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FINAL REPORT
BUOYANT VENUS STATION
FEASIBILITY STUDY

Volume IV - Communications and Power

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FOREWORD

This final report on the Buoyant Venus Station Feasibility Study is submitted by the Martin Marietta Corporation, Denver Division, in accordance with Contract NAS1-6607.

The report is submitted in six volumes as follows:

Volume I - Summary and Problem Identification;

Volume II - Mode Mobility Studies;

Volume III - Instrumentation Study;

Volume IV - Communication and Power;

Volume V - Technical Analysis of a 200-lb BVS;

Volume VI - Technical Analysis of a 2000- and 5000-lb BVS.

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FINAL REPORT

BUOYANT VENUS STATION FEASIBILITY STUDY

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PART I - COMMUNICATIONS

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COMMUNICATION STUDIES - TASK 4.3

The contractor shall study the requirements and limitations of station to orbiter communications. Where applicable, the communication system shall include released probe data links. The contractor shall select, subject to Government approval, a station to orbiter communications system. The contractor shall define the physical and operational characteristics and limitations. From consideration of the various mission mode trajectories and the orbiter S/C orbital characteristics, the contractor shall establish the requirements for station data storage and transmission to the orbiter S/C.

SUMMARY

The buoyant Venus station (BVS) telecommunications subsystem consists of radio and data management functions associated with commanding the station operation, telemetering data to the orbiter, measuring range from the orbiter to the station, and receiving data from radiosondes dropped from the station.

Communications over maximum ranges of 14 000 km are required for some station locations assuming the standard orbit of 1000 km periapsis and 10 000 km apoapsis.

Frequencies in the range of 200 to 400 MHz have been considered with no clear cut advantage for one end of the band over the other until the actual station size, weight allocations, and antenna type are considered.

Communications transmission periods of from 3 to 15 minutes at data rates of 30 to 10 000 bit/sec are reasonable depending on station weight class and location.

Coherent PSK/PM systems for the orbiter/station links have an advantage over noncoherent systems in both higher data rates for the power used and in accomplishing the ranging function. Noncoherent FSK links do not lend themselves to integrated ranging techniques.

For coherent systems one must resort to frequency search modes and keeping subcarriers out of the frequency search range. This results in finite acquisition times and wider bandwidths than otherwise required.

Transmission of data from the station to the orbiter is under control of the orbiter and occurs only when the orbiter initiates it. Once begun the transmitter "on" time must be limited by the station programer unless it is sooner terminated by orbiter command.

Ten- to 40-W solid-state transmitters with good efficiency (30 to 60%) are considered feasible for 1970 technology, however, design of phase-modulated transmitters in this frequency range may be required.

INTRODUCTION

This part of Volume IV treats the telecommunication portion of feasibility study for a BVS under the study ground rules defined in the following section.

Parametric bounds and variables are identified along with the results of limited trade studies and discussion of possible approaches to use in the conceptual design of the various weight class of stations described in the other volumes.

TECHNICAL GUIDELINES FROM NASA-LRC

Technical guidelines established by the NASA Statement of Work L 6801 Exhibit A have a direct bearing on this communication study:

1) An orbiter shall serve as a relay station for transmittal of data to earth, and shall be assumed to possess all required receiving, storage, and transmitting capabilities;
2) The nominal orbit shall be assumed to have a periapsis altitude of 1000 km and apoapsis altitude of 10 000 km;
3) The station to orbiter communications system shall be assumed not to require directional orientation of antennas or high-gain antennas;
4) Initial conditions at inflation of the buoyant station(s) shall be consistent with subsonic velocity above the visible cloud layer for the three Venus atmospheres.

SYMBOLS

BRF	radio frequency bandwidth
B_v	video bandwidth
D	Doppler frequency
ERP	effective radiated power
f_c	center frequency of filter
f_o	frequency of reference subcarrier
FSK	frequency shift key
G_R	gain of receiving antenna above isotropic
G_T	gain of transmitting antenna above isotropic
K	Boltzmann's constant
N_o	noise power density
P_e	bit error probability
PN	pseudonoise
PSK/PM	phase shift key/phase modulation

P_T transmitter power

SNR_i input signal-to-noise ratio

SNR_o signal-to-noise ratio out

T system noise temperature, $^{\circ}K$

T_a antenna temperature

V velocity of orbiter, m/sec

W watts

λ wavelength, m

θ true anomaly of orbit, deg

$2B_{LO}$ noise bandwidth of the phase lock loop at threshold

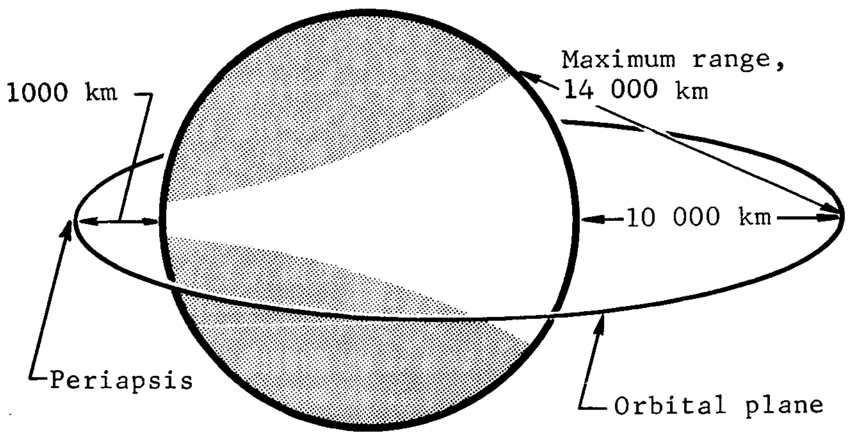
COMMUNICATIONS GEOMETRY AND VARIABLES

Four basic communications links have been identified. These are the telemetry link between the buoyant station to orbiter relay; the command link from the orbiter relay to the buoyant station; the telemetry link between the dropsondes and the buoyant station; and ranging between the orbiter and the station to determine station position locations.

There are several communications variables that can be used to determine feasibility and limitations of the links. (Other variables have been established by the orbit.) These variables are communications range and period, radio frequency band, antenna gain and minimum elevation angle, modulation technique, transmitter power, and data rates.

The Orbit

The communications geometry for the station orbiter is shown in figure 1. The orbit is the standard orbit given in the statement of work. The diagonally shaded area represents locations of the station for which no communications between the station and orbiter are possible because the station will be below the communication horizon.



Note:	1. Shaded area is below horizon.
	2. Orbital period, 228 minutes.
	3. Range variation, 1000 to 14 000 km.
	4. Communications period, 0 to 106 minutes.

Figure 1. - Communications Geometry

The shaded area is fixed in relation to the orbit and the planet rotates very slowly in relation to the fixed geometry since the rotational period of the planet on its axis is approximately 250 days. The orbiter completes one orbit in approximately 228 minutes.

Communications Range and Period

The communications range and view period for the orbiter/station links vary from 1000 to less than 14 000 km and from zero to approximately 106 minutes depending on station location. The effective view times are also a function of station antenna beam width and link margins.

Table I shows view times for various station locations in the orbital plane as a function of station antenna half power beam width. As the station drifts away from the orbital plane, the available period per orbit can reduce to zero at a rate depending on the winds, location of the orbit, and initial location of the station.

TABLE I. - AVAILABLE COMMUNICATION PERIODS, STATION TO ORBITER

True anomaly, θ, degrees from periapsis	Maximum communications periods for various station antenna half power beam widths, minutes		
	90°	100°	140°
0	3.1	3.2	5.3
45	8.0	13	20
90	22	23	40
135	29	55	84
180	69	78	106

These facts are apparent if the assumed wind patterns described in Volume III are considered and figure 1 is reexamined.

Ranges assumed for the dropsonde to station link are zero to 100 km with a maximum communications period of 1 hr.

Radio Frequency Band Selection

Many factors must be considered in selecting suitable frequencies (bands) for the various links. The major factors are listed in table II.

TABLE II. - FACTORS IN SELECTION OF FREQUENCY BAND

Critical frequency, ionoshpere	estimated to be 2 to 20 MHz
Attenuation (moderate rain)	at 3000 MHz 1×10^{-3} dB/km (positive slope with frequency)
Attenuation (sleet, snow)	less than rain
Therefore	200 MHz to 3000 MHz (good from standpoint of above)
Doppler and frequency instability plus desire for solid state	reduce above to 200 MHz to 400 MHz
Transmitter weight and efficiency	high efficienty and low weight desired

The lower frequency limit must be well above the critical frequency for the Venus atmosphere (if one exists) and must be compatible with size and weight limitations for antenna and other equipment.

In arriving at an upper frequency limit for the links, attenuation due to rain and/or ice particles, bandwidths due to frequency instability, transmitter weight and efficiency, and limitations on solid state power amplifier techniques must be considered.

Atmospheric attenuation. - It has been estimated (ref. 1) that the critical frequency on Venus varies from 2 to 20 MHz (the larger value for locations near the subsolar point). Hence a lower limit 10 times the highest critical frequency was chosen.

The upper limit has been chosen as 400 MHz, which is well below frequencies affected by rain and ice particles, and it meets the remaining criteria as well.

A further consideration is the availability of frequency allocations. Allocations in the 400 MHz range should be relatively easy to obtain, while the 215 to 260 MHz telemetry band will be closed after 1970. However, because of an approximately 6 dB penalty for going to the higher frequency, all of the preliminary link calculations have been made at 200 MHz, and for comparison purposes, a few have also been made at 400 MHz. Assuming hemispherical coverage for all antennas, a 400 MHz link is approximately 6 dB worse than the 200 MHz approach because of the added space loss. (The antenna gains are approximately the same for each since by the technical guidelines, high gain antennas are not allowed.)

Transmitter weight, volume, power, and efficiency. - A comparison of solid state transmitter weight, volume, and efficiency from various sources shows that for a given output power in the 5 to 25 W range, the weight and volume is for all practical purposes constant across the frequency range of 200 to 500 MHz, but increases with power as shown by figure 2.

Efficiency decreases with increase in frequency and the slope is greater (more negative) for the higher power transmitters as shown in figure 3.

Weight and size shown in figures 2 and 3 are considerably more conservative than the data shown in table III (ref. 2), which are for 20 W transmitters at 250 MHz and 400 MHz, respectively. In contrast, the efficiency in figure 3 is more optimistic.

TABLE III. - WEIGHT AND SIZE DATA, 20-W TRANSMITTERS

Parameter	Frequency	
	250 MHz	400 MHZ
Efficiency, %	35	16
Volume, cu in.	35	35
Weight, oz	35	40
RF power level, W	20	20

Information from additional sources (vendors) tends to support the lower weight and volumes and the higher efficiencies. Therefore these values have been used in conceptual design of the various stations reported in Volumes V and VI of this report.

Antenna weights vs frequency. - Antenna weights at these frequencies can vary widely over the band depending upon the type selected. Results of a previous study of antenna weights for various antenna types versus frequency is shown in figure 4. Although these weights are conservative, they show the trend in weight as related to frequency.

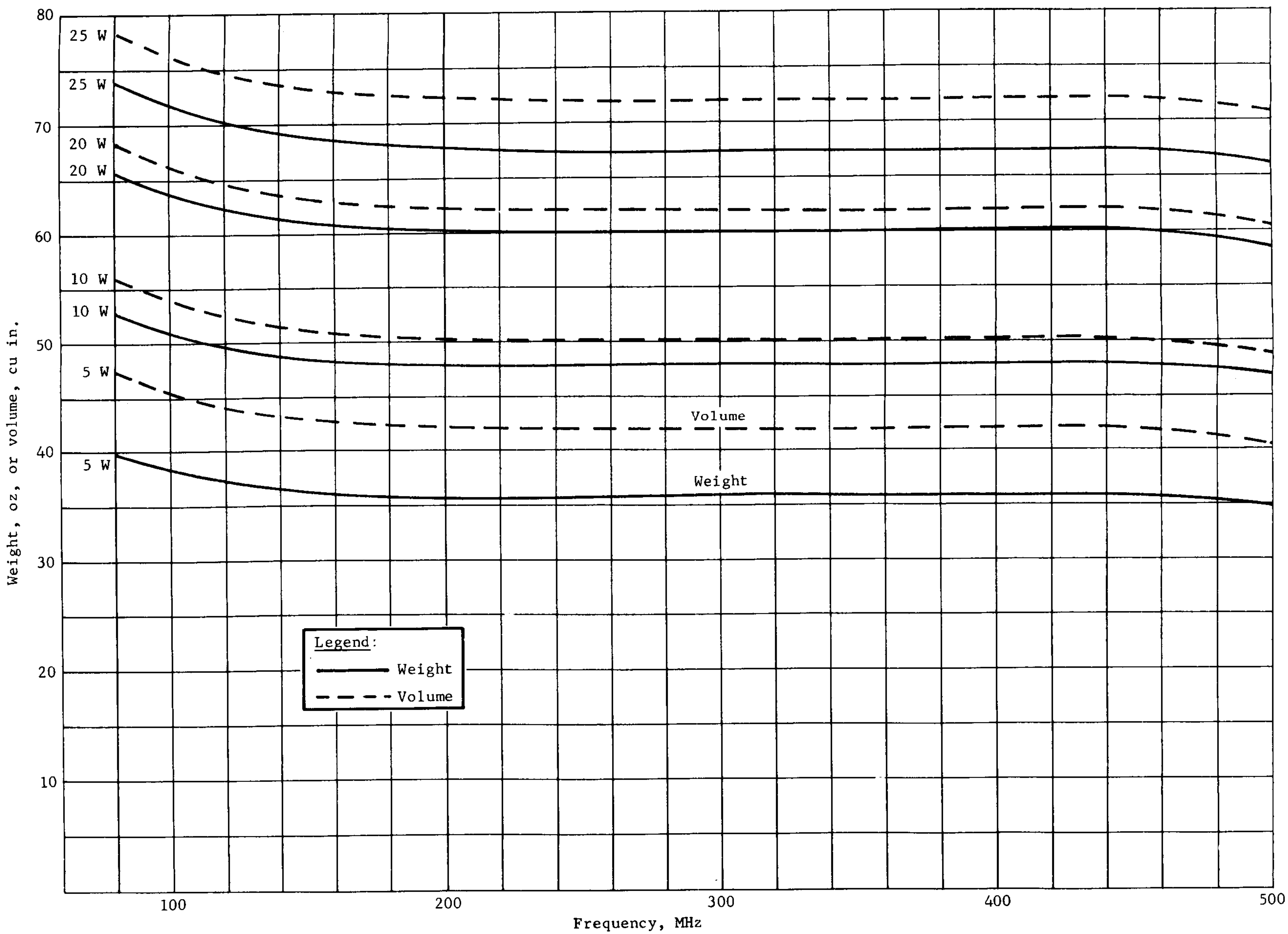


Figure 2. - Transmitter Design Weight and Volume vs Frequency, Solid-State Design

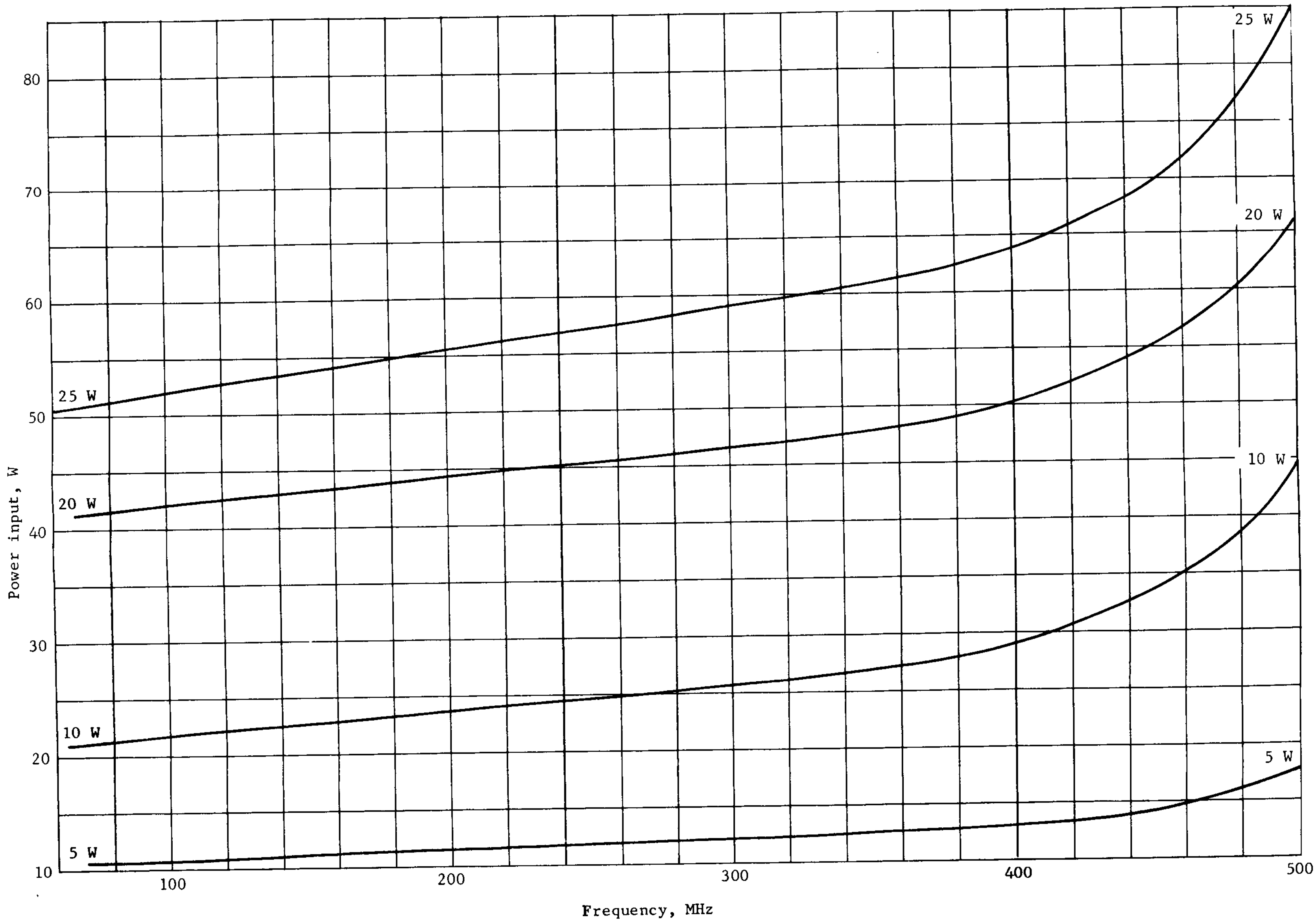


Figure 3. - Transmitter Design Power Input-Output vs Frequency, Solid-State Design

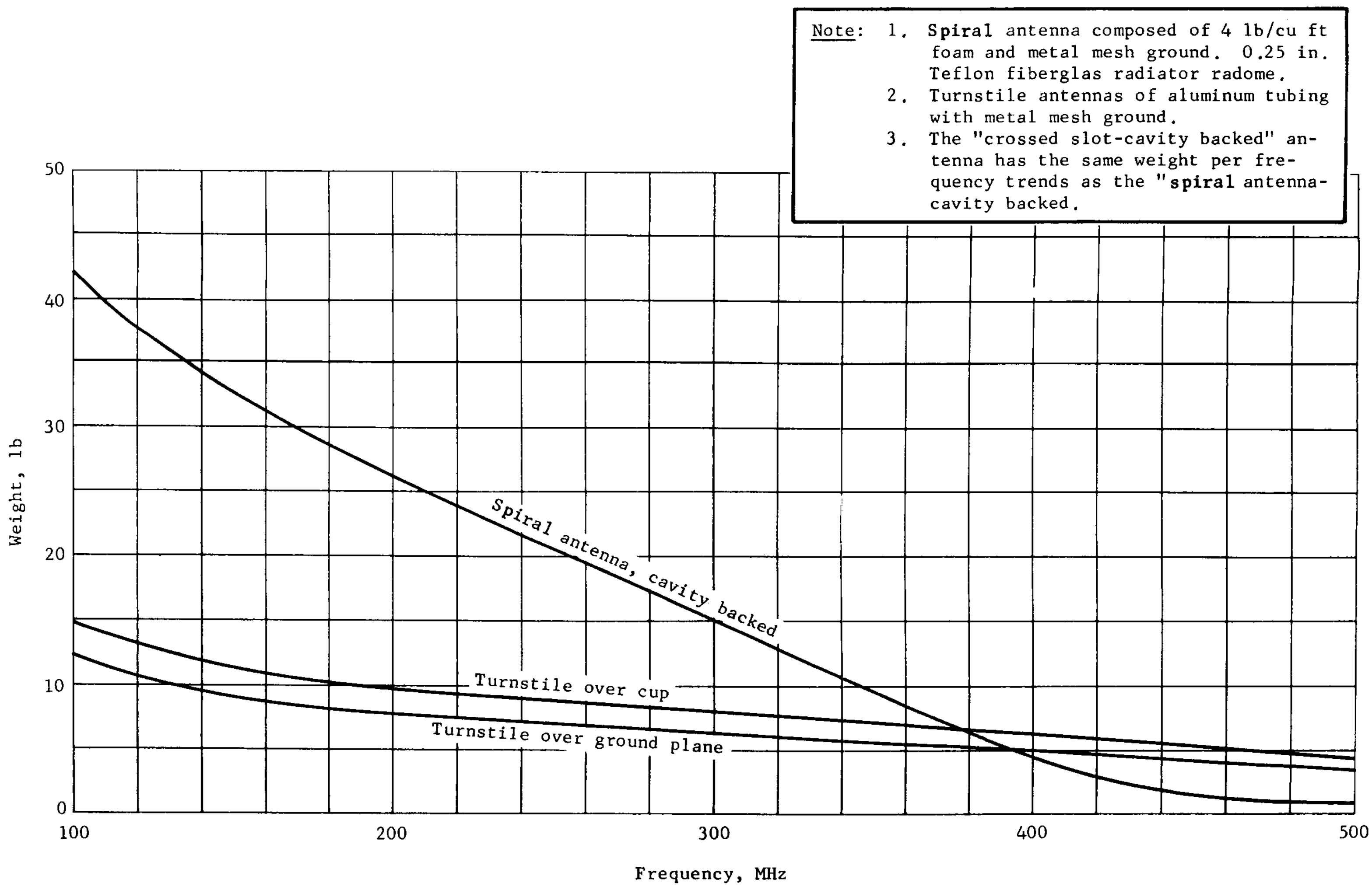


Figure 4. - Antenna Weight vs Frequency

The crossed slot-cavity backed and the spiral-cavity backed antenna weights decrease approximately linearly with frequency between 200 and 400 MHz, while the turnstile over cup and turnstile over ground plane are relatively constant over that range.

The final frequency band selections for each size class of station depend on whether the station is weight limited and on the type of antenna that can be used since the antenna can be the deciding factor where weight optimization is the governing criterion.

Modulation Techniques and Data Rates

Modulation techniques. - Frequency shift key (FSK), phase shift key PSK/PM, and FM modulation methods including PN code ranging have been considered for one or more of the various links. Comparisons are made in the chapters associated with the basic communication links.

Data rates. - Station average data acquisition rates are estimated to range from 100 bit/sec for a maximum instrument payload to a rate of ¼ bit/sec for an absolute minimum station. The station storage capacity, transmission link data rates, and programing must, of course, be compatible with the instrumentation and sampling rates. This subject is discussed in the chapters treating each of the data links.

STATION TO ORBITER DATA LINK

Parametric Bounds on Range and Data Rates

The communications range for this link varies from 1000 to approximately 14 000 km. The data rates to be considered are based on estimated station data accumulation rates of from 1 million bits on some orbits for the larger stations to 3600 bits/orbit on the smaller station. This gives a data rate range roughly 30 bits/sec to 5 or 10 kilobits/sec, depending on the time one selects as a minimum communications transmission period. Actually the higher data rates may be reduced if the concept of only partially dumping the station data store on a given orbit and completing the dump on following orbits is accepted. Scanning or picture data are subject to data compression as discussed in Volume V.

Frequency Uncertainty and Bandwidths

In determining bandwidth requirements for noncoherent systems, frequency uncertainties of the station transmitter, the orbiter receiver, and the doppler shift, as well as the modulation bandwidth must be considered.

Total frequency uncertainty for carrier frequencies of 200, 400, and 500 MHz are shown in table IV. A frequency tolerance of ±0.005% for the transmitter and 0.001% for the receiver were assumed. A straight line flight path simplification was used in calculating the maximum Doppler, $D = V/\lambda \cos \varphi$ where D = Doppler, V = velocity, λ = wavelength, and $\varphi = 10°$.

These values are considered very conservative. In the preliminary design of the various stations, transmitter stabilities of ±0.002% were assumed. The data presented in this volume are based on the ±0.005% frequency tolerance for worst-case conditions.

TABLE IV. - FREQUENCY UNCERTAINTY BANDWIDTHS

Uncertainty due to	Carrier frequency		
	200 MHz	400 MHz	500 MHz
Transmitter	±10	±20	±25
Receiver	±2	±4	±5
Doppler	±5.3	±10.6	±12
Total uncertainty bandwidth	±17.3 KHz	±34.6 KHz	±42 KHz
Note: Transmitter, ±0.005%; receiver, ±0.001%			

Modulation Techniques

Two basic modulation techniques (noncoherent FSK and coherent PSK/PM) were considered in developing transmitter power antenna gain product requirements for data rates versus communications range for this link. These data are shown in figure 5 for a noncoherent FSK link and in figure 6 for a PCM/PSK/PM link.

The PSK link has an advantage of about 10 dB, however it requires a frequency search mode for the orbiter receiver to take advantage of the narrow bandwidth. Supporting link calculations are given in the next section.

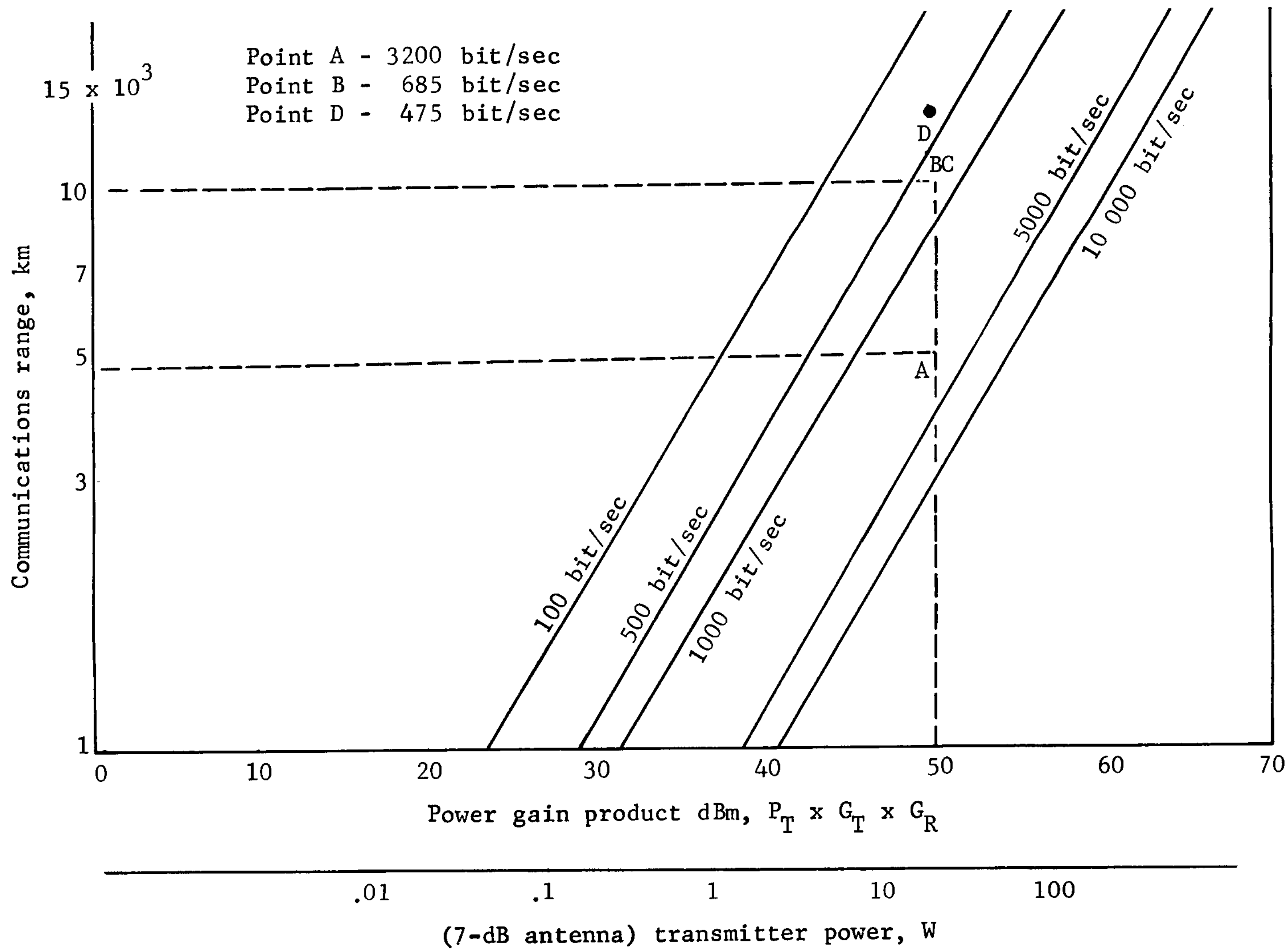


Figure 5. - 200-MHz Noncoherent FSK Modulation Technique

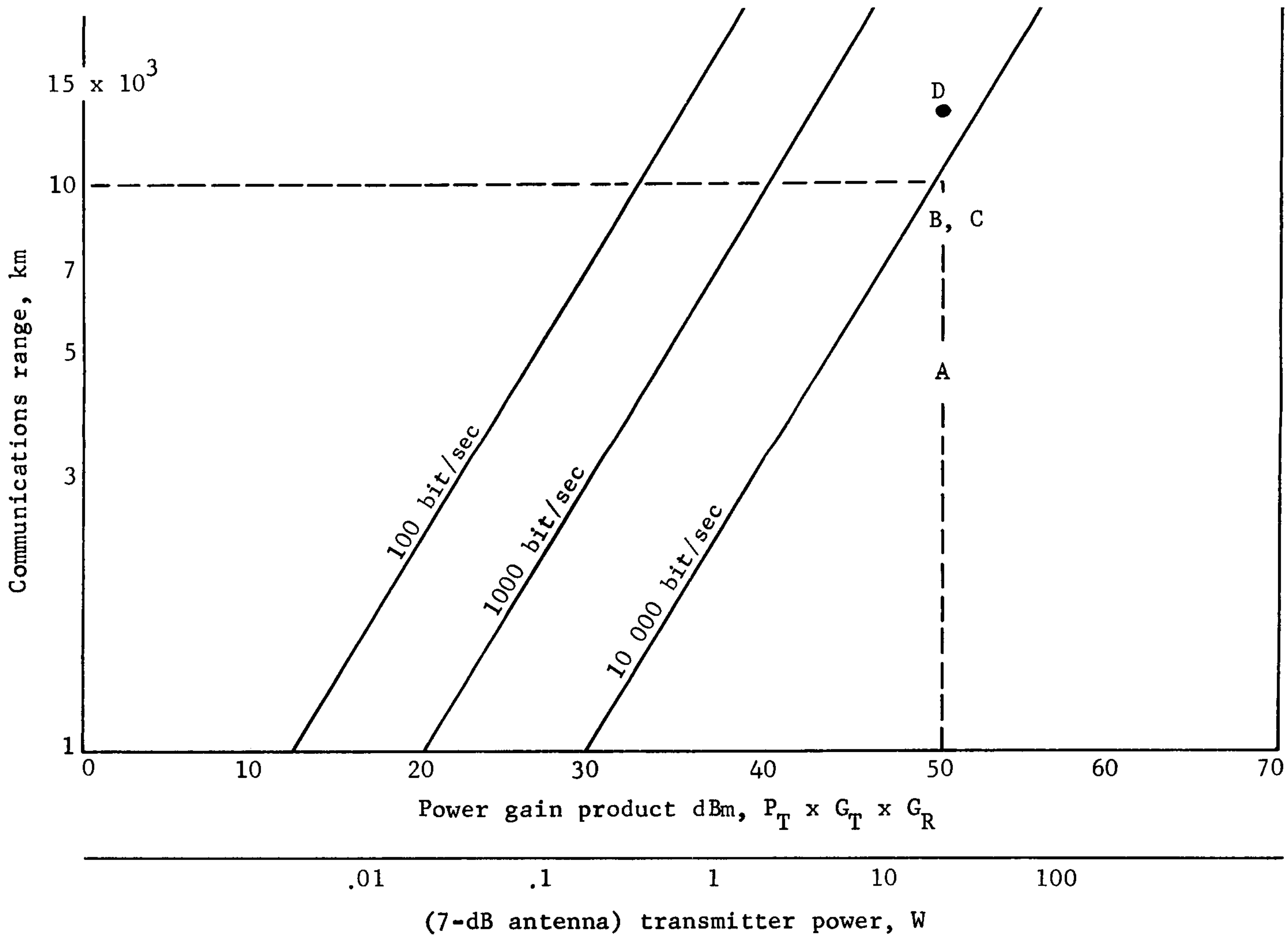


Figure 6. - 200-MHz Coherent System PSK/PM Modulation Technique

Link Calculations for a Station to Orbiter Telemetry Link

FSK link. - A series of link calculations has been made for the station-to-orbiter telemetry link for FSK modulation using split phase PCM format. Sample calculations follow for a data rate of 100 bits/sec and a range of 1000 km. Table V gives effective radiated power required for various bit rates and ranges for a 0-dB orbiter antenna. The results are plotted in figure 5.

Receiver noise power density. - Assuming a receiver noise figure of 5 dB, an antenna temperature of 700°K, and where T_a is antenna temperature, the system noise temperature, T, is:

$$T = (NF-1)\ 290 + T_a$$
$$= (3.16 - 1)\ 290 + 700$$
$$= 726$$
$$T = 1326°K$$

The noise power density number = KT where K = Boltzmann's constant:

$$\text{Number} = -198.6 \text{ dBm} + 31.3 \text{ dB}$$
$$= -167.4 \text{ dBM}$$

Receiver bandwidth. - The receiver bandwidth calculations are:

Transmitter frequency tolerance	±0.005% =	±10 000 Hz
Receiver frequency tolerance	±0.001% =	±2 000 Hz
Maximum doppler (pessimistic)	=	±5 330 Hz
		±17 330 Hz

$$D = \frac{\text{velocity}}{\lambda} \cos 10°$$

$$D = \frac{8.1 \times 10^3 \text{ m/sec}}{1.5 \text{ m}} (0.987)$$

$$D = 5330 \text{ Hz}$$

Total bandwidth (per channel) $B_{RF} = 2(17\ 330) + 2(200)$

$$B_{RF} = 35 \text{ KHz}$$

$$10 \log B_{RF} = 45.4 \text{ dB}$$

TABLE V. - TRANSMITTER ANTENNA GAIN PRODUCTS (dBm) FSK SPLIT PHASE MODULATION

Range, km	Data rate, bit/sec							
	50	100	300	500	1000	2000	4000	10 000
Effective radiated power, dBM, 200 MHz link (0 dB gain orbiter antenna)								
1 000	21.7	23.5	27.2	28.8	31.6	34.2	37.0	40.9
3 000	31.2	33.0	36.7	38.3	41.1	43.7	46.5	50.4
5 000	35.7	37.5	41.2	42.8	45.6	48.2	51.0	54.9
10 000	41.7	43.5	47.2	48.8	51.6	54.7	57.0	60.9
15 000	45.2	47.0	50.7	52.3	55.1	57.7	60.5	64.4
Effective radiated power, dBm, 400 MHz link								
1 000	28.9	30.7			37.8			46.8
3 000	38.4							
5 000	42.9	44.7			51.8			60.8
10 000	48.9							
15 000	52.4	54.2			61.3			70.3
Note: Plotted in figure 5.								

Required input signal to noise at detector. - For FSK in which an envelope detector is used, the required input signal-to-noise ratio (SNR) is related to the output SNR by the following expression:

$$SNR_i = a + \sqrt{a(1+a)}$$

where

$$a = \frac{B_v}{B_{RF}} SNR_o$$

and

B_v = video bandwidth

B_{RF} = RF bandwidth at input to detector

SNR_i = input signal to noise to detector

SNR_o = output signal to noise at video

For this specific example, the output signal to noise required for a bit error probability of 1×10^{-3} is 13.6 dB or 22.9.

The required input signal-to-noise ratio is then

$$a = \frac{200}{35\ 000} (22.9) = 0.131$$

$$SNR_i = a + \sqrt{a(1+a)}$$

$$SNR_i = 0.505$$

$$10 \log SNR_i = -2.97 \text{ dB}$$

The balance of the calaulcations are shown in table VI.

<u>Link calculations for station-to-orbiter telemetry link PSK/PM modulation</u>. - The PSK/PM link for which sample calculations are shown has a single subcarrier that phase modulates the transmitter. The subcarrier consists of a pseudonoise (PN) synchronization code added to a clock subcarrier modulo 2. The resultant signal is biphase modulated by the data to produce the final input to the transmitter.

TABLE VI. - STATION-TO-ORBITER TELEMETRY LINK CALCULATIONS, FSK MODULATION, 100 BIT/SEC

Item	200 MHz	400 MHz
System losses	5 dB	5 dB
Space loss (1000 km)	139 dB	145 dB
Adverse tolerance	4.5 dB	4.5 dB
Total losses	148.5 dB	154.5 dB
System noise/cycle	-167.4 dBm	-167.4 dBm
RF bandwidth	45.4 dB	48.4 dB
Required SNR_i	- 2.97 dB	-4.79 dB
Required receiver power	-125.0 dBm	-123.8 dBm
Required effective radiated power (ERP)[a]	+ 23.5 dBm (for 1000 km)	+ 30.7 dBm (for 1000 km)

[a]$ERP = P_T + G_T$

where

P_T = transmitter power, dBm

G_T = gain of transmitting antenna, dB

G_R = gain of receiving antenna, dB assumed = 0 dB

See table V for ERP values for other data rates and ranges.

The orbiter receiver coherently demodulates the signal using a carrier phase lock loop and coherent subcarrier demodulator (ref. 3).

This type of system, although near optimum, presents a problem when automatic frequency search is required as in this case because the subcarrier and PN code spectrum fall in the carrier frequency search band. This problem may be circumvented by going to a two subcarrier system and eliminating the PN sync (ref. 4).

The lower frequency subcarrier is modulated (modulo 2) by a square wave at half the bit rate and the higher frequency subcarrier, which is twice the frequency of the lower subcarrier, is modulated (modulo 2) by the data as shown in figure 7. (The bit rate shown is 1000 bit/sec.) At the receiving end the carrier phase lock loop locks onto the incoming carrier, the lower frequency subcarrier is doubled and tracked in a closed loop to provide a reference for demodulating the data subcarrier. Unambiguous bit sync is derived in the second phase lock loop and the data are recovered from the data channel using a matched filter (integrate and dump as shown in figure 8).

Margins for this approach are competitive with the single channel approach and have been used for the 200- and 2000-lb class stations described in Volumes V and VI.

A frequency search mode is required of the orbiter receiver to acquire the transmittal signal because of a frequency uncertainty of the order of 35 kHz.

If one assumes a 30-sec one-way sweep cycle the rate of change of frequency that the phase lock loop will encounter is:

$$\Delta\dot{W} = \frac{2\pi \text{ (frequency uncertainty)}}{\text{sweep period}}$$

$$\Delta\dot{W} = \frac{2\pi \ (34\ 600)}{30}$$

$$\Delta\dot{W} = 7250 \text{ rad/sec}^2$$

This can be rounded off to 8000 rad/sec^2 or 1270 Hz/sec. Since the maximum doppler rate is only 43.8 Hz/sec it will be neglected.

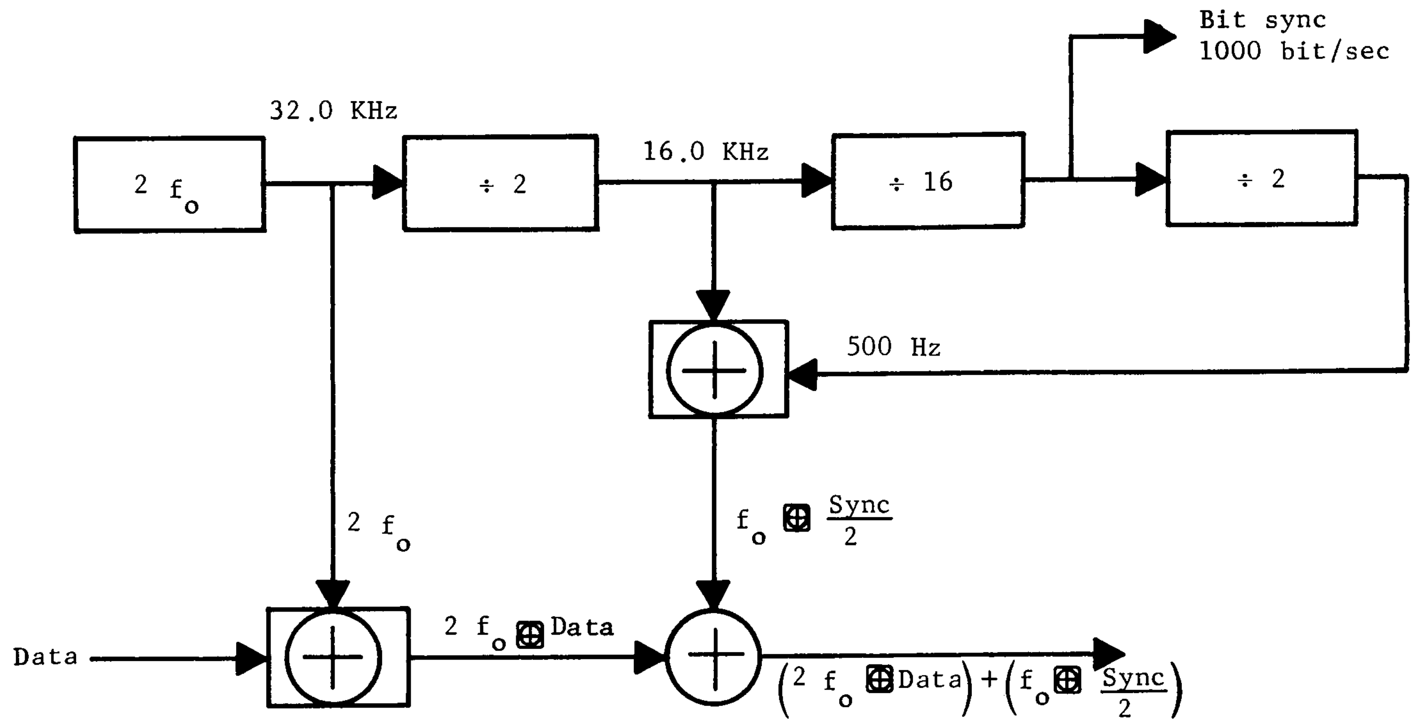


Figure 7. - Two-Channel Modulator

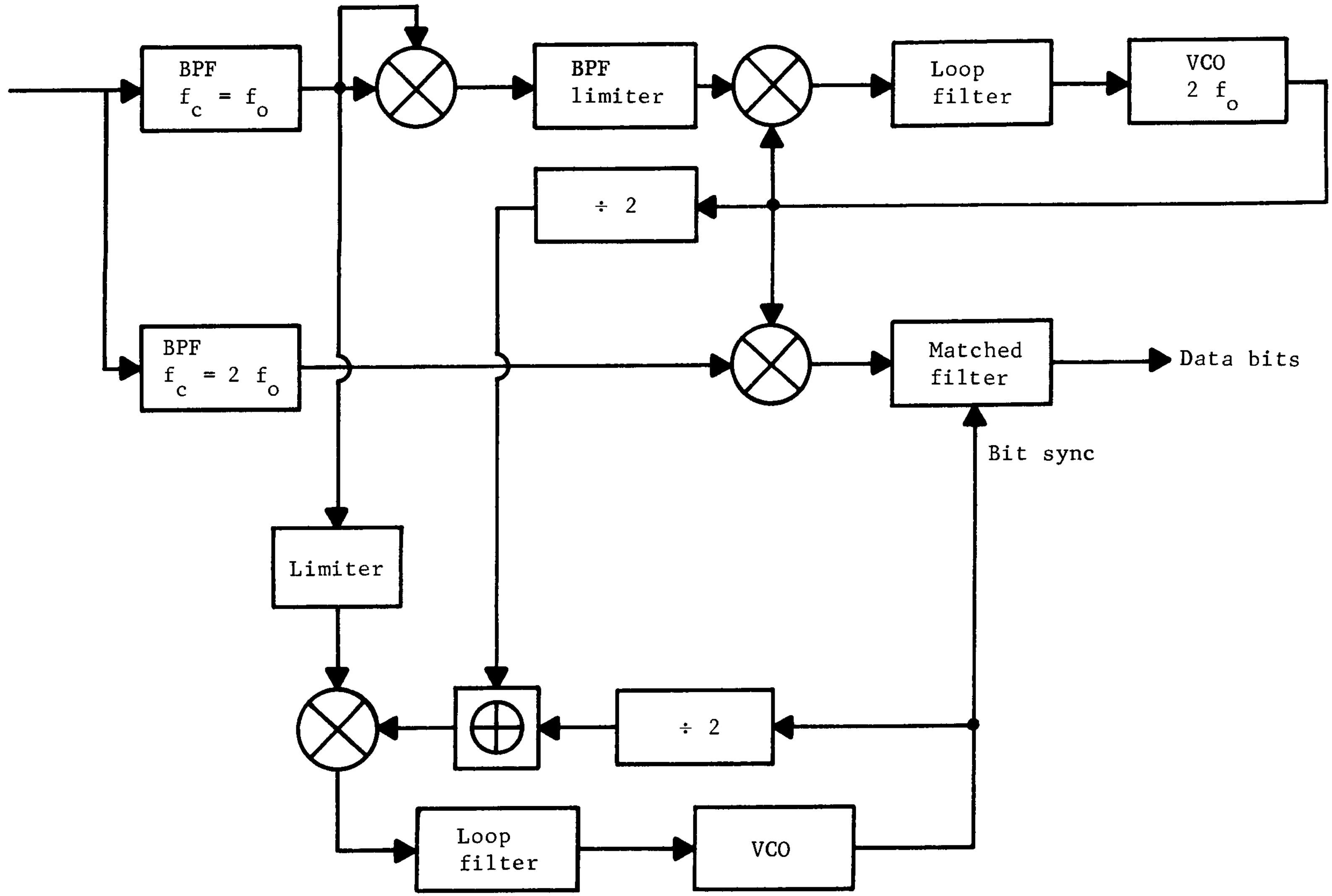


Figure 8. - Two-Channel Demodulator

If no more than a 20° phase error is assumed between the input frequency and the VCO frequency for tracking frequency rates, the $2B_{LO}$ bandwidth required for the carrier tracking loop at threshold is:

$$2\ B_{LO}\ (\text{minimum}) = 20\sqrt{\frac{1}{\Delta\theta}\left(\frac{df}{dt}\right)}$$

$$2\ B_{LO} = 20\sqrt{\frac{1}{20}\ (1270)}$$

$$2\ B_{LO} \cong 160\ \text{Hz}$$

The balance of the link calculations is shown in table VII.

Effect of Station Location on Maximum Telemetry Data Rate

Four station locations have been chosed to examine the results plotted in figure 5 and 6. These locations are shown in figure 7 as points A, B, C, and D. Points A, B, and C are located in the orbital plane at true anomaly θ of 45°, 90°, and 180°, respectively. Point D is located at θ = 180°, but out of the orbital plane so that a maximum communications period of 3 minutes is available.

Assuming a transmitter power of 20 W, and combining station and orbiter antenna gains of 7 dB, figure 5 shows that for an FSK system the maximum data rates that can be used for a bit error probability of 1×10^{-3} vary from 3200 bit/sec for point A to 475 bit/sec for point D. A comparison of the four points on figure 6 shows that were coherent PSK modulation used, the data rates could be increased by a factor of about 10 or, as an alternative, transmitter power could be lowered appreciably.

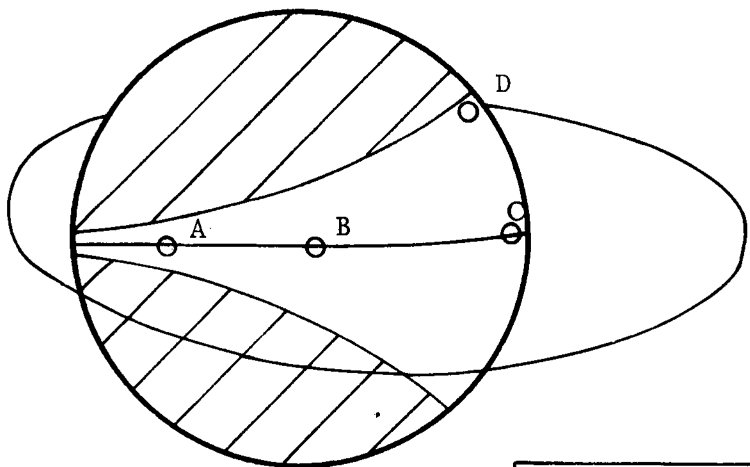
Referring to the table in figure 9 note that for the FSK system the total data transmitted at the maximum rate for each of the four station locations exceeds 1 million bit/orbit except for location D.

To transmit this amount of data per orbit, the transmitter must be on for the full transmission period that is available. For a weight-limited station, this approach is not feasible because of primary power supply limitations; therefore, a fixed period and data rate appears to be most practical.

TABLE VII. - LINK CALCULATION STATION-TO-ORBITER TELEMETRY LINK, PSK/PM MODULATION, SINGLE SUBCARRIER

System losses	5.0 dB	
Space loss (15 000 km)	162. dB	
Adverse tolerance	4.5 dB	
Losses	171.5 dB	
System noise/cycle	-167.4 dBM	
Carrier $2B_{LO} = 160$ ~	22.04 dB	
Threshold S/N	6.0 dB	
Carrier modulation loss	3.57 dB	
Required carrier power	-135.8 dBm	
System noise/cycle	-167.4 dBm	$ERP(dB) = G_T + P_T$
Data and synchronous channel modulation loss	2.52 dB	where:
Bit rate, 100 bit/sec	20.0 dB	G_T = Transmit antenna gain (dB)
Signal energy/bit/cycle	8.0 dB	
Detection loss	0.92 dB	P_T = Transmit power (dBm)
Required subcarrier power	-136.0 dBm	
Required ERP	+ 35.7 dBm	for 100 bit/sec at 15 000 km (assuming 0 dB for receive antenna gain)

Note: ERP values for other bit rates and ranges are shown in figure 6.



Assumed:

Transmitter power - 20 W

Antenna gain $G_T \times G_R = 7$ dB

Modulation, FSK

Station location, A, B, C, or D

	Point A	Point B	Point C	Point D
Maximum range, km	4850	10 000	10 000	12 000
Maximum data rate, bit/sec	3200	685	685	475
Transmission period, minute	17.9	40	106	3
Total data transmitted, bits (assuming full use of transmission period)	3.43 million	1.65 million	4.35 million	85 500

Figure 9. - Maximum Data Rate vs Station Position

For the larger class stations, variable transmission rates and transmitter on-times can be used. Alternatives can be applied to alleviate power-limited situations when necessary, such as reducing sampling rates of the instrumentation.

COMMAND CONTROL

Near real-time direct command from earth via the orbiter as a relay is not practical because of transmission delay time and occurrence of periods when the earth-to-station relay link is not possible because of the link geometry. However, command programs for the station may be transmitted to and stored in the orbiter for relay to the station at appropriate times.

For the smaller class of station only real-time commands for the orbiter storage were considered. For the larger station, the station itself can handle stored as well as real-time commands, if necessary.

Approaches to the Problem

Several command control approaches were considered. For the smallest station, sequential tone and PCM tone command systems were considered because of weight limitations for the station. A PCM/FSK approach could also be taken; however, the tone systems have seen considerable service.

For the larger stations, a PCM system is considered mandatory with coding provisions to provide low probability of acceptance of false commands. Either FSK or PCM/PM systems can be implemented. Possible integration of the command system with the ranging must also be examined.

The control of when to transmit and for how long is one of the important station control requirements. This can be handled in several ways. In the small station, the transmitter can be turned on by detection of the presence of a beacon or command carrier from the orbiter; turn off can be controlled by a fixed delay time. If command capacity and weight allocations are available, the fixed delay time can be adjusted to two or more different values by command after the condition of the primary power system is determined and over a period of time where degradation effects may be determined.

Command Rates and Capacity

Command rates for small stations can be on the order of one command per 10 sec for tone systems, while the larger stations, depending upon their complexity, can use rates an order of magnitude or so greater.

The simple tone systems can handle up to 21 or more real-time commands, PCM tone command systems up to 70 real-time commands, and a typical programable command decoder that handles delayed as well as real time commands has a capacity for handling 256 discrete command words and 32 stored commands.

The commercially available command decoder units in general may not meet the sterilization requirements without some modification.

Conclusions

The command system must be selected to best fulfill the needs for a particular station's programing and sequencing functions. These are best determined during the design of a particular size station. Further considerations are discussed in Volumes V and VI for the various station designs in which detail sequences and command lists were prepared.

The ranging function must also be considered in the selection of the command modulation technique.

Sample link calculations for a PCM/FSK command link for a 15 000-km range and a 10 bit/sec data rate are shown in table VIII. With no antenna gain in the link the orbiter transmitter requirement is approximately 10 W.

Calculations for a tone sequential AM command system and a PSK/PM system are discussed in the 200-lb station volume (Vol V).

RANGING ORBITER TO STATION

The Requirement

A requirement has been identified to measure range between the orbiter and the station twice or more on each communications pass to help identify station location, as discussed in later sections. Range accuracy on the order of ±2 km appears to be desirable.

TABLE VIII. - COMMAND LINK CALCULATIONS ORBITER-TO-BUOYANT STATION, 200 MHz, FREQUENCY SHIFT KEY, 10 BIT/SEC

System loss	5.0 dB	
Space loss	162.5 dB	
Adverse tolerance	4.5 dB	
Losses	172.0 dB	
Receiver noise/cycle	-167.4 dBm	1362°K system temperature
Bandwidth	45.4 dB	34.8 kHz bandwidth
SNR_i	- 8.6 dB	SNR_o = 15 dB
Required receive signal	-130.6 dBm	
Required ERP	+ 41.4 dBm	10 bits/sec, 15 000 km

$$ERP = G_T + P_T$$

where:

G_T = Gain of transmit antenna (dB)

P_T = Transmitter power (dBm)

G_R = Gain of receive antenna (dB) = 0 dB

Possible Approaches

Two approaches can be considered for this application, all of which measure the round trip delay time. These are:

1) Two-day radio - turnaround using range tones;

2) Two-way radio - turnaround digital PN code.

Range tones. - One method of implementing range is to use range tones and measure the relative phase delay. The highest frequency tone is used for measuring the smallest desired increment and the lower tones to resolve ambiguity.

A system used by Goddard Space Flight Center for the STADAN net required six ranging tones (8, 32, 160, 800, 4000 and 20 000 Hz)(ref. 5).

Two subcarriers at 4 kHz and 20 kHz would be required. The four lower frequency tones would modulate the 4 kHz subcarrier to move the baseband frequencies out of the receiver's rf tracking loop bandwidth.

The disadvantage of such a system is the number of required phase lock loops in the orbiter receiving system (one for each tone), the coherent relationships required in generating the tones, the acquisition time, and the complexity in integration of the system with the command and telemetry where power is limited.

Turnaround ranging using PN ranging codes. - A pseudonoise ranging code may be used to determine range in a manner similar to that used by the NASA Deep Space Net in a turnaround ranging mode. The system requires PN code generators and a range extractor for the orbiter plus two-way coherent phase modulation and links.

By using three code sequences to make up the total sequence length, code acquisition time may be reduced to a few seconds after two-way rf carrier lock has been established.

The advantage of such a system is realized when coherent command and telemetry links are used since an integrated command, telemetry, and tracking system can result. Link calculations for such a system are shown in Volumes V and VI.

Conclusions

For integrated coherent command telemetry and tracking the PN ranging code system has an advantage over range tones.

For further discussion see Volume V and VI for the tradeoffs in design of the various size stations.

RANGING FROM THE BUOYANT STATION TO THE SURFACE

The capability of ranging from the station to the surface to an accuracy of ±1 km is a desirable function for the station. No hardware now appears suitable for this application and for the range (100 km) desired. However, a study is now in progress by at least two vendors under NASA contract to complete design studies for an altimeter to operate at ranges up to 15.5 km for use in planetary missions. The contract statement of work "Langley Research Center Statement of Work L-6050, August 19, 1965" sets design goals for extending the range to 100 km for future applications. These design goals are shown in table IX.

Preliminary results reported by Westinghouse Electric Corporation (ref. 6) under NASA Contract NAS1-5953 indicate that with a pulsed radar technique, their choice for the shorter range unit, the ±1 km accuracy is attainable at the 100-km range with increased power and/or antenna gain and other minor changes.

The weight and power objectives of 10 lb and 20 W will be exceeded. Westinghouse estimates weights on the order of 42 to 60.5 lb and average power of approximately 32 W. Principal development items are identified as antenna, TWT transmitter, and integrated microwave circuitry.

A suitable radar altimeter for measuring ranges up to 100 km could be developed given the required lead time. Further, since the data rates indicated in table IX are considerably higher than those required for the buoyant station, both the power and weight requirements could be reduced to some degree.

Reports from the other NASA contractor for a parallel study are not presently available.

TABLE IX. - DESIGN GOALS FOR SPACE PROBE RADAR ANTENNA (PER NASA LRC STATEMENT OF WORK L-6050, AUG 19, 1965)

Parameter	Performance
Altitude range	30 m to 100 km (approximately 100 ft to 330 000 ft)
Accuracy	Maximum error shall not exceed $\pm 1\%$ or ± 5 m, whichever is larger
Frequency	X-band, 8 - 10 GHz
Velocity (maximum	10 km/sec
Acceleration (maximum)	250 g
Antenna size (maximum)	24-in-diam paraboloid (or equivalent)
Power required (maximum)	20 W
Size (maximum)	500 cu in. (including antenna)
Weight (maximum)	15 lb (including antenna)
Data rate and format (radar altimeter)	One reading every 10 sec minimum; 10 readings/sec maximum; least significant bit = 3 m; 16 bit binary code.
Data rate and format (surface roughness)	One reading every 100 sec minimum; 1 reading/sec maximum; resolution = 10 m minimum to 10 km maximum; maximum rate = 5000 bit/sec
Radar reflectivity coefficient	Similar to earth extremes (-20 to -30 dB)
S/N for desired data	Provide a minimum S/N = 20 dB at maximum to minimum range
Mission lifetime	Approximately 2 years total (2 years in inactive or standby condition +2 days in the operating condition)

DROP SONDE TO STATION LINK

Significant parameters affecting the drop sonde-to-station communications link are:

1) Station data storage capacity limitation;
2) Required data rate: 1 to 100 bit/sec;
3) Communications period: 5 to 100 minutes;
4) Communications range, 0 to 100 km;
5) Weight allocation.

The wide variations in the bounds for communications period and bit rate are a result of the variations in the estimated time it takes for dropsondes of different size and ballistic coefficient to fall from the station to the planet's surface for three atmospheric density models. Frequency shift key modulation or on-off keying can readily be used for this link at the data rates shown.

Link calculations for a 100 bit/sec data rate at a nominal 200 MHz carrier frequency are shown in table X.

Assuming a total antenna gain of 0 dB, a minimum transmitter power of 26 mW is required. Included is a 10 dB margin for fading. Further link calculations and design data are given in the design sections of Volumes V and VI for the various size stations.

Reducing the variation in communications period for the dropsonde is very desirable since a severe penalty can be incurred by including data storage capacity for the worst case of high data rate and slow descent.

LINK INTEGRATION CONSIDERATIONS

Some of the more challenging questions that arise in the integration of the various station/orbiter communications links are:

1) Should coherent systems PSK/PM be used and rely on frequency search modes for receivers with finite acquisition times at ends (command and telemetry)?

TABLE X. - LINK CALCULATIONS DROP SONDE TO STATION TELEMETRY LINK, FSK, SPLIT PHASE, 200 MHz

System losses	5.0 dB	
Space loss 200 MHz (100 km)	119.0 dB	
Adverse tolerance	4.5 dB	
Losses	128.5 dB	
Receiver noise/cycle	-167.4 dBm	
Receiver bandwidth rf = 25 kHz	44.0 dB	
Required SNR_i $P_e = 10^{-5}$	- 0.87 dB	
Required signal power	-124.3 dBm	
Required ERP	+ 4.2 dBm	
Add fading margin	+ 10 dB	
ERP with fading margin	+ 14.2 dBm	100 bit/sec, 100 km
with 0 dB gain for antennas, power = 26.2 mW		

$$A = \frac{200}{25\ 000}(31.6)$$

$$SNR_i = A + \sqrt{A(1 + A)}$$

$$SNR_i = 0.819$$

$$10 \text{ Log } SNR_i = 0.87 \text{ dB for } 15 \text{ dB } SNR_o$$

2) If noncoherent systems such as FSK, audio tone digital, or even audio tone sequential (AM modulation) are used for the command link and FSK (no subcarriers) for the telemetry, how is the ranging function handled, especially when power is limited and restricted to low gain antennas?

3) If coherent systems are not used in which telemetry, ranging, and command are integrated, can frequencies be allocated so that a single antenna can be used for all functions (except dropsonde reception)?

All of these design questions had to be faced in designing the 200-lb class station where weight was extremely limited and in designing the larger stations. Elimination of the ranging requirement would greatly simplify the problem; however, this requirement was considered basic and was included in each of the designs.

An example of a completely integrated coherent system with turnaround ranging using a PN range code was shown for the 200-lb class station. Link caclulations and block diagrams are shown in Volume V.

CONCLUSIONS

There are four basic communications links associated with the BVS. They are: command, orbiter to station; telemetry, station to orbiter; telemetry, dropsonde to station; and ranging, orbiter to station.

The ranging function presents the greatest problem when noncoherent links are considered for the telemetry and command, respectively (for example FSK telemetry and AM tone command).

A fully integrated coherent system can be implemented using frequency search techniques; however, care must be taken to keep subcarriers and sidebands out of the search band. This type of system provides almost a 10 dB higher telemetry data rate and enables use of a PN ranging code provided the PN modulation is started after two-way carrier lock is established.

The beginning of data transmission from the station to orbiter should be under the control of the orbiter with a maximum "on" time to prevent damage to the primary power system and to establish a fixed duty cycle for the system. For larger, more complex stations the maximum "on" time may be changed by command to take advantage of longer view times (if the primary power system is flexible).

Initial location of the station in relation to the orbit and the wind patterns will have a significant effect on the number of days communications can be established between the orbiter and the station.

A simple noncoherent FSK system can be implemented for the drop-sonde-to-station link. In this case split phase or bi-phase PCM format should be used to aid in deriving biy synchronization in the station.

Type and capacity of the command system depends on the size and complexity of station operation. Anything from simple sequential tone to PCM stored command programers can be used depending on the need and the desire to integrate the command link with the telemetry and ranging.

The frequency range of 200 to 400 MHz is suitable for use on all links. The choice is highly dependent on the type of antenna used when weight is the determining factor. A 6-dB penalty in transmitter power accompanies use of the higher frequency. Of course, eventually one will be limited by the ability to get the desired frequency allocations.

Finally, there is no reason to believe that feasibility of the BVS concept is jeopardized by inherent limitations on communications. Care must be exercised, however, in planning favorable orbit and station locations.

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FINAL REPORT

BUOYANT VENUS STATION FEASIBILITY STUDY

VOLUME IV - COMMUNICATION AND POWER

PART II - POWER

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POWER SYSTEM STUDIES - TASK 4.4

The contractor shall determine the power requirements of each buoyant station studied and shall investigate the capability of various types of power systems to meet these requirements. The impact of power requirements and power system limitations on station weight and mission lifetime shall be investigated. The study shall include power systems for release probes where applicable.

SUMMARY

Of the three established practical battery systems only the nickel cadmium battery has been successfully sterilized. Extensive work is underway on the silver-zinc system because of its greater inherent energy density. The silver-cadmium system has not been investigated. Other hydrid systems such as the zinc-oxygen system give promise of yet greater energy densities, but are still under development. For long-term operations, one week to six months, batteries are of interest only when used with a constant power source to supply peak loads. The radioisotope thermoelectric generator (RTG) is favored over the solar cell array because it is comparatively environment-insensitive and its waste heat can provide thermal control to other payload components.

Wind-driven generators and radioisotope thermoelectric generators were not considered as power sources for release probes because of uncertainty of the Venusian atmosphere and high specific weight. Silver-zinc, magnesium perchlorate, and ammonia batteries have sufficiently high rates of discharge to satisfy the short-time

power requirements. Only the silver-zinc and the magnesium perchlorate batteries appear amenable to sterilization. It is recommended that the reserve magnesium perchlorate battery be considered for release probes for the Venus Buoyant Station (VBS) because of its high-energy density; 38 W-hr/lb compared to 15 to 20 W-hr/lb for the reserve silver-zinc battery. If ground tests of the actual flight probe batteries are required, manually activated silver-zinc batteries would be substituted.

INTRODUCTION

Paragraph 4.4 of the statement of work requires the contractor to determine the power requirements of each buoyant station studied and to investigate the capability of various types of systmes to meet the requirements. The individual power requirements of the 200 lb and the 2000 lb stations are outlined in Volumes V and VI. This part of the report investigates the capabilities of various types of systems.

Battery characteristics are examined for their applicability to short-term missions, while radioisotope thermoelectric generators (RTG) and solar cell panels are compared for their suitability for use on long-term missions. The influence of sterilization on components will be discussed.

The statement of work also includes power systems for release probes. In the study it is assumed that before sonde release the system will be in the Venusian atmosphere at altitudes from 10 to 80 km with temperatures ranging from 194.2°K to 674.8°K and pressures ranging between 7×10^{-2} to 2.67×10^{4} millibars. After release the atmospheric temperatures and pressures will increase until at impact, temperatures ranging from 650°K to 750°K and pressures ranging from 5.07×10^{3} to 4.05×10^{4} millibars will be encountered.

The small sonde will require 5 W peak power and 5 W-hr energy, while the large sonde will require 25 W peak power and 25 W-hr energy.

SYMBOLS LIST

A-h	ampere-hour
AU	astronomical unit
in.	inch
kg	kilogram
lb	pound
m	meter
mm	millimeter
mRem	milliroentgen equivalent man
psig	pounds per square inch gage
W-hr	watt hour
°C	degrees Celcius
°F	degrees Fahrenheit
°K	degrees Kelvin

BATTERIES

Secondary Systems

The secondary systems of greatest recent interest and that have become established as practical batteries are the nickel-cadmium, silver-cadmium, and the silver-zinc. The nickel-cadium has enjoyed widespread use in commerical devices, while the silver batteries have largely been limited to military and space service. The nickel-cadmium battery has the longest life. The silver-zinc system yields the highest energy density, but is characterized by a relatively short activated life stemming from the nature of the separators and the solubility of the zinc electrode, as well as its tendency to "tree." The silver-cadmium system combines some of the advantages and disadvantages of the nickel-cadmium and the silver-zinc systems since it adopts one electrode from each. Both the silver-cadmium and the silver-zinc systems exhibit the poor voltage regulation of the silver electrode. The silver cadmium system also yields less energy since its operating potential is 0.4 V lower than that of the silver-zinc system. This is in consequence of changing from the short-lived zinc-electrode to the more stable cadmium electrode.

The nickel-cadmium and the silver-cadmium cells are designed for cyclic operation. Cycle life is a function of temperature and depth of discharge. In general, nickel-cadmium cells are capable of 10 000 cycles and silver-cadmium cells are capable of 1000 cycles. Even though some designs of silver-zinc batteries are designated as primary, they are actually capable of several charge-discharge cycles. This permits them to be used in checkout tests necessary to establish confidence in the operability of the system before launch.

Performance. - A large number of chemical and mechanical characteristics may be varied in the design of a battery depending on the characteristics desired. Consequently the energy density may vary considerably for a given chemical system. For comparison purposes, data are given in Table I on energy output of unsterilized batteries, and retention of charge for the three battery systems as a function of temperature.

TABLE I. - COMPARISON OF THREE BATTERY SYSTEMS

Rate, hr	Energy density, W-hr/lb								
	Ni-Cd			Ag-Cd			Ag-Zn		
	80 °F	0 °F	-40 °F	80 °F	0 °F	-40 °F	80 °F	0 °F	-80 °F
5	17	12	10	30	24	14	50	38	5
1	12	10	8	27	19	0	46	35	0
¼	11	8	5	21	13	0	39	30	0

Temperature, °F	Charge retention, days to 50% rated capacity	
	Ni-Cd	Ag-Cd and Ag-Zn
75	300	730
120	25	170
160	10	20

Sterilization. - Sterilization efforts to meet NASA requirements have been carried out on nickel-cadmium and silver-zinc batteries. Hermetically sealed nickel-cadmium cells have been heat-sterilizable without large deterioration in capacity and no apparent structural damage. Tests (ref. 2) showed sterilization in the discharged-shorted condition produced a general and immediate loss of about 25% of initial capacity, but continued cycling showed little additional loss in capacity and consistent charge-discharge characteristics up to the end of the 300-cycle test conducted. Performance is about the same as expected for a nickel-cadmium cell that had not been heat sterilized. Little work has been done or reported on silver-cadmium cells. Some of the problems caused (ref. 3) by the sterilization heat cycle in silver-zinc batteries are:

1) Reduction of the silver oxide in the cathode;
2) Increased solubility of the zinc anode in the potassium hydroxide electrolyte;
3) Dendritic growth from the electrodes, causing puncture of the separators;
4) Rapid deterioration of organic separators causing electrolyte contamination;
5) Formation of gases (hydrogen) evolving from zinc at high temperatures;
6) Terminal and case sealing problems.

The battery performance suffers from these problems as follows:

1) Loss of capacity (35 to 50%);
2) Reduced wet stand life (6 to 8 months);
3) Pressure buildup and cell rupture;
4) Reduced open circuit voltage.

A $1 million a year program is underway in an attempt to develop silver-zinc batteries capable of withstanding sterilization. JPL has contracted with Radiation Applications, Inc., and Narmco for the development of separators. Contracts are also in effect for work on complete batteries with Delco-Remy and Electric Storage Battery Company. NASA Lewis Laboratories have been supporting work at the Astropower Laboratory of Douglas Aircraft on an inorganic separator. Their proprietary effort was originally directed merely at obtaining a wider temperature range for operating silver-zinc batteries. The separator has now proven itself capable of sustaining well over 1500 cycles at 25°C in 5-A-h cells (ref. 4). At 100°C, these cells have lasted over 600 cycles. The depths of discharge, 35 and 50%, respectively, open up the possibility of long-lived temperature-tolerant, secondary silver-zinc cells. Their wet-stand life at 25°C (but not for 100°C) for 1 month is good. They have been sterilized at 145°C for three 36-hr periods and according to preliminary test data, suffered no significant damage. Each of the two sterilized cells has been cycled more than 200 times at room temperature so far, one at 10% and the other at 17% depth of discharge. Development work on these sealed pressure relieved cells is continuing under NASA Lewis Laboratories sponsorship. Recently NASA Ames negotiated a contract with Astropower. The second quarterly report (ref. 5) has been issued. The objective of the contract is to develop a 1 A-h heat sterilizable sealed cell capable of withstanding sterilization pressure and still performing a duty cycle after 10 months wet life. The specified temperature range during transit and operation is 283°K to 302.4°K (50°F to 85°F). The first quarter was devoted mainly to study of component contribution to pressure and gas composition during the sterilization procedure. Several tests on silver and zinc oxide electrodes, separators and interseparators, sealants, and combinations in various electrolytes gave pressures as high as 3600 mm Hg absolute and as low as 1400 mm Hg. The gas analysis generally showed a typical decrease in oxygen content, very little hydrogen, and organic gases at times up to 5% (mainly methane). In the second quarter experimental cells were assembled. The vented cells developed for NASA Lewis have a specific output of 48.4 W-hr/kg (22 W-hr/lb). Astropower expects that the specific output of production sealed cells will be considerably higher.

Packaging. - Cells are available in a wide variety of sizes in both cylindrical and prismatic shapes. Cylindrical sizes vary in capacity from 0.5 to 5 A-h, while prismatic shapes are available in capacities from 1 to 500 A-h. Nickel-cadmium cells are housed in stainless steel while silver-cadmium and silver-zinc cells are encased in plastic or a plastic-metal combination. For space use, sealed cells with very low gas leakage rates are obtained on stainless steel cans by use of hermetic ceramic-to-metal seals. More recently cells are also being developed using bonded rubber seals. Silver-zinc and silver-cadmium cells have been epoxy sealed. A variety of arrangements have been used to support cells so that they may be used as a battery. Cylindrical cells have been epoxy encapsulated while prismatic cells have been packaged in both canisters and open frames. Cylindrical cells are inherently able to withstand the buildup of internal pressures; prismatic cells need mechanical constraint to prevent bulging that would cause an increase in internal resistance.

Reliability. - Achieving high reliability is largely a matter of applying a battery so that its capabilities are not exceeded in the areas of temperature, charging control, and discharge rate and capacity. Test programs simulating the actual conditions to be encountered are essential to develop confidence in the battery to accomplish its mission. Under these conditions reliabilities approaching one have been achieved.

Future Developments. - Additional research to increase understanding of the couple processes and development of cells to improve their specific output and life under space conditions is being continued. Working under NASA contracts to develop a separator for a sterilizable silver-zinc battery, Monsanto Research Corporation is evaluating ligand polymers; Westinghouse Electric Corporation is examining composite membranes; and Southwest Research Institute is investigating grafting of polyethylene film with acrylic acid using irradiation. Electric Storage Battery Company is incorporating separator material developed by RAI Research Corporation into sterilizable cells able to resist high impact shocks. Martin Marietta at Denver has an internal program for testing and evaluating sterilized batteries.

Primary

Silver-zinc. - The value of silver-zinc batteries lies in the unusual electrochemical properties of the silver components employed. These provide a large output per unit of weight during discharge and the current is delivered at a nearly constant voltage. Silver batteries may be discharged at very high rates. Under discharge the two silver oxides, silver dioxide and silver peroxide are reduced to metallic silver. Since metallic silver is more dense than the oxides, the pores of the plates open as discharge progresses. This accounts in part for the flat characteristic of the discharge curves.

Silver-zinc batteries are made in both the reserve and the nonreserve forms. The reserve design incorporates an automatic activation system consisting of an electrically fired gas generator that supplies propulsive force to drive the electrolyte contained in a metal tube electrolyte reservoir coiled around the battery probes. This arrangement is widely used for weapon systems where it is essential to have the system inert before use because of the unknown time of storage. The energy density varies from about 15 to 20 W-hr/lb including the activation equipment. Stand time after activation is about 12 hr since a minimum amount of separator material is used between the plates. Temperature range is 4.4°C to 49°C; when the battery is electrically heated before release, the low temperature is extended to -54°C.

Nonreserve silver-zinc batteries of the sealed type have been used for scientific space missions where it is desired to check out performance before launch. The batteries used in Mariner IV, rated at 27 V and 50 A-h, had a specific energy of 38 W-hr/lb, as shown in reference 6. The capacity retention after one year storage at 32°C was 90%. Design to give a long stand life necessitates that many more layers of plate separator material be used for the nonreserve battery than for the reserve type. This results in a decreased terminal voltage under high discharges, which will probably be accented with the separator materials needed in a sterilizable battery.

Zinc-oxygen. - Recently there has been considerable interest in the zinc-oxygen battery system and development is underway by several companies (ref. 7). This is actually a type of fuel cell using metallic zinc as the fuel and gaseous oxygen as the oxidizer. The system is closely related to the zinc-carbon electrode air cells, but uses pure oxygen in place of air. Relatively high rates can be imposed and extremely high efficiencies are realized from the active materials. The system consists of electroplated porous zinc oxides and powdered silver cathodes containing platinum as a catalyst. The oxygen is imposed on the systems at 150 psig nominal, with no pressure difference between the cathode and anode compartments. The system is handled as a primary battery, with the exception that oxygen must be supplied before power can be obtained. The cells are fabricated with the zinc electrode in a charged state and are activated with electrolyte just before use. Neglecting actual oxygen tankage, the system can be expected to approximately double the energy output of presently available silver-zinc batteries. If individual oxygen tankage is a requirement, it is estimated that the output would be about 1½ times that of the best silver-zinc batteries. A battery with a capability of approximately 12 kWh at an 8-hr rate would produce 90-100 W-hr/lb, including its own oxygen tankage. A unit intended to operate at a 400-hr rate would deliver somewhat greater than 150 W-hr/lb. For operating periods of several hundred hours, this system would appear to have sufficient advantages over silver-zinc batteries in its specific output and over fuel cells in simplicity, that it could become a contender as a space power system. The cell operation produces a reaction product that is essentially zinc oxide, a solid material that remains within the pores of the fuel electrode and creates no problem of disposal. Contact with the manufacturer indicates that no tests have been conducted to determine if the battery can be heat sterilized. It is possible that the platinum catalyst in the cathode may be adversely affected. The upper temperature is also limited by the side reaction of zinc in the electrolyte. At temperatures above 38°C (100 °F), the tendency for this reaction to occur increases. Continuous operation above 49°C (120°F) is not recommended.

Magnesium perchlorate batteries. - In the last few years the U.S. Army Electronics Command has been sponsoring development of production-feasible magnesium/magnesium perchlorate/mercuric oxide batteries (ref. 8). The use of magnesium perchlorate has several advantages over the electrolytes normally used in magnesium batteries, the most pronounced being the slower corrosion rate of magnesium in this solution. The major problem with the use of this system is the control of the heat involved from the heat of solution of the magnesium. The batteries are built in the reserve configuration, and may be activated with a solution of 5-normal electrolyte or by the addition of water to anhydrous electrolyte crystals contained within the cell. The cells may be modified for high- and low-rate applications by varying the positive and negative plate thickness and increasing or decreasing the number of plates per cell. Tests have been made in the temperature range from -40°C to 52°C.

A high-rate battery discharging its nominal capacity in 1 hr has given 38 W-hr/lb. Wet stand life is one to two months.

Ammonia batteries. - The liquid ammonia-activated battery has been established to a reasonable point of adequacy for fuse-type applications (ref. 9). The batteries have several advantages including high capacities per unit weight and volume, a wide operating temperature range, and ability to be packaged to meet a diversity of operating conditions. The high vapor pressure of liquid ammonia at ordinary temperatures necessitated the development of special packaging techniques. The batteries can be reserve activated under full-load conditions in 100 msec and can be stored for considerable periods of time over a wide temperature range with virtually no change in ultimate battery performance.

Sheet magnesium anodes with either mercuric sulfate or silver chloride cathodes have been used in multicell work and yield approximately 2.3 V open circuit. These cathodes are capable of 20- to 30-minute rate discharge over the -54°C to 74°C temperature range. Units built have given a total of 15 W-hr for an energy density of 4.5 W-hr/lb. Projected energy density is 8 W-hr/lb.

Mercury cells. - Mercury cells were developed in the early part of World War II. They consist of a depolarizing cathode of mercuric oxide, an electrolyte of potassium hydroxide with potassium zincate and a zinc anode. The cells are not adapted to short-time use since the internal resistance is relatively high. It increases nearly linearly until 90% of the discharge is completed. (After that the resistance increases more rapidly). The energy density declines from 45 W-hr/lb at the 100-hr rate to 25 W-hr/lb at the 24-hr rate.

Alkaline zinc-manganese dioxide cells. - The alkaline zinc-manganese dioxide system has been established commercially to supersede the old standard LeClanche dry cell where a low cost, fairly high current-producing cell is needed for continuous duty applications. The LeClanche discharge reaction quickly produces salts that impede further ion movement into the cathode, and as a result, its output varies critically with drain rate. The alkaline cathode, on the other hand, regenerates electrolyte during discharge and remains highly conductive. This results in over a seven to one increase at heavy drain (ref. 10) as shown below for a "D" size cell discharged at 0.5 A to 0.8 V.

	Time, hr	Capacity, W-hr
Alkaline	20	7.6
LeClanche	2.5	1

This system for low discharge rates shows an energy density of 30 W-hr/lb. Discharge in 20 hr is considered a high rate. The cell is therefore not adaptable for short-time probe use because of the large drop in terminal voltage.

RADIOISOTOPE THERMOELECTRIC GENERATORS (RTG)

The RTG generates a usable voltage by converting the heat produced by the decay of a radioactive isotope directly into electricity by maintaining a temperature difference across thermocouples. With existing designs about 5% of the heat is converted to electricity. The balance is radiated at a temperature level of 150°C to 205°C (300°F to 400°F). Existing designs use lead telluride couples. Experimental work and system studies have been made using silicon-germanium alloy couples. This system would reject heat at 260°C to 315°C (500°F to 600°F).

Of the nine nuclides available for power purposes Plutonium 238 is the most desirable because of its long half-life (87.6 years) and low radiation level. At a distance of 1 m from an unshielded 5-W RTG, the neutron dose would be 3.4 mRem/hr while the gamma dose would be only 0.5 mRem/hr. The maximum permissible exposure is 100 mRem for a 40-hr work week, or an average of 2.5 mRem/hour. Since the average rate may be exceeded for limited periods of time only, nominal control of access to the RTG is required. No damage will be caused to electronic equipment.

By far the most significant hazards are those associated with inhalation and ingestion of fuel particulates. Plutonium offers relatively small direct radiation dose rates, even for inventories in the megacurie range. However, just one inhaled particle in the micron size range is sufficient to cause a body to receive a maximum allowed radioactive burden. For this reason the AEC requires a safety analysis to be performed once the fuel form, generator design, and mission trajectories have been defined. The analyses determine conformity of design to nuclear safety criteria and establish design constraints to meet the general criteria. The basic nuclear safety criteria are stated in terms of the probability of inflicting radiobiological damage to a number of people. Those accidents which can give rise to a hazardous condition are launch fires, reentry burnup, land impact, ocean submergence and earth burial. In designing the fuel capsule, therefore, consideration is given to fuel form, its chemical activity and compatibility with the container materials, the capsule impact stresses, and the burial, operational, and reentry heat transfer requirements. A number of RTGs developed by Martin Marietta have seen space use. Two SNAP-3 generators, rated at 2.5 W were launched in June and November 1961 to power Navy navigational satellites. SNAP-9A generators with a 25 W rating were launched with additional navigational satellites in September and December of 1963. Launch of the SNAP-19 generator system is scheduled for late 1967 or early 1968. It will be used as a portion of the power supply for the Nimbus-B meteorological satellite.

A 50-W RTG (SNAP-27) will be emplaced on the moon by an astronaut as part of an Apollo experiment. The RTG technology, developed over 10 years, is now available for application to the Venus program.

Thermoelectric Materials

The factor that affects generator efficiency the most is the thermoelectric material used to convert heat to electricity. Two materials are commonly considered for space power, lead telluride and silicon-germanium alloy. Lead telluride possesses a higher figure of merit (a part of the efficiency expression) than silicon-germanium so that it can be operated at a lower temperature and still give the same efficiency. It has a maximum operating hot junction temperature of 1100°F compared to 1800°F for silicon-germanium alloy. Nuclear safety considerations, however, limit the useful hot junction temperature to between 1500°F and 1600°F. There are other differences between the two materials that are significant. Silicon-germanium alloy is not harmed by either oxidizing atmosphere or vacuum. Lead telluride is degraded by oxidation and must be sealed with an inert gas such as argon. Silicon-germanium alloy is also stronger mechanically then lead telluride, and able to withstand the dynamic loading of a rocket launch with less support.

Heat Dissipation

The level at which heat is dissipated becomes an important consideration when the presence of a sterilization canister surrounding the space capsule is considered. The RTG may be sealed inside the canister before sterilization with heat conducted through the skin of the canister before dissipation. To achieve minimum weight (ref. 11) design studies have shown that the optimum cold junction temperature is 350°F to 400°F for lead telluride compared to 500°F to 600°F for silicon-germanium alloy.

A parametric study was made on RTGs that would dissipate their waste heat directly to space by radiation or to the Venusian atmosphere by convection/radiation from radial fins and housings. Silicon-germanium alloy was selected for the thermoelectric conversion system. Design temperatures were as follows:

Design point ambient temperature	350°F	176.7°C
Maximum steady-state ambient operating temperature	600°F	315.6°C
Maximum 1-hr transient operating temperature	700°F	371.1°C

Packaging

For the Venus mission the heat source, thermoelectric modules, and thermal insulation would be packaged in a cylindrical housing. Fins would be provided to increase the effective radiating area and to permit convective cooling when in the Venusian atmosphere. Results of the Martin Marietta parametric study are shown in Tables II and III.

TABLE II. - RTG WEIGHT SUMMARY

Item	Power output, W			
	20	30	40	60
Heat source, lb	9.0	14.5	19.0	29.5
Thermoelectric modules, lb	4.1	6.2	8.2	12.4
Thermal insulations, lb	2.5	3.0	3.5	4.5
Housing and heat source support, lb	2.2	2.5	3.5	5.0
Radiator fins, lb	4.0	6.0	8.0	12.0
Total RTG weight				
Pounds	21.8	32.2	42.2	63.4
Kilograms	9.9	14.6	19.2	28.8

TABLE III. - RTG PHYSICAL CHARACTERISTICS

Item	Power output, W			
	20	30	40	60
Height, in.	11	11	22	22
cm	28	28	56	56
Diameter housing, in.	5	5.5	5	5.5
cm	12.7	14.0	12.7	14.0
Diameter across fin tips, in.	13.75	14.25	13.75	14.25
cm	35.0	36.2	35.0	36.2
Output voltage, V	5.0	7.5	10.0	15.0
Number of radial fins	5	6	5	6

Sterilization

As seen above, the minimum operating temperature is 176.7°C, which is well above the sterilization temperatures of 135°C and 145°C. The sterilization problem then does not concern the biota load on the RTG, but prevents internal temperatures within the RTG from exceeding design limits when the sterilization canister is heated. The internal temperatures of the RTG can be limited by:

1) Short-circuiting the RTG, which lowers its hot junction temperature by about 50°C;

2) Providing forced circulation cooling of the RTG fins during the sterilization cycle.

In spite of extensive research efforts no breakthroughs have occurred in thermoelectric materials. Rather continued progress has been made in processing and combining of existing compounds and in improved engineering practices. This trend is expected to continue with a substantial increase in efficiency and consequent improved specific output.

Reliability

RTGs are inherently reliable because they have no moving parts. Of all the generators manufactured by Martin Marietta, only one has experienced a complete loss of power (open circuit). One unit successfully powered a remote weather station on the island of Axel Heiberg from August 1961 to August 1965.

The SNAP-7 program was initiated in 1950 at the Baltimore Division of the Martin Marietta Corporation. The designed life of these generators is 10 years. A total of more than 19 years of continuous operation has been accumulated. This particular development program has proved the inherent reliability and usefulness of such power supplies for long-term unattended operation in remote stations, as well as for undersea beacons and communications links. The SNAP-9A program was initiated in August 1961. The first system was successfully placed in orbit on September 28, 1963; the second in December; the third failed to achieve orbit when launched in April 1964. The second SNAP-9A is still successfully powering the payload. The satellite with the first unit developed an electrical short circuit three months after launch, however, telemetry, powered by an auxiliary solar-powered circuit, shows that the generator continues to operate.

Current development work by Martin Marietta on the SNAP-17, SNAP-19, and SNAP-29 continues to emphasize high reliability in thermoelectric systems.

SOLAR CELL ARRAYS

Two factors cause the expected output from a solar cell array to be intermittent. The buoyant station may be present for long periods on the dark side of Venus, or it may be below the tops of the clouds. For continuous operation of equipment a battery would have to be used. Present knowledge of the buoyant station is not adequate to permit battery sizing. The other alternative would be to use sunlight when it is available and to accept intermittent operation of the station.

The distance of Venus from the sun varies from 0.718 AU to 0.728 AU, resulting in 2710 to 2640 W/m^2 of radiation being available. Solar cell arrays using silicon solar cells have had extensive use in space and there is a strong incentive to continue use of system because of reliability considerations. Of the energy in the sunlight about 10% can be converted to electricity by silicon solar cells at normal temperatures. As the sun is approached, the equilibrium temperature of a solar cell panel increases. Increased temperature results in a decrease in cell efficiency until finally at 230°C no electrical power is generated. Figure 1 shows a plot of the output that may be expected from a typical array as a function of distance from the sun. The array is normal to the sun's rays and is in space. Arrays may be built to withstand launch conditions with a unit weight of 4.9 kg/m^2 (1 lb/ft^2). Thus the specific output of an array in sunshine at Venus can be 30.9 W/kg (14.0 W/lb). For fixed installation on a buoyant station a number of panels would have to be installed or else the solar cells would need to be distributed around the periphery of the station. This would lower the specific output by a factor of five, assuming panels were placed on the top and on four sides of the station.

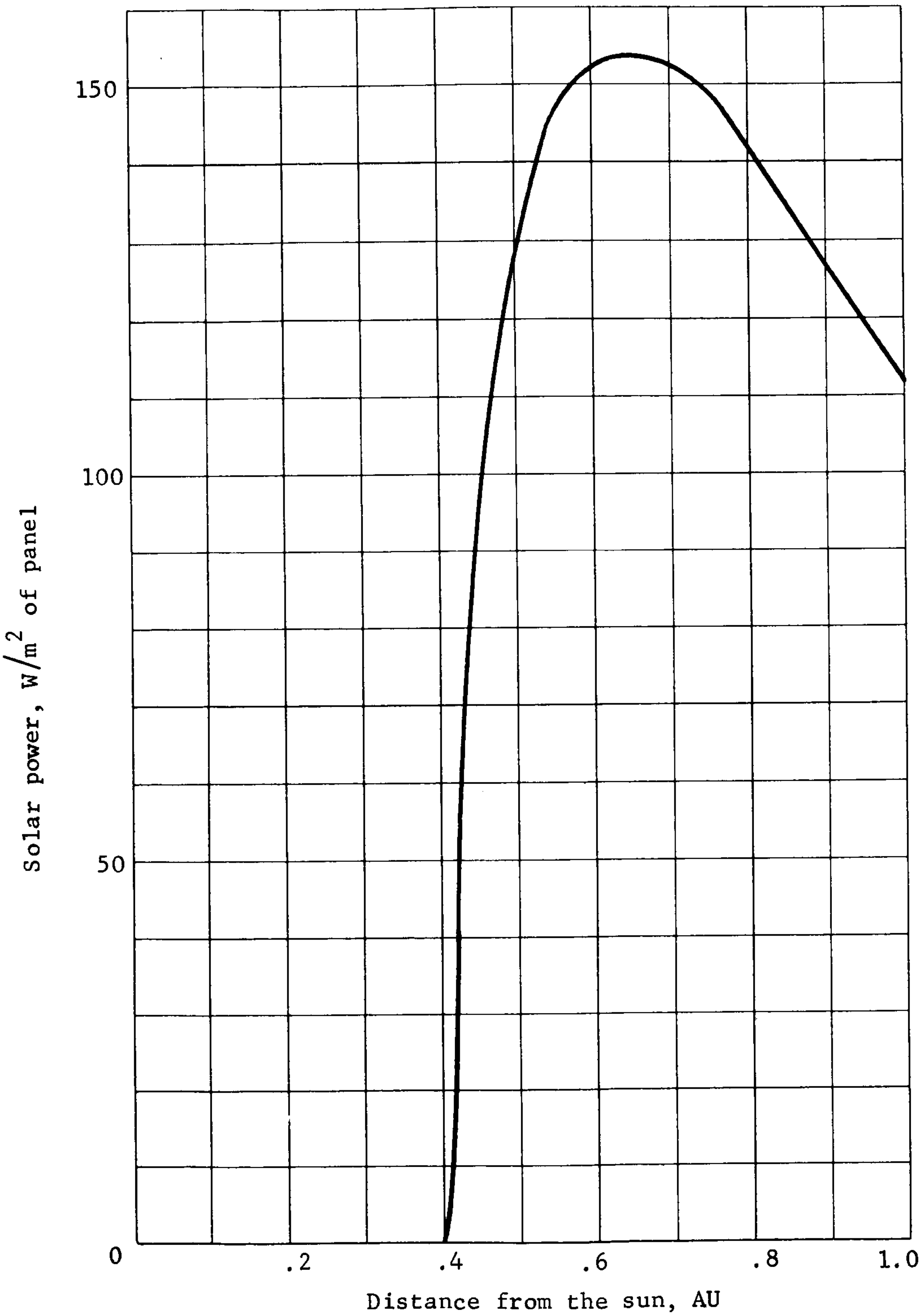

Figure 1. - Solar Cell Array Output

WIND-DRIVEN TURBINE

The wind-driven turbine offers a possibility for powering a dropsonde by virtue of the potential energy it possesses when released. As demonstrated in Volume III, the sonde reaches high velocities quickly in the upper atmosphere and is slowed when denser portions of the atmosphere are encountered. The sonde is converting its potential and kinetic energy into heating the air. The rate of conversion is lowest at the terminal velocity just before surface encounter. Figure 2 shows the rate in watts at which energy is being dissipated to the atmosphere by a 5-lb sonde as a function of ballistic coefficient for the three model atmospheres.

Some of the energy normally dissipated to the atmosphere may be diverted to useful form by a small wind-driven turbine. Energy and momentum considerations show that the maximum power that can be extracted is 59.2% of that available in the intercepted airstream. The power required by the sonde is 5 W. Applying a 25% turbine efficiency and a 50% generator efficiency, it is found that 67.6 W must be available in the intercepted airstream. This compares with the limits of 80 and 1360 W shown in figure 2. Since the energy extracted by the turbine should be only a fraction of that available, ballistic coefficients lower than 0.5 should be avoided for a sonde equipped with a wind-driven generator.

In considering turbine design note that dynamic pressure is only a function of the ballistic coefficient while velocity is both a function of ballistic coefficient and atmosphere model. The differences in velocity might be accounted for in design by using a variable pitch turbine. The estimated weight of a 5-W system is 0.6 lb for the generator, 0.1 lb for rectifier and control, and 0.3 lb for the turbine, for a total of 1.0 lb.

COMPARISON AND SELECTION OF POWER SOURCES

Buoyant Station

The selection of a power system for the buoyant station depends on mission duration. A battery is energy limited while the RTG and solar cell arrays are power limited. Consequently, the battery weight required for a mission is proportional to its duration, while the weight of the other two systems depends on maximum power demands and is nearly independent of mission duration.

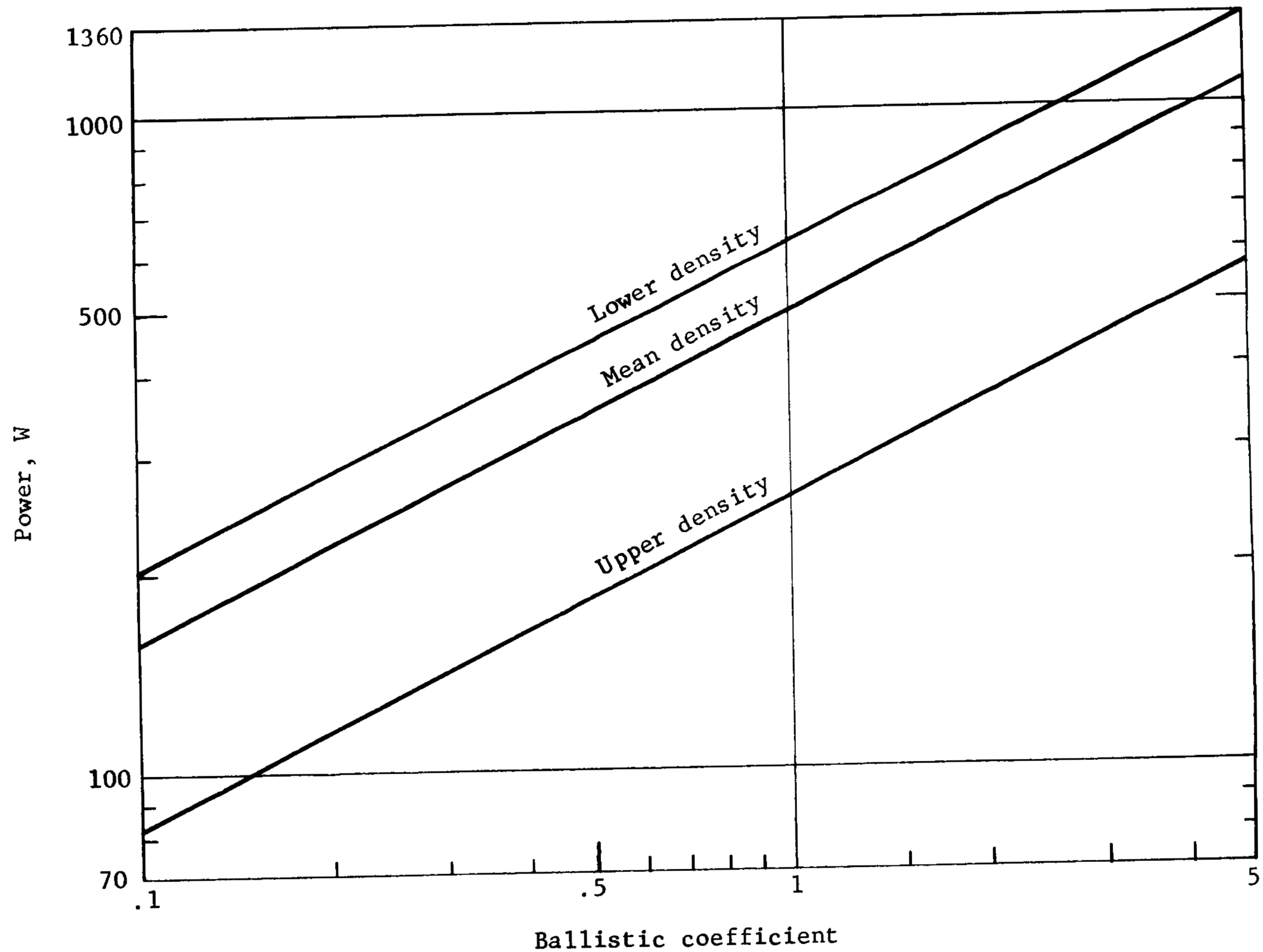

Figure 2. - Rate of Energy Dissipation of 5-lb Sonde at Terminal Velocity

Figure 3 shows weights versus mission duration plotted for four systems each supplying 60 W continuously. The two batteries shown have energy densities of 55 W-hr/kg (25 W-hr/lb) and 220 W-hr/kg (100 W-hr/lb). These values are representative of what may be expected of a sterilized silver-zinc battery and a zinc-oxygen battery, respectively. The RTG and solar cell array weights are those previously given. The figure shows that for mission durations of more than a few days the constant power sources possess a weight advantage.

Power systems have peak loads imposed on them because of intermittent communication demands. Thus, for practical systems, minimum weight will be achieved if a battery is used to supply peak loads while the constant power source supplies the average load, including battery recharging. Such an arrangement is shown in figure 4 where an RTG is used as the power source. At light loads power is supplied from the RTG through the converter-regulator, while the battery is under charge. When peak power demands are encountered the battery is switched to the loads. The arrangement offers some redundancy to the RTG since the battery is able to carry the loads long enough for telemetry information concerning a malfunction to be received.

Release Probe

Two constraints strongly affecting the selection of the battery system are requirements that the battery be able to deliver its capacity in 1 hr and that it be heat sterilizable. The environmental factors encountered during the various phases of the mission before probe release are not governing since they are similar to those encountered on other missions where batteries have been used. After probe release, temperatures above battery capabilities will be encountered. This will require use of thermal insulation and heat absorbing materials. High pressures will also be encountered, which will require that cell cases be either sufficiently strong to withstand the pressures or release vents provided.

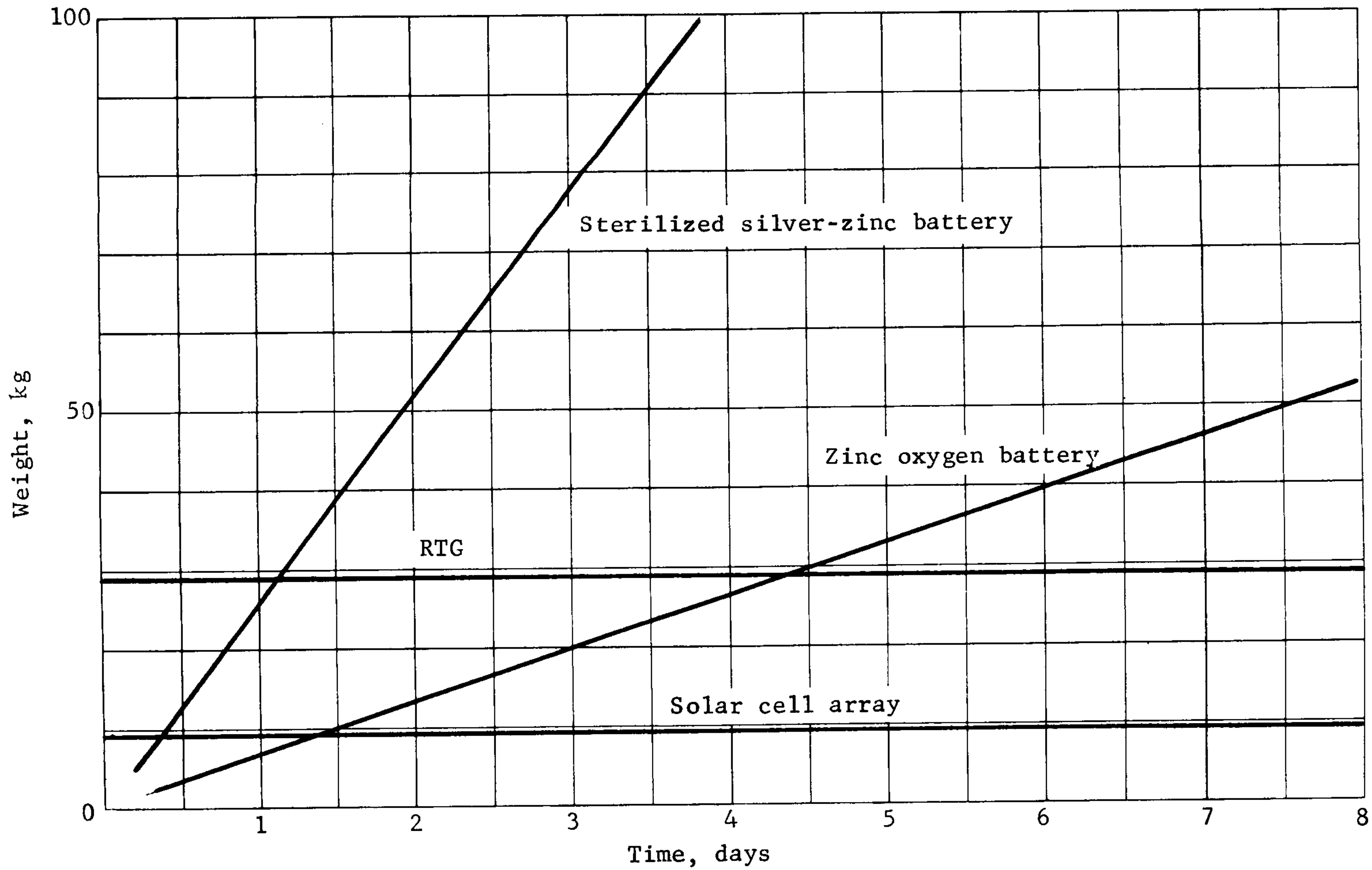

Figure 3. - Mission Power Weight vs Mission Duration

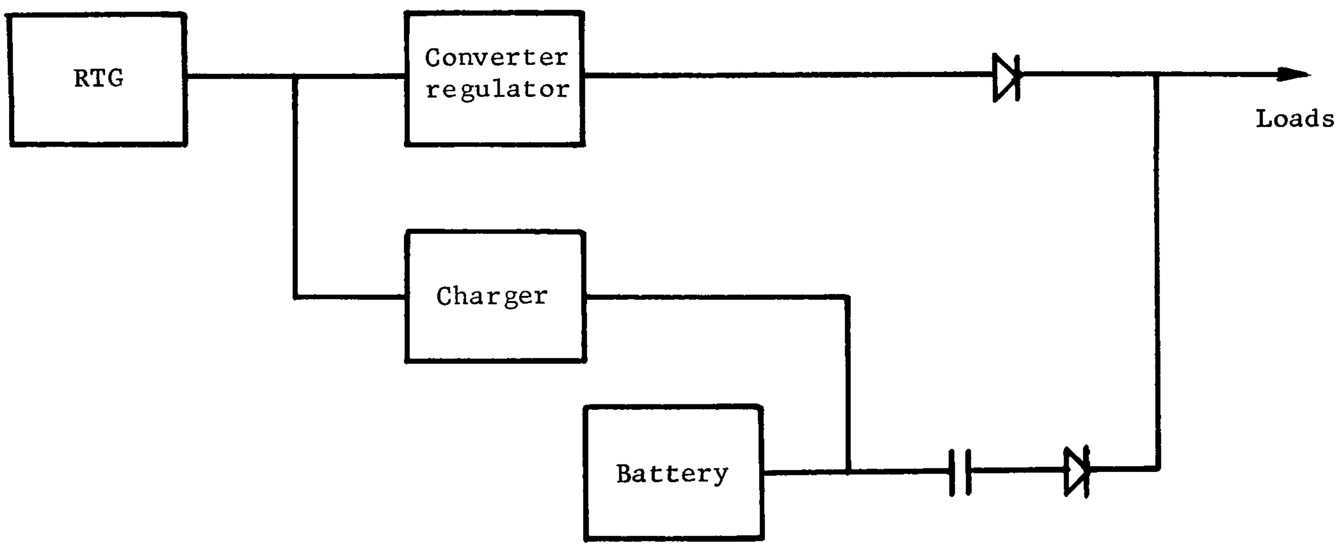

Figure 4. - Power Systems Block Diagram

The folowing tabulation compares energy densities for the battery systems considered.

System	Energy density, W-hr/lb
Silver-zinc	
Reserve	15 to 20
Nonreserve	30
Magnesium perchlorate	38
Ammonia	8
Mercury	45 at low rates
Alkaline zinc-MnO_2	30 at low rates

Of the systems listed, sterilization programs are underway on only the silver-zinc system. One of the problems encountered in heat sterilization is reaction between the electrolyte and the separator material. This would be avoided in the reserve-type battery where the electrolyte is separately contained.

Some difficulties have been encountered with the sterilization of pyrotechnics. However, electrical match materials and gas generators that will withstand sterilization temperatures have been developed (ref. 12). It appears at this time that little penalty may be associated with the sterilization of reserve batteries. As previously stated, reserve batteries cannot be pretested, however, since a number of release probes will be used, this may not be a serious objection.

The magnesium perchlorate and ammonia batteries can also be made in the reserve configuration. Containment of the 5-normal magnesium perchlorate solution is simple since the vapor pressure at the sterilization temperature of 135°C is only one atmosphere. Containment of ammonia will be difficult since the sterilization temperature is above the critical point of ammonia, 132.35°C and 111.3 atmospheres of pressure.

In the magnesium perchlorate battery, magnesium is used as the anode and either mercuric oxide or manganese dioxide as the cathode. Contact with the manufacturer indicates that manganese dioxide would be favored in a sterilized form of the battery as being more stable than mercuric oxide. In the reserve configuration no major problems are foreseen in surviving the sterilization cycle. The magnesium perchlorate battery has fewer problems and is simpler to manufacture than the silver-zinc battery.

The mercury and alkaline zinc-manganese dioxide systems are inherently low rate, and therefore not suitable for release probe use.

The primary recommendation is, therefore, that the reserve magnesium perchlorate battery be considered for release probes for the BVS because of its high energy density and adaptability to sterilization. Should ground tests of flight probe batteries be a firm requirement manually activated silver-zinc batteries would be substituted.

CONCLUSIONS

To date no specific mission duration has been established for the BVS. Indications are that it will span from one week to six months, based upon preliminary science data requirements. For this duration, the obvious candidate for a power system is a radioisotope thermoelectric generator (RTG), used with a nickel-cadmium battery to supply peak loads. The extreme limits of continuous operation for the nickel-cadmium battery are 233°K to 322°K. When the RTG is used, however, a source of heat is available, which may be used to moderate the effects of low temperatures on the battery and other components of the payload. With thermal control equipment this could enable operation at the lower temperature limits shown (194.2°K). The RTG itself is able to operate at 588.6°K, which is much higher than the capability of the battery or other payload equipment. Thus, should the battery and other equipment be insulated for short-term operation in a hot atmosphere, excursions could be made into lower altitudes with the RTG in operation. Although investigation of entry conditions is not part of this contract, an investigation of the transient thermal condition during atmospheric entry will have to be made for the RTG. Adequate data on atmosphere circulation are not available to permit the design of a reliable solar-cell system.

Possible candidates for powering release probes are wind-driven generators, RTGs, and batteries. The present wide divergence between the atmospheric models would present difficulties in establishing a design for a wind turbine to drive a generator. An RTG, which weighs about a pound per watt of output, is not economical from the weight or cost viewpoints for very short missions. Batteries are well adapted to provide loads for short time periods. Use of thermal insulation and heat absorbing materials can minimize the effects of high atmospheric temperatures during the period of use.

A comparison of the different systems is shown in table IV.

TABLE IV. - POWER SYSTEM COMPARISON

System	Sterilizable	Temperature range, °C	Specific output, W-hr/lb	Development status
Batteries				
Silver-zinc	Probably	15 to 35	25 sterilized	Sterilizable batteries being developed
Nickel-cadmium	Yes	-40 to 49	10 sterilized	Sterilizable batteries available
Zinc-oxygen	Questionable	-15 to 49	90 to 100 projected	No sterilization effort
Magnesium perchlorate	Probably	-40 to 52	38 unsterilized	No sterilization effort
Ammonia	Difficult	-54 to 74	4.5 unsterilized	No sterilization effort
RTG	Yes	-273 to 135	about 1 W/lb	Space proven
Solar cells	Yes	-100 to 60	10 W/lb in sunlight at 1 AU	Space proven
Wind-driven turbine generator	Yes	-40 to 100	5 W/lb	Specific designs require development

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