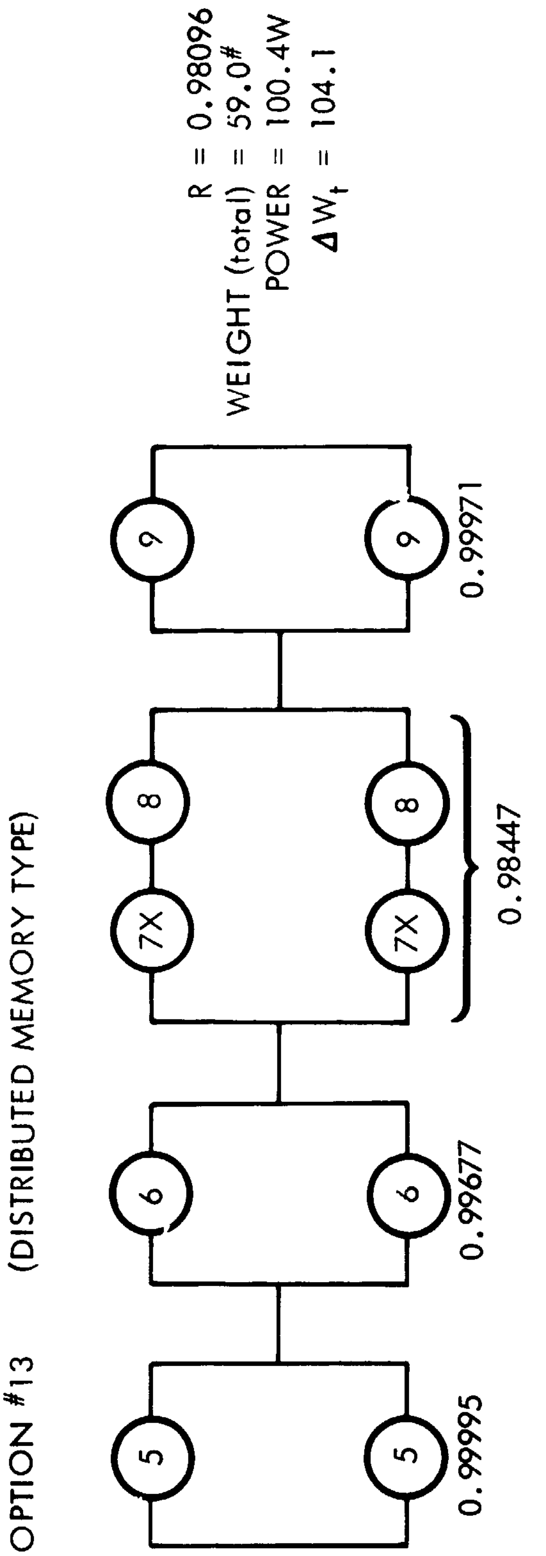
CS & C (Continued)



CS & C (Continued)



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5
0.99995
6
6
0.99677
7X
8
7X
8
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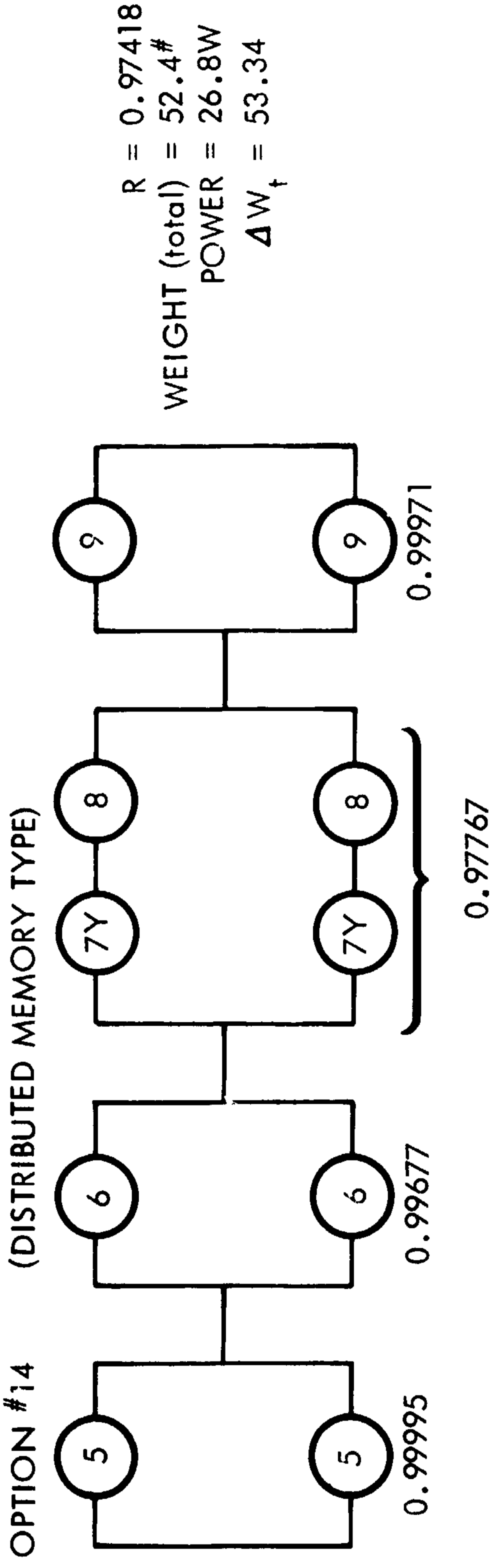
9
9
0.99971
R = 0.98096
WEIGHT (total) = 59.0#
POWER = 100.4W
ΔW_t = 104.1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7Y
8
7Y
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9
9
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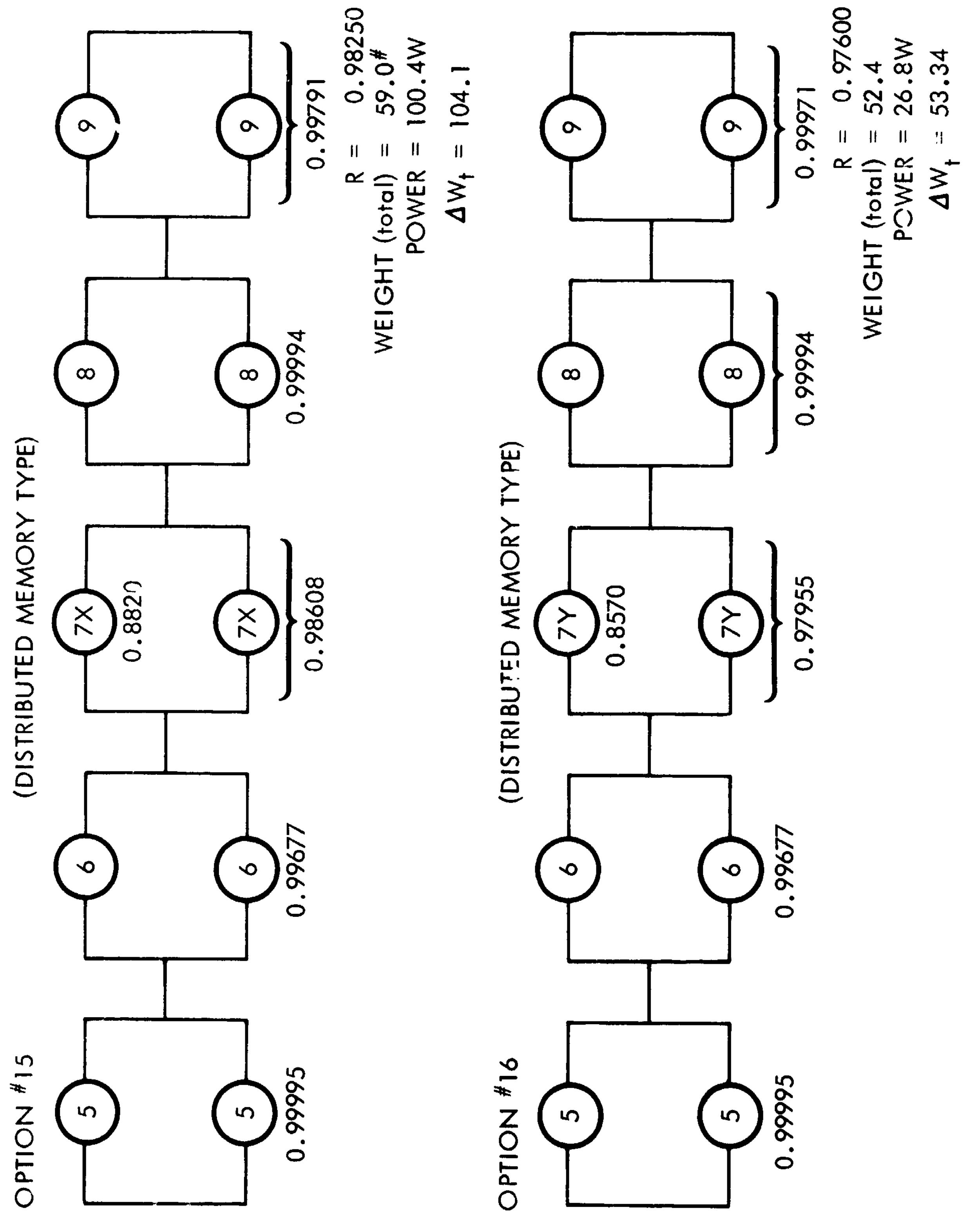
WEIGHT (total) = 52.4#
POWER = 26.8W
ΔW_t = 53.34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(DISTRIBUTED MEMORY TYPE)
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8
8
0.99994
9
9
0.99791
R = 0.98250
WEIGHT (total) = 59.0#
POWER = 100.4W
ΔWt = 104.1
OPTION #16
(DISTRIBUTED MEMORY TYPE)
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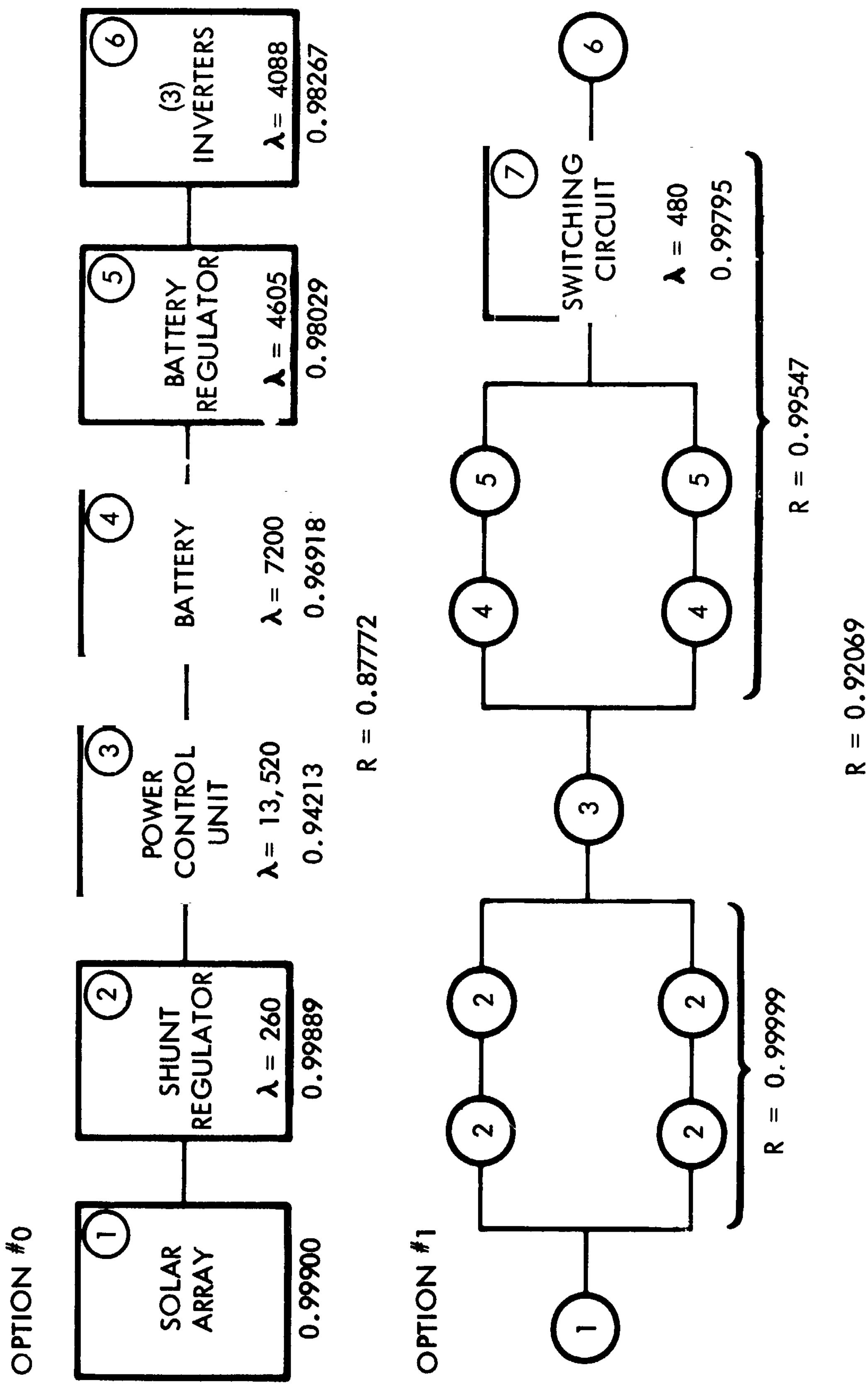
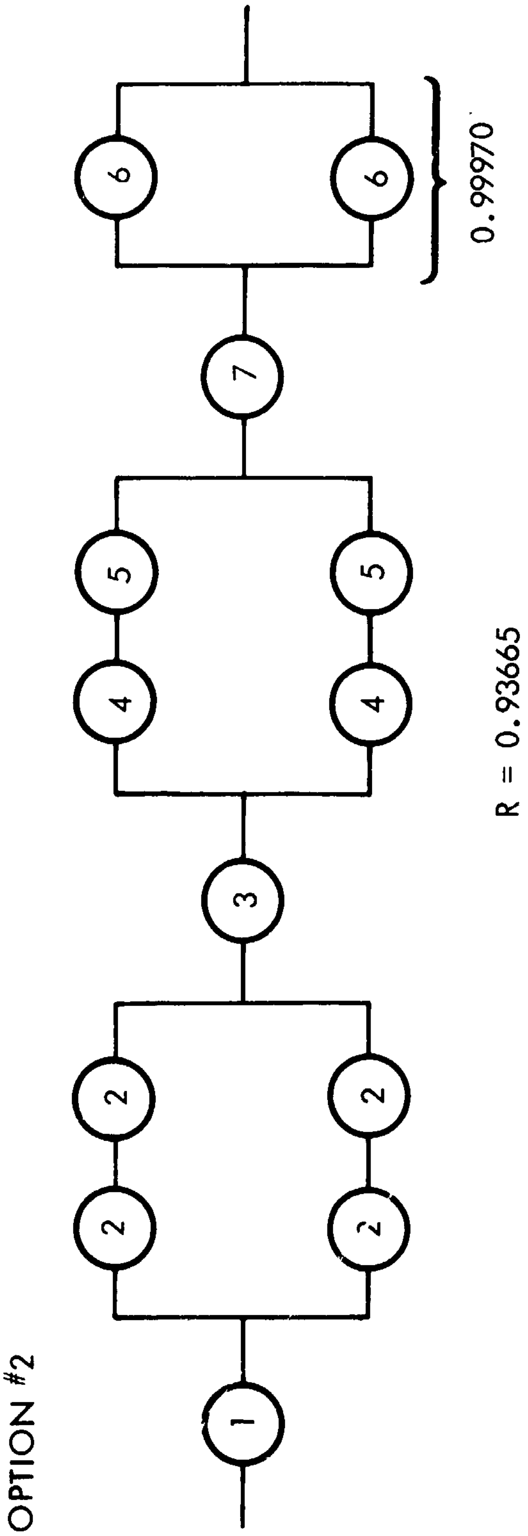
6
6
0.99677
7Y
0.8570
7Y
0.97955
8
8
0.99994
9
9
0.99971
R = 0.97600
WEIGHT (total) = 52.4
POWER = 26.8W
ΔWt = 53.34

Final Assessments (4280 Hours) Power Subsystem

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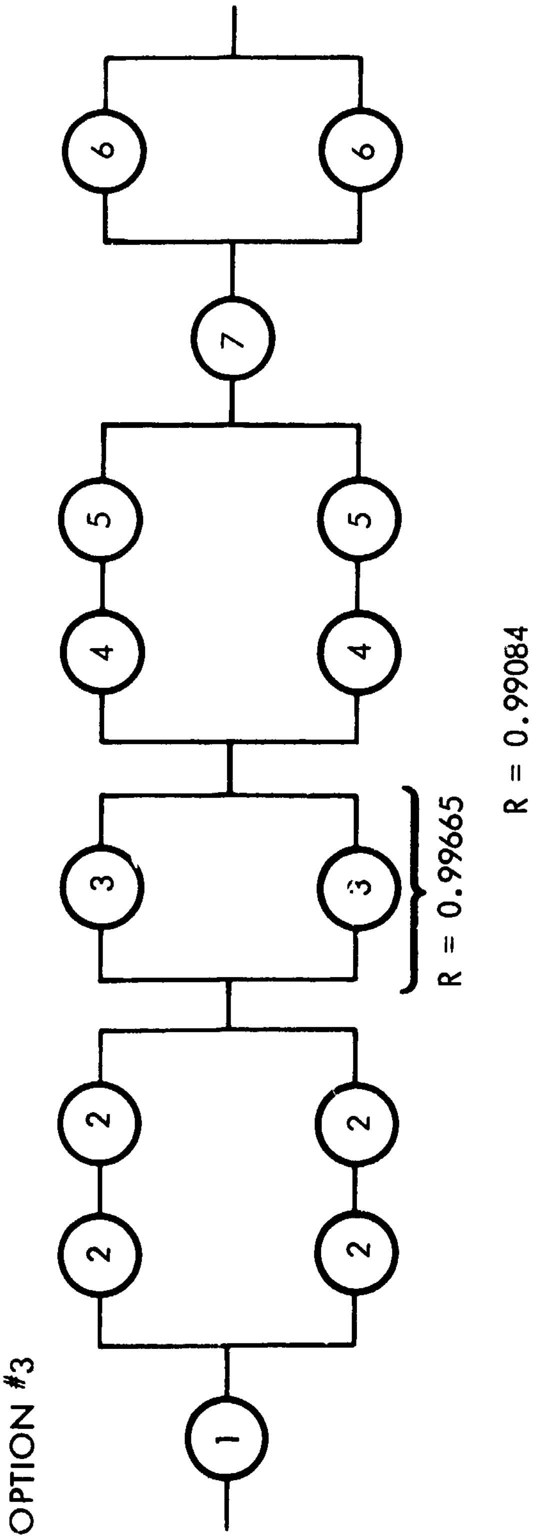
OPTION #3
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R = 0.99084

4. MECHANICAL EQUIPMENT ASSESSMENTS

4.1 Separation and Destruct Subsystem

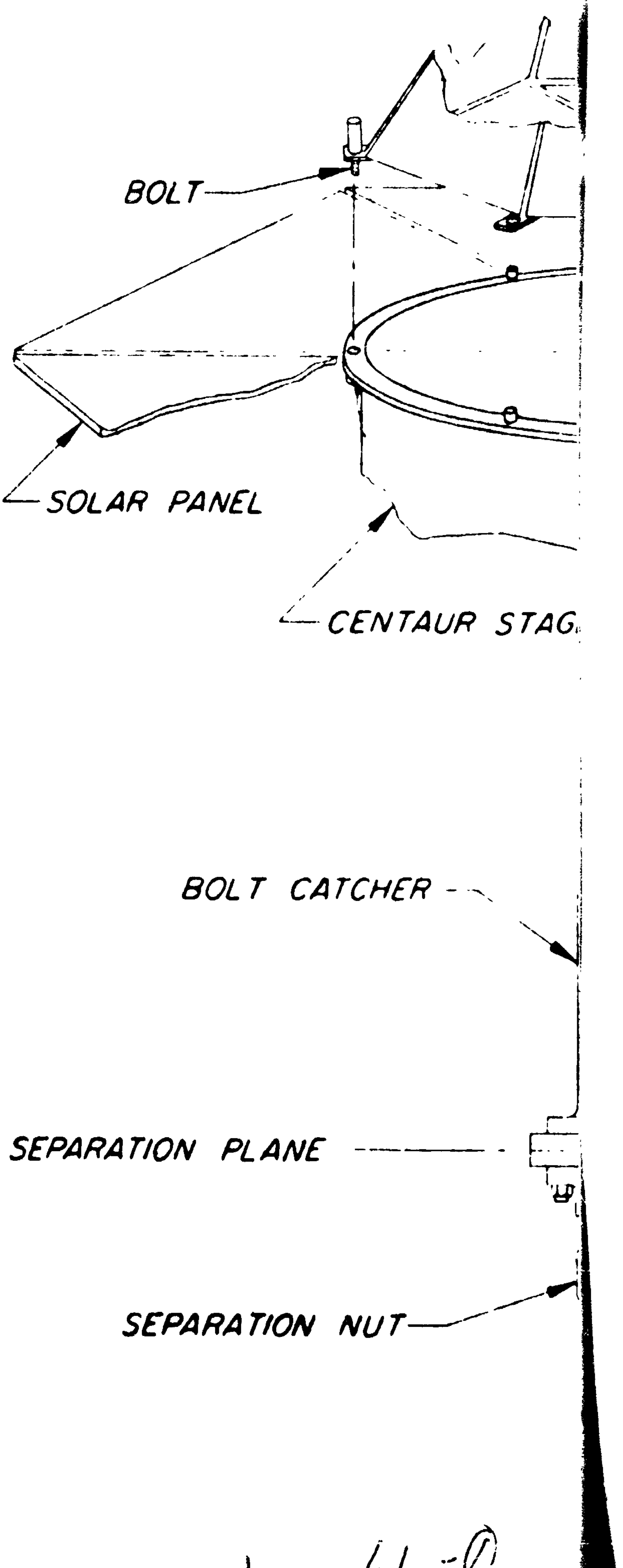
A reliability analysis was accomplished to predict the reliability of various configurations of separation planes and the spacecraft destruct system. The configurations considered are contained in the attached Douglas drawings (SK-BM00-5000 through SK-BM00-5003). Each of the separation planes and destruct systems is considered separately in the following sections. In addition, the reliability analysis of each major component and the mathematical models used in the analysis are attached.

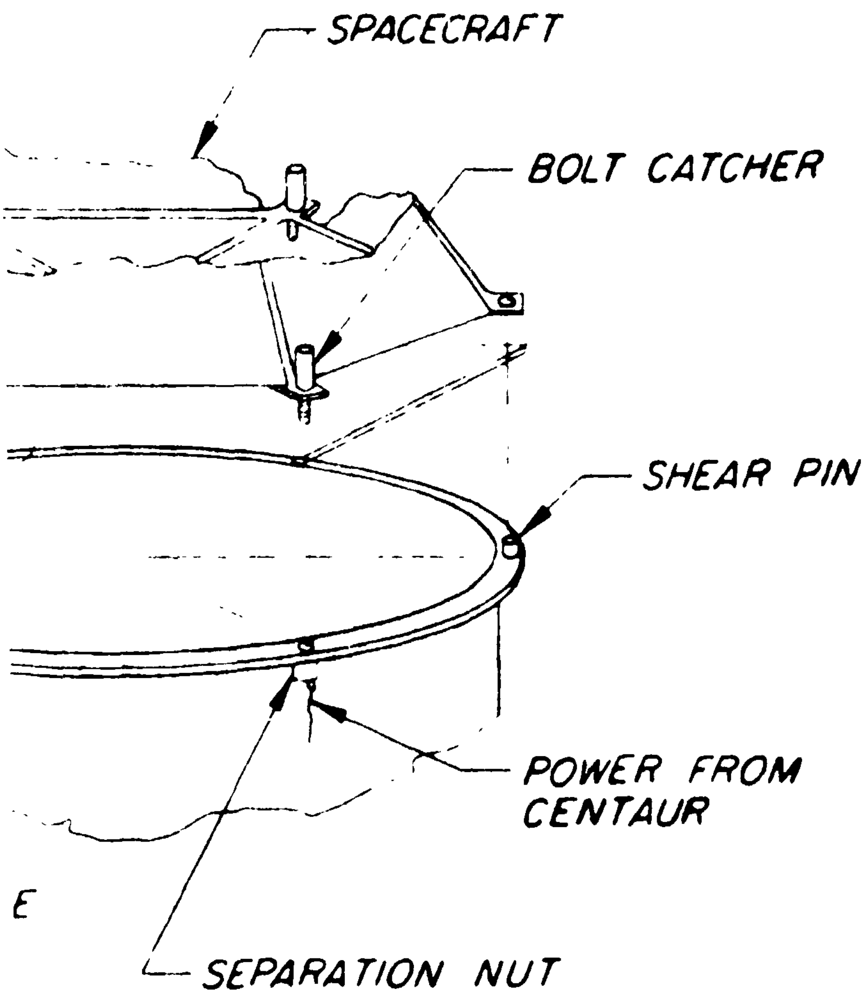
Table 1 presents a summary of the reliability predictions and a brief description of each subsystem considered. The reliability of the separation systems become limited by the probability of the squibs firing prematurely since the squibs do not act redundantly in this mode but in series.

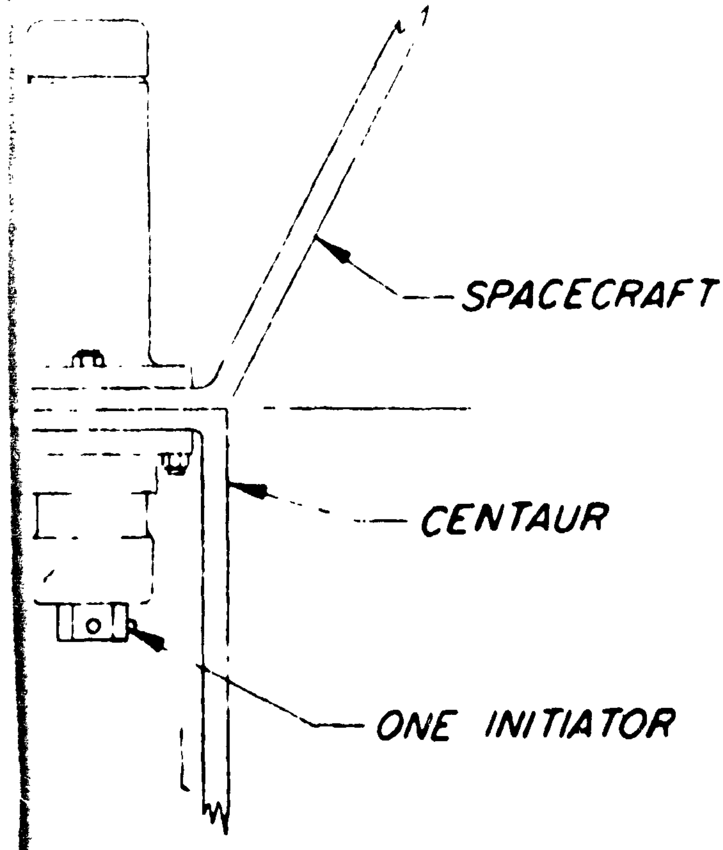
4.1.1 Separation Configuration A1 and A2 (SK-BM00-5000)

This configuration is to be used with the monopropellant system and has three attach points. Table 2 lists the reliability predictions for the three configurations considered. The six squibs are considered to be redundant sets of two in each nut such that the firing of either or both would permit separation.

The "B" configuration is similar except that the separation nuts contain only one squib each. The "A" configuration provides a 100% back-up at each attached point using a separate method to separate the spacecraft-pinpullers. Although Table 2 shows configuration "A" to have the highest predicted reliability, the weight is more than doubled over configuration"R". In configuration "A", if both systems separated at each point, a considerable amount of debris would be loose in the vicinity of the spacecraft. In addition, if only the pinpullers released, considerable weight would be added to the spacecraft since a portion of the LV would be separated with it.


BOLT
SOLAR PANEL
CENTAUR STAG
BOLT CATCHER
SEPARATION PLANE
SEPARATION NUT


SPACECRAFT
BOLT CATCHER
SHEAR PIN
POWER FROM CENTAUR
E
SEPARATION NUT


SPACECRAFT
CENTAUR
ONE INITIATOR

I. SEPARATION SYSTEMS

1.0 ANALYSIS SUMMARY OF SEPARATION METHOD SELECTION

	Wt.	Simplicity	Availability	Shock	Gassing	Rel.	Safety	Total
Ex. Bolt	10	9	10	2	7	8	7	53
Sep. Nut	9	8	9	9	9	9	9	62
Collect	6	6	5	10	10	9	10	56
Pin Puller	8	8	10	9	9	9	9	62
FLSC	10	10	10	1	6	10	2	49

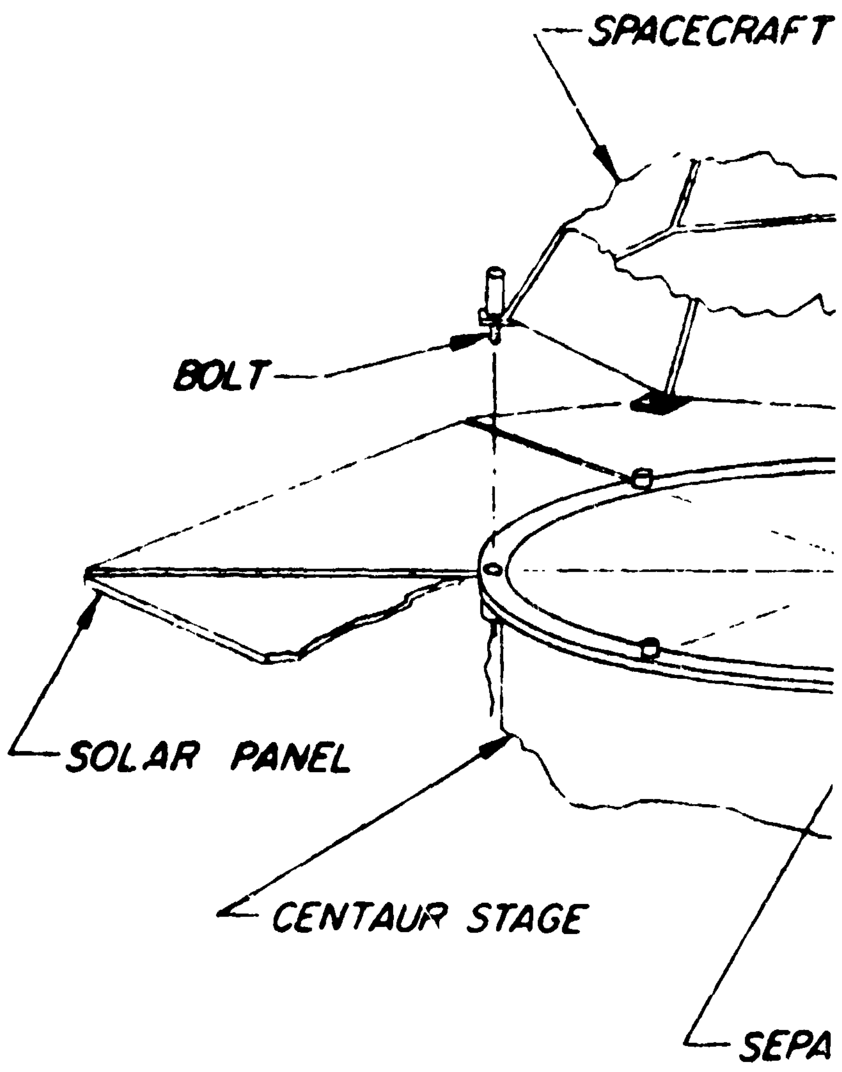
2.0 Separation Nut System was selected from 1.0. Baseline system is illustrated at left.

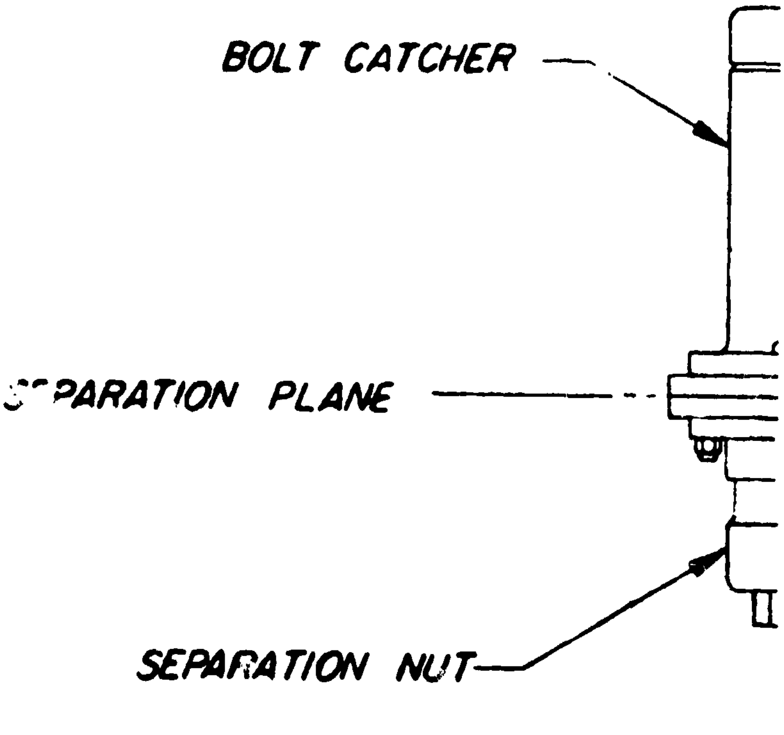
3.0 WEIGHT - RELIABILITY - CONFIGURATION TRADEOFF CHART

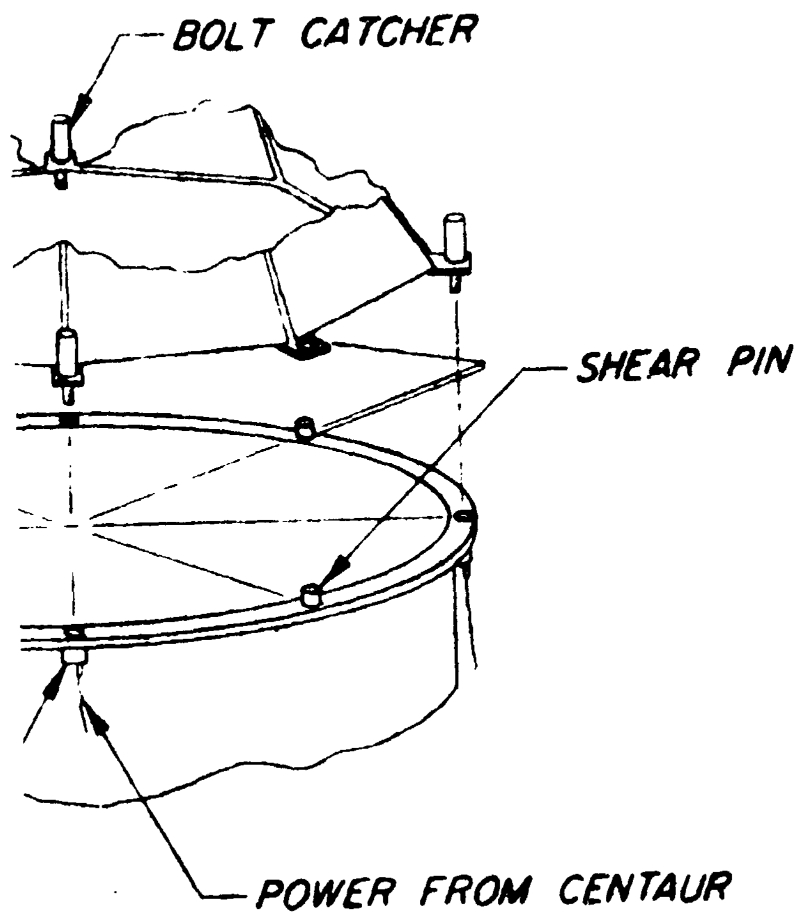
Subsystem		Subsystem Description	Reliability	Weight
S/C - LV	B	3-nuts, 3-dual bridge cartridges	0.998521	2.217
Separation	R	3-nuts, 6-dual bridge cartridges	$0.9^{(3)}816$	2.307
A1 & A2	A	"R" + 3-pin pullers & 6-cartridges	$0.9^{(4)}280$	5.715

Separation S/C from LV
Configuration A-1 & A2
DAC SK-BM00-5000

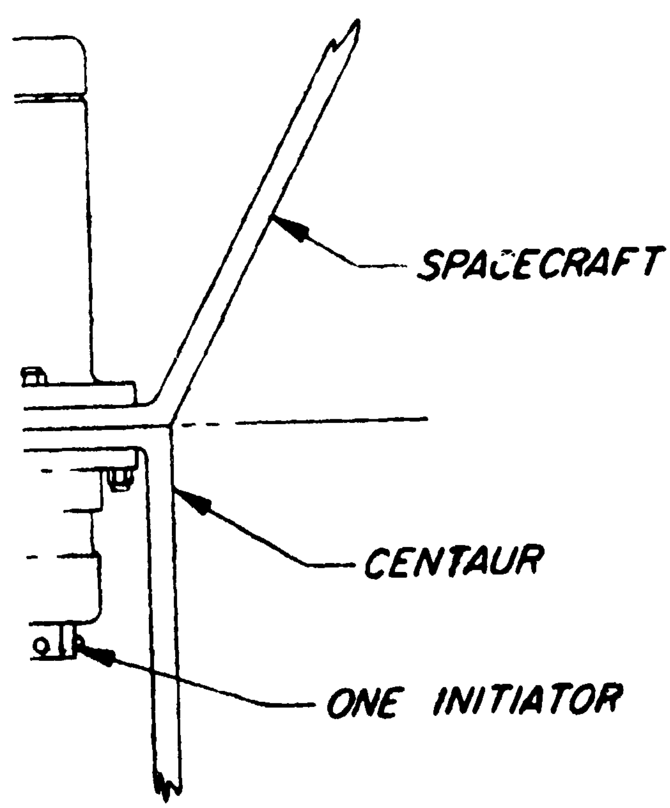
(3)


SPACECRAFT
BOLT
SOLAR PANEL
CENTAUR STAGE
SEPA


BOLT CATCHER
PARATION PLANE
SEPARATION NUT


BOLT CATCHER
SHEAR PIN
POWER FROM CENTAUR

RATION NUT


SPACECRAFT
CENTAUR
ONE INITIATOR

I. SEPARATION SYSTEMS

1.0 ANALYSIS SUMMARY OF SEPARATION METHOD SELECTION

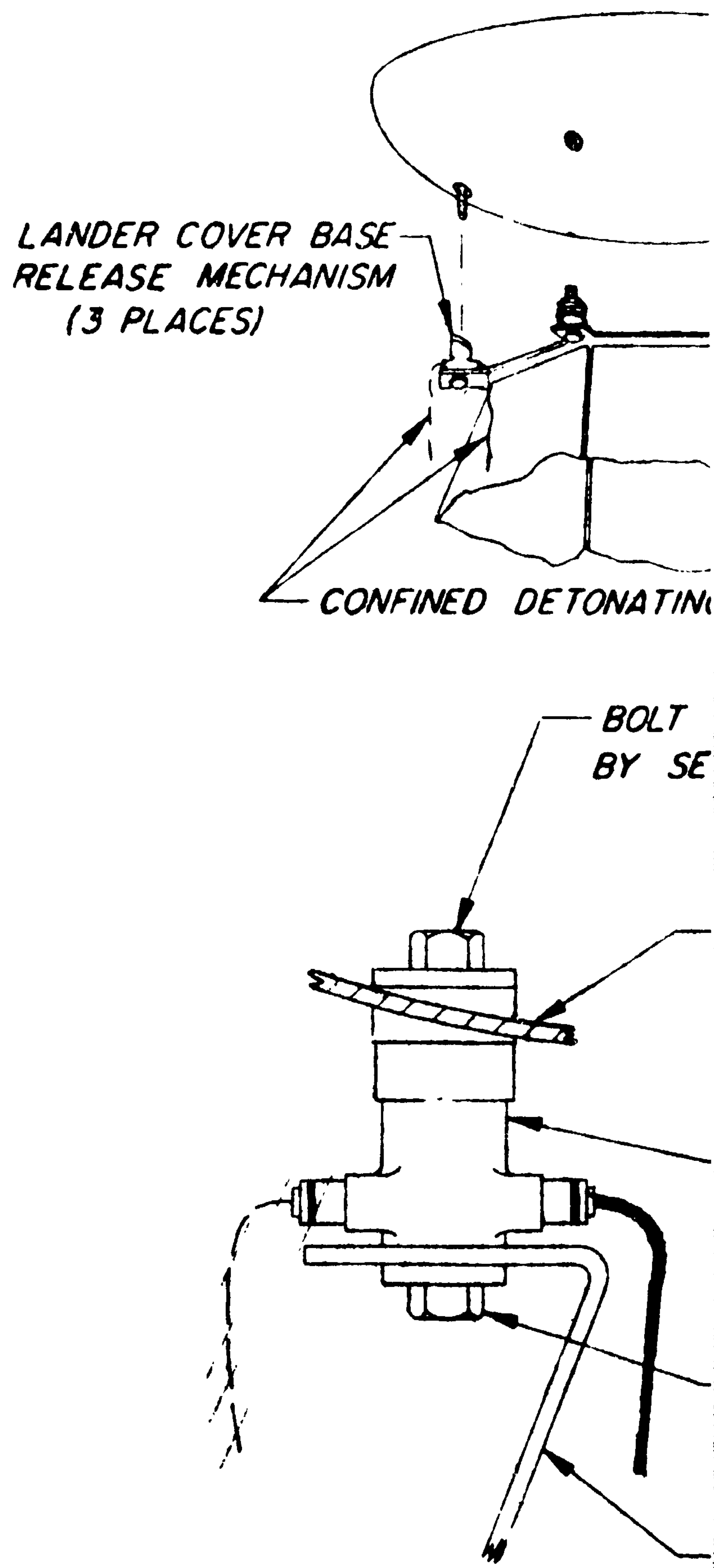
	Wt.	Simplicity	Availability	Shock	Gassing	Rel.	Safety	Total
Ex. Bolt	10	9	10	2	7	8	7	53
Sep. Nut	9	8	9	9	9	9	9	62
Collect	6	6	5	10	10	9	10	56
Pin Puller	8	8	10	9	9	9	9	62
FLSC	10	10	10	1	6	10	2	49

2.0 Separation Nut System was selected from 1.0. Baseline system is illustrated at left.

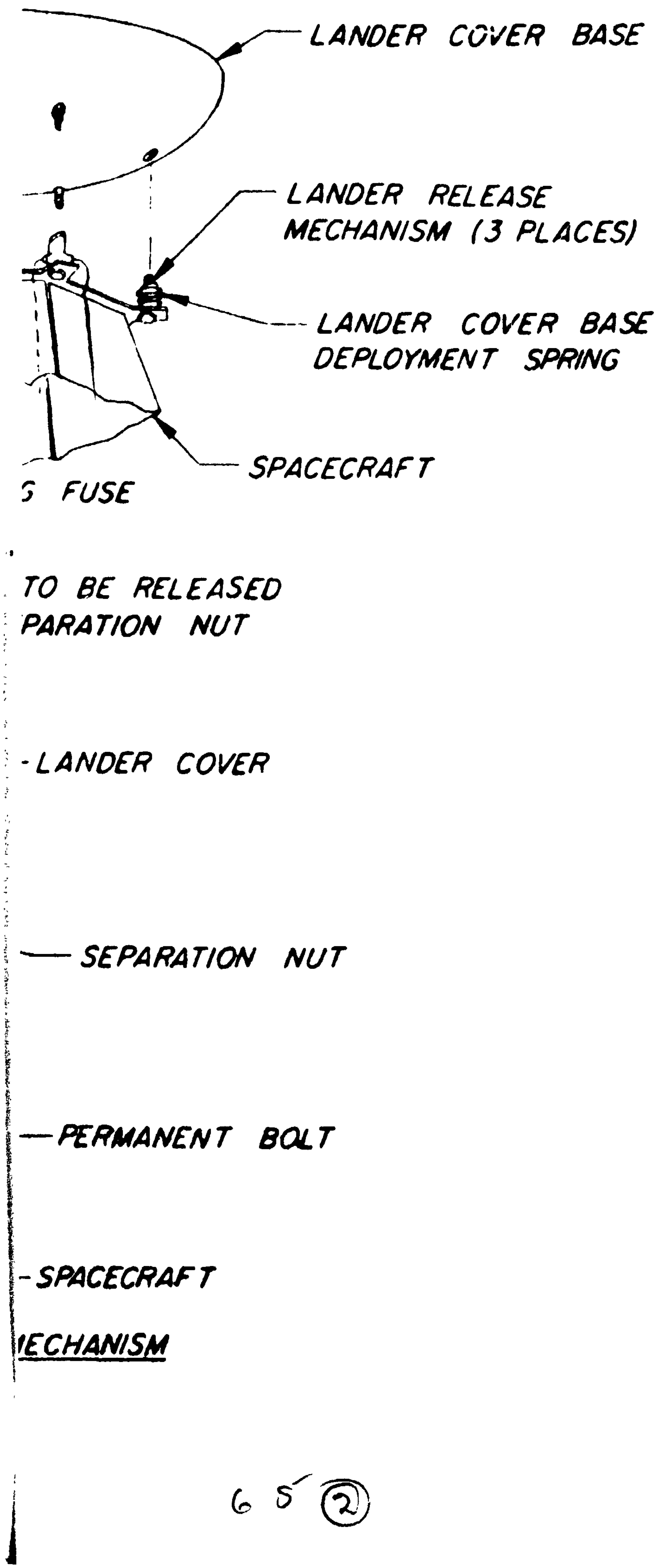
3.0 WEIGHT - RELIABILITY - CONFIGURATION TRADEOFF CHART

Subsystem		Subsystem Description	Reliability	Weight
S/C - LV	B	4-nuts, 4-dual bridge cartridges	0.998019	2.956
Separation	R	4-nuts, 8-dual bridge cartridges	0.9(3)760	3.076
B1 & B2	A	"R" + 4-pin pullers & 8 cartridges	0.9(4)040	6.712

Separation S/C from LV
Configuration B1 & B2
DAC SK-BM00-5001



LANDER COVER RELEASE M


LANDER COVER BASE
LANDER RELEASE MECHANISM (3 PLACES)
LANDER COVER BASE DEPLOYMENT SPRING
SPACECRAFT
FUSE
TO BE RELEASED
PARATION NUT
LANDER COVER
SEPARATION NUT
PERMANENT BOLT
SPACECRAFT
ECHANISM

I. SEPARATION SYSTEMS

1.0 ANALYSIS SUMMARY OF SEPARATION METHOD SELECTION

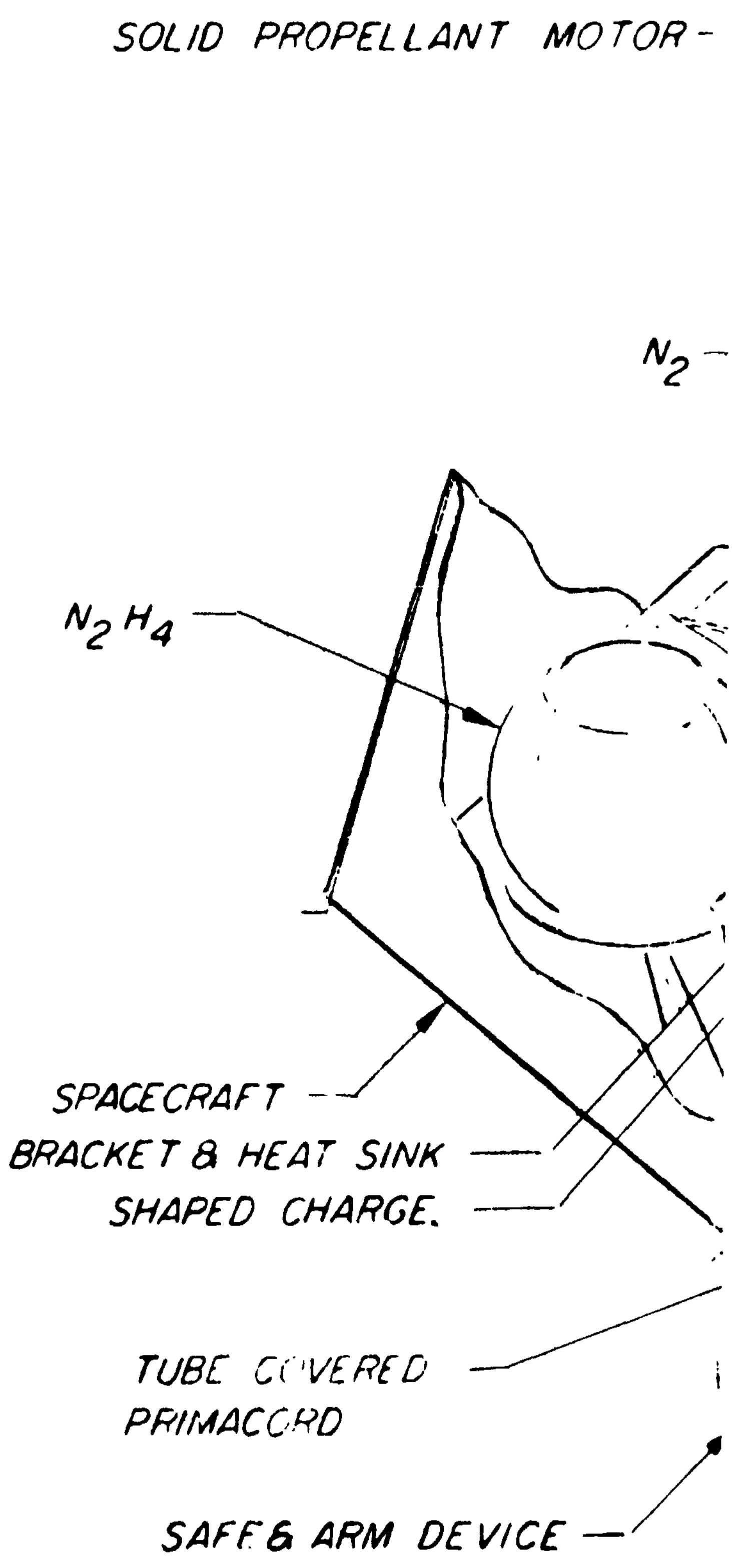
	Wt.	Simplicity	Availability	Shock	Gassing	Rel.	Safety	Total
Ex. Bolt	10	9	10	2	7	8	7	53
Sep. Nut	9	8	9	9	9	9	9	62
Collect	6	6	5	10	10	9	10	56
Pin Puller	8	8	10	9	9	9	9	62
FLSC	10	10	10	1	6	10	2	49

2.0 Separation Nut System was selected from 1.0. Baseline system is illustrated at left.

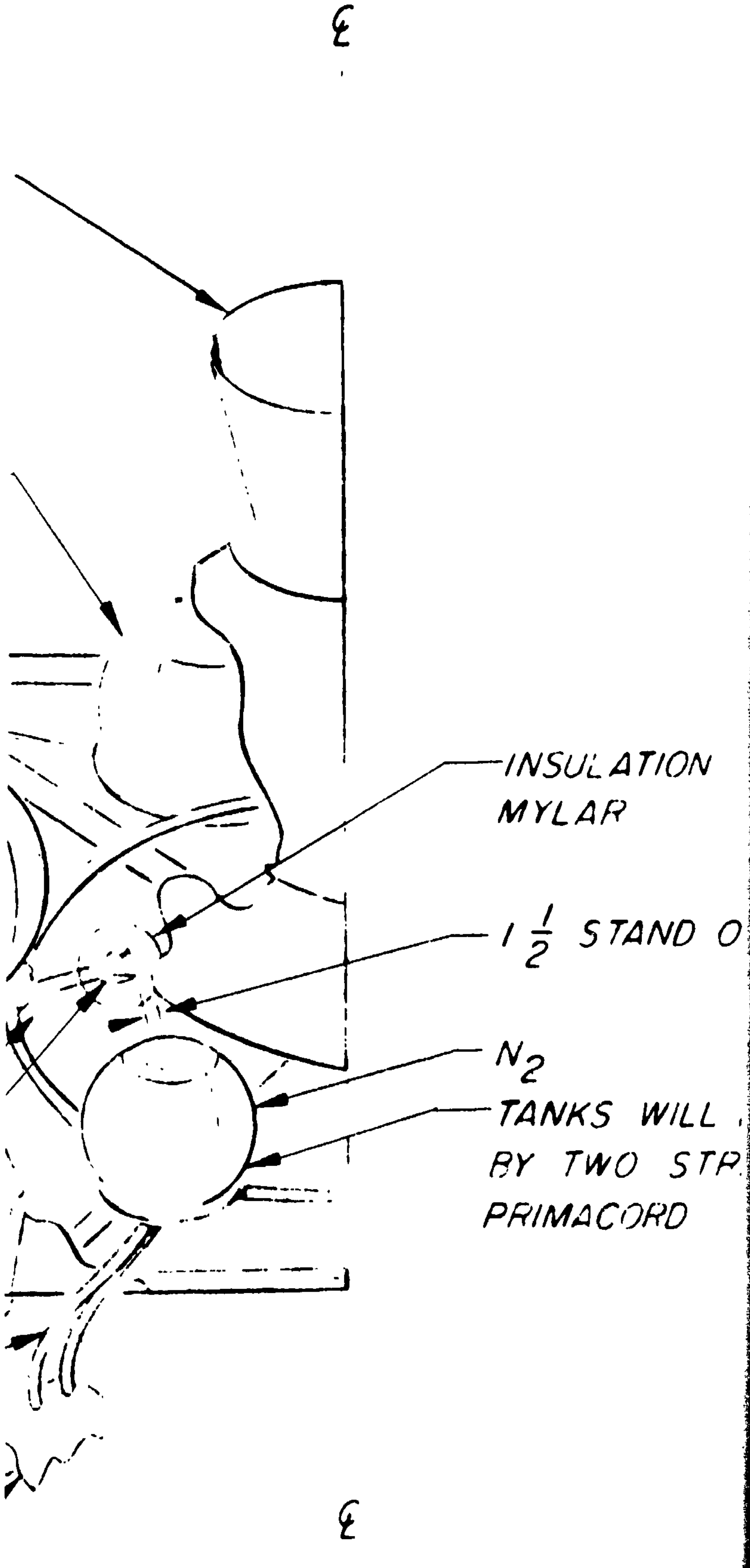
3.0 WEIGHT - RELIABILITY - CONFIGURATION TRADEOFF CHART

Subsystem		Subsystem Description	Reliability	Weight
S/C - Lander Base	B	3-nuts, 3-dual bridge ctg. 3-leads	0.998476	2.967
Separation	R	3-nuts, 6-dual bridge ctg. 6-leads	0.9(3)816	3.807
A1 & A2	A	"R" + "R"	0.9(4)280	7.614

Separation S/C from LV
Configuration A1 & A2
DAC SK-BM00-5002



SOLID PROPELLANT MOTOR
N2
N2 H4
SPACECRAFT
BRACKET & HEAT SINK
SHAPED CHARGE.
TUBE COVERED
PRIMACORD
SAFE & ARM DEVICE



℄
INSULATION
MYLAR
1 1/2 STAND O
N_2
TANKS WILL
BY TWO STR
PRIMACORD
℄

67 (2)

I. DESTRUCT SYSTEM

1.0 WEIGHT RELIABILITY - CONFIGURATION TRADEOFF CHART

Subsystem		Subsystem Description	Reliability	Weight
Destruct	B	Destruct in Centaur		0 *1
	R	2-S&A, 2-primacord, 4-shaped chg.	0.9(7)74	5
A1 & A2	A	2-S&A, 4-primacord, 8-shaped chg.	0.9(7)738	6
Destruct	B	Destruct in Centaur		0 *1
	R	2-S&A, 2-primacord, 0-shaped chg.	0.9(7)729	3
B1 & B2	A	2-S&A, 4-primacord, 4-shaped chg.	0.9(7)738	5

2.0 Reference Configuration A1 & B1 is shown at left.

3.0 In Liquid Motor Configuration no shaped charges are required.

*1 NOTE: A heavier total weight must be carried in the Centaur to do equal damage to the spacecraft.

ALUMINUM

FF

BE DESTROYED

ANDS OF

Destruct Charge and
Safe and Arm Devices
Configuration A1 & A2
DAC SK-BM00-5003

(3)

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Table 1. Preliminary Reliability Predictions of Voyager Separation and Destruct Subsystems

Subsystem	Subsystem Description	Reliability Prediction
S/C-LV Separation A1 and A2 (SK-BM00-5000)	B. 3 nuts, 3 dual bridgewire squibs R. 3 nuts, 6 dual bridgewire squibs A. R (above) + 3 pin pullers and 6 dual bridgewire squibs	0.998521 $0.9^{(3)}816$ $0.9^{(4)}280$
S/C-LV Separation B1 and B2 (SK-BM00-5001)	B. 4 nuts, 4 dual bridgewire squibs R. 4 nuts, 8 dual bridgewire squibs A. R (above) + 4 pin pullers and 8 dual bridgewire squibs	0.998018 $0.9^{(3)}760$ $0.9^{(4)}040$
Separation S/C from LV Configuration A1 & A2 (Lander Base Cover) (SK-BM00-5002)	B. 3 nuts, 3 dual bridgewire squibs, 3 leads R. 3 nuts, 6 dual bridgewire squibs, 6 leads A. R (above) + R (above)	0.998476 $0.9^{(3)}816$ $0.9^{(4)}280$
Destruct Charge and Safe and Arm Devices Configuration A1 & A2 (SK-BM00-5003)	B. Destruct in Centaur R. 2 S & A, 2 primacord, 4 shaped-charges A. 2 S & A, 4 primacord, 8 shaped-charges	- - $0.9^{(7)}711$ $0.9^{(7)}738$
Destruct Charge and Safe and Arm Devices Configuration B1 & A2 (SK-BM00-5003)	B. Destruct in Centaur R. 2 S & A, 2 primacord, 0 shaped-charges A. 2 S & A, 4 primacord, 4 shaped-charges	- - $0.9^{(7)}729$ $0.9^{(7)}738$

4.1.2 Configuration B1 and B2 (SK-BM00-5001)

This configuration is to be used with the bipropellant system and has four attached points. Table 3 presents the reliability predictions for the configurations under consideration.

Table 2. Preliminary Reliability Preduction of S/C-LV Separation-Configuration A1 and A2

System Description	Reliability Prediction
B. 3 nuts, 3 dual bridgewire squibs	0.998521
R. 3 nuts, 6 dual bridgewire squibs	$0.9^{(3)}816$
A. "R" (above) + 3 pin pullers and 6 dual bridgewire squibs.	$0.9^{(4)}280$

Table 3. Preliminary Reliability Prediction of S/C-LV Separation-Configuration B1 and B2

System Description	Reliability Prediction
B. 4 nuts, 4 dual bridgewire squibs	0.998018
R. 4 nuts, 8 dual bridgewire squibs	$0.9^{(3)}760$
A. R (above) + 4 pin pullers and 8 dual bridgewire squibs	$0.9^{(4)}040$

The reliability of the completely redundant system ("A" in Tables 2 and 3) is limited by the probability of premature firing of the squibs. Although redundant squibs result in a higher probability for the firing mode, they also result in a lower probability for the premature firing mode since this becomes a series relationship (i.e. the more squibs used the more likely it is that a failure will occur). In Table 3 the reliability prediction (to the number of places shown) is the probability of no premature firing of the 16 squibs used. Although the release of one of the separation points would probably not result in spacecraft separation, it undoubtedly would have some detrimental effect on the spacecraft and/or launch vehicle.

4.1.3 Configuration A1 and A2 (Lander Base Cover) (SK-BM00-5002)

The separation of the lander base cover is identical to the spacecraft separation (3 points in SK-BM00-5000) except for the addition of leads (fuses) from the squibs located in the bottom of the spacecraft to the cover. Table 4 gives the reliability of the configuration considered. Only the B configuration varies from the figures presented in Section 3, because the addition of the leads; however, when redundant leads are used they have no effect on the calculations to the number of places shown in the table.

Table 4. Preliminary Reliability Prediction of S/C-LV Separation (Lander Base Cover)-Configuration A1 and A2

System Description	Reliability Prediction
B. 3 nuts, 3 dual bridgewire squibs and 3 leads.	0.998476
R. 3 nuts, 6 dual bridgewire squibs and 6 leads.	$0.9^{(3)}816$
A. R (above) + R (above)	$0.9^{(4)}280$

4.1.4 Destruct Charge and Safe and Arm (S&A) Devices Configuration A1 and A2 and B1 and B2 (SK-BM00-5003)

The reliability of the spacecraft destruct system is shown in Table 5. For the purposes of this analysis it was assumed that the system would be in the safe position, otherwise the reliability of the system would be limited by the probability of premature detonation of the squibs which is $0.9^{(4)}88$.

The A1 and A2 configuration is used with the monopropellant configuration where shaped-changes are required to destruct the solid motor case. The B1 and B2 configurations are used with the bipropellant configuration and the shaped-changes are not required except as a backup (redundant) to the primacord.

Table 5. Preliminary Reliability Prediction of Destruct Charge and Safe and Arm Devices, Configuration A1, A2, B1 and B2 (SK-BM00-5003)

System Description	Reliability Prediction
A1 and A2	
B. Destruct in Centaur	- -
R. 2, S & A, 2 primacord and 4 shaped-charges	$0.9^{(7)}711$
A. 2, S & A, 4 primacord and 8 shaped-charges	$0.9^{(7)}738$
B1 and B2	
B. Destruct in Centaur	- -
R. 2 S & A, 2 primacord	$0.9^{(7)}729$
A. 2 S & A, 4 primacord and 4 shaped-charges	$0.9^{(7)}738$

4.1.5 Reliability Analysis and Mathematical Models

Tables 6 through 11 present the reliability analyses of the major parts used in the separation and destruct systems. The tables present significant information about the equipment, the major failure modes considered, possible causes, any backup provisions and the failure rates, and their sources, used in the analysis.

The reliability of each one shot device is simply the complement of its failure rate (λ), where the failure rate is essentially the probability of failure (Q)

$$R = 1 - \lambda = 1 - Q$$

If two devices are used redundantly, both must fail and the reliability is determined from

$$R = 1 - Q^2$$

The total system reliability is determined from the product rule

$$R_T = \prod_{i=1}^{N} R_i$$

Table 6 A

Item: Squib

Subsystem: Separation

Description:

Explosive bridge wire with powder charge. (Redundant bridgewires).

Operational Notes:

One shot device. Apollo standard initiator.

Environmental Notes:

Failure is most likely to result from vibration during the boost phase.

Primary Failure Mode:

Failure to fire, with proper force, when proper electrical input is present.

Possible Causes:

1. Poor workmanship (bridgewire coating)
2. Bad lot of charges
3. Broken bridgewires.

Effects on Subsystem/Mission:

Catastrophic, failure to separate.

Backup Provisions:

Redundant squibs and bridgewires.

Inherent Preventives:

Lot qualification of squibs.

Secondary Failure Mode:

Premature firing.

Possible Causes:

1. Stray R.F. signals
2. Boost environment
3. Bad lot of charges.

Effects on Subsystem/Mission:

Catastrophic, early separation.

Backup Provisions:

None - However probably more than one squib would have to fire before separation would occur.

Inherent Preventives:

Bruceton test - shielding from R.F. signals

Table 6B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles	1 cycle
	Basic Failure Rate, Units: per 10^{-6} Firings	294
	Environment/Application Factor	-
	Actual Failure Rate, Units: per 10^{-6} Firings	294
	Failure Rate Source Code	a
	Probability of Failure, x 10^{-6}	294*
Secondary Failure Mode	Failure Classification	A
	Time/Cycles Units: x 10^{-6} Cycles	1
	Basic Failure Rate, Units: x 10^{-6} Firings	6*
	Environment/Application Factor	-
	Actual Failure Rate, Units: x 10^{-6} Cycles	6
	Failure Rate Source Code	a
	Probability of Failure x 10^{-6}	6
	Probability of Failure, All Modex, x 10^{-6}	300
	Reliability	$0.9^{(3)}700$

Failure Rate Data Sources: TRW experience - 2000 Squib firings with no failures. Prob. of failure of 3 x 10^{-4} at 50% confidence. Farada indicates 47,332 firings with 10 failures. Failure rate 211 x 10^{-4} which is in good agreement with TRW data.

Notes:

A Catastrophic failure; mission abort.

* Experience from Bruceton type testing indicates premature failure mode is approx. 1/50 of total failure probability.

Table 7A

Item: Separation Nut

System/Subsystem: Separation

Item Type/Description:

Captive with a removable flange base and dual cartridge capability.

Operational Notes:

Gas retaining, no fragmentation. All parts are retained and locked in place.

Primary Failure Mode:

Failure to release the bolt when proper force is present.

Possible Causes:

1. Failure of the separator piston to fracture (workmanship).
2. Failure of the locking piston to move forward (jammed).
3. Failure of the segments to displace from the bolt (workmanship)

Effects on Subsystem/Mission:

Catastrophic -- separation would not occur.

Inherent Preventives:

Lot qualification of separation nuts.

Secondary Failure Mode:

Failure to eject the bolt.

Possible Causes:

Failure of the ejector piston.

Effects on Subsystem/Mission:

None if bolt is free could be pulled away by retro force.

Inherent Preventives:

Lot qualification of separation nuts.

Table 7B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles, Units: x 10^{-6} Firings	1 cycle
	Basic Failure Rate, Units: x 10^{-6} Firings	48*
	Environment/Application Factor	-
	Actual Failure Rate, Units: x 10^{-6} Firings	48
	Failure Rate Source Code	b
	Probability of Failure, x 10^{-6}	48
Secondary Failure Mode	Failure Classification	M
	Time/Cycles, Units: x 10^{-6} Firings	1 cycle
	Basic Failure Rate, Units: x 10^{-6} Firings	<1**
	Environment/Application Factor	-
	Actual Failure Rate, Units: x 10^{-6} Firings	<1
	Failure Rate Source Code	b
	Probability of Failure, x 10^{-6}	<1
	Probability of Failure, All Modex, x 10^{-6}	48
	Reliability	$0.9^{(4)}52$

Failure Rate Data Sources: b) Farada, Revision dated 3/1/65 Pg. 2.279, Source 138. Martin Co. Report M-63-3 dated Oct. 63 (K factors not applicable). Explosive bolt used.

Notes:

A ibid
M Minor failure - noncatastrophic.
* The 3 causes of failure are broken down as follows:
1. Failure rate of 40 used (explosive bolt fracture).
2. Failure rate for this cause is considered to be an order of magnitude less than in 1.
3. Same as 2.

** Failure rate was assumed to be negligible <1 x 10^{-6}.

Note: Pin puller assumed to be the same as a nut.

Table 8A

Item: Cable Assy (Squib Firing Circuit)

Operational Notes:

A 2 pin connector, 2 solder joints and 2 wires were considered for each cable assy.

Environmental Notes:

Failure could occur as a result of vibration during boost.

Primary Failure Mode:

Failure to provide electrical signal to squib.

Possible Causes:

1. Poor pin contact.
2. Broken wire.

Effects on Subsystem/Mission:

None (would be catastrophic if all failed)

Backup Provisions:

Redundant squibs used; therefore cable assy are considered redundant.

Inherent Preventives:

Circuit continuity check.

Table 8B

	Mission
Failure Classification	A
Time/Cycles Units: Hours	0.3
Basic Failure Rate, Units: x 10^{-6} Hours	0.438*
Environment/Application Factor	1000
Actual Failure Rate, Units: x 10^{-6} Hrs	438
Failure Rate Source Code	c
Probability of Failure, x 10^{-6}	145
Reliability	$.9^{(3)}855$

Failure Rate Data Sources: c) Farada Revision dated 3/1/65.

Source 138: Martin Co. Report M63-3.*

Notes:

A ibid

* Total Failure Rate determined as follows:

1.	Connector	- .2/pin x 2 pins	=	.400
2.	Wire	- .015 x 2	=	.030
3.	Solder Joint	- .004 x 2	=	.008
				.438

Table 9A

Item: Shaped Charge

System/Subsystem: Separation

Primary Failure Mode:

Failure to cut when proper force is applied.

Possible Causes:

1. Workmanship (voids or cracks in the material if at a member to be cut).
2. Boost environment (structural failure resulting in cracks).

Effects on Subsystem/Mission:

Separation would not occur.

Backup Provisions:

None

Inherent Preventives:

None

Secondary Failure Mode:

Vehicle damage from flying fragments.

Possible Causes:

Inability to contain all fragments.

Effects on Subsystem/Mission:

Mission degradation to mission abort.

Table 9B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles	1
	Basic Failure Rate, Units: x 10^{-6} Cycles	27
	Failure Rate Source Code	b
	Probability of Failure, x 10^{-6}	27
Secondary Failure Mode	Failure Classification	B
	Time/Cycles Units: Cycles	1
	Basic Failure Rate, Units: x 10^{-6}	3*
	Actual Failure Rate, Units:	3
	Failure Rate Source Code	b
	Probability of Failure x 10^{-6}	3

Failure Rate Data Sources: b) ibid

Notes:

A Catastrophic failure, mission abort

B Noncatastrophic failure, mission degradation

* An estimate of 10% of total failure rate from Farada (30) was used for this failure mode.

Table 10A

Item: Safe and Arm Device

System/Subsystem: Destruct

Item Type/Description

Two position rotary device.

Operational Notes:

Device must be capable of arming, firing or disarming on command.

Environmental Notes:

Must function during and/or after boost environment.

Primary Failure Mode:

Failure to arm/disarm.

Possible Causes:

1. Electrical failure (open/short).

Effects on Subsystem/Mission:

1. Unable to destruct.

Backup Provisions

Redundant safe and arm devices.

Inherent Preventives:

Electrical checks.

Table 10B

	Mission
Failure Classification	N
Time/Cycles Units: Hours	0.3
Basic Failure Rate, Units: Hrs.	0.54
Environment/Application Factor	1000
Actual Failure Rate, Units: x 10^{-6}	540
Failure Rate Source Code	d
Probability of Failure, All Modes	162
Reliability	$0.9^{(3)}838$

Failure Rate Data Sources: d) Farada, Pg. 2.183 Revised 3/1/65, Source 138 (Martin Co.) (Rotary switch was used assuming two sets of contacts.)

Notes:

N No effect on mission success.

Table 11

Separation system reliability determined as follows for the monopropellant system:

$$R_s = (R_{squib}, R_{nut}, R_{cable\ assy})^3$$

Quantity raised to the third power because 3 attach points are used.

The individual reliabilities were determined as follows:

$R_{squib} = (P^2 + 2PQ) \cdot (R_{premature})^2$ where P is probability of success and Q us probability of failure (1 - P) since the squibs are redundant, substituting P = 1 - Q

$R_{squib} = (1 - Q^2) . (R_{premature})^2 = 1 - (.000294)^2 . (999994)^2$

$R_{premature}$ is squared since either squib could fire early.

$R_{squib} = (1 - .0^{(7)}864) . 9^{(4)}88$

$= 9^{(7)}136 . 9^{(4)}88$

$R_{squib} = 9^{(4)}879136$

$R_{nut} = 0.9^{(4)}52$

$R_{cable\ assy} = e^{-.483 . 1000 . .3 . 10^{-6}} = e^{-145 \times 10^{-6}}$

$= 0.9^{(3)}855$ each

Since the cable assemblies are redundant:

$R_{cable\ assy} = 1 - Q^2 = 1 - (145 \times 10^{-6})^2 = 1 - .0^{(7)}21$

$= 0.9^{(7)}79$

$R_s = (0.9^{(4)}879136 . 9^{(4)}52 . 9^{(7)}79)^3$

$R_s = 0.9^{(3)}816$

The reliability of the separation system for the bipropellant system has four separation points and is determined as follows:

$R_{s'} = (R_{squib} \cdot R_{nut} \cdot R_{cable\ assy})^4$

$= (0.9^{(4)}879136 . 9^{(4)}52 . 9^{(7)}79)^4$

$R_{s'} = 0.9^{(3)}760$

The reliability of the safe and arm devices was determined using

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

where λ is the total failure rate and includes the environmental factor (1000) and t is the boost time (0.3 hour).

If λt is small (<0.01) the first equation can be used.

4.2 Baseline Propellant Feed Systems

A preliminary reliability analysis was made of the baseline mono- and bipropellant feed systems. In addition to computing nominal reliability values, a rough error analysis was performed to assess the magnitude of the uncertainty associated with this preliminary analysis. The results are:

		Monopropellant System	Bipropellant System
	Best	.9955	.9925
Mission Reliability:	Nominal	.9923	.983
	Worst	.896	.821

An environmental factor of 1,000 is used during powered portions of the mission; a factor of 1 is used for unpowered flight. A mission duration of seven months is used since the baseline engine configuration is no longer used after entering Mars orbit. Exceptions are such items as pressure vessels which must not rupture during an additional six months in Mars orbit.

Criticality factors are used in place of failure classifications on the following tables. These are estimates of the probability of mission failure given that a failure mode occurs. This accounts for small leaks and other failures which may not be catastrophic to the mission.

The uncertainty of the reliability predictions is primarily due to uncertainty of component failure rates. Many failure rates are known only to the nearest order of magnitude. Most failure rates were taken from the FARADA handbook, 1 April 1965 issue. Some were taken from DAC studies on the Saturn S-IVB stage.

Table 1A

Item: Helium Sphere

System/Subsystem: Propellant Feed

Operational Notes:

Titanium - 2 req'd

Primary Failure Mode:

Rupture

Possible Causes:

Tank damaged after acceptance by improper handling

Effects on Subsystem/Mission:

Catastrophic at any time during mission

Inherent Preventives:

Proof pressure test for acceptance

Table 1B

	Nom.	Worst	Best	Boost	Transit	Mission
Failure Classification				1	1	
Time/Cycles Units: hrs				0.43	10,800	
Basic Failure Rate, Units: x 10^{-9} hrs	80	1590	20	80	80	
Environment/ Application Factor				1000	1	
Actual Failure Rate, Units: x 16^{-9} hrs				80,000	80	
Probability of Failure, x 10^{-6}				34	865	899
Reliability				$0.9^4 66$	$0.9^3 135$	$0.9^3 101$

Table 2A

Hand: Hand Valve
System/Subsystem: Propellant Feed, Mono and Bi-prop.
Operational Notes:
Capped during flight. Used on ground for filling He and propellant tanks.
3 req'd for mono, 5 req'd for bi-prop.
Primary Failure Mode:
Leakage in flight
Possible Causes:
Vibration: Improperly seated or capped.
Effects on Subsystem/Mission:
Could be negligible or catastrophic depending on magnitude of leak rate.
Backup Provisions:
Outlet sealed and capped.

Table 2B

	Nom.	Worst	Best	Boost	Transit	Mission
Failure Classification				1	0.5	
Time/Cycles Units: hrs				0.43	5040	
Basic Failure Rate, Units: x 10^{-9} hrs	60	224	20	60	60	
Environment/ Application Factor				$\frac{1000}{0.1}$	$\frac{1}{0.1}$ *	
Actual Failure Rate, Units: x 16^{-9} hrs				6000	6	
Probability of Failure, x 10^{-6}				2.58	15.2	17.8
Reliability				$0.9^5 74$	$0.9^4 848$	$0.9^4 822$

*Application factor of 0.1 is used to account for value being capped.

Table 3A

Item: Filter

System/Subsystem: Propellant Feed

Item Type/Description:

Gas or Propellant

Primary Failure Mode:

Release particles or fragments

Possible Causes:

Vibration

Effects on Subsystem/Mission:

Negligible to catastrophic depending on effect of particles downstream, which also depends on size of particles

Secondary Failure Mode:

Incomplete filtration

Possible Causes:

Loose element

Effects on Subsystem/Mission:

Normally not serious

Other Failure Modes:

Burnt case; clogging

Effects on Subsystem/Mission:

Catastrophic, although highly unlikely

Table 3B

		Nom.	Worst	Best	Boost	Transit	Mission
Primary Mode	Failure Classification				0.5	0.5	
	Time/Cycles Units: hrs				0.2	0.23	
	Basic Failure Rate, Units: x 10^{-9} hrs	120	1080	18	120	120	
	Environment/ Application Factor				1000	1000	
	Actual Failure Rate, Units: x 10^{-9} hrs				120,000	120,000	
	Probability of Failure, X 10^{-6}				12.0	13.8	
Secondary Mode	Failure Classification				0.1	0.1	
	Time/Cycles Units: hrs				0.2	0.23	
	Basic Failure Rate, Units: x 10^{-9} hrs	80	720	12	80	80	
	Environment/ Application Factor				1000	1000	
	Actual Failure Rate, Units: x 10^{-9}				80,000	80,000	
	Probability of Failure x 10^{-6}				1.60	1.84	
	Probability of Failure, All Modes				73.6	15.6	29.2
	Reliability				0.9^4864	0.9^4844	0.9^4708

Table 4A

Item: Regulator

System/Subsystem: Propellant Feed

Operational Notes:

Regulates 3000 psi to 250 psi. Must positively lock up when downstream pressure exceeds 250 psi.

Primary Failure Mode:

Leakage during lockup

Possible Causes:

Contamination on seat.

Effects on Subsystem/Mission:

Negligible to catastrophic depending on leakage rate

Inherent Preventives:

Upstream filter

Secondary Failure Mode:

Over pressurization or fail wide open

Possible Causes:

Excessive regulator drift. Binding or jamming. Contamination

Effects on Subsystem/Mission:

Major to catastrophic depending on degree of drift

Other Failure Modes:

Underpressurization or fail closed

Possible Causes:

As above

Effects on Subsystem/Mission:

As above

Table 4B

		Nom.	Worst	Best	Boost	Transit	Mission
Primary	Failure Classification				0.5	0.5	
	Time/Cycles Units: Hours				0.43	5,040	
	Basic Failure Rate, Units: $\times 10^{-9}$ hrs	500	7,870	200	500	500	
	Environment/Application Factor				1000	1	
	Actual Failure Rate, Units: $\times 10^{-9}$ hrs				500,000	500	
	Probability of Failure, $\times 10^{-6}$				108	1260	
Secondary	Failure Classification					0.8	
	Time/Cycles Units: Hours					0.22	
	Basic Failure Rate, Units: $\times 10^{-9}$ hrs	300	4,620	47.3		300	
	Environment/Application Factor					1	
	Actual Failure Rate, Units: $\times 10^{-9}$ hrs					300	
	Probability of Failure $\times 10^{-6}$					0.053	
Other Modes	Failure Classification					0.8	
	Time/Cycles Units: Hours					0.22	
	Basic Failure Rate, Units: $\times 10^{-9}$	200	3,150	31.5		200	
	Environment/Application Factor					1	
	Actual Failure Rate, Units: $\times 10^{-9}$					200	
	Probability of Failure $\times 10^{-6}$					0.035	
	Probability of Failure, All Modes $\times 10^{-6}$				108	1260	1368
	Reliability				$0.9^{3}892$	$0.9^{2}874$	$0.9^{2}863$

Table 5A

Item: Vent and Relief Valve

System/Subsystem: Propellant Feed

Operational Notes:

Used as a solenoid vent valve on the ground; as a spring loaded relief valve during flight. Preflight reliability is not considered in analysis.

Primary Failure Mode:

Stuck; failure to relieve when required

Possible Causes:

Improper handling or installation

Effects on Subsystem/Mission:

Catastrophic only if relief function is needed during flight.

Secondary Failure Mode:

Leakage

Possible Causes:

Contamination on seat

Effects on Subsystem/Mission:

Negligible to catastrophic depending on magnitude of leak rate and at what time during the mission it starts leaking.

Table 5B

		Nom.	Worst	Best	Boost	Transit	Mission
Primary	Failure Classification				0.1	0.1	
	Time/Cycles Units: hrs				0.43	10,800	
	Basic Failure Rate, Units: $\times 10^{-9}$ hrs	375	4,270	150	375	375	
	Environment/ Application Factor				1000	1	
	Actual Failure Rate, Units: $\times 10^{-9}$				375,000	375	
	Probability of Failure, $\times 10^{-6}$				16.1	405	
Secondary	Failure Classification				0.8	0.5	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: $\times 10^{-9}$ hrs	125	1,425	50	125	125	
	Environment/ Application Factor				1000	1	
	Actual Failure Rate, Units: $\times 10^{-9}$				125,000	125	
	Probability of Failure $\times 10^{-6}$				43	315	
	Probability of Failure, All Modes $\times 10^{-6}$				59	720	779
	Reliability				$0.9^{4}41$	$0.9^{3}280$	$0.9^{3}221$

Table 6A

Item: Propellant Tank

System/Subsystem: Propellant Feed

Operational Notes:

Contains fuel, oxidizer or monopropellant

4 req'd for Bi propellant system

2 req'd for monopropellant system

Primary Failure Mode:

Rupture

Possible Causes:

Tank damaged after acceptance by being dropped, struck, etc.

Effects on Subsystem/Mission:

Catastrophic at any time during mission

Backup Provisions:

Relief valve

Inherent Preventives:

Proof pressure test for acceptance

Table 6B

	Nom.	Worst	Best	Boost	Transit	Mission
Failure Classification				0.1*	0.1*	
Time/Cycles Units: hrs				0.43	10,800	
Basic Failure Rate, Units: x 10^{-9} hrs	120	1,640	20	120	120	
Environment/ Application Factor				1000	1	
Actual Failure Rate, Units: x 10^{-9} hrs				120,000	120	
Failure Rate Source Code	a					
Probability of Failure, x 10^{-6}				5.16	130	135
Reliability				0.9^548	0.9^387	0.9^3865

Failure Rate Data Sources: a) DAC Saturn S-IVB

Notes:

* Considers backup effect of relief valve

$\sigma_{Ri} = 0.0^3818$

$1 + \sigma_{Ri}^2 = 1.0^66699$

Table 7A

Item: Propellant Tank Bellows

System/Subsystem: Propellant Feed (Bi-Prop. Tank)

Item Type/Description:

347 Stainless Steel

Primary Failure Mode:

Binding or failure to expell fluid

Effects on Subsystem/Mission:

Catastrophic

Secondary Failure Mode:

Rupture or leakage

Effects on Subsystem/Mission:

Would not be serious if leakage is small. Some propellant would be unavailable for combustion.

Table 7B

		Nom.	Worst	Best	Boost	Transit	Mission
Primary	Failure Classification				1	1	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: $x\ 10^{-9}$	150	225	50	150	150	
	Environment/ Application Factor				1000	1	
	Actual Failure Rate, Units: $x\ 10^{-6}$				150,000	150	
	Failure Rate Source Code	a					
	Probability of Failure, $x\ 10^{-6}$				64.5	755	
Secondary	Failure Classification				0.5	0.5	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: $x\ 10^{-9}$	150	225	50	150	150	
	Environment/ Application Factor				1000	1	
	Probability of Failure $x\ 10^{-6}$				32.2	378	
	Probability of Failure, All Modes $x\ 10^{-6}$				96.7	1133	1230
	Reliability				$0.9^{4}033$	$0.9^{2}887$	$0.9^{2}870$

Failure Rate Data Sources: a) DAC Saturn S-IVB

Table 8A

Item: Bladder

System/Subsystem: Propellant Feed (Monopropellant Tank)

Item Type/Description:

Butyl Rubber

Primary Failure Mode:

Leakage due to permeation, cracks, pinholes, etc. or complete rupture

Possible Causes:

Material deterioration; damage in handling

Effects on Subsystem/Mission:

Would not be serious if leakage is small. Some propellant would be unavailable for combustion.

Secondary Failure Mode:

Blockage of tank outlet

Effects on Subsystem/Mission:

Catastrophic

Inherent Preventives:

Design to prevent blockage

Table 8B

		Nom.	Worst	Best	Boost	Transit	Mission
Primary Mode	Failure Classification	90 %			0.5	0.5	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: x 10^{-9} hrs	450	630	162	450	450	
	Environment/ Application Factor				1000	1	
	Failure Rate Source Code	a					
	Probability of Failure, x 10^{-6}				96.7	1130	
Secondary Mode	Failure Classification	10 %			1	1	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: x 10^{-9}	50	70	18	50	50	
	Environment/ Application Factor				1000	1	
	Failure Rate Source Code	a					
	Probability of Failure x 10^{-6}				21.5	252	
	Probability of Failure, All Modes x 10^{-6}				118.2	1382	1500
	Reliability				$0.9^{3}882$	$0.9^{2}862$	$0.9^{2}850$

Failure Rate Data Sources: a) DAC Saturn S-IVB

Table 9A

Item: Check Valve

System/Subsystem: Propellant Feed

Item Type/Description:

Spring loaded closed

Primary Failure Mode:

Failure to prevent backflow

Possible Causes:

Stuck open, contamination, binding

Effects on Subsystem/Mission:

Catastrophic in bi-propellant system if two valves leak; may not be serious if only one leaks a small amount

Secondary Failure Mode:

Stuck in closed position

Possible Causes:

Jammed due to contamination

Effects on Subsystem/Mission:

Catastrophic

Table 9B

		Nom.	Worst	Best	Boost	Transit	Mission
Primary Mode	Failure Classification	80 %			0.8	0.8	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: x 10^{-9} hrs	600	3,760	160	600	600	
	Environment/ Application Factor				1000	1	
	Actual Failure Rate, Units: x 10^{-9} hrs				600,000	600	
	Probability of Failure, x 10^{-6}				206	2420	
Secondary Mode	Failure Classification	20 %			1	1	
	Time/Cycles Units: hrs				0.43	5,040	
	Basic Failure Rate, Units: x 10^{-9}	150	940	40	150	150	
	Environment/ Application Factor				1000	1	
	Actual Failure Rate, Units: x 10^{-9}				150,000	150	
	Probability of Failure x 10^{-6}				64.5	757	
	Probability of Failure, All Modes				271	3177	3448
	Reliability				0.9^3729	0.9^2682	0.9^2655

Table 9A

Item: Fittings and Connections
System/Subsystem: Propellant Feed and Engine
Item Type/Description: Mechanical connections, not welded or brazed *
Primary Failure Mode: Leakage
Possible Causes: Improper connection
Effects on Subsystem/Mission: Could be negligible or catastrophic depending on magnitude of leak rate

Remarks:

* Welded and brazed fittings are considered to have negligible prob. of failure.

Table 9B

	Nom.	Worst	Best	Boost	Transit	Mission
Failure Classification				1	0.5	
Time/Cycles Units:				0.43	5,040	
Basic Failure Rate, Units: x 10^{-9} hrs	30	710	20	30	30	
Environment/ Application Factor				1000	1	
Actual Failure Rate, Units: x 10^{-9} hrs				30,000	30	
Failure Rate Source Code	a					
Probability of Failure, x 10^{-6}				12.9	76	89
Reliability per Fitting				$0.9^{4}87$	$0.9^{4}24$	$0.9^{4}11$

Failure Rate Data Sources: a) DAC Saturn S-IVB

Table 10A

Item: Ducts and Lines	
Operational Notes:	1/4" and 1/2" Aluminum
Primary Failure Mode:	Rupture
Possible Causes:	Improper handling
Effects on Subsystem/Mission:	Catastrophic

Table 10B

	Nom.	Worst	Best	Boost	Transit	Mission
Failure Classification				0.1 *	0.1 *	
Time/Cycles Units:				0.43	10,800	
Basic Failure Rate, Units: $\times 10^{-9}$	150	1000	150	150	150	
Environment/ Application Factor				1000	1	
Probability of Failure, $\times 10^{-6}$				6.45	162	168
Reliability				0.9^5755	0.9^3838	0.9^3832

Notes:

* Considers backup effect of relief valve

Table 11. Propellant Feed System

Monopropellant Baseline Configuration

Component	Unit Reliability	Number Req'd	Total Reliability
Helium sphere	$0.9^{3}101$	2	$0.9^{2}820$
Hand valve	$0.9^{4}822$	3	$0.9^{4}466$
Filter	$0.9^{4}708$	2	$0.9^{4}416$
Regulator	$0.9^{2}863$	1	$0.9^{2}863$
Relief valve	$0.9^{3}221$	1	$0.9^{3}221$
Propellant tank	$0.9^{3}865$	2	$0.9^{3}730$
Bladder	$0.9^{2}850$	2	$0.9^{2}700$
Fittings & connections	$0.9^{4}11$	3	$0.9^{3}733$
Lines & Ducts Assy	$0.9^{3}832$	1	$0.9^{3}832$
		Total	$0.9^{2}225$

Table 12. Propellant Feed System

Baseline Bi-propellant Configuration

Component	Unit Reliability	Number Req'd	Total Reliability
Helium sphere	$0.9^3 101$	2	$0.9^2 820$
Hand valve	$0.9^4 822$	5	$0.9^3 811$
Filter	$0.9^4 708$	3	$0.9^4 214$
Regulator	$0.9^2 863$	1	$0.9^2 863$
Relief valve	$0.9^3 221$	2	$0.9^2 884$
Propellant tank	$0.9^3 865$	4	$0.9^3 460$
Expulsion bellows	$0.9^2 870$	4	$0.9^2 508$
Check valve	$0.9^2 655$	2	$0.9^2 310$
Fittings & connections	$0.9^4 11$	5	$0.9^3 555$
Lines & Ducts Assy	$0.9^3 832$	1	$0.9^3 832$
		Total	0.9825

Table 13. Uncertainty of Reliability Predictions

A rough estimate of the uncertainty of the reliability predictions of the propellant feed system can be obtained from the range between the best and worst case failure rates. These are summarized as follows:

	No. Req'd		Failure rate, x 15^9 hrs	
Component	Mono	Biprop	Worst	Best
Helium Sphere	2	2	1,590	20
Hand Valve	3	5	224	20
Filter	2	3	1,800	30
Regulator	1	1	7,870	200
Relief Valve	1	2	5,700	200
Propellant Tank	2	4	1,640	20
Bellows	0	4	450	100
Bladder	2	0	700	180
Check Valve	0	2	4,700	200
Fittings	3	5	710	20
Line Assy	1	1	1,000	150
	17	29		

Monopropellant ave. $\frac{28,832}{17} = 1,694$ $\frac{1170}{17} = 68.8$

Bipropellant ave. $\frac{51,280}{29} = 1,770$ $\frac{1960}{29} = 67.5$

Table 13 (Continued)

Equivalent Mission Failure Rate

$$R_S : R_B R_T$$

where

R_S = Mission reliability of system

R_B = System reliability during boost

R_T = System reliability during transit

$$R_S = e^{-\lambda t_B K_B cn} \cdot e^{-\lambda t_T K_T cn}$$

where

λ = Ave. mission failure rate per component

t_B = Ave. boost duration = 0.43 hr

t_T = Ave. transit duration = 5,040 hrs

K_B = Boost environmental factor = 1000

K_T = Transit environmental factor = 1

c = Criticality factor, ave. ≈ 0.7

n = No. of components = 17 for monoprop.
= 29 for biprop.

$$R_S = \exp - \left| \lambda nc(t_B K_B + t_T K_T) \right|$$

Mono:

$$0.9^2 225 = \exp - [\lambda (17)(0.7)(430 + 5040)] = \exp - [65,100\lambda]$$

$$\lambda_{mono} = \frac{0.00778}{65,100} = 119.4 \times 10^{-9} \text{ hrs}$$

Biprop:

$$0.9825 = \exp - [\lambda (29)(0.7)(430 + 5040)] = \exp - [111,000\lambda]$$

$$\lambda_{bi} = \frac{0.0177}{111,000} = 159 \times 10^{-9} \text{ hrs}$$

Table 13 (Continued)

An estimate of the best and worst reliability is obtained by using the coefficients of λ found above.

Mono:

$$R_{S(worst)} = \exp - [65,100(1,694 \times 10^{-9})]$$

$$= \underline{\underline{0.8957}}$$

$$R_{S(best)} = \exp - [65,100(68.8 \times 10^{-9})]$$

$$= \underline{\underline{0.9^2552}}$$

Biprop:

$$R_{S(worst)} = \exp - [111,000(1770 \times 10^{-9})]$$

$$= \underline{\underline{0.8214}}$$

$$R_{S(best)} = \exp - [111,000(67.5 \times 10^{-9})]$$

$$= \underline{\underline{0.9^2254}}$$

4.3 Selected Propellant Feed System

Following is a reliability analysis of an integrated pressurization and propellant feed system for the monopropellant midcourse propulsion system of the Voyager spacecraft. This is the configuration adopted in the selected spacecraft design.

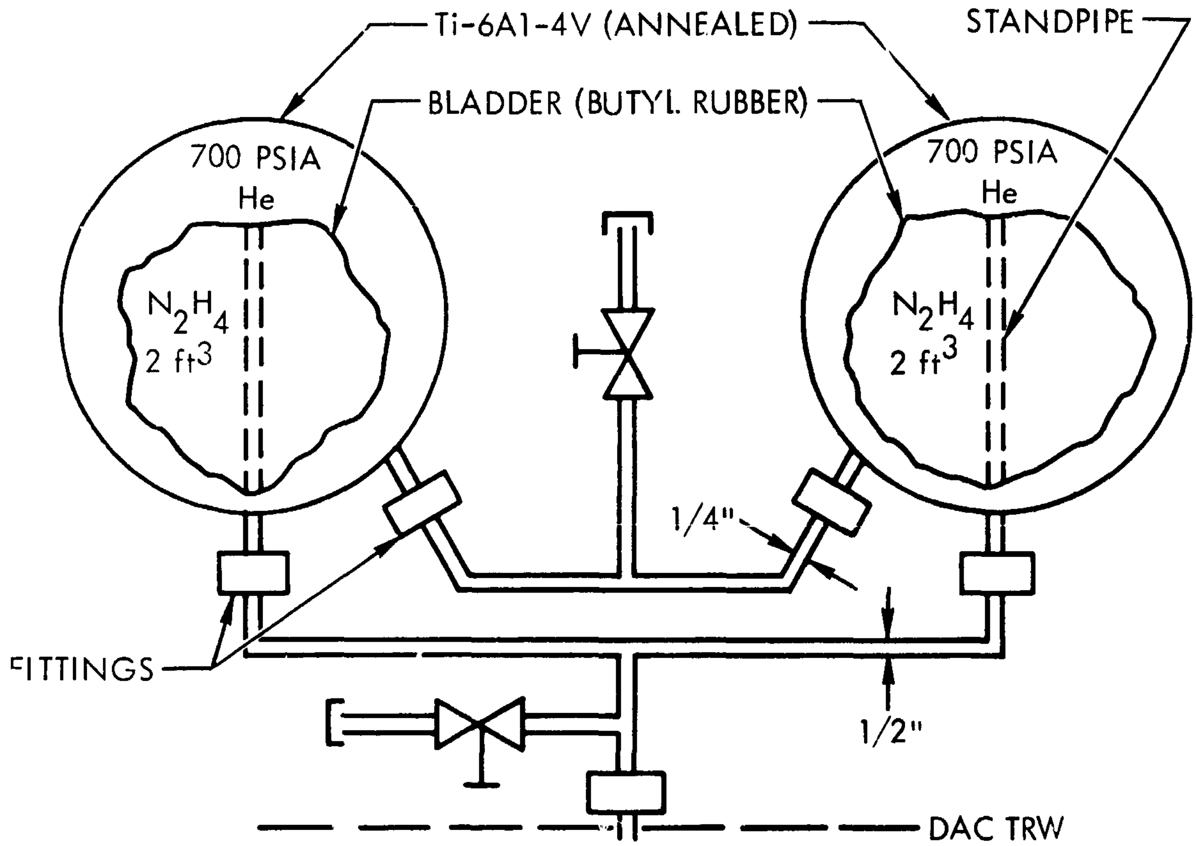
The analysis is based on the same assumptions and ground rules used in the preliminary reliability analysis of Section 4.2.* The analysis has been divided into five mission phases as follows:

		Reliability
Phase 1 -	Launch and Boost	$0.9^{(3)}050$
Phase 2 -	Transit of 4,280 hours including four midcourse correction firings.	$0.9^{(2)}723$
Phase 3 -	Retropropulsion and 2 days in Martian orbit.	$0.9^{(4)}792$
Phase 4 -	One additional month in Martian orbit.	$0.9^{(3)}720$
Phase 5 -	Five additional months in Martian orbit.	$0.9^{(3)}095$
	Total Mission:	$0.9^{(2)}508$

The major unreliability of the system occurs in Phase 2 where the midcourse correction firings are made. The unreliability which occurs in other phases is primarily due to leakage and the hazard associated with tank rupture and other structural failure within the pressurized system.

*Some of the component failure rates and failure probabilities differ between Sections 4.2 and 4.3. This is because the selected system was analyzed at a later date using revised failure rates based on a more extensive search for applicable failure rate data.

The selected monopropellant configuration is shown below.



Pressure vessels are 24" dia. The failure rate data used is an updated version of the applicable reliability assessment data sheets in Appendix "A" of DAC-VOYAGER memo DAC-VM-28 dated 6-24-65.

Failure Probability (10^{-6})

Component	Phase 1	Phase 2	Phase 3	Phase 4	Phase 5
Pressure Vessel (2)	786.4	881.6		100	5
Capped Hand Valve (2)	0.186	2.68			
Bladder (2)	86.4	776.24			
Standpipe (2)	2.46	35.10			
Fittings (5)	75.0	1075	20.85	180	900
T1-A1 Interface					
	950.446	277062	20.85	280.000	905

$$R_{mission} = \left[R_{phase\ 1}\right] \left[R_{phase\ 2}\right] \left[R_{phase\ 3}\right] \left[R_{phase\ 4}\right] \left[R_{phase\ 5}\right]$$

$$R_{mission} = (9^{(3)}050)\ (9^{(2)}723)\ (9^{(4)}792)\ (9^{(3)}720)\ (9^{(3)}095) = 9^{(2)}508$$

Table 1A

Item: Fittings (titanium-aluminum) interface

System/Subsystem: Propulsion/Pressurization Subsystem

Environmental Notes:

Leakage rate must be less than 10^{-6} SCCM

Primary Failure Mode:

Leakage at fitting

Possible Causes:

Damaged threads, improper torquing

Effects of Subsystem/Mission:

Loss of mission depending on severity

Backup Provisions:

Shut-off solenoid valves and regulator bank for fittings at pressurization tank. No backup for failure of fitting at fuel tanks.

Inherent Preventives:

Quality control and inspection procedures

Table 1B

	Nominal	Worst	Best	1	2A	2B	3A	3B	4	5
Failure Classification				0.5	0.1	0.5	0.1	0.5	0.5	0.5
Time/Cycles Units: hrs.				0.3	0.056	4280	0.167	50	720	3600
Basic Failure Rate, Units: $\times 10^{-9}$	100	710	40	100	100	100	100	100	100	100
Environment/Application Factor				1000	1000	1.0	1000	1.0	1.0	1.0
Actual Failure Rate, Units: $\times 10^{-9}$				15000	560	214000	1670	2500	36000	180000
Failure Rate Source Code	(a)	(b)	(c)							
Probability of Failure, $\times 10^{-6}$				15.0	0.560	214	1.67	2.5	36	180
Reliability/fitting				$9^{(4)}850$	$9^{(6)}440$	$9^{(3)}786$	$9^{(5)}833$	$9^{(3)}750$	$9^{(4)}64$	$9^{(3)}820$
					$9^{(3)}786$ (2A+2B)		$9^{(5)}586$ (3A+3B)			

Failure Rate Data Sources: (a) (b) (c) D. R. Earles, Reliability Engineering Data Series, April 1962

Table 2A

Item: Pressure Vessel

System/Subsystem: Propellant Feed

Operational Notes:
Titanium

Primary Failure Mode:
Leak

Possible Causes:
Defective welding, micrometeoroid impingement

Effects on Subsystem/Mission:
Dependent on degree of leak

Backup Provisions:
None

Inherent Preventives:
Closed loop control of all welding manufacturing processes, adequate meteoroid shielding, quality control inspection and testing in accordance with NPC 200-2, NPC 200-3

Secondary Failure Mode:
Rupture

Possible Causes:
Faulty weld or connection; flaw in material

Effects on Subsystem/Mission:
Tank explosion/loss of mission

Backup Provisions:
None

Inherent Preventives:
Same as above

Table 2B

Mode		Nominal	Worst	Best	1	2A	2B	3A	4	5
Primary Mode	Failure Classification				0.8	0.5	0.5	0.3		
	Time/Cycles Units: hrs.				0.3	0.056	4280	0.167		
	Basic Failure Rate, Units: x 10^{-9}	180	5000	60	130	180	180	180		
	Environment/Application Factor				1000	1000	1.0	1000		
	Actual Failure Rate, Units: x 10^{-9}				43200	5040	385200	9018		
	Failure Rate Source Code	(a)	(b)	(c)						
	Probability of Failure, x 10^{-6}				43.2	5.04	385.2	9.018*		
Secondary Mode	Failure Rate Source Code	(d)								
	Probability of Failure, x 10^{-6}				50	50		50*	50	5
	Probability of Failure, All Modes				393.2	55.4	385.2	59.018	50	5
	Reliability				$9^{(3)}607$	$9^{(4)}450$	$9^{(3)}615$	$9^{(4)}410$	$9^{(4)}500$	$9^{(5)}5$
						$9^{(3)}560$ (2A and 2B combined)				

Failure Rate Data Sources:

(a) FARADA p2. 384 Source 138 (Martin) } Primary Failure Mode
(b) FARADA p2. 384 Source 36 (Wright-Patterson AFB) } Primary Failure Mode
(c) FADADA p2. 385 Source 138 (Martin) } Primary Failure Mode
(d) FADADA p2. 385 Source 85 (Bell) - Secondary

Notes: *Applies only to a propellant configuration.
l Phase 5 is considered to be one order of magnitude lower than Phase 2.

Table 3A

Item: Standpipe

System/Subsystem: Propulsion/Propellant Feed

Item Type/Description:

Perforated pipe that facilitates fuel loading

Operational Notes:

Pipe will be made of titanium and welded into place

Environmental Notes:

Standpipe will be exposed to hydrazine environment

Primary Failure Mode:

Standpipe broken

Possible Causes:

Defective welding

Effects on Subsystem/Mission:

Flow of fuel into bladder will be erratic

Backup Provisions:

None

Inherent Preventives:

Radiographic inspection of welds

Secondary Failure Mode:

Burrs and imperfections on perforation holes

Possible Causes:

Inadequately controlled machine processes and inspection procedures

Effects on Subsystem/Mission:

Contaminant in fuel tank and engine resulting in possible engine damage, possible loss of mission

Backup Provisions:

None

Inherent Preventives:

Carefully deburr all perforations; polish standpipe

Table 3B

		Nominal	Worst	Best	1	2	3
Primary Mode	Failure Classification				0.9	0.9	0.9
	Time/Cycles Units: hrs.				0.3	4280	50
	Basic Failure Rate, Units: $\times 10^{-9}$	4.0	28.4	1.6	4.0	4.0	4.0
	Environment/Application Factor				1000	1.0	1.0
	Actual Failure Rate, Units: $\times 10^{-9}$				1080	15408	180
	Failure Rate Source Code	(a)	(b)	(c)			
	Probability of Failure, $\times 10^{-6}$				1.08	15408	0.18*
Secondary Mode	Failure Classification				0.5	0.5	0.5
	Time/Cycles Units: hrs.				0.3	4280	50
	Basic Failure Rate, Units: $\times 10^{-9}$	1.0	7.1	0.4	1.0	1.0	1.0
	Environment/Application Factor				1000	1.0	1.0
	Actual Failure Rate, Units: $\times 10^{-9}$				150	2140	25
	Failure Rate Source Code	(a)	(b)	(c)			
	Probability of Failure, $\times 10^{-6}$				0.15	2.14	0.205*
	Probability of Failure, All Modes				1.23	17.548	0.205
	Reliability				$9^{(5)}880$	$9^{(4)}820$	$9^{(6)}800$

Failure Rate Data Sources: (a) (b) (c) D. R. Earles, Reliability Engineering Data Series, April, 1962 (mechanical fitting) failure rate.

Notes:

The secondary failure mode is 20% of the total. Also, the failure rate for welds is considered to be 5% of that for fittings (mechanical).

*Applies only to bipropellant configuration.

Table 4A

Item: Bladder (Butyl Rubber)

System/Subsystem: Propulsion (Propellant Feed)

Item Type/Description:

Thin membrane type bag which contains the M MH or N_2 H_4 fuel located in tank.

Operational Notes:

Partial pressure of He or N_2 should be equal to or greater than the vapor pressure of the Hydrazine.

Primary Failure Mode:

Leakage due to permeation, cracks, pin holes, etc. or complete rupture.

Possible Causes:

Faulty Manufacturing Process.

Effects on Subsystem/Mission:

Burst/loss of mission.

Inherent Preventives:

Quality Control (NPC 200-2 and 200-3)

Secondary Failure Mode:

Blockage of tank outlet.

Effect on Mission:

Loss of Mission.

Table 4B

Mode		Nominal	1	2A	2B	3
Primary Mode	Failure Classification		0.8	0.5	0.5	0.3
	Time/Cycles Units: hrs.		0.3	0.056	4280	0.167
	Basic Failure Rate, Units: $\times 10^{-9}$	180	180	180	180	180
	Environment/Application Factor		1000	1000	1.0	1000
	Actual Failure Rate, Units: $\times 10^{-9}$		43200		387000	9018
	Failure Rate Source Code	(a)				
	Probability of Failure, $\times 10^{-6}$		43.2	not included	387	9.018*
Secondary Mode	Failure Classification		-	1.0	-	1.0
	Time/Cycles Units: hrs.		-	0.056	-	0.167
	Basic Failure Rate, Units: $\times 10^{-9}$	20	-	20	-	20
	Environment/Application Factor		-	1000	-	1000
	Actual Failure Rate, Units: $\times 10^{-9}$		-	1120	-	3340
	Probability of Failure, $\times 10^{-6}$		-	1.12	-	3.34*
	Probability of Failure, All Modes, $\times 10^{-6}$		43.2	1.12	387	12.358
	Reliability		$9^{(4)}568$	$9^{(5)}888$	$9^{(3)}613$	$9^{(4)}876$
				$9^{(3)}612$ (2A and 2B combined)		

Failure Rate Data Sources: (a) FARADA p. 2.278, Source 17 (Radioplane)

Notes: Blockage failure mode is 10% of total.

*Phase 3 is applicable only for bipropellant configuration. Phase 2A failure probability not included because it is only 1.5% of failure probability for Phase 2B (Primary Failure Mode).

Table 5A

Item: Hand Valve (manual)

System/Subsystem: Propulsion/Propellant Feed

Operational Notes:

This valve is used on the ground for filling gaseous helium or nitrogen tank and propellant tanks.

Environmental Notes:

Valve will be capped during flight but it might be susceptible to leakage through stem.

Primary Failure Mode:

Valve leaks during flight.

Possible Causes:

Faulty capping, imperfections in body of valve and seals.

Effects on Subsystem/Mission:

Loss of pressurization gas or propellant resulting in reduced impulse from engine. Erratic engine performance. Possible loss of mission.

Inherent Preventives:

Quality control and inspection (NPC 200-2 and NPC 200-3)

Table 5B

	Nominal	Worst	Best	1	2A	2B	3A*	3B
Failure Classification				0.5	0.5	0.5	0.5	0.01
Time/Cycles Units: hrs.				0.3	0.056	4280	0.167	50
Basic Failure Rate, Units: $\times 10^{-9}$	62	6520	9	62	62	62	62	62
Environment Application/Factor				10.00	10.00	0.01	10.00	0.01
Actual Failure Rate, Units: $\times 10^{-9}$				93.0	17.36	1326.8	51.77	0.31
Failure Rate Source Code	(a)	(b)	(c)					
Probability of Failure, $\times 10^{-6}$				0.093	0.0174	1.327	0.0518	0.00031
Probability of Failure All Modes				0.093	0.0174	1.327	0.052	100031
Reliability				$9^{(7)}070$	$9^{(7)}983$	$9^{(5)}867$	$9^{(7)}480$	$9^{(4)}690$
					$9^{(5)}866$ (2A–2B)		$9^{(7)}477$ (3A–3B)	

Failure Rate Data Sources:

(a) FARADA p. 2.403, Source 83 (Grumman Aircraft)

(b) FARADA p. 2.403, Source 110 (Boeing Aircraft)

(c) FARADA p. 2.403, Source 114 (Lockheed Aircraft)

Notes:

*Phase 3A is not applicable for monopropellant. Environmental factors are lowered two orders of magnitude to account for valve being capped.

4.4 Low-Gain Antenna Deployment

Reliability analyses were performed for four different configurations of the low gain antenna deployment system. The purpose of the analyses was to provide a preliminary reliability prediction for each configuration for comparison.

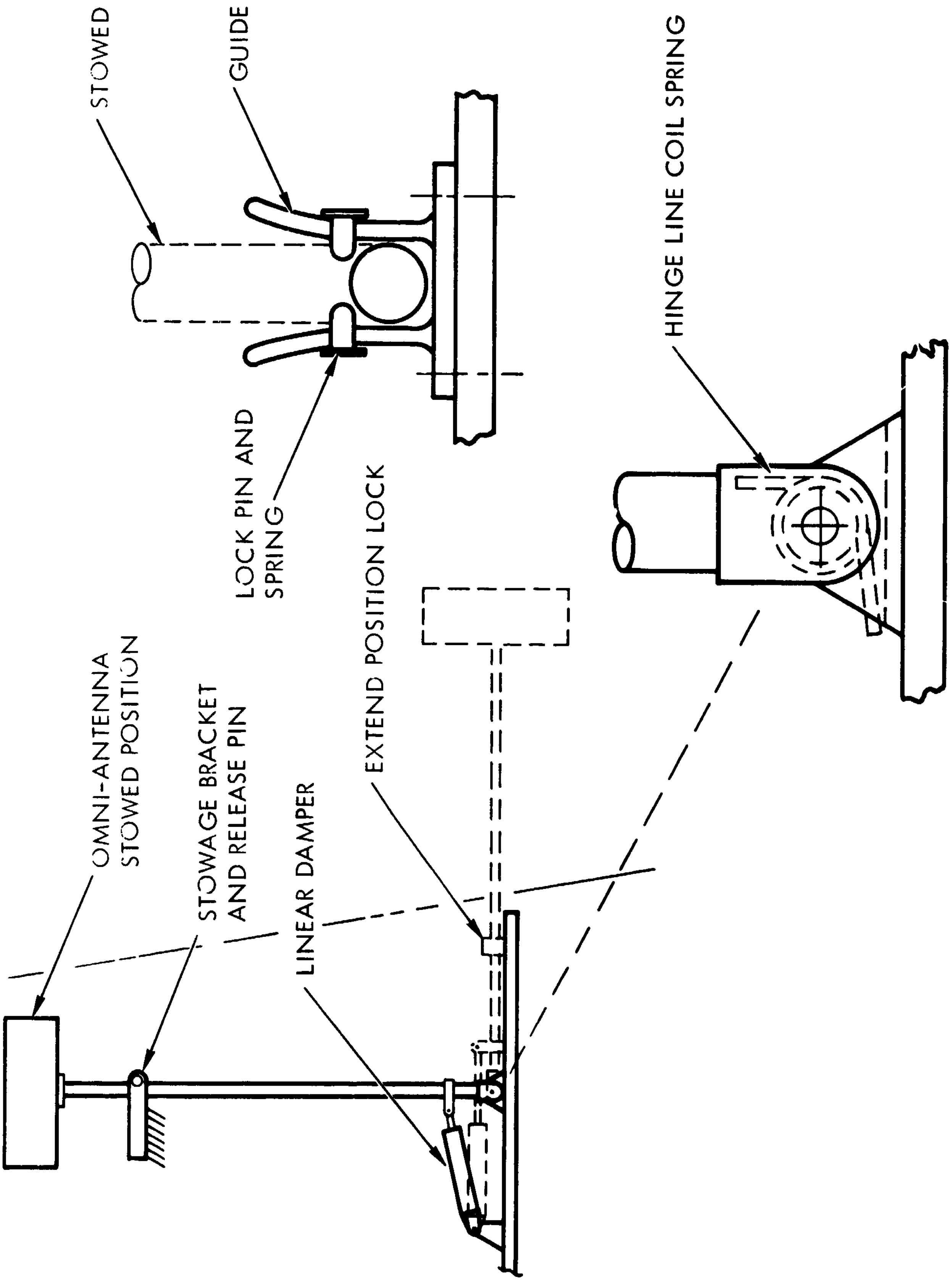
Each configuration, contained in the attached Douglas drawings, (SK-BM00-7001 through SK-BM00-7004) is treated separately as a section which includes the reliability predictions. In addition, this report contains the mathematical models used and reliability analysis of the parts required to compute the preliminary reliability predictions.

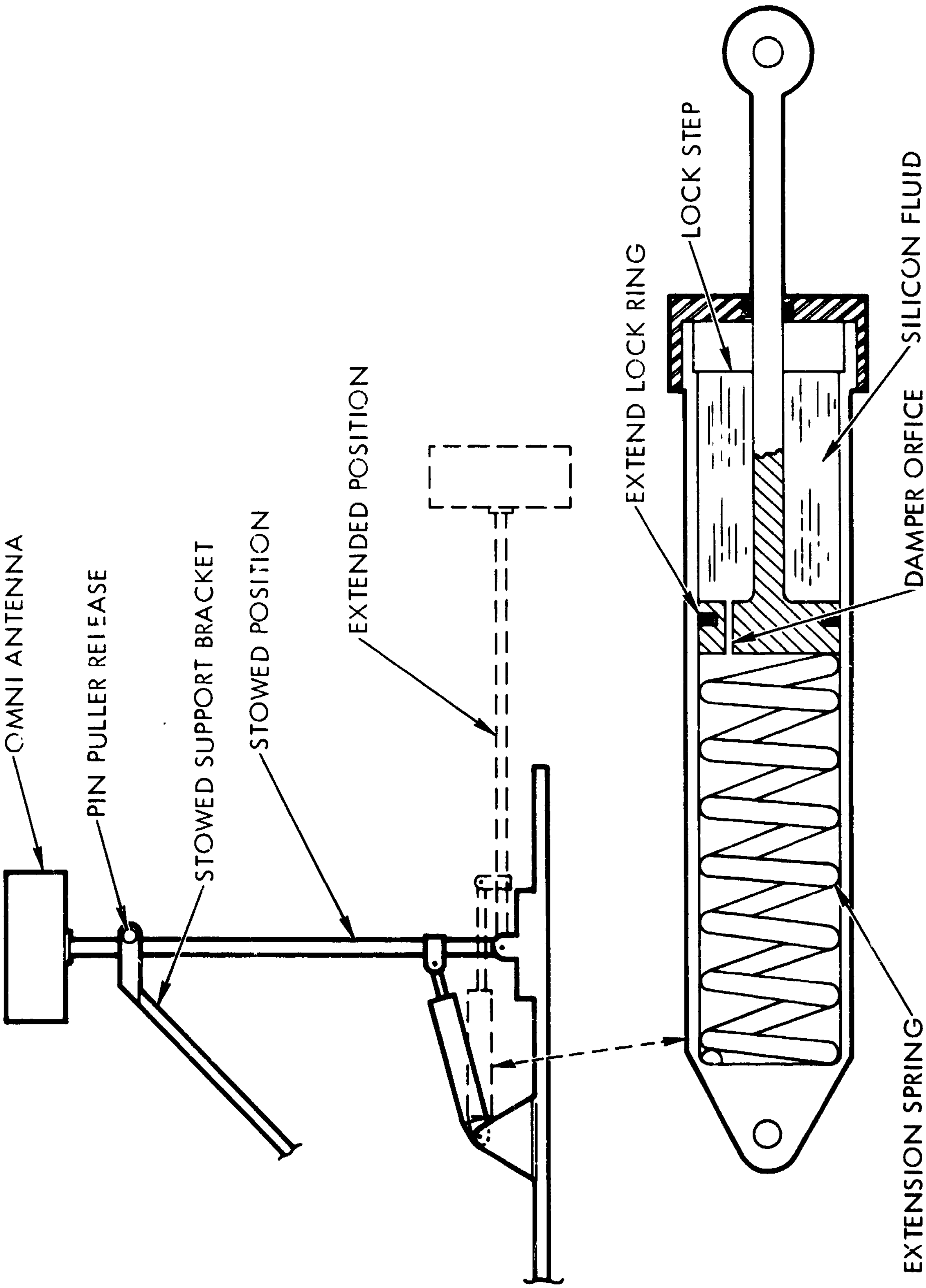
Table 1 presents a summary of the preliminary reliability predictions for each of the configurations considered in the analysis. The table shows that the deployment system which uses no squibs had the highest reliability prediction. In the remaining configurations the squibs are the least reliable portion of the system; the next most unreliable component is the damper/actuator which limits the reliability improvement.

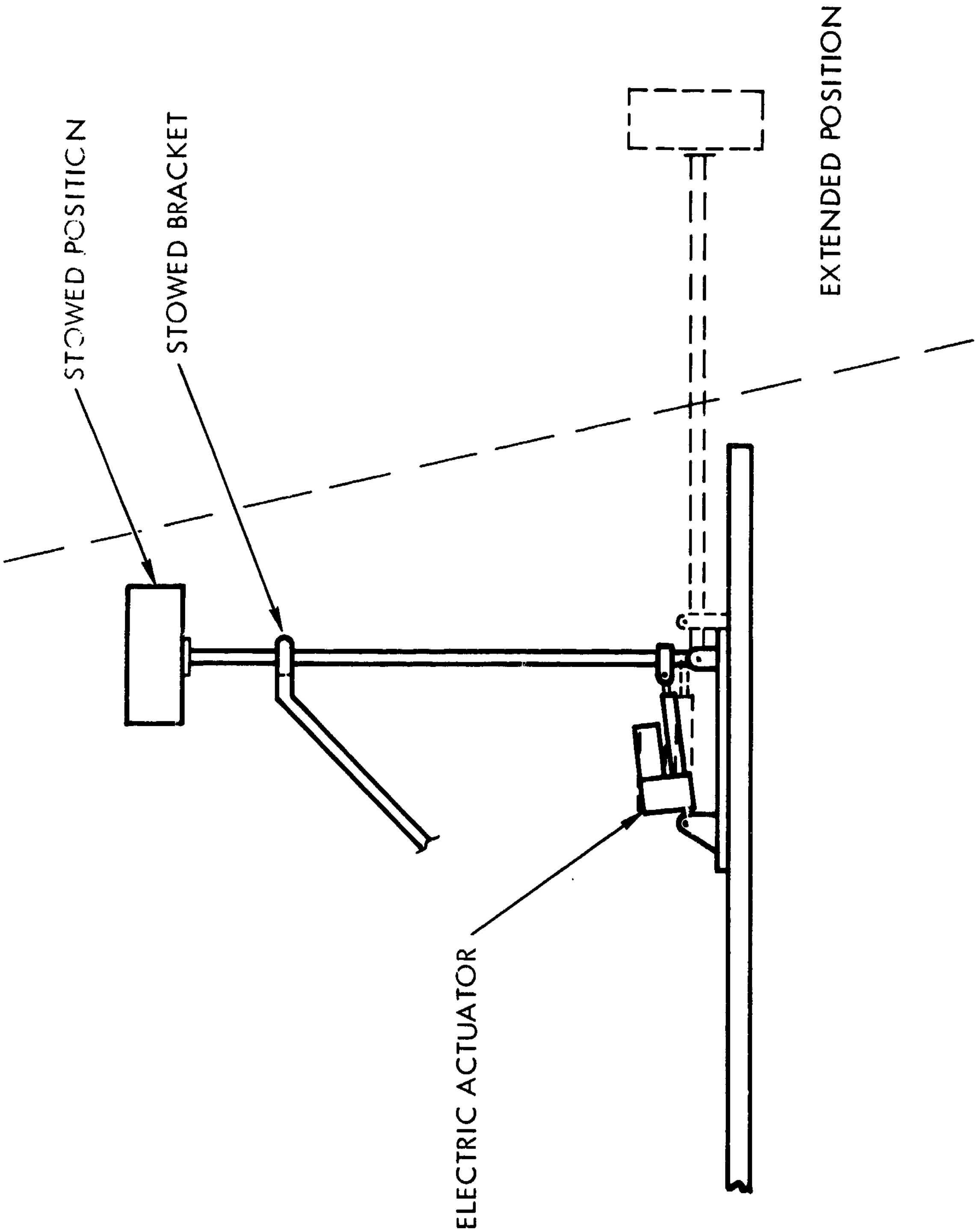
4.4.1 Hinge Spring Extension System (SK-BM00-7001)

The hinge spring extension system consists of a pin puller which keeps the antenna in the stowed position and releases the antenna for deployment upon firing of the squib(s). The system also consists of a coil spring for deployment, a hydraulic damper to prevent the antenna from structural failure during deployment and two lock pins and springs to lock the antenna in place.

Table 1 presents the reliability predictions for using the single and dual (redundant) squibs in the pin puller. Although the squib in the pin puller is the most unreliable portion of the system, the damper becomes the limiting factor when the squibs are made redundant and the gain in reliability is not larger ($<$ an order of magnitude). The primary failure mode of the damper is the loss of fluid. A major loss of fluid would prevent the damper from providing the necessary damping force causing the antenna to break during deployment. How much fluid loss could be tolerated has not been determined. Since the reliability figure used for


STOWED
GUIDE
HINGE LINE COIL SPRING
LOCK PIN AND SPRING
EXTEND POSITION LOCK
OMNI-ANTENNA STOWED POSITION
STOWAGE BRACKET AND RELEASE PIN
LINEAR DAMPER


OMNI ANTENNA
PIN PULLER RELEASE
STOWED SUPPORT BRACKET
STOWED POSITION
EXTENDED POSITION
EXTEND LOCK RING
LOCK STEP
SILICON FLUID
DAMPER ORFICE
EXTENSION SPRING


STOWED POSITION
STOWED BRACKET
EXTENDED POSITION
ELECTRIC ACTUATOR

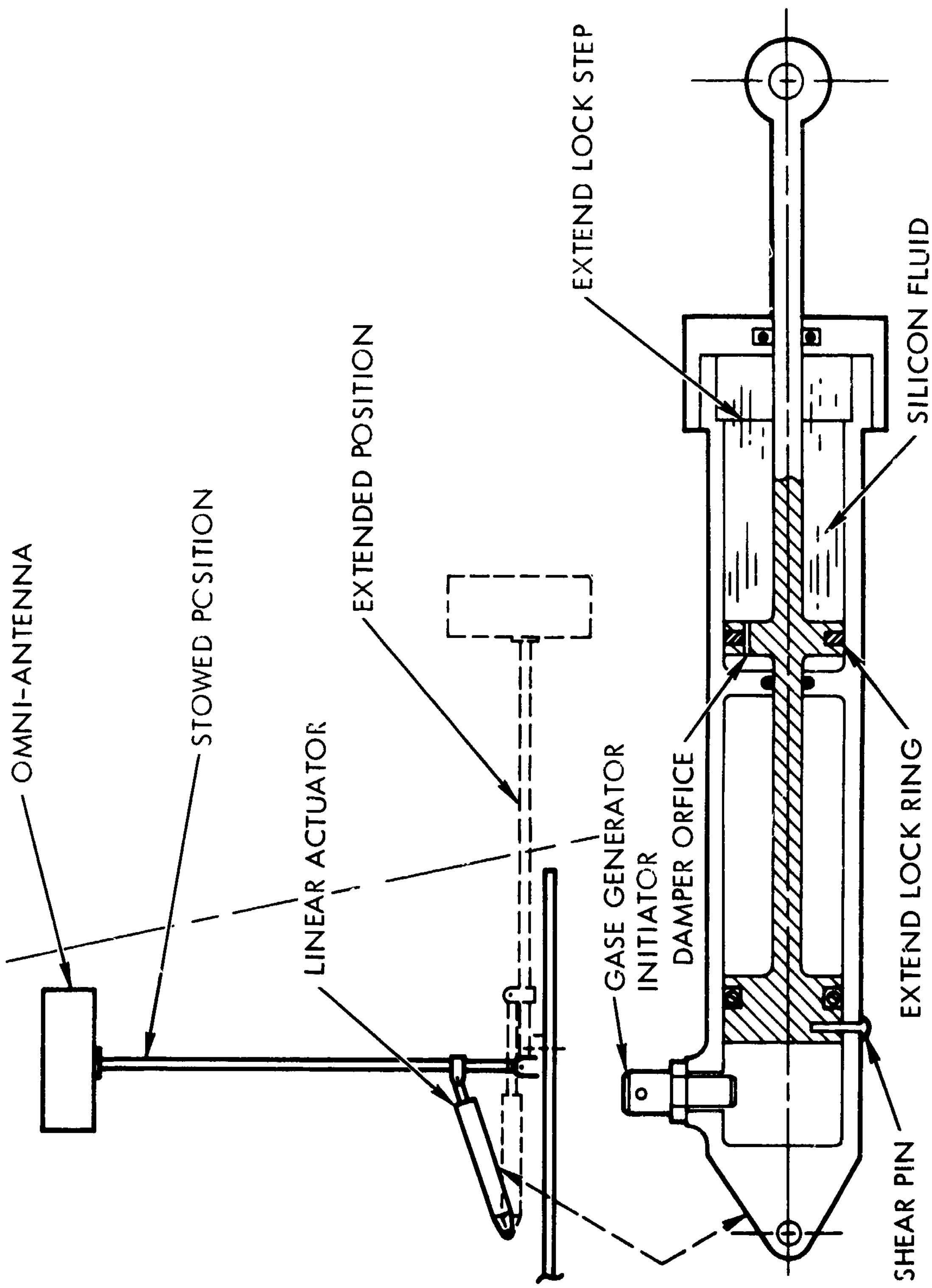

OMNI-ANTENNA
STOWED POSITION
LINEAR ACTUATOR
EXTENDED POSITION
EXTEND LOCK STEP
SILICON FLUID
GASE GENERATOR
INITIATOR
DAMPER ORFICE
EXTEND LOCK RING
SHEAR PIN

Table 1. Preliminary Reliability Predictions for Low-Gain Antenna Deployment Configurations

Subsystem Configurations	Reliability Predictions
1. Hinge Spring Extension System SK-BM00-7001 a. Single Squib b. Redundant Squibs	 $0.9^{(3)}258$ $0.9^{(3)}691$
2. Linear Spring Actuator System SK-BM00-7002 a. Single Squib b. Redundant Squibs	 $0.9^{(3)}314$ $0.9^{(3)}747$
3. Electric Omni-Antenna Deployment System SK-BM00-7003	$0.9^{(3)}838$
4. Gas Linear Actuator System SK-BM00-7004 a. Single Squib b. Redundant Squibs	 $0.9^{(3)}441$ $0.9^{(3)}785$

the damper is considered to provide for zero leakage, the total figure for the system may be conservative.

4.4.2 Linear Spring Actuator System (SK-BM00-7002)

This system is similar to the previous system except the extension spring and locking device are all contained in the damper, which in effect makes it an actuator. The reliability predictions are shown in Table 54. The increase in reliability results from the simplified locking device.

4.4.3 Electric Deployment (SK-BM00-7003)

This configuration uses no ordnance devices and depends upon an electric actuator to keep the antenna in the stowed position and to deploy and lock the antenna. The reliability prediction for the system is $0.9^{(3)}838$, the highest reliability of all the configurations considered.

4.4.4 Gas Linear Actuator (SK-BM00-7004)

This configuration is similar to the spring-actuator configuration except that the actuator uses gas pressure, from squibs, rather than a spring to deploy the antenna. The antenna is held in the stowed position by a shear pin which must be broken to start the antenna deployment. The reliability predictions of the configurations considered are shown in Table 2.

4.4.5 Reliability Analysis and Mathematical Model

The reliability analysis for each of the major parts considered in each of the four different configurations is contained in Tables 3 through 10. For the most part the failure modes of the components were determined and the failure rate of the parts contributing to this failure mode were used (e.g. in Tables 6A and 6B the failure rates for seals and orifices were used rather than the failure rate of a damper or actuator). This method of prediction gives a better indication of the possible problem areas.

The reliability of each one shot device is simply the complement of its failure rate (λ), where the failure rate is essentially the probability of failure (Q)

$$R = 1 - \lambda = 1 - Q$$

If two devices are used redundantly, both must fail before the system is considered failed and the reliability is

$$R = 1 - Q^2$$

Table 2. Preliminary Reliability Prediction for Gas Linear Actuator System (SK-BM00-7004)

Configuration	Reliability Prediction
1. Single Squib	$0.9^{(3)}441$
2. Redundant Squibs	$0.9^{(3)}785$

Table 3A

Item: Squib

System/Subsystem: Low Gain-Antenna Deployment

Item Type/Description:

Explosive bridge wire with powder charge. (Redundant bridge-wires)

Operational Notes:

One shot device.

Environmental Notes:

Failure is most likely to result from vibration during the boost phase.

Primary Failure Mode:

Failure to fire, with proper force, when proper electrical input is present.

Possible Causes:

1. Poor workmanship (bridgewire coating).
2. Bad lot of charges.
3. Broken bridgewires.

Effects on Subsystem/Mission:

Catastrophic, failure to deploy.

Backup Provisions:

Redundant squibs.

Inherent Preventives:

Lot qualification of squibs.

Secondary Failure Mode:

Premature firing.

Possible Causes:

1. Stray R.F. signals.
2. Boost environment.
3. Bad lot of charges.

Effects on Subsystem/Mission:

Catastrophic, early deployment.

Backup Provisions:

None

Inherent Preventives:

Bruceton tests, shielding from R.F. signals.

Table 3B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles x 10^{-6}	1
	Basic Failure Rate, Units: Per 10^6 Firings	294
	Environment/Application Factor	-
	Actual Failure Rate, Units: Per 10^6 Firings	294
	Failure Rate Source Code	a
	Probability of Failure, x 10^{-6}	294
Secondary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles x 10^{-6}	1
	Basic Failure Rate, Units: Per 10^6 Firings	6*
	Environment/Application Factor	-
	Actual Failure Rate, Units: Per 10^6 Firings	6
	Failure Rate Source Code	a
	Probability of Failure x 10^{-6}	6
	Probability of Failure, All Modes	300
	Reliability	0.9(3)700

Failure Rate Data Sources: a) TRW experience - 2000 squib firings with no failures. Prob. of failure of 3 x 10^{-4} at 50% C.L. Telecon with Farada indicated 47,332 firings with 10 failures. Prob. of failure, 2.11 x 10^{-4}.

Notes:

A. Catastrophic.

* Experience from Bruceton type testing indicates premature failure mode is approximately 1/50 of total failure probability.

Table 4A

Item: Pin Puller

System/Subsystem: Low Gain-Antenna Deployment

Item Type/Description:

Captive with dual cartridge capability.

Operational Notes:

Gas retaining, no fragmentation.

Primary Failure Mode:

Failure to release pin when proper force is present.

Possible Causes:

1. Failure of pin to move (jammed)
2. Broken pin

Effects on Subsystem/Mission:

Catastrophic — deployment would not occur.

Backup Provisions:

None

Inherent Preventives:

Lot qualification of pin pullers.

Table 4B

	Mission
Failure Classification	A
Time/Cycles,Units: x 10^{-6} Firings	1
Basic Failure Rate, Units, x 10^{6} Firings	48
Environment/Application Factor	-
Actual Failure Rate, Units: x 10^{-6} Firings	48
Failure Rate Source Code	b
Probability of Failure, x 10^{-6}	48
Reliability	$0.9^{4}52$

b) Farada, revised 3/1/65. Pg. 2.279, Source 138. Explosive bolt used - estimates for locking, etc. - 10% of failure rate.

Notes:

A ibid

Table 5A

Item: Cable Assy (Squib Firing Circuit)

System/Subsystem: Low-Gain Antenna Deployment

Item Type/Description:

A 2-pin connector, 2-solder joints and two wires were considered for each cable assy.

Environmental Notes:

Failure could occur as a result of vibration during boost.

Primary Failure Mode:

Failure to provide electrical signal to squib.

Possible Causes:

1. Poor pin contact
2. Broken wire or connection.

Effects on Subsystem/Mission:

Catastrophic - no deployment.

Backup Provisions:

Redundant squibs used; therefore cable assy are considered redundant.

Inherent Preventives:

Circuit continuity check.

Table 5B.

	Mission
Failure Classification	A
Time/Cycles Units: Hrs	0.3
Basic Failure Rate, Units: x 10^{-6} Hrs.	0.438*
Environment/Application Factor	1000
Actual Failure Rate, Units: x 10^{-6} Hrs.	438
Failure Rate Source Code	c
Probability of Failure, x 10^{-6}	145
Reliability	$0.9^{(3)}855$

Failure Rate Data Sources: c) Farada, Revised 3/1/65, Source 138

Notes:

A ibid.

* Total failure rate determined as follows:

1.	Connector	- 0.2/Pin x 2	=	0.400
2.	Wire	- 0.015 x 2	=	0.03
3.	Solder Joint	- 0.004 x 2	=	0.008
				0.438

Table 6A

Item: Hydraulic Damper

System/Subsystem: Low Gain-Antenna Deployment

Operational Notes:

Used in SK-BM00-7001

Environmental Notes:

Failure could occur during boost environment.

Primary Failure Mode:

Leakage of fluid past seals.

Possible Causes:

1. Boost environment.
2. Poor workmanship.

Effects on Subsystem/Mission:

None to catastrophic, depending on amount of leakage (structural failure of antenna).

Backup Provisions:

3 antennas in system (redundant).

Inherent Preventives:

None.

Secondary Failure Mode:

Failure to extend antenna.

Possible Causes:

1. Clogged or wrong size orifice.
2. Jamming of piston.

Effects on Subsystem/Mission:

No communication.

Backup Provisions:

3 antennas in system (redundant).

Inherent Preventives:

None.

Table 6B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Hrs.	0.3
	Basic Failure Rate, Units: x 10^{-6} Hrs.	0.35*
	Environment/Application Factor	1000
	Actual Failure Rate, Units: x 10^{-6} Hrs.	350
	Failure Rate Source Code	c
	Probability of Failure, x 10^{-6}	105
Secondary Failure Mode	Failure Classification	A
	Time/Cycles Units: Hrs.	0.3
	Basic Failure Rate, Units: x 10^{-6} Hrs.	0.45**
	Environment/Application Factor	1000
	Actual Failure Rate, Units: x 10^{-6} Hrs.	150
	Failure Rate Source Code	c
	Probability of Failure x 10^{-6}	45
	Probability of Failure, All Modes	150
	Reliability	$9^{(3)}850$

Failure Rate Data Sources: c) ibid.

Notes:

* Failure rate of parts as follows:

Pg. 2.316 - "O" Ring - 0.05

Pg. 2.369 - Seal (Sliding) - 0.30

** Pg. 2.348 - Orifice - 0.15

A ibid.

Table 7A

Item: Coil Spring

System/Subsystem: Low Gain-Antenna Deployment

Operational Notes:

Used on SK-BM00-7001

Environmental Notes:

Failure would probably occur during boost.

Primary Failure Mode:

Break - fail to act as a spring. (Vacuum welding)

Possible Causes:

1. Boost environment.
2. Poor workmanship.

Effects on Subsystem/Mission:

Catastrophic - antenna would not deploy.

Backup Provisions:

3 Antennas in system (redundant).

Inherent Preventives:

None.

Table 7B

	Mission
Failure Classification	A
Time/Cycles Units: Hrs. x 10^{-6}	0.3
Basic Failure Rate, Units: x 10^{-6} Hrs.	0.11
Environment/Application Factor	1000
Actual Failure Rate, Units: x 10^{-6} Hrs.	110
Failure Rate Source Code	c
Probability of Failure, x 10^{-6}	33
Reliability	$0.9^{(4)}67$

Failure Rate Data Sources: c) ibid - Pg. 2. 374

Notes:

A ibid.

Table 8A

Item: Linear Spring Actuator

Item Type/Description:

Used on SK-BM00-7002.

Environmental Notes:

Failure could occur during boost.

Primary Failure Mode:

Leakage past the seals.

Possible Causes:

1. Boost environment.
2. Poor workmanship.

Effects on Subsystem/Mission:

None to catastrophic depending on amount of leakage (structural failure of antenna).

Backup Provisions:

3 antennas in system (redundant).

Inherent Preventives:

None.

Secondary Failure Mode:

Failure to deploy antenna.

Possible Causes:

Spring breakage or welding.

Effects on Subsystem/Mission:

Catastrophic.

Backup Provisions:

3 antennas in system (redundant).

Inherent Preventives:

None.

Other Failure Modes:

Failure to lock in place.

Possible Causes:

Broken lock ring.

Effects on Subsystem/Mission:

Degraded performance.

Backup Provisions:

Same as above.

Inherent Preventives:

None.

Table 8B

		Mission
Primary Failure Modes	Failure Classification	A
	Probability of Failure, x 10^{-6}	150*
Secondary Failure Modes	Failure Classification	A
	Probability of Failure, x 10^{-6}	33**
Other Failure Modes	Failure Classification	M
	Time/Cycles Units: Hrs x 10^{-6}	0.3
	Basic Failure Rate, Units: x 10^{-6}	0.033
	Environment/Application Factor	1000
	Actual Failure Rate, Units: x 10^{-6}	33
	Failure Source Code	d
	Probability of Failure x 10^{-6}	10
	Probability of Failure, All Modes	193
	Reliability	$0.9^{(3)}807$

Failure Rate Data Sources: d) Estimated based on other components

Notes:

A ibid
M Minor
* See Table 6B
** See Table 7B

Table 9A

Item: Electric Actuator

System/Subsystem: Low Gain-Antenna Deployment

Item Type/Description:

Used on SK-BM00-7003.

Primary Failure Mode:

Electrical Failure.

Possible Causes:

Motor.

Effects on Subsystem/Mission:

Catastrophic - no deployment.

Backup Provisions:

3 antennas in system (redundant).

Inherent Preventives:

Electrical checks.

Secondary Failure Mode:

Mechanical failure.

Possible Causes:

Structural failure of screw jack or gears.

Effects on Subsystem/Mission:

Catastrophic - no deployment.

Backup Provisions:

3 antennas in system (redundant).

Inherent Preventives:

None.

Table 9B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: x 10^{-6} Hrs.	0.3
	Basic Failure Rate, Units: x 10^{-6} Hrs.	0.3
	Environment/Application Factor	1000
	Actual Failure Rate, Units: x 10^{-6} Hrs.	300
	Failure Rate Source Code	e
	Probability of Failure, x 10^{-6}	90
Secondary Failure Mode	Failure Classification	A
	Time/Cycles Units: x 10^{-6} Hrs.	0.3
	Basic Failure Rate, Units: x 10^{-6} Hrs.	0.275
	Environment/Application Factor	1000
	Actual Failure Rate, Units: x 10^{-6} Hrs.	275
	Failure Rate Source Code	f
	Probability of Failure x 10^{-6}	82.5
	Probability of Failure, All Modes	172.5
	Reliability	$9^{(3)}828$

Failure Rate Data Sources: e) Farada, Revised 3/1/65, Pg. 1.100, Source 138 (Motor) f) Farada, Revised 3/1/65, Pg. 2.318, Source 82 (Screw Jack)

Table 10A

Item: Shear Pin

System/Subsystem: Low Gain-Antenna Deployment

Item Type/Description:

Used on SK-BM00-7004.

Primary Failure Mode:

Failure to shear when proper force is present.

Possible Causes:

Workmanship.

Effects on Subsystem/Mission:

Catastrophic - no deployment.

Backup Provisions:

3 antennas in system - (redundant).

Inherent Preventives:

None.

Secondary Failure Mode:

Premature shear.

Possible Causes:

1. Boost environment.
2. Workmanship.

Effects on Subsystem/Mission:

Catastrophic - early deployment.

Backup Provisions:

3 antennas in system - (redundant).

Inherent Preventives:

None.

Table 10B

	Mission
Failure Classification	A
Time/Cycles Units: x 10^{-6} cycles	1
Basic Failure Rate, Units: x 10^{-6}	6
Environment/Application Factor	-
Actual Failure Rate, Units: x 10^{-6}	6
Failure Rate Source Code	g
Probability of Failure, x 10^{-6}	6
Failure Classification	A
Time/Cycles Units: x 10^{-6} cycles	1
Basic Failure Rate, Units: x 10^{-6}	<1
Environment/Application Factor	-
Actual Failure Rate, Units: x 10^{-6}	<1
Failure Rate Source Code	g
Probability of Failure x 10^{-6}	<1
Probability of Failure, All Modes	6
Reliability	$9^{(5)}4$

Failure Rate Data Sources: g) Farada, revised 3/1/65, Pg. 2.351, Source 123.

Total system reliability is determined from the product rule

$$R_T = \prod_{i=1}^{N} R_i$$

In those instances where time is a factor the following is used to determine the reliability

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

where

K = Environmental and/or operational factor

λ = Failure rate

t = Mission time (boost 0.3 hour)

The boost time was selected since the actual operation time during deployment is small (seconds) and the majority of the failure modes considered would be caused by the boost environment rather than the system operation.

If $K\lambda t$ is small (<0.01) then the first equation can be used.

4.5 Explosive Bolts and Shaped Charges

This study presents a preliminary comparison between explosive bolts, explosive nuts, and a shaped-charge device for use in determining the method of separation for the Voyager spacecraft. The study is divided into two parts; (a) the general pros and cons of explosive bolts, explosive nuts, and shaped-charges and (b) a reliability analysis comparing several different configurations of the explosive bolts, explosive nuts, and shaped-charges. This portion contains a numerical analysis and a failure mode analysis.

The three primary reliability predictions are shown below:

6 Explosive Bolts: (all must fire)	0.99796
Single Shaped Charge: (Single Squib)	$0.9^{(3)}67$
12 Explosive Nuts: (2 at each point, redundant)	$0.9^{(5)}31$

In order to obtain the reliability of the nut configuration (a stud with a nut on either end) it would be necessary to use redundant shaped charges. Using this configuration would actually give a higher reliability; however, the increase in weight and the problems of shock and contamination argue against this approach.

The primary failure mode of all devices is the failure of the squib to fire with the required force when the electrical signal is present. An additional failure mode is present with the nuts which concerns the stud hanging up. The probability of this failure can be reduced by the use of sufficient ramp angles and oversized holes and control of tip-off angles.

4.5.1 Comparison

a. Explosive Bolts

The primary pros and cons associated with the use of explosive bolts are as follows:

(1) Explosive bolts have been used as a method of separation for many years and the probability of their success is well known.

(2) Explosive bolts, because of their size, are easily manufactured, transported and handled.

(3) The fact that all of the bolts in the system must fire to accomplish separation reduces the probability of success.

(4) The explosive bolts are not easily made redundant; however, they can be made redundant by using a spacer between the separation planes or an explosive nut as the restraining device.

(5) Fragmentation can be easily contained.

b. Shaped-Charged Devices

The primary pros and cons associated with the use of shaped-charge devices are as follows:

(1) Shaped-charge devices are a recent innovation; however, uses on such weapon systems as Minuteman and various payload fairings should provide a good confidence in their probability of success.

(2) The manufacturing (flatness and roundness), transportation and handling of shaped-charges could be a problem especially if the sections are large.

(3) The reliability of a single shaped-charge is high (probably higher than a single explosive bolt).

(4) A shaped-charge is readily adaptable to various redundant configurations.

c. Explosive Nuts

The pros of a(1), a(2), and a(5) also apply here and the cons of paragraph a(3) and a(4) are eliminated. The configuration considered uses a stud through the interface with a nut on both ends either of which will provide separation. Another failure mode is introduced: the probability of the stud hanging up.

4.5.2 Explosive Bolts vs. Shaped Charges

a. Failure Rates

The failure rates used in this study are based upon both TRW experience with cartridges and FARADA* information. TRW has fired over 2,000 squibs without encountering a failure. Using 2,000 firings without a failure, the statistical probability of success is 0.9997 at a 50% confidence level (best estimate). This figure is the total reliability used for the squibs in this study. This figure is considered to be conservative since many companies have data for a larger sample size. It has been reported that Hi-Shear has fired over 8,000 squibs without failure, a reliability of 0.999923 for the squib.

In order to determine the probability of the bolt fracturing as required and the shaped-charge cutting as required, FARADA was

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* Failure Rate Data Handbook, Bureau of Naval Weapons, 1 June 1962.

consulted. Although shaped-charges were not listed in FARADA, primacord was; therefore prima-cord was used as the failure rate for shaped-charges. The failure rate listed in FARADA for the two devices is as follows:

- Explosive bolts: 40 failures per 10^6 firings
- Prima-cord (shaped charge): 30 failures per 10^6 firings

Based upon this information, the shaped-charge device was considered to be more reliable than a single explosive bolt.

b. Model

The reliability of each one shot device (squib, explosive bolt or nut) is simply the complement of its failure rate (λ), where the failure rate is essentially the probability of failure (Q)

$$R = 1 - \lambda = 1 - Q.$$

If two devices are used in a redundant configuration, both must fail and the reliability is

$$R = 1 - Q^2.$$

Separation system reliability is determined from the product rule

$$R_T = \prod_{i=1}^{N} R_i.$$

c. Analysis

The reliability of several different configurations are shown in Table 1. Redundant squib firing circuits were assumed in all cases and therefore were not considered in the analysis. The explosive nut configuration uses a stud through the interface connected on both ends by an explosive nut. If either nut fires, the point is free to separate. A single squib firing circuit is used for each nut thereby becoming redundant, as are the nuts.

Table 1. Reliability Comparison of Separation Methods

Component	Reliability	Failures per Million Attempts
Explosive Bolt	$0.9^{(3)}66$ (each)	340.0 (each)
1. 6 out of 6	0.99796	2040.0
2. 4 out of 4	0.99864	1360.0
3. 3 out of 3	0.99898	1020.0
Shaped-Charge		
1. Single squib	$0.9^{(3)}67$	330.0
2. Dual squibs	$0.9^{(4)}7$	30.0
3. Redundant shaped-charge with redundant squibs in each	$0.9^{(9)}1$	0.0009
Explosive Nuts	$0.9^{(3)}66$ (each)	340.0 (each)
(2 used redundancy at each point)	$0.9^{(6)}88$ (per point)	0.12 (per point)
1. 6 out of 6 points	$0.9^{(6)}31$	0.69
2. 4 out of 4 points	$0.9^{(6)}54$	0.46
3. 3 out of 3 points	$0.9^{(6)}65$	0.35

Note: $0.9^{(6)}65 = 0.99999965$

Since the reliability numbers for all configurations are greater than 0.99, the third column of the table has been presented to aid in the interpretation of the reliability. The column presents the number of failures predicted per million attempts. The table indicates that the explosive bolt (with no redundancy) is the most unreliable and that the redundant shaped charge has the highest reliability. The explosive nuts, however, are more reliable than a single shaped-charge with redundant squibs. On a straight reliability comparison the redundant shaped-charge could be recommended.

A failure mode analysis is presented in the attached worksheets. Four separate items are considered: the squib, the bolt, the stud, and the shaped charge. The explosive nut is considered to be the same as the bolt. Sample calculations are shown on the final worksheets showing how the reliability of the total item is determined as well as some of the calculations of the reliability figures of merit in Table 1. As is indicated on the worksheets, the most probable mode of failure is failure of the squib to fire. The reliability of the squib is nearly an order of magnitude less than the mechanical reliability of either the bolt, the nut, or the shaped charge. The primary failure modes considered for these two items result from workmanship errors. In summary, if the squib fires with the proper force, the probability of the bolt fracturing or the shaped-charge cutting is high; however, the existence of the bolt and shaped-charge failure modes should not be overlooked and close quality control is required to obtain these high probabilities of success.

The probability of the stud hanging up is an estimate based on a comparison with the bolt. It is assumed that all precautions would be taken with the design to preclude the hanging up of the stud. The failure mode was not considered in the reliability calculations because several studs would have to hang up to prevent separation, which has negligible probability.

Table 2A

Item: Squib

System/Subsystem: Separation

Item Type/Description:

Exploding bridgewire with power charge.

Operational Notes:

One shot device.

Environmental Notes:

Failure is most likely to result from vibrations during the boost phase.

Primary Failure Mode:

Failure to fire when proper electrical imput is present.

Possible Causes:

1. Poor workmanship (Bridgewire coating).
2. Broken bridgewire due to boost environment.
3. Poor electrical connection.
4. Bad lot of charges.

Effects on Subsystem/Mission:

Catastrophic, failure to separate.

Backup Provisions:

None for the shaped charge and bolt configuration squibs redundant in the nut configuration.

Inherent Preventives:

Lot qualification of all squibs. Circuit continuity check.

Secondary Failure Mode:

Premature firing.

Possible Causes:

1. Boost environment.
2. A bad lot of charges.
3. Stray R. F. signals.

Effects on Subsystem/Mission:

Catastrophic, early separation.

Backup Provisions:

None for shaped charge; several would have to fire in the bolt and nut configuration to accomplish early separation.

Inherent Preventives:

Testing to determine probabilty of all fire and no fire currents. Shielding from R. F. signals.

Table 2B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles, Units:	1 cycle
	Actual Failure Rate, Units: Per 10^6 Firing	294
	Failure Rate Source Code	a
	Probability of Failure, x 10^{-6}	294*
Secondary Failure Mode	Failure Classification	A
	Time/Cycles, Units:	1 cycle
	Actual Failure Rate, Units:	6
	Failure Rate Source Code	a
	Probability of Failure, x 10^{-6}	6*
	Probability of Failure, All Modes x 10^{-6}	300
	Reliability	$0.9^{(3)}700$

Failure Rate Data Sources: a) TRW experience - 2000 squib firings with no failures gives probability of failure of 3×10^{-4} at 50% confidence.

Notes:

A Catastrophic failure; mission abort.

* Experience from Brewston type testing indicates premature failure mode is approximately 1/50 of total failure probability.

Table 3A

Item: Bolt (Explosive)

System/Subsystem: Separation.

Operational Notes:

One shot device. It is assumed that all fragments are contained.

<u>Primary Failure Mode:</u>

Failure to fracture when proper force is present.

Possible Causes:

Poor workmanship (The section is oversize)

Effects on Subsystem/Mission:

Separation would not occur.

Backup Provisions:

None.

Inherent Preventives:

None.

Table 3B

	Mission
Failure Classification	A
Time/Cycles, Units: Cycle	1
Actual Failure Rate, Units:	40
Failure Rate Source Code	b
Probability of Failure, x 10^{-6}	40
Probability of Failure, All Modes	40
Reliability	$0.9^{(4)}60$

Failure Rate Data Sources: b) Farada, revision dated 3/1/65, Pg 2.279, source 138: Martin Co. report M-63-3, dated Oct 63. (k factor not applicable)

Notes: A. Catastrophic failure, mission abort.

Table 4A

Item: Shaped Charge

System/Subsystem: Separation

Primary Failure Mode:

Failure to cut when proper force is applied.

Possible Causes:

1. Workmanship (voids or cracks in the material if at a member to be cut.)
2. Boost environment (structural failure resulting in cracks)

Effects on Subsystem/Mission:

Separation would not occur.

Backup Provisions:

None.

Inherent Preventives:

None.

Secondary Failure Mode:

Vehicle damage from flying fragments.

Possible Causes:

Inability to contain all fragments.

Effects on Subsystem/Mission:

Mission degradation to mission abort.

Table 4B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles	1
	Basic Failure Rate, Units: x 10^{-6} cycle	27
	Failure Rate Source Code	b
	Probability of Failure, x 10^{-6}	27
Secondary Failure Mode	Failure Classification	B
	Time/Cycles Units: Cycle 6	1
	Basic Failure Rate, Units: x 10^{-6}	3*
	Actual Failure Rate, Units:	3
	Failure Rate Source Code	b
	Probability of Failure x 10^{-6}	3
	Probability of Failure, All Modes	30
	Reliability	$0.9^{(4)}7$

Failure Rate Data Sources: b) ibid.

Notes:

A. Catastrophic failure, mission abort.

B. Non catastrophic failure, mission degradation.

* An estimate of 10% of total failure rate from farada (30) was used for this failure mode.

Table 5A

Item: Stud

System/Subsystem: Separation

Primary Failure Mode:

Hanging up in the bolt hole when the nuts have fired successfully.

Possible Causes:

1. High tipoff angle.
2. Cocked due to explosive force.
3. Stud end collared from explosive force

Effects on Subsystem/Mission:

Catastrophic, the spacecraft would not separate.

Backup Provisions:

None.

Inherent Preventives:

1. Ramp angle.
2. Oversized hole.
3. Control of tip off angle.
4. Some separation force is present and it is assumed that several studs would be required to hinge up before separation could not be accomplished.

Table 5B

	Mission
Failure Classification	A
Time/Cycles Units: Cycle	1
Basic Failure Rate, Units: x 10^{-6} cycle	5
Failure Rate Source Code	c
Probability of Failure, x 10^{-6}	5
Reliability	$0.9^{(5)}5$

Failure Rate Data Sources: c) Failure rate estimated based on a comparison to the explosive bolt.

Notes:

A. Catastrophic failure, mission abort.

Table 6

The total reliability of the explosive devices is determined as follows:

$$R_{bolt/nut} = R_{squib} \ R_{bolt/nut} = 0.9997 \cdot 0.99996$$
$$= 0.99966$$
$$R_{s/c} = R_{squib} \ R_{bolt/nut} = 0.9997 \cdot 0.99997$$
$$= 0.99967$$

The reliability of all six bolts fracturing as required was determined as follows:

$$R_6 = (R_1)^6 = (0.99966)^6 = 0.99796$$

The reliabilities of 4 and 3 bolts were determined using the same method except the exponent was 4 and 3.

The reliability of redundant items was determined as follows using the nuts as a sample:

$$(P + Q)^2 = 1 = P^2 + 2PQ + Q^2$$

where

$$Q = 1 - P$$

The problem is that at least one of the two nuts must function; therefore the first two terms are used. (The probability that both nuts will fire plus the probability that one will fire and one will fail.)

$$R = P^2 + 2PQ$$

Substituting 1 - Q for P the following equation is determined:

$$R = 1 - Q^2 = 1 - (0.00034)^2 = 1 - 0.0000001156$$
$$R = 0.9999998844 \text{ per attach point.}$$

4.6 Structure

4.6.1 Stress-Strength Approach

An estimate of the structural reliabilities may be obtained from the values used for the factor of safety and the margin of safety in the design of the spacecraft structure.

Each of the structural members in the Voyager is subjected to one or more stresses of varying magnitude during the mission. Of the stresses which are applied to an individual member, often one stress is predominant and when failure occurs, it is almost always due to this stress. This critical stress is used in the sizing of each member. The second factor which determines the reliability of a member is its strength or ability to withstand stress A part will fail only if the applied stress or stresses exceed its strength. The probability that this occurs is defined as the unreliability of the part.

Let X be the strength of the part and Y be the maximum stress placed on the part during the mission where X and Y are independent random variables Then reliability is defined as

$$R = P(X > Y).$$

Assuming that the probability densities of X and Y are reasonably approximated by independent normal distributions; i. e. , X is normal with mean μ_X and standard deviation σ_X, Y is normal with mean μ_Y and standard deviation σ_Y, we may let

$$D = X - Y$$

and write $R = P(X > Y) = P(D > 0)$ By the addition theorem for normal variables, D has a normal distribution with

$$\text{Mean of D} \equiv \mu_D = \mu_X - \mu_Y$$

$$\text{Standard Deviation of D} \equiv \sigma_D = \sqrt{\sigma_X^2 + \sigma_Y^2}.$$

Note that distributional assumptions of normality which may not be precisely satisfied for X and Y will tend to be more satisfied for D since by the central limit theorem, the difference of two variables will be more normal than either of the two.

Thus,

$$R = P(D > 0) = \int_0^{\infty} \frac{1}{\sigma_D \sqrt{2\pi}} \exp - \frac{1}{2} \left(\frac{D - \mu_D}{\sigma_D} \right)^2 dD$$

Letting

$$t = \frac{D - \mu_D}{\sigma_D}$$

$$dt = \frac{dD}{\sigma_D}$$

$$R = \int_{-\frac{\mu_D}{\sigma_D}}^{\infty} \frac{1}{\sqrt{2\pi}} \exp - \frac{t^2}{2} dt = \int_{-\infty}^{\frac{\mu_D}{\sigma_D}} \frac{1}{\sqrt{2\pi}} \exp - \frac{t^2}{2} dt$$

by symmetry of the integrand.

$$R = \Phi\left(\frac{\mu_D}{\sigma_D}\right) = \Phi\left(\frac{\mu_X - \mu_Y}{\sqrt{\sigma_X^2 + \sigma_Y^2}}\right) = \Phi(Z) \qquad (1)$$

where Φ is the cumulative distribution function of the standardized normal variable.

This procedure can be used as the basis for estimating the reliability of structural members. In stress analysis, it is customary to design in terms of a safety factor, S. F., margin of safety, M. S., and the stresses or loads. These are related by the equation

$$\text{S. F. (M. S. + 1)} = \frac{\text{Allowable Stress}}{\text{Limit Load}}$$

The allowables are determined based upon material properties found in military handbooks. Either 99% or 95% guaranteed values are used. Similarly, the limit load is usually taken as some value above the expected maximum stress.

Thus employing normal distribution notations:

$$S\ F. (M. S. + 1) = \frac{\mu_X - n\sigma_X}{\mu_Y + m\sigma_Y}$$

For the Voyager a safety factor of 1. 25 is specified. A margin of safety as close to zero on the positive side as possible is desired Substituting these values in the above, one has

$$\frac{\mu_X - n\sigma_X}{\mu_Y + m\sigma_Y} = 1.25,$$

or

$$\left(\frac{\mu_X}{\mu_Y}\right)\left(\frac{1 - n\left(\frac{\sigma_X}{\mu_X}\right)}{1 + m\left(\frac{\sigma_Y}{\mu_Y}\right)}\right) = 1.25 \qquad (2)$$

The quantity $\nu = \sigma/\mu$ is called the coefficient of variation and measures the spread of the distribution relative to the mean. If the four quantities, n, m, ν_X, ν_Y were known, they would be substituted and the ratio between the means determined from Equation (9).

Rewriting the quantity $Z = \frac{\mu_X - \mu_Y}{\sqrt{\sigma_X^2 + \sigma_Y^2}}$, one obtains

$$Z = \frac{\left(\frac{\mu_X}{\mu_Y}\right) - 1}{\sqrt{\left(\frac{\sigma_X}{\mu_X}\right)^2 \left(\frac{\mu_X}{\mu_Y}\right)^2 + \left(\frac{\sigma_Y}{\mu_Y}\right)^2}}$$

$$Z = \frac{\frac{\mu_X}{\mu_Y} - 1}{\sqrt{\nu_X^2 \left(\frac{\mu_X}{\mu_Y}\right)^2 + \nu_Y^2}}$$

Thus reliability can be determined as on equation (1)

$$R = \Phi(Z)$$

if n, m, ν_X, and ν_Y are known.

4.6.1.1 Estimation of n, m, ν_X, and ν_Y on Voyager Design

The allowable and limit stresses are not known with complete precision in any given situation. Materials vary from batch to batch, loads vary from vehicle to vehicle. Thus the quantities, allowable and limit stresses, can be considered as variables with probability distributions, again approximable by the normal. The needed quantities are estimated in the following paragraphs.

4.6.1.1.1 Estimation of n

For the Voyager, 99% guaranteed values were used in determining allowables.

Thus

$$\mu_X - n\sigma_X = \text{allowable stress}$$

is such that 99% of samples chosen will withstand the tabulated value of stress without breaking, cracking, or otherwise failing catastrophically in any way. As may be determined from a normal table, the number of standard deviations below the mean corresponding to 1 - 0.99 = 0.01 is 2.33. Thus n is estimated as 2.33.

4.6.1.1.2 Estimation of ν_X

Two tabulated values are given in the Handbook of Material Properties*, one 95% guaranteed, the other 99%. For the type of aluminum sheet which makes up the large majority of Voyager

$$70 - \mu_X = (-2.33)(4.412)$$

$$= -10.278$$

$$\mu_X = 80.278 \text{ ksi}$$

So that

$$\frac{\sigma_X}{\mu_X} = \nu_X = \frac{4.412}{80.278} = 0.055$$

4.6.1.1.3 Estimation oi M

The corresponding estimation of m and ν_Y is somewhat more subjective. Hence, conservative, best, and optimistic estimates for each are obtained and combined to arrive at an eventual range of values for the Voyager structure reliability. The range of values reflects the uncertainty associated with the load properties.

In the case of estimating m, it is difficult to assess the degree of conservatism exhibited in determination of the loads since these items were specified by JPL. An estimate is that actual loads would be less than those specified with a probability of 90%. A range of

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* MIL-HNBK-5, August 1962, Revised November 1, 1963; pp 3.2.70C

80% to 95% was assumed with 90% taken as a best estimate. In terms of m, these percentages correspond to

80%:	$m = 0.84$	Conservative
90%:	$m = 1.282$	Best Estimate
95%:	$m = 1.645$	Optimistic

4.6.1.1.4 Estimation of ν_Y

In order to obtain some feeling regarding possible load variability, the following question was asked of several Structures people (both TRW and Douglas), "Assuming that actual load data is available from previous pay loads which are similar enough to be meaningful to the Voyager structure, what do you estimate as the probability that actual limit loads will fall within ± 25% of conscientiously obtained best estimates of limit loads?" Note that the question was phrased not in terms of possible conservatism of JPL specified loads, but predicated upon the assumption that these loads were best estimates. Thus, answers reflect estimates of the uncertainty associated with load prediction with possible JPL conservatism being accounted for in the estimation of m.

Answers to this question ranged over a considerable spread although most felt that the past actual load information would greatly improve the accuracy of present load estimates. A conservative answer to the question is felt to be 80%, a best 90%, and an optimistic 95%. The computation of ν_Y from these values is done as follows:

Assuming an 80% probability that the actual loads will be within ± 25% of the predicted implies that 80% of the probability distribution will lie between $\mu_Y + .25\mu_Y$ and $\mu_Y - .25\mu_Y$. This implies, from the normal table, that

$$1.25\mu_Y = \mu_Y + 1.282\sigma_Y$$

$$0.25\mu_Y = 1.282\sigma_Y$$

$$0.195 = \frac{0.25}{1.282} = \frac{\sigma_Y}{\mu_Y} = \nu_Y \text{ conservative}$$

structural items, these two values are 73 ksi and 70 ksi, respectively Thus

$$P(X < 70) = .01$$

$$P(X < 73) = .05$$

or

$$P\left(\frac{X - \mu_X}{\sigma_X} < \frac{70 - \mu_X}{\sigma_X}\right) = .01$$

$$P\left(\frac{X - \mu_X}{\sigma_X} < \frac{73 - \mu_X}{\sigma_X}\right) = .05$$

Again, consulting a normal table shows that the .01 and .05 lower values are -2.33 and -1.65. Thus

$$\frac{70 - \mu_X}{\sigma_X} = -2.33$$

$$\frac{73 - \mu_X}{\sigma_X} = -1.65$$

Solving these equations for μ_X and σ_X:

$$70 - \mu_X = -2.33\sigma_X$$

$$73 - \mu_X = -1.65\sigma_X$$

$$-3 = -.68\sigma_X$$

$$4.412 \text{ ksi} = \sigma_X$$

or

$$0.75\mu = \mu_Y - 1.282\sigma_Y$$

$$1.282\sigma_Y = 0.25\mu_Y$$

Conservative

$$\nu_Y = \frac{\sigma_Y}{\mu_Y} = \frac{0.25}{1.282} = 0.195$$

Similarly, the best estimate of ν_Y is given by

$$\text{Best } \nu_Y = 0.25/1.645 = 0.152$$

And the optimistic by

$$\text{Optimistic } \nu_Y = 0.25/1.96 = 0.128$$

4.6.1.2 Computation of Reliability of One Structural Member

Since all structural members follow the same general design guidelines, it is a good first approximation to assume that all have equal reliability. On the basis of values for n, m, ν_X, ν_Y, the reliability is then obtained from

$$R = \Phi(Z)$$

by consulting a normal table The values obtained by this process are summarized in Table 1 for various sets of values of n, m, ν_X, ν_Y. As may be seen from the values of R in Table 82, a range of from 0.9^3041 to 0.9^5796 has been obtained depending upon the degree of conservatism assumed for m and ν_Y.

4.6.1.2.1 Estimation of σ_R

In an effort to describe the variation of estimates of R statistically, assuming that the estimates have a normal distribution would

Table 1.

	m	n	μ_X	μ_Y	
1	0.84	2.33	0.055	0.195	
2	0.84	2.33	0.055	0.152	
3	1.282	2.33	0.055	0.195	
4	1.282	2.33	0.055	0.152	
5	1.282	2.33	0.055	0.128	← Best Estimate
6	1.645	2.33	0.055	0.152	
7	1.645	2.33	0.055	0.128	

Input Set	μ_X/μ_Y	Z	R	$\mathcal{R}$	
1	1.6686	3.1026	0.9^3041	6.94866	
2	1.6168	3.5026	0.9^3770	8.37718	
3	1.7921	3.6253	0.9^3856	8.84541	
4	1.7131	3.9882	0.9^4667	10.30992	
5	1.6690	4.2473	0.9^4892	11.43595	← Best Estimate
6	1.7922	4.3729	0.9^5387	12.00232	
7	1.7356	4.6070	0.9^5796	13.10256	

be unrealistic since R can vary only between 0 and 1. However, the transformed variable,

$$\mathcal{R} = \ln\left(\frac{R}{1 - R}\right)$$

has a range from $-\infty$ to $+\infty$ as R ranges from 0 to 1. Corresponding values of $\mathcal{R}$ are given in the final column of Table 1. These values are plotted in Figure 1.

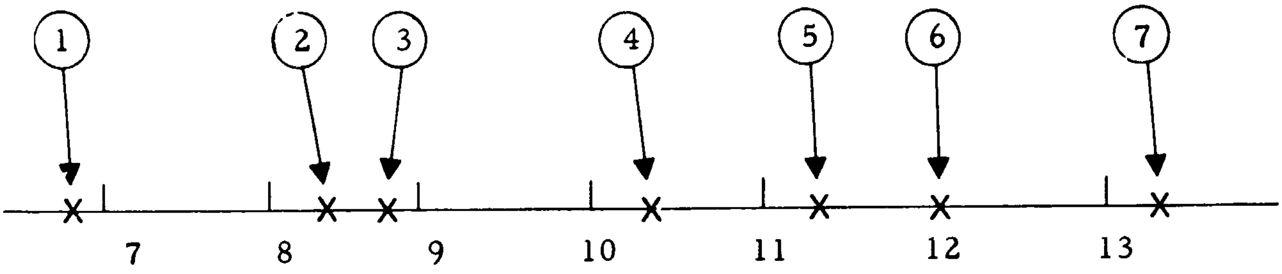


Figure 1. $\mathcal{R}$ Values

Since the range of values for m and ν_Y may be thought of as roughly about a $\pm 2\sigma$ to $\pm 3\sigma$ range, one would expect values of R derived from combining one conservative value with one best value to lie roughly at -2σ to -3σ in the R or $\mathcal{R}$ distributions. Similarly, one optimistic estimate combined with a best estimate would place the resulting $\mathcal{R}$ value at about $+2\sigma$ to $+3\sigma$ in the $\mathcal{R}$ distribution. Combining two conservative estimates yields a result at about -5σ and two optimistic estimates yield a result at about $+5\sigma$. Assuming the mean $\mathcal{R}$ to lie near the result obtained by combining two best estimates, one may obtain a set of $\sigma_{\mathcal{R}}$ estimates from the values of $\mathcal{R}$ obtained. Thus,

$$\sigma_{\mathcal{R}1} = (10.30992 - 6.94866)/5 = 0.672$$

$$\sigma_{\mathcal{R}2} = (10.30992 - 8.37718)/3 = 0.644$$

$$\sigma_{\mathcal{R}3} = (10.30992 - 8.84541)/2 = 0.732$$

$$\sigma_{\mathcal{R}5} = (11.43595 - 10.30992)/2 = 0.563$$

$$\sigma_{\mathcal{R}6} = (12.00232 - 10.30922)/3 = 0.564$$

$$\sigma_{\mathcal{R}7} = (13.10256 - 10.30992)/5 = 0.559$$

These estimates are fairly homogenous. Averaging them obtains an estimate for $\sigma_{\mathcal{R}}$ of

$$\sigma_{\mathcal{R}} = 0.622$$

Thus the $\mathcal{R}$ distribution is assumed normal with

$$\mu_{\mathcal{R}} = 10.30992$$

$$\sigma_{\mathcal{R}} = 0.622$$

:. 6. 2 Estimation of R_S and σ_{R_S}

Making the conservative assumption that Voyager mission failure will result if any structural member fails, the structure system reliability may be computed as

$$R_S = R^k$$

where k is the number of independent structural members. At this preliminary stage of the design only a rough estimate of k could be obtained, a value of k = 80. Thus, using the best estimate for R of $.9^4667$,

$$\hat{R}_S \equiv \mu_{R_S} = (.9^4667)^{80} = .997339$$

In order to estimate σ_{R_S}, one may expand R_S in a series

$$R_S = R^{80} = \mu_{R_S} + \left(\frac{dR_S}{d\mathcal{R}}\right)_{\mathcal{R}=\mu_{\mathcal{R}}} (\mathcal{R} - \mu_{\mathcal{R}})$$

neglecting higher order terms. Now $\sigma^2_{R_S}$ may be found as

$$\sigma^2_{R_S} = E\left(R_S - \mu_{R_S}\right)^2 = \left(\frac{dR_S}{d\mathcal{R}}\right)^2_{\mathcal{R}=\mu_{\mathcal{R}}} \sigma^2_{\mathcal{R}}$$

or

$$\sigma_{R_S} = \left(\frac{dR_S}{d\mathcal{R}}\right)_{\mathcal{R}=\mu_{\mathcal{R}}} \sigma_{\mathcal{R}}$$

By definition,

$$\mathcal{R} = \ln\left(\frac{R}{1 - R}\right)$$

Solving for R,

$$e^{\mathcal{R}} = \frac{R}{1 - R}$$

$$(1 - R)e^{\mathcal{R}} = R$$

$$e^{\mathcal{R}} - Re^{\mathcal{R}} = R$$

$$e^{\mathcal{R}} = R(1 + e^{\mathcal{R}})$$

$$R = \frac{e^{\mathcal{R}}}{1 + e^{\mathcal{R}}} = \frac{1}{1 + e^{-\mathcal{R}}} = (1 + e^{-\mathcal{R}})^{-1}$$

Thus,

$$R_S = R^{80} = (1 + e^{-\mathcal{R}})^{-80}$$

$$\frac{dR_S}{d\mathcal{R}} = +80(1 + e^{-\mathcal{R}})^{-81}\ e^{-\mathcal{R}}$$

$$\frac{dR_S}{d\mathcal{R}} = \frac{80\ e^{-\mathcal{R}}}{(1 + e^{-\mathcal{R}})^{80}\ (1 - e^{-\mathcal{R}})} = 80\mu_{R_S}\ (1 - R)$$

Substituting $\mathcal{R} = \mu_{\mathcal{R}} = 10.30992$,

$$\left(\frac{dR_S}{d\mathcal{R}}\right)_{\mathcal{R} = \mu_{\mathcal{R}}} = 0.002657$$

Putting this value along with $\sigma_{\mathcal{R}} = 0.622$, one obtains,

$$\sigma_{R_S} = (0.002657)(0.622)$$

$$\sigma_{R_S} = 0.001653$$

Thus, the system reliability estimate is found to have a mean $\mu_{R_S} = 0.997339$ and a standard deviation $\mu_{R_S} = 0.001653$.

4.6.3 Deviation of Range of Values for R_S

For the same reasons discussed in Section 4.6.1.2.1, a normality assumption on R_S would not be realistic. Once again define

$$\mathcal{R}_S = \ln\left(\frac{R_S}{1 - R_S}\right) \tag{3}$$

and expand this expression:

$$\mathcal{R}_S = \mathcal{R}_S + \left(\frac{d\mathcal{R}_S}{dR_S}\right)_{R_S=\mu_{R_S}} \left(R_S - \mu_{R_S}\right)$$

Again the standard deviation of $\mathcal{R}_S$ may be found as

$$\sigma_{\mathcal{R}S} = \sqrt{E\left(\mathcal{R}_S - \mu_{\mathcal{R}S}\right)^2} = \left(\frac{d\mathcal{R}_S}{dR_S}\right)_{R_S=\mu_{R_S}} \sigma_{R_S} \tag{4}$$

Now,

$$\frac{d\mathcal{R}_S}{dR_S} = \frac{1 - R_S}{R_S} \quad \frac{1 - R_S + R_S}{\left(1 - R_S\right)^2} = \frac{1}{R_S(1 - R_S)}$$

Substituting $R_S = \mu_{R_S} = 0.997339$

$$\left(\frac{d\mathcal{R}_S}{dR_S}\right)_{R_S=\mu_{R_S}} = \frac{1}{(0.997339)(0.002661)} = \frac{1}{0.0026539} = 376.8$$

Putting these values in (4) along with $\sigma_{R_S} = 0.001653$.

$$\sigma_{\mathcal{R}_S} = (376.80)(0.001653)$$

$$= 0.62285$$

Substituting $\mu_{R_S} = 0.997339$ in (3) one obtains as

$$\mu_{\mathcal{R}_S} = \ln\left(\frac{0.997339}{0.002661}\right) = \ln(374.798)$$

$$= 5.92639$$

Since normality may be assumed for $\mathcal{R}_S$, a $\pm 3\sigma$ confidence interval may be formed as

$$\mathcal{R}_{S,\ \text{lower limit}} = \mu_{\mathcal{R}_S} - 3\sigma_{\mathcal{R}_S}$$

$$\mathcal{R}_{S,\ \text{upper limit}} = \mu_{\mathcal{R}_S} + 3\sigma_{\mathcal{R}_S}$$

$$\mathcal{R}_{S,\ \text{lower limit}} = 5.92639 - 3(0.62285)$$
$$= 5.92639 - 1.86855$$
$$= 4.05784$$

$$\mathcal{R}_{S,\ \text{upper limit}} = 5.92639 + 1.86855$$
$$= 7.79494$$

Now the transformation from Z_S to R_S is one-to-one and preserves probability, i.e.

$$P(\mathcal{R}_S \leq \mathcal{R}_{S_0}) = P(h(\mathcal{R}_S) \leq h(\mathcal{R}_{S_0}))$$

where

$$h(\mathcal{R}) = (1 + e^{-\mathcal{R}})^{-1}$$

Consequently, a $\pm 3\sigma$ confidence interval for R_S may be obtained by using the values of R_S which correspond to $\mathcal{R}_{S,\ \text{lower limit}}$ and and $\mathcal{R}_{S,\ \text{upper limit}}$.

These are found to be

$$R_{S,\ lower\ limit} = (1 + e^{-4.05784})^{-1} = (1 + 0.017286)^{-1}$$

$$= 0.983008$$

$$R_{S,\ upper\ limit} = (1 + e^{-7.79494})^{-1} = (1 + 0.0004118)^{-1}$$

$$= 0.999588$$

4.7 Solar Panel Deployment

A reliability analysis has been conducted on four configurations under consideration for use as the method of deploying the ten solar panels of the Voyager Spacecraft to determine a reliability prediction for each of the configurations such that trade-offs could be made with other parameters and a system selected.

The configurations considered in the analysis are shown on the attached Douglas Drawings (SK-BM00-9001 through SK-BM00-9004). Table 1 presents a summary of the preliminary reliability predictions for each of the systems considered. Predictions are given for the use of a single squib as well as redundant squibs.

The table shows that the pin puller and swivel catch system has the highest prediction. The reliability predictions of all systems become limited by the damper/actuator.

4.7.1 Pin Puller and Spring Hinge System (SK-BM00-9001)

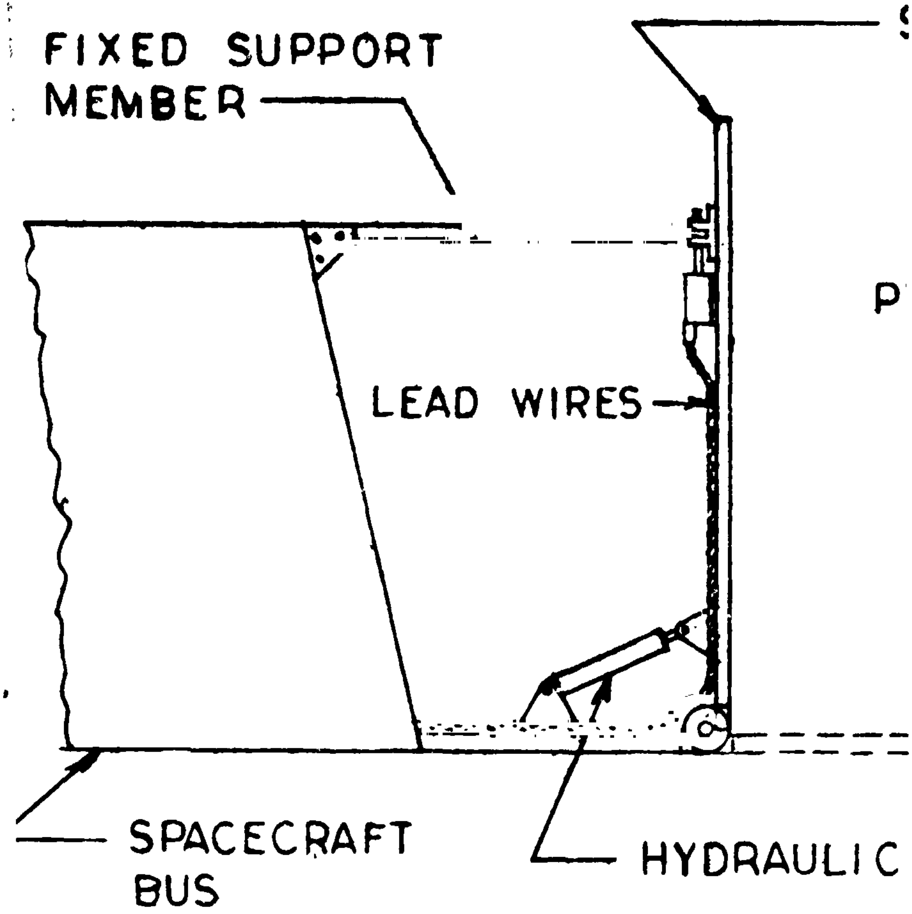
This configuration consists of a pin puller which holds the panel in the stowed position and releases it on command, two springs to deploy and lock the panel in place and a hydraulic damper to prevent structural failure of the panel during deployment. The total solar array system consists of 10 panels, each with identical deployment hardware, and all 10 panels are required for successful operation.

Table 2 presents the preliminary reliability predictions for the total solar panel deployment (10 panels). It is noted from the table that the reliability is improved by the use of redundant squibs,

EVALUATION OF SOLAR PANEL DEPLOYMENT SYSTEMS

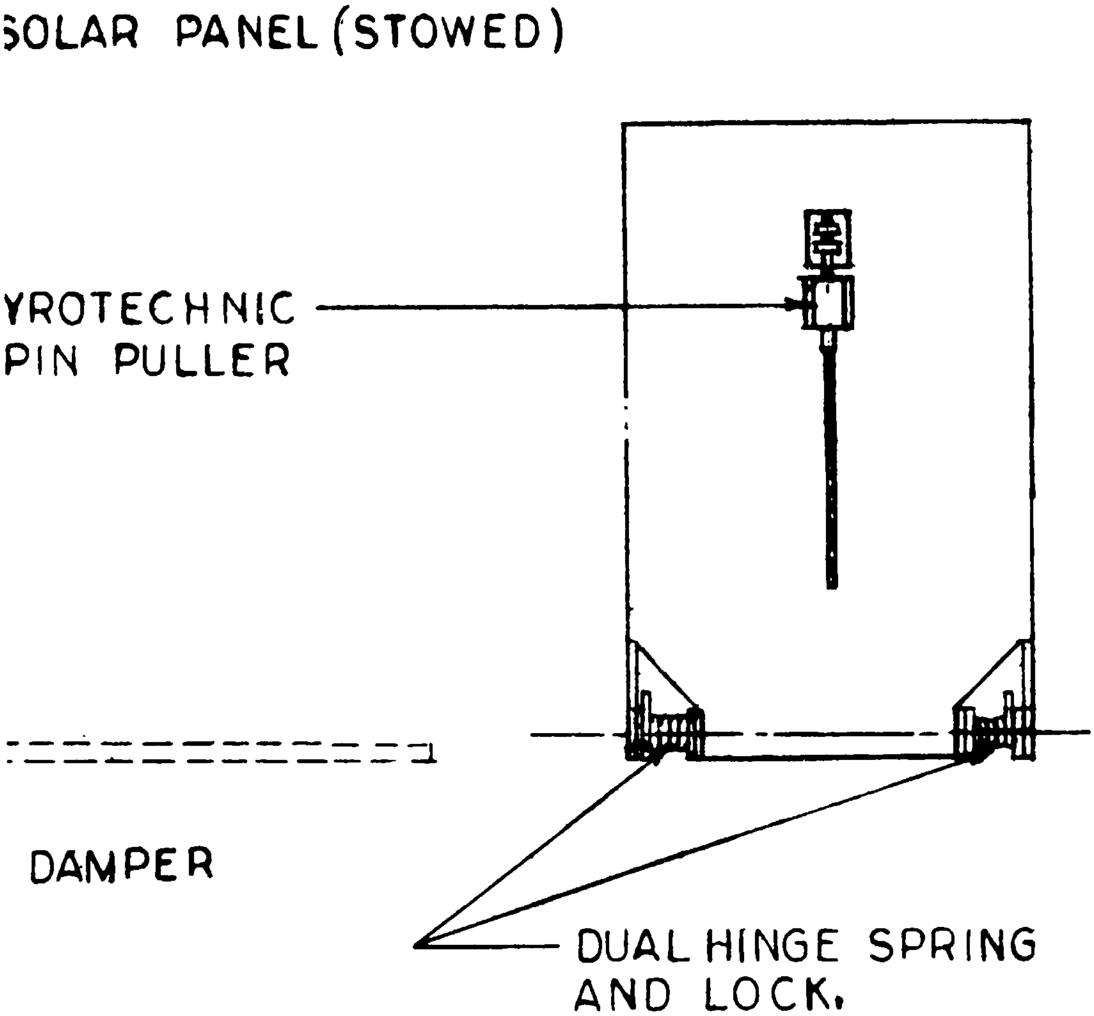
1. WEIGHT . . . . ______
2. COST . . . . ______
3. RELIABILITY. . . . . ______
4. SIMPLICITY . . . . ______
5. EXTERNAL POWER REQUIRED. ______
6. DEVELOPMENT REQUIRED . . ______
7. AVAILABILITY . . . . ______
8. HISTORY OF USAGE . . . . ______
9. FAIL SAFE. . . . . ______

TOTAL. . . . . ______



PIN PULLER AND SPF

172 (2)



RING HINGE SYSTEM

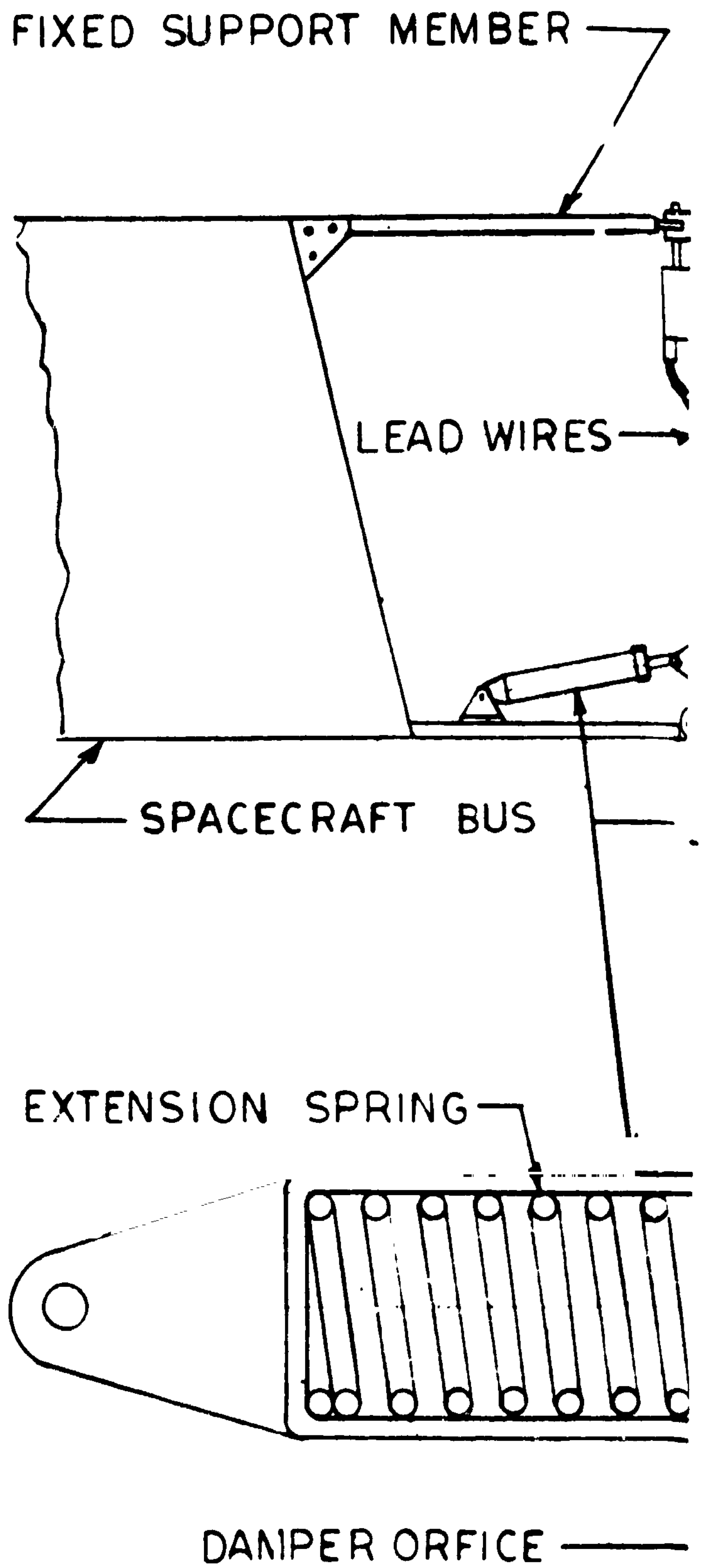
172B

W. CARR DAC
SK-BM00 9001

EVALUATION OF SOLAR PANEL DEPLOYMENT SYSTEMS

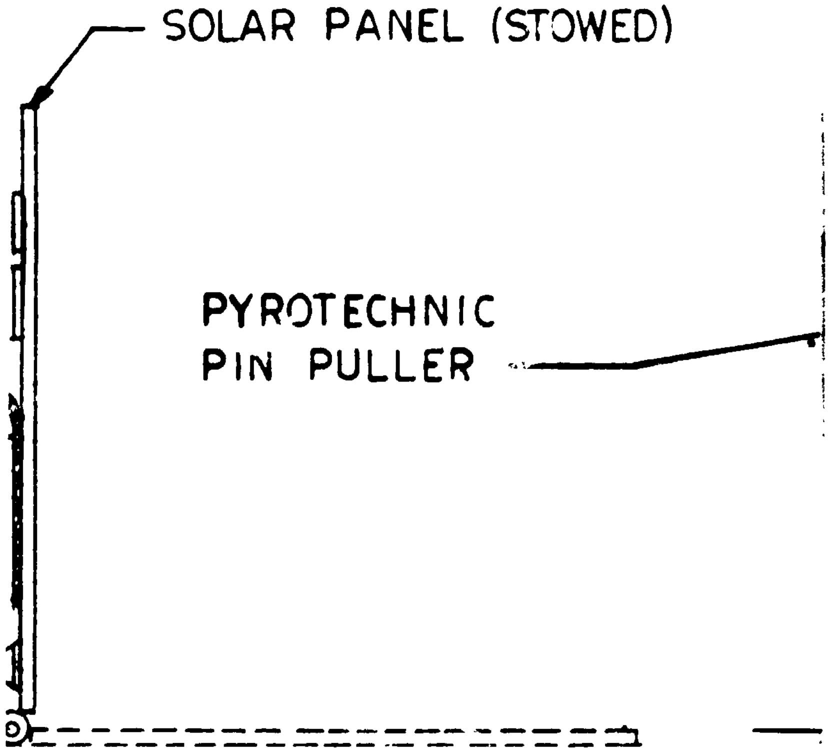
1. WEIGHT . . . . ______
2. COST . . . . ______
3. RELIABILITY. . . . . ______
4. SIMPLICITY . . . . ______
5. EXTERNAL POWER REQUIRED. ______
6. DEVELOPMENT REQUIRED . . ______
7. AVAILABILITY . . . . ______
8. HISTORY OF USAGE . . . . ______
9. FAIL SAFE. . . . . ______

TOTAL. . . . . ______

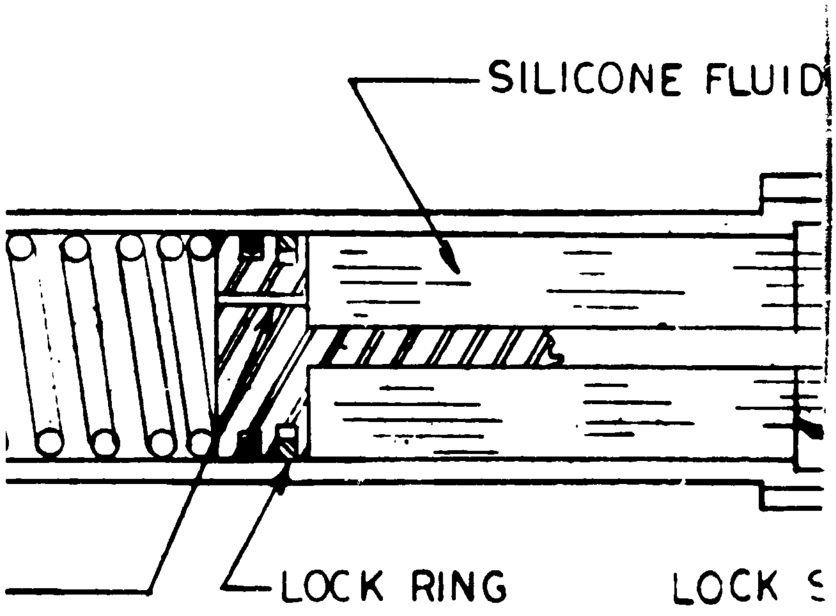


SPRING ACTUATOF

1730

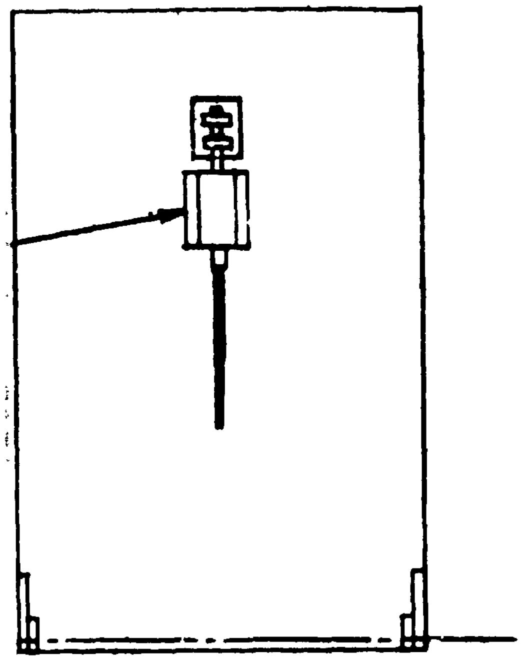


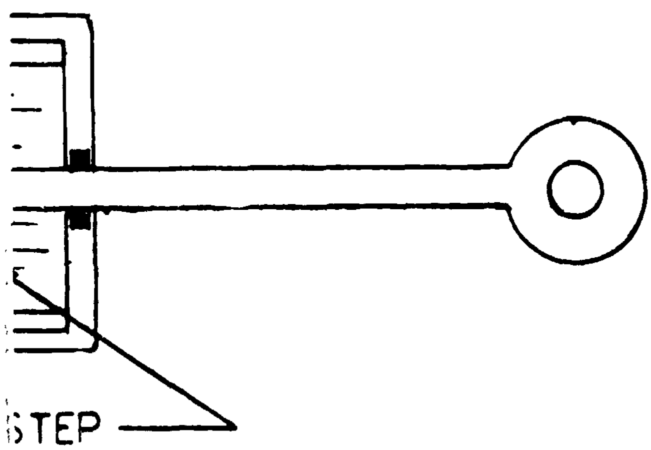
LINEAR ACTUATOR,
DAMPER AND EXTEND
LOCK UNIT.



R AND PIN PULLER

173(2)





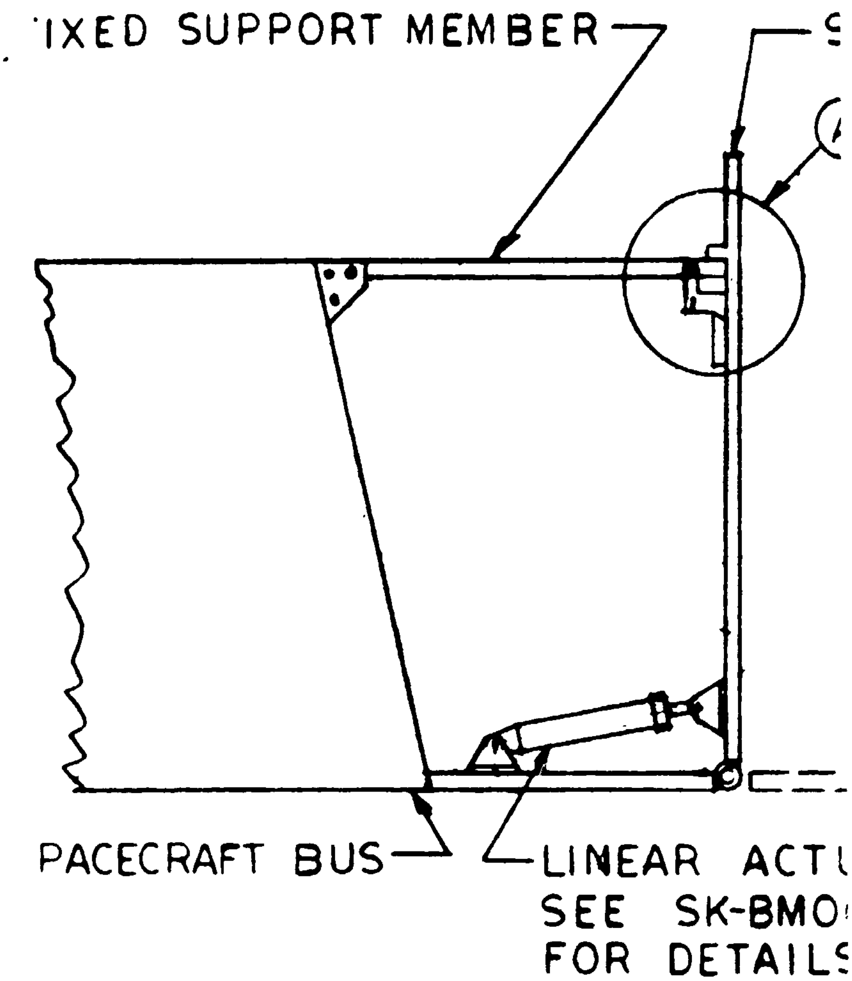
W CARR DAC

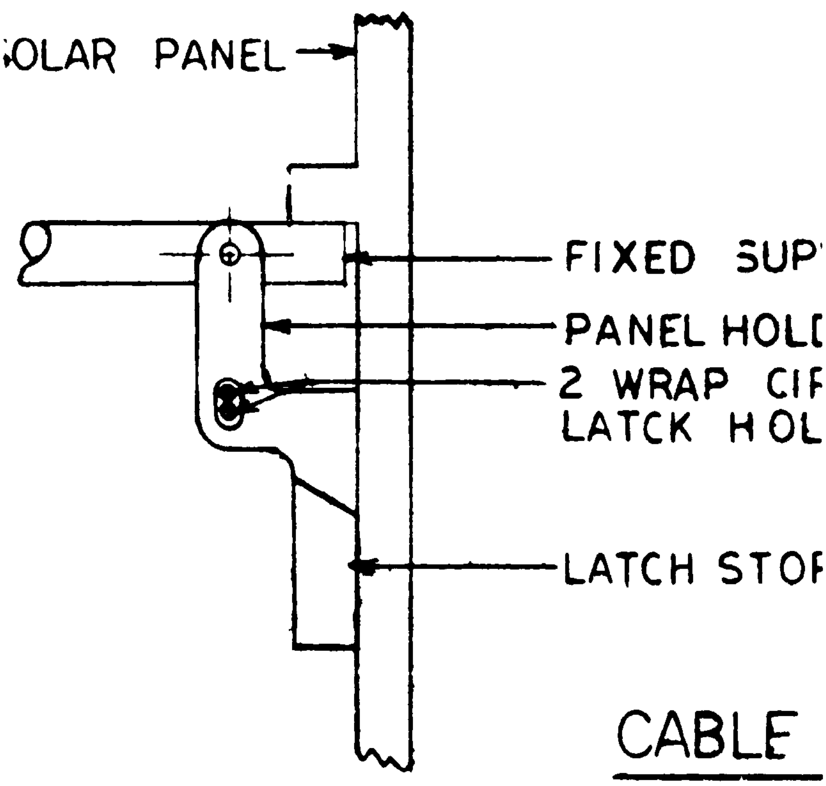
SK-BM00 9002

EVALUATION OF SOLAR PANEL DEPLOYMENT SYSTEMS

1. WEIGHT . . . . ______
2. COST . . . . ______
3. RELIABILITY. . . . . ______
4. SIMPLICITY . . . . ______
5. EXTERNAL POWER REQUIRED. ______
6. DEVELOPMENT REQUIRED . . ______
7. AVAILABILITY . . . . . . ______
8. HISTORY OF USAGE . . . . ______
9. FAIL SAFE. . . . . ______

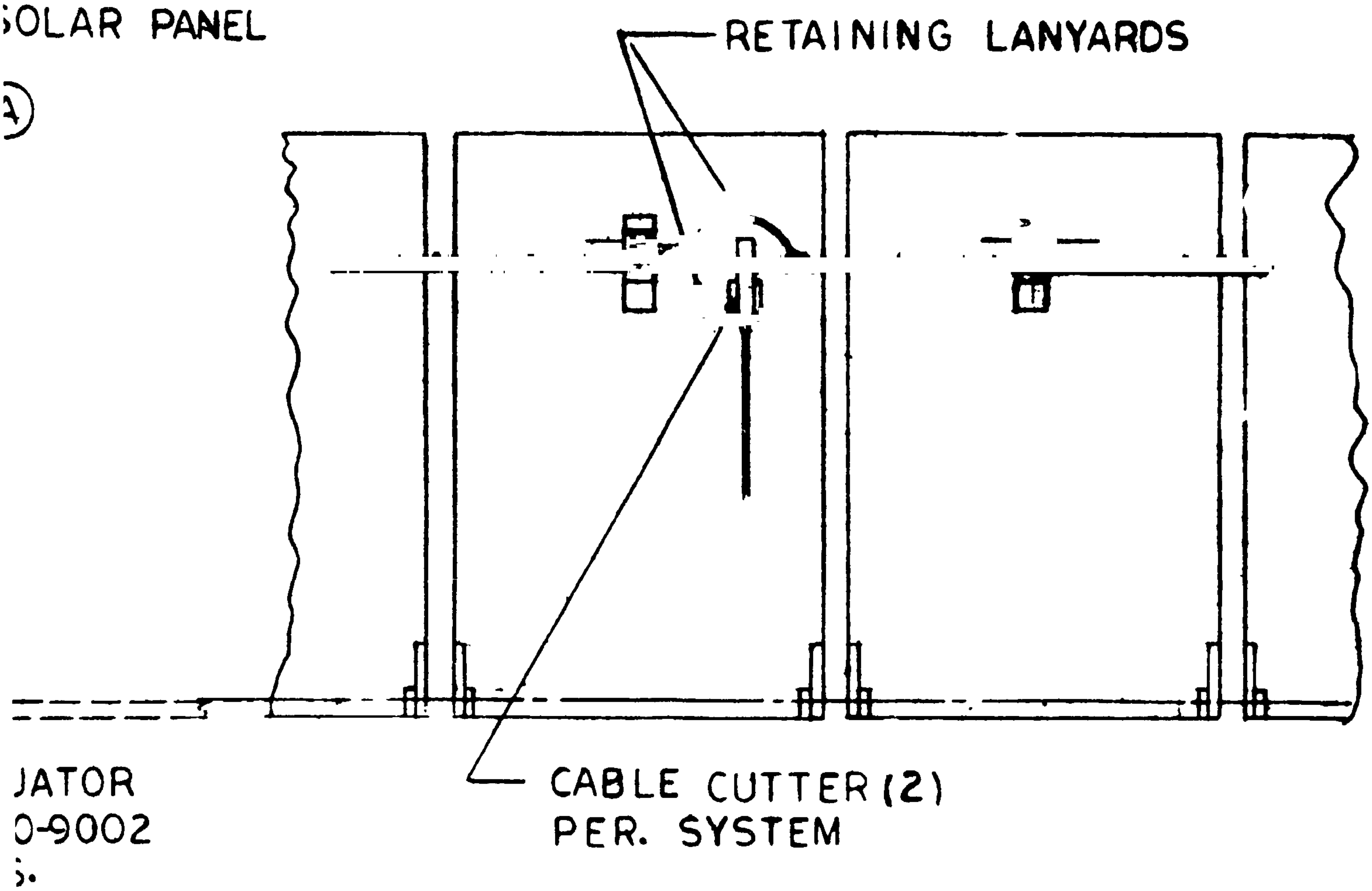
TOTAL. . . . . ______





CABLE

DETAIL 'A'



PORT MEMBER

DING LATCH
RCUMVENTIONAL
DING COR D.

R

CUTTER SYSTEM

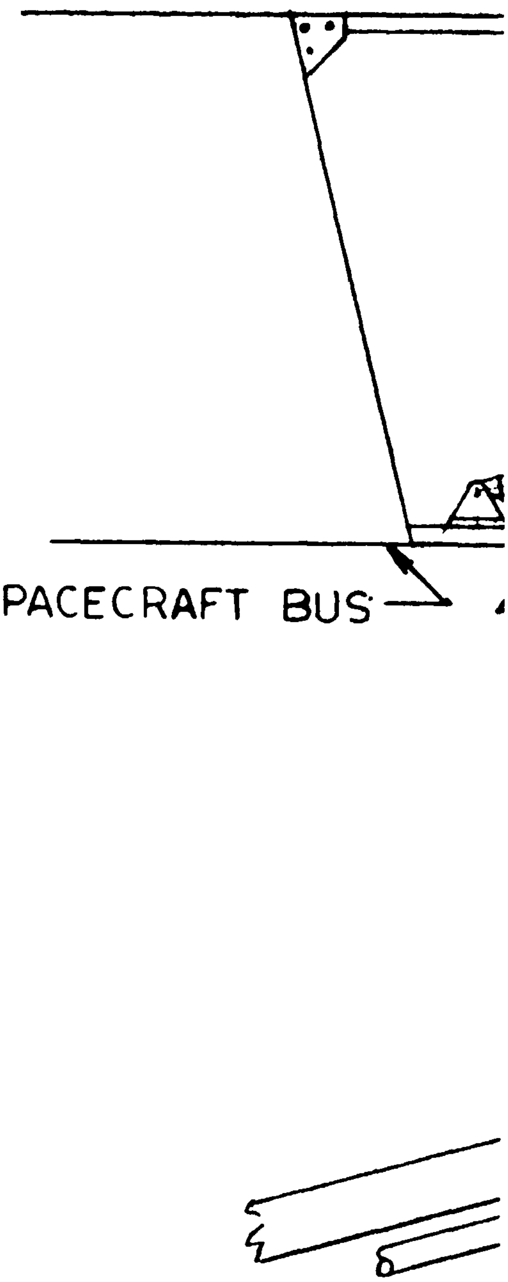
W. CARR DAC
SK-BM00-9003

EVALUATION OF SOLAR PANEL DEPLOYMENT SYSTEMS

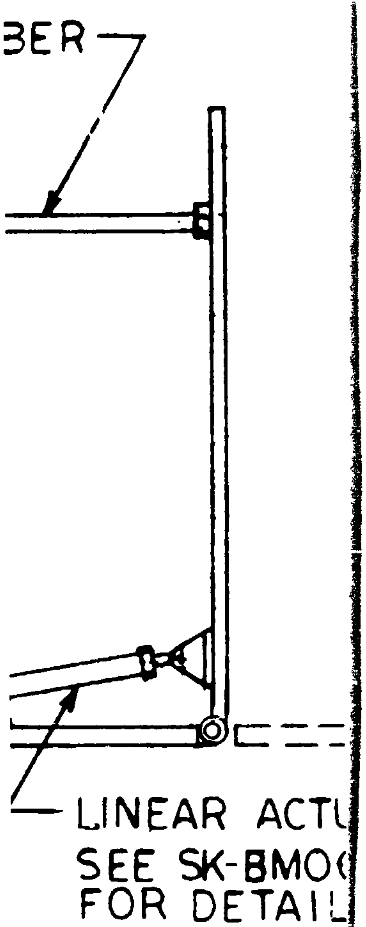
1. WEIGHT . . . . ______
2. COST . . . . ______
3. RELIABILITY. . . . . ______
4. SIMPLICITY . . . . ______
5. EXTERNAL POWER REQUIRED. ______
6. DEVELOPMENT REQUIRED . . ______
7. AVAILABILITY . . . . ______
8. HISTORY OF USAGE . . . . ______
9. FAIL SAFE. . . . . ______

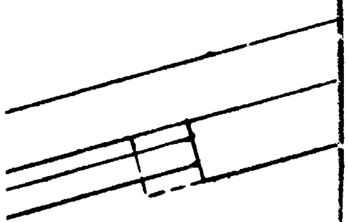
TOTAL. . . . . ______

FIXED SUPPORT MEM



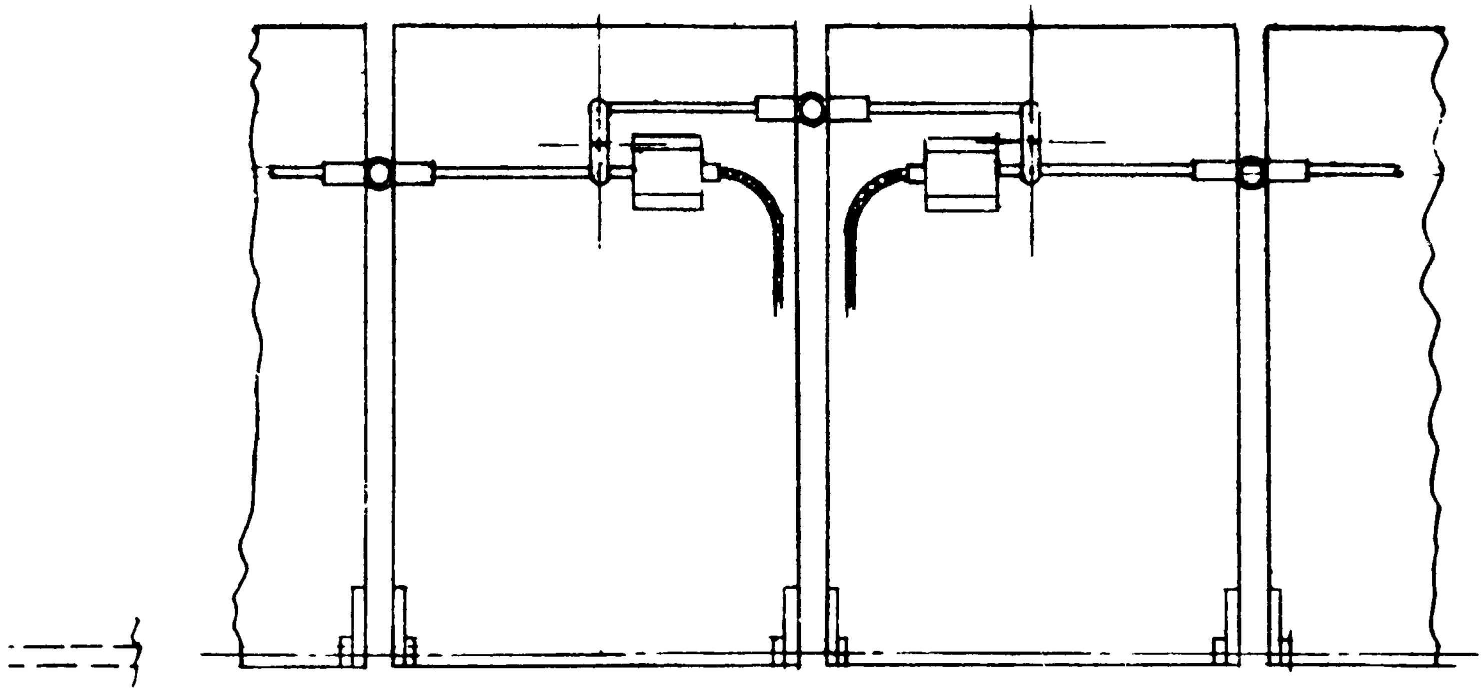
PIN F



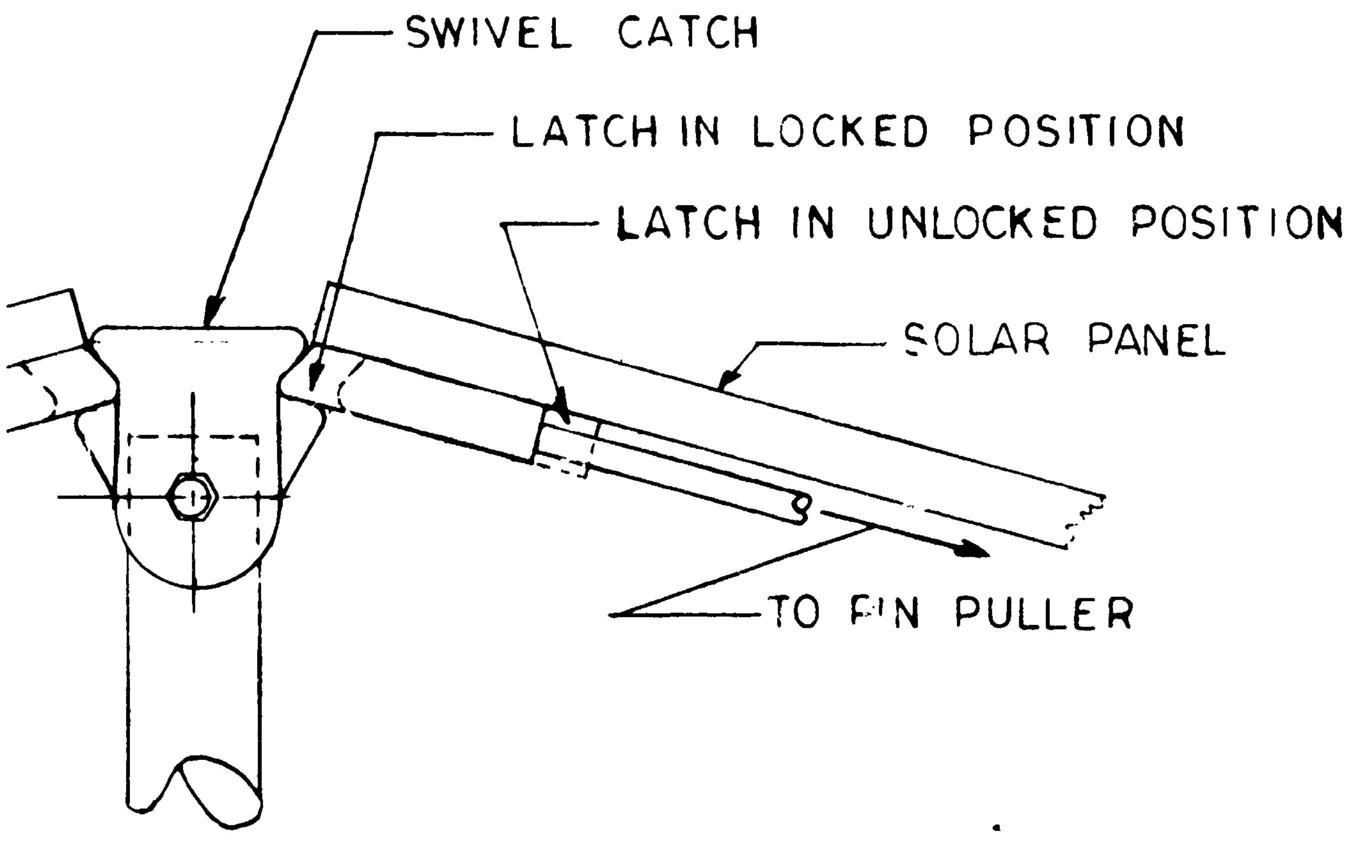


PULLER A

175(2)



JATOR
)-9002



ND SWIVEL CATCH

W. CARR DAC
SK-BM00-9004

Table 1. Preliminary Reliability Prediction for Solar Panel Deployment Systems — (Total of 10 Panels)

Configuration	Reliability Prediction
1. Pin Puller and Spring Hinge System SK-BM00-9001	
a. Single Squib	0.99291
b. Redundant Squibs	0.99724
2. Spring Actuator and Pin Puller System SK-BM00-9002	
a. Single Squib	0.99314
b. Redundant Squibs	0.99747
3. Cable Cutter System SK-BM00-9003	
a. Single Cable Cutter	0.99757
b. Redundant Cable Cutters	0.99804
4. Pin Puller and Swivel Catch System SK-BM00-9004	
a. Single Squib	0.99804
b. Redundant Squibs	0.99806

but does not reach the 0.999 level. This is due to the fact that next to the squibs, the damper is the most unreliable portion of the system; therefore, it becomes the limiting factor for reliability improvement when the squibs are improved.

The primary failure mode of the damper is loss of fluid (See Tables 2 and 3), which could allow the panel to fail structurally if this loss is great. Since the amount of fluid loss acceptable has not been determined, the reliability prediction assumes any loss to be a failure; therefore, the prediction for the damper is no doubt conservative. Since all configurations use a damper, the figures are valid for comparison.

Table 2. Preliminary Reliability Prediction for the Pin Puller and Spring Hinge System (10 Panels) (SK-BM00-9001)

Configuration	Reliability Prediction
1. Single Squib	0.99291
2. Redundant Squibs	0.99724

4.7.2 Spring Actuator and Pin Puller System (SK-BM00-9002)

This configuration consists of a pin puller, as in Section 4.7.1, but the extension spring, damper and locking device are all contained in the single spring actuator. The comments in Section 4.7.1 are applicable here. Table 3 presents the preliminary reliability predictions for this configuration. Predictions are given for the pin pullers using both a single squib and redundant squibs.

Table 3. Preliminary Reliability Prediction for the Spring Actuator and Pin Puller System (10 Panels) (SK-BM00-9002)

Configuration	Reliability Prediction
1. Single Squib	0.99314
2. Redundant Squibs	0.99747

4.7.3 Cable Cutter System (SK-BM00-9003)

The cable (cord) cutter system consists of the linear actuator, as in Section 4.7.2; however, the panels are held in the stowed position by a latch which, in turn, is held by a double wrap of cord. A cord cutter is used to cut the cord which releases the latch and allows the actuators to extend the panels.

The preliminary reliability predictions for the system are shown in Table 4 for both a single cord cutter and redundant cord cutters. (In order for the panels to release, only one cord must be cut since it is a double wrap). The table shows a higher reliability than the configurations discussed previously since a cord cutter is not required at each of the 10 panels. The effect of the actuator is shown in the redundant cable cutter prediction if compared to the previous sections. The prediction for the actuator is 0.99807 as opposed to the system prediction of 0.99804.

Table 4. Preliminary Reliability Prediction for the Cable Cutter System (10 Panels) (SK-BM00-9003)

Configuration	Reliability Prediction
1. Single Cable Cutter	0.99757
2. Redundant Cable Cutters	0.99804

4.7.4 Pin Puller and Swivel Catch (SK-BM00-9004)

This configuration, although similar to that presented in Section 4.7.2, allows, through the swivel catch, failures of a portion of the pin pullers and the deployment will be successful. The firing of one pin puller will release the two attach points on that panel and one attach point on each adjacent panel; therefore, every other pin puller could fail and the deployment would be successful.

Table 5 presents the reliability predictions for the solar panel system for two configurations, one using a single squib per pin puller and the other using redundant squibs. The table shows that this system has the highest reliability prediction of all four considered; however, it is not significantly higher than the system presented in Section 4.7.3.

Table 5. Preliminary Reliability Prediction of the Pin Puller and Swivel Catch System (10 Panels) (SK-BM00-9004)

Configuration	Reliability Prediction
1. Single Squib	0.99804
2. Redundant Squibs	0.99806

4.7.5 Reliability Analysis

Tables 6 through 12 present the reliability analysis necessary to determine the reliability predictions presented in the previous sections. The tables contain a failure mode and effect analysis and the failure rate and source used for the major parts/components used in each configuration considered. By reducing each component to its major failure modes and determining the failure rate of those parts contributing to these modes (See Tables 9A and 9B), a more realistic prediction is obtained than if the failure rate of the component were used (e.g. damper/actuator).

In the configurations presented in SK-BM00-9003 and 9004, several items such as cord, latch, swivel, etc., are used; however, no analysis sheets are presented. This is because each item by itself has a failure rate less than 1×10^{-6}; however, they were considered in the calculations because of the number of items (cord, latch, etc.,) and the fact that there are 10 panels in the system. Several of these items would lend themselves to a reliability analysis based upon a stress analysis.

Table 6A

Item: Squib

System/Subsystem: Solar Panel Deployment

Item Type/Description:

Explosive bridgewire with powder charge. (Redundant bridgewires)

Operational Notes:

One shot device.

Environmental Notes:

Failure is most likely to result from vibration during the boost phase.

Primary Failure Mode:

Failure to fire with proper force, when proper electrical input is present.

Possible Causes:

1. Poor workmanship (bridgewire coating).
2. Bad lot of charges.
3. Broken bridgewires.

Effects on Subsystem/Mission:

Catastrophic — Failure to deploy.

Backup Provisions:

Redundant squibs.

Inherent Preventives:

Lot qualification of squibs.

Secondary Failure Mode:

Premature firing.

Possible Causes:

1. Stray R.F. signals.
2. Boost environment.
3. Bad lot of charges.

Effects on Subsystem/Mission:

Catastrophic - early deployment.

Backup Provisions:

None.

Inherent Preventives:

Bruceton tests, shielding from R.F. signals.

Table 6B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles x 10^{-6}	1
	Basic Failure Rate, Units: per 10^{-6} Firings	294
	Environment/Application Factor	-
	Actual Failure Rate, Units: per 10^{-6} Firings	294
	Failure Rate Source Code	a
	Probability of Failure, x 10^{-6}	294
Secondary Failure Mode	Failure Classification	A
	Time/Cycles Units: Cycles x 10^{-6}	1
	Basic Failure Rate, Units: Per 10^{-6} Firings	6*
	Environment/Application Factor	-
	Actual Failure Rate, Units: Per 10^{-6} Firings	6
	Failure Rate Source Code	a
	Probability of Failure x 10^{-6}	6
	Probability of Failure, All Modes	300
	Reliability	$0.9^{(3)}700$

Failure Rate Data Sources: a) TRW experience - 2000 firings with no failures. Prob. of failure 3 x 10^{-4} at 50% C.L. Telecon with Farada indicated 47,332 firings with 10 failures. Prob. of failure 2.11 x 10^{-4} (good agreement).

Notes:

A catastrophic

* Experience from Bruceton type testing indicates premature failure mode is approx. 1/50 of total failure rate probability.

Table 7A

Item: Pin Puller

System/Subsystem: Solar Panel Deployment

Item Type/Description:

Captive with dual cartridge capability.

Operational Notes:

Gas retaining, no fragmentation.

Primary Failure Mode:

Failure to release pin when proper force is present.

Possible Causes:

1. Failure of pin to move (jammed).
2. Broken pin.

Effects on Subsystem/Mission:

Catastrophic - deployment would not occur.

Backup Provisions:

None.

Inherent Preventives:

Lot qualification of pin pullers.

Table 7B

	Mission
Failure Classification	A
Time/Cycles Units: x 10^{-6} Cycles	1
Basic Failure Rate, Units: Per 10^6 Firings	48
Environment/Application Factor	-
Actual Failure Rate, Units: Per 10^6 Firings	48
Failure Rate Source Code	b
Probability of Failure, x 10^{-6}	48
Reliability	$0.9^{(4)}52$

Failure Rate Data Sources: b) Farada, revised 3/1/65, Pg. 2.279, Source 138, Explosive bolt used - estimate for locking etc - 10% of failure rate.

Notes:

A ibid.

Table 8A

Item: Cable Assy (Squib Firing Circuit)

Item Type/Description:

A 2 pin connector, 2 solder joints and two wires were considered for each cable assy.

Environmental Notes:

Failure could occur as a result of vibration during boost.

Primary Failure Mode:

Failure to provide electrical signal to squib.

Possible Causes:

1. Poor pin contact.
2. Broken wire or connection.

Effects on Subsystem/Mission:

Catastrophic - no deployment.

Backup Provisions:

Redundant squibs; therefore cable assemblies are considered redundant.

Inherent Preventives:

Circuit continuity check.

Table 8B

	Mission
Failure Classification	A
Time/Cycles Units: Hrs.	0.3†
Basic Failure Rate, Units: $\times 10^{-6}$ Hrs.	0.438*
Environment/Application Factor	1000
Actual Failure Rate, Units: $\times 10^{-6}$ Hrs.	438
Failure Rate Source Code	c
Probability of Failure, $\times 10^{-6}$	145
Reliability	$0.9^{(3)}855$

Failure Rate Data Sources: c) Farada, Revised 3/1/65, Source 138.

Notes:

A ibid.

* Total failure rate determined as follows:

1.	Connector	- 0.2/Pin x 2	=	0.400
2.	Wire	- 0.015 x 2	=	0.030
3.	Solder Joint	- 0.004 x 2	=	0.008
				0.438

† Booster duration.

Table 9A

Item: Hydraulic Damper

System/Subsystem: Solar Panel Deployment

Operational Notes:

Used in SK-BM00-9001.

Environmental Notes:

Failure could occur during boost environment.

Primary Failure Mode:

Leakage of fluid past seals.

Possible Causes:

1. Boost environment.
2. Poor workmanship.

Effects on Subsystem/Mission:

None to catastrophic depending on amount of leakage (structural failure of panel).

Backup Provisions:

None – all 10 panels required.

Inherent Preventives:

None.

Secondary Failure Mode:

Failure to extend antenna.

Possible Causes:

1. Clogged or wrong sized orifice.
2. Jamming of piston.

Effects on Subsystem/Mission:

System degraded.

Backup Provisions:

None – all 10 panels required.

Inherent Preventives:

None.

Table 9B

		Mission
Primary Failure Mode	Failure Classification	A
	Time/Cycles Units: Hrs.	0.3
	Basic Failure Rate, Units: $\times 10^{-6}$ Hrs.	0.35*
	Environment/Application Factor	1000
	Actual Failure Rate, Units: $\times 10^{-6}$ Hrs.	350
	Failure Rate Source Code	c
	Probability of Failure, $\times 10^{-6}$	105
Secondary Failure Mode	Failure Classification	A
	Time/Cycle Units: Hrs.	0.3
	Basic Failure Rate, Units: $\times 10^{-6}$ Hrs.	0.15**
	Environment/Application Factor	1000
	Actual Failure Rate, Units: $\times 10^{-6}$ Hrs.	150
	Failure Rate Source Code	c
	Probability of Failure $\times 10^{-6}$	45
	Probability of Failure, All Modes	150
	Reliability	$0.9^{(3)}850$

Failure Rate Data Sources: c) ibid.

Notes:

* Failure rate of parts as follows:

Pg. 2.316 - "O" Ring - 0.05

Pg. 2.369 - Seal (Sliding) - 0.30

** Pg. 2.348 - Orifice - 0.15.

A ibid.

Table 10A

Item: Spring.

System/Subsystem: Solar Panel Deployment.

Operational Notes:

Used on SK-BM00-9001.

Environmental Notes:

Failure would probably occur during boost.

Primary Failure Mode:

Break - fail to act as a spring (vacuum welding).

Possible Causes:

1. Boost environment.
2. Poor workmanship.

Effects on Subsystem/Mission:

Catastrophic - Panel would not deploy.

Backup Provisions:

None.

Inherent Preventives:

None.

Table 10B

	Mission
Failure Classification	A
Time/Cycles Units: Hrs. x 10^{-6}	0.3
Basic Failure Rate, Units: x 10^{-6} Hrs.	0.11
Environment/Application Factor	1000
Actual Failure Rate, Units: x 10^{-6} Hrs.	110
Failure Rate Source Code	c
Probability of Failure, x 10^{-6}	33
Probability of Failure, All Modes	33
Reliability	$9^{(4)}67$

Failure Rate Data Sources: c) ibid - Pg. 2.374.

Notes:

A ibid.

Table 11A

Item: Linear Spring Actuator.

System/Subsystem: Solar Panel Deployment.

Item Type/Description:

Used on SK-BM00-9002 through SK-BM00-9004.

Environmental Notes:

Failure could occur during boost.

Primary Failure Mode:

Leakage past the seals.

Possible Causes:

1. Boost environment.
2. Poor workmanship.

Effects on Subsystem/Mission:

None to catastrophic depending on amount of leakage (structural failure of panel).

Backup Provisions:

None.

Inherent Preventives:

None.

Secondary Failure Mode:

Failure to deploy panel.

Possible Causes:

Spring breakage or vacuum welding.

Effects on Subsystem/Mission:

Catastrophic.

Backup Provisions:

None.

Inherent Preventives:

None.

Other Failure Modes:

Failure to lock in place.

Possible Causes:

Broken lockring.

Effects on Subsystem/Mission:

Degraded performance to catastrophic.

Backup Provisions: None.

Inherent Preventives: None.

Table 11B

		Mission
Primary Failure Mode	Failure Classification	A
	Probability of Failure, x 10^{-6}	150*
Secondary Failure Mode	Failure Classification	A
	Probability of Failure, x 10^{-6}	33**
Other Failure Modes	Failure Classification	A
	Time/Cycles Units: Hrs.	0.3
	Basic Failure Rate, Units: x 10^{-6}	0.033
	Environment/Application Factor	1000
	Actual Failure Rate, Units: x 10^{-6}	33
	Failure Rate Source Code	d
	Probability of Failure x 10^{-6}	10
	Probability of Failure, All Modes	193
	Reliability	$9^{(3)}807$

Failure Rate Data Sources: d) Estimates based on other components.

Notes:

A ibid

* See Table 9B

** See Table 10B.

Table 12A

Item: Code Cutter
System/Subsystem: Solar Panel Deployment
Item Type/Description: Used on SK-BM000-9003.
Primary Failure Mode: Failure to completely cut cord.
Possible Causes: 1. Jammed piston 2. Dull cutter.
Effects on Subsystem/Mission: Catastrophic - failure to deploy.
Backup provisions: Redundant cutters.
Inherent Preventives: Lot qualification of cord cutters.

Table 12B

	Mission
Failure Classification	A
Time/Cycles Units: Cycles	1
Basic Failure Rate, Units: x 10^{-6} cycles	40
Environment/Application Factor	-
Actual Failure Rate, Units: x 10^{-6} cycles	40
Failure Rate Source Code	e
Probability of Failure, x 10^{-6}	40
Reliability	$9^{(4)}60$

Failure Rate Data Source: e) Farada, Revised 3/1/65, Pg. 2.279, Source 138, Based on explosive bolt.

4.7.6 Mathematical Models

The reliability of each one shot device is simply the complement of its failure rate (λ), where this failure rate is essentially the probability of failure (Q)

$$R = 1 - \lambda = 1 - Q$$

If two devices are used redundantly, both must fail in order to fail the system and the reliability is determined from

$$R = 1 - Q^2$$

In the case of the pin puller and swivel catch system (SK-BM00-9004) where only every other pin puller was required to function, the following was used to determine the reliability

$$R = (P + Q)^n$$

P = Probability of success

Q = Probability of failure

n = Number of items (10)

Since all combinations of failure would not result in success of the system, the failures had to be ordered such that success did occur:

$$R = P^{10} + 10P^9Q + 35P^8Q^2 + 50P^7Q^3 + 25P^6Q^4 + 2P^5Q^5$$

The reliability of time dependent components was determined from the following

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

K = Environmental/application factor (1000)

λ = Failure rate

t = Time (0.3 hr)

The boost operating time was used because the operation during deployment is short (seconds) and in many cases the boost environment was considered to be the major contributor to failure. In the case where $K \cdot \lambda \cdot t$ is small (<0.01) the first equation is applicable.

Total system reliability was determined from the product rule.

$$R_T = \prod_{i=1}^{N} R_i$$

4.8 Monopropellant Engine (With Solid-Propellant Retro)

The reliability analysis has been performed for each of five distinct mission phases as follows:

Mission Phase 1. For the period from liftoff through boost and the spacecraft injection (0.3 hour).

Mission Phase 2. For the period after injection through transit and capsule separation. This phase was further reduced into:

A. Midcourse maneuver (4 cycles or 0.056 hour)

B. Cruise (4,280 hours)

Mission Phase 3. For the period after capsule separation and including accomplishment of successful retropropulsion for the spacecraft. The analysis shows:

A. Retropropulsion (0.022 hour)

B. Cruise (50 hours)

Mission Phase 4. For the period after successful spacecraft orbit attainment and extending for one month in orbit (720 hours)

Mission Phase 5. For an additional 5 months in orbit (3,600 hours).

The analysis was performed for the five mission phases and includes both the baseline and augmented configuration. All powered flight portions of the mission (launch, midcourse and retro) have an environmental "k" factor of 1000 associated with them. In addition all four midcourse maneuvers were considered during Phase 2 although at least one of the four maneuvers may be an orbit adjust. This has no effect on the mission reliability assessment, but does tend to lower slightly the Phase 2 reliability while raising Phase 4.

Criticality factors are shown in the failure classification box of the attached reliability analysis forms. These are estimates of the probability of mission failure given that a failure mode occurs. This accounts for small leaks and other failures which may not be catastrophic to the mission.

The results are:

Engine System Reliability

	Phase 1	2	3	4	5	Total Mission
Baseline	0.999173	0.995471	0.974234	0.999964	0.99982	0.968811
Augmented	0.999173	0.997954	0.974234	0.999964	0.999817	0.97122

The above values are for the liquid engine, solid motor and their associated thrust vector controls as shown in Figures 1 and 2. When combined with the pressurization and propellant feed system the following propulsion reliabilities are obtained. These numbers are for the selected propulsion configuration.

Propulsion System Reliability

Phase 1	2	3	4	5	Total Mission
0.998314	0.99519	0.974214	0.999684	0.998912	0.966446

The failure rates used in the analysis are shown in the detailed reliability analysis sheets attached and were primarily based on the FARADA handbook, 1 April 1965 issue, as well as combined TRW data from various programs.

The probability of failure per failure mode for each component for every Mission phase is shown in Table 1. This table is for one component only with the math model taking into account the number of components per system.

4.8.1 Phase 1

This represents the most severe environment from a vibration standpoint and thus has a "K factor" of 1000 applied in the analysis. The engine is nonoperative at this time and only the connection upstream from the valves and the jet vane assembly is subject to failure.

Thus,

$$R_1 = (1 - E_1)(1 - F_1) = (1 - 15 \times 10^{-6})(1 - 812.4 \times 10^{-6})$$

$$= (0.999985)(9991876) = 999173$$

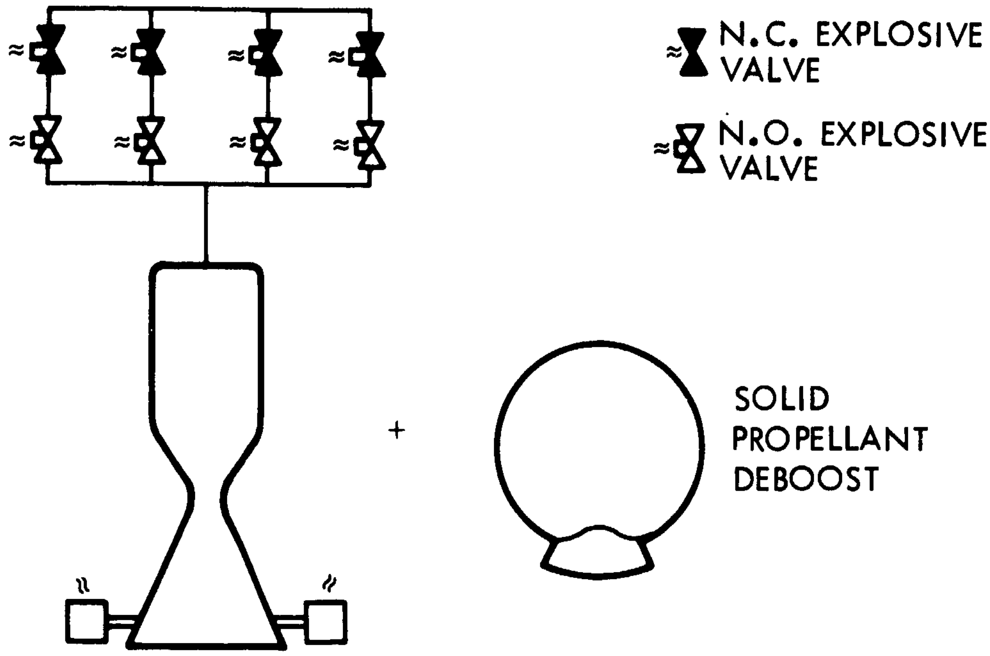


Figure 1.

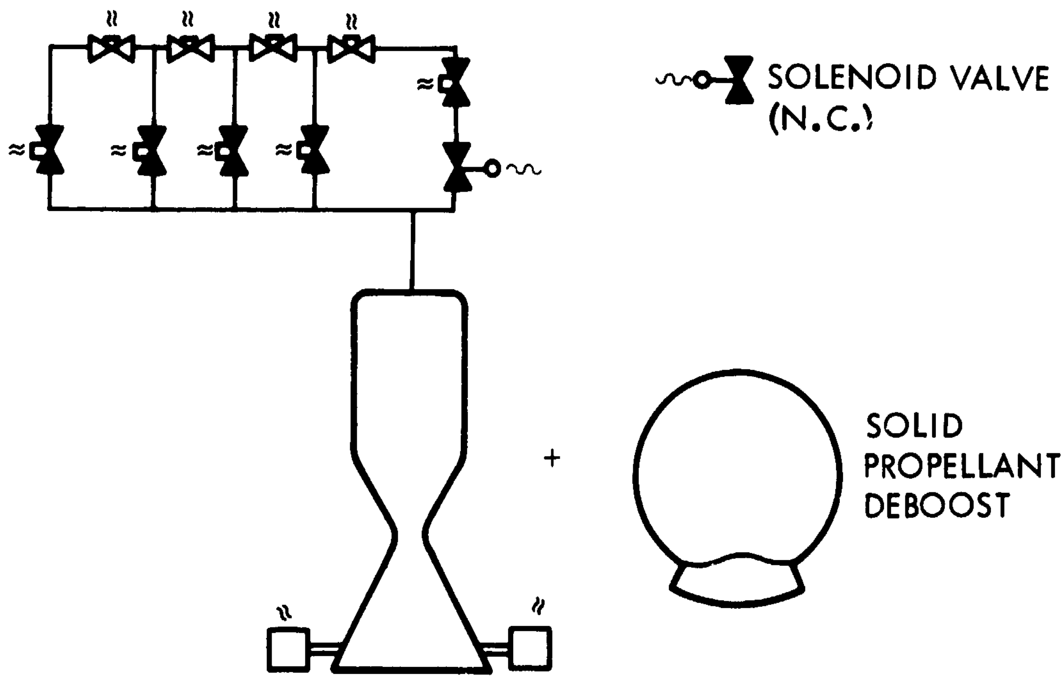


Figure 2.

Table 1. Reliability (Baseline A)
Failure Mode Probability Summary
Monopropellant Engine (A)

Component	Failure Mode	Prob of Failure (Phase) x 10^{-6}				
		1	2	3	4	5
A Explosive Valve (N. C.)	a. Failure to Fire	N. A.	294	N. A.	N. A.	N. A.
	b. Premature	N. A.	6	N. A.	N. A.	N. A.
B Explosive Valve (N. O.)	a. Failure to Fire	N. A.	294	N. A.	N. A.	N. A.
	b. Premature	N. A.	6	N. A.	N. A.	N. A.
	c. Leak after closing	N. A.	29.4	N. A.	N. A.	N. A.
C Fuel Injector	Plugging	N. A.	256	N. A.	N. A.	N. A.
D Combustion Chamber and Nozzle	Weld Failure	N. A.	371.5	N. A.	N. A.	N. A.
E Fittings and Connect	Leakage	15	214.56	3.6	36	180
F Jet Vane Assy	Incorrect Position or Inoperative	812.4	1178.6	71.6	N. A.	N. A.
G Ignitor	Failure to Ignite	N. A.	N. A.	2237	N. A.	N. A.
H Solid Propellant Motor	a. Burnthrough	N. A.	N. A.	6400	N. A.	N. A.
	b. Overpressure	N. A.	N. A.	6700	N. A.	N. A.
	c. Structural	N. A.	N. A.	5500	N. A.	N. A.
I Thrust Vector Control (Liquid Injection)	Incorrect or Inoperative	N. A.	N. A.	5000	N. A.	N. A.

where

R_1 = Engine system reliability for Phase 1.

E_1 = Component E (Table 1) probability of failure for Phase 1.

The above applies for the remainder of the math model where a capital letter refers to a specific component in Table 1, a numeral subscript to the applicable phase, and a small letter subscript to the failure mode as shown in Table 1, i.e., B_{2b} means explosive valve (N.O.) Phase 2 probability of premature. H_3 means solid propellant motor, Phase 3, all failure modes combined.

4.8.2 Phase 2

This phase includes the necessary midcourse maneuver and the long cruise. The monopropellant engine operates here and thus all its failure modes are applicable. All four branches of the explosive valves (Figure 1) have to operate.

Thus

$$R_2 = (1 - 4A_2)(1 - 4B_2)(1 - C_2)(1 - D_2)(1 - E_2)(1 - F_2)$$

$$= \left[1 - 4(300 \times 10^{-6})\right]\left[1 - 4(329.4 \times 10^{-6})\right](1 - 256 \times 10^{-6})$$

$$(1 - 371.5 \times 10^{-6})(1 - 214.56 \times 10^{-6})(1 - 1178 \times 10^{-6})$$

$$R_2 = (0.9988)(0.998682)(0.999744)(0.999629)(0.999785)$$

$$(0.998822)$$

$$= 0.995471$$

4.8.3 Phase 3

This phase concerns the retropropulsion sequence and as such is primarily concerned with the solid propellant motor. We are, however, including connection leakage here in the event that orbit adjust is necessary. The orbit adjust sequence is assumed to be at least one of the four monopropellant firings which are included, somewhat out of sequence, in Phase 2. This does not affect our assessed reliability for the mission but does lower our Phase 2 value while raising our Phase 4 assessment.

Thus

$$R_3 = (1 - E_3)(1 - F_3)(1 - G_3)(1 - H_3)(1 - I_3)$$

$$= (1 - 3.6 \times 10^{-6})(1 - 71.6 \times 10^{-6})(1 - 2237 \times 10^{-6})$$

$$(1 - 18,600 \times 10^{-6})(1 - 5000 \times 10^{-6})$$

$$= (0.9999964)(0.9999284)(0.997763)(0.9814)(0.995)$$

$$= 0.974234$$

4.8.4 Phase 4

This phase concerns the spacecraft one-month Mars orbit and is subject to orbit adjust as discussed during Phase 3.

Thus

$$R_4 = (1 - E_4) = (1 - 36 \times 10^{-6}) = 0.999964$$

4.8.5 Phase 5

This phase concerns the additional 5 months in orbit. Since all functions have been performed, the only concern here is that of propellant leakage which could contaminate the spacecraft or cause other hazards.

Thus

$$R_5 = (1 - E_5) = (1 - 180 \times 10^{-6}) = 0.99982$$

From the above we obtain the entire mission reliability, as follows:

$$R_{mission} = (R_1)(R_2)(R_3)(R_4)(R_5) = (0.999173)(0.995471)(0.974234)(0.999964)(0.99982)$$
$$= 0.968811$$

4.8.6 Selected Configuration

The augmented configuration is identical to the baseline except for the valves (Figure 2). The augmented configuration also uses explosive valves except in a different arrangement which includes an additional N.C. valve in series with a normally closed solenoid valve. Thus, the solenoid valve offers an effective multi-operating backup should any of the other explosive valves fail. In addition the N.C. explosive valve in series with the solenoid eliminates the solenoid's major failure mode (leakage) up to the time it is required to operate.

R_{valves} = All four sets of explosive valves work + one set fails, the solenoid valve operates with its N.C. Explosive valve + two sets fail, the solenoid operates twice its N.C. Explosive valve operates, and the solenoid does not leak + three sets fail, the solenoid operates 3 times, the N.C. Exp. valve operates and the solenoid does not leak + all four sets fail, the solenoid operates 4 times. The N.C. Exp. valve operates and the solenoid does not leak.

$$R_{valves} = (M)^4 + 4(M)^3 (1 - M) (N) (P) + 6(M)^2 (1 - M)^2 (N)^2 (P) (Q) + 4(M) (1 - M)^3 (N)^3 (P) (Q) + (1 - M)^4 (N)^4 (P) (R)$$

where

M = $1 - A_2 + B_2$ (reliability of N. C. x N. O. explosive valve for all failure modes)

N = Reliability of one solenoid valve cycle (open and close)

P = $1 - A_2$ (reliability of N. C. explosive valve)

Q = The reliability of the solenoid valve for 800 hours in the leakage mode.

R = The reliability of the solenoid valve for 5000 hours in the leakage mode.

$$\begin{aligned} R_{valves} &= (0.999371)^4 + 4(0.999371)^3 (0.000629) (0.9999993) (0.9997) \\ &+ 6(0.999371)^2 (0.000629)^2 (0.9999993)^2 (0.9997) (0.999886) \\ &+ 4(0.999371) (0.000629)^3 (0.9999993)^3 (0.9997) (0.999886) \\ &+ (0.000629)^4 (0.9999993)^4 (0.9997) (0.999285) \\ &= 0.997486 + 0.0025102 + 0.00000237 + 0 + 0 \\ &= 0.9999725 \end{aligned}$$

The above represents an oversimplification of the Analysis for the valve configuration in that it does not look at the effects of each failure mode but instead combines all modes. Since a detailed analysis requires considerably more time, the above is felt to be a valid approximation for the purposes of this analysis. It appears, however, that the reliability value obtained with the approximation is somewhat higher than the actual.

$$\begin{aligned} R'_2 &= (R_{valves}) (1 - C_2) (1 - D_2) (1 - E_2) (1 - F_2) \\ &= (0.9999725) (0.999744) (0.999629) (0.999785) (0.998822) \\ &= 0.997954 \end{aligned}$$

Table 2A

Item: Explosive (Squib) Valve

System/Subsystem: Propulsion (A)

Primary Failure Mode:

Failure to fire.

Possible Causes:

Faulty squib, shorted wiring.

Effects on Subsystem/Mission

Catastrophic.

Inherent Preventives:

Redundant bridgewires, careful checkout.

Secondary Failure Mode:

Premature firing.

Possible Causes:

Stray RF signals, static discharge.

Effects on Subsystem/Mission:

Catastrophic.

Inherent Preventives:

Adequate shielding, 100% no-fire test both current and capacitor discharge.

Other Failure Modes:

Leaks after closing.

Possible Causes:

Contamination.

Effects on Subsystem/Mission:

Negligible to catastrophic.

Inherent Preventives:

Adequate filtration in conjunction with thorough cleaning and careful assembly of system.

Table 2B

		Nominal	2	Mission
Primary Mode	Failure Classification		1	
	Time/Cycles, Units: Cycle		1	
	Basic Failure Rate, Units: x 10^{-6}	294	294	
	Failure Rate Source Code	a		
	Probability of Failure, x 10^{-6}		294	
Secondary Mode	Failure Classification		1	
	Time/Cycles Units: Cycle		1	
	Basic Failure Rate, Units: x 10^{-6}	6*	6	
	Failure Rate Source Code	a		
Other Modes	Time/Cycles Units: Cycle		1	
	Basic Failure Rate, Units: x 10^{-6}	29.4	29.4	
	Failure Rate Source Code	b		
	Probability of Failure x 10^{-6}		29.4	
	Probability of Failure, All Modes		$0.0^{(3)}329$	$0.0^{(3)}329$
	Reliability		$0.9^{(3)}671$	$0.9^{(3)}671$

Failure Rate Data Sources: a) TRW Experience - 2000 Squib Firings with no failure Probability of Failure 3×10^{-4} a 50% conf telephone call to Farada office indicated 47,332 firings and 10 failures (Minn. Honeywell) for a failure rate of 211×10^{-4}. b) Failure rate for this mode considered to be an order of magnitude less than primary mode.

Notes:

* Experience from Bruceton testing indicates premature failure mode is approx 1/50 of total failure probability.

Table 3A

Item: Solenoid Valve.

System/Subsystem: Propulsion (A).

Item Type/Description:

Spring loaded normally closed.

Primary Failure Mode:

Leakage.

Possible Causes:

Contamination.

Effects on Subsystem/Mission:

Could be catastrophic.

Backup Provisions:

Normally closed explosive valve in series with solenoid. This prevents leakage until solenoid is needed.

Inherent Preventives:

Adequate filtration in conjunction with thorough cleaning and careful assembly of system.

Secondary Failure Mode:

Failure to open.

Possible Causes:

Open or short circuit. Mechanical interference.

Effects on Subsystem/Mission:

Catastrophic.

Backup Provisions:

Only has to operate if one of the explosive valves fails.

Inherent Preventives:

Adequate testing and checkout.

Other Failure Modes:

Failure to close.

Possible Causes:

Mechanical interference.

Effects on Subsystem/Mission:

Catastrophic.

Backup Provisions:

Only has to operate if one of the explosive valves fails.

Inherent Preventives:

Same as above.

Table 3B

Mode		Nominal	Worst	Best	2A	2B	3	4	5	Mission
Primary Mode	Failure Classification					0.5	0.5	0.5	0.5	
	Time/ Cycles Units: hrs.					4280	50	720	3600	
	Basic Failure Rate, Units: x 10^{-9} hrs.	286	67300	90		286	286	286	286	
	Environment/Application Factor					1	1	1	1	
	Actual Failure Rate, Units: x 10^{-9}					286	286	286	286	
	Failure Rate Source Code	A	(e)	(d)						
	Probability of Failure, x 10^{-6}					612	7.15	102.96	514.8	
Secondary Mode	Failure Classification				1					
	Time/Cycles Units: cycles				1					
	Basic Failure Rate, Units: x 10^{-9} cycles	420	995	38	420					
	Actual Failure Rate, Units: x 10^{-9} cycles				420					
	Failure Rate Source Code	(a)	(c)	(b)						
	Probability of Failure x 10^{-9}				420					
	Failure Classification				1					
	Time/Cycles Units: cycles				1					

Table 3B Continued

		Nominal	Worst	Best	2A	2B	3	4	5	Mission
Other Modes	Basic Failure Rate, Units: x 10^{-9} cycles	280	665	25	280					
	Environment/Application Factor				1					
	Actual Failure Rate, Units: x 10^{-9} cycles				280					
	Failure Rate Source Code	(a)	(c)	(b)						
	Probability of Failure x 10-9				280					
	Probability of Failure, All Modes				$0.0^{(6)}7$	$0.0^{(3)}612$	$0.0^{(5)}715$	$0.0^{(3)}103$	$0.0^{(3)}5148$	$0.0^{(2)}1238$
	Reliability				$0.9^{(6)}3$	$0.9^{(3)}388$	$0.9^{(5)}285$	$0.9^{(3)}897$	$0.9^{(3)}4852$	$0.9^{(2)}8762$

Failure Rate Data Sources:

(a) FARADA page 2.416 Source 136 Wright-Patterson

(A) FARADA page 2.417 Source 80 General Dynamics

(b) Walter Kidde Report R-1681 "Solenoid Valve Qual Test"

(c) FARADA page 2.417 Source 136

(d) FARADA page 2.413 Source 84 Grumman

(e) FARADA page 2.417 Source 125 Space-General

Table 4A

Item: Fuel Injector

System/Subsystem: Propulsion (A)

Primary Failure Mode:

Plugging.

Possible Causes:

Contamination, damage due to improper handling.

Effects on Subsystem/Mission

Minor to complete loss of performance.

Inherent Preventives:

Adequate filtration, stringent cleanliness and handling procedures.

Table 4B

	Nominal	Worst	Best	2A	Mission
Failure Classification				1*	
Time/Cycles Units: hrs.	0.056	0.333	0	0.056	
Actual Failure Rate, Units: hrs.	04572	30456	$0.0^{(3)}6624$	04572	
Failure Rate Source Code	a. b.				
Probability of Failure, x 10^{-6}				256	256
Reliability				$0.9^{(3)}744$	$0.9^{(3)}744$

Notes: * Failure Classification of . 1 used since data is based on a bipropellant variable thrust injector.

Failure Rate Data Sources: a) STL memo 64-9701, 3-127, "Reliability Input for September Surveyor Progress Report "8 October 1964.
b) Lem Descent Engine Component Failure Summary, Dated 04-06-65.

Notes:

Surveyor 37,320 sec of firing with no applicable failures

Lemde 17,160 sec of firing with no applicable failures

54,480

$$\chi^2\ 0.50,0 = 1.386$$

$$\chi^2\ 0.01,0 = 0.0201$$

$$\chi^2\ 0.99,0 = 9.21$$

$$\lambda = \frac{X^2}{2\ T}$$

$$\lambda Nom = \frac{1.386}{1.09 \times 10^5 \text{ sec}} = 1.27 \times 10^{-5} \text{ sec} = 04572 \text{ hr}$$

$$\lambda best = \frac{0201}{1.09 \times 10^5 \text{ sec}} = 1.84 \times 10^{-7} \text{ sec} = 0006624 \text{ hr}$$

$$\lambda worst = \frac{9\ 21}{1.09 \times 10^5 \text{ sec}} = 8.46 \times 10^{-5} \text{ sec} = 30456 \text{ hr}$$

Table 5A

Item: Combustion Chamber and Nozzle

System/Subsystem: Propulsion (A)

Item Type/Description:

All welded Haynes 25 alloy.

Primary Failure Mode:

Weld failure.

Possible Causes:

Porosity or microcracks.

Effects on Subsystem/Mission:

Leaks or rupture of combustion chamber. Minor to catastrophic.

Inherent Preventives:

Inspection of welds (X-ray), proof test.

Table 5B

	Nominal	Worst	Best	2A
Failure Classification				0. 1*
Time/Cycles Units: hrs.	0. 056	0. 333	0	0. 056
Actual Failure Rate, Units:	0. 06608	0. 4391	0. 00096	0. 06608
Failure Rate Source Code	(a) (b)			
Probability of Failure, x 10⁻				0. 0(3)3715
Reliability				0. 9(3)6285

Notes: * Failure classification of . 1 used since data is based on bipropellant ablative liner combustion chamber.

Failure Rate Data Sources: (a) STL Memo 64-9701. 3-127 "Reliability Input for September Surveyor Progress Report " 8 October 1964.
(b) LEM Descent Engine Component Failure Summary, dated 4-6-65.

Notes:

Surveyor	20, 593	sec of firing with no applicable failures
Lemde	17, 160	sec of firing with no applicable failures
	37, 753	

$$\chi^2_{0.50,0} = 1.386 \qquad \lambda_{Nom} = \frac{1.386}{20.974} = 0.06608$$

$$\chi^2_{0.01,0} = 0.0201 \qquad \lambda_{Best} = \frac{0.0201}{20.974} = 0.00096$$

$$\chi^2_{0.90,0} = 9.21 \qquad \lambda_{worst} = \frac{9.21}{20.974} = 0.4391$$

$$\lambda = \frac{\chi 2}{2T}$$

Table 6A

Item: Fittings and Connections

System/Subsystem: Propulsion (A)

Item Type/Description:

* Mechanical Connections, not welded or brazed. 1 required.

Primary Failure Mode:

Leakage.

Possible Causes:

Contamination and faulty assembly.

Effects on Subsystem/Mission:

Negligible to catastrophic depending on magnitude of leak rate.

Inherent Preventives:

Careful assembly and adequate inspection.

Remarks:

* Welded and brazed fittings are considered to have a negligible probability of failure.

Table 6B

	Nominal	Worst	Best	1	2A	2B	3A	3B	4	5	Mission
Failure Classification				0.5	0.1	0.5	0.5	0.5	0.5	0.5	
Time/Cycles Units: hrs.	0.056	0.333	0	0.3	0.056	4280	0.022	50	720	3600	
Basic Failure Rate, Units: x 10^{-9} hrs.	100	710	40	100	100	100	100	100	100	100	
Environment/ Application Factor				1000	1000	1	1000	1	1	1	
Actual Failure Rate, Units: x 10^{-6}				100	100	0.1	100	0.1	0.1	0.1	
Failure Rate Source Code	(a)	(b)	(c)								
Probability of Failure, x 10^{-6}				15	0.56	214	1.1	2.5	36	180	499.16
Reliability				$9^{4}85$	$9^{6}44$	$9^{3}786$	$9^{5}89$	$9^{5}75$	$9^{4}64$	$9^{3}82$	$9^{3}551$

Failure Rate Data Sources:

(a), (b), (c) D. R. Earles, Reliability Engineering Data Series, April 1962.

Table 7

Item: Jet Vane Assembly

System/Subsystem: Propulsion (A)

Item Type/Description:

This assembly consists of four jet vane actuators. Each of which drives a deflection vane directly, without gearing.

Operational Notes:

Jet vanes used for midcourse correction thrust vector control.

Primary Failure Mode:

Improper vane positioning.

Possible Causes:

Potentiometer wiper contact problem.

Effects on Subsystem/Mission:

Could be catastrophic.

Inherent Preventives:

Thorough ground testing and checkout.

Secondary Failure Mode:

Stuck vane.

Possible Causes:

Contamination or bearing problem.

Effects on Subsystem/Mission:

Catastrophic.

Inherent Preventives:

Adequately sealed unit.

Table 7A

	Nominal	1	2A	2B	3A	3B	4	Mission
Failure Classification		1	1	1	1	1	1	
Time/Cycles Units: hrs.		0.3	0.056	4280	0.022	50	720	
Basic Failure Rate, Units: x 10^{-9} hrs.	2708	2708	2708	240	2708	240		
Environment/Application Factor		1000	1000	1	1000	1		
Actual Failure Rate, Units: x 10^{-6}		2708	2709	0.24	2708	0.24		
Failure Rate Source Code	a							
Probability of Failure, x 10^{-}		$0^{3}8124$	$0^{3}1516$	$0^{2}1027$	$0^{4}596$	$0^{4}12$		$0^{2}206$
		Secondary mode included above						
Reliability		$9^{3}1876$	$9^{3}8484$	$9^{2}8973$	$9^{4}404$	$9^{4}88$		$9^{2}794$

a) See Next Page

Table 7B

Jet Vane Assembly

Item	Quantity Per Assy	Failure Rate x 10^{-9}	Total Failure Rate x 10^{-9}
1. D. C. Torque Motor	4	200	800
2. Dual Potentiometer	4	92	368
3. Bearing	8	100	800
4. O-Ring Seal	4	600 x 0.1*	240
5. Connector 8 leads	4	80	320
6. Vane on Output Shaft	4	45	180
			2,708

Motor, Brushless	FARADA Page 2. 102 Source 86 (Autonetics)
Potentiometer	FARADA Page 2. 111 Source 96 (Minn. Honeywell)
Seals "O-Ring"	FARADA Page 2. 368 Source 9 (Minn. Honeywell)
Vane, Exhaust Guide	FARADA Page 2.419 Source 111 (Boeing)

* 0. 1 criticality factor. Since seal failure does not cause, but may contribute, to mission failure.

Table 8A

Item: Ignitor (Solid Motor)
System/Subsystem: Propulsion (A)
Primary Failure Mode:
Failure to ignite.
Possible Causes:
Open bridgewires, faulty ignition charge.
Effects on Subsystem/Mission:
Catastrophic.

Table 8B

	Nominal	Worst	Best	3	Mission
Failure Classification				1	
Time/Cycles Units: cycles				1	
Actual Failure Rate, Units: x 10^{-6} cycles	2237	5176	1328	2237	
Failure Rate Source Code	(a)				
Probability of Failure, x 10^{-6}				2237	
Reliability				9^2776	9^2776

(a) Minuteman Stages 1 thru 3, Tests. (753 tests with one failure)

$$\text{Nom}_{50\%\text{ conf}} = 0.002237$$

$$\text{Worst}_{90\%\text{ conf}} = 0.005176$$

$$\text{Best} = \frac{752}{753} = 0.001328$$

Table 9A

Item: Solid Propellant Motor

System/Subsystem: Propulsion (A)

Item Type/Description:

Spherical motor with submerged nozzle. 10,000 lb-thrust. 800,000 lb-sec impulse.

Primary Failure Mode:

Burnthrough.

Possible Causes:

Insulation bond separation, insulation quality or thickness deficiency.

Effects on Subsystem/Mission:

Catastrophic.

Inherent Preventives:

Stringent manufacturing controls coupled with adequate quality control and inspection (X-ray).

Secondary Failure Mode:

Overpressure.

Possible Causes:

Propellant deficiencies and voids, erosive or unstable burning.

Effects on Subsystem/Mission:

Catastrophic.

Inherent Preventives:

Same as above.

Other Failure Modes:

Structural.

Possible Causes:

Inadequate weld; joint failure, seal failure.

Effects on Subsystem/Mission:

Catastrophic.

Inherent Preventives:

Same as above.

Table 9B

		Nominal	3	Mission
Other Modes	Basic Failure Rate, Units:	0.0064	0.0064	
	Failure Rate Source Code		(a)	
Secondary Mode	Basic Failure Rate, Units:	0.0067	0.0067	
	Failure Rate Source Code		(a)	
Other Modes	Basic Failure Rate, Units:	0.0055	0.0055	
	Failure Rate Source Code		(a)	
	Probability of Failure, All Modes		0.0186	0.0186
	Reliability		0.09814	0.09814

Failure Rate Data Sources:

(a) AIAA Paper No. 65-165 "Development of Malfunction Sensors for Use on Large Solid Rocket Motors" AIAA 6th Solid Prop. Rocket Conf. Feb 1-3, 1965.

Table 10A

Item: Thrust Vector Control (Solid Motor)
System/Subsystem: Propulsion (A)
Item Type/Description:
Liquid Injection
Operational Notes:
Thrust vector control for solid rocket.

Table 10B

	Nominal	3	Mission
Actual Failure Rate, Units: x 10^{-6}	5000	5000	
Failure Rate Source Code	(a)		
Probability of Failure, x 10^{-6}		5000	5000
Reliability		0.995	0.995

Failure Rate Data Sources:

(a) JPL Technical Memorandum No. 33-219, Page 3 (May 10, 1965)

Notes:

Liquid Injection per Wing VI Minuteman shows 53 trials with 0 failures

$$R_{50\%\ conf.} = 0.98698$$

Phase 5 changes in that we now have to account for solenoid valve leakage provided one or more of the explosive valves failed.
Thus,

$$R_5' = (R_5) \left[1 - (1 - M^4)(S)\right]$$

where

$$\begin{aligned}
S &= \text{Prob of failure of the solenoid valve for 3600 hrs in the leakage mode.} \\
&= (0.99982) \left[1 - (0.002514)(286 \times 10^{-9})(3600)\right] \\
&= (0.99982)(1 - 0.0000026) = (0.99982)(0.9999974) \\
&= 0.999817 \\
R_{mission} &= (R_1)(R_2')(R_3)(R_4)(R_5') \\
&= (0.999173)(0.997954)(0.974234)(0.999964) \\
&\quad (0.999817) \\
&= 0.97122
\end{aligned}$$

4.9 Bipropellant Engine

The attached analysis was performed for the five mission phases defined in 4.8 and includes both the baseline and reference configuration. All powered flight portions of the mission (launch, midcourse and retro) have an environmental "k" factor of 1000 associated with them. In addition, all four midcourse maneuvers were considered during phase 2 although it is felt that at least one of the four maneuvers probably will be an orbit adjust. This has no effect on the mission reliability assessment but does tend to lower slightly the phase 2 reliability while raising phase 4.

Criticality factors are shown in the failure classification box of the attached reliability analysis forms. These are estimates of the probability of mission failure given that a failure mode occurs. This accounts for small leaks and other failures which may not be catastrophic to the mission.

The results are:

ENGINE SYSTEM RELIABILITY

PHASE

	1	2	3	4	5	MISSION
BASELINE	0.998762	0.991649	0.980118	0.999722	0.99861	0.969110
REFERENCE	0.998847	0.992892	0.980131	0.999928	0.99964	0.971621

The above values are for the engine, and its translation control as shown in Figures 1 and 2. When combined with the pressurization and propellant feed system* the following propulsion reliabilities are obtained:

PROPULSION SYSTEM RELIABILITY

PHASE

	1	2	3	4	5	MISSION
BASELINE	0.996055	0.983025	0.979186	0.999046	0.996727	0.955494
REFERENCE	0.996120	0.985296	0.979684	0.999286	0.997841	0.958754

* Ref: A-830-BM00-59 (DAC-VM-28) 24 June 1965, Page 5 (Bipropellant)

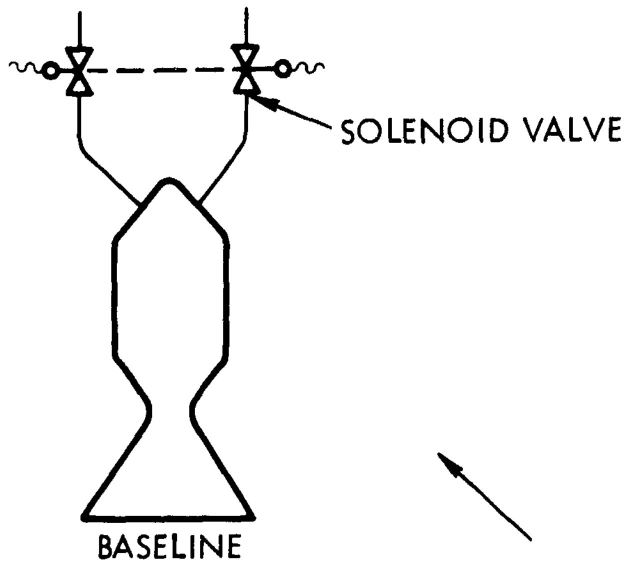
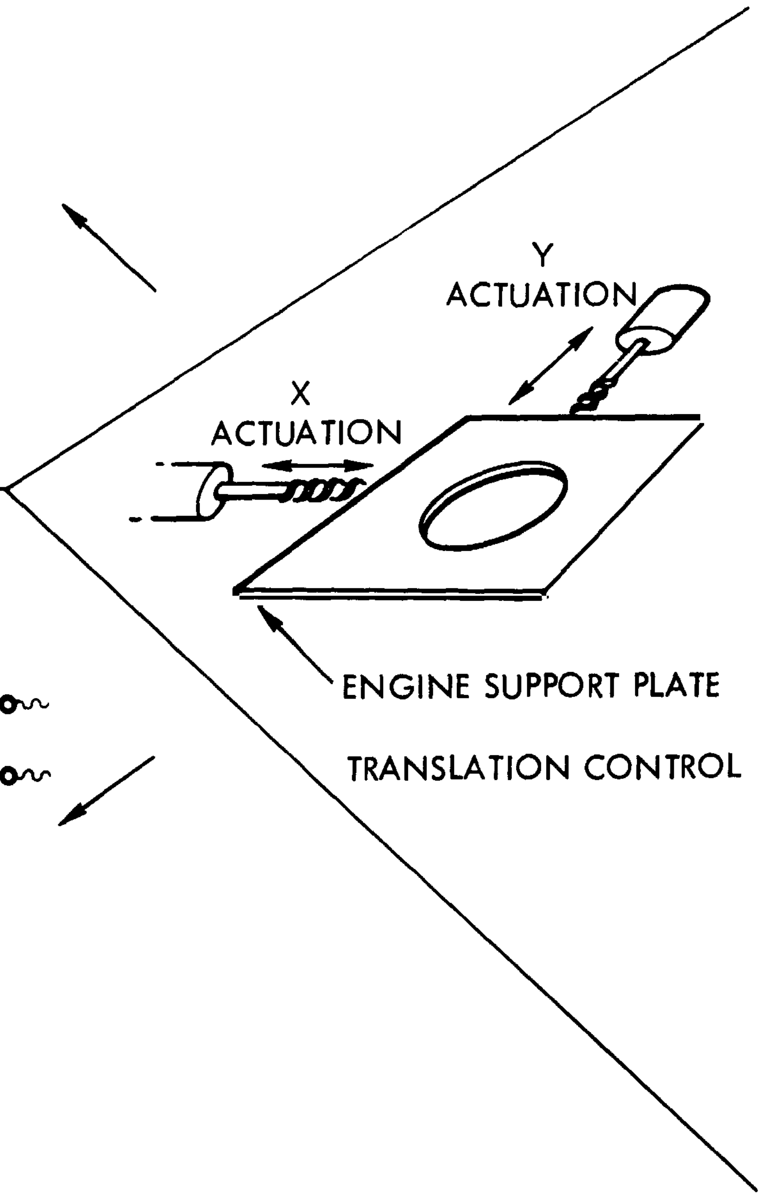


Figure 1. Baseline
Translation Control



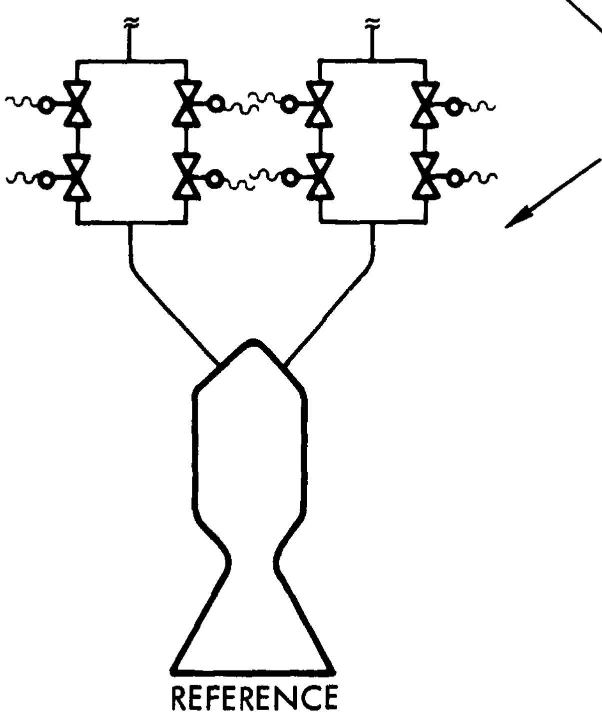


Figure 2. Reference
Translation Control

The failure rates used in the analysis are shown in the detailed reliability analysis sheets attached and were primarily based on the FARADA handbook, 1 April 1965 issue, as well as combined TRW data from various programs.

The probability of failure per failure mode for each component for every mission phase is shown in Table 1. This Table is only for one component with the math model taking into account the number of components per system.

Table 1. Failure Mode Probability Summary
Bipropellant Engine (B)

Component	Failure Mode	Prob of Failure (Phase) x 10^{-6}				
		1	2	3	4	5
A Solenoid Valve	a. Leakage	42.9	612	7.15	103	515
	b. Failure to Open	N. A.	2.1	4.2	N.A.	N.A.
	c. Failure to Close	N. A.	1.4	2.8	N.A.	N.A.
B Injector	Poor Combustion Pattern	N. A.	1540	4580	N.A.	N.A.
C Combustion Chamber and Nozzle	Material Outgassing	N. A.	3710	11070	N.A.	N.A.
D Fittings and Connect.	Leakage	15	214.56	11.35	36	180
E Translation Control	Inoperative	561	209.44	625.58	N.A.	N.A.

Reliability (Baseline B)

4.9.1 Phase 1

This represents the most severe environment from a vibration standpoint and thus has a "K factor" of 1000 applied in our analysis. The engine is nonoperative at this time and the valves, connections and translation control are subject to their failure modes. The translation control is dormant at this time but has to survive the launch environment. During this phase we applied a criticality factor of 0.5 to its operating failure rate since the mechanism is less likely to fail in this environment while dormant.

Thus

$$R_1 = (1 - 2A_1)(1 - 2D_1)(1 - 2E_1)$$

$$= (1 - 85.8 \times 10^{-6})\ (1 - 30 \times 10^{-6})\ (1 - 1122 \times 10^{-6})$$

$$= (0.9999142)\ (0.999970)\ (0.998878)$$

$$= 0.998762$$

R_1 = Engine system reliability for Phase 1

A_1 = Component A (Table 1) probability of failure for Phase 1.

The above applies for the remainder of the math model where a capital letter refers to a specific component in Table 1, a number of subscript to the applicable phase and a small letter subscript to the failure mode as shown in Table 1 i.e., - A_{2c} means solenoid valve, Phase 2, failure to close. A_3 means solenoid valves, Phase 3, all failure modes. $A_{2b,c}$ means solenoid valve, Phase 2, failure to open and/or close.

4.9.2 Phase 2

This phase includes the necessary midcourse manuvers and the long cruise. The solenoid valves have to open and close four times in this phase.

$$R_2 = (1 - 2A_{2a})\ (1 - 8A_{2b,c})\ (1 - B_2)\ (1 - C_2)\ (1 - 2D_2)$$
$$(1 - 2E_2)$$
$$= (1 - 1224.08 \times 10^{-6})\ (1 - 28 \times 10^{-6})\ (1 - 2560 \times 10^{-6})$$
$$(1 - 3710 \times 10^{-6})\ (1 - 429.12 \times 10^{-6})\ (1 - 418.88 \times 10^{-6})$$
$$R_2 = (0.998776)\ (0.999972)\ (0.99744)\ (0.99629)\ (0.999571)$$
$$(0.999581)$$
$$= 0.991649$$

4.9.3 Phase 3

This phase concerns the retropropulsion sequence and the 50-hour cruise. Valve leakage is included here to allow for any necessary orbit adjusts.

Thus,

$$R_3 = (1 - 2A_3)\ (1 - B_3)\ (1 - C_3)\ (1 - 2D_3)\ (1 - 2E_3)$$
$$(1 - 14.2 \times 10^{-6})\ (1 - 7630 \times 10^{-6})$$
$$(1 - 11070 \times 10^{-6})\ (1 - 22.7 \times 10^{-6})$$
$$(1 - 1249.16 \times 10^{-6})$$
$$= (0.9999858)\ (0.99237)\ (0.98893)\ (0.9999773)$$
$$(0.998751)$$
$$= 0.980118$$

4.9.4 Phase 4

This phase concerns the spacecraft one-month Mars orbit. The problem encountered here is that of leakage at the connections or solenoid valves. This failure mode could prevent orbit adjust were it necessary or in general is undesirable in that it may contaminate the spacecraft or cause other hazards.

$$R_4 = (1 - 2A_4)\ (1 - 2D_4)$$

$$(1 - 206 \times 10^{-6})\ (1 - 72 \times 10^{-6})$$

$$= (0.999794)\ (0.999928)$$

$$= 0.999722$$

4.9.5 Phase 5

This phase concerns the additional 5 months in orbit and is therefore similar to the above Phase 4 conditions.

Thus

$$R_5 = (1 - 2A_5)\ (1 - 2D_5) = (1 - 1030 \times 10^{-6})$$

$$(1 - 360 \times 10^{-6})$$

$$= (0.99897)\ (0.99964)$$

$$= 0.99861$$

From the above we obtain the entire mission reliability, as follows:

$$R_{Mission} = (R_1)\ (R_2)\ (R_3)\ (R_4)\ (R_5)$$

$$= (0.998762)\ (0.991649)\ (0.980118)\ (0.999722)$$

$$(0.99861)$$

$$= 0.969110$$

4.9.6 Reference configuration

The reference configuration is identical to the baseline except for valves (see Figures 1 and 2). The reference configuration uses quad redundant solenoid valves in both the fuel and oxidizer sections. The valves in this memo are applicable here since they represent the same mission profile. One difference is that the valves in the engine are mechanically linked. This will have a minor effect on the analysis since we do lose some independence in the valve operating modes. This is somewhat off-set by the fact that either quad valve set failing is catastrophic to the mission.

In the analysis,

G_a = Probability of failure for leakage (one quad valve set)

G_b = Probability of failure to open (one quad valve set)

G_c = Probability of failure to close (one quad valve set)

In phase 1, only the leakage mode is present for the quad solenoid valves with other components remaining as in the baseline analysis.

Thus, for Phase 1,

$$R_1' = (1 - 2D_1)\ (1 - 2E_1)\ (1 - 2G_{1a})$$

$$= (1 - 30 \times 10^{-6})\ (1 - 1122 \times 10^{-6})$$

$$\left[1 - (2)\ (0.147^{*} \times 10^{-9})\right]$$

$$= (0.99997)\ (0.998878)\ (0.9^{9}706)$$

$$= 0.998847$$

The reasoning for all the additional phases is the same as for the baseline analysis except that the quad redundant valves are subsitituted for the single solenoid valves.

Phase 2

$$R_2' = (1 - B_2)\ (1 - C_2)\ (1 - 2D_2)\ (1 - 2E_2)\ (1 - 2G_{2a})$$

$$(1 - 2G_{2b,\,c})$$

$$= (1 - 2560 \times 10^{-6})\ (1 - 3710 \times 10^{-6})\ (1 - 429.12 \times 10^{-6})$$

$$= (1 - 418.88 \times 10^{-6})$$

$$\left[1 - (2)\ (30.3^{*} \times 10^{-9})\right]\quad \left[1 - (2)\ (1^{**} \times 10^{-9})\right]$$

$$= (0.99744)\ (0.99629)\ (0.999571)\ (0.999581)$$

$$(0.9^{7}394)\ (0.9^{8}8)$$

$$= 0.992892$$

---

* = $1 - 0.9^{9}853$; page 14 of referenced memo (Phase 1 Leakage)

** = $1 - 0.9^{7}697$ and $1 - 0.9^{9}+$; page 14 of referenced memo Phase 2

Phase 3

$$R_3' = (1 - B_3)\ (1 - C_3)\ (1 - 2D_3)\ (1 - 2E_3)\ (1 - 2G_{3a})$$

$$(1 - 2G_{3b,\,c})$$

$$= (1 - 7630 \times 10^{-6})\ (1 - 11070 \times 10^{-6})\ (1 - 22.7 \times 10^{-6})$$

$$(1 - 1249.16 \times 10^{-6})\ \left|1 - (2)\ (0.35\overset{*}{4} \times 10^{-9})\right|$$

$$\left|1 - (2)\ (1 \times 10^{-9})\right|$$

$$= (0.99237)\ (0.98893)\ (0.999773)\ (0.998751)$$

$$(0.9^{9}29)\ (0.9^{8}8)$$

$$= 0.980131$$

$$* \quad = \text{leakage for 50 hrs} = \left(\frac{30.3 \times 10^{-9}}{4280}\right)\ (50) = 0.354 \times 10^{-9}$$

Phase 4

$$R_4' = (1 - 2D_4)\ (1 - 2G_{4a})$$

$$= (1 - 72 \times 10^{-6})\ \left|1 - (2)\ (5.\overset{*}{1} \times 10^{-9})\right|$$

$$= (0.999928)\ (0.9^{7}898)$$

$$= (0.999928)$$

$$* \quad = \text{leakage for 720 hrs} = \left(\frac{30.3 \times 10^{-9}}{4280}\right)\ (720) = 5.1 \times 10^{-9}$$

Phas: 5

$$R'_5 = (1 - 2D_5)\ (1 - 2G_{5a})$$

$$= (1 - 360 \times 10^{-6})\ \left|1 - (2)\ (25.49^{*} \times 10^{-9})\right|$$

$$= (0.99964)\ (0.9^{7}49)$$

$$= (0.99964)$$

$*$ = leakage for 3600 hrs $= \left(\frac{30.3 \times 10^{-9}}{4280}\right)\ (3600)$

$$= 25.49 \times 10^{-9}$$

$$R' = (R'_1)\ (R'_2)\ (R'_3)\ (R'_4)\ (R'_5)$$

$$= (0.998847)\ (0.992892)\ (0.980131)\ (0.999928)\ (0.99964)$$

$$= 0.971621$$

Table 2

Mode		1	2A	2B	3A	3B	4	5	Nom	Worst	Best	Mission
Primary Mode	Failure Classification	0.5		0.5		0.5	0.5	0.5				
	Time/Cycles Units: hrs.	0.3		4280		50	720	3600				
	Basic Failure Rate, Units: x 10^{-9}	286		286		286	286	286	286	67300	90	
	Environment/ Application Factor	1000		1		1	1	1				
	Actual Failure Rate, Units: x 10^{-9}	286000		286		286	286	286				
	Failure Rate Source Code								A	(e)	(d)	
	Probability of Failure, x 10^{-9}	42900		612040		7150	102960	514800				
Secondary Mode	Failure Classification		1		1							
	Time/Cycles Units: cycles		1		1							
	Basic Failure Rate, Units: x 10^{-9} cycles		420		420				420	995	38	
	Environment/ Application Factor		5		10							
	Actual Failure Rate, Units: x 10^{-9} cycles		2100		4200							
	Failure Rate Source Code								(a)	(c)	(b)	

Table 2 Continued

	1	2A	2B	3A	3B	4	5	Nom	Worst	Best	Mission
Probability of Failure x 10^{-9}		2100		4200							
Failure Classification		1		1							
Time/Cycles Units: cycles		1		1							
Basic Failure Rate, Units: x 10^{-9} cycles		280		280				280	665	25	
Environment/ Application Factor		5		10							
Actual Failure Rate Units: x 10^{-9} cycles		1400		2800							
Failure Rate Source Code								(a)	(c)	(b)	
Probability of Failure x 10^{-9}		1400		2800							
Probability of Failure All Modes	0^4429	0^535	0^3612	0^57	0^5715	0^3103	0^3515				0^2129
Reliability	9^4571	9^565	9^3388	9^53	9^5285	9^3897	9^3485				9^2871

Notes:

(a) FARADA p. 2.416 Source 136 Wright-Patterson
(A) FARADA p. 2.417 Source 80 General Dynamics
(b) Walter Kidde Report R-1681 "Solenoid Valve Qual. Test"
(c) FARADA p. 2.417 Source 136
(d) FARADA p. 2.413 Source 84 Grumman
(e) FARADA p. 2.417 Source 125 Space-General

Other Modes

Table 3A

Item: Injector, Bipropellant

System/Subsystem: Propulsion (B)

Item Type/Description:

Coaxial, single element injector.

Primary Failure Mode:

Non-uniform combustion pattern.

Possible Causes:

Contamination, damage due to improper handling.

Effects on Subsystem/Mission:

Erosion of combustion chamber which could result in a minor loss of performance or burnthrough and catastrophic failure.

Inherent Preventives:

Adequate filtration, stringent cleanliness and handling procedures.

Table 3B

	2a	3a	Nom	Worst	Best	Mission
Failure Classification	1	1				
Time/Cycles Units: hrs.	0.056	0.167				
Actual Failure Rate, Units: hrs.	0.04572	0.05472	0.04572	0.30456	$0.0^{3}6624$	
Failure Rate Source Code	a, b					
Probability of Failure, x 10^{-}	$0.0^{2}256$	$0.0^{2}763$				0.0102
Reliability	$0.9^{2}744$	$0.9^{2}237$				0.9898

Failure Rate Data Sources:

(a) STL Memo 64-9701.3-127 "Reliability Input for September Surveyor Progress Report "8 October 1964
(b) LEM Descent Engine Component Failure Summary, dated 4-6-65

Notes:

Surveyor	37,320	sec of firing with no applicable failures
LEMDE	17,160	sec of firing with no applicable failures
	54,480	

$$\chi^2_{.50,0} = 1.386$$

$$\chi^2_{.01,0} = 0.0201$$

$$\chi^2_{.99,0} = 9.21$$

$$\lambda = \frac{X2}{2T}$$

$$\lambda_{Nom} = \frac{1.386}{1.09 \times 10^5} = 1.27 \times 10^{-5} \text{ sec} = 0.04572 \text{ hr}$$

$$\lambda_{Best} = \frac{0.0201}{1.09 \times 10^5} = 1.84 \times 10^{-7} \text{ sec} = 0.0006624 \text{ hr}$$

$$\lambda_{Worst} = \frac{9.21}{1.09 \times 10^5} = 8.46 \times 10^{-5} \text{ sec} = 0.30456 \text{ hr}$$

Table 4A

Item: Combustion Chamber and Nozzle

System/Subsystem: Propulsion (B)

Item Type/Description:

Phenolic refrasil ablative liner encased in metal shell (Haynes 25) Bipropellant engine.

Operational Notes:

Liner thickness overdesigned.

Primary Failure Mode:

Material outgassing (plugs injector, weakens material properties which leads to a higher erosion rate).

Effects on Subsystem/Mission:

Minor to catastrophic depending on degree of outgassing.

Inherent Preventives:

Selection of low outgassing materials.

Table 4B

	2A	3A	Nom	Worst	Best	Mission
Failure Classification	1	1				
Time/Cycles Units: hrs	0.056	0.167	0.056	0.333	0	*
Actual Failure Rate, Units: hrs	0.06608	0.06608	0.06608	0.4391	$0.0^{3}96$	
Failure Rate Source Code	a, b					
Probability of Failure, x 10^{-}	$0.0^{2}371$	0.01107				0.01478
Reliability	$0.9^{2}629$	0.98893				0.98522

Failure Rate Data Source:

(a) STL Memo 64-9701.3-127 "Reliability Input for September Surveyor Progress Report" 8 October 1964

(b) LEM Descent Engine Component Failure Summary, dated 4-6-65

Notes:

* Midcourse Phase 2

Surveyor 20,593 sec of firing with no applicable failures

LEMDE 17,160 sec of firing with no applicable failures

37,753

$\chi^2_{0.50,0} = 1.386$ $\qquad \lambda_{Nom} = \frac{1.386}{20.974} = 0.06608$

$\chi^2_{0.01,0} = 0.0201$ $\qquad \lambda_{Best} = \frac{0.021}{20.974} = 0.00096$

$\chi^2_{0.99,0} = 9.21$ $\qquad \lambda_{Worst} = \frac{9.21}{20.974} = 0.4391$

$$\lambda = \frac{\chi^2}{2T}$$

Table 5A

Item: Fittings and Connections

System/Subsystem: Propulsion (B)

Item Type/Description:

*Mechanical connections, not welded or brazed. 2 required.

Primary Failure Mode:

Leakage

Possible Causes:

Contamination and faulty assembly.

Effects on Subsystem/Mission:

Negligible or catastrophic depending on magnitude of leak rate.

Inherent Preventives:

Careful assembly and adequate inspection.

* Welded and brazed fittings are considered to have a negligible problem of failure.

Table 5B

	1	2A	2B	3A	3B	4	5	Nom	Worst	Best	Mission
Failure Classification	0.5	0.1	0.5	0.1	0.5	0.5	0.5				
Time/Cycles Units: hrs.	0.3	0.056	4280	0.167	50	720	3600	0.056	0.333	0	
Basic Failure Rate, Units: $\times 10^{-9}$ hrs.	100	100	100	100	100	100	100	100	710	40	
Environment/ Application Factor	1000	1000	1	1000	1	1	1				
Actual Failure Rate, Units: $\times 10^{-9}$ hrs.	100000	100000	100	100000	100	100	100				
Failure Rate Source Code								(a)	(b)	(c)	
Probability of Failure, $\times 10^{-9}$	15000	560	214000	8850	2500	36000	180000				$.0^{3}459$
Reliability	$0.9^{4}85$	$0.9^{6}44$	$0.9^{3}786$	$0.9^{5}115$	$0.9^{5}75$	$0.9^{4}64$	$0.9^{3}82$				$0.9^{3}541$

Failure Rate Data Sources:

(a), (b), & (c) D. R. Earles, Reliability Engineering Data Series, April 1962

Table 6A

Item: Translation Control

System/Subsystem: Propulsion (B)

Item Type /Description:

Consists of two servo actuators which move the engine support plate in the x and y planes.

Operational Notes:

Used for midcourse and retro propulsion maneuver.

Primary Failure Mode:

Inoperative.

Possible Causes:

Actuator jammed.

Effects on Subsystem/Mission:

Catastrophic.

Backup Provisions:

None.

Inherent Preventives:

Careful checkout of assembly.

Table 6B

	1	2A	3A	Mission
Failure Classification	0. 5*	1	1	
Time/Cycles Units:	0. 3	0. 056	0. 167	
Basic Failure Rate, Units: x 10^{-6} hrs	3. 74	3. 74	3. 74	
Environment/Application Factor	1000	1000	1000	
Actual Failure Rate, Units: x 10^{-6}	3740	3740	3740	
Failure Rate Source Code	a			
Probability of Failure, x 10^{-6}	561	209. 44	624. 58	$0^{2}139$
Reliability	$9^{3}439$	9ˇ791	$9^{3}375$	$9^{2}861$

Failure Rate Data Sources:

a) FARADA Page B-1, Source 179 (Norair Div. of Northrop)

Notes:

* 0. 5 criticality factor used for phase 1 since mechanism is nonoperative at this time and is therefore less susceptible to failure.

4.10 Thermal Louver Mechanisms

This analysis of nine proposed Voyager spacecraft thermal louver mechanisms is based on the current design status, mission profile, mechanism operation, certain necessary assumptions, and available failure rate data. The mission profile is common to all mechanisms and is defined in Table 1.

Table 1. Mission Profile

Mission Phase	Mission Time (hrs)	Environmental K-Factor
1. Lift-off and boost (including injection)	0.3	1000
2. Post-injection through capsule separation		
a. Midcourse maneuver (accomplished during Phase 2b)	0.056	1000
b. Cruise	4,280	1
3. Post-separation (capsule) through post-retro cruise		
a. Retropropulsion	0.022	1000
b. Cruise (after retro)	50	1
4. Mars Orbit	720	1
5. Additional Mars Orbit	3,600	1

Since the configuration is different for each of the nine mechanisms, nine analyses are presented. In each, there is a brief mechanism description and schematic followed by a critical component breakdown, and, finally, mechanism mission reliability calculations and results.

The nine thermal louver mechanisms fall roughly into two classes: those that contain a relatively large number of components and generate relatively large operating forces and those of simple design that generate relatively small operating forces. Those mechanisms generating large forces operate a 3 inch louver and require 72 such louvers to complete the entire thermal control system. Those mechanisms generating small forces operate 2 inch louvers and require 108 such louvers to complete the entire thermal control system. It is assumed that the 3 inch louver thermal control system allows as many as two louvers to fail and still achieve mission success, while the 2 inch system allows as many as three louvers to fail and still achieve mission success. Those mechanisms generating large forces utilize a spring and sliding seal on the ends of their louvers. These sliding seals fit flush against the spacecraft frame. The mechanisms generating small forces do not utilize sliding end seals since they probably would have difficulty overcoming any stiction developed between the seal and the spacecraft frame.

This reliability analysis is not of sufficient refinement to include performance considerations. Thus, a paradox develops in the reliability calculations. Those mechanisms generating large enough forces to insure good performance are penalized in the reliability calculations because they require more parts, while those mechanisms generating small forces tend to be assigned relatively higher reliabilities even though their performance may be at a lower level. The assumption is made, for the purposes of this analysis, that all mechanisms yield adequate performance to assure mission success if they operate as designed.

4.10.1 Wax Filled Thermal Actuation with Rack and Pinion

This mechanism consists of a thermal actuator, overshoot spring, rack and pinion, louver, torsional return bar, and sliding end seals (Figure 1). The thermal actuator consists of a case, plunger, Teflon diaphragm, and wax fill. The sliding end seals consist of a seal and a small spring.

The thermal actuator, riding on the overshoot spring, is partially embedded in a thermal sink that serves as a mount for the heat generating electronic equipment. As the electronic equipment looses heat to the sink, the thermal actuator is also affected. A characteristic of the thermal actuator wax fill is that in the solid and liquid states, its volumetric

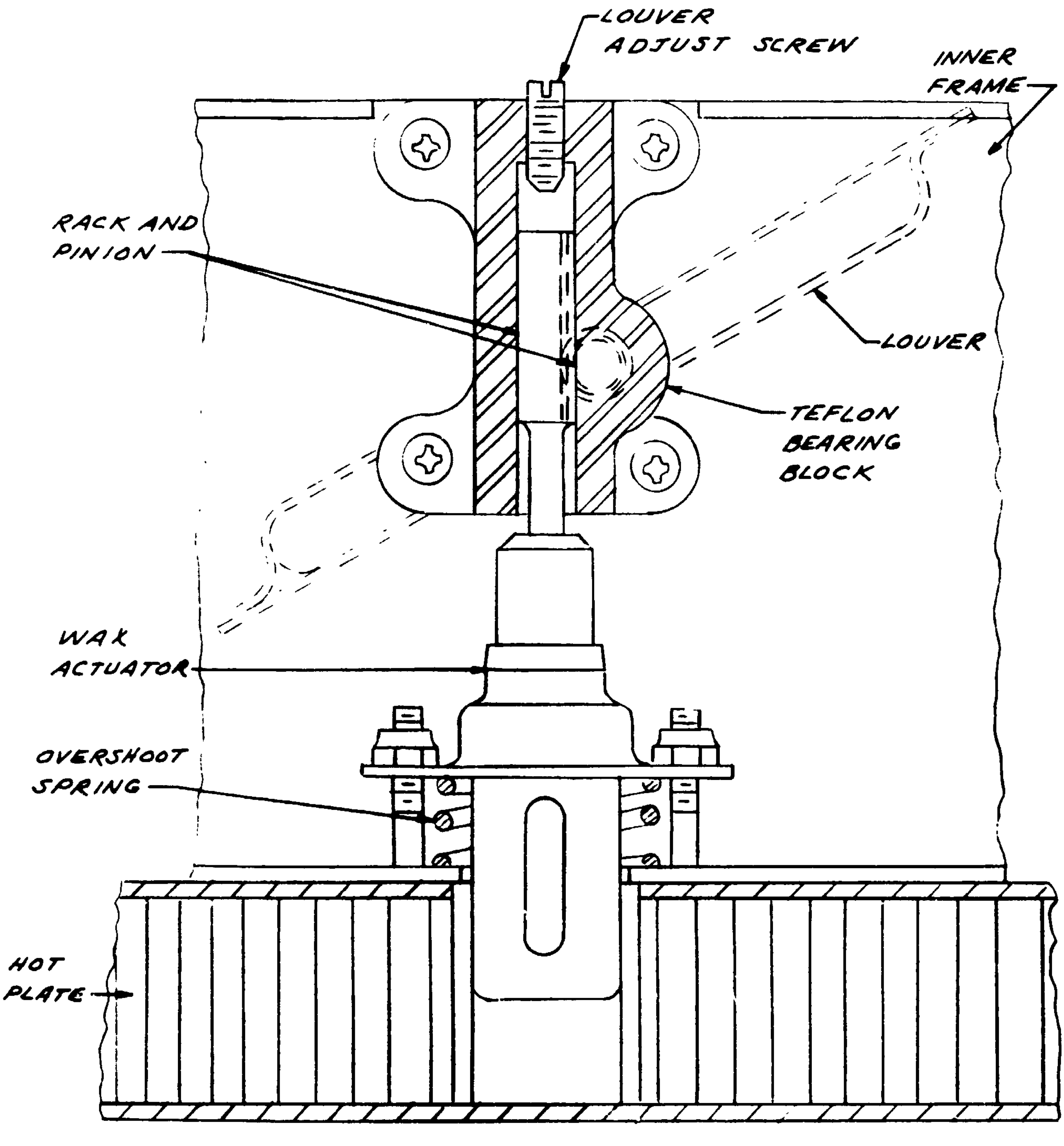

LOUVER
ADJUST SCREW
INNER
FRAME
RACK AND
PINION
LOUVER
TEFLON
BEARING
BLOCK
WAX
ACTUATOR
OVERSHOOT
SPRING
HOT
PLATE

Figure 1.

increase with temperature rise is small; however, in the change of phase between solid and liquid, volume increases greatly with a relatively small temperature increase. It is around this temperature range that the mechanism is designed to operate. The louvers remain closed until the heat sink and thermal actuator wax fill reach the temperature at which the wax begins its change of phase from a solid to a liquid. At this temperature the wax experiences a large volume increase. This increase in volume is transformed to louver opening motion, against torsional return bar force, through the thermal actuator piston, Teflon diaphragm, and the rack and pinion. With increasing temperature the thermal actuator piston is driven against the louver adjust screw stop. At this position the louver is fully opened. Any further increase in wax volume and associated piston travel after the piston is driven against the louver adjust screw is absorbed by compressing the overshoot spring and driving the thermal actuator further into the heat sink well. If the thermal actuator were embedded solidly in the heat sink with no overshoot spring, further increase in temperature after the piston had been driven against the louver adjust screw stop could cause serious structural damage to the spacecraft. As the louver is opened, the heat sink is exposed to low space temperatures and is cooled. Decreases in heat sink temperature cause the thermal actuator wax fill to contract allowing the torsional return bar to close the louver.

The louver end sliding seals are merely a device to increase efficiency by decreasing "gap fraction" (the space between the louver ends and the spacecraft frame). The sliding seals are forced against the spacecraft frame by small compressed springs. Since the thermal actuator can apply in the order of 50 lbs force to its piston, it is assumed that any stiction forces between the sliding seals and the spacecraft frame can be easily overcome.

Table 2

Louver Mechanism Component and Handbook Component	Lower Limit Failure Rate x 10^{-6} and Source ( )
Overshoot Spring Spring, Simple Return Force	0.001 (1)
Actuator Diaphragm Diaphragm	0.1 (1)
Rack and Pinion Gears	0.002 (1)
Teflon Bearing Block Bearing, Ball, Light Duty	0.035 (1)
Torsional Return Spring (Twist Bar) Spring, Simple Return Force	0.001 (1)
Bearing, Sleeve Bearing, Translatory, Sleeve	0.210 (1)
Gap Fraction Sliding Seal Spring, Simple Return Force	*
TOTAL MECHANISM FAILURE RATE	0.147 x 10^{-6} hrs

* Failure of this component is not considered critical since the result would not be mechanism failure but only decreased efficiency.

(1) Reliability Engineering Data Series, Failure Rates, April 1962, AVCO Corporation, Research and Advanced Development Division.

Necessary data for reliability calculations is as follows:

a. Mission phase time and environmental K-factors from Table 1.

b. Total failure rate from Table 2.

c. Number of mechanisms required to complete the system = 72.

Reliability formula for each mechanism:

$$R_m = \prod_{i=1}^{5} R_i$$

$$R_i = e^{-\lambda K t_i}$$

where:

R_m = mechanism mission reliability

R_i = mechanism mission phase reliability

λ = total mechanism failure rate

K = environmental factor

t = mission phase time

Calculation:

$$R_1 = e^{-0.147 \times 10^{-6}[(1000)(0.3)]} = 0.9^4 56$$

$$R_2 = e^{-0.147 \times 10^{-6}[(1000)(0.056) + 1(4,280)]} = 0.9^3 363$$

$$R_3 = e^{-0.147 \times 10^{-6}[(1000)(0.022) + 1(50)]} = 0.9^4 89$$

$$R_4 = e^{-0.147 \times 10^{-6}[(1)(720)]} = 0.9^3 894$$

$$R_5 = e^{-0.147 \times 10^{-6}[(1)(3600)]} = 0.9^3 471$$

$$R_m = 0.9^2 8673$$

Reliability formula for total system:

$$R_s = \prod_{j=1}^{5} R_{sj}$$

$$R_{sj} = R_i^{n} + nR_i^{n-1}(1-R_i) + \frac{n(n-1)}{2!}R_i^{n-2}(1-R_i)^2 + \frac{n(n-1)(n-2)}{3!}R_i^{n-3}(1-R_i)^3$$

where:

R_s = system mission reliability

R_{sj} = system mission phase reliability

R_i = mechanism mission phase reliability

n = number of mechanisms in the system

NOTE: If n = 108 the formula (R_{sj}) is applied through the fourth term. If n = 72 the formula (R_{sj}) is applied through the third term.

Calculations:

$$R_{s1} = (R_1)^{72} + 72(R_1)^{71}(1-R_1)^1 + \frac{72(71)}{2}(R_1)^{70}(1-R_1)^2$$

$$= (0.9^456)^{72} + 72(0.9^456)^{71}(0.000\ 044) + 2556(0.9^456)^{70}(0.000\ 044)^2$$

$$= 0.9^26836 + 0.003\ 153 + 0.000\ 004 \quad = 0.9^58$$

$$R_{s2} = 0.955170 + 0.043835 + 0.000\ 991 \quad = 0.9^567$$

$$R_{s3} = 0.999\ 208 + 0.000\ 791 + 0.000\ 000 \quad = 0.9^6$$

$$R_{s4} = 0.992\ 396 + 0.007\ 574 + 0.000\ 028 \quad = 0.9^58$$

$$R_{s5} = 0.962\ 627 + 0.036\ 683 + 0.000\ 687 \quad = 0.9^57$$

$$R_s = (0.9^58)(0.9^56)(0.9^58)(0.9^57) \quad = 0.9^488$$

4.10.2 Wax Filled Thermal Actuator with Cable and Pulley Drive

This mechanism is identical to that of 4.10.1 except that a cable and pulley arrangement is used in place of a rack and pinion (Figure 2). The operation is identical.

The failure rate in source (1) for cables is identical to that of gears. Thus the total mechanism failure rate is the same as in 4.10.1.

The necessary data for reliability calculations, the mechanism mission reliability calculations, and the system mission phase reliability calculations are the same as in 4.10.1. All results are also identical.

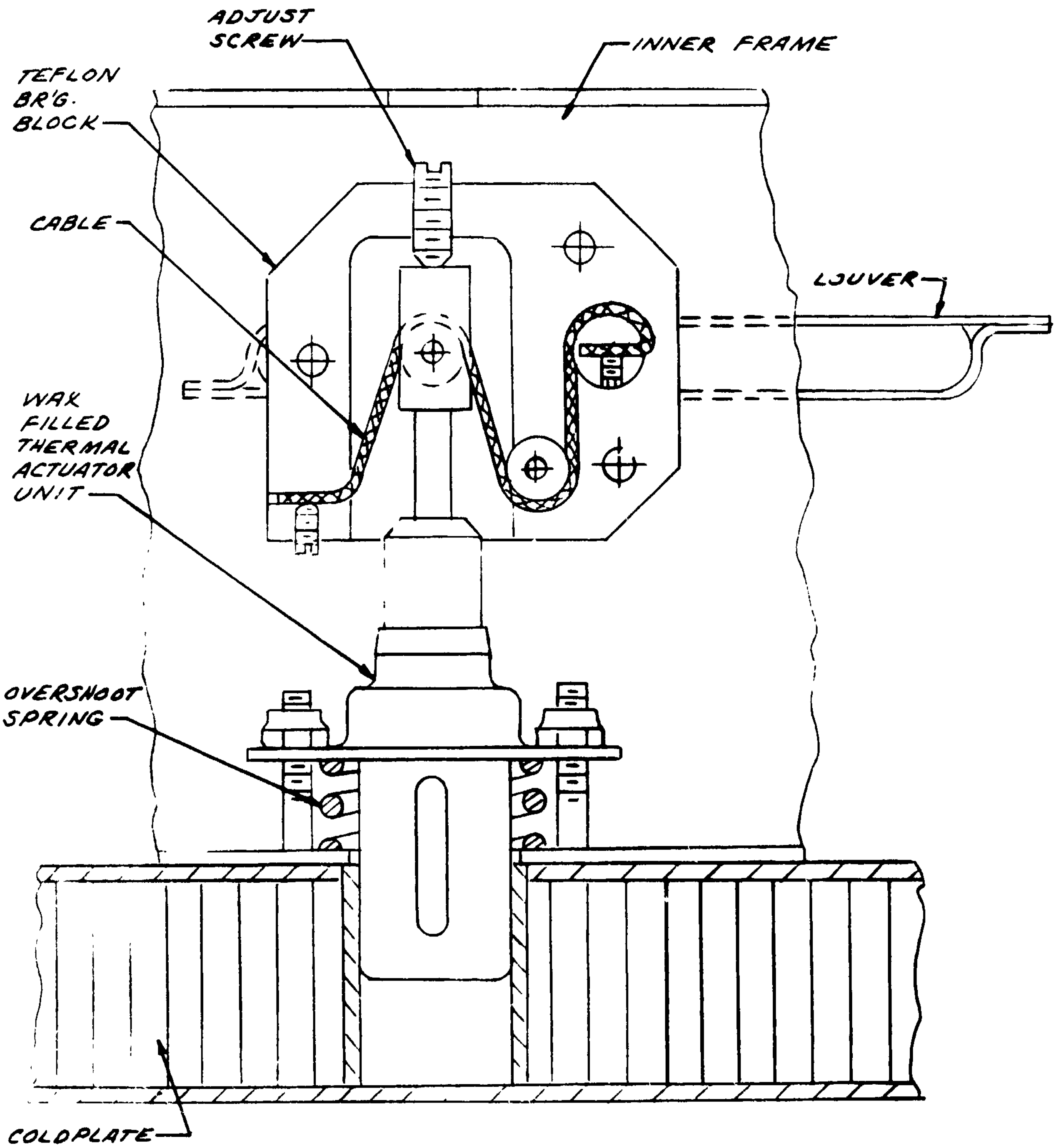

ADJUST SCREW
INNER FRAME
TEFLON BR'G. BLOCK
CABLE
LOUVER
WAX FILLED THERMAL ACTUATOR UNIT
OVERSHOOT SPRING
COLDPLATE

Figure 2.

4.10.3 The Bimetal Louver System

This system consists of 216 bonded bimetal strip mechanisms partially embedded in the heat sink that serves the electronic heat generating equipment (Figure 3). The strips are set at 1 inch intervals. When the electronic gear is cold, the expansion properties of the bimetal strips are such that adjacent metal strips bend over to cover the heat sink surface. As heat to the heat sink increases, the adjacent bimetal strips tend to straighten, exposing the sink surface to the colder space environment.

The mechanism consists of two dissimilar metals and the bond holding them together. There is no failure rate data available on these three items. There is no experimental test data on this mechanism available at this time. Thus this mech ..sm cannot be assigned a quantitative reliability prediction number at this time.

4.10.4 The Wax Filled Bellows Actuator

This mechanism consists of a wax filled bellows actuator, mechanical gearing, torsional return bar, and sliding end seal (Figure 4). The mechanical gearing is assumed to be equivalent to the rack and pinion of 4.10.1. The mechanism operation is similar to that of 4.10.1 except that there is no overshoot spring and the actuator is in a stationary mount on the heat sink.

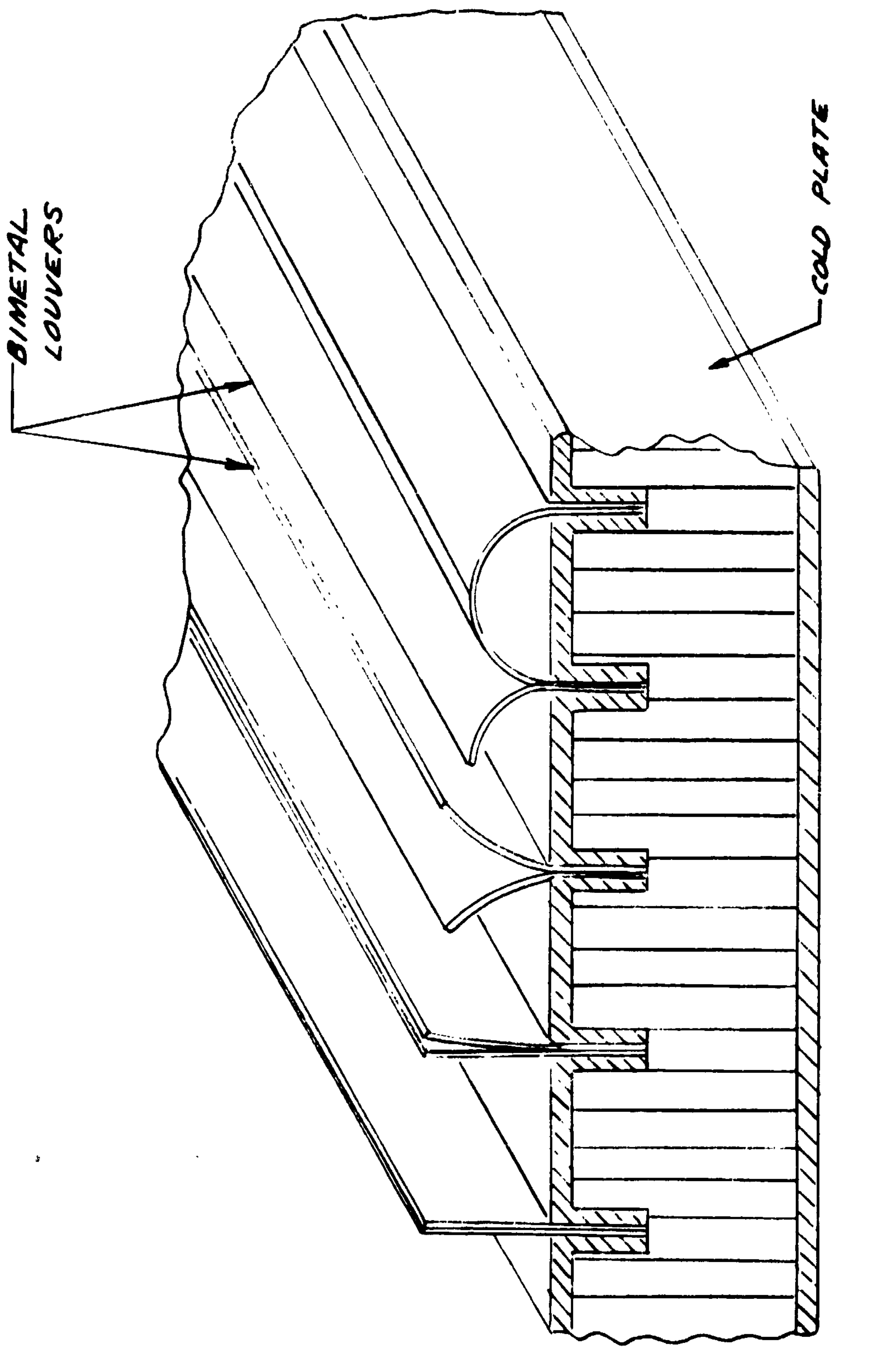


Figure 3.

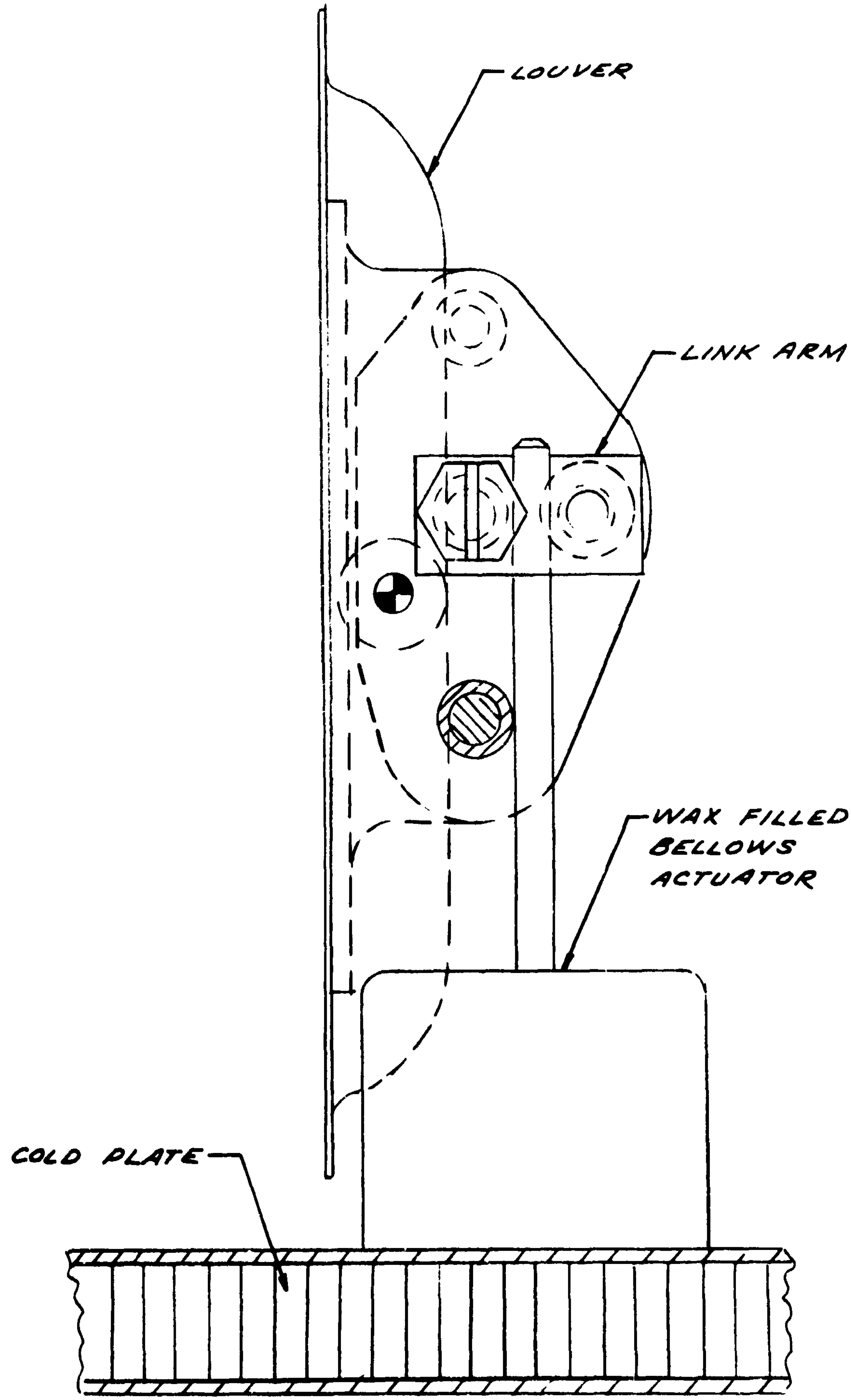


Figure 4.

Table 3

Louver Mechanism Component and Handbook Component	Lower Limit Failure Rate x 10^{-6} and Source
Wax Filled Bellows Actuator Bellows	0.090 (1)
Link Arm (Rack and Pinion) Gears	0.002 (1)
Bearing Block Bushings	0.020 (1)
Torsional Return Spring (Twist Bar) Spring, Simple Return Force	0.001 (1)
Bearing, Sleeve Bearing, Translatory, Sleeve	0.008 (1)
Gap Fraction Sliding Seal Spring, Simple Return Force	*
TOTAL MECHANISM FAILURE RATE	0.121 x 10^{-6}

* Failure of this component is not considered critical since the result would not be mechanism failure but decrease efficiency.

(1) Reliability Engineering Data Series, Failure Rates, April 1962, AVCO Corporation, Research and Advanced Development Division.

Necessary data for reliability calculations is as follows:

a. Mission phase time and environment K-factors from Table 1.

b. Total failure rate from preceding table.

c. Number of mechanism required to complete the system = 72.

Reliability formulas for the mechanism and the system are the same as in 4.10.1.

Calculation results:

$R_1 = 0.9^4 64$

$R_2 = 0.9^3 475$

$R_3 = 0.9^5 1$

$R_4 = 0.9^4 13$

$R_5 = 0.9^3 564$

$R_m = 0.9^2 8907$

$R_{s1} = 0.9^6$

$R_{s2} = 0.9^5 7$

$R_{s3} = 0.9^6$

$R_{s4} = 0.9^5 8$

$R_{s5} = 0.9^5 7$

$R_s = 0.9^5$

4.10.5 The Freon Filled Bellows Actuator

The only difference between this mechanism and the preceding mechanism is the actuator material (Figure 5). This mechanism utilizes Freon where the other uses wax. However, this difference is not reflectable in this reliability analysis and the results for this mechanism are therefore identical.

4.10.6 Spiral Bimetal Actuator

The spiral bimetal actuator mechanism consists of a bearing assembly, bimetal actuator spiral spring, and a louver all attached to the heat sink by an inner frame (Figure 6). Temperature effects on the bimetal spring cause it to flex, thereby opening and closing the louver. This mechanism develops a torque of about 0.01 inch-pound.

The bimetal actuator spring is similar to that used on the Pioneer, while the louver is similar to that used in the OGO.

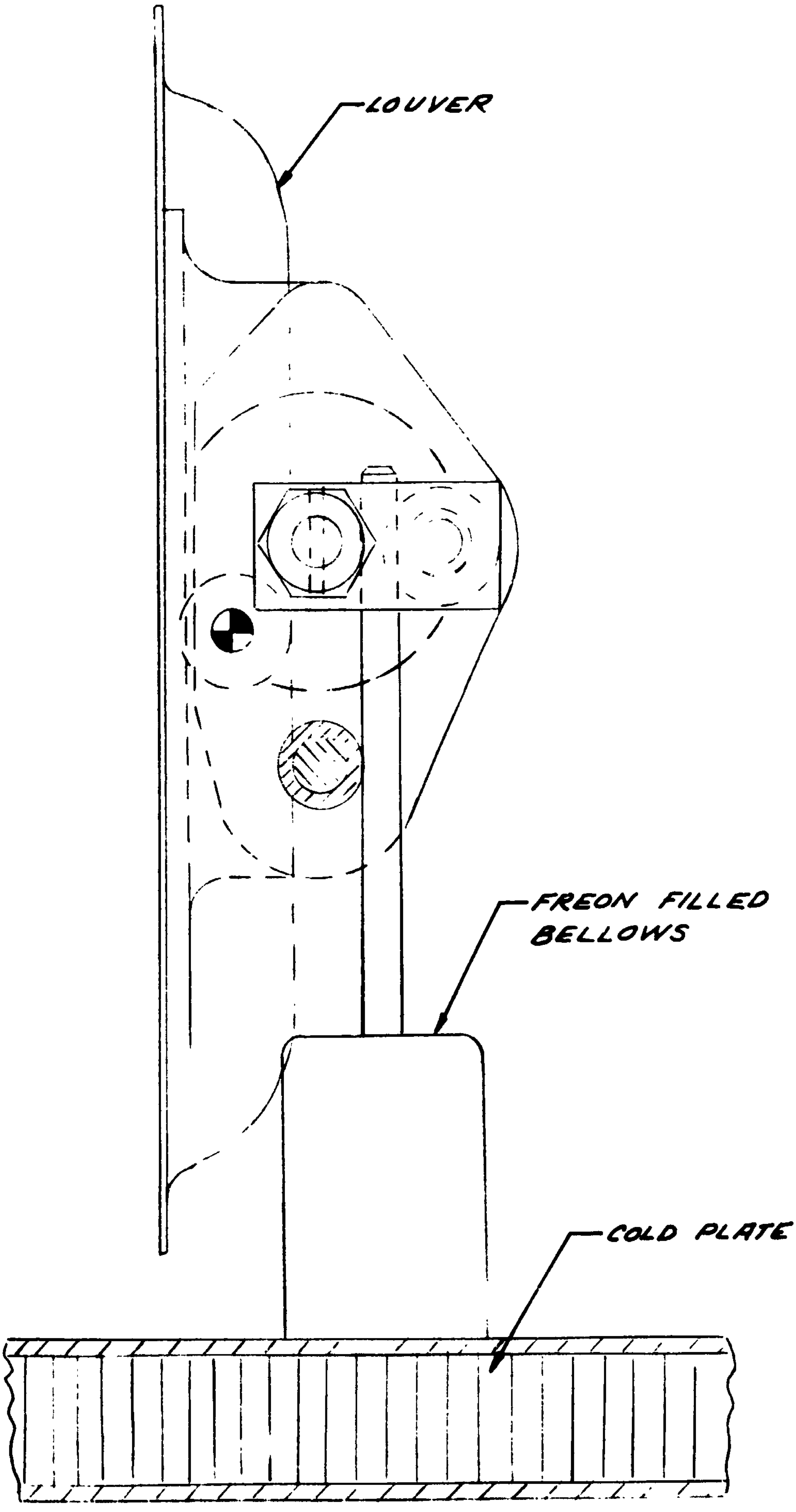


Figure 5.

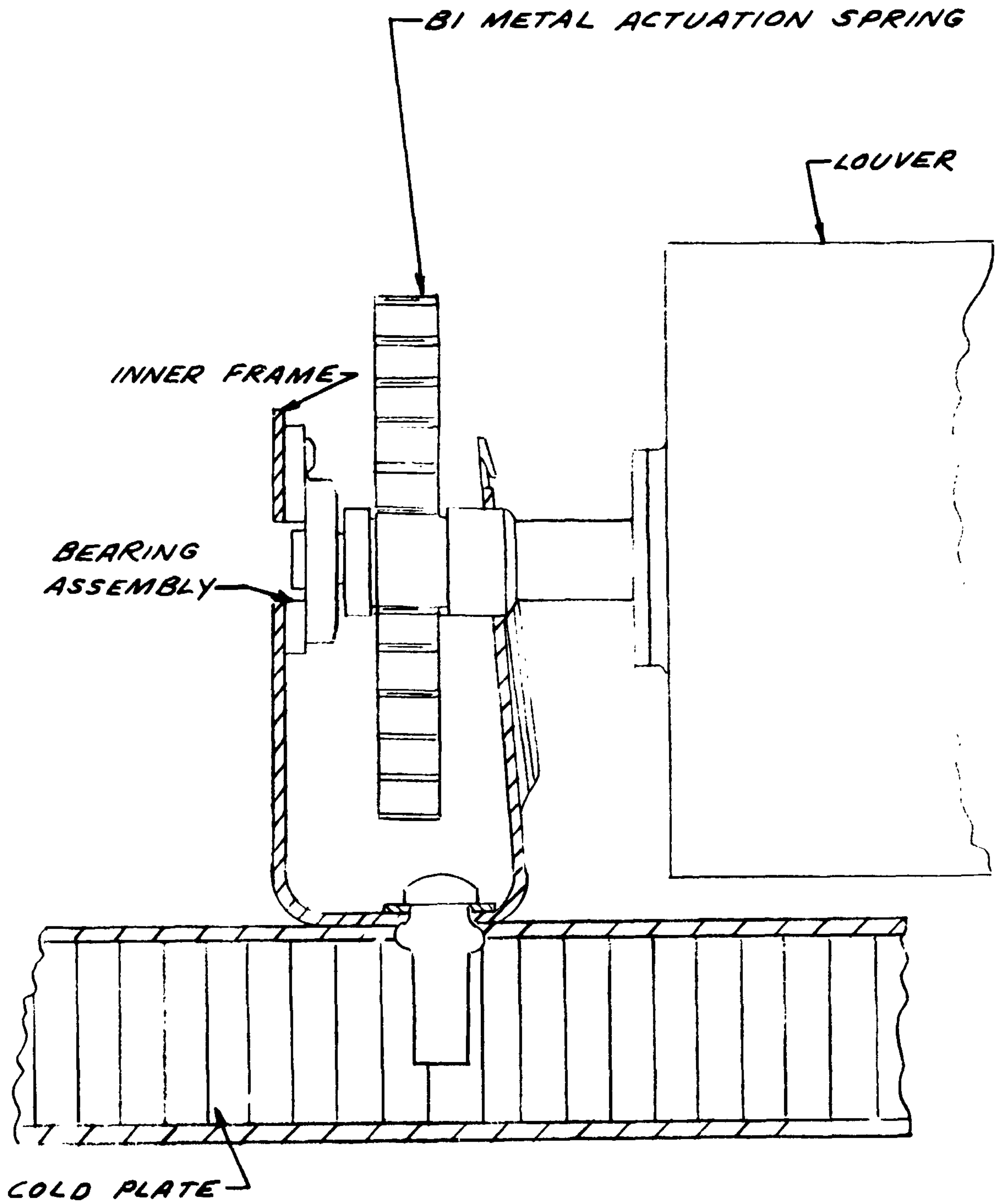

BI METAL ACTUATION SPRING
LOUVER
INNER FRAME
BEARING
ASSEMBLY
COLD PLATE

Figure 6.

Test data on the similar projects are used in the reliability calculations. Test results are given in the following report:

> Test Report 2321-6012-TU-000, "Orbiting Geophysical Observatory Temperature Control Louvers Life Test", (OGO-V21-25), 23 April 1963.

In these tests there were 510,000 individual louver cycles with no failures. The mechanism failure rate can be calculated:

$$Q = 1 - R,$$

where

Q = the mechanism failure rate/cycle

R = the mechanism reliability/cycle

Reliability (R) can be calculated:

$$R^n = 1 - \gamma,$$

where

n = the number of test cycles = 510,000

γ = the desired confidence level (50% confidence level is used here)

Thus

$$R^n = 1 - \gamma$$

$$R^n = 0.5$$

$$\ln R = \frac{\ln 0.5}{n} = \frac{-0.693}{0.51 \times 10^6} = -1.36 \times 10^{-6}$$

$$R = e^{-1.36 \times 10^{-6}} = e^{-\lambda c}$$

where

λ = failure rate in cycles

c = 1 cycle

Since

$$Q = 1 - R$$

$$Q = 1 - e^{-1.36 \times 10^{-6}} = 1 - e^{-\lambda}$$

but for small exponents $Q \approx \lambda$

Thus

$$Q \approx \lambda = 1.36 \times 10^{-6} \approx 1.4 \times 10^{-6}$$

The failure rate is thus calculated to be $1.4 \times 10^{-6} \frac{\text{Failures}}{\text{Cycle}}$

It is estimated that the louver system will be subjected to 500 operative cycles during the mission. For the purposes of this analysis, it is assumed that the mission phase cyclic louver operation will be proportional to the mission phase duration. Thus the operative cycles assigned for each phase is as follows:

Phase	Operative Louver Cycles
1	0
2	247
3	3
4	42
5	208

Using $\lambda = 1.4 \times 10^{-6} \frac{\text{failures}}{\text{cycle}}$ and $R = e^{-\lambda c}$ where c = cycles/ phase as above, mechanical mission phase reliability is calculated to be

$R_1 > 0.9^6$

$R_2 = 0.9^3 6542$

$R_3 = 0.9^5 58$

$R_4 = 0.9^4 412$

$R_5 = 0.9^3 7088$

$R_m = 0.9^3 3$

System mission phase reliability calculations:

$R_{s1} > 0.9^6$

$R_{s2} = 0.9^4 88$

$R_{s3} = 0.9^6$

$R_{s4} = 0.9^6$

$R_{s5} = 0.9^6$

$R_s = 0.9^4 84$

4.10.7 Wax Filled Bourdon Tube Actuator

This mechanism consists of two U-shaped wax-filled bourdon tubes and a louver (Figure 7). One leg of each tube is attached to the heat sink and the other is attached to the louver. The wax has the characteristics previously described. Thus as the temperature rises the wax expands, tending to open the curved tubes and attached louver.

Necessary data for reliability calculations is as follows:

a. Mission phase time and environmental K-factors from Table 1.

b. Total failure rate from Table 4.

c. Number of mechanisms required to complete the system = 108.

Mechanism mission phase reliability calculations:

$R_1 = 0.9^{5}4$

$R_2 = 0.9^{4}09$

$R_3 = 0.9^{5}8$

$R_4 = 0.9^{4}85$

$R_5 = 0.9^{4}24$

$R_m = 0.9^{3}81$

System mission phase reliability calculations:

$R_{s1} = 0.9^{6}$

$R_{s2} = 0.9^{6}$

$R_{s3} = 0.9^{6}$

$R_{s4} = 0.9^{6}$

$R_{s5} = 0.96$

$R_s = 0.9^{5}5$

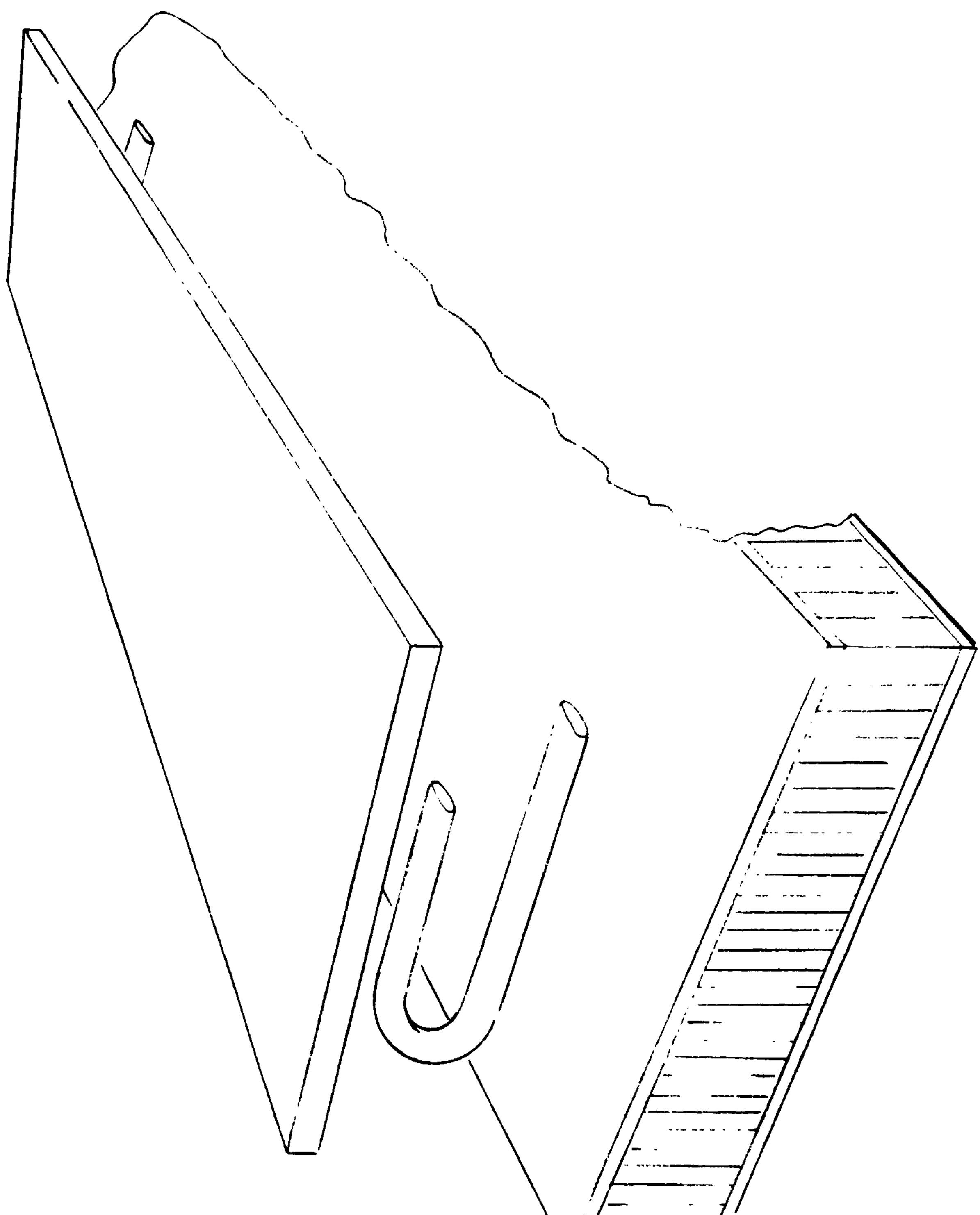


Figure 7.

Table 4

Louver Mechanism Component and Handbook Component	Failure Rate x 10^{-6} hrs.
Wax-Filled Bourdon Tube (2)	
*Joints Welded (4) 4 x 0.00537 =	0.021 (2)
Total Mechanism Failure Rate	0.021×10^{-6}

* The assumed failure mode for this mechanism is wax extruding from the point where the bourdon tube was filled and sealed.

(2) The Sippican Corporation, Sippican Report FA-A202234-B, Revised 1 May 1961.

4.10.8 Gas Filled Bourdon Tube Actuator

This mechanism is similar to that of 4.10.7 except that a gas is used here instead of wax a. d a sensor-expander with connecting tubing has been added to achieve more heat sink temperature test points (Figure 8). The mechanism operation is the same.

Table 5

Louver Mechanism Component and Handbook Component	Failure Rate x 10^{-6} and Source ( )
Gas Filled Bourdon Tube (2) with Sensor-Expander and Connecting Tubing	
*Joints Welded (10) 10 x 0.00537 =	0.054×10^{-6} (2)
Total Mechanism Failure Rate	0.054×10^{-6}

* The assumed failure mode for this mechanism is gas leaking from a joint or sealing.

(2) The Sippican Corporation, Sippican Report FA-A202234-B, Revised 1 May 1961.

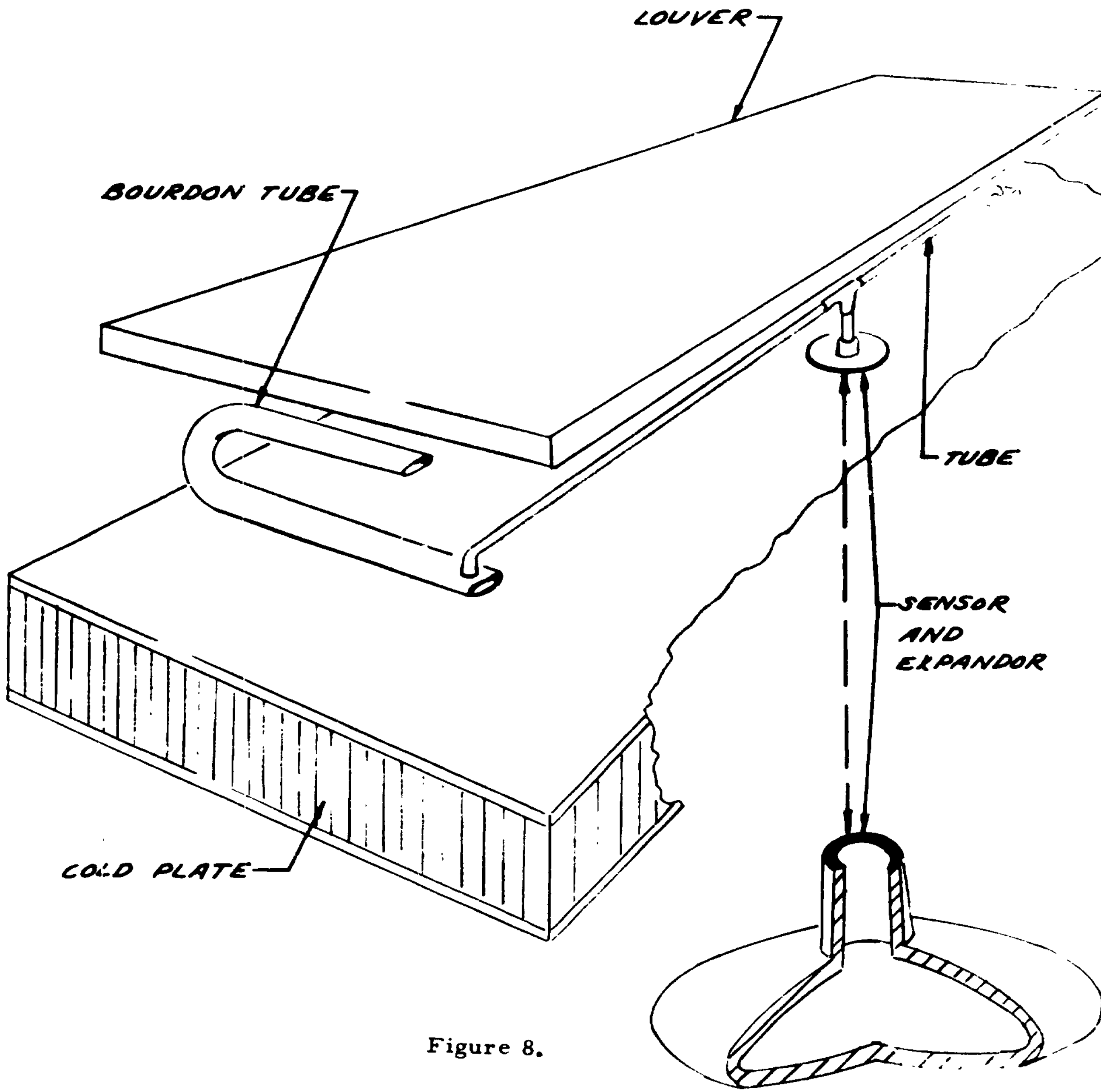


Figure 8.

Necessary data for reliability calculations are as follows:

a. Mission phase time and environmental K-factors from Table 1.

b. Total failure rate from preceding table.

c. Number of mechanisms required to complete the system = 108.

Mechanism mission phase reliability calculations:

$R_1 = 0.9^4 85$

$R_2 = 0.9^3 766$

$R_3 = 0.9^5 6$

$R_4 = 0.9^4 61$

$R_5 = 0.9^3 806$

$R_m = 0.9^3 513$

System mission phase reliability calculations:

$R_{s1} = 0.9^6$

$R_{s2} = 0.9^6$

$R_{s3} = 0.9^6$

$R_{s4} = 0.9^6$

$R_{s5} = 0.9^6$

$R_s = 0.9^5 5$

4.10.9 The Bimetal Helix Actuator

The bimetal helix actuator consists of a bimetal spring and a louver (Figure 9). One end of the bimetal helix is attached to the heat sink while the other is attached to the louver. Temperature effects on the bimetal helix cause it to flex, thereby opening and closing the louver.

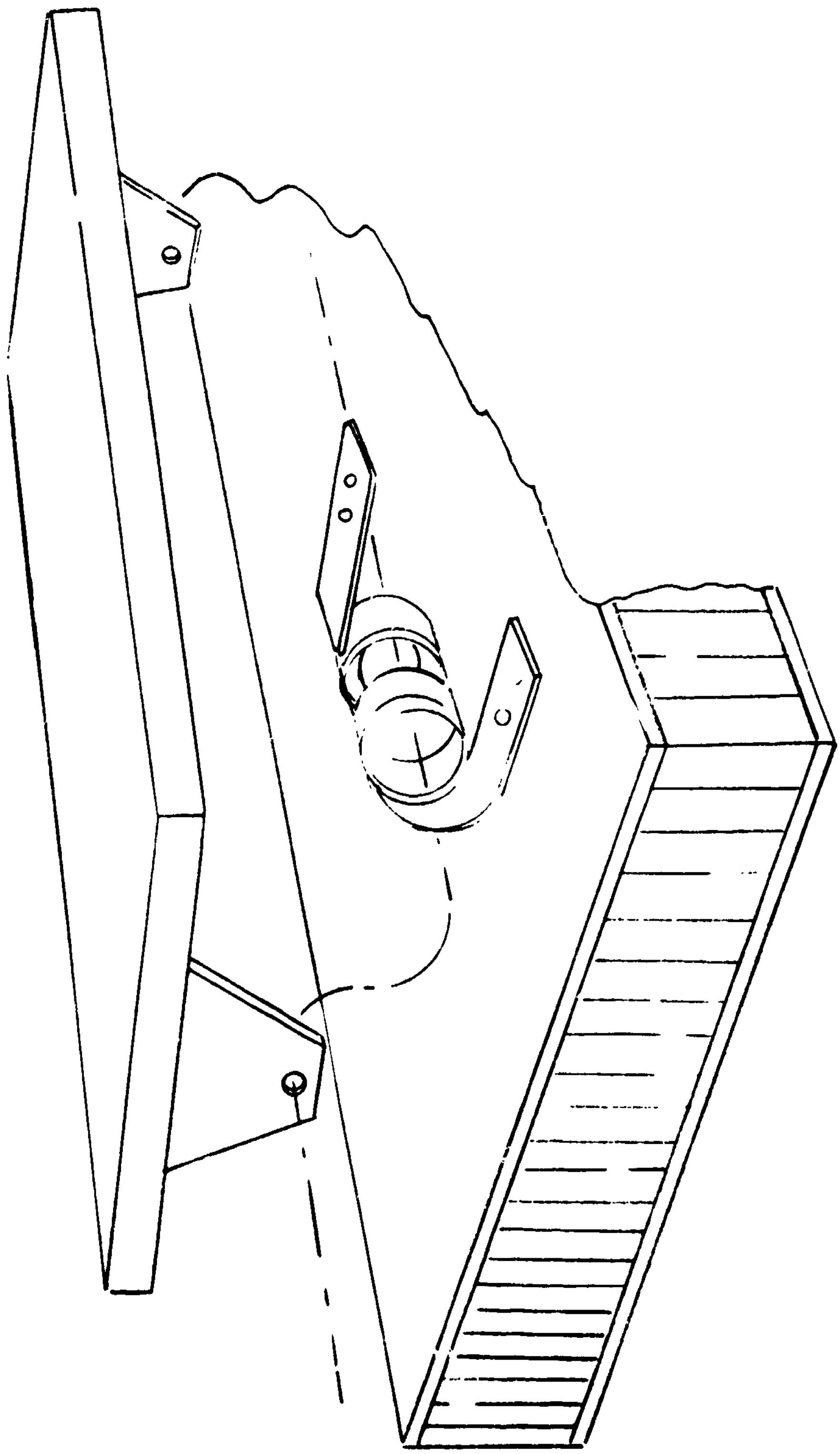


Figure 9.

No available handbook failure rate data exists on a bimetal helix. There is no apparent applicable test data available on a bimetal helix. It is noted, however, that similarities exist between this mechanism and the spiral bimetal actuator. Therefore, as an estimate, the mechanism reliability is assumed to be identical to that of 4.10.6.

4.11 Strip Heaters

This analysis of the Voyager strip heater system is based on the current design status, mission profile, system operation, certain necessary assumptions, and available failure rate data. Following the mission profile and a brief system description, a section is devoted to mission success criteria and calculated results. The calculated results follow directly from mission success criteria and individual component analyses.

For reliability calculations, the mission profile is assumed to consist of the following phases:

	Mission Phase	Mission Time (hrs)	Environmental K-Factor
1.	Lift-off and boost (including injection)	0.3	1000
2.	Post-injection thru capsule separation		
	a. Midcourse maneuvers (accomplished during phase 2b)	0.056	1000
	b. Cruise	4,280.000	1
3.	Post-separation (capsule thru post-retro cruise)		
	a. Retropropulsion	0.022	1000
	b. Orbit (after retro)	50.000	1
4.	Mars Orbit (one month)	720.000	1
5.	Additional Mars Orbit (5 months)	3,600.000	1

The Voyager spacecraft strip heater system contains six independent strip heaters located as follows, one each on the

- 3 ft. dish antenna gimbal
- 6 ft. dish antenna gimbal
- POP gimbal
- POP package
- each of two external experimental packages

Each strip heater is comprised of one TWR standard strip heater element and one TRW standard hermatically sealed normally closed thermostatic switch as described in the component analysis. This strip heater design is being used successfully on the OGO spacecraft program.

Power is applied to the heaters early in the mission and is continuously available throughout. Heat generated by a strip heater is controlled by its associated thermostatic switch. When the desired preset temperature is obtained, the normally closed contacts of the switch separate thereby opening the circuit. When the component temperature falls below the preset value the contacts again close thereby reactivating the heater.

The following success criteria have been established:

(1) If a heater fails "on" (ie. the contacts do not open) near earth, the system will fail due to overheating.

(2) If a heater fails "on" near Mars, the system will not fail since it would normally be in near continuous operation due to the extreme cold.

(3) If the heater fails "off" (open circuit) the system fails.

For the purposes of this analysis the separation point between criterion (1) and (2) is arbitrarily assumed to occur midway into the first spacecraft cruise (after 2, 140 hrs). The phase and mission reliabilities for one strip heater are calculated on Tables 1 and 2.

Phase	Component Phase Reliabilities	Strip Heater Mission Phase Reliability
1	(0.9^57) (0.9^57)	0.9^54
2	$(0.9^6)(0.9^457)(0.9^6)(0.0^479)$	0.9^436
3	$(0.9^6)(0.9^6)(0.9^6)(0.9^6)$	0.9^56
4	$(0.9^53)(0.9^6)$	0.9^52
5	$(0.9^464)(0.9^57)$	0.9^461
	Strip Heater Mission Reliability	0.9^3879

The corresponding calculated reliabilities for the strip heater system (six independent strip heaters) is as follows:

Phase	Strip Heater Mission Phase Reliability	Strip Heater System Mission Phase Reliability
1	0.9^54	0.9^464
2	0.9^436	0.9^3616
3	0.9^56	0.9^476
4	0.9^52	0.9^452
5	0.9^461	0.9^3766
	Mission Reliability	0.9^3274

Table 1A

MECHANICS DIVISION RELIABILITY STAFF

RELIABILITY ANALYSIS FORM

SIDE 1: QUALITATIVE DATA

TRW SPACE TECHNOLOGY LABORATORIES

PROGRAM/PROJECT: VOYAGER SPACECRAFT	PREPARED BY: R.A. PAUL[illegible]	DATE: [illegible]
ITEM: HEATER, STRIP, FLEXIBLE (STL STANDARD)	SYSTEM/SUBSYSTEM: STRIP HEATER SYSTEM	
PART NO: PT4-170[illegible]	MANUFACTURER: —	I.D. NO.: —

GENERAL DATA

ITEM TYPE/DESCRIPTION: HEATER, STRIP FLEXIBLE AS DESCRIBED IN STL SPECIFICATION [illegible]

OPERATIONAL NOTES: AS PER SECTION II

ENVIRONMENTAL NOTES: AS PER SECTION I

PRIMARY FAILURE MODE

PRIMARY FAILURE MODE: OPEN CIRCUIT (BROKEN OR BURNED WIRE, FAULTY SOLDER JOINTS) HEATER BECOMES UNBONDED FROM THE PART TO BE WARMED.

POSSIBLE CAUSES: MECHANICAL STRESSES, EXCESSIVE POWER

EFFECTS ON SUBSYSTEM/MISSION: SYSTEM FAILURE

BACKUP PROVISIONS: NONE

INHERENT PREVENTIVES: PROVEN DESIGN

SECONDARY FAILURE MODE

SECONDARY FAILURE MODE

POSSIBLE CAUSES

EFFECTS ON SUBSYSTEM/MISSION

BACKUP PROVISIONS

INHERENT PREVENTIVES

OTHER FAILURE MODES

OTHER FAILURE MODES

POSSIBLE CAUSES

EFFECTS ON SUBSYSTEM/MISSION

BACKUP PROVISIONS

INHERENT PREVENTIVES

REMARKS

Table 1B

	1	2a	2b	3a	3b	4	5	Mission
Failure Classification	C	C	C	C	C	C	C	C
Time/Cycles Units: hrs	0.3	0.056	4280	0.022	50	720	3600	
Basic Failure Rate, Units: x 10^{-6} hrs	0.01	0.01	0.01	0.01	0.01	0.01	0.01	
Environment/Application Factor	1000	1000	1	1000	1	1	1	
Actual Failure Rate, Units: x 10^{-6} hrs	10	10	0.01	10	0.01	0.01	0.01	
Failure Rate Source Code	(1)	(1)	(1)	(1)	(1)	(1)	(1)	
Probability of Failure, x 10^{-6}	3	0.56	42.80	0.22	0.5	7.2	36	90
Probability of Failure, All Modes x 10^{-6}	3	0.56	42.8	0.22	0.5	7.2	36	90
Reliability	$0.9^{5}7$	0.9^{6}	$0.9^{4}57$	0.9^{6}	0.9^{6}	$0.9^{5}3$	$0.9^{4}64$	$0.9^{4}10$

Failure Rate Data Sources: (1) Reliability engineering data series failure rates, April 1962, AVCO Corporation*

Notes:

C: Critical failure (system failure)

*** Since this strip heater is a proven successful design, the lower limit generic failure rate for heater elements is utilized here.**

Table 2A

Item: Switch, Thermostatic (TRW Standard)

System/Subsystem: Strip Heater System

Item Type/Description:

Switch, thermostatic, SPST, snap acting, hermetically sealed, normally closed.

Primary Failure Mode:

Fail closed.

Possible Causes:

Oxidation of contacts, pitted contacts - arcing, insulation breakdown, shorts.

Backup Provisions:

None.

Inherent Preventives:

Proven design.

Secondary Failure Mode:

Fail open (open circuit)

Possible Causes:

Vibration, shock.

Effects on Subsystem/Mission:

System failure.

Backup Provisions:

None.

Inherent Preventives:

Proven design.

Table 2B

Mode		1	2a	2b	3a	3b	4	5	Mission
Primary Mode	Failure Classification	C	M	C*	M	M	M	M	C*
	Time/Cycles Units: hrs.	0.3	0.056	2140 2140	0.022	50	720	3600	
	Basic Failure Rate, Units: $\times 10^{-6}$ Hrs**	0.0082	0.0082	0.0082	0.0082	0.0082	0.0082	0.0082	
	Environment/Application Factor	1000	1000	1	1000	1	1	1	
	Actual Failure Rate, Units: $\times 10^{-6}$ Hrs	8.2	N.A.	0.0082	N.A.	N.A.	N.A.	N.A.	
	Failure Rate Source Code	(1)	(1)	(1)	(1)	(1)	(1)	(1)	
	Probability of Failure, $\times 10^{-6}$	2.46	N.A.	17.55	N.A.	N.A.	N.A.	N.A.	20.01
Secondary Mode	Failure Classification	c	c	c	c	c	c	c	c
	Time/Cycles Units: Hrs.	0.3	0.056	4280	0.022	50	720	3600	
	Basic Failure Rate, Units: $\times 10^{-6}$ Hrs.	0.0008	0.0008	0.0008	0.0008	0.0008	0.0008	0.0008	
	Environment/Application Factor	1000	1000	1	1000	1	1	1	
	Actual Failure Rate, Units: $\times 10^{-6}$ Hrs.	0.8	0.8	0.0008	0.8	0.0008	0.0008	0.0008	
	Failure Rate Source Code	(1)	(1)	(1)	(1)	(1)	(1)	(1)	
	Probability of Failure $\times 10^{-6}$	0.24	0.0448	3.424	0.0176	0.04	0.576	2.88	7.222
	Probability of Failure, All Modes $\times 10^{-6}$	2.70	0.0448	20.97	0.0176	0.04	0.576	2.88	27.23
	Reliability	0.9[5]7	0.9[6]	0.9[4]79	0.9[6]	0.9[6]	0.9[6]	0.9[5]7	0.9[4]73

Failure Rate Data Sources: (1) Reliability engineering data series failure rates, April 1962, AVCO Corp.
**Since this switch is a proven successful design the lower limit generic failure rate is utilized here.

Notes: c Critical Failure (system failure)
C* Critical or minor depending upon when in the phase the failure occurs
M Minor failure (system success)

Basic Failure Rate Calculation: Basic failure rate from source (1) = 0.009×10^{-6}/Hr Assume 1 contact set

Assume: $\frac{\text{Fail Closed}}{\text{Fail Open}} = \frac{10}{1} = \frac{x}{y} \Rightarrow x = 10y \Rightarrow x - 10y = 0$

$x + y = 0.009 \qquad x + y = 0.009$

$+ 11y = +0.009$

Thus: $y \approx 0.0008 \times 10^{-6}$
$x \approx 0.0082 \times 10^{-6}$

4.12 Magnetometer Boom Mechanisms

This analysis of five proposed Voyager spacecraft magnetometer boom mechanisms is based on the current design status, mission profile, mechanism operation, certain necessary assumptions, and available failure rate data. The mission profile is common to all mechanisms and is defined in Table 1. Since the operation is different for each of the five mechanisms, the analysis is handled on an individual system basis. In each analysis a brief system description is followed by a critical component breakdown, and finally reliability calculations and results.

Four of the five proposed boom mechanism configuration schematics show an 18 inch fixed boom backup to the 20 foot boom. This 18 inch fixed boom consists of a single structural member deemed sufficiently strong to withstand all mission loads. The 18 inch boom is thus assumed to have a negligible mission failure probability for the purposes of this analysis. The 18 inch fixed boom is not sufficient in terms of performance to obtain the data required during the long cruise portion of the mission. It is sufficient and necessary to obtain the required data after retropropulsion since there is a $\pm 1^{\circ}$ three axis orientation requirement during Mars orbit of which the 20 foot boom is not capable. During transit the requirement is only $\pm 3^{\circ}$. Therefore, in this analysis, to obtain the data required, it is assumed that there must be a 20 foot boom non-failed during the long cruise and at least an 18 inch boom non-failed during the orbit phases of the mission.

4.12.1 The DeHavilland STEM Type Boom

The DeHavilland STEM boom is a production type item that has been utilized on the OGO spacecraft and several other aerospace projects. In the proposed application, the mechanism consists of a brushless AC motor, gearing and windup drum, and an extendable boom all encased in an hermetically sealed box. The natural lower energy level state of the boom is an extended tubular shape. It is forced flat, wound onto the windup drum, and stored in the hermetically sealed box until needed.

Table 1 Mission Profile

Mission Phase	Time(hrs)	Environmental K-Factor
1. Lift-off and boost (including injection)	0.3	1000
2. Post-injection thru capsule separation		
a. Midcourse maneuvers (accomplished during phase 2b)	0.056	1000
b. Cruise (Boom deployment is assumed to be accomplished in the first hour of cruise in this analysis. Boom retraction is assumed to be accomplished in the last hour of cruise in this analysis).	4,280.000	1
3. Post-separation (capsule) thru post-retro cruise		
a. Retropropulsion	0.022	1000
b. Cruise (after retro)	50.000	1
4. Mars Orbit (1 month)	720.000	1
5. Additional Mars Orbit (5 months)	3,600.000	1

The boom remains retracted during launch and boost. Upon spacecraft injection, the motor-gearing brake is released, whereupon the boom extends to its natural position which in this case is a 20-foot flexible tube. The boom remains in the extended position until just prior to retropropulsion, at which time it must be retracted to 18 inches to withstand the retro loads. The boom then operates for the rest of the mission in the 18 inch mode. If the motor-gear brake fails to hold the boom in the 18 inch position after retropropulsion, system failure will occur since the boom would extend to the 20-foot position. Even though the magnetometer package would operate from that point, the $\pm 1^{\circ}$ three axis orientation requirement could not be met. After retropropulsion, there are no anticipated loads of magnitude sufficient to fail the boom in its extended position.

A breakdown of the critical components is given in Table 2 including operating times, environmental K-factors, operating K-factors, and failure rates.

Since there is no component redundance in the mechanism, the mission reliability can be calculated from the following formula:

$$R_{\substack{\text{Mechanism for the}\\ \text{Mission duration}}} = (P) \prod_{i=1}^{5} (e^{-x})_i$$

where:

P = the probability of success of any "one-shot" components

$(e^{-x})_i$ = reliability of the mechanism during the i^{th} phase

$$x = \sum \lambda \left[\begin{pmatrix}\text{Duration of}\\ \text{phase i}\\ \text{applicable}\\ \text{to the}\\ \text{component}\end{pmatrix} \begin{pmatrix}\text{Phase i envir-}\\ \text{onmental K-}\\ \text{factor appli-}\\ \text{cable to the}\\ \text{component}\end{pmatrix} \begin{pmatrix}\text{Phase i}\\ \text{operational}\\ \text{K-factor}\\ \text{applicable to}\\ \text{the component}\end{pmatrix} \right]$$

where:

λ = failure rate of the component.

For phase 1:

$$x = 0.01x10^{-6}(0.3)(1000)(1) + 1.25x10^{-6}(0.3)(1000)(0.1) + 0.12x10^{-6}(0.3)(1000)(1) = 0.000076$$

$$R = (e^{-x})_1 = e^{-0.000\,076} = 0.9^4 24$$

For phase 2: (subphases 2a and 2b)

$$x = 0.01x10^{-6}\left[4,280.000)(1)(1) + (0.056)(1000)(1)\right] + 1.25x10^{-6}\left[(1)(1)(1) + (4,278)(1)(0.1) + (1)(1)(1) + (0.056)(1000)(0.1)\right] + 0.12x10^{-6}\left[(1)(1)(1) + (4,278)(1)(0.1) + (1)(1)(1) + (0.056)(1000)(0.1)\right]$$

$$= 0.000\,640$$

$$R = (e^{-x})_2 = e^{-0.000\,640} = 0.9^3 360$$

Table 2

Critical Component Breakdown: Component	Applicable Mission Hours	K-Factors: Environmental (Mission Phase) 1	2 a	2 b	3 a	3 b	4	5	Operational: Op	Non-Op	Failure Rate x 10^{-6} & Source
Hermatically sealed case (Use o-ring failure rate)	4,280.378	1000	1000	1	1000	-	-	-	1	-	0.01 (1)
AC Motor (Brushless)	4,280.378	1000	1000	1	1000	-	-	-	1	0.1	1.25 (2)
Gearing	4,280.378	1000	1000	1	1000	-	-	-	1	0.1	0.12 (1)

(1) Reliability Engineering Data Series
Failure Rates, April 1962
AVCO Corporation
Research and Advanced Development Division

(2) MIL-HDBK-217

For phase 3:

$$x = 0.01 \times 10^{-6} [(0.022)(1000)(1)] + 1.25 \times 10^{-6} [(0.022)(1000)(0.1)] + 0.12 \times 10^{-6} [(0.022)(1000)(1)] + 0.01 \times 10^{-6} [(50)(1)(1)] + 1.25 \times 10^{-6} [(50)(1)(0.1)] + 0.12 \times 10^{-6} [(50)(1)(1)]$$

$$= 0.000\,018$$

$$= e^{-0.000\,018} = 0.9^{4}\,816$$

For phase 4:

$$x = 0.01 \times 10^{-6} [(720)(1)(1)] + 1.25 \times 10^{-6} [(720)(1)(0.1)] + 0.12 \times 10^{-6} [(720)(1)(1)] = 0.000\,184$$

$$R = e^{-0.000\,184} = 0.9^{3}\,816$$

For phase 5:

$$x = 0.01 \times 10^{-6} [(3600)(1)(1)] + 1.25 \times 10^{-6} [(3600)(1)(0.1)] + 0.12 \times 10^{-6} [(3600)(1)(1)] = 0.000\,918$$

$$R = e^{-0.000\,918} = 0.9^{3}\,082$$

Finally:

$$R_{\substack{\text{mechanism for the}\\ \text{mission duration}}} = P^{*} \prod_{i=1}^{5} (e^{-x})_i = (0.9^4 24)(0.9^3 360)$$

$$(0.9^4 82)(0.9^3 816)(0.9^3 082) = 0.9^2 816$$

*P = 1 since no "one shot" components exist in this mechanism.

4.12.2 STEM Type Boom with a Fixed Backup Boom

The addition of the fixed backup boom allows less stringent operation of the retractable boom during the mission. The retractable boom remains in its retracted position during launch and boost as previously. It is also extended to its full length when the spacecraft achieves injection and remains in that position until retropropulsion as before. In this case, however, the 20 foot boom need not be retracted to 18 inches before retropropulsion since if it is functionally destroyed in the extended position by the retro acceleration load, the 18 inch fixed backup boom still allows mission success. If the STEM boom is buckled by loads in the extended position, it will return to its original shape as soon as the loads are removed. It is felt that a buckled 20-foot STEM boom would not cause damage to other spacecraft systems during retro firing. Since the boom need not be retracted before retropropulsion, the last required operation for mission success of the AC motor, gearing and windup drum, and hermetically sealed case occurs within one hour into the first mission cruise phase when the retractable boom is first extended.

A breakdown of the critical components is given in Table 3 including operating times, environmental K-factors, operating K-factors, and failure rates.

Since there is no component redundancy in the mechanism, the mission reliability can be calculated by the following formula:

Formula identical to that of 4.12.2.

Results are:

The calculated mechanism reliability for the 1st mission phase is: $0.9^{4}24$

The calculated mechanism reliability for the 2nd* mission phase is: 0.9^{6}

Mission phases 3, 4, and 5 are not applicable as per 4.12.2.

Finally:

$$R_{\text{mechanism for the mission duration}} = P^{**} \prod_{i=1}^{5} (e^{-x})_i = (0.9^{4}24)(0.9^{6}) =$$

$$0.9^{4}24 \approx 0.9^{4}2$$

* 1 hour of mission phase 2b (mission phase 2a is not applicable as per 4.12.2.

** P = 1 since no "one shot" components exist in this mechanism

Table 3

Critical Component Breakdown		K-Factors									
		Environmental (Mission Phase)							Operational		Failure Rate x 10^{-6} & Source
		1	2		3		4	5			
Component	Applicable Mission Hours		a	b	a	b			Op	Non-Op	
Hermatically sealed case	1.3	1000	-	1	-	-	-	-	1	-	0.01 (1)
AC Motor (Brushless)	1.3	1000	-	1	-	-	-	-	1	0.1	1.25 (2)
Gearing	1.3	1000	-	1	-	-	-	-	1	0.1	0.12 (1)

(1) Reliability Engineering Data Series
Failure Rates, April 1962
AVCO Corporation
Research and Advanced Development Division

(2) MIL-HDBK-217

4.12.3 Non-Retractable Boom with a Fixed Backup Boom

The non-retractable boom consists of two structural sections and two joints. One joint connects the boom sections, while the other joins the boom to the spacecraft. Both joints are identical and consist of an actuator spring, a damper, and an extended position lock. The damper operates on the principle of a vane moving through a viscous fluid. The damper case is attached to one segment of the boom, while a shaft with several vanes attached to it is connected to the other segments of the boom. The viscous fluid and several necessary seals complete the damper. A squib operated cable cutter and nylon cord serve to retain the boom in the retracted position.

During launch and boost the boom is retained in the retracted position. During the first hour of cruise after spacecraft injection, the cable cutter squib is activated and the nylon retaining cord cut. The joint spring actuators, which are in a loaded state while the boom is retracted, immediately extend, thus deploying the boom. The dampers serve to slow the spring action, thus avoiding any damage caused by the otherwise rapid boom deployment. When the boom is fully deployed, spring operated extended position locks activate and hold the joints rigid and the boom remains deployed for the duration of the mission. The boom will be structurally able to withstand retropropulsion loads.

A breakdown of the critical components is given in Table 4 including operating times, environmental K-factors, operating K-factors, and failure rates.

Since there is no component redundancy in the mechanism, the mission reliability can be calculated by the previous formula.

The calculated mechanism reliability for the 1st mission phase is: 0.9^3772

The calculated mechanism reliability for the 2nd mission phase is: 0.9^3697

The calculated mechanism reliability for the 3rd mission phase is: 1.0*

*The reliability is not 1.0, but is greater than 0.9^6, which is the limit of the mathematical scope of this analysis.

Table 4

Critical Component Breakdown

Component	Applicable Mission Hours	K-Factors: Environmental (Mission Phase) 1	2 a	2 b	3 a	3 b	4	5	Operational Op	Operational Non-Op	Failure Rate x 10^{-6} and Source	
Cable Cutter Squib	1 cycle after 1.3 hr.	-	-	-	-	-	-	-	-	-	Probability the squib does not fire prematurely = 0.999994 Probability the squib fires = 0.999706	(4)
Nylon Cord*	1.3	1000	-	1	-	-	-	-	1	-	0.438	(3)
Base Joint Actuator Spring	1.3	1000	-	1	-	-	-	-	1	-	0.11	(3)
Damper (2 0-rings)	1.3	1000	-	1	-	-	-	-	1	-	0.04	(3)
Extended Position Lock (Spring loaded latch)	8,650.378	1000	1000	1	1000	1	1	1	1	0.1	0.001	(1)
2nd Joint Actuator Spring	1.3	1000	-	1	-	-	-	-	1	-	0.11	(3)
Damper (2 0-rings)	1.3	1000	-	1	-	-	-	-	1	-	0.04	(3)
Extended Position Lock (Spring loaded latch)	8,650.378	1000	1000	1	1000	1	1	1	1	0.1	0.001	(1)

(1) Reliability Engineering Data Series, Failure Rates, April 1962, AVCO Corporation, Research and Advanced Development Division

(2) MIL-HDBK-217

Table 4 (Continued)

(3) FARADA - Source 138

(4) TRW Experience - 2000 squib firings with no failures. Probability of failure of 3×10^{-4} at 50% C. L. Telecon with FARADA indicated 47, 332 firings with 10 failures. Probability of failure 2.11×10^{-4}.

*This failure rate includes:

1.	Connector	0.2/pin x 2	=	0.400×10^{-6}
2.	Wire	0.015 x 2	=	0.03×10^{-6}
3.	Solder Joint	0.004 x 2	=	0.008×10^{-6}
				0.438×10^{-6}

The calculated mechanism reliability for the 4th mission phase is: 0.9^6

The calculated mechanism reliability for the 5th mission phase is: $0.9^5 3$

Finally:

$$R_{\text{mechanism for the mission duration}} = P \prod_{i=1}^{5} (e^{-x})_i = (0.9^3 772)(0.9^3 697)$$

$$(1.0)(0.9^6)(0.9^5 3) = 0.9^3 462 \approx 0.9^3 46$$

4.12.4 Retractable Boom with a Fixed Backup Boom

This mechanism consists of a brushless AC motor and associated gearing, cable, two boom sections, and two hinge joints. Each joint utilizes one spring actuator. An 18 inch fixed backup boom is provided.

During launch and boost the motor, gearing, and cable restrain the boom in the retracted position against joint spring actuator pressure. After spacecraft injection and during the first hour of cruise, the motor is activated allowing the boom to deploy slowly against the spring pressure. When the boom is fully deployed, the motor is shut down and the boom held in place by the spring actuator pressure. The boom remains in this state until just prior to retropuopulsion at which time the motor is reactivated and the bocm retracted to its initial position. This action is necessary since the boom is lightweight and would fail during retro loading and possibly damage other spacecraft systems. In this analysis, it is assumed that the boom is extended after retro and remains extended for the duration of the mission unless one of the spring actuator fails in which case the boom is retracted and the mission completed on the backup boom. Thus, a double failure is necessary to fail the mechanism after retro. That is, one of the spring actuators must fail and the motor, gearing or cable must fail, allowing the boom to flop around and damage the spacecraft.

A breakdown of the critical components is given in Table 5 including operating times, environmental K-factors, operating K-factors, and failure rates.

Since there is no component redundancy in the mechanism up to phase 3b, the mission reliability can be calculated by the preceding formulas.

In phases 3b, 4, and 5 the motor, gearing, and cable are considered in standby with the spring actuator. The usual standby reliability formula is utilized in these phases.

$$R = e^{-\lambda_1 K_E K_O t} + \frac{\lambda_1 K_E K_O}{\lambda_2 K_E K_O - \lambda_1 K_E K_O} \left(e^{-\lambda_1 K_E K_O t} - e^{-\lambda_2 K_E K_O t}\right)$$

where:

λ_1 = the failure rate of two spring actuators

K_E = the phase environmental K-factor

K_O = the phase operational K-factor

λ_2 = the combined failure rates of the motor, gearing, and cable

t = the appropriate phase time

The calculated mechanism reliability for the five mission phases are as follows:

(1) $0.9^{(3)}855$

(2) $0.9^{(3)}299$

(3) $0.9^{(4)}88$

(4) $0.9^{(6)}$

(5) $0.9^{(6)}$

$$R_{\text{mechanism for the mission duration}} = P \prod_{i=1}^{n} (e^{-x})_i$$

$$= (0.9^3 855)(0.9^3 299)(0.9^4 88)(0.9^6)(0.9^6)$$

$$= 0.9^3 140 \approx 0.9^3 14$$

4.12.5 Non-Retractable, Expendable Folding Boom with Fixed Backup Boom

This mechanism consists of a cable cutter, nylon cord, two boom sections, two hinge joints, and an umbilical thruster. The cable cutter

Table 5

Critical Component Breakdown		K-Factors									
		Environmental (Mission Phase)							Operational		Failure Rate x 10^{-6} and Source
		1	2		3		4	5			
Component	Applicable Mission Hours		a	b	a	b			Op	Non-Op	
Retractor and Supply Drum											
AC Motor (Brushless)	8,650.378	1000	1000	1	1000	1	1	1	1.0	0.1	1.25 (2)
Gearing	8,650.378	1000	1000	1	1000	1	1	1	1.0	0.1	0.12 (1)
Cable	8,650.378	1000	1000	1	1000	1	1	1	1.0	0.1	0.02 (3)
Base Joint											
Spring, Actuator	8,650.378	1000	1000	1	1000	1	1	1	1.0	0.1	0.11 (3)
2nd Joint											
Spring, Actuator	8,650.378	1000	1000	1	1000	1	1	1	1.0	0.1	0.11 (3)

(1) Reliability Engineering Data Series
Failure Rates, April 1962
AVCO Corporation
Research and Advanced Development Division

(2) MIL-HDBK-217

(3) Farada - Source 138

is squib operated while the umbilical thruster is spring operated. Each hinge joint utilizes a spring actuator, a spring operated extended position lock, and a damper. An 18 inch fixed backup boom is also provided.

During launch and boost the nylon cord holds the boom in the retracted position against the joint spring actuator pressure. After spacecraft injection and during the first hour of cruise, the cable cutter squib is fired and the nylon cord severed. The boom sections extend under hinge spring actuator pressure subject only to the restraint provided by the hinge dampers. When the boom is fully extended, it is held rigid by the extended position lock. The boom remains in this position throughout the first cruise including the midcourse maneuver. In the last hour prior to retropropulsion, the boom is jettisoned by the umbilical thruster. The remainder of the mission is completed utilizing the 18 inch backup boom.

A breakdown of the critical components is given in Table 6 including operating times, environmental K-factors, operating K-factors, and failure rates.

Since there is no component redundancy in the mechanism, the mission reliability can be calculated by the preceding formulas.

The calculated mechanism reliability for the first mission phase is $.9^3613$ and for the second is $.9^27394$. The other three phases are not applicable.

$$R_{\text{mechanism for the mission duration}} = \prod_{i=1}^{5} (e^{-x})_i$$

$$= (.9^3613)(.9^27394) = .9^27008 \approx .9^2701$$

4.13 Stabilization and Control Subsystem

The attached analysis was performed for the five mission phases previously defined and includes both the baseline and seven other options including the selected configuration. All powered flight portions of the mission (launch, midcourse and retro) have an environmental "k" factor of 1000 associated with them. The results are tabulated in Table 1 which shows the reliabilities of the various options for each mission

Table 6

Critical Component Breakdown		K-Factors									
		Environmental (Mission Phase)							Operational		
Component	Applicable Mission Hours	1	2a	2b	3a	3b	4	5	Op	Non-Op	Failure Rate $X10^{-6}$ and Source
Cable Cutter Squib	1 cycle after 1.3 hr	-	-	-	-	-	-	-	-	-	Probability the Squib does not fire prematurely = 0.999994 Probability the Squib fires = 0.999706 (4)
Nylon Cord*	1.3	1000	-	1	-	-	-	-	1	-	0.438 (3)
Base Joint											
Spring, Actuator	1.3	1000	-	1	-	-	-	-	1	-	0.11 (3)
Damper (2 O-rings)	1.3	1000	-	1	-	-	-	-	1	-	0.04 (3)
Extended Position Lock (Spring loaded latch)	4,280.356	1000	1000	1	-	-	-	-	1	0.1	0.001 (1)
2nd Joint											
Spring, Actuator	1.3	1000	-	1	-	-	-	-	1	-	0.11 (3)
Damper (2 O-rings)	1.3	1000	-	1	-	-	-	-	1	-	0.04 (3)
Extended Position Lock (Spring loaded latch)	4,280.356	1000	1000	1	-	-	-	-	1	0.1	0.001 (1)
Umbilical Thruster**	4,280.356	1000	1000	1	-	-	-	-	1	-	0.531 (3)

(1) Reliability Engineering Data Series Failure Rates, April 1962 AVCO Corporation, Research and Advanced Development Division

(2) MIL-HDBK-217

Table 6 (Continued)

(3) FARADA - Source 138

(4) TRW Experience - 2000 squib firings with no failures. Probability of failure of 3×10^{-4} at 50 percent C.L. FARADA indicates 47,332 firings with 10 failures. Probability of failure 2.11×10^{-4}

*This failure rate includes:

1.	Connector	0.2/pin x 2	= 0.400×10^{-6}
2.	Wire	0.015 x 2	= 0.03×10^{-6}
3.	Solder Joint	0.004 x 2	= 0.008×10^{-6}
			0.438×10^{-6}

** This failure rate includes:

1.	Bushing	0.05 x 1	= 0.05×10^{-6}
2.	Spring	0.11 x 1	= 0.11×10^{-6}
3.	Spring	0.001 x 6	= 0.006×10^{-6}
4.	O-rings	0.02 x 6	= 0.12×10^{-6}
5.	Connector	0.2/pin x 1	= 0.20×10^{-6}
6.	Wire	0.015 x 3	= 0.045×10^{-6}
			0.531×10^{-6}

Table 1. Stabilization and Control Reliability

Option	Weight in lbs	Mission Phase 1	2	3	4	5	Mission	Weight increase from Baseline (0)	Relia. increase from Baseline (0)
0	59	0.985143	0.864623	0.996742	0.973782	0.877540	0.725668	0	0
1	96	0.987118	0.926530	0.998212	0.987939	0.939218	0.847126	37	0.1214
2	68-1/2	0.991289	0.936092	0.998112	0.987831	0.938105	0.858286	9-1/2	0.1326
3	94-1/2	0.990184	0.959412	0.998838	0.985188	0.967548	0.904498	35-1/2	0.1788
4	90-1/2	0.985966	0.911255	0.997932	0.984726	0.926331	0.817870	31-1/2	0.0922
5	64-1/2	0.987065	0 889107	0.997208	0.977903	0.898144	0.768648	5-1/2	0.0429
6	63	0.991132	0.920662	0.997834	0.985068	0.925236	0.829868	4	0.1042
7*	100	0.991342	0.975492	0.999117	0.997969	0.981007	0.945916	41	0.2202

*Selected configuration

Option

0 Baseline
1 Redundant canopus with over-ride + redundant reaction control
2 Redundant canopus with over-ride + redundant signal electronics
3 Redundant signal ele + redundant reaction control + canopus over-ride
4 Redundant reaction control + canopus over-ride
5 Redundant canopus with canopus over-ride
6 Redundant signal electronics with canopus over-ride
7* Redundant canopus with over-ride, redundant reaction control, redundant signal electronics.

NOTE: "Redundant signal electronics" refers to several selected redundancies in the electronics package and therefore does not mean complete redundancy.

phase as well as for the entire mission. Also shown are weights, changes in weights and reliability for the various options.

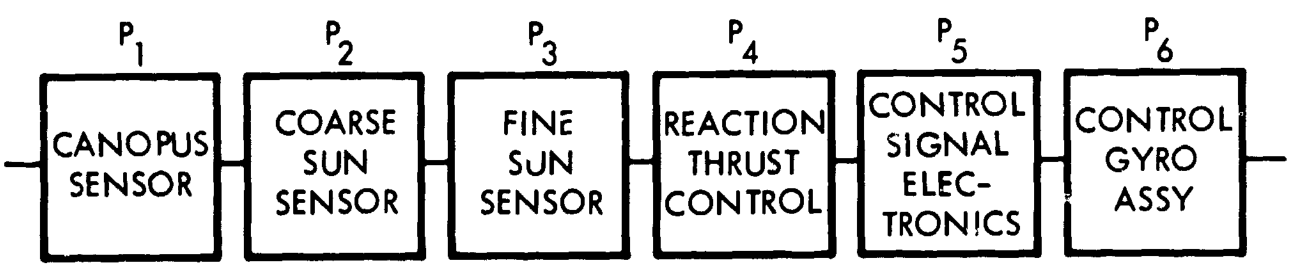
The seven options listed in Table 1 are sketched as follows. In the sketches, the numbers refer to the following:

1. Canopus sensor
2. Coarse sun sensor
3. Fine sun sensor
4. Control gyro assembly
5. Reaction thrust control
6. Control signal electronics

The stabilization and control subsystem provides full attitude stabilization of the flight spacecraft using the sun and the star canopus as the basic attitude references. The system is composed of six elements.

1. Canopus sensor
2. Coarse sun sensor (2 required)
3. Fine sun sensor
4. Control gyro assembly
5. Reaction thrust control
6. Control signal electronics

The baseline reliability block diagram is as follows:



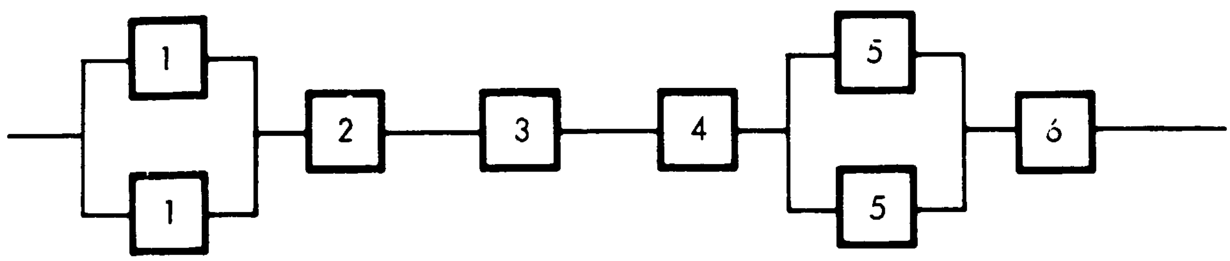
where

P_1, P_2, P_3 etc. denote the reliability of each block.

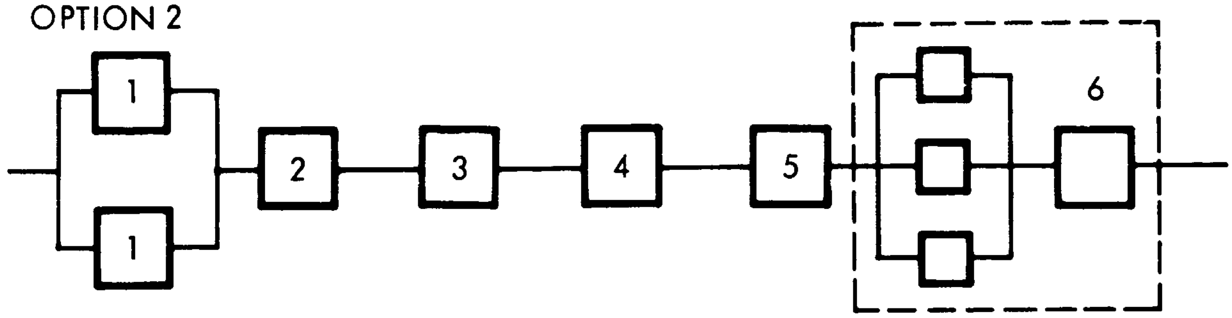
Thus

$$\text{System reliability } (R_S) = \prod_{i=1}^{6} P_i$$

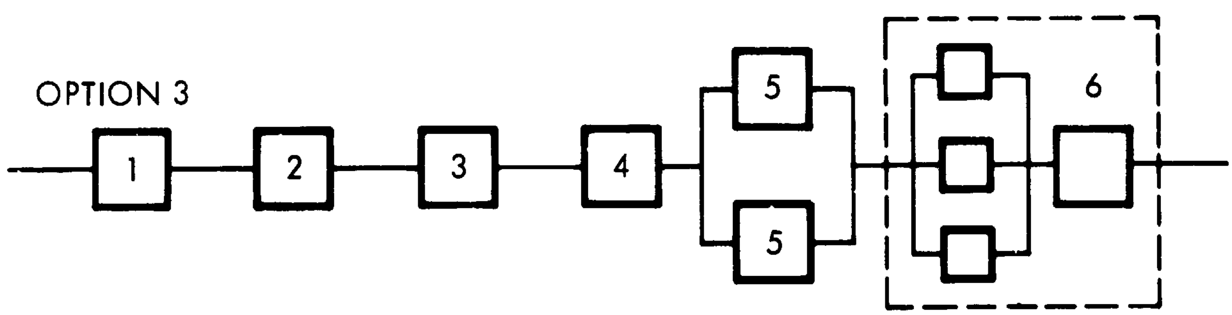
OPTION 1



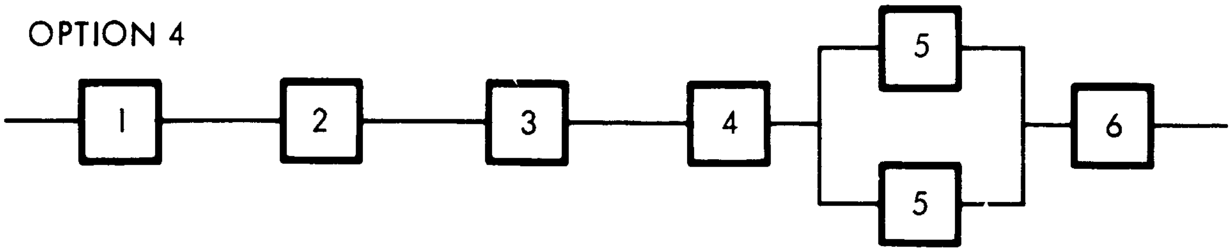
OPTION 2



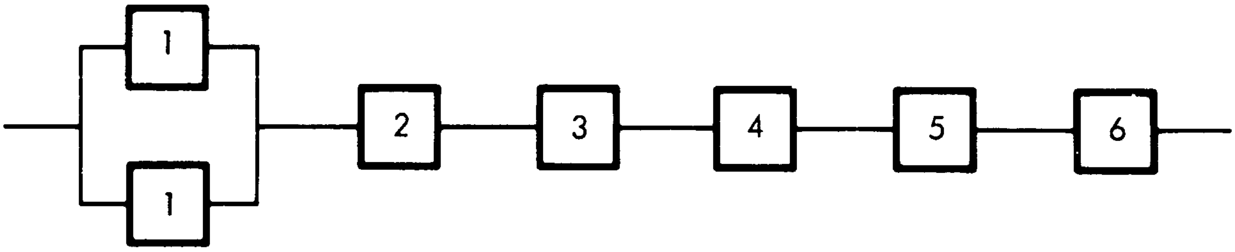
OPTION 3



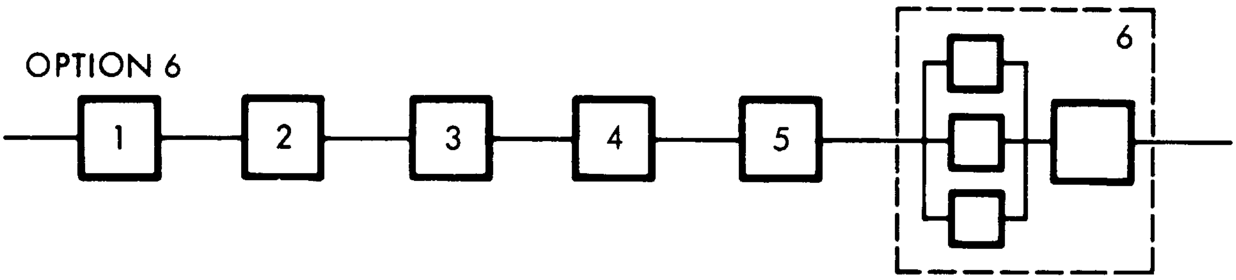
OPTION 4



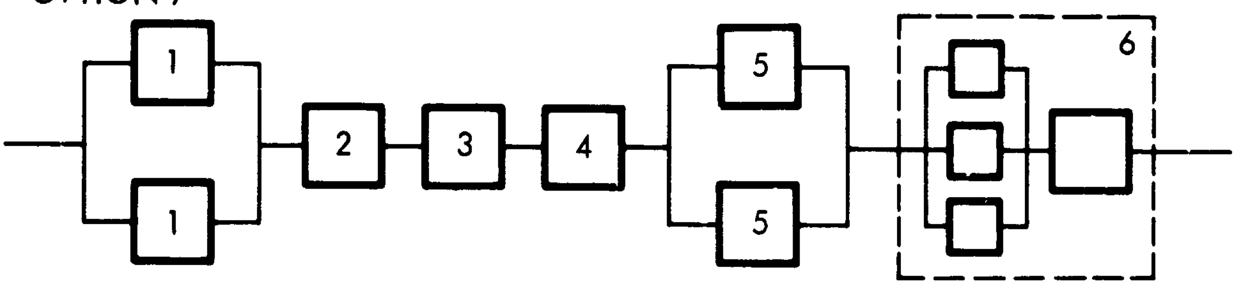
OPTION 5



OPTION 6



OPTION 7



The following terms are used throughout the analysis:

k_a = Environmental "k" factor of 1000 applied to all powered portions of the mission, i.e., launch, midcourse corrections, etc.

k_b = Environmental "k" factor of 1 applied to all cruise portions of the mission

t_p = Time in environment per phase where p goes from 1 through 5 and

t_1 = 0.3 hr

t_{2a} = 0.056 hr

t_{2b} = 4280 hrs

t_{3a} = 0.022 hr

t_{3b} = 50 hrs

t_4 = 720 hrs

t_5 = 3600 hrs

P_{1i} = Reliability of block (subsystem element) i for Mission Phase 1.

$$P_{1i} = e^{-\lambda_i k_a t_1}$$

where

λ^i = failure rate of the i^{th} block (subsystem)

In order to simplify later calculations, we will factor out the "kt" portions for the various phases.

Thus, for

Phase 1 $(k_a t_1) = (1000)(0.3) = 300$

Phase 2 $(k_a t_{2a} + k_b t_{2b}) = (1000)(0.056) + (1)(4280) = 4336$

Phase 3 $(k_a t_{3a} + k_b t_{3b}) = (1000)(0.022) + (1)(50) = 72$

Phase 4 $(k_b t_4) = (1)(720) = 720$

Phase 5 $(k_b t_5) = (1)(3600) = 3600$

The above holds true for most parts of the analysis. Where exceptions exist, they are noted. The failure rates used are as those described in this appendix; exceptions and additions are noted.

Other failure rates used were reduced to lab conditions by division with the following k factors;

Missile = 1000)		
Aircraft = 50)	Reliability engineering data series - D.R. Earles - April 1962, page 29.	
Ground = 8)		

The resultant basic failure rate was then multiplied by the appropriate "k" factor consistent with the mission profile.

4.13.1 Canopus Sensor

This sensor provides a basic attitude reference for the system and for all practical purposes is in operation throughout the mission. The electronics associated with this sensor has an MTBF of 154,000 hours (Barnes Engineering Data).

Thus

$$\lambda = \frac{1}{MTBF} = \frac{1}{154,000} = 6494 \times 10^{-9} \text{ hours}$$

Also the sun shutter is activated by a solenoid.

$\lambda = 440 \times 10^{-9}$ cycle; FARADA Page 2.175 Source 50 (Boeing)

For purposes of this analysis, it was assumed that the shutter will operate 1000 times throughout the mission. In addition, there exists a capability of ground control under certain conditions should the canopus sensor fail. This is reflected in the analysis by the use of a criticality factor of 0.6.*

*Probability of mission failure should this subsystem fail

	$e^{-\lambda kt}$	$e^{-0.6\lambda kt}$
$P_{11} = e^{-(6494 \times 10^{-9})(300)} =$	0.998052	0.998831
$P_{21} = e^{-\left[(6494 \times 10^{-9})(4336) + (440 \times 10^{-9})(500)\right]}$	0.97219	0.98324
$P_{31} = e^{-\left[(649 \times 10^{-9})(72) + (440 \times 10^{-9})(10)\right]}$	0.999532	0.99972
$P_{41} = e^{-\left[(6494 \times 10^{-9})(720) + (440 \times 10^{-9})(90)\right]}$	0.995324	0.997195
$P_{51} = e^{-\left[(6494 \times 10^{-9})(3600) + (440 \times 10^{-9})(400)\right]}$	0.97687	0.98609
$P_{m1} = (P_{11})(P_{21})(P_{31})(P_{41})(P_{51}) =$	0.94298	0.965443

where

$$P_{m1} = \text{mission reliability for Block 1} = P_1$$

The Canopus sensor was considered fully redundant in several of the options explored (see Table 1).

Thus,

$$P_{11}' = 1 - (1 - P_{11})^2 = 0.9999986$$

$$P_{21}' = 1 - (1 - P_{21})^2 = 0.99972$$

$$P_{31}' = 1 - (1 - P_{31})^2 = 0.999999$$

$$P_{41}' = 1 - (1 - P_{41})^2 = 0.9999922$$

$$P_{51}' = 1 - (1 - P_{51})^2 = 0.999807$$

$$P_{m1}' = 1 - (1 - P_{m1})^2 = 0.998806$$

Where P' refers to redundant configuration with ground override.

4.13.2 Coarse Sun Sensor (2 Required)

These sensors provide the other basic attitude reference for the system and are in operation for the entire mission. A criticality factor of 0.5 is associated with these sensors since under certain conditions the fine sun sensor in combination with one of the coarse sensors can successfully perform the intended function. Table 2 is a breakdown of the parts associated with the coarse sun sensors. This table also shows the corresponding failure rates for the individual items as well as for the sum.

Table 2

Item	Quantity	Failure Rate $x 10^{-9}$hrs	Total Failure rate $x 10^{-9}$ hrs
1-inch diam. solar cell	4	75	300
Metal film resistor	22	10	220
Capacitor (Tantalum)	4	20	80
Capacitor (ceramic)	4	15	60
Linear integrated circuits (uA702)	4	80	320
			980

$$P_{12} = e^{-(0.5)(980 \times 10^{-9})(300)} = 0.999853$$

$$P_{22} = e^{-(0.5)(980 \times 10^{-9})(4336)} = 0.997875$$

$$P_{32} = e^{-(0.5)(980 \times 10^{-9})(72)} = 0.999965$$

$$P_{42} = e^{-(0.5)(980 \times 10^{-9})(720)} = 0.999649$$

$$P_{52} = e^{-(0.5)(980 \times 10^{-9})(3600)} = 0.998236$$

$$P_{m2} = (P_{12})(P_{22})(P_{32})(P_{42})(P_{52}) = 0.995584$$

4.13.3 Fine Sun Sensor

This sensor operates for the entire mission and also has a 0.5 criticality factor associated with it in that the coarse sun sensors can under certain conditions perform the intended function. Table 3 is a breakdown of the parts associated with the fine sun sensor. This table also shows the corresponding failure rates for the individual items as well as for the sum.

Table 3

Item	Quantity	Failure Rate x 10^{-9} hrs	Total Failure Rate x 10^{-9} hrs
Radiation tracking transducer	1	300*	300
Resistor (Metal film)	22	10	220
Capacitor (Tantalum)	4	20	80
Capacitor (Ceramic)	2	15	30
Linear Integrated Circuit (uA702)	2	80	160
			790

*Assumed failure rate of 4 times that of a solar cell.

$$P_{13} = e^{-(0.5)(790 \times 10^{-9})(300)} = 0.999882$$

$$P_{23} = e^{-(0.5)(790 \times 10^{-9})(4336)} = 0.998287$$

$$P_{33} = e^{-(0.5)(790 \times 10^{-9})(72)} = 0.999972$$

$$P_{43} = e^{-(0.5)(790 \times 10^{-9})(720)} = 0.999716$$

$$P_{53} = e^{-(0.5)(790 \times 10^{-9})(3600)} = 0.998578$$

$$P_{m3} = (P_{13})(P_{23})(P_{33})(P_{43})(P_{53}) = 0.996439$$

4.13.4 Control Gyro Assembly

This assembly has an approximate 2% duty cycle and is used primarily during spacecraft orientation for Midcourse maneuvers and retro. It will also be used during occultation while in Mars orbit. Table 4 is a breakdown of the parts associated with the control gyro assembly. This table also shows the corresponding failure rates for the individual items as well as for the sum.

Table 4

Item	Quantity	Failure Rate x10⁻⁹hrs	Total Failure rate x10⁻⁹hrs
a. Gyro (integrating)	3	1000*	3000
b. Resistors	840	10	8400
c. Capacitors	60	30	1800
d. Capacitors	60	20	1200
e. Diodes G. P.	66	15	990
f. Transistors G. P.	183	50	9150
g. Transformers	3	120	360
h. Diodes, Zener	24	40	960
i. Linear Integrated Ckt. (uA702)	12	80	960
			26,820

* FARADA Page 2.321 Source 136 (Wright-Patterson)

$$P_{14} = e^{-(26,820 \times 10^{-9})(300)} = 0.991954$$

$$P_{24} = e^{-(26,820 \times 10^{-9})(141.6)*} = 0.996202$$

$$P_{34} = e^{-(26,820 \times 10^{-9})(23)*} = 0.999383$$

$$P_{44} = e^{-(26,820 \times 10^{-9})(14.4)**} = 0.999614$$

$$P_{54} = e^{-(26,820 \times 10^{-9})(72)**} = 0.998069$$

$$P_{m4} = (P_{14})(P_{24})(P_{34})(P_{44})(P_{54}) = 0.985290$$

* 2% of cruise and 100% of maneuver time

** 2% of cruise time.

REACTION THRUST CONTROL AND SIGNAL ELECTRONICS

Detailed analysis is minimum. (Copy attached.)

Reaction Thrust Control

This assembly operates for the entire mission after launch and keeps the spacecraft in the proper desired attitude throughout the mission. Table 5 is a breakdown of the parts associated with the reaction thrust control. This table also shows the corresponding failure rates for the individual items as well as for the sum. The solenoid valves were assumed to operate a total of 30,000 cycles during the mission.

Item	Quantity	Failure Rate $\times 10^{-9}$ hrs	Total Failure Rate $\times 10^{-9}$ hrs
a. Regulator	1	(a) 8680 hrs	8680
b. Solenoid valves	6	(b) 700 cyc.	-
c. Pressure transducer	2	(c) .7 cyc.	-
d. Pressure vessel	1	(d) 80	80
e. Nozzle	6	(e) 40.2	242.12
f. Fill valve	1	(f) .62	.62
g. Plumbing set	1	(g) 100	100
			9102.74

TABLE 5

a) FARADA Page 2.361 source 70 (North American Aviation)

b) Leakage assumed negligible due to dual seat valve FARADA page 2.416 source 136 (Wright-Patterson)

c) FARADA page 2.388 source 136 (Wright-Patterson)

d) Reliability Engineering Data series, Avco - April 1962 - page 83

e) FARADA page 2.348, source 123 (Chance Vought)

f) FARADA page 2.403, source 83 (Grumman) x .01 to account for valve being capped.

g) Reliability Engineering Data series, Avco - April 1962 - page 74, assuming all connections will be brazed or welded.

$$P_{15} = 1 - (180.62^{*} \times 10^{-9})(300) = .999946$$

$$P_{25} = 1 - [(9102.74 \times 10^{-9})(4336) + (5000)(701.4 \times 10^{-9})] = .95791$$

$$P_{35} = 1 - [(9102.74 \times 10^{-9})(72) + (500)(701.4 \times 10^{-9})] = .998994$$

$$P_{45} = 1 - [(9102.74 \times 10^{-9})(720) + (4500)(701.4 \times 10^{-9})] = .99029$$

$$P_{55} = 1 - [(9102.74 \times 10^{-9})(3600) + (20{,}000)(701.4 \times 10^{-9})] = .95427$$

$$P_{m5} = (P_{15})(P_{25})(P_{35})(P_{45})(P_{55}) = .904269$$

* Pressure vessel, fill valve and plumbing set are applicable in Phase 1.

The relatively low reliability of this assembly made it subject to improvement. This assembly was made fully redundant in several of the options considered. The reliabilities for the fully redundant reaction thrust control subsystem are:

$$P'_{15} = 1 - (1 - P_{15})^2 = .99999999$$

$$P'_{25} = 1 - (1 - P_{25})^2 = .998228$$

$$P'_{35} = 1 - (1 - P_{35})^2 = .999999$$

$$P'_{45} = 1 - (1 - P_{45})^2 = .999906$$

$$P'_{55} = 1 - (1 - P_{55})^2 = .99791$$

$$P'_{MS} = 1 - (1 - P_{MS})^2 = .997255$$

Control Signal Electronics

This subsystem operates for the entire mission and is responsible for the proper operation of the stabilization and control system. Table 6 is a breakdown of the parts associated with the control signal electronics. This table shows the corresponding failure rates for the individual items as well as for the sum.

	ITEM	QUANTITY	FAILURE RATE $\times 10^{-9}$ HRS	TOTAL FAILURE RATE $\times 10^{-9}$ HRS
a.	Resistor	356	8	2848
b.	Capacitors (tantalum)	24	20	480
c.	Diodes G.P.	28	15	420
d.	Flip Flop Integrated Ckts	25	35	875
e.	Gate Integrated Ckts	25	35	875
f.	Medium Power Transistor	24	130	3120
g.	Low Power Transistor	59	50	2950
h.	Power Diodes	8	100	800
i.	Medium Transformers	2	120	240
j.	Capacitor (Paper Ceramic)	36	30	1080
k.	Linear Integrated Ckt. (uA702)	21	80	1680
				15,368

TABLE 6

$$P_{16} = e^{-(15368 \times 10^{-9})(300)} = .99539$$

$$P_{26} = e^{-(15368 \times 10^{-9})(4336)} = .93556$$

$$P_{36} = e^{-(15368 \times 10^{-9})(72)} = .998894$$

$$P_{46} = e^{-(15368 \times 10^{-9})(720)} = .98896$$

$$P_{56} = e^{-(15368 \times 10^{-9})(3600)} = .94620$$

$$P_{M6} \quad (P_{16})(P_{26})(P_{36})(P_{46})(P_{56}) = .870454$$

The relatively low reliability of this assembly made it an appropriate candidate for improvement. This was accomplished with the addition of selected redundancies, such as triple redundant valve drivers, with the following results.

$$P'_{M6} = (P'_{16})\ (P'_{26})\ (P'_{36})\ (P'_{46})\ (P'_{56})$$

$$= (99965)\ (.9850)\ (.9998)\ (.99991)\ (.9883) = .971956$$

Thus, by increasing the weight by approximately 50%, we closely approach the reliability that would be obtained by complete redundancy.

RELIABILITY TRADEOFF STUDIES INVOLVING PROPOSED ALTERNATE CONFIGURATIONS (AUGMENTED VERSION) OF THE PROPULSION PRESSURIZATION AND PROPELLANT FEED SUBSYSTEM

This section presents the analysis and reliability assessments of several proposed component arrangement, and their impact on the total subsystem weight. Specifically, the method used to configurate the propulsion pressurization and propellant feed subsystem was to vary the shutoff and pressurization design configuration (denoted by G-) and also to to vary the pressure regulation design configuration (denoted by H-).

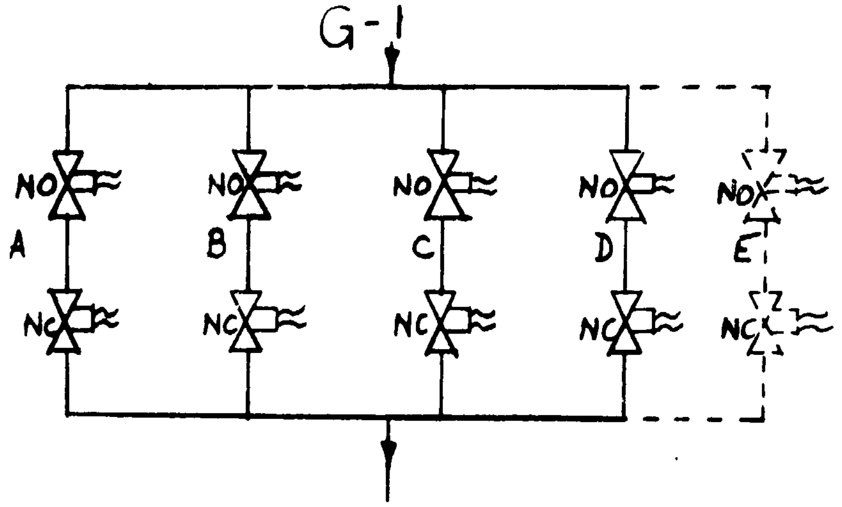
Included in this section is a schematic diagram of each of the component configurations along with necessary basic assumptions made in deriving the equations used to assess the reliability.

The results, showing the reliability assessments and the associated increase in weight above that of the baseline configuration, are presented in Table I. This tabular format makes evident significant trends pertaining to the reliability and weight parameters.

TABLE I

MODIFICATIONS TO A-2 AND B-2 BASELINE CONFIGURATIONS

	Modification	Monopropellant Reliability (Mission)	Monopropellant Weight	Bipropellant Reliability (Mission)	Bipropellant Weight
	G-1	$9^{(2)}748$	+ 0.5 lbs.	$9^{(2)}686$	+ 1.0 lbs.
	G-2	$9^{(4)}032$	+ 2.5 lbs.	$9^{(3)}879$	+ 3.5 lbs.
	G-3	$9^{(7)}307$	+ 3.3 lbs.	$9^{(7)}300$	+ 3.3 lbs.
	H-1	$9^{(6)}769$	+ 7.9 lbs.	$9^{(5)}867$	+ 7.9 lbs.
	H-2	$9^{(3)}809$	+ 1.9 lbs.	$9^{(3)}233$	+ 1.9 lbs.
	H-3	$9^{(5)}135$	+ 5.5 lbs.	$9^{(5)}101$	+ 5.5 lbs.
(1)	G-1 and H-1	$9^{(2)}748$	+ 8.4 lbs.	$9^{(2)}685$	+ 8.9 lbs.
(2)	G-1 and H-2	$9^{(2)}729$	+ 2.4 lbs.	$9^{(2)}610$	+ 2.9 lbs.
(3)	G-1 and H-3	$9^{(2)}747$	+ 6.0 lbs.	$9^{(2)}685$	+ 6.5 lbs.
(4)	G-2 and H-1	$9^{(4)}030$	+10.4 lbs.	$9^{(3)}878$	+11.4 lbs.
(5)	G-2 and H-2	$9^{(3)}712$	+ 4.4 lbs.	$9^{(3)}112$	+ 5.4 lbs.
(6)	G-2 and H-3	$9^{(3)}895$	+ 8.0 lbs.	$9^{(3)}870$	+ 9.0 lbs.
(7)	G-3 and H-1	$9^{(6)}700$	+11.2 lbs.	$9^{(5)}860$	+11.2 lbs.
(8)	G-3 and H-2	$9^{(3)}808$	+ 5.2 lbs.	$9^{(3)}232$	+ 5.2 lbs.
(9)	G-3 and H-3	$9^{(5)}128$	+ 8.8 lbs.	$9^{(5)}094$	+ 8.8 lbs.
	G-4	$9^{(2)}881$	+ 1.0 lbs.		
	G-5	$9^{(4)}725$	+ 1.9 lbs.		
(10)	G-4 and H-1	$9^{(2)}880$	+ 8.9 lbs.		
(11)	G-4 and H-2	$9^{(2)}862$	+ 2.9 lbs.		
(12)	G-4 and H-3	$9^{(2)}880$	+ 6.5 lbs.		
(13)	G-5 and H-1	$9^{(4)}723$	+ 9.8 lbs.		
(14)	G-5 and H-2	$9^{(3)}782$	+ 3.8 lbs.		
(15)	G-5 and H-3	$9^{(4)}639$	+ 7.4 lbs.		



Fifth branch is applicable to the bipropellant only.

This particular configuration utilizes a normally open and a normally closed explosive valve, in series for each branch, to be used to pressurize and subsequently shut off the pressurization function. Since four engine starts are necessary for the monopropellant configuration and five engine starts are necessary for the bipropellant configuration, four and five pressurization paths are required respectively. For analysis purposes, these paths are labeled from left to right as A, B, C, D, and E. It becomes readily apparent that this particular configuration has no redundancy built into it. Also, the operation of each pressurization branch is independent of its neighbor, and all branches have an equal probability of success.

THEORY OF OPERATION

For the purpose of analysis, it will be assumed that branch "A" is the first desirable pressurization path. Upon command, the normally closed valve is actuated. This allows the pressurization gas to flow through the pressurization module for the specified amount of time, expelling the fuel into the engine combustion chamber, facilitating engine firing until the appropriate mid-course correction has been made. Once this happens, the normally open valve in branch "A" is commanded closed and pressiruzation of the system ceases. For the monopropellant configuration, this procedure would be repeated for pressurization paths "B", "C" and "D". For the bipropellant configuration, this procedure would also be performed for pressurization path "E" to facilitate the necessary retropropulsion maneuver.

INVESTIGATION OF PROBABLE FAILURE MODES

A reliability analysis of explosive squib valves was conducted in conjunction with the reliability assessments performed on the augmented and baseline versions of the monopropellant and bipropellant configurations (see DAC-VOYAGER Memo DAC-VM-28, dated 6-24-65). The failure modes determined were:

1. Explosive valve fails to fire (both normally open and normally closed)
2. Explosive valve fires prematurely (both normally open and normally closed)
3. Normally open explosive valve leaks after closing.

G-1 (continued)

The probabilities of failure associated with these three failure modes are respectively:

1. $P_{f1} = 294 \times 10^{-6}$
2. $P_{f2} = 6 \times 10^{-6}$
3. $P_{f3} = 29.4 \times 10^{-6}$

DERIVATION OF RELIABILITY EQUATIONS

The probability of failure associated with branches A, B, C, D and E is the same and is equal to the probability that either the normally open or normally closed valve fails to fire plus the probability that either explosive valve fires prematurely plus the probability that the normally open valve leaks after closing. Algebraically, this can be represented as follows:

$$P_{f_{\text{per branch}}} = 2P_{f1} + 2P_{f2} + P_{f3} \pm \text{2nd and higher order terms}$$

$$\text{Reliability}_{\text{per branch}} = 1 - P_{f_{\text{per branch}}} = 1 - \left[2P_{f1} + 2P_{f2} + P_{f3}\right]$$

$$\text{Reliability}_{\text{mono config}} = \left[R_{\text{Branch A}}\right]\left[R_{\text{Branch B}}\right]\left[R_{\text{Branch C}}\right]\left[R_{\text{Branch D}}\right]$$

$$\text{Reliability}_{\text{Bi config}} = \left[R_{\text{Branch A}}\right]\left[R_{\text{Branch B}}\right]\left[R_{\text{Branch C}}\right]\left[R_{\text{Branch D}}\right]\left[R_{\text{Branch E}}\right]$$

Since the reliability of the branches is equal:

$$R_{\text{mono config}} = \left[R_{\text{per branch}}\right]^4$$

$$R_{\text{bi config}} = \left[R_{\text{per branch}}\right]^5$$

$$P_{f_{\text{per branch}}} = \left[(2)(294) + 2(6) + 29.4\right](10^{-6}) = (629.4)10^{-6}$$

$$R_{\text{per branch}} = 1 - P_{f_{\text{per branch}}} = 1 - 629.4(10^{-6})$$

G-1 (continued)

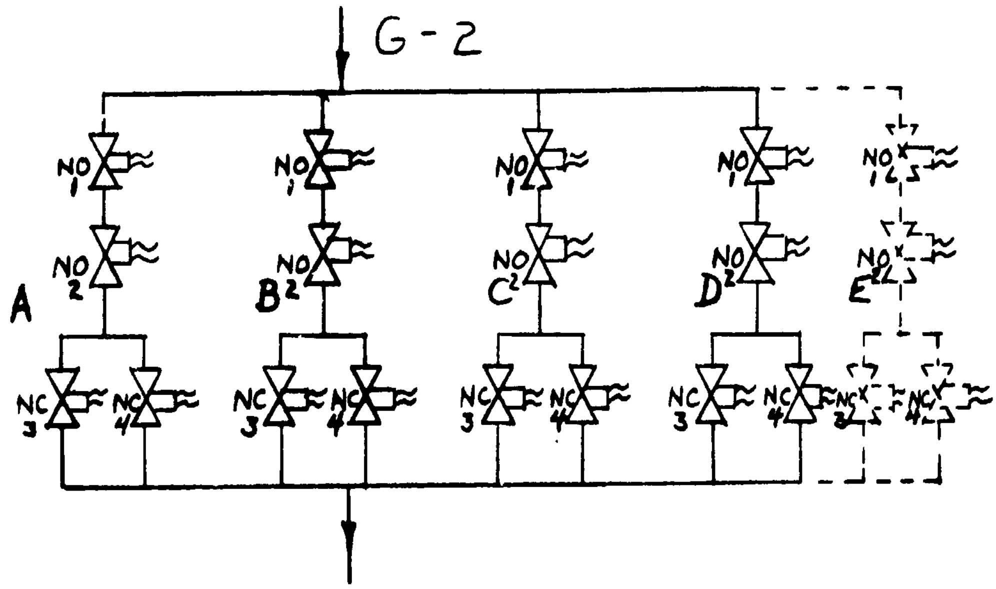
$$R_{\substack{per\\branch}} = .9^{(3)}3706$$

$$R_{\substack{mono\\config}} = \left[R_{\substack{per\\branch}}\right]^4 = 0.9993706^4$$

$$R_{\substack{mono\\config}} = .9^{(2)}748$$

$$R_{\substack{bi\\config}} = \left[R_{\substack{per\\branch}}\right]^5 = .9^{(3)}3706^5$$

$$R_{\substack{bi\\config}} = (.9^{(2)}686)$$



Fifth branch (E) used only in bipropellant configuration.

INTRODUCTION

This particular configuration is very similar to the G-1 configuration. The major difference is the introduction of additional explosive valves in each branch to facilitate redundancy. The redundancy is accomplished by utilizing two normally open explosive valves in series with two normally closed explosive valves in parallel. This arrangement of explosive valving is used in each of five pressurization paths. Since four engine starts are necessary for the monopropellant configuration and five engine starts are necessary for the bipropellant configuration, four and five pressurization paths are required respectively. For analysis purposes, these paths are labeled from left to right as A, B, C, D and E. It becomes readily apparent that this particular configuration offers dual redundant protection against the "fails to fire" mode of failure for both the normally open and normally closed explosive valve. Dual redundant protection is also offered against the "leakage" failure mode of a normally open explosive valve after it has been fired. The probability of failure of each branch associated with the premature firing of the explosive valves is twice that of the G-1 configuration because of the use of additional explosive valves.

THEORY OF OPERATION

For the purpose of analysis, it will be assumed that branch "A" is the first desirable pressurization path. Upon command, the normally closed valves #3 and #4 are actuated. It is only necessary for one of these two explosive valves to actuate to achieve successful pressurization of the system. The pressurization gas then flows through the pressurization module for the specified amount of time, expelling the fuel into the engine combustion chamber, facilitating engine firing until the appropriate mid-course correction has been made. Once this happens, the normally open valves #1 and #2 of branch A are commanded closed. Again, it is only necessary for one of these two valves to successfully close to shut off the pressurization of this system. For the monopropellant configuration, this procedure would be repeated for pressurization paths "B", "C" and "D". For the bipropellant configuration, this procedure would also be performed for pressurization path "E" to facilitate the necessary retropropulsion maneuver.

INVESTIGATION OF PROBABLE FAILURE MODES

A reliability analysis of explosive squib valves was conducted in conjunction with the reliability assessments performed on the augmented and baseline versions of the monopropellant and bipropellant configurations (see DAC-Voyager Memo DAC-VM-28, dated 6-24-65) The failure modes determined were:

1. Explosive valve fails to fire (both normally open and normally closed)
2. Explosive valve fires prematurely (both normally open and normally closed)
3. Normally open explosive valve leaks after closing.

The probabilities of failure associated with these three failure modes are respectively:

1. $P_{f_1} = 294 \times 10^{-6}$
2. $P_{f_2} = 6 \times 10^{-6}$
3. $P_{f_3} = 29.4 \times 10^{-6}$

DERIVATION OF RELIABILITY EQUATIONS

The probability of failure associated with branches A, B, C, D, and E is the same and is equal to the probability that both normally closed valves #3 and #4 fail to fire plus the probability that both normally open valves #1 and #2 fail to fire plus the probability that either explosive valve #1, #2, #3 or #4 fires prematurely, plus the probability that both normally open valves #1 and #2 leak, after having been fired. Algebraically, this can be represented as follows:

$$P_{f_{\text{per branch}}} = 2P_{f_1}^2 + 4P_{f_2} + P_{f_3}^2 \pm \text{other 2nd and higher order terms}$$

$$R_{\text{per branch}} = 1 - P_{f_{\text{per branch}}} = 1 - (2P_{f_1}^2 + 4P_{f_2} + P_{f_3}^2)$$

$$\text{Reliability}_{\text{mono config}} = \left[R_{\text{branch A}}\right]\left[R_{\text{branch B}}\right]\left[R_{\text{branch C}}\right]\left[R_{\text{branch D}}\right]$$

$$\text{Reliability}_{\text{bi config}} = \left[R_{\text{branch A}}\right]\left[R_{\text{branch B}}\right]\left[R_{\text{branch C}}\right]\left[R_{\text{branch D}}\right]\left[R_{\text{branch E}}\right]$$

Since the reliability of the branches is equal:

$$R_{\text{mono config}} = \left[R_{\text{per branch}}\right]^4$$

$$R_{\text{bi config}} = \left[R_{\text{per branch}}\right]^5$$

$$P_{f_{\text{per branch}}} = 2(294 \times 10^{-6})^2 + 4(6 \times 10^{-6}) + (29.4 \times 10^{-6})^2 = 24.1737 \times 10^{-6}$$

$$R_{\text{per branch}} = 1 - P_{f_{\text{per branch}}} = 1 - 24.17\,(10^{-6})$$

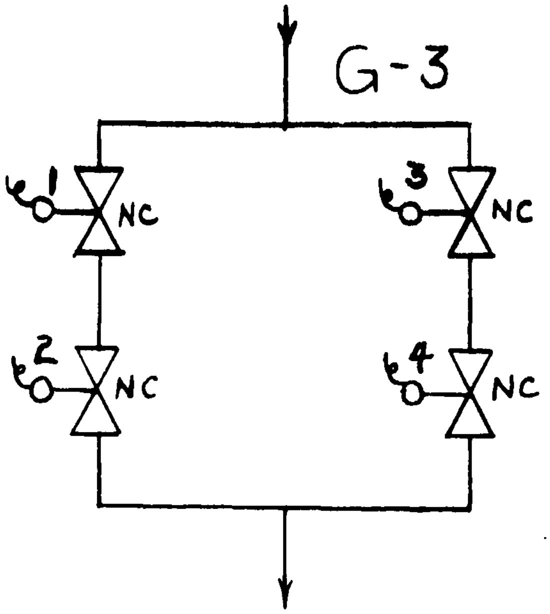
$$R_{\text{per branch}} = 9^{(4)}758^{+}$$

$$R_{\text{mono config}} = \left[R_{\text{per branch}}\right]^4 = \left[9^{(4)}758\right]^4$$

$$\boxed{R_{\text{mono config}} = 9^{(4)}032}$$

$$R_{\text{bi config}} = \left[R_{\text{per branch}}\right]^5 = 9^{(4)}758^{\,5}$$

$$\boxed{R_{\text{bi config}} = 9^{(3)}879}$$



This configuration was analyzed in detail as part of the reliability analysis of the augmented versions of the monopropellant and bipropellant configurations in DAC-Voyager Memo DAC-VM-28, dated 6-24-65. This arrangement of solenoid valve is colloquially called the "quad solenoid" configuration.

THEORY OF OPERATION

This configuration has power applied to the solenoid valves only when the system is being pressurized to facilitate engine firing. It is necessary for either valves #1 and #2 or valves #3 and #4 to open successfully to facilitate system pressurization. It is also necessary for either valves #3 or #4; or valves #1 or #2 to both close successfully and not leak to bring about system depressurization. Therefore, this configuration is redundant for both the "leakage" mode of failure and the "fails to open" or "fails to close" mode of failure.

INVESTIGATION OF PROBABLE FAILURE MODES

Each single solenoid has the following failure modes:

1. Solenoid valve fails to open
2. Solenoid valve fails to close
3. Solenoid valve leaks after closing/or normally closed valve leaks.

The probabilities of failure associated with these failure modes are respectively:

1. $(1-s_o) = 0.42 \times 10^{-6}$/cycle
2. $(1-s_c) = 0.056 \times 10^{-6}$/cycle
3. $(1-S\ ) = 131.58 \times 10^{-6}$/hr mission

4 cycles will be required for phase 2 associated with the required mid-course corrections, and 1 cycle will be required for phase 3 to facilitate retropropulsion.

The phases mentioned here are as defined in DAC-Voyager Memo DAC-VM-28, dated 6-24-65. Also, a phase 5 has been added for this analysis. Phase 5 is defined as an additional five-month period during which the spacecraft orbits Mars. Actually, this particular portion of the pressurization system need function properly only during Phases 1, 2, and 3. Only failure mode 3 is applicable during Phase 1; failure modes 1, 2 and 3 are applicable during Phase 2; failure modes 1 and 2 are applicable during Phase 3 for the bipropellant configuration only.

DERIVATION OF RELIABILITY EQUATIONS

*Refer to pp. 30-31, 51 of DAC-VM-28 Memo dated 6-24-65.

$$*R_{\substack{mono\\config}} = \left[R_{phase\ 1}\right]\left[R_{phase\ 2}\right]$$

$$*R_{\substack{bi\\config}} = \left[R_{phase\ 1}\right]\left[R_{phase\ 2}\right]\left[R_{phase\ 3}\right]$$

Phase 1

$$R_{phase\ 1} = R_{\substack{quad\ solenoid\\leakage}} = \left[1 - (1-S_{\ell_{phase\ 1}})^2\right]^2$$

Phase 2

$$R_{phase\ 2} = \left[R_{\substack{quad\ solenoid\\leakage\ phase\ 2}}\right]\left[R_{\substack{quad\ solenoid\\cycle\ phase\ 2}}\right]$$

$$R_{\substack{quad\ solenoid\\leakage\ phase\ 2}} = \left[1 - (1-S_{\ell_{phase\ 2}})^2\right]^2$$

$$*R_{\substack{quad\ solenoid\\cycle}} = s_o^4 s_c^4 + 4s_o^4 s_o^3 (1-s_c) + 4s_o^4 s_c^2 (1-s_c)^2 + 4s_o^3 (1-s_o) s_c^3$$

$$+ 8s_o^3 (1-s_o) s_c^2 (1-s_c) + 2s_o^2 (1-s_o^2) s_c^2 + 4s_o^2 (1-s_o)^2 s_c (1-s_c)$$

This equation gives the reliability of the quad solenoid for failure in the "open" or "closed" mode. The reliability of the quad solenoid configuration was calculated to be greater than $1 - 10^{-9}$.

Phase 3

$$R_{phase\ 3} = \left[1 - (1-S_{\ell_{phase\ 1}})^2\right]^2 \left[R_{\substack{quad\ solenoid\\4\ cycles\ phase\ 2}}\right]\left[1 - (1-S_{phase\ 2})^2\right]^2$$

$$R_{\substack{mono\\config}} = \left[1 - (8.58 \times 10^{-6})^2\right]^2 \left[9^{(9)}4\right]\left[1 - (123 \times 10^{-6})^2\right]^2$$

$$R_{\substack{mono\\config}} = 9^{(9)}853 \quad 9^{(9)}+ \quad 9^{(7)}697$$

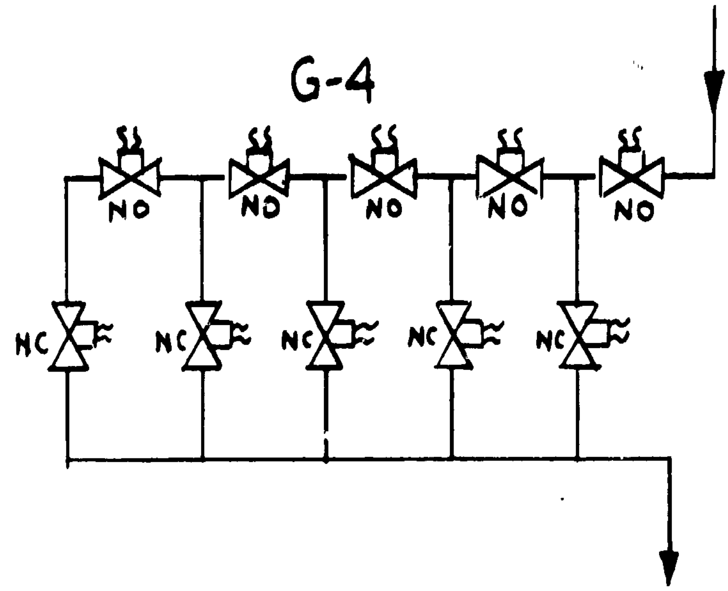
$$\boxed{R_{\substack{mono\\config}} = .9^{(7)}307}$$

$$R_{\substack{bi\\config}} = \left[1 - (1 - S_{\ell_{phase\ 1}})^2\right]^2 \left[R_{\substack{quad\ solenoid\\4\ cycles\ phase\ 2}}\right] \left[1 - (1 - S_{\ell_{phase\ 2}})^2\right]^2 \times \left[R_{\substack{quad\ solenoid\\1\ cycle\ phase\ 3}}\right]^*$$

$$R_{\substack{bi\\config}} = 9^{(9)}853\ \ 9^{(9)} + \ \ 9^{(7)}697\ \ 9^{(9)} +$$

$$\boxed{R_{\substack{bi\\config}} = .9^{(7)}300}$$

* Page 51 of DAC-VM-28 Memo, dated 6-24-65



a = probability that a normally closed valve opens successfully.
b = probability that a normally open valve closes successfully.
c = probability that a normally open valve does not leak after closing.

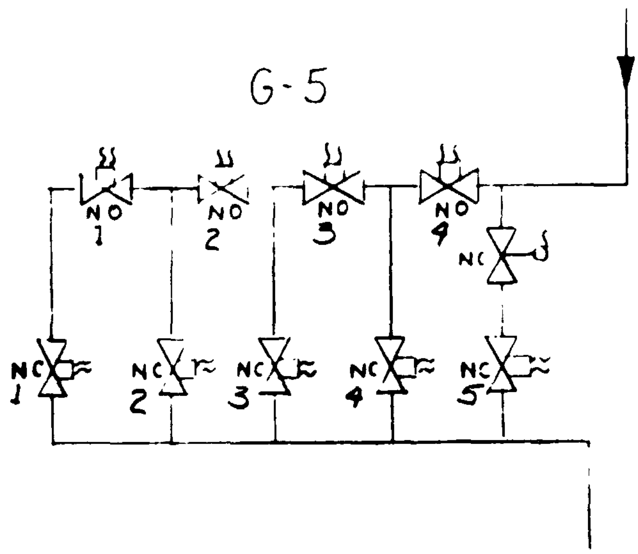
$$R_{G-4} = (abc)^4 + 3a^4b^3(1-b)c^3abc + 4a^3(1-a)b^3c^4abc + a^4b^3(1-b)$$
$$+ (0.5)a^4b^4c^3(1-c) + 0.5\ a^4b^4c^2(1-c)^2(10) + \text{3rd and higher order terms}$$

$a = \bar{e}_c = 0.999706$
$(1-a) = 0.000294$
$a^2 = 0.99941209$
$a^4 = 0.9988245$[illegible]
$a^7 = 0.9979438$
$a^8 = 0.99765044$
$a^9 = 0.99735713$
$b = \bar{e}_o = 0.999706$

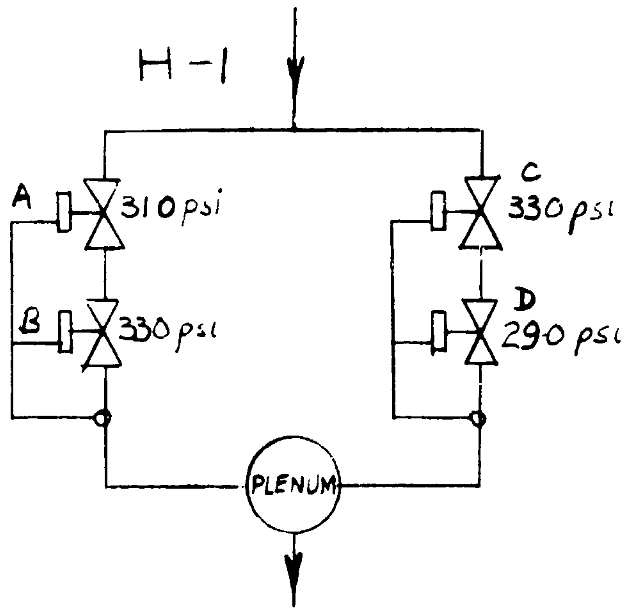
$e = \bar{e}_{OL} = 0.999994$
$(1-c) = 0.000006 = 6 \times 10^{-6}$
$(1-c)^2 = 36 \times 10^{-12}$
$c^2 = 0.999989$
$c^3 = 0.999982$
$c^4 = 0.999976$
$c^5 = 0.999970$

$R_{G-4} =$ 0.9976265
+0.0008796
+0.0000117
+0.0002934
+0.0000003
+0.0000000

0.9988115



The reliability analysis and assessments of the G-5 configuration will be submitted as part of the TRW input. The result of this analysis is given in Table I and Figure 1.



This particular configuration offers protection against both the underpressurization and overpressurization modes of failure.

A logic diagram has been attached showing the results of all outcomes including success and failures. The regulators are designated A, B, C, and D.

The following ground rules and definitions are applicable to the derivation of the reliability expression for the success criterion associated with phase 2 and phase 3.

$$R_{\substack{\text{mono}\\ \text{config}}} = R_{\text{phase 2}}$$

$$R_{\substack{\text{bi}\\ \text{config}}} = \left[R_{\text{phase 2}}\right]\left[R_{\text{phase 3}}\right]$$

$O_A = O_B = O_C = O_D \equiv$ Probability of regulator not failing in the overpressurization mode.

$\bar{O}_A = \bar{O}_B = \bar{O}_C = \bar{O}_D \equiv$ Probability of regulator failing in the overpressurization mode.

$U_A = U_B = U_C = U_D \equiv$ Probability of regulator not failing in the underpressurization mode.

$\bar{U}_A = \bar{U}_B = \bar{U}_C = \bar{U}_D \equiv$ Probability of regulator failing in the underpressurization mode.

$O_2 = 9^{(3)}708$

$\bar{O}_2 = 0^{(3)}292$

Referring to the logic diagram, the probability of the outcome is as follows:

H-1 (continued)

$U_2 = 9^{(3)}809$	$(O_A U_A) + (O_A \overline{U}_A O_D U_D) + [O_A \overline{U}_A O_D U_D] + (O_A \overline{U}_A \overline{O}_D O_C U_C)$
$\overline{U}_2 = 0^{(3)}191$	$+ [O_A \overline{U}_A \overline{O}_D O_C \overline{U}_C] + [O_A \overline{U}_A \overline{O}_D \overline{O}_C] + (\overline{O}_A O_B U_B) +$
$O_3 = 9^{(3)}128$	$(\overline{O}_A O_B \overline{U}_B C_D U_D) + \overline{O}_A O_B \overline{U}_B O_D \overline{U}_D + (\overline{O}_A O_B \overline{U}_B \overline{O}_D O_C U_C)$
$\overline{O}_3 = 0^{(3)}872$	$[\overline{O}_A O_B \overline{U}_B \overline{O}_D O_C \overline{U}_C] + [\overline{O}_A O_B \overline{U}_B \overline{O}_D \overline{O}_C] + [\overline{O}_A \overline{O}_B]$
$U_3 = 9^{(3)}423$	The terms in parentheses contribute to success; the terms in brackets contribute to failure.
$\overline{U}_3 = 0^{(3)}577$	

Phase 2

Probability of Success is the sum of the following terms:

$O_2 U_2$	$= (9^{(3)}708)(9^{(3)}809)$	9995170558
$O_2 U_2 \overline{U}_2$	$= 9^{(3)}708^{\,2}(9^{(3)}809)(0^{(3)}191)$	0001908520
$(2) O_2^2 \overline{O}_2 \overline{U}_2 U_2$	$= (2)\, 9^{(3)}708^{\,2}(0^{(3)}292)(0^{(3)}191)(9^{(3)}809)$	0000001115
$O_2 \overline{O}_2 U_2$	$= (9^{(3)}708)(0^{(3)}292)(9^{(3)}809)$	0002918589
$O_2^2 \overline{O}_2^2 U_2 \overline{U}_2$	$= 9^{(3)}708^{\,2}\; 0^{(3)}292^{\,2}\; 9^{(3)}809\; (0^{(3)}191$	0000000000
		= .9999998782

Probability of Failure is the sum of the following terms:

$O_2^2 \overline{U}_2^2$	$= (9^{(3)}708)^2(0^{(3)}191)^2$	0000000364
$O_2^2 \overline{U}_2^2 \overline{O}_2$	$= (9^{(3)}708)^2(0^{(3)}191)^2(0^{(3)}292)$	0000000000
$\overline{O}_2^2 \overline{U}_2 O_2$	$= (0^{(3)}292)(0^{(3)}191)(9^{(3)}708)$	0000000558
$O_2^2 \overline{U}_2^2 \overline{O}_2$	$= (9^{(3)}708)^2(0^{(3)}191)^2(0^{(3)}292)$	0000000000
$\overline{U}_2^2 \overline{O}_2^2 O_2^2$	$= (0^{(3)}191)^2(0^{(3)}292)^2(9^{(3)}708)^2$	0000000000
$\overline{O}_2^3 O_2 \overline{U}_2$	$= (0^{(3)}292)^2(9^{(3)}708)(0^{(3)}191)$	0000000000
$\overline{O}_2^2$	$= (0^{(3)}292)^2$	0000000853
		.0000001775

Phase 3

Probability of Success is the sum of the following terms:

O_3U_3	$= (9^{(3)}128)(9^{(3)}423)$	9985515031
$O_3^2U_3\bar{U}_3$	$= (9^{(3)}128)^2(9^{(3)}423)(0^{(3)}577)$	0005756618
$2\ O_3^2O_3\bar{U}_3\bar{U}_3$	$= (2)(9^{(3)}128)^2(0^{(3)}872)(0^{(3)}577)(9^{(3)}423)$	0000010039
$O_3\bar{O}_3U_3$	$= (9^{(3)}128)(0^{(3)}872)(9^{(3)}423)$	0008707369
$O_3^2\bar{O}_3^2U_3\bar{U}_3$	$= (9^{(3)}128)(0^{(3)}872)^2(9^{(3)}423)(0^{(3)}577)$	0000000004
	=	9999989061

Probability of Failure is the sum of the following terms:

$O_3^2\bar{U}_3^2$	$= (9^{(3)}128)^2(0^{(3)}577)^2$	0000003323
$O_3^2\bar{U}_3^2\bar{O}_3$	$= (9^{(3)}128)^2(0^{(3)}577)^2(0^{(3)}872)$	0000000003
$\bar{O}_3^2\bar{U}_3O_3$	$= (0^{(3)}872)^2(0^{(3)}577)(9^{(3)}128)$	0000000003
$O_3^2\bar{U}_3^2\bar{O}_3$	$= (9^{(3)}128)^2(0^{(3)}577)^2(0^{(3)}872)$	0000000003
$\bar{U}_3^2\bar{O}_3^2O_3^2$	$= (0^{(3)}577)^2(0^{(3)}872)^2(9^{(3)}128)$	0000000000
$\bar{O}_3^3O_3\bar{U}_3$	$= (0^{(3)}872)^3(9^{(3)}128)(9^{(3)}423)$	0000000006
$\bar{O}_3^2$	$= (0^{(3)}872)^2$	0000007602
		0000010940

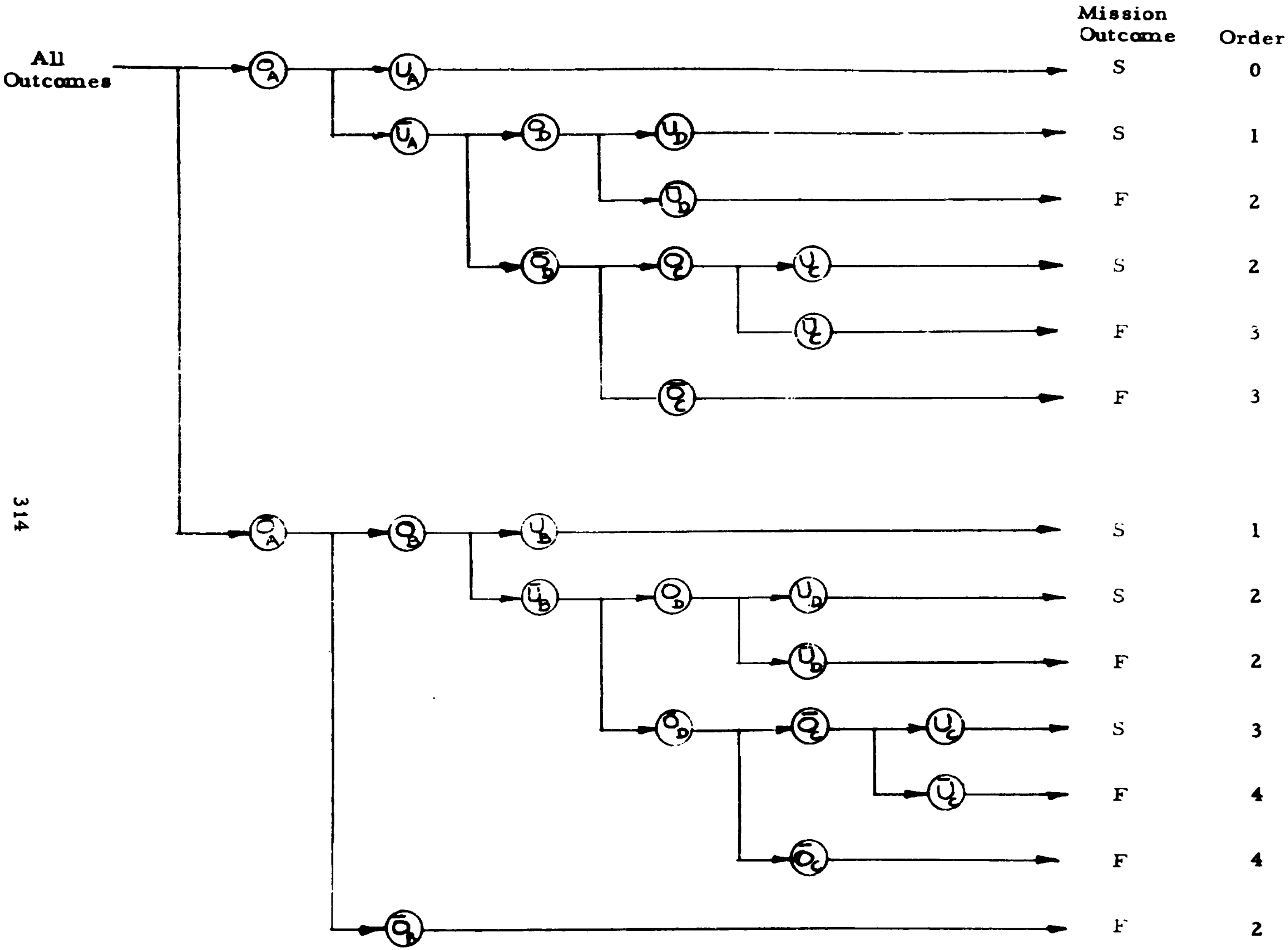
$$R_{\substack{\text{mono}\\ \text{config}}} = R_{\text{phase 2}}$$

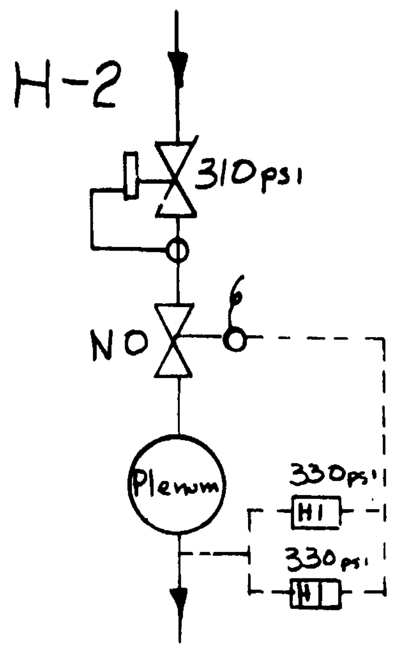
$$\boxed{R_{\substack{\text{mono}\\ \text{config}}} = 9^{(6)}769}$$

$$R_{\substack{\text{bi}\\ \text{config}}} = \left[R_{\text{phase 2}}\right]\left[R_{\text{phase 3}}\right] = (9^{(6)}769)(9^{(5)}891)$$

$$\boxed{R_{\substack{\text{bi}\\ \text{config}}} = 9^{(5)}867}$$

H-1 LOGIC DIAGRAM





This configuration offers additional protection against failures that might result due to overpressurization of the regulator. The two pressure switches are set at a value of 330 psi. If for some reason the regulator fails to regulate at 310 psi and the downstream pressure tries to increase, the two pressure switches pick up when the downstream pressure reaches 330 psi. When the two pressure switches pick up, the normally open solenoid valve is commanded closed. If either pressure switch fails to pick up or prematurely drops out, the solenoid valve would remain in the normally open position, resulting in a failure.

DERIVATION OF RELIABILITY EQUATIONS:

$$R_{mission} = \left[R_{phase\ 2}\right]\left[R_{phase\ 3}\right]$$

$P_{2\,reg\,under}$ = Probability of success associated with the underpressurization mode of the regulator for phase 2

$$P_{2\,reg\,under} = 9^{(3)}809$$

$P_{2\,reg\,over}$ = Probability of success associated with the overpressurization mode of the regulator for phase 2

$$P_{2\,reg\,over} = 9^{(3)}708$$

$\bar{P}_{2f\,reg\,over}$ = Probability of failure associated with the overpressurization mode of the regulator for phase 2

H-2 (continued)

$\overline{P}_{2f\,\substack{reg\\over}} = 0^{(3)}292$

$P_{\substack{pres\\sw}} = 9^{(6)}440$

S_o = Probability the solenoid valve cycles successfully

$S_o = 9^{(6)}524$

Assume 4 cycles(n_2) necessary during phase 2 and 1 cycle (n_3) necessary during phase 3.

$S_{\ell\,phase\ 2}$ = Probability the solenoid does not leak during phase 2 when it has been closed, = $9^{(3)}$

$P_{3\,\substack{reg\\under}}$ = Probability of success associated with the underpressurization mode of the regulator for phase 3

$P_{3\,\substack{reg\\under}} = 9^{(3)}423$

$P_{3\,\substack{reg\\over}}$ = Probability of success associated with the overpressurization mode of the regulator for Phase 3

$P_{3\,\substack{reg\\over}} = 9^{(3)}128$

$\overline{P}_{3f\,\substack{reg\\over}}$ = Probability of failure associated with the overpressurization mode of the regulator for phase 3

$\overline{P}_{3f\,\substack{reg\\over}} = 0^{(3)}872$

$$R_{phase\ 2} = \left[P_{2\,\substack{reg\\under}}\right]\left[P_{2\,\substack{reg\\over}}\right] + \overline{P}_{2f\,\substack{reg\\over}}\left[(S_o)^{n_2}\,(P_{\substack{pres\\sw}})^{n_2}\,S_{\ell\,phase\ 2}\right]$$

H-2 (continued)

$$R_{phase\ 3} = \left[P_{3\ \substack{reg\\under}}\right]\left[P_{3\ \substack{reg\\over}}\right] + \overline{P}_{3f\ \substack{reg\\over}}\left[(S_o)^{n_3}(P_{\substack{pres\\sw}})^{n_3}\right]$$

$$R_{phase\ 2} = \left[9^{(3)}809\right]\left[9^{(3)}708\right] + 0^{(3)}292\left[(9^{(6)}524)^4(9^{(6)}440)^4(9^{(3)}877)\right]$$

$$R_{phase\ 2} = 0.9995170558 + 0^{(3)}292\,(9^{(3)}8728565)$$

$$R_{phase\ 2} = 9^{(3)}809$$

$$R_{phase\ 3} = \left[9^{(3)}423\right]\left[9^{(3)}128\right] + 0^{(3)}872\left[(9^{(6)}524)^1(9^{(6)}440)^1\right]$$

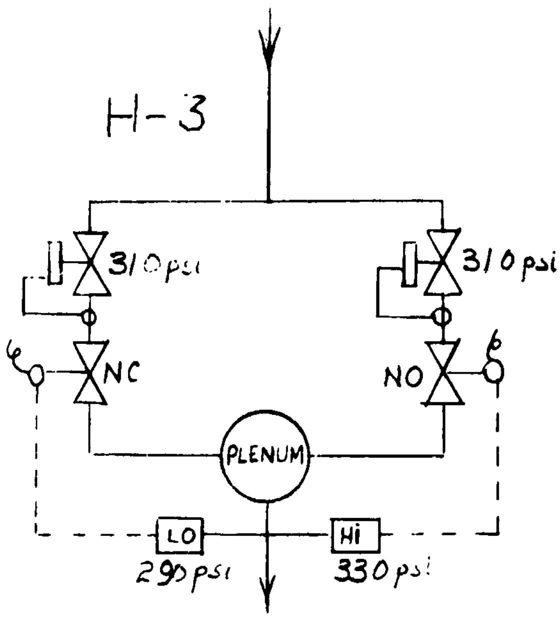
$$R_{phase\ 3} = 0.9985515031 + 0^{(3)}8719991$$

$$R_{phase\ 3} = 9^{(3)}424$$

$$R_{\substack{mono\\config}} = 9^{(3)}809$$

$$R_{\substack{bi\\config}} = \left[R_{phase\ 2}\right]\left[R_{phase\ 3}\right] = (9^{(3)}809)(9^{(3)}424)$$

$$R_{\substack{bi\\config}} = 9^{(3)}233$$



This particular configuration offers protection from both the underpressurization and overpressurization modes of failure. If the pressure of the system attempts to go over 330 psi, the high pressure switch picks up and closes the normally open solenoid valve. The high pressure switch keeps cycling until the system again regulates at the nominal pressure of 310 psi. If the pressure of the system attempts to decrease to a value below 290 psi, the low pressure switch picks up and opens the normally closed valve on the left. The regulator in this branch then regulates at 290 psi.

DERIVATION OF RELIABILITY EQUATIONS

1. The failure modes and probabilities of failure associated with the regulators are equal to the values shown in the "Failure Mode Probability Summary" of DAC-Voyager Memo DAC-VM-28, dated 6-24-65.

2. Assume 100 cycles of operation for each pressure switch. Of these, 4 actuations will be applicable to phase 2 and 1 actuation will be applicable to phase 3.

3. The probability of failure of the pressure switch shall be 0.28×10^{-6}/actuation. $= P_{f_{pr\,sw}}$

4. Phases 1 and 4 are not applicable for either the bipropellant or monopropellant configuration except for the solenoid leakage of Phase 1. Phase 3 is only applicable for the bipropellant configuration.

Therefore: $\text{Reliability}_{system} = \left[R_{\substack{under\\pressure}}\right]\left[R_{\substack{over\\pressure}}\right] = \left[1-P_{f_u}\right]\left[1-P_{f_o}\right]$

$P_{f_{2U}}$ = Probability of failure associated with the underpressurization mode of the regulator for phase 2.

$P_{f_{2U}} = 191 \times 10^{-6}$

H-3 (continued)

$P_{f_{2O}}$ = Probability of failure associated with the overpressurization mode of the regulator for phase 2

$P_{f_{2O}}$ = 292×10^{-6}

$P_{f_{3U}}$ = Probability of failure associated with the underpressurization mode of the regulator for phase 3

$P_{f_{3U}}$ = 577×10^{-6}

$P_{f_{3O}}$ = Probability of failure associated with the overpressurization mode of the regulator for phase 3

$P_{f_{3O}}$ = 872×10^{-6}

S_o = Probability the solenoid will open and stay open = 0.42×10^{-6}

S_{cycle} = Probability the solenoid valve does not cycle successfully

S_{cycle} = 0.476×10^{-6} (four cycles are required for phase 2 and one cycle is required for phase 3.)

n = number of cycles

S_ℓ = Probability the normally open solenoid leaks after closing

$S_{\ell_{phase\ 1}}$ = 8.58×10^{-6}

$S_{\ell_{phase\ 2}}$ = 123×10^{-6}

$$R_{\substack{mono \\ config}} = \left[R_{phase\ 1}\right]\left[R_{phase\ 2}\right]$$

$$R_{\substack{bi \\ config}} = \left[R_{phase\ 1}\right]\left[R_{phase\ 2}\right]\left[R_{phase\ 3}\right]$$

$R_{phase\ 1}$ = Probability the normally closed solenoid valve doesn't leak

$$R_{phase\ 1} = 1 - S_{\ell_{phase\ 1}} = 1 - 8.58 \times 10^{-6} = 9^{(5)}142$$

$$R_{phase\ 2} = \left[1 - P_{f_{2O}}\right]\left[1 - P_{f_{2U}}\right] + P_{f_{2O}}\left[\left[1 - (n_2) P_{\underset{sw}{pr}}\right]\left[1 - (n_2) S_{cycle}\right]\left[1 - S_{\ell_{phase\ 2}}\right]\right] +$$

$$P_{f_{2U}}\left[1 - P_{\underset{\frac{sw}{2}}{pr}}\right]\left[(1 - S_o)(1 - P_{f_{2O}})(1 - P_{f_{2U}})\right] + \ldots$$

$$P_{phase\ 3} = \left[1 - P_{f_{3O}}\right]\left[1 - P_{f_{3U}}\right] + P_{f_{3O}}\left[1 - (n_3) P_{\underset{sw}{pr}}\right]\left[(1 - (n_3) S_{cyc}\right] + P_{f_{3U}} \times$$

$$\times \left[(1 - P_{\underset{\frac{sw}{2}}{pr}})\ (1 - S_o)\ (1 - P_{f_{3O}})\ (1 - P_{f_{3U}})\right]$$

$$R_{phase\ 2} = \left[1 - 292 \times 10^{-6}\right]\left[1 - 191 \times 10^{-6}\right] + 292 \times 10^{-6}\left[1 - 4(+0.28 \times 10^{-6})\right] \times$$

$$\left[1 - (4)(0.476 \times 10^{-6})\right] \times \left[1 - 123 \times 10^{-6}\right] + 191 \times 10^{-6}\left[(1 - (1)(0.14 \times 10^{-6})\right]$$

$$\times \left[(1 - 0.42 \times 10^{-6})\ (1 - 292 \times 10^{-6})\ (1 - 191 \times 10^{-6})\right]$$

$$R_{phase\ 3} = \left[1 - 872 \times 10^{-6}\right]\left[1 - 577 \times 10^{-6}\right] + 872 \times 10^{-6}\left[1 - (1)(0.28 \times 10^{-6})\right]$$

$$\left[1 - (1)(0.476 \times 10^{-6})\right] + (577 \times 10^{-6})\left[1 - (1)(+0.14 \times 10^{-6})\right](1 - 0.42 \times 10^{-6})$$

$$\times (1 - 872 \times 10^{-6})(1 - 577 \times 10^{-6})$$

$$R_{phase\ 2} = 9^{(3)}708 \quad 9^{(3)}809 + 0^{(3)}292 \quad (9^{(5)}888)(9^{(5)}8096)(9^{(3)}877 \quad +$$

$$0^{(3)}191 \quad (9^{(6)}860)(9^{(6)}580)(9^{(3)}708)(9^{(3)}809)$$

$$R_{phase\ 2} = 9^{(3)}5170558 + (0^{(3)}292)(9^{(3)}8733976) + (0^{(3)}191)(9^{(3)}5164959)$$

$$R_{phase\ 2} = 9^{(7)}267$$

$$R_{phase\ 3} = 9^{(3)}128 \quad 9^{(3)}423 \ + 0^{(3)}872 \quad (9^{(6)}780)(9^{(6)}524) \ +$$

$$0^{(3)}577 \quad (9^{(6)}840)(9^{(6)}580)(9^{(3)}128)(9^{(3)}423)$$

$$R_{phase\ 3} = (9^{(2)}8551503) + (0^{(3)}872)(9^{(6)}304) + 0^{(3)}577 \ (9^{(2)}855)$$

$$R_{phase\ 3} = 9^{(6)}666$$

$$R_{\substack{mono\\config}} = R_{phase\ 1} \ R_{phase\ 2} = 9^{(5)}142 \quad 9^{(7)}267$$

$$\boxed{R_{\substack{mono\\config}} = 9^{(5)}135}$$

$$R_{\substack{bi\\config}} = R_{phase\ 1} \ R_{phase\ 2} \ R_{phase\ 3} = 9^{(5)}142 \quad 9^{(7)}267 \quad 9^{(6)}666$$

$$\boxed{R_{\substack{bi\\config}} = 9^{(5)}101}$$

APPENDIX C
MICROELECTRONICS PLANNING AND CONTROL

1. INTRODUCTION

A continuing study is in progress at TRW to determine the state of the art of monolithic integrated circuits and to justify the replacement of discrete parts with integrated circuits on a reliability basis.

In general, it is felt that this point has been reached for low power level repetitive digital functions such as are typified by digital operations, excluding memory, in the central sequencing and command subsystem. For this function approximately five and certainly no more than eight individual monolothic integrated-circuit types are required and are currently available. This small number makes it feasible to qualify such circuits on a timely basis for Voyager use. In addition to this class of digital circuits we have tentatively proposed the use of the Fairchild μ A702 analog DC amplifier in the stabilization and control subsystem. This circuit is in an advanced stage of verification for Vela and Apollo and promises improved reliability over its discrete-part equivalent.

Particular attention has been given to the Voyager mission specification which states:

> "Attempts to advance technology by using parts, materials, and processes which cannot demonstrate a history of reliability shall be prohibited (unless such advances are clearly necessary to meet minimum performance requirements)."

Except for this prohibition, TRW would have proposed more extensive use of integrated circuits than the very limited set indicated above.

The following discussion justifies the position that the selected integrated circuits are compatible with the mission specification constraint. It should also justify the position that as the Voyager spacecraft evolves through the various flight opportunities the reliability can be improved by expanding the use of integrated circuits.

2. TECHNOLOGICAL CONSIDERATIONS

The advent of silicon planar epitaxial passivation technology in the manufacture of semiconductor devices produced transistors and diodes whose reliability and versatility achieved unprecedented levels of success. In the "language" of part failure rates*, this generic class of devices has reached a well documented level of success characterized by:

Transistors	20 bits (i.e., 20×10^{-9} failures per hour)
Diodes	4 bits

In addition, this silicon planar technology made the monolithic integrated circuit a practical reality, which, for selected functions, is rapidly replacing its discrete counterpart. Integrated circuits can be viewed, therefore, as part of the continuum of the materials and processes of transistor technology. Their counterpart is the typical circuit "module" (such as welded circuit modules and sectors of printed circuits) currently used in electronic equipment. As shown in Figure C-1, a comparison of the technological tree of circuit module manufacture versus that for integrated circuits clearly demonstrates that:

a) The cumulative quantity of technological steps required by integrated circuits is considerably smaller than that required for circuit modules.

b) The variety of technologies required by circuit modules (i.e., resistors, capacitors, semiconductors) is much greater than that for integrated circuits.

c) The number of physical locations of fabrication activities for circuit modules is greater than for integrated circuits (i.e., the creation of a circuit module requires fabrication and procurement of a diverse array of parts and materials to be further fabricated into a module by the user, while an integrated circuit is essentially completed at a single manufacturer's facility).

*Failure rates were computed at an ambient temperature of 25°C and 25 per cent rated power.

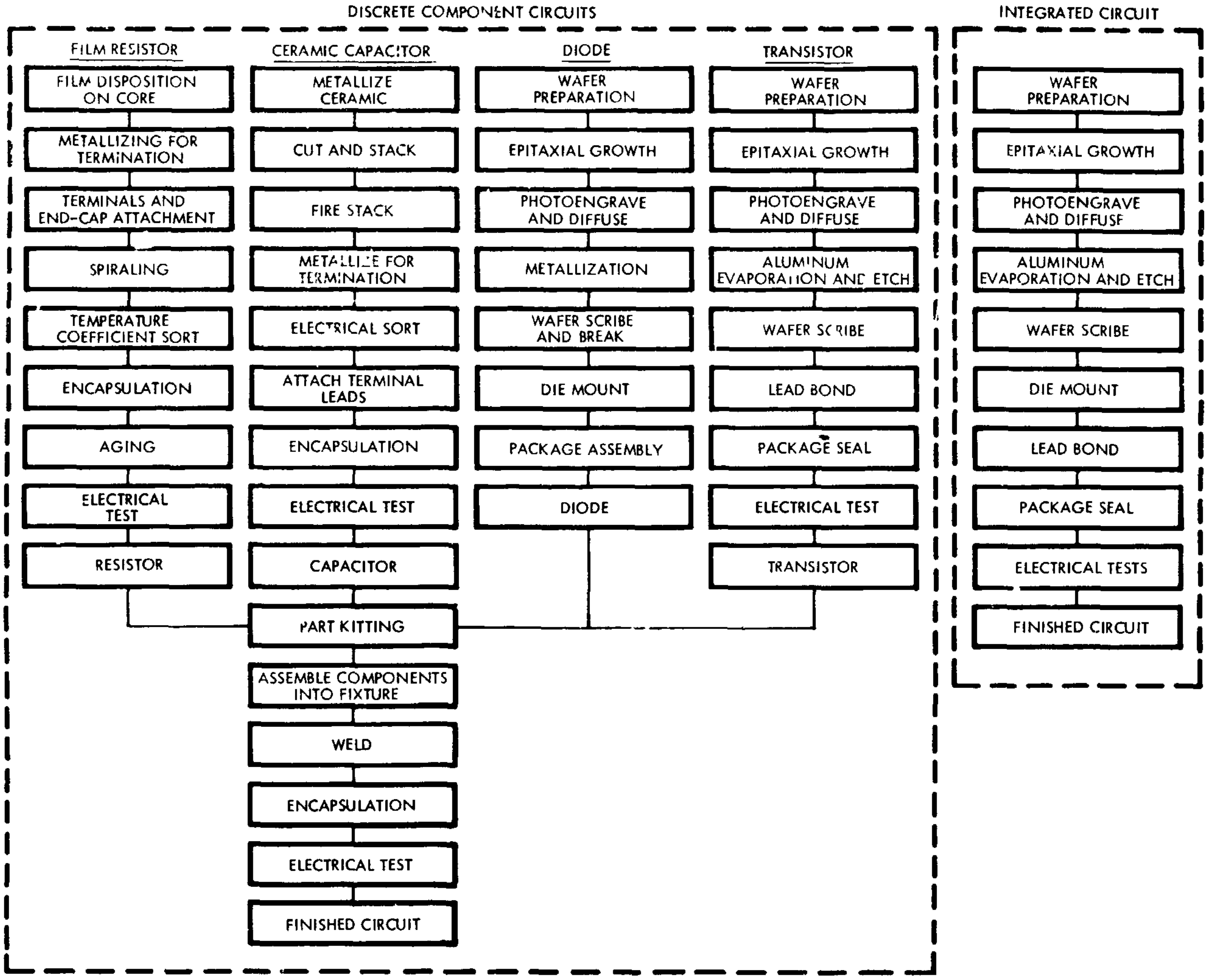


Figure C-1. Comparative Manufacturing Technology, Discrete Component Versus Integrated Circuit

Practical equipment production normally requires multiple sources for critical components. As Figure C-1 shows, the diversity of parts required for a typical welded module greatly increases the complexity of the multiple source problem in comparison with the relatively simple requirements for integrated circuits. The integrated circuit is a much more homogeneous product than the welded module; although both units have the same functional mission, the dispersion of physical and electrical properties of the module is far greater.

In considering the comparison of the two manufacturing technologies, the factor of declining versus ascending production lines is a significant consideration. This is illustrated by recent experience with the family of military parts specifications identified as the "38100 Series," which described the devices in the guidance computer of a major weapons system. The reasons for the release of these specifications were apparently quite valid: there had been a multi-million dollar program to "prove" these devices and issuing military specifications would make the parts available to general industry, thus increasing the benefits to be derived from past expenditures and efforts. However, attempts to purchase these parts from suppliers were largely futile because production lines had been closed down or were declining since the weapons system had moved on to a new design using integrated circuits. Many of the suppliers, of course, were willing to sell the devices at the cost (amortized in parts prices) of reopening closed production activities. Therefore, a decision based solely on the use of "proven" parts requires careful consideration of technological dynamics. Specifically, the choice of discrete electronic parts for digital electronic circuits to be produced during the 1966 to 1970 period is apt to require procurement of devices from declining production activities and technologies, while the choice of integrated circuits will permit procurement from ascending production lines.

3. HUMAN FACTORS

The consequences of the technological differences between integrated circuits and circuit modules identify tabulations of human factors which are distinctly different:

	Integrated Circuit	Circuit Module
Handling and production fragility	Approximately 1 semiconductor	Cumulative of resistors + capacitors + diodes + transistors + assembly + modules
Testing errors	Ratio of probabilities indeterminate but considered clearly in favor of integrated circuits e.g., chances of surge damage	

Surveillance of sources of supply	Concentration can be effected on fewer suppliers	Concentration required on many more suppliers, including "in-house" production facilities

The RFP for the Voyager Phase IA study required an answer to the following:

> "Describe your approach to life-test verification of flight spacecraft hardware as a function of funding available and define criteria for successful completion of life tests."

Let us assume a hypothetical subsystem of spacecraft hardware comprised of a family of digital hardware. We can assume a fixed funding situation in which an attempt is made to provide the system with its family of hardware as reliably as possible. Consider an equation based upon the query,

$$\left(\sum_{1}^{n} \$ \text{ process steps}\right) + \left(\sum_{1}^{m} \$ \text{ control steps}\right) = \text{constant funding}$$

clearly $$\sum_{1}^{n_1} \$\begin{pmatrix}\text{process steps}\\ \text{for circuit}\\ \text{modules}\end{pmatrix} > \sum_{1}^{n_2} \$\begin{pmatrix}\text{process steps}\\ \text{for integrated}\\ \text{circuits}\end{pmatrix}$$

thus $$\sum_{1}^{m_1} \$\begin{pmatrix}\text{control steps}\\ \text{for integrated}\\ \text{circuits}\end{pmatrix} > \sum_{1}^{m_2} \$\begin{pmatrix}\text{control steps}\\ \text{for circuit}\\ \text{modules}\end{pmatrix}$$

The funding available for controlling the fewer process steps of integrated circuits configuration under a constant total funding situation is far in excess of that available for circuit modules using discrete parts.

4. DEVELOPMENT ENGINEERING

Referring to Figure C-2 the following general observations may be tabulated:

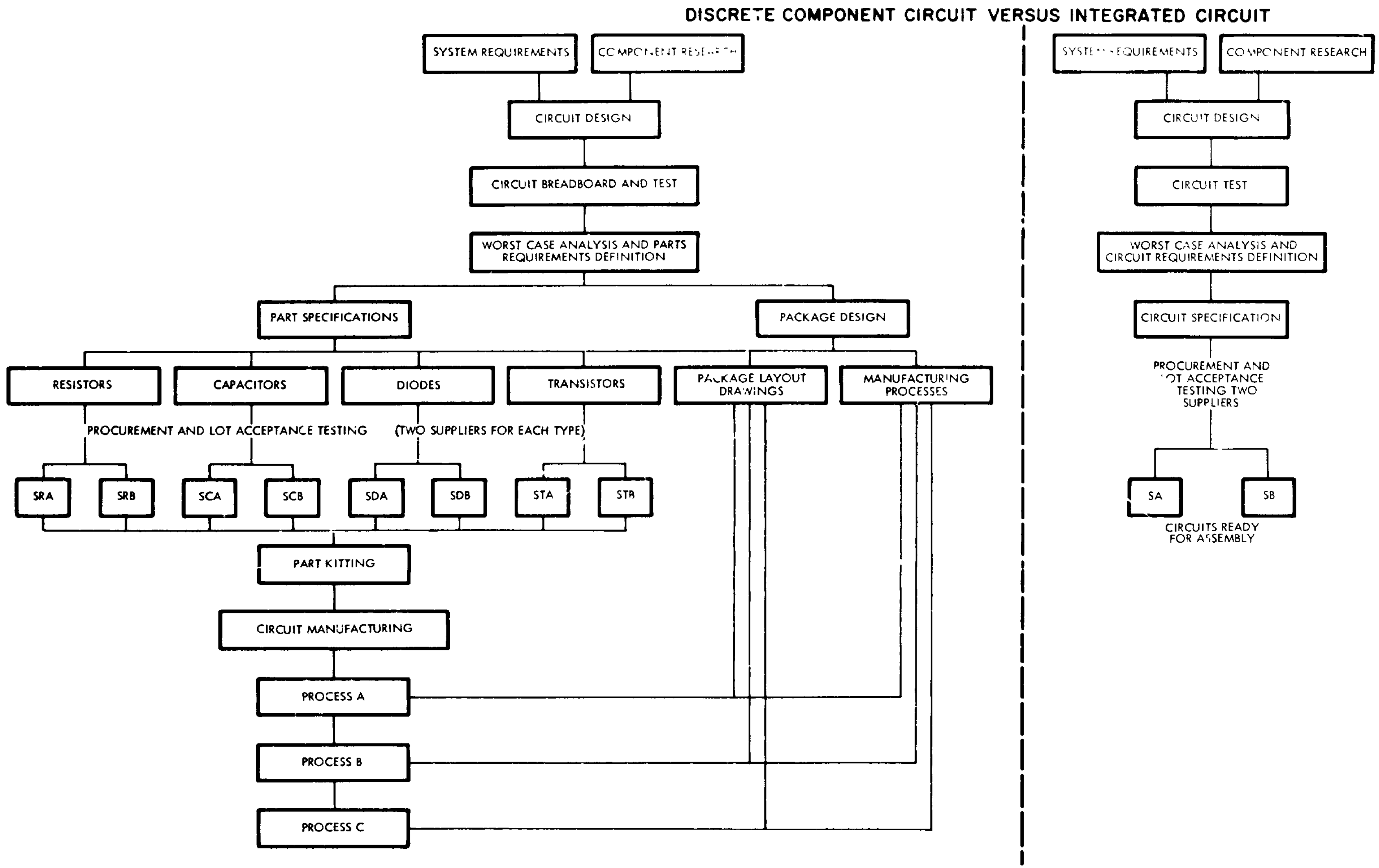


Figure C-2. Comparison of Development Engineering Factors

Activity	Circuit Modules	Integrated Circuits
Circuit and breadboard development	Order and evaluate parts: develop digital circuits; fabricate and evaluate circuit modules	Evaluate integrated circuits
Margins testing	Worst case parts insertion (limit devices); worst case power supply and thermal variations	Worst case power supply and thermal variations
Packaging	Parts selection; packaging design, layout; materials and processes evaluation; prototype package tests	Package predefined and qualified
Misapplication probabilities	Circuit misapplication of parts; physical misapplications of parts	Standard integrated circuits less likely to contain misapplied elements due to Voyager standardization of integrated circuits

In general, development engineering factors indicate an opportunity (analogous to the case cited in Section 3) to effect greater visibility to fewer activities needed to arrive at comparable levels of assembly. The tangible results appear to favor a significant reliability advantage for integrated circuits.

5. SYSTEMS ENGINEERING FACTORS

5.1 Redundancy

In a given application where the available power and weight allowances are permissive of the utilization of either discrete elements or integrated circuits, a distinct advantage is still obtained by the use of integrated circuits via the use of redundancy. TRW studies indicate this advantage to be applicable to Voyager.

5.2 System Mechanical Design

The over-all physical design of a digital subsystem, being drastically simplified and reduced in size by the use of integrated circuits, has "simpler" mechanical and thermal properties. Thus the analyses of shock and vibration transmissibilities and thermal patterns are drastically reduced and available to increased scrutiny. This advantage is also available for selected analog circuits where residual effects are low, matched characteristics are critical, and functions are highly standardized. Certain operational amplifiers are in this category but must be expected to have somewhat lower confidence or higher failure rates for equal complexity.

6. LOGISTIC FACTORS

Referring to Table C-1 it should be noted that the simplification of the problems of acquisition of parts and materials via the reduction of variety of required sources of supply again permits more intense concentration on fewer activities. Associated with this is simplification of parts traffic patterns (shipping, receiving, handling), specification negotiations, and vendor surveys.

Table C-1. Logistics Comparison, Discrete Component Versus Digital Integrated Circuit

Resistor Supplier A Resistor Supplier B Capacitor Supplier A Capacitor Supplier B Diode Supplier A Diode Supplier B Transistor Supplier A Transistor Supplier B	Facility Survey Specification Negotiation Qualification Test Production Monitor Lot Acceptance Test	DISCRETE PARTS 40 steps to receipt of parts for flight hardware
Integrated Circuit Supplier A Integrated Circuit Supplier B	Facility Survey Specification Negotiation Qualification Test Production Monitor Lot Acceptance Test	INTEGRATED CIRCUITS 10 steps to receipt of complete flight circuits

7. STATISTICAL DATA

Statistical reliability data for integrated circuits is divided as follows:

a) Table C-2 is a tabulation of the failure rates for integrated circuits experienced in four digital configurations; these rates were reported to the TRW Systems Reliability Staff. It should be noted that the composite "laboratory" experience indicates a failure rate of about 87 x 10^{-9} failures/hour of integrated circuit technology of the 1964 vintage.

b) Figure C-3, taken from a TRW Systems Reliability Staff study, illustrates the anticipated change in λ for integrated circuits to 1970. This study estimates that digital integrated circuit technology will be characterized by $\lambda \cong 40 \times 10^{-9}$ failures/hour by 1965; about 1970, a figure of $\lambda \cong 15 \times 10^{-9}$ failures/hour will be approached (i.e., 15 bits).

c) Table C-3 is a tabulation of the failure rates under two reference conditions for the discrete circuit elements commonly found in computer logic.

Table C-2. Tabulation of Digital Integrated Circuit Life Test Data and Resulting Average Bit Failure Rate Estimate

System	Circuit	Test Hours	No. Devices	No. Failures	Observed in %/1000 hours
Apollo Guidance	Fairchild RTL	19×10^6	--	1	0.0053
Magic I Airborne Computer	Fairchild RTL	15.25×10^6	--	2	0.0131
Airborne PCM Computer	Texas Instrument Series 51 and 52	--	280	--	0.0711
Grumman E-ZA Aircraft Tactical Early Warning System	Texas Instrument Series 51	--	30,000	--	0.089
Nominal Average Digital Integrated Circuit Failure Rate (including additional sources)					0.0087
Predicted Average Digital Integrated Circuit Failure Rate for January 1965 (See Figure C-3)					0.004

Table C-3. Tabulation of Bit Failure Rates for Discrete Components in a Digital Application (Hi-Rel Procurement)

Part Type	Bit Failure Rate	
	T_A = 50°C and 40% Rated Power	T_A = 25°C and 25% Rated Power
Silicon diode	8	4
Silicon Transistor	30	20
Resistor, carbon composition	1	1
Resistor, metal film	10	5
Capacitor, fixed, ceramic	20	6
Capacitor, fixed, glass	20	4
Connection, welded	0.5	0.5
Connection, soldered	0.5	0.5

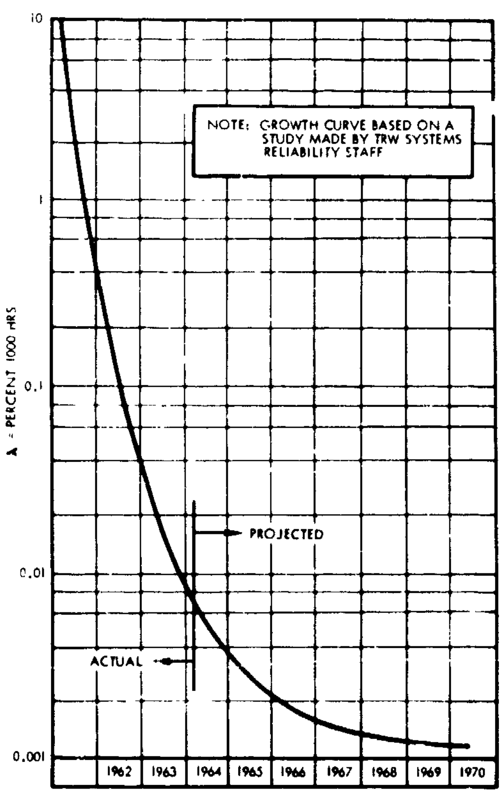
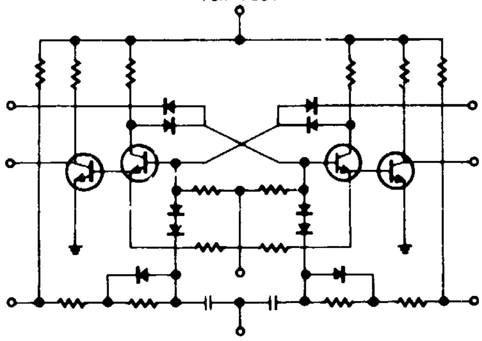


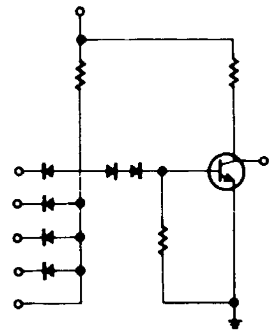
Figure C-3. Integrated Monolithic Circuits Reliability Growth

d) Figure C-4 contains two representative logic circuits for which λ had been computed based on the preceeding data. Using the assumptions in Paragraphs a, b, and c above, the comparative reliability of an integrated circuit versus its circuit module counterpart is indicated to be:

EXAMPLE A
FLIP-FLOP



EXAMPLE B
4 INPUT DTL GATE



ESTIMATE OF DISCRETE COMPONENT
FLIP-FLOP RELIABILITY*

COMPONENT	NO.	BIT RATE EACH	TOTAL BIT RATE
RESISTORS	14	1	14
CAPACITORS	2	4	8
DIODES	10	4	40
TRANSISTORS	4	20	80
TOTAL	30		142
AVERAGE DIGITAL IC BIT RATE**			40
IMPROVEMENT WITH IC (1965)			102

ESTIMATE OF DISCRETE COMPONENT
4-INPUT GATE RELIABILITY*

COMPONENTS	NO.	BIT RATE EACH	TOTAL BIT RATE
RESISTORS	3	1	3
DIODES	6	4	24
TRANSISTORS	1	20	20
CONNECTIONS	10	0.5	5
TOTAL	20		52
AVERAGE DIGITAL IC BIT RATE**			40
IMPROVEMENT WITH IC (1965)			12

*COMPONENT BIT RATES TAKEN FROM TRW SYSTEMS RELIABILITY MANUAL

**AVERAGE DIGITAL IC BIT RATES TAKEN FROM "MICROELECTRONIC RELIABILITY" BY THE TRW SYSTEMS RELIABILITY STAFF (REPORT NO 4303-6001-TU000)

Figure C-4. Comparison of Estimated Failure Rates, Digital Integrated Circuit Versus Discrete Component Equivalent Circuit

Example A, Flip Flop

λ (circuit module, 1965 projected) $\cong 142 \times 10^{-9}$ failures/hour

λ (integrated circuit, 1965 projected) $\cong 40 \times 10^{-9}$ failures/hour

Example B, 4 Input DTL Gate

λ (circuit module, 1965 projected) $\cong 52 \times 10^{-9}$ failures/hour

λ (integrated circuit, 1965 projected) $\cong 40 \times 10^{-9}$ failures/hour

e) Referring again to Figure C-3, it should be noted that the slope of the integrated circuit reliability curve is expected to show significant reliability improvement throughout the remainder of the 1965-1966 period. The projection of its circuit module counterpart is expected to remain substantially constant during this period thus assuring an increasing margin of improvement.

f) Reliability bit failure rate data with its basis on highly variable information has, nevertheless, a reasonable pattern of consistency and agreement with systems experience. Using the rationale of bit failure rates, it is indicated that technology has currently reached a reliability crossover point in the assessment of integrated circuits versus circuit modules for digital applications; in the future, integrated circuits will be favored for selected functions.

8. QUALITY ASSURANCE DATA

8.1 Comparison of "Strengths" and Fragilities Profiles

Table C-4 tabulates some comparative strengths of integrated circuits versus circuit modules. Although no attempt can be made to translate these differences into reliability estimates, the profile indicated for integrated circuits is in excess of that for circuit modules.

Table C-4. Comparison of Stress Capabilities of Discrete Component Circuit Modules Vs. Integrated Circuits

STRESS	Circuit Module Capabilities	Integrated Circuit Capabilities
Mechanical shock	200 g	20,000 g
Vibration	5 to 15 g, 2000 cps	206 g, 5 to 2000 cps
Constant acceleration	200 g	40,000 g
Thermal cycle	-40 to +70°C	-65 to +150°C
High temperature storage	70°C	200°C
Moisture resistance	10 days per MIL-STD-202, Method 106	10 days per MIL-STD-750, Method 1056.1
Hermeticity	Non-hermetic	Leak rate: 5×10^{-8} cc/sec
Vibration fatigue (96 hours)	15 g, 60 cps	20 g, 60 cps

8.2 Industrial Quality Assurance Levels

Through detailed specification negotiations and vendor commitments, TRW Systems has confirmed that the semiconductor industry can and will supply integrated circuits to quality assurance specifications substantially equivalent to high reliability discrete transistors and diodes. Primarily because integrated circuit technology is an extension of transistor technology, both classes of devices are available to the following composite requirements:

Load life tests (maximum ratings)	$\lambda = 5\%$	
Accelerated storage life tests (maximum ratings)	$\lambda = 5\%$	
Shock, vibration, centrifuge (per MIL-S-19500C)		LTPD = 10% cumulative
Thermal shock, moisture resistance, temperature cycling, (per MIL-S-19500C)		LTPD = 10% cumulative

It should be noted, however, that typical quality assurance life testing of integrated circuits is currently performed at considerably lower thermal or dissipative levels than those used for discrete diodes and transistors. Consequently, industrial quality assurance levels which certify performance to $\lambda = 5\%/1000$ hr for integrated circuits should be interpreted as corresponding to a more conservative estimate of "in use" reliability.

8.3 Screening Techniques and Capabilities

Figure C-5 and Tables C-5 and C-6 illustrate the screening techniques required for 100 per cent inspection in TRW Systems integrated circuit specifications, together with their relationship to the current tabulation of failure modes. As with other devices, screening techniques for integrated circuits have their basis in well-founded experience and reasoning with failure mechanisms, coupled with the techniques of electrical burn-in, environmental testing, and parameter selection. The practice of coupling parameter selection with monitoring degradation sensitive parameters forms the foundation for highly effective parameter drift screening requirements.

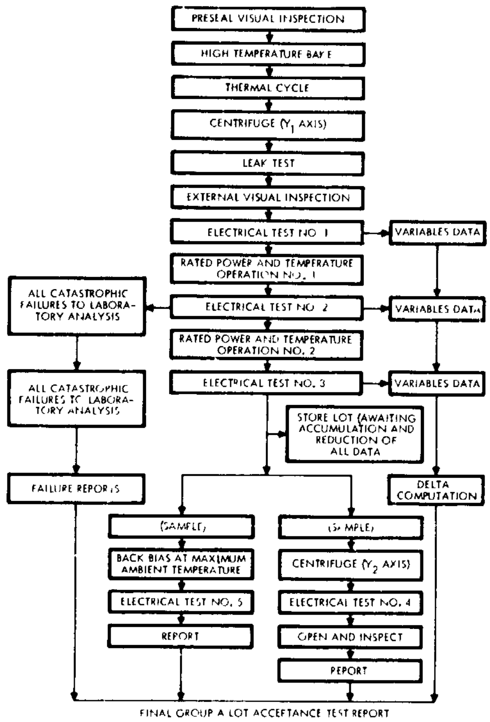


Figure C-5. Integrated Circuit Screening Flow Chart

Table C-5. Description of Integrated Circuit Screening Steps

Preseal visual inspection	All circuits are carefully inspected under 20 and 80X magnification and defective circuits eliminated. Detailed inspection criteria shall be generated for defective circuits.
High temperature bake	150 to 200°C for 168 hours minimum; circuit nonoperating.
Thermal cycle	-55 to + 150°C, 3 cycles minimum per MIL-STD-750, Method 1056, Condition B or equivalent.
Centrifuge (Y_1 or Y_2 Axis)	MIL-STD-750, Method 2006 or equivalent for specified axis; A = 40,000 g
Leak test	As verified by both a gross and fine leak test, the leak rate shall be less than 5 x 10^{-8} cm^3/sec.
Electrical Test No. 1	Read and record all specified electrical parameters.
Electrical Tests No. 2, 3, 4, and 5.	Read and record certain degradation sensitive parameters.
Variables data from Tests No. 2, 3, 4, and 5 and Delta Computations	Variables data identifiable to specific devices will be taken for each device at each test and for each parameter, and the delta in each parameter will be computed and recorded for successive tests. The distribution of deltas will be determined at each test and deltas will be considered failures. An allowable defect rate of 5% will be imposed on a cumulative basis.
Rated power and temperature operation No. 1	At an ambient temperature of 125°C with maximum rated supply voltages applied, the circuits will be operated in a ring counter configuration for 240 hours.
Rated power and temperature operation No. 2	Same operating and ambient conditions as Operation No. 1 above except operating time will be 760 hours.
Back bias at maximum ambient operating temperature	Apply rated Dc voltages to selected terminals in such a way as to back bias the largest number of junctions in the circuit. Store at 125°C for 168 hours.
Open and inspect	Open circuits and inspect for deterioration as a result of previous screening steps and for workmanship defects.
Failure analysis on catastrophic failures	All opens, shorts or otherwise seriously degraded circuits which have failed during the screening process will be thoroughly analyzed and their failure mode cataloged. The appearance of unpredictable and/or previously unidentified failure modes shall be cause for lot rejection.

Table C-6. Tabulation of Integrated Circuit Failure Modes

Major Failure Modes in Integrated Circuits		Effective Screens
Open Bonds to Chip		
Caused by:	Poor metalization adherence Under-bonding Over-bonding Gold-aluminum eutectic	Preseal visual inspection High temperature bake Thermal cycle Centrifuge (Y_1 axis)
Open Leads		
Caused by:	Nicks or cuts in lead Thinning of lead at bond	Preseal visual inspection Centrifuge (Y_1 and Y_2 axis) Vibration
Leads Shorting		
Caused by:	Excessive lead length	Preseal visual inspection Centrifuge (Y_2 axis) and electrical test Centrifuge (Y_1 axis) and electrical test Vibration noise
Open Bonds to Terminal		
Caused by:	Poor bonding technique	Preseal visual inspection Centrifuge
Opens in Chip Metallization		
Caused by:	Scratches Gold-aluminum eutectic Metallization deterioration at oxide steps Deterioration of aluminum to silicon contact at oxide window Metallization corrosion	Preseal visual inspection High temperature bake and electrical test Extended operation at rated power and ambient temperature and electrical test
Shorts on Chip		
Caused by:	Oxide breakdown Metallization smear Poor bond placement Misregistration of marking Metallic particles in package	Preseal visual inspection Extended operation at rated power and ambient temperature and electrical test.
Catastrophic Failure		
Caused by:	Cracked chip	Preseal visual inspection Thermal cycle Centrifuge
Severe Electrical Degradation		
Caused by:	Surface channeling Surface contamination	Extended operation at rated power and ambient temperature.

TRW Systems' capabilities for performing screening inspection of integrated circuits and for developing the experimental foundations of screening formulas are necessary measures to insure the continuous confirmation of the receipt of integrated circuits commensurate with reliability assessments.

At the present time, discrete parts probably have an advantage over integrated circuits with respect to screening efficiency. This is primarily due to the accessibility of each electrode of each circuit element, which makes incipient drifts and circuit element parameter variations more discoverable. Improved screening techniques for integrated circuits require development of methods to circumvent this disadvantage. Analogous test equipment for discrete parts has a more detailed diagnostic capability than integrated circuit test equipment; again compensatory techniques are required. Since such techniques are not fully established, screening methods for discrete parts are currently probably more sophisticated. To some extent, the greater sensitivity of present screening techniques for discrete parts probably yields an advantage in identifying failure modes and corrective action procedures. The assembly of discrete parts into a circuit module, however, tends to remove this inequality to the extent that electrode inaccessibility is re-established.

9. CONCLUSIONS

The use of selected integrated circuits in modern electronic hardware is rapidly becoming consistent with the most advanced design techniques and with stringent requirements for a demonstrated history of reliability. It is necessary, therefore, that any planning for electronic equipment to be designed and manufactured during the 1966 to 1970 period include utilization of integrated circuits. Conversely, it is strongly believed that if such planning were to prohibit the use of integrated circuits and enforce the use of discrete parts only, equipment design and manufacture would be seriously hampered by declining production lines, technological obsolescence, and increasingly limited reliability.

APPENDIX D

NOMINAL 1971 TRAJECTORY AND ORBIT

This appendix serves to define and describe sample earth-Mars trajectories for the 1971 mission, and a sample spacecraft orbit about Mars. These samples are used throughout this report as background for many different analyses.

1. INTERPLANETARY TRAJECTORY

Figure D-1 shows the basic interplanetary trajectory constraints imposed by the Preliminary 1971 Voyager Specification against the coordinates, launch date, and arrival date. These constraints, indicated by shading, represent these parameter limits:

• Launch energy	$C_3 \leq 18$ km^2/sec^2
• Arrival asymptotic velocity	$V_\infty \leq 5$ km/sec
• Declination of launch asymptote	$5^\circ \leq \lvert DLA \rvert \leq 33^\circ$
• Inclination of transfer plane	$INC \geq 0.1^\circ$

Sample transfer trajectories, No. 1 through No. 6, are also indicated in Figure D-1. The launch and arrival dates for these six trajectories are listed in Table D-1. In this appendix, when "the nominal trajectory" or "the sample trajectory" is used, the one specified is No. 3, with launch date May 19, 1971, and arrival date November 12, 1971.

The geometry of this interplanetary trajectory is shown in Figure D-2, and the variation of geometrical quantities with time is shown in Figure D-3, for the spacecraft in interplanetary transfer and in orbit about Mars. Trajectory No. 3, compared with others of the 1971 opportunity, exhibits relatively low launch energy requirements and low arrival V_∞. Other characteristics are given in Table D-2.

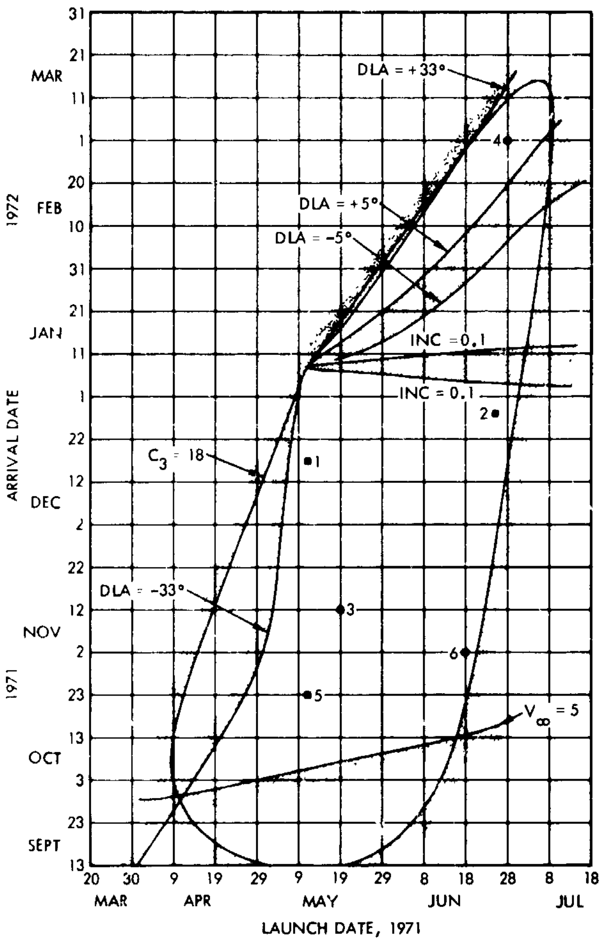
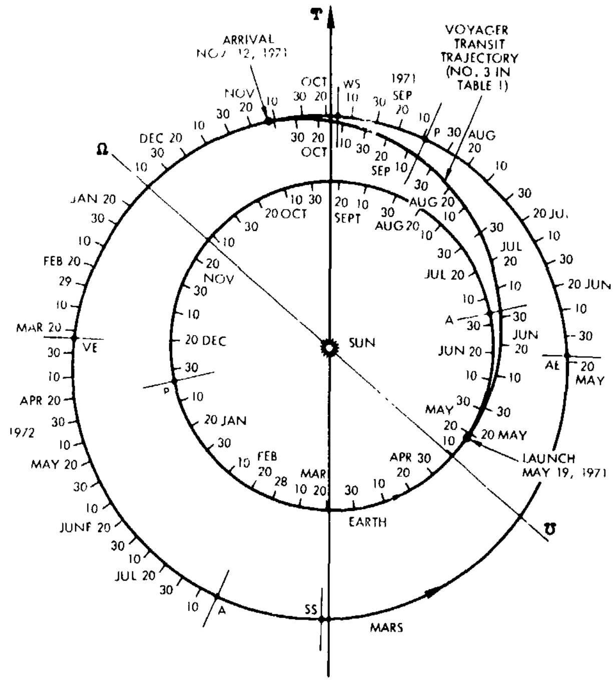


Figure D-1. 1971 Earth-Mars Trajectories

Table D-1. 1971 Earth-Mars Trajectories

Trajectory Number	Launch Date	Arrival Date
1	May 11, 1971	December 17, 1971
2	June 25, 1971	December 28, 1971
3	May 19, 1971	November 12, 1971
4	June 28, 1971	March 1, 1972
5	May 11, 1971	October 23, 1971
6	June 18, 1971	November 2, 1971



LEGEND		
Ω	ASCENDING NODE OF MARS ORBIT	
℧	DESCENDING NODE OF MARS ORBIT	
A	APHELION	
P	PERIHELION	
VE	VERNAL EQUINOX	NORTHERN HEMISPHERE
SS	SUMMER SOLSTICE	NORTHERN HEMISPHERE
AE	AUTUMNAL EQUINOX	NORTHERN HEMISPHERE
WS	WINTER SOLSTICE	NORTHERN HEMISPHERE

Figure D-2. Voyager Sample Trajectory

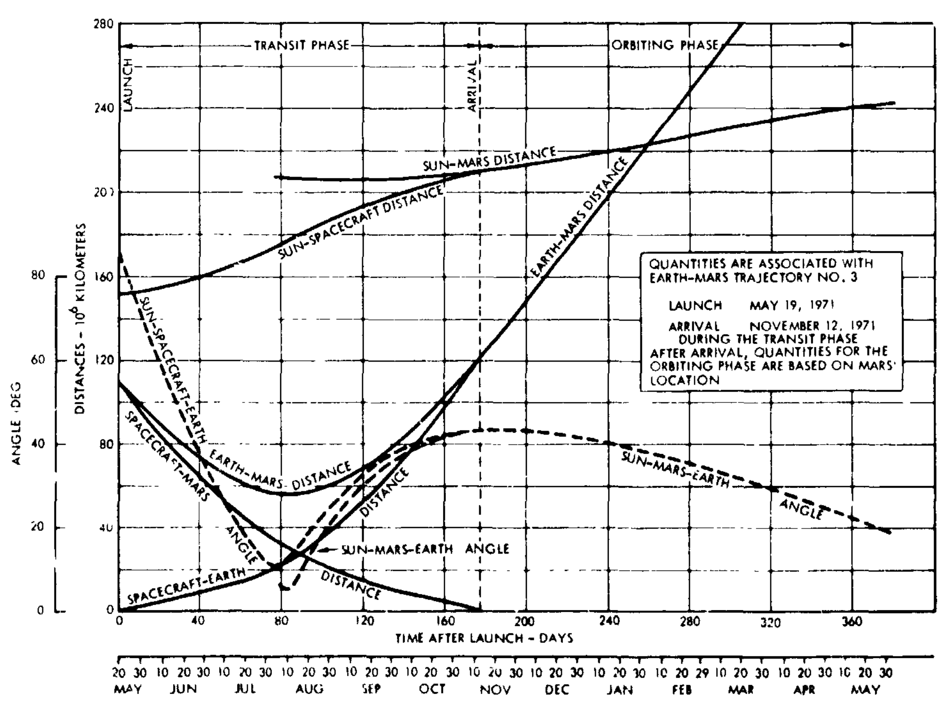


Figure D-3. Geometrical Quantities Versus Time

Table D-2. Characteristics of Earth-Mars Trajectory No. 3

Launch date	May 19, 1971
Arrival date	November 12, 1971
Time of flight	177 days
Departure asymptote (from earth)	
V_{∞}	2.92 km/sec
C_3	8.53 km^2/sec^2
Angle to ecliptic	-16 deg
Angle to sun-earth line	88 deg
Approach asymptote (to Mars)	
V_{∞}	3.25 km/sec
Angle to plane of Mars' orbit	-3 deg
Angle to Mars-sun line	119 deg
Interplanetary Orbit	
True anomaly at arrival	142.5 deg
True anomaly at launch	4.5 deg
Heliocentric central angle	138 deg
Inclination to ecliptic	1.5 deg
Perihelion distance from sun	151.2×10^6 km
Aphelion distance from sun	220.5×10^6 km
Eccentricity	0.1853

2. ORBIT ABOUT MARS

The nominal orbit about Mars, used throughout this report, is entered by a periapsis-to-periapsis transfer from interplanetary trajectory No. 3. It has this basic definition:

Inclination to Mars' equator	45 deg
Altitude at periapsis	2,000 km
Altitude at apoapsis	20,000 km

The nominal orbit proceeds in an easterly direction, and the initial passage takes place over the sunlit side of Mars' southern hemisphere. To determine the characteristics of this orbit, the following constraints were assumed:

Radius of Mars	3,330 km
Gravitational constant of Mars	42,920 km^3/sec^2

Figure D-4 shows the geometry of the hyperbolic approach trajectory and the elliptical orbit. For the approach, the following quantities apply:

Periapsis distance from Mars' center, r_p	5,330 km
Asymptotic approach velocity, V_∞	3.250 km/sec
Velocity at periapsis (areocentric)	5.163 km/sec
Eccentricity, e_h	2.312
ϵ (see Figure D-4)	64.37 deg
a_h (see Figure D-4)	4,063 km
Impact parameter, B	8,472 km

For the elliptical orbit, the following quantities apply:

Periapsis distance from Mars' center, r_p	5,330 km
Apoapsis distance from Mars' center, r_a	23,330 km
Semi-major axis, a	14,330 km
Eccentricity, e	0.6280
Velocity at periapsis	3.620 km/sec
Velocity at apoapsis	0.827 km/sec
Period of orbit	14.46 hours

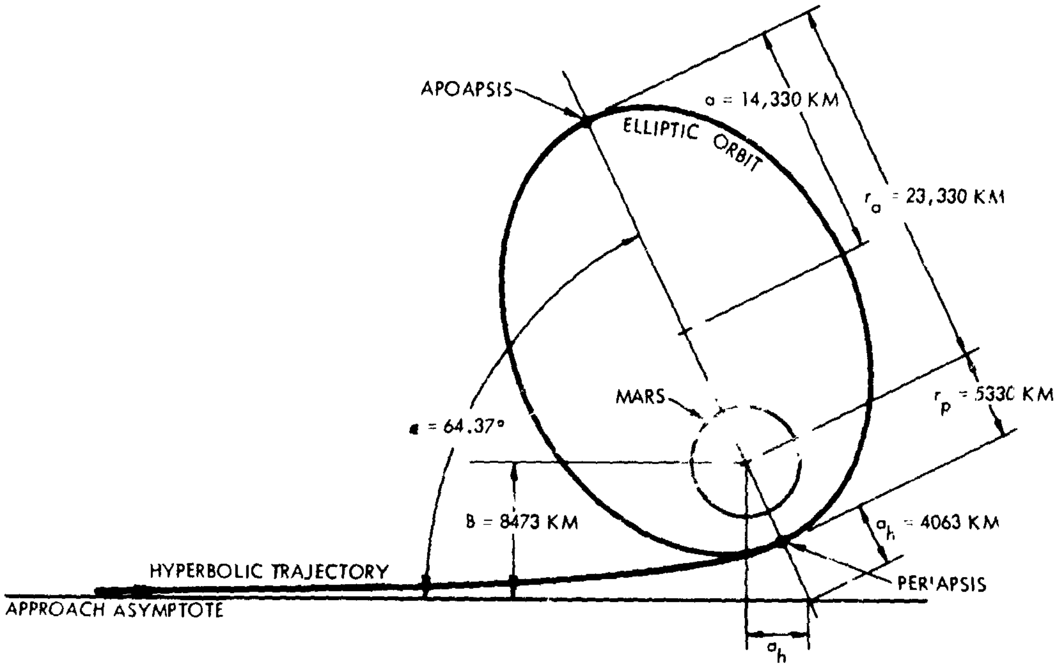


Figure D-4. Nominal Approach Trajectory
and Orbit Geometry

The impulsive retropropulsion required for periapsis-to-periapsis insertion has these characteristics:

Specific impulse (assumed), I_{sp}	300 sec
Velocity increment, ΔV	1.543 km/sec
Mass ratio (initial to final)	1.690

The orientation of the orbit plane is described by these angular measurements of the direction of its north pole (as of the arrival date):

Cone angle (sun-Mars-orbit pole)	69 deg
Clock angle (from Canopus reference, clockwise about Mars-sun line)	176.1 deg
Inclination of orbit plane to Mars' equator	45 deg
Inclination of orbit plane to Mars' orbit plane	26.7 deg

Figure D-5 illustrates the orientation of the orbit plane and other characteristics of the orbit on a Mercator projection of the celestial sphere referenced to Mars' equatorial coordinates and gives these characteristics at the time of arrival. Figures D-6 and D-7 show the same properties 90 and 180 days later, respectively. For Figures D-6 and D-7, the sun and earth (indicated by S and E) have progressed eastward along the Martian "ecliptic," and the orbit plane has regressed westward. Apsidal advance has not changed the right ascension of periapsis significantly, but its declination has progressed northward from -27.7 degrees at arrival to -4.9 degrees 130 days later.

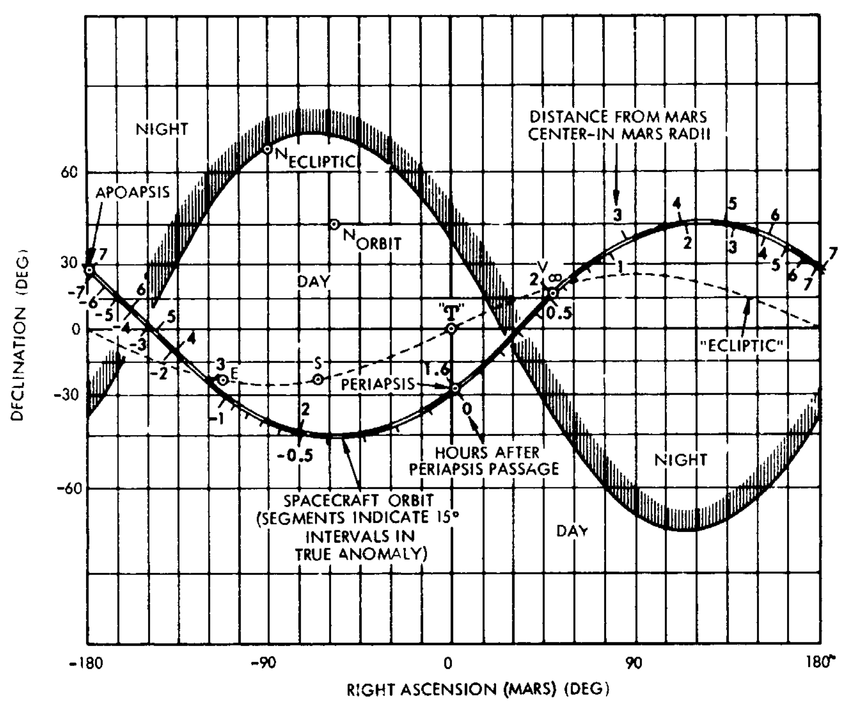


Figure D-5. Nominal Orbit at Arrival

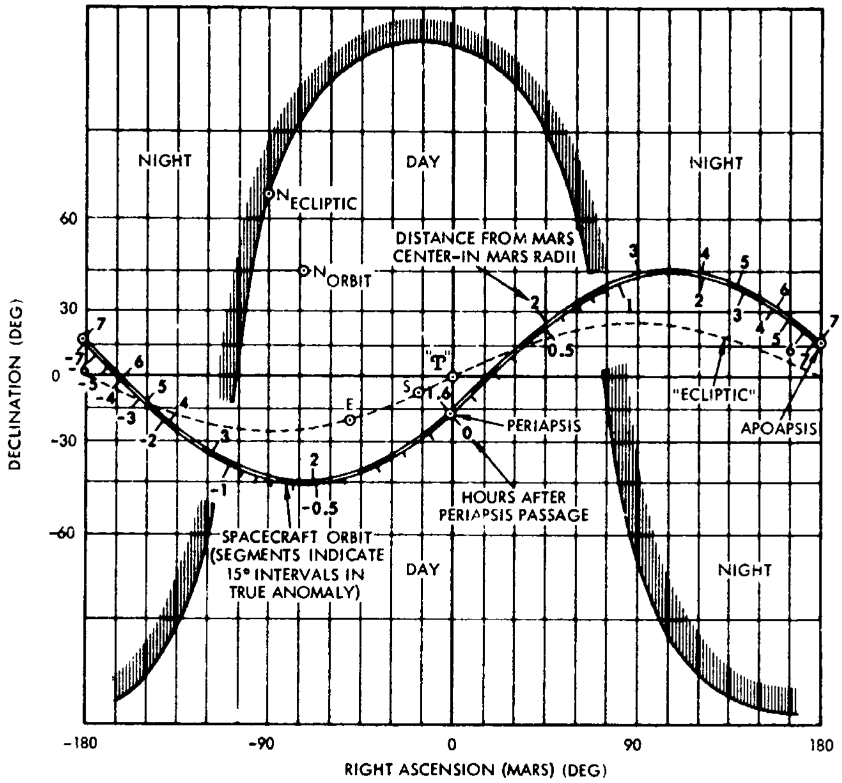


Figure D-6. Nominal Orbit 90 Days after Arrival

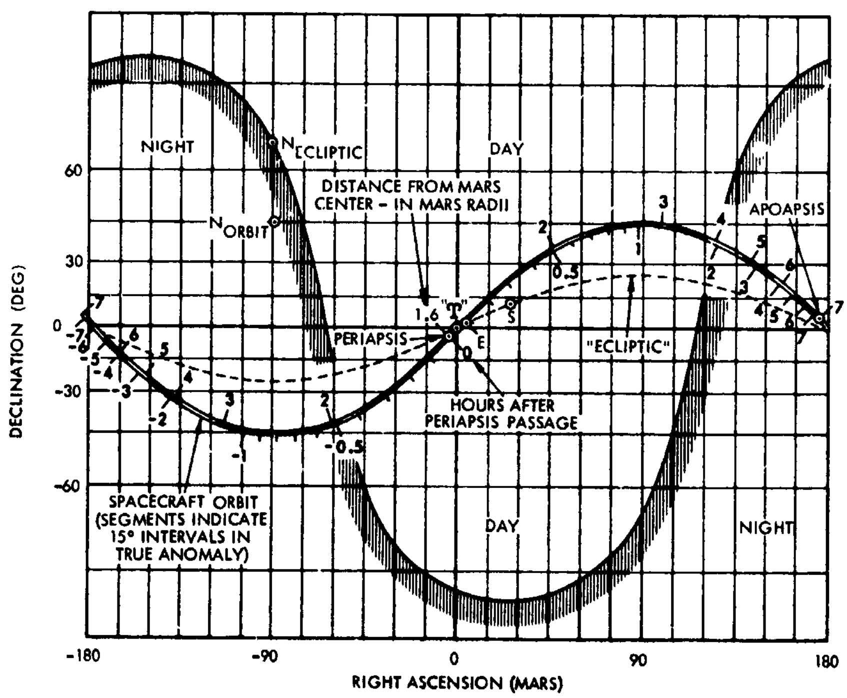


Figure D-7. Nominal Orbit 180 Days after Arrival

APPENDIX E

DOPPLER AND DOPPLER RATE LIMITS FOR VOYAGER ORBITER-TO-EARTH COMMUNICATIONS

Doppler frequency shift and the rate of change of the doppler frequency shift for one- or two-way communication between two points are proportional to the first and second time derivatives of the distance between the two points, i.e., to the components of relative velocity and relative acceleration along the line connecting the two points. This appendix determines limiting values for the relative axial velocity and acceleration for the communication path between the Voyager spacecraft in orbit about Mars and a DSN (Deep Space Network) station on the earth.

The orbiter-ground station distance is the sum of three parts:

a) (Due to the orbit) The component of the vector $\bar{r}$ from the center of Mars to the orbiting spacecraft which lies along the earth-Mars line.

b) The distance from the center of Mars to the center of the earth.

c) (Due to the ground station) The component of the vector from the center of the earth to the DSN station which lies along the earth-Mars line.

The contributions of these three sources to the corresponding velocity and acceleration are evaluated separately.

1. DUE TO THE ORBIT

The coordinates describing the spacecraft in orbit are shown in Figure E-1. It can be shown that the component of $\bar{r}$ along the z-axis (the earth-Mars direction) is

$$z = \cos \psi = \frac{a\,(1 - e^2) \sin i \sin (\omega + v)}{1 + e \cos v},$$

and that the derivative of z is

$$\dot{z} = \sqrt{\frac{\mu}{a}} \frac{\sin i}{\sqrt{1 - e^2}} \left[\cos(\omega + v) + e \cos \omega\right]$$

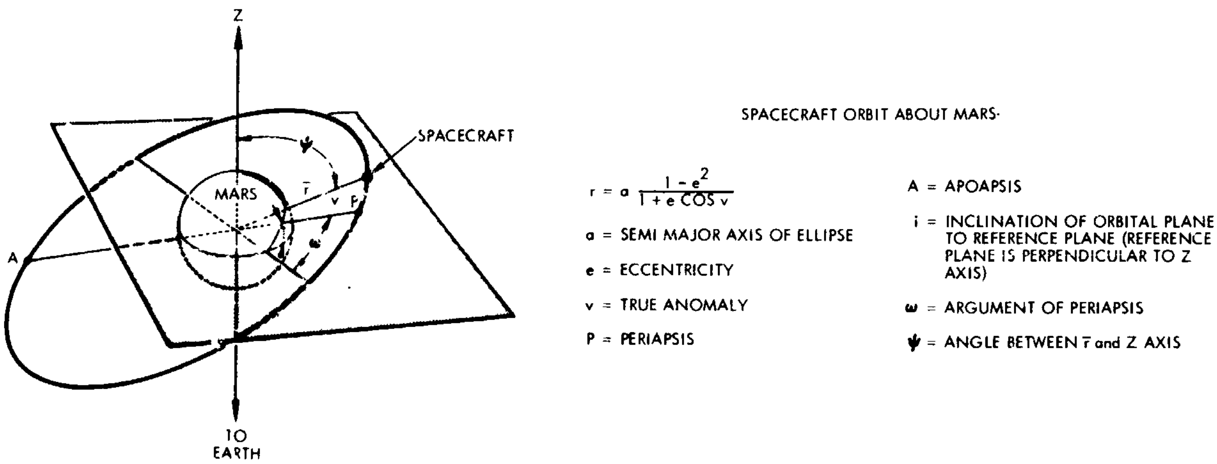


Figure E-1. Geometry of Orbit about Mars

μ is the gravitational parameter of Mars, and equals 42,920 km^3/sec^2. Of interest here are the maximum value of $\dot{z}$ and the range $(\dot{z}_{max} - \dot{z}_{min})$ as v varies through 360 degrees. These are related to the maximum doppler shift from the mean frequency, and the maximum total doppler frequency range, respectively, and are given by

$$|\dot{z}|_{max} = \sqrt{\frac{\mu}{a}} \frac{\sin i}{\sqrt{1 - e^2}} (1 + |e \cos \omega|),$$

$$(\dot{z}_{max} - \dot{z}_{min}) = 2\sqrt{\frac{\mu}{a}} \frac{\sin i}{\sqrt{1 - e^2}}$$

A second differentiation results in

$$\ddot{z} = - \frac{\mu \sin i \sin(\omega + v) (1 + e \cos v)^2}{a^2 (1 - e^2)^2}$$

Again we are interested in the maximum value of $\ddot{z}$ as v varies through 360 degrees.

$$\ddot{z}_{max} = \frac{\mu \sin i}{a^2 (1 - e^2)^2} F(\omega, e),$$

where F (ω, e) is the maximum value attained by

$$f(\omega, e, v) = \left| \sin(\omega + v) (1 + e \cos v)^2 \right|$$

as v is varied. The function F (ω, e) is illustrated in Figure E-2.

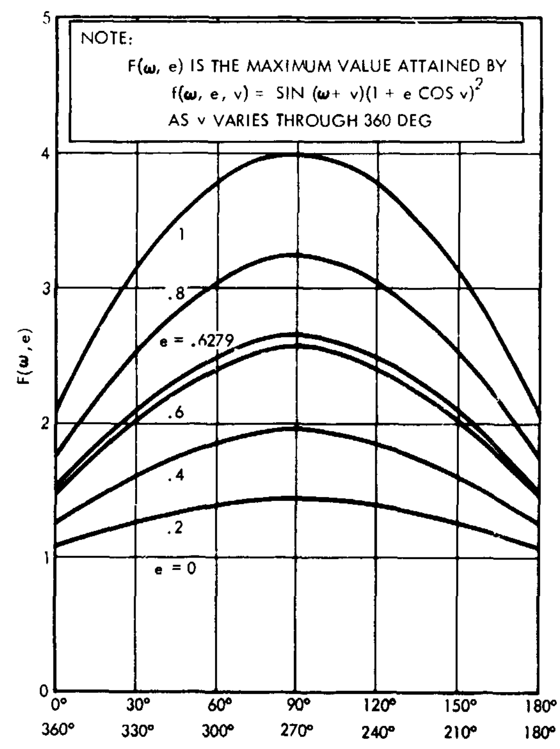


Figure E-2. Function F (ω, e)

$|\ddot{z}|_{max}$ is proportional to the maximum rate of change of doppler frequency which must be accommodated by the telecommunications link.

The functions $|\dot{z}|_{max}$, $(\dot{z}_{max} - \dot{z}_{min})$, and $|\ddot{z}|_{max}$ have been evaluated for the following conditions:

a) The spacecraft in a nominal orbit (2000 and 20,000 km minimum and maximum altitudes, inclined 45 degrees to Mars' equator) shortly after encounter, November 12, 1971. $i = 85.0$ degrees, $\omega = 28.0$ degrees.

b) The same orbit February 10, 1972 (90 days after encounter). i and ω are changed by orbit perturbation and by rotation of the earth-Mars line. $i = 71.2$ degrees, $\omega = 36.1$ degrees.

c) The same orbit May 10, 1972 (180 days after encounter). $i = 91.2$ degrees, $\omega = 101.0$ degrees.

d) The spacecraft in an orbit of the same size (2000 by 20,000 km), but with i and ω taking worst-case values.

e) The spacecraft in an extreme orbit (minimum altitude, 1600 km, maximum altitude approaching ∞), i and ω taking worst-case values.

a), b), and c) are orbits entered from an earth-Mars transit trajectory with launch date May 19, 1971, and arrival date November 12, 1971. (See Appendix D.) The results are:

	$\lvert\dot{z}\rvert_{max}$ (km/sec)	$(\dot{z}_{max} - \dot{z}_{min})$ (km/sec)	$\lvert\ddot{z}\rvert_{max}$ (m/sec^2)
Nominal orbit, encounter	3.444	4.431	1.161
Nominal orbit, encounter + 90 days	3.172	4.210	1.173
Nominal orbit, encounter + 180 days	2.491	4.444	1.497
Same size orbit, worst orientation	3.620	4.448	1.510
Extreme orbit, worst orientation	4.170	4.170*	1.763

*Whereas $\lvert\dot{z}\rvert_{max}$ and $\lvert\ddot{z}\rvert_{max}$ are greatest for the highly eccentric orbit (e), $(\dot{z}_{max} - \dot{z}_{min})$ is greatest for smaller, more circular orbits. For orbits satisfying the 50-year lifetime requirements, $(\dot{z}_{max} - \dot{z}_{min})$ has its greatest value of about 4.53 km/sec for a circular orbit of 5000 km altitude.

2. EARTH-MARS DISTANCE

Variation in the earth-Mars distance leads to large values in $|\dot{z}|_{max}$, but, as the period of (approximate) cyclical repetition is the synodic period of 25.5 months, this contribution is very predictable, and changes so slowly that $|\ddot{z}|_{max}$ is almost negligible. The values are:

	$|\dot{z}|_{max}$ (km/sec)	$|\ddot{z}|_{max}$ (m/sec^2)
Maximum value	15.02	0.0024
Time of maximum values	During the optimum period for encounter, Nov. 1971 through Feb. 1972	At opposition, Aug. 10, 1971

3. DUE TO THE GROUND STATION

To determine the worst-case contribution of the effect of the earth's diurnal rotation on the doppler data, we assume the latitude of the DSN station and the declination of Mars are each 0 degrees. Taking R = 6378 km $\dot{\theta} = \frac{2\pi}{86,400} = 0.00007277$ rad/sec, we obtain

$$|\dot{z}|_{max} = R\,\dot{\theta} = 0.464 \text{ km/sec}$$

$$(\dot{z}_{max} - \dot{z}_{min}) = 2\,R\,\dot{\theta} = 0.928 \text{ km/sec}$$

$$|\ddot{z}|_{max} = R\,\dot{\theta}^2 = 0.034 \text{ m/sec}^2$$

APPENDIX F

CELESTIAL OBJECTS COMPETING WITH CANOPUS

1. INTRODUCTION

The purpose of this appendix is to examine possible ambiguity in the use of celestial references for attitude control due to competing celestial objects. For the selected Voyager spacecraft configuration and modes of operation the sensor which is most susceptible to this sort of ambiguity is the Canopus sensor. There are two phases in the mission in which the Canopus sensor would be subject to this sort of ambiguity:

a) In the Canopus acquisition mode, when the roll axis is pointed toward the sun, and the spacecraft is in a controlled roll which is to be terminated when the Canopus sensor recognizes the star Canopus. In this mode, other celestial objects appearing in the same band of cone angles swept out by the Canopus sensor may cause a premature lock if the brightness and spectral characteristics of the object are sufficiently close to Canopus.

b) In the interplanetary cruise mode, when both the roll axis and the roll orientation have been established, a second object appearing within the same range of cone angles and clock angles covered by the field of view of the Canopus sensor may cause the roll reference to be diverted from Canopus. In addition, very bright objects, not within the field of view but close to it, may introduce enough light by scattering to cause loss of lock on Canopus.

2. CANOPUS SENSOR MECHANIZATION

The Canopus sensor and the mechanization proposed for its use on the Voyager spacecraft are both based on the Mariner C approach. (See Section IV-1, Volume 5.) The means by which the sensor discriminates against celestial objects which may compete with Canopus are the following:

a) By establishing upper and lower brightness thresholds, the sensor discriminates against objects which have a brightness 2.5 times that of Canopus or greater, and against objects which have a brightness of less than 0.4 times that of Canopus. This discrimination alone restricts the number of possible competing objects to the earth, Mars, Venus, Jupiter, Saturn, and approximately 11 stars.

b) Because of the characteristics of the spectrum of Canopus it is possible to employ discrimination against celestial objects which have substantially different spectral characteristics from Canopus.

c) The Canopus sensor has a field of view in the cone angle direction of 11 degrees. Because the cone angle of Canopus as seen from the spacecraft during the transit and orbital phases of the Voyager mission encompasses a range from 75 to 103 degrees (in 1971), the 11-degree field of view is insufficient for the whole mission, and it must be updated several times during the course of the mission to accommodate the range. (The change of cone angle during transit for several representative earth-to-Mars trajectories and during orbital operations is given in Figure F-1.

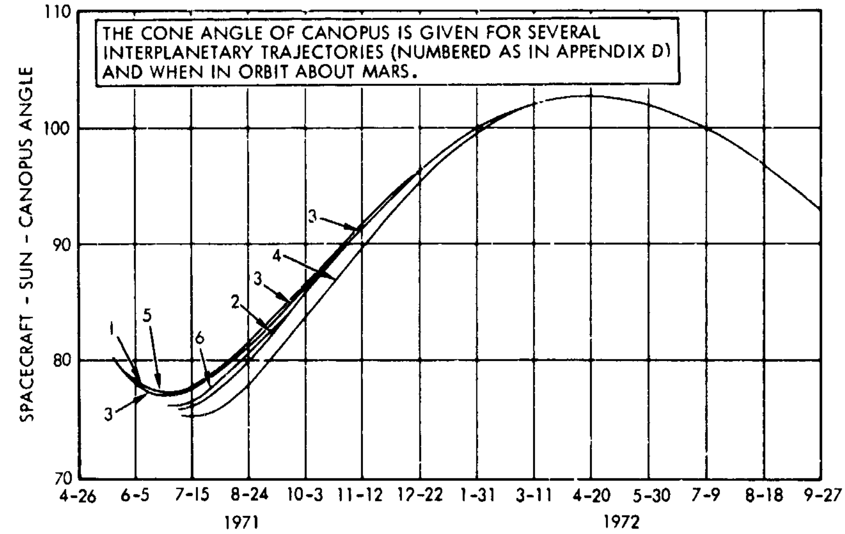


Figure F-1. Spacecraft-Sun-Canopus Angle (Cone Angle of Canopus), 1971

An updating scheme is not chosen in this appendix, because it would be a different scheme for each trajectory.) At a given time in the mission only objects whose cone angles are within the same 11-degree band currently in use by the Canopus sensor would be capable of conflicting with the lock on Canopus.

d) In the Canopus acquisition mode, the clock angles which must be traversed by the Canopus sensor depend on the relation of the initial spacecraft roll attitude to the desired attitude. This range could be anything from a small angle up to almost 360 degrees, if the programmed roll is always in the same sense. Therefore, in the Canopus acquisition mode the clock angle range does not discriminate against competing objects. However, in the cruise mode for a competing object to displace Canopus for the attention of the Canopus sensor it would have to be within the 4-degree field of view of the sensor in the clock angle direction.

3. OBJECTS COMPETING DURING CANOPUS ACQUISITION MODE

For the purpose of this analysis, objects were considered as potentially competing with Canopus if the apparent visual magnitude (as would be measured at the location of the spacecraft) is between +0.6 and -2.4. No discrimination has been introduced for different spectral characteristics.

Figure F-2 presents a map, essentially in the plane of the ecliptic, indicating a sample earth to Mars trajectory in 1971, the cone angle band which would be swept by a spacecraft in the Canopus acquisition mode at several representative locations on its trajectory, and the approximate directions to the principal competing celestial objects. The directions to stars brighter than +0.6 in magnitude are shown by arrows and an indication of the out-of-ecliptic component. (It is noted that the apparent magnitude of a star is normally listed as seen through the earth's atmosphere; outside of the atmosphere most stars are brighter by approximately 0.3 magnitude.)

IDENT	NAME OF STAR	APPARENT VISUAL MAGNITUDE
1.	SIRIUS	-1.7
2.	CANOPUS	-0.9
3.	α CENTAURI	-0.6
4.	ARCTURUS	-0.4
5.	VEGA	-0.3
6.	CAPELLA	-0.2
7.	RIGEL	-0.1
8.	PROCYON	0.1
9.	ACHERNAR	0.2
10.	AGENA	0.4
11.	BETELGEUSE	0.4
12.	ALTAIR	0.5
13.	ALDEBARAN	0.6
14.	ACRUX	0.6

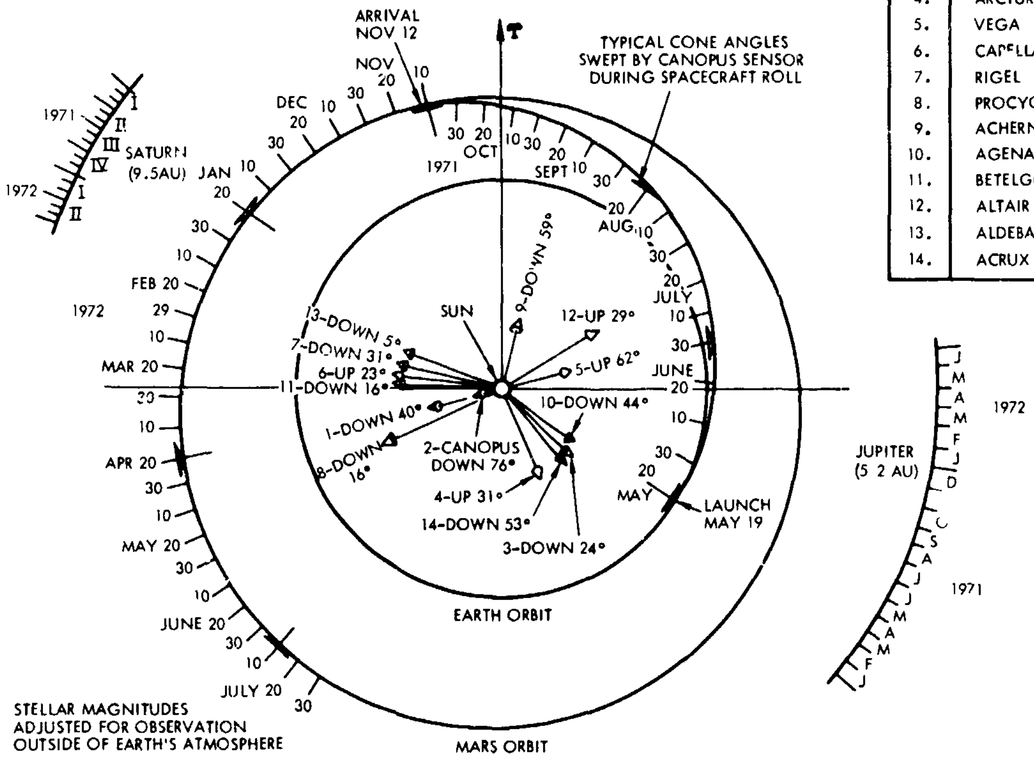
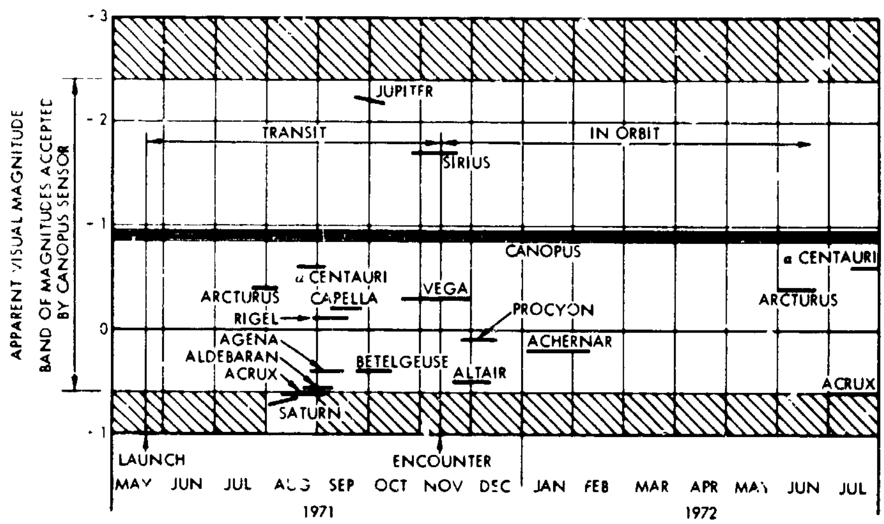


Figure F-2. Celestial Objects Competing with Canopus

Examination of Figure F-2 indicates that the sun and Venus, while substantially brighter than Canopus, will never appear in the field of view of the Canopus sensor in the Canopus acquisition mode. The earth and moon may appear in the field of view, but such appearance can only be early in the mission, and the magnitude would be considerably above the upper threshold, and therefore would not cause the Canopus sensor to lock. Similarly, the planet Mars may be in the field of view, depending on the approach geometry of the trajectory, but if so it is only when the spacecraft is close to encounter or in orbit about Mars, and again its magnitude would be considerably above the upper threshold. Jupiter and Saturn have apparent brightnesses in the range which could cause ambiguity; Jupiter is very close to the upper brightness threshold and Saturn to the lower brightness threshold. No other planets are bright enough to cause conflict.

For the stars and planets mentioned above as competitors for the attention of the Canopus sensor, Figure F-3 shows their brightness in comparison with the band of magnitudes accepted by the Canopus sensor and it also indicates the time intervals during the mission in which the objects have cone angles close enough to that of Canopus to cause possible conflict. As a specific updating schedule for the Canopus sensor was not assumed, the criterion adopted for cone angle conflict is that the object remain within ±5.5 degrees of the cone angle of that object when it equals the cone angle of Canopus. Depending on the actual updating schedule, these periods of conflict would vary somewhat. The primary value of Figure F-3 is to indicate the approximate timing and duration of possible conflicts.



OBJECTS BRIGHTER THAN + 0.6 MAGNITUDE (THE LOW THRESHOLD OF THE CANOPUS SENSOR) ARE SHOWN. WITHIN THE TIME INTERVAL INDICATED, THE CONE ANGLE OF EACH OBJECT ENCOMPASSES THE RANGE ± 5.5° ABOUT THE CONE ANGLE OF THAT OBJECT WHEN IT IS THE SAME AS THE CONE ANGLE OF CANOPUS.

Figure F-3. Celestial Objects Competing with Canopus

4. OBJECTS COMPETING DURING CRUISE MODE

When the spacecraft is in the cruise mode—that is, the roll axis points at the sun and the roll attitude is controlled by locking of the Canopus sensor on the star Canopus—loss of the roll reference can occur for one of the following reasons:

- Occultation of Canopus by an obstructing body
- Appearance of an object in the 4 by 11 degrees field of view, other than Canopus, with brightness greater than magnitude +0.6.
- The existence of a bright object nearby (not in the field of view) whose scattered radiation causes loss of lock on Canopus.

If the possible source of any of these incidents is Mars, the third cause is likely to be the one which ensues. It is not likely to be a stellar source, because there are no bright stars close to Canopus. For similar reasons planets other than Mars will not compete during the cruise mode. Another possibility is that the conflicting light sources come from dust particles near the spacecraft, in which case the second cause of loss of Canopus would ensue. Further possible sources are Phobos and Deimos, which, depending on proximity to the orbiting spacecraft, could induce either the second or third reasons for loss of lock.

As to loss of lock on Canopus caused by scattered light from Mars, this subject has been covered and summarized in Reference 1, page 22, which outlines the field of view (relative to the spacecraft) in which the appearance of Mars acts as a conflict. It is not examined further in this appendix.

5. SUN AND EARTH SENSORS

All of the spacecraft configurations which were considered by TRW for the Voyager employed two-axis sun sensors for the purpose of pointing the roll axis of the spacecraft toward the sun. (Configuration C, which is earth-oriented to achieve the communications link with greatest data capability, is sun-oriented during the earlier phases of the mission when communications requirements are not so great.) Because of the great brightness of the sun, in comparison with all other celestial objects, there is little difficulty in avoiding any ambiguity by establishing a sufficiently high threshold.

Spacecraft concepts based on Configuration C require a means to maintain the roll axis in an attitude pointing towards the earth. Although this means may be open loop, based on the use of other references, or may be by a loop closed through an RF tracking antenna, the optical earth sensor has been proposed as the means of achieving this attitude control. This appendix does not present an analysis of conflicting objects for the earth sensor in the detail given above to the Canopus sensor. However, it is worth pointing out that the earth sensor would operate under several handicaps, in comparison with the Canopus sensor:

a) The apparent magnitude of the earth as seen from the spacecraft covers a wide range, from very bright to as low as approximately magnitude +0.5. Therefore discrimination against competing objects by brightness selection must either be less effective than that for a Canopus sensor or complicated by some updating scheme. (Delaying the shift from sun-orientation to earth-orientation will alleviate the wide brightness range which must be accommodated, however.)

b) In the course of the trajectory between the earth and Mars, the earth would pass quite close to the spacecraft-sun line. The minimum value of the earth-spacecraft-sun angle can be anywhere from 0 to 14 degrees for reasonable trajectories. It is obvious that when the direction to the earth is very close to the direction to the sun it would be difficult for an optical tracker to maintain lock on the earth. However, it is noted for Configuration C that the spacecraft would not make use of the earth orientation until this point in the transit trajectory had been passed.

c) Eight to 10 months after arrival in orbit the direction to the earth again approaches the direction to the sun, as seen from the spacecraft, and the effectiveness of the earth sensor is once again doubtful.

REFERENCES

1. JPL Project Document No. 46, V-MA-004-002-14-03, "Voyager Mission Guidelines," May 1, 1965.

APPENDIX G

APPROACH GUIDANCE SENSOR

During Phase 1A, consideration has been given to the use of an approach guidance sensor to provide greater accuracy in the capsule descent trajectory than is possible using the DSIF along. This instrument would be used prior to capsule separation. The approach is to detect the position of Mars relative to its star background by use of a television camera and to transmit the composite pictures to earth for use in the orbit determination programs. Such an instrument is not required for the 1971 mission and hence is not included in the selected design. It might be required for later missions however, which is the justification of this appendix.

1. SENSOR CONFIGURATION AND PERFORMANCE REQUIREMENTS

The terminal guidance sensor would consist of a high sensitivity image orthicon television camera equipped with an optical system suitable to cover a 15 x 15 degree field on the tube photocathode. The raster will be digitally scanned and the planet outlines and star positions designated in the output by position only with amplitude unpreserved. To insure the desired accuracy, a reference reticle will be etched or projected upon the photocathode. The black-white, start-point, stop-point only requirements of the transmitted video insure that data compression may be utilized if necessary.

Analysis of the task to be performed by the terminal guidance sensor indicates that the most stringent performance requirements lie in accuracy and target brightness accommodation. Other less acute requirements related to sensitivity and environmental resistance. Accuracy in sensing the line-of-sight orientation is determined by the resolution in the planet-center, star separation angle measurement. For useful terminal guidance it is required that the line of sight orientation be sensed with an accuracy of at least 0.75 milliradian (3σ).

One of the well known limitations of the image orthicon is brightness accommodation. The usual image orthicon, when set up for optimum performance, operates over a brightness range of approximately 50:1 on the photocathode without loss of grey scale or blooming. It is shown below that the brightness ratio between the Mars image and the star image on the photocathode is:

2790:1 for +4.0 star magnitude

17,450:1 for +6.0 star magnitude

A solution to this problem utilizes a camera with point-to-point-beam current control allowing a range of 10,000:1. Prototype devices have been made. A simpler solution, however, has been found which depends instead upon a compromise between accommodation and signal linearity.[1] If the camera voltages are not set to maintain signal linearity at all illuminance levels present it was found that the simulated planet and stars dimmer than magnitude +4.0 could be resolved well. Since the terminal guidance sensor does not depend upon measurement of target intensity, this solution is acceptable. A second problem associated with the simultaneous detection of targets of grossly different total power is light-scattering in the optical system. The analysis of section 2 of this appendix shows that this effect is tolerable at the range of 500,000 km, becoming a possible limiting sensitivity factor at this point for stars of magnitude +6.0. In order to utilize the terminal guidance sensor at significantly closer ranges, separate optical paths for the star and planet images is required.

Examination of the star field in the area of Mars as seen from the approaching spacecraft reveals that a sensor detection threshold of from +4.0 to +6.0 visual magnitude will be required to provide a sufficient number of stars for data averaging. For instance, with a January 1969

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[1] Woestemeyer, F. B., "Approach Phase Guidance for Interplanetary Missions", AIAA Paper No. 64-655; AIAA/ION Astrodynamics Guidance and Control Conference; August, 1964.

launch date, assuming a 15 x 15 degree sensor field of view, 15 stars brighter than 5th magnitude are visible. Although for this case a sensitivity of +5.0 magnitude or even +4.0 would be sufficient, a sensitivity goal of +6.0 magnitude for all missions is desirable. The illuminance levels at the optical entrance aperture corresponding to these two latter star visual magnitudes are:[2]

$$E = 6.27 \times 10^{-9} \text{ lumens/ft}^2, \quad m = +4.0$$

$$E = 9.86 \times 10^{-10} \text{ lumens/ft}^2, \quad m = +6.0$$

A brief decription of the recommended instrument is provided in Table G-1.

2. APPROACH GUIDANCE SENSOR DESIGN CONSIDERATIONS

2.1 Sensitivity

The over-all accuracy obtained in terminal guidance sensing is in part dependent upon the number of stars used to identify the field. To assure that a sufficient number (~10) of stars are detected, a sensitivity goal of +6.0 magnitude has been established. In the example presented (January 1969 launch) a total of 15 stars of magnitude +5.0 or brighter are available. It is possible, however, that different launch dates would require detection of stars in the increment +5.0 to +6.0 for adequate coverage.

The illumination on an orthicon resolution element may be calculated by assuming that all of the energy focused in the central disc of the star image falls upon one element. Then, assuming that 50 per cent of the collected energy is focused in the central disc, and that resolution in the horizontal direction is equal to the vertical scan line number:

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[2] Allen, C. W., Astrophysical Quantities, University of London, the Athlone Press; 1963.

Table G-1. Instrument Description-Terminal Guidance Sensor

General (Approximate Physical Dimensions)	
Volume	1 ft^3
Weight	15 lb
Power	25 watts (operating)
Optical System	
Lens	114 mm, f/2.0 (refractive)
Field of view	15 x 15 deg
Reticle grid	Etched or projected, 8 x 8
Pickup Tube	
Tube type	image orthicon (3 in. diameter)
Exposure time	30 msec
Frame period	5 min (data transmission limited)
Scanning lines	1000
Raster dimensions	1 x 1 in.
Performance	
Minimum detectable star	+6.0 magnitude
Accuracy	0.50 mr (3σ)
Nominal operating range	500,000 km

$$E_{re} = \frac{(9.80 \times 10^{-10}) \left(\frac{\pi d^2}{4}\right) (0.50)}{(1/12)^2 (1000)^{-2}}$$

where d = entrance aperture diameter and $\frac{fl}{f/no}$ = 57 mm = 0.187 ft so

$$E_{re} = 1.95 \times 10^{-3} \text{ foot-candles}$$

Illumination at this level in the stellar image is readily detectable by the orthicon camera. Figure G-1 shows orthicon signal output current as a function of photocathode highlight illumination for two image orthicons. Note that the illumination level computed above falls on or above the high end of both curves shown. A detailed computation of the signal

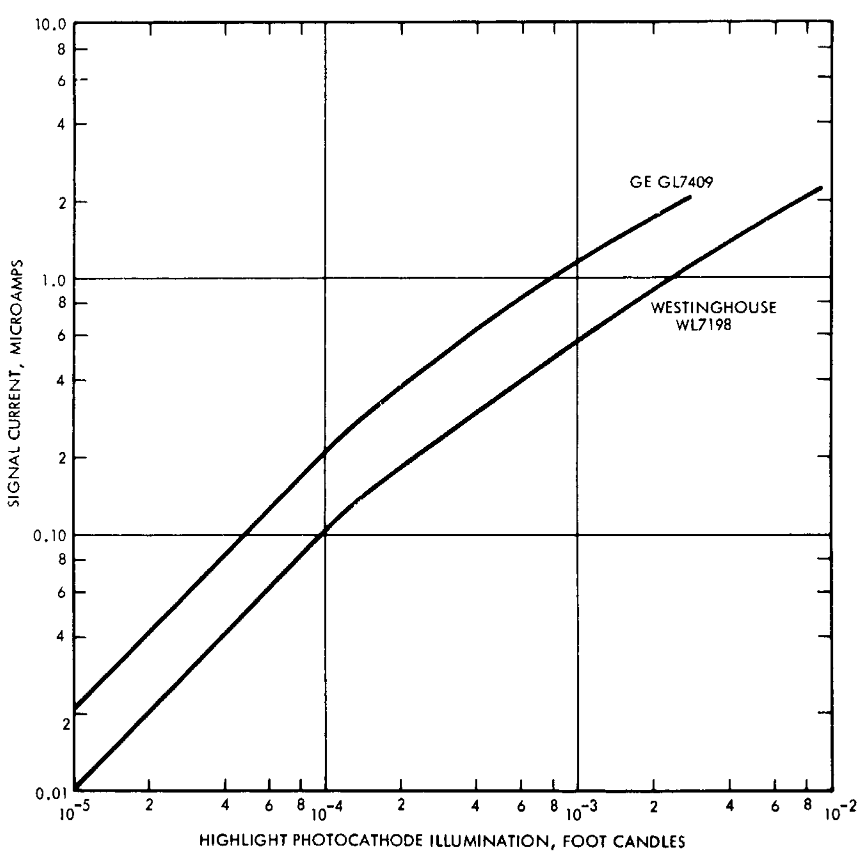

Figure G-1. Typical Image Orthicon Transfer Curves to noise ratio expected in the signal output can be made[3] but a brief consideration of Figure G-1 indicates that accuracy and illuminance range accommodations are greater problems than sensitivity.

It will be shown below that the photocathode illumination due to the Martian planet is many times that of the star image, so sensitivity is not a problem in detection of Mars.

2.2 Accuracy

Two sources of uncertainty will be present in extracting the terminal guidance information from the transmitted picture:

- Uncertainties in the exact location of the planet and star centers
- Uncertainties in the distance between the two centers

Other usual sources of error such as mechanical alignment and processor noise are circumvented by this simplified sensing mode.

The resolution and image spreading ("blooming") encountered in scientific applications of image orthicons have been the subject of considerable study.[4] For the purpose of this analysis a horizontal resoluation of 1000 elements will be assumed. Due to pulling of the beam toward the charged element the electronic start image will cover several elements, even though it is optically confined to one. If the image center is taken to be the geometric center of the apparent star image an ambiguity of ±0.5 resolution element occurs. This is equivalent to ±0.0075 degree in the field of view and is a worst case error for star position determination in that the ambiguity may be partially resolved as the star field becomes recognized since the proper separation angles between stars are very accurately known.

(3) "The Comparative Performance of Electron Tube Photodectors in Terrestial and Space Navigation Systems"; N. P. Faverty, IEEE Trans, in Aerospace Navigational Electronics, ANE-10, 3, 9/63.

(4) "Resolution Capability of the Image-Orthicon Camera Tube Under Non-standard Scan Conditions", W. C. Livingston, J. SMPTE, 72, 10, 1. 771, 10/63.

If a continuous semicircular planet crescent were provided in the image orthicon picture the center of the planet could be exactly located. Since the crescent is instead represented by a semicircular array of points, each with a ± 0.0075 degree uncertainty, the precision in center determination is instead ± 0.0075 deg/$\sqrt{N}$ where N is the number of edge picture elements, 25 being the minimum number at 500,000 km. The uncertainty in planet center is then ± 0.0015 degree.

The second source of uncertainty in measurement of the separation between star and planet may be reduced by projection (or scribing) of a grid reticle on the orthicon faceplate (see Figure G-2). An eight by eight grid, for instance, will insert a reference signal every 125 elements in the horizontal direction and one per 125 vertical lines. A good deflection system will be able to control the beam position during

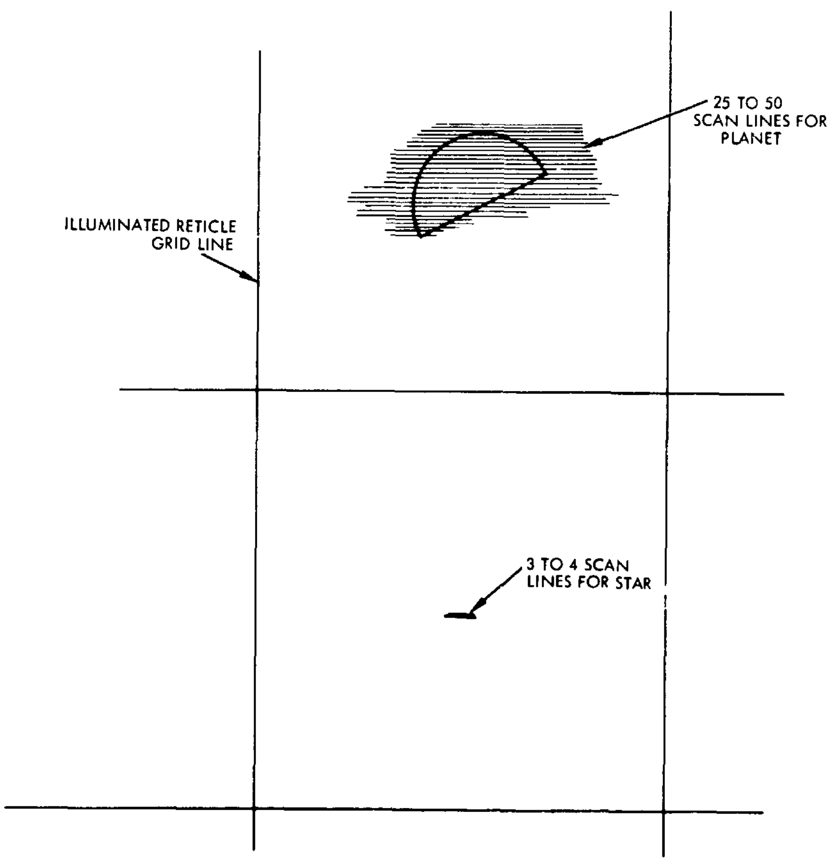

Figure G-2. Expanded Scale Drawing of Roster Area Containing Planet and Star Image readout to 1 per cent across the tube face; therefore, for the worst case (star or planet in the center of a grid element) an ambiguity of 0.63 resolution elements or ±0.0094 degree will occur in the position of either star or planet point relative to the grid. The net center-to-center distance uncertainty for a single star-planet measurement is then ±0.019 degree.

The maximum uncertainties specified above may be designated as the 3σ value and summed by the root squares method giving a single axis error of 0.0204 degree. The net error resulting from the two axes of uncertainty may be described by a circular error of radius 0.029 degree or 0.51 milliradian. This error is less than the required value of 0.75 milliradian.

A detailed error analysis has been made for the case where no reference grid is introduced. Acceptable results were obtained, assuming multiple star detections and a random star distribution in the field. The results of this analysis, shown in Figure G-3, indicate that consideration might be given to camera operation without a reticle.[5]

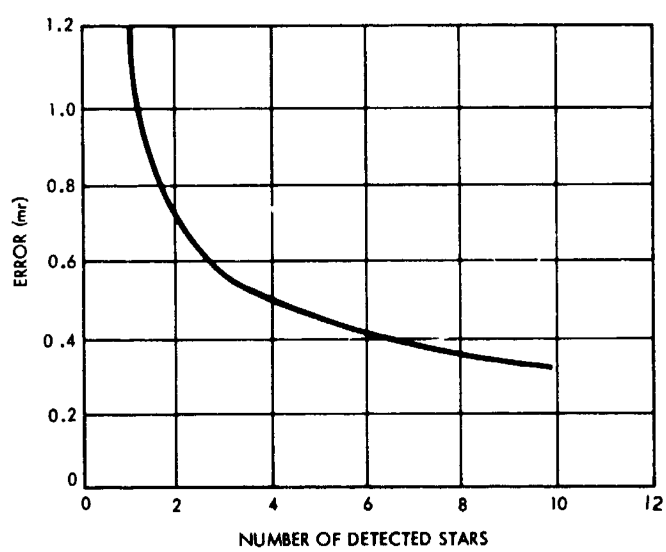

Figure G-3. Probable Error Versus Number of Detected Stars

[5] Woestemeyer, op. cit.

2.3 Accommodation and Scattering

Two major problems in the design of an imaging electro-optical device relate to the presence of two targets of widely different power content in the same frame. The first, scattering, is an increase of the entire frame illuminance due to stray light from the bright target. The second, accommodation, results from the necessity for the pickup tube to satisfactorily reproduce highlight intensity differences present in the frame.

In order to assess the effect of either of these problems on system performance it is necessary to calculate the relative brightness of Mars and the stellar target. The average illuminance of the Martian image may be estimated using the visual magnitude of the planet and its image area. For simplicity the calculation is done as though the phase angle is zero since image illuminance is essentially independent of phase angle (for $\alpha < 90$ deg). The visual magnitude of Mars at the nominal terminal guidance range (500, 000 km) is given by:[6]

$$m_v = -1.45 + 5 \log r\Delta$$

where

r = sun-planet distance, 1.5237 AU (mean opposition)

Δ = spacecraft-planet distance, 3.3445×10^{-3} AU

so

$$m_v = 1.45 - (2.29)(5)$$

$$m_v = -12.90$$

---

(6) Allen, op. cit.

In order to compute the image illuminance of Mars it is necessary to determine the area of the image. The half-angle subtended by the planet at 500,000 km is:

$$\phi = \frac{3,392}{500,000}$$

$$\phi = 6.784 \text{ mrad}$$

For the proposed focal length of 114 mm or 0.376 feet then, the image area is:

$$A_m = \pi \left[(6.784 \times 10^{-3})\ (0.376)\right]^2$$

$$A_m = 2.04 \times 10^{-5} ft^2$$

The star image may be assumed to fill one resolution element. therefore,

$$A_s = \frac{\left[2\ (0.376) \tan 7.5^\circ\right]^2}{(1000)^2}$$

$$A_s = 9.89 \times 10^{-9} ft^2$$

The illuminance ratio R between the two images may now be calculated from magnitude and image area:

$$R = \frac{\dfrac{(2.51)^{12.90}}{2.04 \times 10^{-5}}}{\dfrac{(2.51)^{-6.0}}{9.89 \times 10^{-9}}} = \frac{(2.51)^{18.90}}{2.07 \times 10^{3}} = \frac{3.62 \times 10^{7}}{2.07 \times 10^{3}}$$

$$R = 17,450, \quad m = +6.0$$

or

$$R = 2790, \quad m = +4.0$$

This ratio may vary somewhat for different martian seasons, depending upon the planet-sun distance. It was stated previously that a normal image orthicon camera would be unable to reproduce a scene containing a brightness range of this magnitude. Evidence reported,[7] however, indicates that a camera can function with a scene of this type if the planet image is grossly overexposed. This mode of operation permits accurate readout of the planet edge location but provides no measure of intensity. This solution is quite compatible with the terminal guidance sensor mode of operation. A less desirable solution would be a servo loop controlling the camera beam current according to the signal intensity. The frequency response of such a system would have to be sufficiently rapid to permit accommodation from element to element.

The presence of the bright planet in the field of view will raise the average illumination level over the entire photocathode due to scatter light in the optical system. It is known from extensive experience that a well designed optical system will scatter in the order of 1 per cent of the "in field" light. The ratio of the average scatter illumination to star illumination is then readily calculated from the brightness ratio derived above, thus

$$E_m = 17450\ E_s$$

$$F_m = 17450\ E_s A_m$$

$$F_{scat} = (0.01)\ F_m$$

and

$$E_{scat} = \frac{F_{scat}}{A_p}$$

$$E_{scat} = \frac{174.5\ E_s A_m}{A_p}$$

---

[7] Woestemeyer, op. cit.

where

$$A_p = \text{photocathode area} = 9.89 \times 10^{-3} ft^2$$

$$A_m = \text{Mars image area} = 1.02 \times 10^{-5} ft^2$$
(only one-half of planet illuminated)

so

$$E_{scat} \cong 0.175Es$$

Therefore the average scatter illumination in the field of view is equal to 17 per cent of the +6.0 magnitude star image illumination. This ratio assures that the scatter light will not prevent the detection of a +6.0 magnitude star, even when some latitude is permitted for bright spots.

2.4 Alternatives

The preceding discussion has been concerned with outlining a recommended sensor configuration. Several alternatives are discussed in this section.

2.4.1 Detector

The decision to utilize an image orthicon pickup tube was predicated upon the need for high sensitivity and a relatively large photocathode area. Other detectors offer tradeoff possibilities which are summarized in Table G-2.

The image intensifier orthicon was rejected because it offers unnecessary sensitivity at the expense of resolution and size. With this device an extremely large number of stars would be detectable, adding little to the terminal guidance accuracy at a large cost in equipment weight.

Table G-2. Comparison of Image Tube Characteristics

Detector	Type and Minimum Operating Light Level (foot-candles)	Limiting Resolution TV Lines	Maximum Cathode Diameter (in.)	Dimensions Length x Diameter (in.)	Spectral Response	Model and Manufacturer
Image orthicon	$10^{-5}/10^{-6}$	650/350	1.8	15.45 x 3.06	S-20	7967 RCA
Image intensifier orthicon	$\frac{6 \times 10^{-6}}{8 \times 10^{-8}}$	500/380	2.0	22.4 x 4.150	S-20	C74093A RCA
Vidicon	$10^{-1}/10^{-2}$	750	0.625	6.5 x 1.135		7735A RCA
Intensifier vidicon(1)	$\frac{5 \times 10^{-3}}{5 \times 10^{-4}}$	600	0.625	9.5 x 1.75		RCA development
SEC vidicon(1)	$\frac{5 \times 10^{-3}}{5 \times 10^{-4}}$	650	0.625	11.25 x 2.75	S-20	Westinghouse
Image dissector	not available	1500	1.1	8.2 x 1.5	S-20	F4011, ITT

1. Intensifier and SEC vidicon performance is calculated
2. All illumination data referred to 30 frame/sec operation

The conventional vidicon has insufficient sensitivity and sensitive area dimensions to perform the required task without penalties in at least two areas. In order to extend the detection capability of the vidicon the exposure time would have to be increased, at least by two orders of magnitude or to three seconds. This increase in exposure time in turn increases the threat of accuracy loss due to vehicle motion during exposure. To cover the same angular field the vidicon camera would have to be equipped with a 1.6 in. focal length optical system requiring a drastic decrease in f number and entrance aperture. The small size of the vidicon has made it the choice for use with the attitude verifying television camera aboard OAO. This device has poorer accuracy and requires longer exposure than the subject instrument.

Some of the disadvantages of the conventional vidicon are overcome in the image intensifier or secondary emission conduction (SEC) vidicon. These devices are theoretically capable of good resolution at the required illumination level. The second objection to the conventional vidicon applies here also, namely a relatively small photocathode. It is possible, however, that a reduced entrance aperture could be compensated for by a relatively slight exposure time increase. Although not the preferred choice at this time the intensifier SEC vidicons are a promising future consideration.

The image dissector, the only nonstorage device listed in Table G-1, suffers a fundamental disadvantage in a wide field detection task requiring short exposure. The detection dwell time for the image dissector is only the scanning period for a single picture element in contrast to the storage case where the dwell time occupies an entire frame. Assuming that the limiting noise source is cathode emission shot noise, the orthicon and vidicon possess a $\sqrt{N}$ sensitivity advantage over the image dissector where N is the number of picture elements in the frame. It is unlikely that the image dissector will be given further consideration for this task.

The image orthicon is recommended for the terminal guidance sensor in preference to the detectors described above. In summary the following characteristics are pertinent:

Advantages

- High sensitivity and resolution
- Large field of view coverage
- Proven environmental resistance
- Favorable spectral response

Disadvantages

- Over-all dimensions
- Numerous regulated voltages and adjustments

2.4.2 Optical System

A single field of view refractive optical system is chosen at this time on the basis that it would provide the necessary field coverage and light collecting power without excessive scattering. Image quality and resolution are not of great concern in selection of the optics, since the image orthicon will limit system image quality. It is important, however, that aberrations and distortions not be excessive since they cannot be eliminated by reference to the reticle grid.

The major consideration in discussion of an alternative optical configuration is the possible need to obtain terminal guidance data significantly closer than 500,000 km. For instance, if a range of 125,000 km were desirable, a 1 per cent scatter of planet flux by the optical system would illuminate the cathode 2.5 times the level of a +6.0 magnitude star. This problem might be overcome in two different ways, design refinement to eliminate excessive scatter or utilization of a dual field of view device. The former approach could lead to a reduction of scatter light of less than an order of magnitude. The operating range might be reduced to 125,000 km in this way at some additional expense in optical design.

The second solution would call for the design of a device having two fields of view imaged either simultaneously or sequentially on the camera face. A sequential device, either turreted or switched by means of a mirror, requires a moving part and an additional source of alignment uncertainty. The two fields might be imaged simultaneously with a neutral density filter reducing the intensity of the planet. One suggestion is to replace the entrance apertures of a Thompson-Starling photometer head with objectives of the proper focal length.

AUG 12 1965

SIGNIFICANT ERRATA. TRW Systems, Phase 1A
Study Report, Voyager Spacecraft
August 11, 1965

Volume 1. Summary

Substitute new p. 79 attached.

Volume 2. 1971 Voyager Spacecraft

p. 18. Item h) "necessary landed operations" should read "necessary lander operations."

p. 143. Section 3.4.1.a. second line should read "threshold of 0.25 gamma"

p. 282. Lines 3 and 4. Delete "or incorrect spacecraft address"

p. 284. Figure 5. Change "128 Word DRO Core Memory" to "256 Word DRO Core Memory"

p. 327. Denominator of second term on right hand side of equation should read

$$\left(\frac{1}{\epsilon_1} + \frac{1}{\epsilon_2} - 1\right)\left(N - 1\right)$$

p. 351. Figure 1, Section F-F. "separation nut" should read "bolt catcher"

Volume 3. Voyager Program Plan

Substitute new p. 12 attached.

p. 13. Figure 2-3. PTM Assemblies in item 7 move 1.5 months to right

p. 16. Figure 2-6. First milestone date should be September 1, 1969, instead of mid-January 1970, and all subsequent dates should be correspondingly adjusted 4.5 months earlier.

p. 20. Table 2-2. Third item in 1969 column should read "coincident with completion of proof test model assemblies. Fifth item in this column change "2 weeks" to "3.5 months." Fourth item in 1971 column, change "4 months" to "5 months."

p. 67. Figure 5-2. Under Intersystem Interface Specification add a block entitled "Spacecraft to OSE Interface Specification"

p. 120. Last line of paragraph c should read "shown in Table 5-2."

p. 126. Figure 5-13. Year should be 1966 instead of 1965.

p. 153. Figure 5-18. Ignore all numbers associated with lines in figure.

p. 167. Figure 5-21. In line 20 change "design revisions" to "design reviews"

p. 254. Second paragraph, third line, "The capability of the transmitter to select" should read "The capability of the transmitter selector to select."

p. 258. Section heading n should read Experiment Data Handling

p. 604. Section 3.2.1 beginning of second paragraph should read "The hydrazine fuel . . . "

Volume 4. Alternate Designs: Systems Considerations

p. 103. Figure 3-19. Caption should read "Radial Center of Mass. . . "

p. 151. Last paragraph, second line, "For the baseline, the reliability. . ." should read "The reliability . . . "

p. 158. 8th line, replace "0.06 pound/watt" by "0.6 pound/watt"

p. 215. Figure 3-50. Dot in ellipse at right should be 0.

p. 230. Section 5.3.2, second paragraph, 7th line, should read "Figure 3-52.

p. 239. Second line, "with a variable V" should read "with a variable ΔV"

p. 247. First line, "3250 km/sec" should read "3.250 km/sec"

p. 261. Figure 3-64. Interchange coordinates, clock angle and cone angle

p. 293. Figure 3-81. An arrow should connect "Low-gain spacecraft antenna" and the dashed line at 73 × 10^6 km

Volume 4. Alternate Designs: Systems Considerations Appendix

p. 6. Figure A-2. The shaded portion under the lower curve should extend to the right only as far as 325 lb.

p. 9. Tabl(A-1, part (1). In last column heading change "W_3" to "W_1". In part (4) last column heading change "W_3" to "W_4"

p. 22. Second line below tabulation, replace "575 × 35" by "570 × 35"

p. 29. Tabulation at bottom of page, change "18" to "30" and "400" to "240"

p. 207. Numerator of equation for λ best at bottom of page should read "0.0201," and numerator of equation for λ worst should read "9.21"

p. 209. Table 5B, fifth line. Delete " × 10¯." Also p. 213, Table 7A, seventh line, and p. 232, Table 3B, fifth line.

p. 217. Top portion of Table 9B should be labeled "primary mode" instead of "other modes"

p. 326. In equations following words "clearly" and "thus" insert " $>$" before second summation.

Volume 5. Alternate Designs: Subsystem Considerations

p. 3-15 Fifth line, "... is extended, spacecraft" should read "... is extended, two spacecraft"

p. 3-38 Last line, change " $= \frac{32}{4500} = M$" to " $\left(\frac{32}{4500}\right)\left(M\right)$"

p. 3-51 Two equations at bottom of page should read

$$D = 4\pi A/\lambda^2$$

$$A = \frac{D\lambda^2}{4\pi} = \frac{1000\,\lambda^2}{4\pi}$$

p. 3-67 Third line, last parenthesis " $\left(\frac{\pi}{2} + \phi\right) -$ "

p. 3-82 6th line should read "50 degrees" instead of "50-140 degrees," and seventh line should read "140 degrees" instead of "50-140 degrees"

p. 3-111 Last line, change "50 Mc" to "1 Mc"

p. 3-137 Item g) for "... followed by 5 frames of real time" substitute "... followed by 11 frames of low rate science data and 5 frames of real time" pp. 3-150 and 3-151 are interchanged.

p. 3-156 Last line, should read "gates, a 7 bit"

p. 5-21 Second paragraph, third line, for "others since they are" substitute "others which are"

p. 5-33 Bjork equations should identify 0.18 as an exponent, and the exponent for (ρ_p/ρ_t) in the Hermann and Jones equation should be 2/3 in both cases.

p. 5-33 Figure 5-12 should be replaced with Figure C-7 of Appendix C.

p. 5-40 Three lines above Table 5-10 substitute "permanent set" for "experiment"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix I

p. B-11 Bottom of page, for "$r^{2/3}$" substitute "$(V/C)^{2/3}$ r"

p. C-4 The title of Figure C-2 should read "Figure C-2. Meteoroid Influx Rate Circular Orbit Mars", and the title of Figure C-3 should read "Figure C-3. Meteoroid Influx Rate Cruise"

p. C-5 At bottom of page, add the following: "*Within 50,000 km of Mars"

p. C-6 Line 13 should read: "... of low density ($\rho_p < 2.4$ gm/cm^3..."

p. C-6 Figure C-4. The ordinate "2" should read "100"

pp. C-17, C-21 The figures C-6 and C-7 on pages C-17 and C-21 should be reversed.

p. C-28 The title of Figure C-8 should read "Meteoroid Shield Test Specimen"

p. C-29 The title of Figure C-9 should read "Cutaway of Meteoroid Shield Test Specimen

p. C-34 In Section 1.8 the first sentence should be replaced by the following two sentences: "Preceding sections of this appendix contain derivations of the probability of penetrations of the spacecraft outer skin by meteoroids. It is clear that to design an outer skin of sufficient thickness to reduce the probability of no penetrations to a low level, such as 0.05 to 0.01, would be prohibitive in terms of the weight required."

p. C-35 In the first equation, the expression "(t in m^2)" in two places should read "(t in cm)" and "A" in two places should read "(A in m^2)"

p. C-38 In Table C-2, all values in inches should be in centimeters. A zero should be inserted immediately following the decimal point, for example: (0.020-inch) = 0.05080, (0.020-inch) = 0.06096, (0.020-inch) = 0.04064, etc.

p. C-40 In Section 1.8.7 Computation of R_i's, the sixth line should read "... than 10^6 are neglected"

p. C-45 In listing under "Values of t Used for Extreme Environment Analysis," under Inch, the first number should read 0.020 instead of 0.202

p. C-52 In 1.10 NOMENCLATURE, "K_2" should be defined as "$K^{-2/3}(4 \pm 2)$" and "B" should be

$$\frac{1000\, \rho_t V^2}{9.806 H_t}$$

pp. C-150 and C-151 should be reversed.

p. C-208 Along the ordinate in the graph, "Stress $\times 10^{-3}$" should read "Stress $\times 10^{-2}$"

Volume 5. Alternate Designs: Subsystem Considerations. Appendix II

p. F-23 Lines 7 and 10 change all subscript τ to T

p. F-24 Line 14, change "ME_1" to "mE_1"

p. F-29 Figure F-9 title should be "Reflection Phase Angle ϕ (deg)" and Figure F-10 title should be "Reflection Magnitude R"

p. F-30 Last line, change "0.27" to "0.175"

p. F-31 Lines 14 and 15, change "14,700 ft/sec to 460 ft/sec" to 14,700 ft/sec minus 460 ft/sec" and "14,700 ft/sec to 10,000 ft/sec" to "14,700 ft/sec minus 10,000 ft/sec"

p. F-32 Last line in item 4), change "27 per cent" to "17.5 per cent"

p. F-35 Table F-4, under Assumed Parameter for item 2 insert "$\pm 2 \times 10^{-5}$", for item 3 insert "$\pm 3 \times 10^{-5}$", and for item 4 insert "$\pm 2 \times 10^{-5}$"

p. F-53 Item d. Noise Figure, change "4 db" to "3.5 db"; Gain, change "20 db" to "10 db", last line change "10 db" to "4 db"

p. F-58 Figure F-21. Change 102 kc to 112 kc.

p. F-59 Line 22, change to "M_1 = 21.5 deg or 0.375 radians (rms, peak)"

p. F-60 Line 2, change to

$$"M_2 = \sqrt{(1.1)^2 - (0.375)^2}"$$

p. F-60 Line 3, change to "M_2 = 1.03 radians (rms) or 1.46 radians (peak)"

p. G-6 Paragraph 1.4, second line, change "from $E_M = 10' E_o$ to $10^4 E_o$ . . ." to read "from $E_M = 10^{-1} E_o$ to $10^4 E_o$ . . ."

Volume 6. Operational Support Equipment

p. 25 Figure 6. Caption should be "Typical Grounding Scheme"

p. 39 Section 1.3.3, change opening of first sentence to read "Launch pad equipment consists of the ground power and RF consoles and the test flight program power and control equipment . . ."

p. G-31 Figure 1. Lines enclosing Data Format Generator should be solid.

p. G-102 Last line substitute "4500" for "45"

p. G-113 In Section 4.4.2, change "25 per cent" to "250 per cent"

p. G-184 Section 4.5, substitute "6.5 feet" for "six feet"

p. G-311 Fifth line, change "30 per cent" to "20 per cent"

p. G-398 Section 4.2 should begin with "The hoist beam is . . ."

p. G-419 Second line "4 optical alignment targets" instead of 8. Same correction top of p. G-421.

p. G-423 Section 4.9.2, substitute "20 per cent" for "50 per cent"

Volume 7. 1969 Flight Test Spacecraft and OSE

p. 90 First line should read "Launch pad equipment consists of the ground power and RF consoles and . . . "

p. 107 Last line, change Volume 5 to Volume 6.