i wp.

X-551-71-494

PREPRINT

NASA TM X- 65795

ALTIMETER AND RADIOMETER FOR A VENUS ORBITER MISSION

J. W. BRYAN

K. R. RICHTER

DECEMBER 1971

GSFC

GODDARD SPACE FLIGHT CENTER
GREENBELT, MARYLAND

X-551-71-494

ALTIMETER AND RADIOMETER FOR A VENUS ORBITER MISSION

J. W. Bryan

K. R. Richter

December 1971

GODDARD SPACE FLIGHT CENTER

Greenbelt, Maryland

PRECEDING PAGE BLANK NOT FILMED

ALTIMETER AND RADIOMETER FOR A VENUS ORBITER MISSION

J. W. Bryan

K. R. Richter

ABSTRACT

This paper presents the concept, constraints and capabilities of a radar altimeter type contour mapper for a Venus Orbiter Mission. The system was developed for the Goddard proposed Planetary Explorer Universal Bus concept. A system with a height precision of 30 meters over a surface area of 7200 square kilometers is achieved. Using this system and the orbit proposed in the orbiting "Bus" concept, the northern hemisphere of Venus is mapped in one Venus day. The radar receiver system, as conceived, is used in a radiometer mode to obtain a map of the diurnal and longitudinal variations of the Venus surface temperature with a resolution of 3.0 degrees Kelvin.

PRECEDING PAGE BLANK NOT FILMED

CONTENTS

ILLUSTRATIONS

TABLES

ALTIMETER AND RADIOMETER FOR A VENUS ORBITER MISSION

INTRODUCTION

The purpose of this document is to present a plan at the conceptual level for the inclusion of a radar altimeter type terrain contour mapper for a Venus Orbiter Mission. A secondary purpose is to present a conceptual design of such a radar, to examine the contraints imposed upon the design by the proposed "Bus" concept (ref. 1) and to indicate trade-offs possible in the design of such an instrument.

Although Venus is our closest planet it is still somewhat of a mystery. The reason — perpetual cloud cover. Radar with its ability to penetrate clouds and map contours is one of the few instruments which can be used to study the topography of Venus. Earth based Radar studies by the Jet Propulsion Laboratory (refs. 2, 3,) have yielded much information about Venus such as spin rate, direction of rotation and orientation of the spin axis. During these observations some gross topographic features have been observed. However, due to the vast range, the instrumentation was unable to discern much in the way of surface features.

The proposed system will map the surface contours with a height resolution of 30 meters. The height is measured above a 7200 square kilometer area at the equator and less than 1000 square kilometer area at periapsis.

PROPOSED ORBIT

The proposed orbit for the orbital mission is a 24 hour, elliptical, polar orbit with a periapsis of 400km, and an apoapsis of 67,000km.

Periapsis will occur at 45° north latitude. During the proposed 24 hour orbit the planet will rotate approximate 1.55°. Considering this rotation and the planet radius (6040 km), the subsatellite point moves 164 km per pass at the equator and zero at the north pole. If the radar is active during each periapsis pass from the equator to the north pole, the northern hemisphere will be mapped in one Venus day (approximately 243 earth days). If an orbit reduction is then accomplished along with a 180° spin axis rotation the southern hemisphere can be plotted in similar fashion.

Assuming that a mean reflecting surface exists, a radar system can be designed to operate within the orbit mission constraints. The major constraints which have been considered are:

a. The orbiter spin axis will always remain parallel to the planet spin axis.

b. The orbiter spin rate will be 15 rpm

c. Height above the planet at the planet equator is 1500km maximum.

d. Height at periapsis (45° N. Latitude) is 400km

e. Subsatellite point velocity over the planet surface varies between 9.1 km/sec at 45° N. latitude and 7.95km/sec at the N. pole and equator.

f. The angle between the normal to the planet surface at the subsatellite point and some external reference such as the sun, earth or stars, can be defined to ±2.0 degrees. This angle must be determined in a plane that is normal to the satellite orbit plane and contains the satellite and the subsatellite point.

g. The angle between the normal to the planet surface and the spacecraft spin axis in the satellite orbit plane can be determined and programmed to an accuracy of ±2.0°.

The basic function of this radar altimeter is to determine the height of the satellite above a mean surface at the subsatellite point. Due to the large illuminated surface area the height above any particular point on the surface is not measured. To establish a physical interpretation of the measurement, the height measurement is defined as the distance between the satellite and a mean of the spherical planet surface. The relation between this surface and the actual planet surface cannot be determined since the surface will consist of reflecting facets whose distribution is not known.

ANTENNA AND BEAM STEERING

An artist's concept of the Venus Orbiter showing the relative position of the altimeter antenna is shown in Figure 1. The contour mapping radar will supply a measurement of the vertical height of the spacecraft above the mean surface of the planet Venus. It will supply these measurements during each perigee pass as shown in Figure 2. The planar array antenna is conceived as being electronically despun during that portion of the satellite spin when the antenna bore site is within ±45° of the subsatellite point. This despinning action

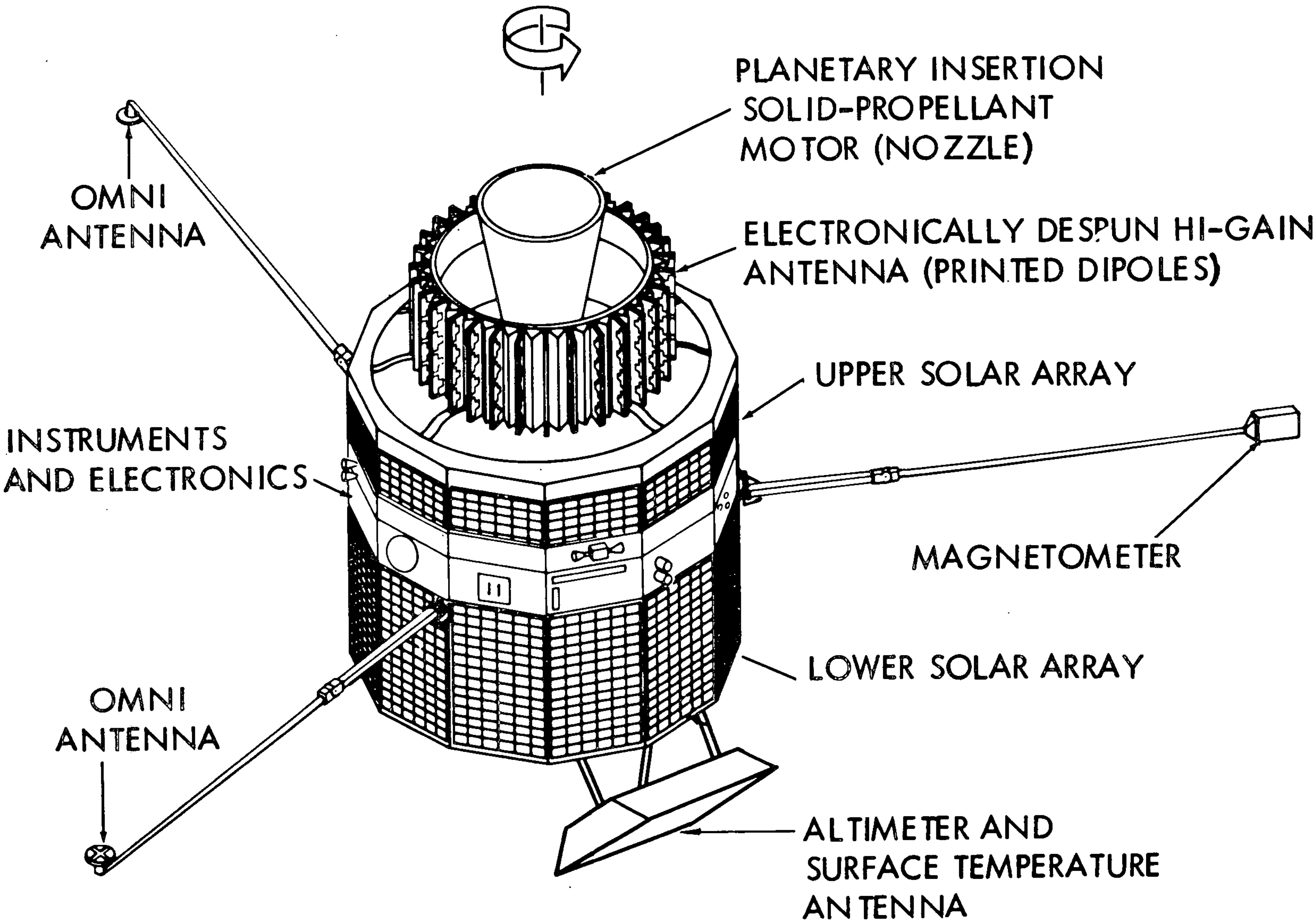

Figure 1. Venus Orbiter Configuration

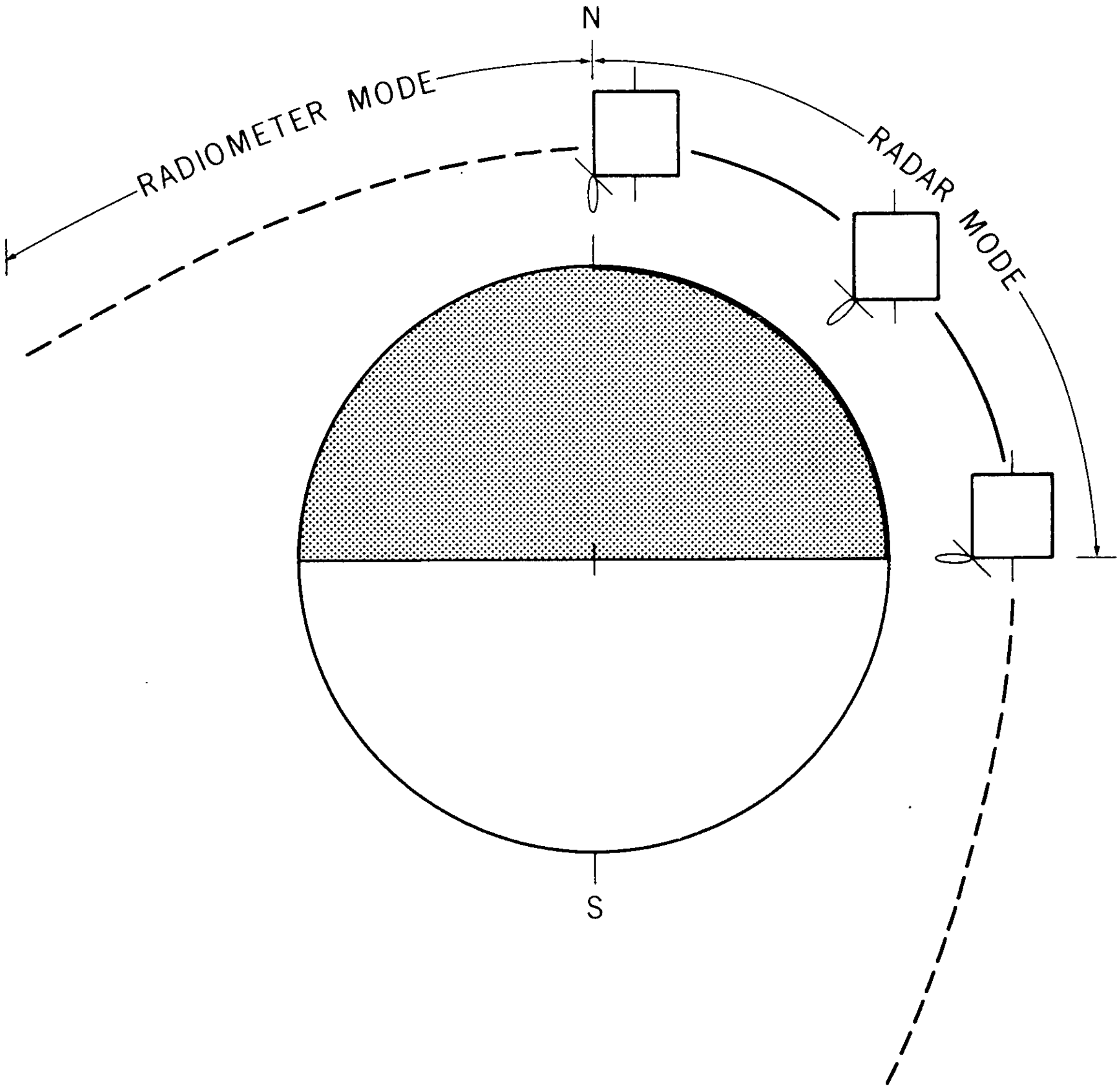

Figure 2. Venus Altimeter and Radiometer Coverage

coupled with an electronic beam steering maintains the radar beam pointing at the subsatellite point. The antenna is an 11 by 11 element phased array. This array is 75 by 75 by 15 centimeters with the back edges tapered to accommodate mounting and the spacecraft-launch vehicle adapter.

Phasing or beam steering will be in 0.25 degree steps. The steps will be synchronized with the spacecraft spin and orbit position relative to the planet such that each time the spin brings the antenna bore site within ±45° of the sub-satellite point the beam will have stepped 0.25° relative to the spin axis to keep the nadir point within the beam. This beam steering is necessary to keep the illuminating radar beam at normal incidence to the planet surface.

The 0.25° steps will require a 330 count stored in a countdown register. After passage over the north pole the register will count up the original 330. This count-up routine is not wasted and will be utilized in a radiometer mode.

The step size for equal sized steps is dictated by the velocity of the sub-satellite point at periapsis. Since the array phasing elements must be designed for this step size, it is deemed cost effective to design all steps the same size. This results in overlapping of the illuminated area at the equator and pole. This overlapping can only help in constructing a vertical contour map of the planet.

Radar beam steering or despinning in the direction orthogonal to the orbit plane requires a reference that is not a portion of the spinning spacecraft. This is actually the electronic despinning function in relation to the spacecraft 15 rpm spin. To maintain the radar beam pointing at nadir the concept is to constantly determine the angle formed by an external reference, the spacecraft and the center of the planet. The radar beam will then be maintained at this angle referred to the reference during the periapsis pass. This despin electronics will control the beam whenever the array bore site is within ±45° of the subsatellite point. Since the radar beam is approximately 4.5 degrees between the one dB points, the beam pointing accuracy must be less than ±2.0 degrees. This is within the state-of-the-art for either the reference sensor or scanning celestial attitude-determination system (SCADS). The 121 element phase array is illustrated in Figure 3 and the characteristics of this array are tabulated in Table I. The beam steering phasing elements are conceived as being housed behind the elements with control voltages being supplied from the spacecraft proper.

Table I

Partially Despun Antenna Array

Size	75 by 75 by 15 centimeters
Active Elements	121
Weight	6.8 kg
Gain (45° off boresite)	25 dB
Steering Power (d.c.)	5 watts
Beam Width (3 dB)	9 degrees
Polarization	Circular
Feed Loss	2 dB

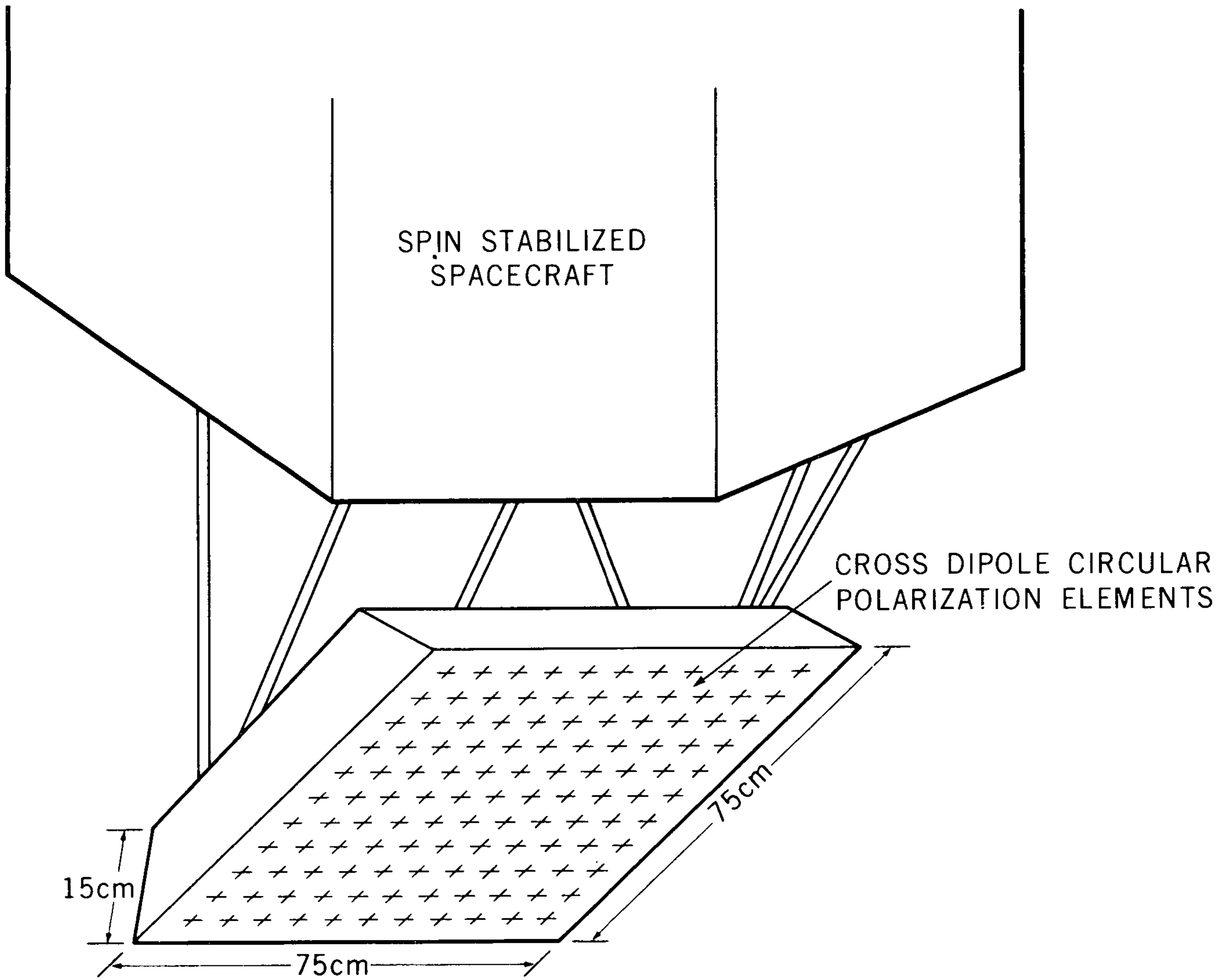

Figure 3. Altimeter Antenna Configuration

The d.c. power required to steer as well as despin the radar beam is estimated at 3.5 watts. This based upon the use of pin diode phase shifters. The countdown register is estimated to draw 0.5 watts d.c. during the active pass and during the count up in the radiometry mode.

SYSTEMS CONCEPTS

A block diagram of the altimeter system is shown in Figure 4. All timing including the transmitter modulation frequency is synthesized from a common oscillator. A 10 kilowatt solid state transmitter generates the 10 cm wavelength pulses. Transmission and reception by the same antenna is accomplished via a duplexer. The superhetrodyne receiver has a 6 dB noise figure and an IF bandwidth of 1.0 MHz. Data processing is limited to 1 second integration of the diode threshold detector output. The digitized altitude is shifted into the memory as a 16 bit word. To gain some information about the reflecting surface as well as the operation of the radar, the Automatic Gain Control (A.G.C.) voltage is included as a 5 bit word in the data storage. AGC is read at a one per second rate. The transmitter power is read and stored at a one per minute rate.

The total data storage required per perigee pass is 6400 bits. This information is transmitted to earth at a slow rate during the long swing out to apoapsis.

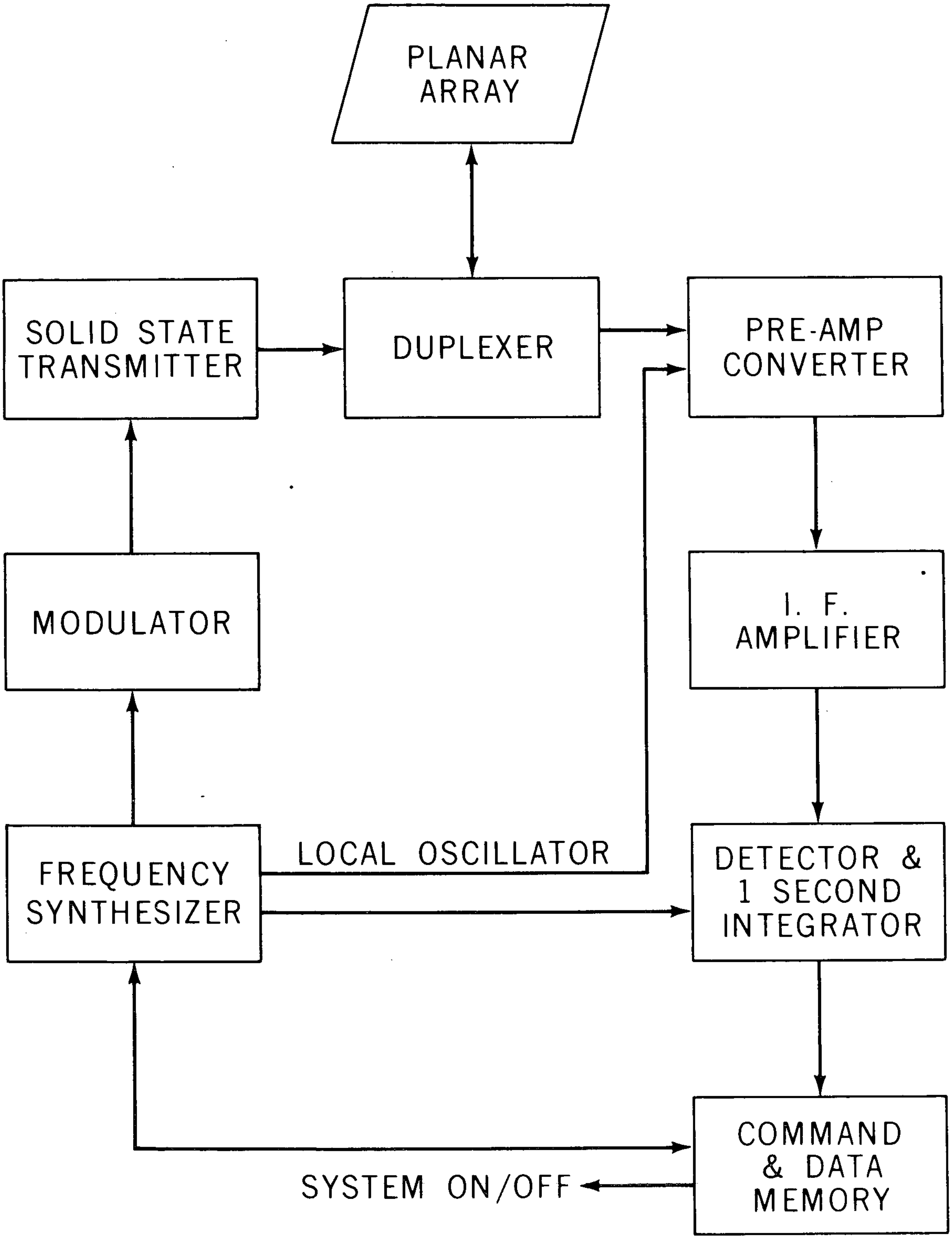

Figure 4. Block Diagram

WEIGHT AND POWER

The weight and power estimates for the contour mapping radar are given in Table II.

Table II

Weight and Power Estimates

Item	Weight	Power	Volume
Transmitter	4.99 kg	10 watts	500 cm^3
Receiver	0.9 kg	5 watts	500 cm^3
Antenna	5.9 kg	5 watts	0.084 m^3
Memory	0.9 kg	1 watt	60 cm^3
Hot Load	0.9 kg	1 milliwatt	40 cm^3
Miscellaneous (cable etc.)	1.8 kg	NA	distributed
Totals	15.4 kg	21 watts	0.085 m^3

These values are based upon the development of a solid state 10 kilowatt peak power transmitter and the continued development of microwave strip line subsystems. The inclusion of the radiometry mode increases the altimeter system weight by the 0.9 kilograms required for the hot load calibrator. Since the radar transmitter is off during radiometer measurements the total d.c. power is reduced by approximately 10 watts during temperature measurements.

RADAR DESIGN

The per pulse signal to noise ratio is given by

$$S/N = \frac{P_t \; G^2 \; \lambda^2 \; \sigma}{(4\pi)^3 \; H^4 \; KT \cdot B \cdot F_n \cdot F_{at}} \quad (1)$$

where

S/N	Signal to noise ratio
P_t	Transmitter Power
G	Antenna Gain
λ	Wave Length
σ	Radar Cross Section
H	Range (Altitude above surface)
K	Boltzmann's constant
T	Receiver Temperature
B	Receiver Bandwidth
F_n	Receiver Noise Figure
F_{at}	Atmospheric Attenuation

The radar system considered is pulse width limited where the radar cross section is (ref. 4):

$$\sigma = \sigma^\circ \; \pi c H \tau \quad (2)$$

where

σ°	Surface Reflectivity per Unit Area
c	Propagation Velocity
τ	Pulse Duration Time

Furthermore, the receiver bandwidth is chosen as the reciprocal value of the pulse duration time (τ).

Rewriting equation (1) one obtains:

$$S/N = \frac{P_t\ G^2\ \lambda^2\ \tau^2\ c\ \sigma^\circ}{64\pi^2\ H^3\ KT\ F_n\ F_{at}} \tag{3}$$

The first important decision which has to be made concerns the wavelength at which the radar should be operated. By inspection of equation (3) one sees that the signal to noise ratio depends on the wavelength only in terms of $\frac{G^2\ \lambda^2}{F_{at}}$. The gain (G) and the effective area (A) of an antenna are related by

$$G = \frac{4\pi}{\lambda^2}\ A. \tag{4}$$

where A is proportional to the geometrical dimensions of the antenna. The physical size of the antenna is limited by the available space and spacecraft dynamics. The gain of a fixed aperture antenna may be increased by the use of shorter wavelengths.

For wavelengths in the cm-region, the attenuation in a neutral atmosphere increases with a factor exp $\frac{1}{\lambda^2}$ for decreasing wavelength (ref. 5). The two way attenuation through the entire atmosphere is given by

$$F_{at} = \exp\ (M/\lambda^2) \tag{5}$$

where M is a constant value which depends on the composition and on the total height of the atmosphere. Then the wavelength dependent term of the signal to noise ratio may be written as:

$$f(\lambda) = \left(\frac{4\pi}{\lambda}\right)^2\ A\ \exp\ (-M/\lambda^2). \tag{6}$$

This function shows a maximum value at

$$\lambda = \sqrt{M} \tag{7}$$

Using the model for the atmosphere of Venus developed at Goddard Space Flight Center (ref. 6) M is calculated as 36.84 which yields an optimum wavelength of 6 cm (frequency (f) = 5 GHz). However, the resulting value for the attenuation in the atmosphere may be low because no propagation experiment is available in the region of cm-wavelength in the lower atmosphere (ref. 7, 8). Therefore 3 GHz (λ = 10 cm) has been selected as operating frequency.

In Table III the transmitter and receiver characteristics are given. These are required to obtain a per pulse signal to noise ratio of 5 dB for the worst case, (1500 km altitude). At the perigee of the orbit (altitude 400 km) the per pulse signal to noise ratio is increased by 17 dB. This value of 17 dB has to be considered as the minimum dynamic range of the receiver. An improvement of signal to noise ratio by approximately 10 dB is achieved by integrating 100 pulses (integration time = 1 sec).

The calculation of the signal to noise ratio is based on a reflectivity per unit area of $\sigma^\circ = 0.08$ which is much lower than the reflectivity of 0.152 obtained by earth based radar measurements.

It must be pointed out that for the estimation of the signal to noise ratio rather pessimistic assumptions on the properties of the atmosphere and the surface reflectivity have been made providing a security margin for the mission.

Table III

System Characteristics

Transmitter	
Peak Power	10kw
Frequency	3 GHz
Pulse Length	1 μsec
Pulse Repetition Frequency	100 Hz
Duty Cycle	10^{-4}
Average Radiated Power	1 W
D.C. Power	10 W
Receiver	
Noise Figure	6 dB
Bandwidth	1 MHz
Detection	Noncoherent
Threshold detector sensitivity	+5 dB
Integration time	1 sec
Clock Frequency	8 MHz
Sampling Gate	1 sec
D.C. Power	5 W

This radar provides a measure of the average altitude of the spacecraft above the planet surface.

The altitude error measurement due to the clock interval ($\Delta t = 1.25 \cdot 10^{-7}$ sec) is:

$$\Delta R = \frac{c}{2} \Delta t \approx 19 \text{ meters} \qquad (8)$$

The error due to the thermal noise in the receiver is smaller than 19m for the worst case (H = 1500km). The total altitude error is less than 30m from the equator to the north pole.

RADIOMETER MODE

The radiometer mode will be activated during that portion of the orbit covering the northern hemisphere of the planet as shown in Figure 2. The electronic despin which maintains the antenna beam in the orbit plane is disabled, the radar transmitter is commanded "OFF" and the radiometer hot load is activated as the spacecraft passes over the north pole. The electronic beam steering which maintains the antenna beam pointing normal to the planet surface is reverted to the "count up" mode as explained on page 6. The antenna beam will scan the surface at right angles to the orbit plane.

The r.f. noise temperature of the planet will be integrated over the one-fourth a spacecraft revolution as the antenna beam sweeps across the planet. At the spin rate of 15 rpm this results in an integration time of one second. The calibration temperatures will be recorded during those portions of the spacecraft spin shown in Figure 5. Each integrated temperature will be digitized into a 9 bit word resulting in a quantization accuracy of less than 2.0°K.

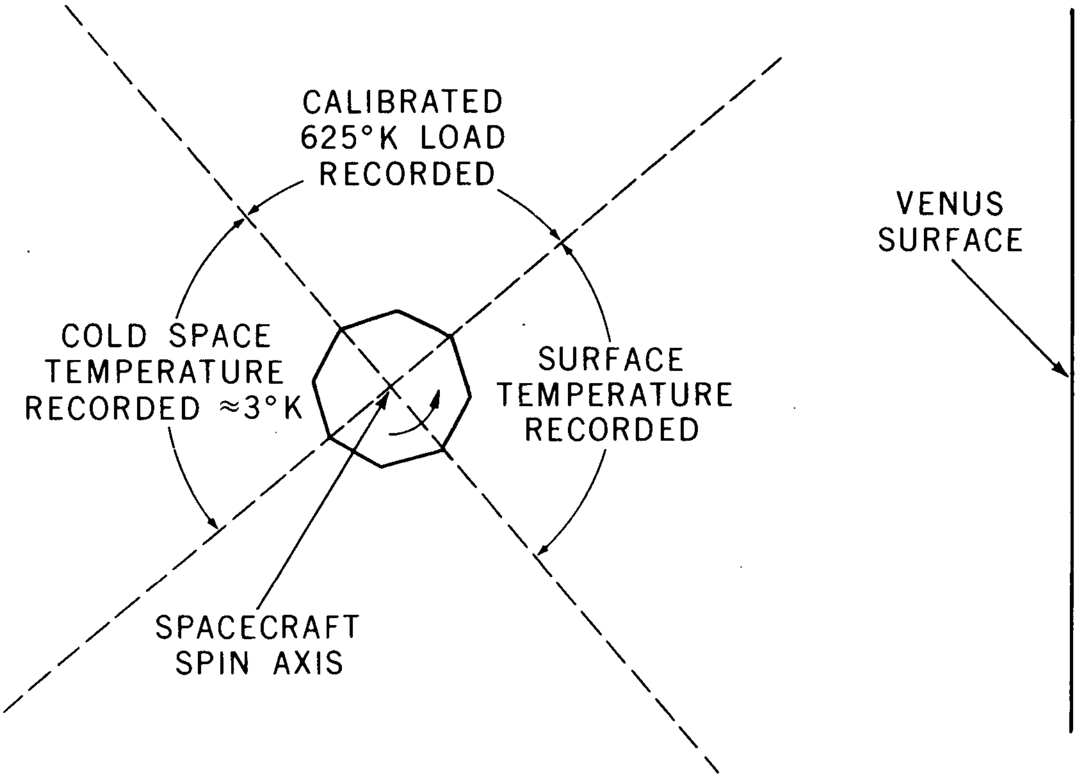

Figure 5. Radiometer Cycle as a Function of Spacecraft Spin

The radiometer temperature resolution is defined as:

$$\Delta T_{rms} = \frac{2\,(T_s + T_p)}{\sqrt{B_n \times t_{int}}} \tag{9}$$

where

T_s is the receiving system temperature (°K)

T_p is the planet r.f. temperature (°K)

B_n is the receiving system noise bandwidth (Hz)

t_{int} is the measurement integration time (seconds)

For this radiometer system using equation 9

$$\Delta T_{rms} = \frac{2\,(865 + 625)}{\sqrt{1.2 \times 10^6}} = 2.72^\circ K$$

where the expected planet temperature has been reduced by the 10 cm microwave attenuation of 0.8 dB through the planet atmosphere.

REFERENCES

1. "Planetary Explorer Phase A Report and Universal Bus Description," GSFC, May 1971.

2. Muhleman, D. O., "Radar Scattering from Venus and the Moon," Astron. J., Vol. 69, No. 1, Feb. 1964.

3. Carpenter, R. L., "Study of Venus by G. W. Radar - JPL 1964 Results," Report No. 32-963, Dec. 1965.

4. Bryan, J. W., "Satellite Altimeter," GSFC Report X-551-70-248, June 1970.

5. Muhleman, D. O., "Interferrometric - Investigations of the Atmosphere of Venus," Radio Sc. 5 #2, 1970, pp 355-362.

6. Ainsworth, J. E., "Comprehensive Study of Venus by Means of A Low Cost Entry-Probe and Orbiter Mission—Series," GSFC X-625-70-203, June 1970.

7. Fjeldbo, G., Kliore, A. J., Eshleman, R., "The Neutral Atmosphere of Venus as Studied with the Mariner V," Astron. J. 76, 1970, pp 132-140

8. Richter, K. R., "Enhanced Microwave Absorption in the Lower Atmosphere of Venus," GSFC X-401-71-282, June 1971.