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PROPOSALS FOR SATELLITE PROGRAM

VOLUME III

SIGNAL CORPS PROGRAM PROPOSALS TO
PHASE I AND PHASE II

9 SEPTEMBER 1955

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SIGNAL CORPS ENGINEERING LABORATORIES

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PROPOSALS FOR SATELLITE PROGRAM

Volume III

Signal Corps Program Proposals to
Phase I and Phase II

9 SEPTEMBER 1955

Regrading Date Cannot Be Predetermined

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PROPOSALS TO SATELLITE PROGRAM

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PROPOSALS TO SATELLITE PROGRAM

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

1. Tracking Systems

a. Optical Location of an Artificial Satellite

[redacted]

[redacted]

II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I

1. Tracking Systems

a. Optical Location of an Artificial Satellite

Prepared by: Physical Sciences Division

SYNOPSIS: Considerations are presented regarding the engineering parameters required to produce optical instrumentation in the satellite program. The instrumentation is designed to fulfill all tasks beginning with the search and determination that a satellite exists, the determination of the precise orbit and changes in orbit in the course of its life, and the detailed study of the satellite itself. This proposal is complicated by a great number of unknown parameters which exist in preliminary planning and the dependence upon atmospheric transmission as it effects optical instrumentation. The parameters which effect location are presented in tables covering both a 20" diameter sphere as well as larger objects. It is shown that in order to make all locations positive, the object should be larger than the 20" sphere.

1. INTRODUCTION:

Optical tracking must be considered in the instrumentation because of these advantages:

(1) It is a passive system.
(2) It is a noninterfering system.
(3) Its location accuracy is high.
(4) The techniques developed could be used for the location and identification of other satellites--friendly or not.
(5) Secondary or multiple stations are relatively inexpensive especially with regard to the use of power.

a. The experience gained through the developments of astronomy and through the much more recent guided missile test range instrumentation indicates that optical location during the day is very difficult even under the most favorable optical environment which exists at White Sands Proving Grounds. The difficulties are operational and increase with the need of obtaining location data quickly for further position

[redacted]

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1. INTRODUCTION. a. (Contd)

information. Optical location during twilight or dawn is considerably more favorable. Under such conditions the object can be seen and photographed with relatively simple equipment. The problems introduced by the rapid change of the sky brightness and the angular velocity of the object introduce operational problems. The use of intrinsic illumination in the satellite, such as a pulsed arc, is a possibility for location at night. The source would require weight in the satellite.

b. If the object is increased in size, optical location becomes much simpler.

c. In order to achieve a good optical location program, it is proposed that an orbit be chosen which will enable the use of existing astronomical facilities. In addition, it is recommended that optical instrumentation developed in connection with Army problems be installed in suitable positions depending on the orbit chosen. This equipment should have aided tracking and servo controls capable of operating in an electronic location network.

2. ANALOGY TO ASTRONOMY:

In considering the instrumentation which might be used for the optical location of an Artificial Satellite, one immediately is attracted by the similarity of the artificial satellite with natural stars, satellites, or meteors. The analogy with astronomy denotes the possibility of using astronomical type instruments for the precise location and especially for the first requirement to determine that a satellite exists. There are, however, some striking differences in the analogy which create difficulties in the location system.

[redacted]

2. ANALOGY TO ASTRONOMY: (Contd)

a. First and foremost the artificial satellite is eclipsed from dusk until dawn and there is never a time when the sun-illuminated satellite can be viewed against a night background.

b. The apparent motion through the sky is rapid and becomes analogous to the location of meteor paths.

c. While the artificial satellite is resolvable with large astronomical type lenses and has a unit brightness some twelve times greater than the moon, the total illumination is comparable to those stars which vary between the third and tenth magnitude. Or, carrying the analogy further, it is very similar to the larger moons of jupiter in angular size and brightness.

3. OPTICAL TRACKING INSTRUMENTATION:

The study of the launching of the satellite, determining that the satellite exists, and then determining its precise orbit requires four types of instrumentation.

a. Tracking equipment for a period during burnout of the final stage long enough to predict the subsequent orbit. This instrumentation will not be considered further in this study.

b. Wide angle search or intercept devices having either airborne mobility or fixed locations in sufficient quantity to cover the probable course. The difficulties inherent in optical wide angle search are so great that they become almost insurmountable. The possibility of location is only attractive during an hour and a half at twilight. Optical search can only be considered secondary and reliance should be placed on other devices such as the Long Range Radar or Radive as indicated in a following chapter.

3. OPTICAL TRACKING INSTRUMENTATION. (Contd)

c. Once an orbit can be predicted so that the location of the satellite can be predicted to approximately one degree of arc and 15 seconds of time, special narrow angle optical trackers will be able to track the satellite.

d. With accurate determination of location, high resolving power and special purpose astronomical instrumentation will be able to study and recognize the satellite in its orbit.

4. PROPOSED OPTICAL INSTRUMENTATION SYSTEM:

An optical instrumentation system is proposed for the detection and location of the satellite. This system will be compatible with other electronic location communication and data gathering systems. The optical tracking unit should be physically located with other radio electronic location devices as they are mutually supplementary
4 and would be most effective. It is proposed to furnish optical equipment in accordance with the data tabulated in the Appendix 1a. This data has been checked as carefully as possible against field experience and literature and represents optical possibilities in a practical way. That is, by careful engineering and under good "seeing" conditions, it is possible to better the listed performance. Under poor conditions the performance will not be achieved. In order to both be compatible with the electronic location system and to fulfill the stated requirements for search, detection and investigation of the missile, it is proposed that the following optical instrumentation be provided:

[redacted]

4. PROPOSED OPTICAL INSTRUMENTATION SYSTEM. (Contd)

a. Wide Angle Intercept Equipment:

(1) Description and Function: This equipment will be used to "image" the area of the sky in which the satellite is expected. It will consist of photographic cameras or photoconductive-electronic detectors which will provide azimuth and elevation data of the satellite. Three of these equipments will be used as a set on a base line of approximately 10 miles to enable location by triangulation and to better insure results against the failure of one unit.

(2) Locations: It is proposed that a minimum of 8 sets be utilized in order to insure that at least a fix is made each satellite revolution. These locations should be spaced along the earth's surface at the latitudes in which morning and even twilight is expected in the satellite's orbit. It is most likely that mixed north and south locations will give best coverage.

(3) Costs:

(a) Initial Equipment and Installation:

$80,000 per set plus $10,000 for installation

Total for 8 sets--$720,000

(b) Technical Operating Costs: Based on 1 set:

Total--8 men.

1. Using enlisted personnel--$15 per day/set

2. Using contracted personnel--$80 per day/set

b. SCEL Precision Optical Tracking Equipment.

(1) Description: This equipment will be used to photograph the satellite and provide its azimuth and elevation data. It will

[redacted]

[redacted]

4. PROPOSED OPTICAL INSTRUMENTATION SYSTEM. b. SCEL Precision Optical Tracking Equipment. (1) Description (Contd)

consist of either 160 inch or 320 inch focal length optics with visual observation and photographic cine recording. Two of the equipments will be used as a set on a 10 mile base line interconnected with a radio location set to enable location by triangulation.

(2) Locations: Eight sets located as indicated under the Wide Angle Intercept Equipment. Inasmuch as this equipment is longer range than the preceding equipment and is compatible with the data link satisfactory coverage may be obtained with four sets.

(3) Costs:

(a) Initial Equipment and Installation:

$225,000 per set (1st four), $200,000 per set thereafter, plus $10,000 for installation; total for 8 sets--$1,780,000
4 sets--$1,000,000

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(b) Operating Costs: Based on 4 men per set;

Total--32 men or 16

1. Using military personnel--$70 per day/set

2. Using contracted personnel--$400 per day/set

c. Amateur Observers: The services of the amateur should also be encouraged for they can conceivably provide an excellent source of "early warning" information which would be of great help especially during the initial stages of this activity. This service should also be available at no cost.

d. Astronomical Telescopes: The problem of resolving the 20" sphere is difficult; once the orbit is known precisely it is proposed this exact data be furnished suitable astronomical observatories and

[redacted]

4. PROPOSED OPTICAL INSTRUMENTATION SYSTEM: d. Astronomical Telescopes (Contd)

their cooperation be solicited. This data should be available at no cost. However, once the orbit is settled and the definite firing schedule is determined, arrangements should be made to insure that at least two positions are available at the expected orbit.

[redacted]

APPENDIX Ia

1. LUMINOSITY OF THE SATELLITE:

The luminosity calculations are based on the initial assumptions that the satellite is illuminated only by a combination of direct sunlight, moonlight, and earthlight; that the satellite is a perfect diffuse reflector of unity reflectance and 1/2 meter diameter; and that the orbit of the satellite is 200 miles above the earth's sea level surface. Skylight makes a negligible contribution to the illumination of the satellite. The satellite illumination from direct sunlight is Es = 14.5 lumens/cm^2. From an extended source the relation of the illumination at a distance D due to a perfectly diffuse spherical source of radius a, reflectance ρ, and irradiation L is

$$E = \rho L \frac{a^2}{a^2 + D^2} \tau$$

where τ = 0.90 is the atmospheric transmission.

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a. The above equation and the following data were used to calculate the satellite illumination Em from moonlight and the satellite illumination E_e from earthlight

ρ_m = 0.07 (the moon's albedo)

L_m = 14.5 lumens/cm^2

a_m = mean radius of the moon = 1.74×10^8

D = 3.841×10^{10} cm

τ = 1

Em = 3.86×10^{-5} lumens/cm^2

For the earth

ρ_e = 0.20 (the earth's albedo)

[redacted]

[redacted]

1. LUMINOSITY OF THE SATELLITE a. (CONT'D)

$L_e = 14.5$ lumens/cm^2 Due to atmospheric absorption the irradiation of the earth is $L_o = 12.91$ lumens/cm^2

A_e = mean radius of earth $= 6.37 \times 10^8$cm

The entire diameter of the earth is not effective and A_e has to be corrected. The corrected or effective earth radius is $A_o = 1.95 \times 10^8$cm for D = 200 miles. The corrected distance D_o for an equivalent disk or spherical source of corrected radius A_o is $D_o = 0.627 \times 10^8$ cm.

$E = 2.08$ lumens/cm^2

The total maximum satellite illumination E_t is as follows:

$$E_t = E_s + E_m + E_e = 14.5 + 3.86 \times 10^{-6} + 2.08$$

$$E_t = 16.58 \text{ lumens/cm}^2$$

(Note that the moonlight contribution E_m is negligible.)

Since the reflectance of the satellite is unity, the maximum luminosity of the 1/2 meter diameter satellite is 16.58 lumens/cm^2.

b. The visibility of the satellite will be effected by the region of the infrared or optical spectrum in which the satellite is viewed. The effect of scattering by dust particles, water vapor, and air molecules can be reduced by using only the infrared region. This also reduces the background illumination and improves the detection inasmuch as the visibility of the satellite depends upon the contrast between its brightness and the sky brightness. However, 42% of the natural sunlight lies in the visible region, 38% lies in what might be called the near infrared region, and only 11% in the intermediate infrared. The lack of radiation in the intermediate infrared region in addition to the diminishing sensitivity of infrared detectors militates against the use of intermediate infrared devices.

[redacted]

[redacted]

1. LUMINOSITY OF THE SATELLITE. b. (Contd)

There is evidence that under most conditions of slight haze photography in the near infrared region to 0.86 micron reduces background illumination and therefore makes it possible to improve daytime minimum photographability by 1 magnitude. Extending further to 1.0 micron in the infrared has produced extraordinary photographs but such photography in the satellite case is considered impractical because of the low sensitivity and the difficulty of handling the film. There are possibilities that infrared detection systems would increase the detectability to an extent not fully accessable at the present time.

In the optical detection of the satellite the background illumination by the sun and sky exerts a great influence. The illumination changes by a factor of 8 orders of magnitude from daylight condition to nighttime. This corresponds to a change of twenty astronomical magnitudes. The seriousness of this is recognized when one considers that stars of the 20th magnitude are barely detectable at night using the best techniques and tools of the astronomer's art. The daytime sky background is caused either by scatter of visible light by air molecules which gives the blue sky or by the scatter of haze particles, water drops or clouds which gives the characteristic white or grey color. These scatter effects are rapidly minimized as one views the increasingly longer infrared wavelengths. Beyond about 3.5 microns the daytime sky is practically as dark as the nighttime sky is in the visible region. The change in the illumination during the summer solstice at the equator and north 30th latitude is shown in Figure 1 as follows:

[redacted]

ILLUMINATION ON THE EARTH AT SUMMER SOLSTICE

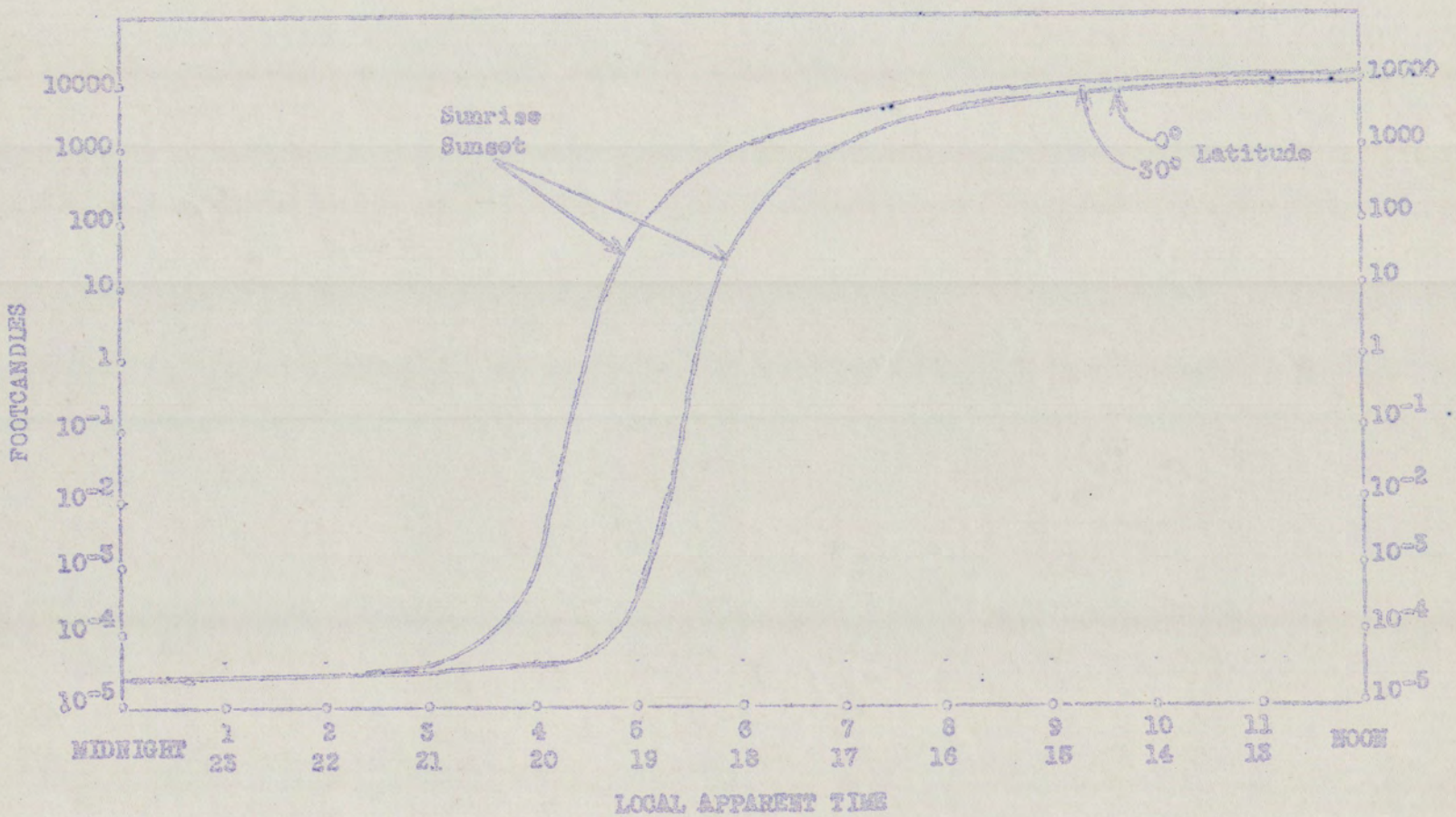

Figure 1.

From Dept. of Navy
BuShips

2. BACKGROUND: (Contd)

The time of the setting of the sun is indicates by arrows in the figure. From that time until the nighttime condition the sky brightness drops rapidly as indicated and presents an increasingly favorable seeing condition. During this time the background decreases. However the luminosity of the satellite diminishes due to attenuation of an increasingly greater atmospheric path through which the sun rays must travel. At nighttime the sky brightness is so feeble that while there is no sun illumination it is possible the satellite could be observed under a condition where a visible source is placed on the satellite. The candlepower required for this illumination can be determined by Figure 2 which shows

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[redacted]

EQUIVALENT POINT SOURCE MAGNITUDES

Showing the intensity at 200 and 800 miles as a function of the equivalent astronomical magnitude.

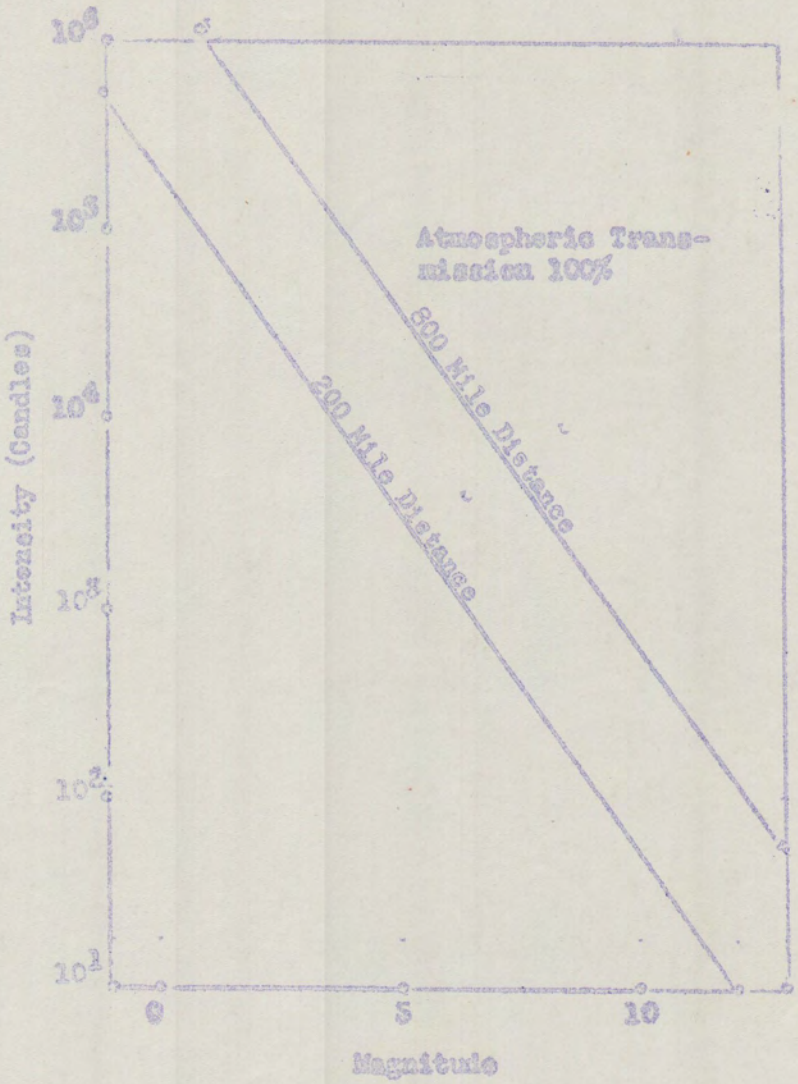

Figure 2

[redacted]

[redacted]

2. BACKGROUND: (CONT'D)

the intensity at the satellite as a function of the astronomical magnitude produced by the illumination. The curve indicates this relationship at two ranges - - 200 miles and 800 miles. The visibility of the satellite in the sky can be increased by a filter. The effect of short-wave scattering can be reduced by either viewing of photographing through a red filter against a blue sky or using a Polarized filter oriented in such a manner as to suppress the polarized components of the scattered background. It is also possible to reduce the background effects by what is termed space filtering where the characteristic signal from the target is caused to be at a frequency different from the characteristic frequency of other objects in the field of viewing. This characteristic frequency is determined by fourier components of a signal chopped by specially chosen chopper configurations. Tracking stars during the daytime is possible by using this space filter.

3. TYPE OF MISSILE:

The optical location and tracking problem can be aided immeasurably by changing the missile from that hypothesized as a 20" sphere. Increasing the size of the sphere will improve the location possibilities. This could be done by having a balloon structure within the sphere which upon attaining the satellite status can be broken free and inflated to an appropriate size. Tables shown later indicate the improvement of the apparent magnitude as the sphere is increased from 20" diameter to 1 meter diameter then to 10 meter diameter. The 10 meter diameter becomes extremely easy to see and would be of considerable interest visually.

[redacted]

3. TYPE OF MISSILE (CONT'D)

a. The use of a diffuse reflecting spherical surface would be much superior to a specular reflecting surface inasmuch as the only gain in brightness in the specular reflecting case is when the sphere can be resolved. However, the disk of the sphere would not be seen but only light from images such as the sun. When the sphere can be resolved photographing the disk of the diffuse sphere itself presents no problem.

b. The gain in detectability by painting the object other than very high diffuse white with titanium dioxide would only be possible under certain conditions of sky background depending upon the specular sensitivity of the detecting element. The gain could never be more than a
15 fraction of an astronomical magnitude. This includes the case of painting the satellite with fluorescent or phosphorescent surfaces. However, marking should be made on the satellite so that when the satellite is resolved it will be possible to study the motion of the surface or any possible deterioration due to erosion by cosmic dust.

c. The size of the object could be increased by what might be termed the "magician's flower" arrangement; that is, the sphere could be packed with metallic chaff which would bloom forth after the sphere is in its orbit. This would produce a large configuration which could be designed to improve both the radar cross section and the optical cross section. This would not substantially affect certain portions of the drag measurements.

d. Illumination of the satellite by a directed searchlight from the ground is a technique whereby detectivity may be obtained at night. As the illumination would be insufficient for making a diffuse sphere

3. TYPE OF MISSILE d. (CONT'D)

visible this requires that the satellite incorporate some arrangement of retrodirective reflectors. It is concluded that in order to conserve weight these reflectors be of small size covering the surface of the sphere and that they should be as nearly perfect as reasonable fabrication techniques permit. With good atmospheric conditions a searchlight of 10^7 beam candle power would make the 200 mile distant satellite equivalent to a 0 magnitude star with a perfect retrodirective reflector. Such a searchlight would have a beam width of 7°. In determining the detectivity from the tables it must be considered that the object is being observed against a background of scattered light from the beam itself. This scattering depends so much on atmospheric conditions that we can consider that we have either the twilight case or under bad scattering conditions, the daylight case. Back scattering illumination can be eliminated by the use of extremely short pulses of light and a gated electronic detector. There are many engineering problems which much be investigated before an adequate proposal can be made.

4. DETECTION DEVICES:

In view of the astronomical background associated with the detection of the satellite, the astronomical magnitude of the object can be easily determined and compared with other backgrounds reduced to astronomical magnitudes. In the following tables this is performed visually and photographically. Further extension of the tables to other pick-up devices such as photoelectric cells can be accomplished by considering the visual case as also the limit to photoelectric detection. In order to achieve location at a low astronomical magnitude, th

4. DETECTION DEVICES; (CONT'D)

parameter must be examined very closely. This is complicated by the orbit of the satellite which has such velocity that it only takes 10.7 minutes for the satellite to go from horizon to horizon as indicated in Fig. 3. Much of this time the satellite is at low elevation and as it rises in elevation the angular acceleration becomes large which complicates the tracking problem. Table I lists the equivalent of astronomical magnitudes of the satellite considering the satellite to be either 1/2 meter, 1 meter or 10 meter object at conditions which have been termed "best case" and "worst case". The "best case" and the "worst case" have been determined by calculating the illumination of the object by direct sunlight and earthlight, the phase change of the satellite, and its distance from the observer. The difference in the tables between daylight and twilight is due largely to the phase changes of the earthlight and sunlight.

[redacted]

PLOT OF
I. ZENITH DISTANCE, II. ANGULAR VELOCITY,
III. CHANGE OF BRIGHTNESS vs TIME

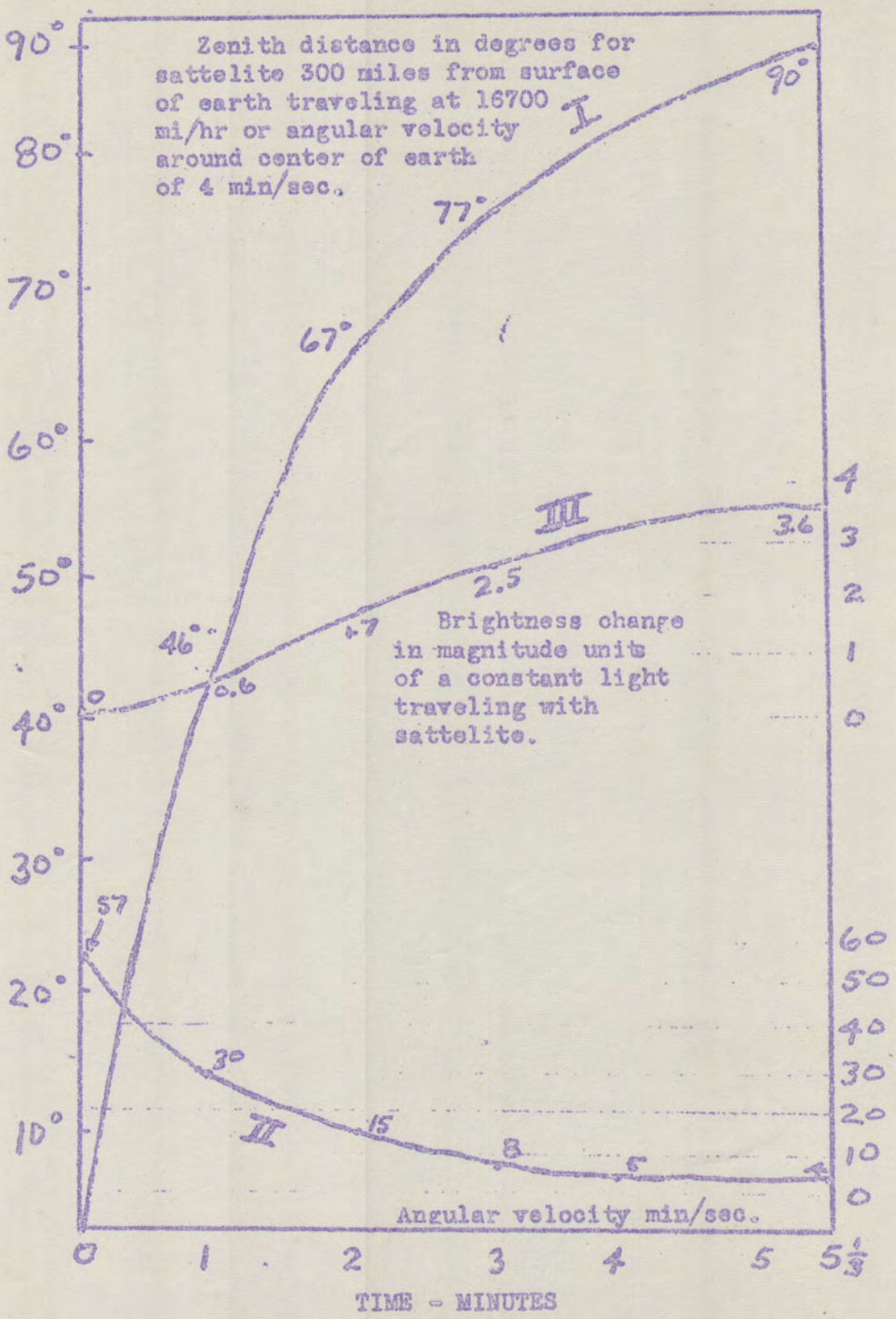

Figure 3

[redacted]

4. DETECTION DEVICES: (Contd)

TABLE I

APPROXIMATE SATELLITE MAGNITUDES DURING DAYLIGHT

Satellite Diameter (meters)	Magnitudes Best case	Worst case
½	3	10
1	2	8
10	-3	3

APPROXIMATE SATELLITE MAGNITUDES DURING TWILIGHT

Satellite Diameter (meters)	Magnitudes Best case	Worst case
½	5	10
1	3	8
10	-2	3

The above magnitudes are a function of phase, distance and earthshine.

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a. Visual Location. Table II shows the limiting magnitude which can be detected, namely, at 10, 20 and 40 power magnification.

[redacted]

4. DETECTION DEVICES: a. Visual Location. (Contd)

TABLE II

Limiting visual magnitudes during twilight and daylight hours. Three telescopic powers are considered. A polarizer set to maximum darkening is used. Data is determined for an observational elevation of ten thousand feet above sea level. To make correction to sea level subtract 0.5 magnitudes.

Daylight.

Satellite 30° above horizon, looking away from sun.

10X -- 0 mag; 20X -- 1 mag; 40X -- 2 mag.

Pre-twilight.

Satellite 30° above horizon, sun within 5° of horizon.

10X -- 2.5 mag; 20X -- 4 mag; 40X -- 5.5 mag.

Twilight.

Satellite above 15°, sun 15° below horizon.

10X -- 10 mag; 20X -- 11 mag; 40X -- 13 mag.

As indicated in Tables I and II, the detection will depend upon the position of the sun. The effects of atmospheric absorption at low elevations have not been considered in the tables. The tables have to be considered as true only under fairly favorable seeing conditions--in general, at elevations below 5° the atmospheric absorption would deteriorate the results. The detectability under two limiting twilight conditions is given. The first is the sun on the horizon, and the second the sun 15° below the horizon. The object travels so quickly that the detectability does not change much with time during a particular passage. That is, the detectability becomes a function of the time at which a satellite arrives

[redacted]

4. DETECTION DEVICES; a. Visual Location (CONT'D)

at the particular longtitude on the earth and remains constant except as affected by the approach of the course of the sun.

b. Photographability: The computation of the photographability of the object is complicated by the assessment of the "jitter" of the tracking over a short period of time. Inasmuch as the angular acceleration of the object at a ground station is appreciable, tracking cannot be done to any extended time by clockwork. The tracking will have to be controlled by a servo or an analog computer which definitely limits the trackability. In any event, the object cannot be tracked more than the 9 minute period when it is above the horizon. Table III indicates the limit of photography for various focal length and F numbers which might be considered. The limiting photographic magnitude can be calculated by assuming that the illumination due to the sky imaged by a lens

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$$E = \frac{\pi}{4} \frac{B_{SKY}}{(f/no.)^2}$$

is equal in the limiting case to the illumination of a point source imaged into a limiting image area A_1

$$E = \pi B_{STAR} \frac{A_{LENS}}{A_1}$$

Then B star converted to astronomical magnitudes from equating the two equations is

$$B_{STAR} = \frac{B_{SKY}}{A f \ell^2} \cdot D_i^2$$

where B_{sky} is given in candles/meter^2 and B_{star} is in lumens/meter^2. Results for various lenses are shown in Table IV.

[redacted]

. DETECTION DEVICES: b. Photographability. (Contd)

TABLE III

LIMIT OF PHOTOGRAPHY.

Optics	Focal length inches	Focal Ratio	50 μ image arc of sky (seconds)	Theoretical Resolved size μ	Image size 20" object at 400 miles μ
Super Schmidt	10	.85	40	0.6	0.2
	25	4.5	16	3.	0.5
	50	12.	8	8.	1.
Palomar Schmidt	121	2.5	3	1.7	2.5
SCEL Tracker	160	14.	2	10.	3.4
SCEL Tracker (modified)	320	28.	1	20.	8.8

TABLE IV

LIMITING PHOTOGRAPHABLE MAGNITUDES

A. During Daylight

Focal length (inches)	Image size 20 μ	Image size 50 μ	Exposure time seconds 50μ *
10	-2	-4	3/1000
25	0	-2	1/500
50	2	0	1/200
160	4	2	1/100
320	6	4	1/16

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* Exposure calculated for ASA 200 film speed, to give exposure density 0.6 above fog density. A 50 micron diameter spot was assumed to receive all the light reaching the lens

B. Sun on Horizon, Satellite in Opposite Hemisphere.

Focal length (in.)	Limiting Magnitude	Exposure Time (sec.)
10	3	1/50
25	4	1/12
50	6	1
121	8	1/8
160	9	6
320	(9)	(6)

[redacted]

[redacted]

4. DETECTION DEVICES: b. Photographability: Table III (Contd)

c. SUN 15° BELOW HORIZON, SATELLITE IN OPPOSITE HEMISPHERE.

Focal length (in.)	Limiting Magnitude	Exposure Time (sec.)
10	10	10
25	3	10
50	8	10
121	13	10
160	10	10
320	10	10

If we assume that the image size is limited to 50 microns by seeing conditions, photographic "bloom", and instrumental "jitter" and further that the tracking deviation from the 50 micron image is controlled such that we can get a 10 second exposure, then Table III indicates the photographic ability of the lenses under the two twilight conditions and during daylight.

23 5. RECOMMENDED OPTICAL INSTRUMENTATION.

A study of the preceeding tables indicates the direction in which one must move in order to provide suitable optical conditions for locating and tracking the satellite. The first conclusion is that a tremendous optical gain will be made in enlarging the object to at least one meter or more. That is, the limiting cases are such that except under the most favorable conditions the 20" object will not be detected. The second conclusion is that effort should be made to provide radio transmissions from the satellite in order to supplement the optical track. The third conclusion is that optical tracking must be accomplished in such a manner as to stabilize the image for as long a time as possible. These conditions are discussed as follows:

[redacted]

[redacted]

5. RECOMMENDED OPTICAL INSTRUMENTATION: (Contd)

a. The design of the satellite. The first improvement which must be considered is making the satellite larger either by launching a larger sphere or expanding the sphere after it has obtained its orbit. A more improved object would be obtained by leaving the final propulsion stage and sphere as one satellite.

b. Wide Angle Intercept Equipment.

(1) General Operating Principle. To provide lens equipment having a 25 inch focal length and /4.5 speed which will be used to image the area of the sky in which the satellite is expected. The image will be detected either by a photoconductive array of individual cells such as provided in Flash Ranging Set AN/GAS-1(XE-3) or a scanning device or a photographic film. If the AN/GAS-1 (Modified) is used it will have a fine grid structure over the photoconductive cells so that as the image moves across the grid a modulated signal will be received, having a frequency characteristic of the expected velocity of the satellite. This is analogous to a petoscope. If a photographic film is used the film will advance at a rate sufficient to stabilize the image so that when the satellite is at a selected angle and a proper velocity the image does not move on the film. The film exposure would be controlled by limiting the area of exposure as a function of time by an adjustable slot.

(2) Accuracy. The field will be designed for 25° coverage and three equipment would constitute a set which when placed on a base line such as ten miles would be used to measure the angle, the time, and to calculate the position as a function of time. The field would be calibrated to have an accuracy of 1 angular mil or e minutes of arc.

[redacted]

[redacted]

5. RECOMMENDED OPTICAL INSTRUMENTATION: b. Wide Angle Search Equipment
(2) Accuracy. (Contd)

On a 10 mile base line the satellite altitude could be measured with a probable error of 0.4 of a mile at 400 miles. The time could be measured to 1/30 of a second.

(3) Power Supply. Minor

(4) Reliability. The 1/2 meter satellite would not be observable with either equipment during the day. It would be detectable during approximately one and a half hours of twilight, morning or night. The 1 meter satellite would be just detectable by the photoconductive, conductive and photographic device at its closest approach during the day and visible to both devices at twilight. The 10 meter satellite will nearly always be visible during the daylight to the photoconductive device and half the time to the photo device. The photoconductive device is considered to be equivalent to the 40X telescope listed in Table II. To insure that the satellite would be detected every circuit of the orbit
25 would require eight stations each containing one set. This circuit of stations would give complete coverage for the latitude of sunset does not change with time except for pertubations. As the sets would be light and portable they could be shifted easily by air to fit changing conditions.

(5) Schedule and funding. Either unit could be modified from existing designs and produced in fifteen months. The price for one set of three lens units would be $80,000 in quantities of eight stations at $640,000.

c. SCEL Precision Optical Tracker. Based on the construction and operation of the Long Range Tracking Instrument shown in Fig. 4, a

[redacted]

LONG RANGE TRACKING INSTRUMENT, (584 Fairchild) . (Laboratory Designed-Constructed)
View . SCR-584 PEDESTAL . Showing Pedestal Assembly, Sun Shield, Camera Housing,
Elevation Counter-Weight and Elevation Drive Motor

DATE 9-11-53 SIGNAL CORPS ENGINEERING LABORATORIES NO. SCEL 38991

Figure 4

[redacted]

5. RECOMMENDED OPTICAL INSTRUMENTATION: c. SCEL Precision Optical Tracker (Contd)

contract was placed to build a precision mount compatible in its guidance through electrical servo input and output with chain radar. This precision mount would be useful in the tracking function in the satellite program.

(1) General Operating Principle. The Precision Mount will consist of a precision bearing assembly operating in azimuth and elevation. The control of the motion will consist of a precision servo system which will operate from data introduced by external sources or by the control of the human operator feeding information to the servo through a "crystal ball control". The operator control sets in smooth velocity and displacement information so that he can track with a probable deviation of 0.1 angular mil or 18 seconds.

(2) Accuracy. Three types of output are available, they
27 are in order of precision: First, a fine servo control accurate within 0.5 to 1 angular mil; second, electrical digital take-off at .1 mil units, and, third, a divided circle precise to 1 second photographed on the recording film. The optics on the Long Range Tracking Instrument consist of a visual tracking telescope designed particularly to give maximum contrast of point targets and a combination reflector-refractor telescope designed for stability and precision. The optics were designed by Dr. James Baker for the quality work required for satellite location. The SCEL tracker has a variable focus sighting telescope as indicated in the tables of magnification 10, 20 and 40 power. The photographic lens consists of a rigid basic 40" reflector sealed with fiducial markings and 1/2, 1, 2 and 4 power projecting refractors making equivalent focal lengths of 20, 40, 80

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5. RECOMMENDED OPTICAL INSTRUMENTATION: c. SCEL Precision Mount.
(2) Accuracy (Contd)

and 160 inches. An 8 power refractor could be built to give 320 inches as shown in the table.

(3) Power supply. 50-65 cycle 115-125 volt ac, single phase.

(4) Reliability.

(a) Visual Tracking. For a 1/2 meter satellite visual tracking can only be done at twilight; however, the instrument can follow radar or computed data tracking to 1 mil. For a 1 meter satellite, the satellite may be seen only at the "best case" during the day and at twilight. For a 10 meter satellite, tracking may be done nearly any time.

(b) Photographically. The photographic lens will do about equivalent to the visual tracking.

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(c) Location Accuracy. The altitude and position can be determined with a probable error of .04 miles at 400 miles and a time accuracy of 1/300 of a second.

(5) Schedule and Funding. The precision instrument mount is just in the preliminary design stages. Additional units could be made available in 24 months by increasing the priority of the program. Two Long Range Tracking Instruments are available. Complete sets with two mounts to a set would cost $225,000 up to four sets. Additional sets would cost $200,000.

d. Astronomical Telescopes. The problem of resolving the 20" satellite is difficult as indicated in Table III. Once the orbit is known precisely, information can be furnished to suitable astronomical observers who have both good seeing conditions and high resolution equipment. It

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5. RECOMMENDED OPTICAL INSTRUMENTATION: d. Astronomical Telescopes (Contd)

will be possible to resolve the object and study the details of motion. Inasmuch as these equipments are very expensive and require special locations, it is important to choose an orbit to pass over these stations.

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6. REFERENCES:

a. R. Tousey and E. O. Hulburt, "Visibility of Stars in the Daylight Sky", Journal of the Optical Society of America, Vol. 38, No. 10, October 1948.

b. R. Tousey and M. J. Kooman, "The Visibility of Stars and Planets During Twilight", Journal of the Optical Society of America, Vol. 43, No. 3, March 1953.

c. Hynek, J. A., "Photographing Stars in the Daytime", Sky and Telescope, January 1951.

d. M. J. Koomen, C. Lock, D. M. Packer, R. Scolnik, R. Tousey, and E. O. Hulburt, "Measurements of the Brightness of the Twilight Sky", Journal of the Optical Society of America, CVol. 42, No. 5, May 1952.

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

1. Tracking Systems

b. Radar Skin Tracking Systems

(1) Diana Type Radars

[redacted]

II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I

1. Tracking Systems

b. Radar Skin Tracking Systems

(1) DIANA Type Radars

Prepared by: Physical Sciences Division

SNYOPSIS: This section is concerned with the possibility of tracking the satellite by use of existing DIANA-type radars. Modifications which are necessary to make the sets suitable for acquiring and tracking the satellite are indicated and time and cost estimates are furnished. Some brief considerations are given concerning possible target modifications which could be made in order to provide a larger radar cross section. Finally, a series of planned tests is mentioned, in which one of the radar sets will be used in its present site to track rockets fired in Cocoa, Florida.

DISCUSSION:

The two radar sets which are considered here make use of the same paraboloidal reflector (see photograph on following page). The radiating antenna for each set is detachable so that changeover can be achieved with a minimum of effort. These sets have frequencies of 151.11 mc/s and 413.25 mc/s. It is concluded that the 151 mc/s set can be more easily adapted to the problem of tracking the satellite (see appendix). Thus it is proposed that this set, with necessary modifications, be taken as a basic item.

The 151 mc/s set can track a totally reflecting sphere having a diameter of 20 inches to a maximum range of about 500 miles. This leads one to conclude that the set will track the satellite satisfactorily only in the nearer portions of the orbit. (Although the optimum frequency, on the basis of cross section considerations, is in the neighborhood of 200 mc/s, this seems hardly significant in view of the large amount of engineering already done at 151 mc. Moreover, later developments may

PARABOLIC ANTENNA REFLECTOR . (Experimental) . Task 182-A
For Use in "Very Long Range Radio Propagation"
Overall View . Showing Complete Assembly of Reflector, Mount and Tower

DATE 4-2-53 SIGNAL CORPS ENGINEERING LABORATORIES NO. SCEL 42445 37

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result in spheres of larger diameters - hence the gain would be only temporary.) Of the various target modifications suggested, it appears that an increase in diameter by a factor of 3 1/2 or 4 would be most desirable.

The accuracy of this set is not of a very high order - in fact the angular accuracy is of the order of a few degrees and the range accuracy is no better than one part in one hundred for tracking a target such as this satellite.

The set will require the following modifications so as to permit tracking of the satellite:

a. Provision for scanner and automatic tracking.

b. Provision for doppler tuning range from + 6.5 kc to - 6.5 kc.

c. A larger dish is desirable for greater gain and improved angular accuracy.

The cost of the set, including mount, antenna, and paraboloidal reflector will be about $800,000. The time involved would depend upon the number of sets required. With modifications, one set should be deliverable in 10 months, while an order of 5 should be deliverable within 2 years.

The existing set will be used in connection with rocket firings beginning in September 1955. From available figures on the altitudes which will be reached, the radar will view the upper 13 to 20 mile portion of the missile trajectory. Feasibility studies will be conducted and the information obtained will be applied to the problem of satellite detection. It is not likely that the information will affect the above considerations in an adverse manner.

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I
1. Tracking Systems
b.1 Radar Skin Tracking (DIANA Type Radars)

Prepared by: Physical Sciences Division

APPENDIX

1. Operating Principle: The two radar sets which are considered here make use of the same paraboloidal reflector. The radiating antenna on each is detachable so that change-over can be achieved with a minimum of effort. The basic principle of operation is just that of the standard pulsed radar. Relevant parameters of the systems are given in the tables below:

Frequency	151.11 mc/s	413.25 mc/s
Power	50 kw (rf)	7 Mw (peak pulse)
Antenna Gain	380	2800
Beam Width	8.5° x 9.5°	3.1° x 3.5°
Prf	1/8 to 100/sec.	1/8 to 100/sec.
Receiver-Bandwidth	Variable (stepwise) 50 c/s up to a few kc.	2 mc
Noise Figure	3 db at 50 c/s	6 db

The effect of the TR system on the noise figure has not been included in the value for the 151 mc/s system, nor has the fact that this figure becomes somewhat worse at the broader bandwidths necessary for the pulse-lengths required here. The overall noise figure may well be 5 db or so in the practical operating situation. (No suitable TR system exists at present for the 413 mc/s set. Moreover, improvement in frequency stability of the transmitter and development of a low noise narrow band receiver will be required before it can be employed for the purpose at hand.)

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2. Range Considerations: If one assumes the satellite to be a totally reflecting sphere having a diameter of 20" (say roughly 50 cm), then the radar cross section at 151 mc/s is 0.35 square meters. For such a sphere, the existing set would have a maximum range of about 500 miles. This result is obtained by using a noise figure of 5 db and a bandwidth of a kilocycle. Pulses shorter than one millisecond will require larger bandwidths and hence will increase the minimum detectable signal. Thus the maximum range may be somewhat less than that given above.

These considerations lead one to conclude that the existing set will track the satellite satisfactorily near perigee but will be wholly inadequate for the larger distances. If arrangements are possible for which the narrow band of 50 cycles can be employed, one gains a factor of about 1.6 in range and the target is just detectable at apogee.

These considerations have not been applied to the 413 mc/s system. For that frequency, the radar cross section is near the first minimum of the scattering curve. The large peak pulse makes this fact not very serious, but the receiver bandwidth and noise figure do not promise encouraging results. With modifications (including stabilization of the transmitter frequency) this set should be adequate for tracking the satellite. However, a rather large engineering undertaking may be involved.

The optimum frequency, on the basis of cross section, is in the neighborhood of 200 mc/s for a 20-inch sphere. One need not limit considerations to a sphere since other shapes provide more effective scatterers. The suggestion of corner reflectors symmetrically placed about the

center of the object has been made. The Harvard group has considered use of such a reflector covered by a spherical surface which is transparent to radio waves but is made optically reflecting by a suitable coating. The use of facets is discarded because the increase in gain is offset by the decrease in size (with respect to the sphere) for almost any reasonable configuration. This suggests the most natural modification, viz. increasing the diameter by a factor of the order of 3 1/2 or 4. This increase would permit tracking at apogee with some margin of safety.

These considerations have ignored the doppler shift which is rather large. Indeed, the large doppler frequencies and the rapid rates of change in these frequencies will make tracking of the satellite by conventional radar techniques rather difficult. The existing DIANA radars will require appropriate modifications as indicated below.

3. <u>Orbit Considerations</u>: There are no fundamental limitations as to orientation of the orbit for radar detection. If radar were to be the sole means of tracking, then the desirability of having every station track the satellite at every passage would dictate an equatorial orbit. Since it is desirable to supplement radar tracking by other means, it may be necessary to choose other orientations even at the expense of multiplying the number of stations required. E.g. for optical observations, the locations of astronomical observatories, information concerning cloud coverage, and possible use of many amateur astronomers should be considered in reaching a decision. In a later model equipped for experiments, the equatorial orbit would be the least interesting of all possible choices.

Simple geometrical considerations show that near apogee the horizon distance of the satellite for a ground station is 29° 43', while near perigee it is 17° 51'. Other values of the horizon distance are intermediate to these. For overhead passages, the station will observe portions of the orbit equal to twice these values. All other passages give less. (These calculations were made for h = 200 mi at perigee and 600 mi at apogee.) Circles having centers at the stations and radii equal to the appropriate horizon distance will define the geometrical limits of visibility for the stations. Due to the rotation of the earth, all inclined orbits will drift across these circles of observation at a rate of about 22 1/2° of longitude per satellite revolution. (Clearly this effect is worse for stations nearer the equator.) The radii of all circles will be in the range $17° 51' \leq \theta \leq 29° 43'$, and will vary in time at every station along with the change in the line of apsides.

4. <u>Accuracy</u>: The angular accuracy is limited by the beam width, which is of the order of 9° between half-power points. Pattern measurements have not been made, but the rate of change of power with angle in the neighborhood of the beam maximum is probably too small to permit angle determinations to better than, say, 4°. This accuracy may be improved by limiting the range of doppler frequencies accepted by the receiver (i.e. tuning range - it is not desirable to cover this range by bandwidth!), but one does this at the expense of observing shorter portions of the orbit at each station.

The range accuracy is good to 1 part in 10^8 as concerns the timing circuits. However, the reading accuracy is not that good - and hence is the limiting factor. This latter may be of the order of one part in a hundred.

5. Power and Communication Requirements: The present set makes use of the following power:

300 kVA	440 V 3Ø
100 kVA	208 V 3Ø
30 kVA	120 V 1Ø

This multiplicity will not be required if the pulsing system is simplified.

Requirements for communications are best expressed as those facilities necessary for data transmission between stations for synchronization, and from each station to a central analysis group.

6. Modifications Required: As mentioned above, the most serious limitation of the radar technique is that of the rapid change of doppler frequency during a single passage of the object. For an overhead passage, the doppler frequency at 151 mc/s will vary from about + 6.5 kc/s to - 6.5 kc/s between horizons. This wide tuning range is not available in the existing set. The frequency change given above takes place within a period of time between 9 and 15 minutes in length. If narrow band techniques are to be used, then rather precise knowledge concerning the motions of the satellite must be available.

In order for the 151 mc/s set to be used, it is necessary that automatic tracking and scanning be available and that the doppler tuning range be increased from ± 1 kc/s to ± 6.5 kc/s. An increase in size of the antenna reflector is desired, say to 75 ft. (A 60 ft paraboloid would be satisfactory, and may be obtainable as a standard item.)

The cost of such a set would be about $800,000, exclusive of station facilities. The use of standard items and engineering knowledge obtained on a previously developed item could reduce the procurement time considerably. Five or six sets, judiciously placed, could go a long way toward

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furnishing the required information - and the procurement time should be between one and two years.

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PROPOSALS TO SATELLITE PROGRAM

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II. Signal Corps Program Proposals to Phase I

1. Tracking Systems

b. Radar Skin Tracking Systems

(2) "V" Beam Intercept System

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I
1. Tracking Systems.
b. Radar Skin Tracking Systems
(2) "V" Beam Intercept System

Prepared by: Radar Division

SYNOPSIS:

A proposal is made for a radar system using the "V"-Beam Intercept Technique. Two antennas are used to provide fan beams at a 45° angle with each other and normal to the earth s surface. When the satellite passes through the beams, latitude can be determined to an accuracy of 0.13°, longitude to within 0.06°, and altitude to within 10 miles. The radar station would be placed on the equator and would provide latitude coverage of $\pm$ 100 miles. The estimated development effort is twenty-one months and $850,000.

"V" Beam Intercept System

1. GENERAL

In an effort to achieve a working radar system capable of providing the maximum trajectory information at the least possible cost of time and money, an extremely simple but practical system is described below. This system can be made available in production in time to monitor the flight of the satellite program proposed for the International Geophysical Year. Such a system will be capable of the following:

(1) Detecting the presence of the satellite in its varying orbit about the Earth.

(2) Measuring its orbital position with respect to longitude with an accuracy better than 0.06 degrees.

(3) Determining its latitude or deviation from an equatorial orbit within an accuracy of 0.13 degrees of latitude and with a height or orbital radial error of less than 10 miles.

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(4) The data can be obtained within two (2) minutes.

A-2 The accuracies promised by such a proposal are far from those desired for precise astronomical calculations but they are available thru the use of an extremely simple instrumentation. The use of this technique, and consequently, of the equipment to be described below, is necessarily limited to the use of an equatorial orbit for the path of the satellite.

The intercept system could be expanded to consist of several radar stations distributed about the Earth's equator. The number of these stations would be determined by the number of discrete orbital points required to check or predict any ephemeris of a given accuracy. The proposal is for one station. A station consists of two (2) complete radars, the first using an array which would radiate an extremely narrow fan beam normal to the trajectory (the nearer beam in Figure 1 and Radar #1 in Figure 2), but whose width would be sufficient to provide coverage for any position of the satellite within the boundaries predicted for the apogeal and perigeal extremes as well as for the lateral or latitude deviations caused by an initial firing error up to 1.5° (See Figure 3). Interception of the target in this first fan beam, would provide absolute time information as well as slant range data. The second fan beam, energized by the second radar (the 45° fan in Figure 1 and Radar #2 in Figure 2), is placed at a 45° angle to the anticipated trajectory (and to the first antenna array) so that it essentially completes a "V"-beam configuration. This beam is also extremely narrow in the direction of the target path but wide enough again to provide for any possible target position. The intercept time

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A-3

and slant range data we derive from this second array is then combined with that from the first to provide height, latitude, and absolute time of intercept or longitude information. Such a calculation may be based on predicted target speed but through the use of two such stations, or "V" patterns, separated by several hundreds of miles, even this speed could be determined.

This proposal does not pretend to provide a monitoring or tracking system unlimited in its potentialities or even marginal in its meeting of required accuracies, but it does describe a simple, relatively inexpensive, and realistically tangible piece of equipment which will determine in a practical way the success or failure of the satellite launching.

2. FUNDAMENTAL RADAR PARAMETERS

1. Type of Radar	Pulsed
2. Operating Frequency	201.0 megacycles per second or 1.49 meters wavelength
3. Target Size	20 inch diameter sphere; equivalent echoing area at 201.0 mcs. is 0.752 sq. meters
4. Transmitter	Ring oscillator driven by a crystal controlled chain oscillator
5. Peak Power Output for each radar	24 megawatts
6. Pulse Repetition Rate	100 pulses per second
7. Pulse Width	10 microseconds
8. Duty Cycle	0.001
9. Average Radiated Power for each radar	24 kilowatts

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10.	Antenna construction	Linear array of dipoles with a cylindrical paraboloid for a reflector
11.	Antenna Apertures and Radiation Coverage	(1) Normal Fan: $1^{\circ} \times 54^{\circ}$ is 8 ft. x 300 ft. Antenna Gain of 27.5 db (2) 45° Fan: $0.76^{\circ} \times 70^{\circ}$ is 7.5 ft x 350 ft. Antenna Gain of 27.5 db
12.	Hits per Target Interception	(1) Normal Fan: 279 max, 69 min. (2) 45° Fan: 212 max, 53 min.
13.	Scan Rate	No scanning required
14.	Display Facilities	Ten (10), ten inch A-scopes, each displaying different, but overlapping 50 mile sectors and, in addition, an automatic signal detection ranging circuit and display A-scope.
15.	Recording Facilities	(1) Photographic recording of a sweeping range A-scope. (2) Automatic recording of echo signal strength vs. absolute time. (3) Automatic recording of doppler frequency shift vs. absolute time. (Optional, as a source of supplementary data)
16.	Receiver Noise Figure	4 db
17.	Receiver Band Width	100 Kc
18.	Visibility Factor	3 db
19.	Losses, System	3 db

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3. ESTIMATE OF DEVELOPMENT EFFORT REQUIRED

A. Antennas - $250,000 16 months.

B. Radars, less antennas - $500,000 18 months.

C. Installation and Testing - $100,000 3 months.

D. Totals $850,000 21 months.

4. DESIGN CONSIDERATIONS:

a. General: A realistic approach to the immediate satellite tracking problem, i.e., observation or tracking of a twenty inch metallic sphere, traveling at five miles* per second at altitudes extending from 200 to 800 miles, within the next two years, and without the assistance of a satellite-borne transmitter, precludes the possibility of initiation of a design and production program for any equipment other than the simplest mechanical structures, with practically no provision for new development effort or state of the art advances. The requirement placed upon the skin-tracking designer resolves itself to a simple selection:

(1) Provide an equipment capable of missile detection and location of the best accuracies obtainable with a design sufficiently fundamental and straightforward which is possible of instrumentation, installation, and testing with the next two years, or

(2) Initiate an extensive study and design program leading to the fabrication of an advanced radar system or systems capable of providing accuracies required for precise ephemeris prediction. The latter requirement has been set at a range accuracy of 300 feet at ranges up to 800 miles.

*All references to miles are in statute miles.

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This would also require an angular accuracy of 0.1 mils. This second approach to the problem could certainly not be realized within the prescribed time limitation. The realistic approach to the problem remains as the first proposal, i.e., a sacrifice of accuracies but the accomplishment of the detection and approximate determination of the satellite's orbit.

b. Antenna Configuration: As shown in Figures 2 and 3, the anticipated trajectory of the satellite will generate an orbit about the Earth's Equator with a cross-section extending to 800 miles height above the Earth as a result of its apogeal positions and as low as 200 miles above the Earth as the result of its perigeal positions. The lateral or latitude deviation of the satellite's orbit from the Equatorial plane is occasioned by the lateral launching error which might be as large as 1.5°. This would cause a latitude shift of slightly over 100 miles maximum, each side of the Equator. The period of advance of the line of apsides (recession of the Earth under the target) is 49 days and it is this factor which yields the cross-sectional area illustrated. We thus have a torus or band, of rectangular cross-section, in the Equatorial plane in which we must be able to detect and locate the target. The simplest configuration conceived to provide complete radar coverage of the orbital path (whose cross-section is shown in Figure 3), and at the same time provide latitude, longitude, and height information of fair accuracy, is a "V"-shaped radiation pattern positioned across the anticipated trajectory, as shown in Figures 1 and 2. One section of this pattern would consist of

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an extremely narrow fan of energy whose broad dimension is placed perpendicular or normal to the anticipated trajectory. To fulfill the geometrical requirements, the beam must be 54° wide (in latitude) and must be capable of detecting the target to a range of 805 miles, as in Figure 3. The beam's depth, or narrow dimension in the direction of the trajectory, must be (1) large enough to insure a detectable number of target hits as the target passes at the 200 mile altitude (the worst case), (2) narrow enough to provide a reasonable longitude or zenith time determination, (3) narrow enough to require only a practicable amount of radiated energy at a power level required to detect the target, and (4) wide or deep enough to allow a practical antenna configuration. For the frequency selected, 201 mcs (see the discussion below on frequency selection), a cylindrical paraboloidal antenna reflector was selected with an 8 ft. dimension to shape the 54° plane and a 300 ft. dimension to shape the narrow plane. This antenna would have a gain in the 54° plane of roughly 11 db and in the narrow plane of about 44 db and would provide a 54° x 1° radiation pattern. Such an antenna configuration, utilizing 25 in-line dipole feeds, could be constructed on simple, fixed, wooden or steel frameworks using chicken-wire mesh or suitable expanded metal lining. The 201 mcs frequency would not require contour accuracies better than 3.5 inches, based on 1/16 wave-length tolerances. A second antenna array, similar to that described above, would be located at a 45° angle to the first as shown in Figure 2. To cover the required trajectory cross-section, this beam would have a coverage of 70°. In order to make the two radar trans-

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A-8 mitters supplying energy to the two antenna arrays identical, and to provide equal illumination in each of the patterns, the narrow dimension of the 45° beam is reduced to 0.76°. This will still provide sufficient hits on the target while in the beam and at the same time extend the long dimension of the 45° beam to about 350 ft. This figure is still realizable with an in-line array of 30 dipoles, with a cylindrical paraboloid reflector constructed in the manner described above. Actual location of the antennas will be on the Equator at sites located at least 100 miles and preferably less than 200 miles apart. Every additional 100 miles of separation will increase latitude errors 1.7 miles for initial firing errors of 1.5°.

c. Frequency Selection: For a given metallic sphere, it has been shown* that the echoing area or backscattering is a function of frequency with definite maxima and minima values. As is shown in Figure 4, a pronounced peak in the value of equivalent echoing area occurs at a frequency of 201.0 megacycles per second. Although this frequency, corresponding to a wavelength of 1.49 meters, will require a larger antenna than use of a higher frequency for the same radiation pattern, the increased echoing area co-efficient (better than 4 to 1 for a ten-fold frequency increase), the availability of power output tubes at the lower frequency, and the use of non-scanning antennas made selection of the 201.0 mcs frequency acceptable. The scattering cross-section of moisture droplets varies inversely as the fourth power of the wavelength. The 1.49 meter wavelength selected above should

* Rayleigh-Stratton Resonance Curves, Ridenour, Radiation Laboratory Series, Vol. 1, page 65.

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certainly minimize this scattering and allow all-weather operation. Further increase of wavelength to minimize this scattering effect due to moisture droplets in the atmosphere would only decrease the effective echoing area.

d. Pulse Repetition Frequency: An interval of 8600 microseconds will elapse before a signal is received from a target at a range of 800 miles. (800 miles x 10.7 microseconds/mile). A factor of 15 percent must be added to permit a recovery time for the sawtooth voltages used in the Range Unit and Indicator. Thus, the minimum interval is now 9890 microseconds which corresponds to the maximum PRF of 101 cycles per second. In order to simplify the design of the synchronizer and the mechanical duplexer a PRF of 100 cps was selected.

e. Duplexer: The large size of the antenna makes it mandatory that some type of duplexer be provided in order to use one antenna for both transmitting and receiving. While the electron tube type of duplexer is more desirable, tubes are not available at the required power level. Because of the low PRF and the minimum range requirement of 200 miles, a mechanical type commutator appears to be practical. In its elementary form a constant speed motor would drive a disc at 25 rotations per second which would open and short the receiver transmission line four (4) times a revolution. Synchronizing pulses for the Range Unit, Indicator, and Transmitter would be obtained from the disc by means of steel inserts on the disc and a stationary pickup coil.

f. Hits per Target Interception: The number of hits per beam-

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width is a function of target speed and pulse repetition frequency for a stationary beam of a given width. Converting beamwidth to length of arc in miles at the target altitude and dividing by target speed* the time of target illumination can be determined. Multiplication of this quantity by the pulse repetition frequency yields the number of hits per beamwidth. The resulting data are as follows:

	Normal Beam Alt. in Miles 200	Normal Beam Alt. in Miles 800	45° Beam Alt. in Miles 200	45° Beam Alt. in Miles 800
Beamwidth in miles (in an equatorial plane)	3.49	13.9	2.55	10.6
Illumination time in seconds	0.698	2.79	0.531	2.12
PRF	100	100	100	100
Number of Hits	69.8	279	53.1	212

g. Pulse Width: For an assumed duty cycle of 0.001 (max. for powers required) the pulse width will be

$$\frac{0.001 \text{ Duty Cycle}}{100 \text{ cycles/second}} = 0.00001 \text{ sec. or } 10 \text{ microseconds}$$

h. Peak Power: The necessary peak power was calculated from the radar range equation after the wavelength was optimized to obtain maximum radar echoing area for a metallic sphere of a given radius, the antenna gain having been set by the radiation pattern required by the geometry of the problem.

To compute the power required for the minimum discernible signal we would solve for P in the following:

* A constant target speed of 5 miles per second is assumed.

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$$P = \frac{R_{max}^4 \times 4\pi \; S_{min} \; \lambda^2}{\sigma A^2 \; f^2}$$

where $R_{max} = 805$ miles $= 4.25 \times 10^6$ ft.

$S_{min} = 0.001$ mmw (See Below) $(m = 10^{-6})$

$\lambda = 1.49$ m $= 4.9$ ft.

$\sigma = 8.06$ sq. ft.

$A = 300 \times 8 = 2400$ sq. ft.

$f = 0.6$ (illumination factor)

$P = 6.0$ megawatts

$S_{min} = \overline{NF}KTB$

where $\overline{NF} = 4$ db Noise Figure $= 2.51$

$K = 1.38 \times 10^{-23}$

$T = 300°$ K.

$B = \frac{1}{10 \text{ msc}} = 100$ kc

$S_{min} = 0.001$ mmw

This represents the minimum power required to discern the target with a Noise Figure of 4 db. For an increase of 3 db, signal to noise ratio, or an increase in S_{min} from 0.001 mmw to 0.002 mmw, the power required would become 12 megawatts, peak. This power will provide signals from the target at 805 miles of range which are 3 db above the minimum detectable signals.

An assumed system loss of 3 db will increase this power output requirement to 24 megawatts. This loss figure includes all waveguide, TR and display indicator persistency losses which might be present in

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a system of these dimensions.

-12 At a duty cycle of 0.001 the average power radiated by each radar will be 24 kw.

i. Data Accuracies:

(1) Longitude:

The fundamental basis for all measurements will be the determination of the instant at which the target passes thru the longitudinal zenith plane of the normal antenna array. Therefore, two factors will influence the accuracy of this measurement, viz. (1) the position of the radiation pattern with respect to the zenith plane and (2) the detection or recording of the absolute time at which the target passes thru the center of this beam. The first consideration may be determined within 0.2 degrees by calibrating the power distribution in the beam by flying a test transmitter thru it at altitudes up to 5 miles (the highest practical) and recording signal amplitude vs. elevation angle using a tracking theodolite. The error contributed by displacement of the fan beam from a zenith plane at the maximum anticipated ranges would be 0.2 degrees x 0.0144 °/mile at the Equator, or 0.03 degrees of longitude.

A further error will be introduced in time measurement by the variation of intercept point as determined by the radar observer or automatically determined by a Doppler device or signal amplitude and power distribution comparator. Manual operator marking of expanded range A-scopes (50 miles) will yield results good to one-eighth the time in the beam or for the worst case (800 mile altitude) 1.77 miles

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-13 or .0255 degrees of longitude. The stability of a crystal controlled oscillator being limited to 1 part in one million, will not permit Doppler shift measurements better than 200 cycles. Since this indication will determine target position in the beam only to within one degree no provision for doppler shift measurement should be required of the radars.

The summation of possible errors in longitude consists (1) of a 0.03 degree error due to orientation of the beam with respect to the zenith plane and, (2) a 0.0255 degree error due to marking the intercept time within the beam, or a total of 0.0555 degrees for apogeal heights.

The actual measurement of longitude consists then of merely knowing the longitude of the station and the instant at which the target passes over it.

(2) Target Speed:

The target's speed has been specified to be roughly five miles per second. Although one radar station consisting of a complete V-beam arrangement will be able to derive longitude, latitude and height information based on this or some other given rate of speed, a more accurate determination can be made by having two V-beam stations spaced alon the trajectory as described in the discussion of antenna configuration. These two stations, or essentially two normal fan arrays, each with its longitude or zenith time sensing ability accurate to less than 3.86 miles (at the apogeal orbit), would for a 500 mile antenna separation be capable of determining the target's average

[redacted]

A-14

speed between the two stations to better than 1.74%. Since the speed of the target is varying as it passes thru various portions of its elliptical orbit, it may be quite important to determine speed based on intercept intervals of less than five hundred miles. In these, and the following calculations on accuracy however, all errors based on variation of the given 5 miles per second speed, or that derived from two stations placed 500 miles apart, are considered insignificant in relation to the 1.74% error indicated above.

(3) Latitude:

The determination of target latitude or lateral deviation from the Equatorial orbit requires only a knowledge of the speed of the target as it passes between the two parts of a single V-beam array and the time it takes to do this. The first figure is either predicted on the basis of other ephemeris information or may be determined as described above, in (2) Target Speed. The time between the two beams will be determined by intercept elapsed time recording and this figure will be accurate to about 1.55 seconds. By knowing the target's average speed with an accuracy of 1.74% and the time between intercepts to $\pm$ 1.55 seconds we will know the distance it has to travel through the two beams to within 9.1 miles. This distance traveled is a direct function of the distance from the Equatorial path and may be indicated in degrees of latitude North or South of the Equator. The accuracy of this measurement will be within 9.1 miles or 0.13 degrees of latitude.

[redacted]

A-15

(4) Altitude:

Once the latitude of a target is known to 0.13 degrees or 9.1 miles and its slant range is also known to a pulse width or 9.3 miles then the altitude of the target may be computed to an accuracy of 9.3 miles (at the worst case of apogee and lateral or latitude deviation). This provides a height accuracy figure of better than 5.5% at 200 miles and 1.16% at 800 miles.

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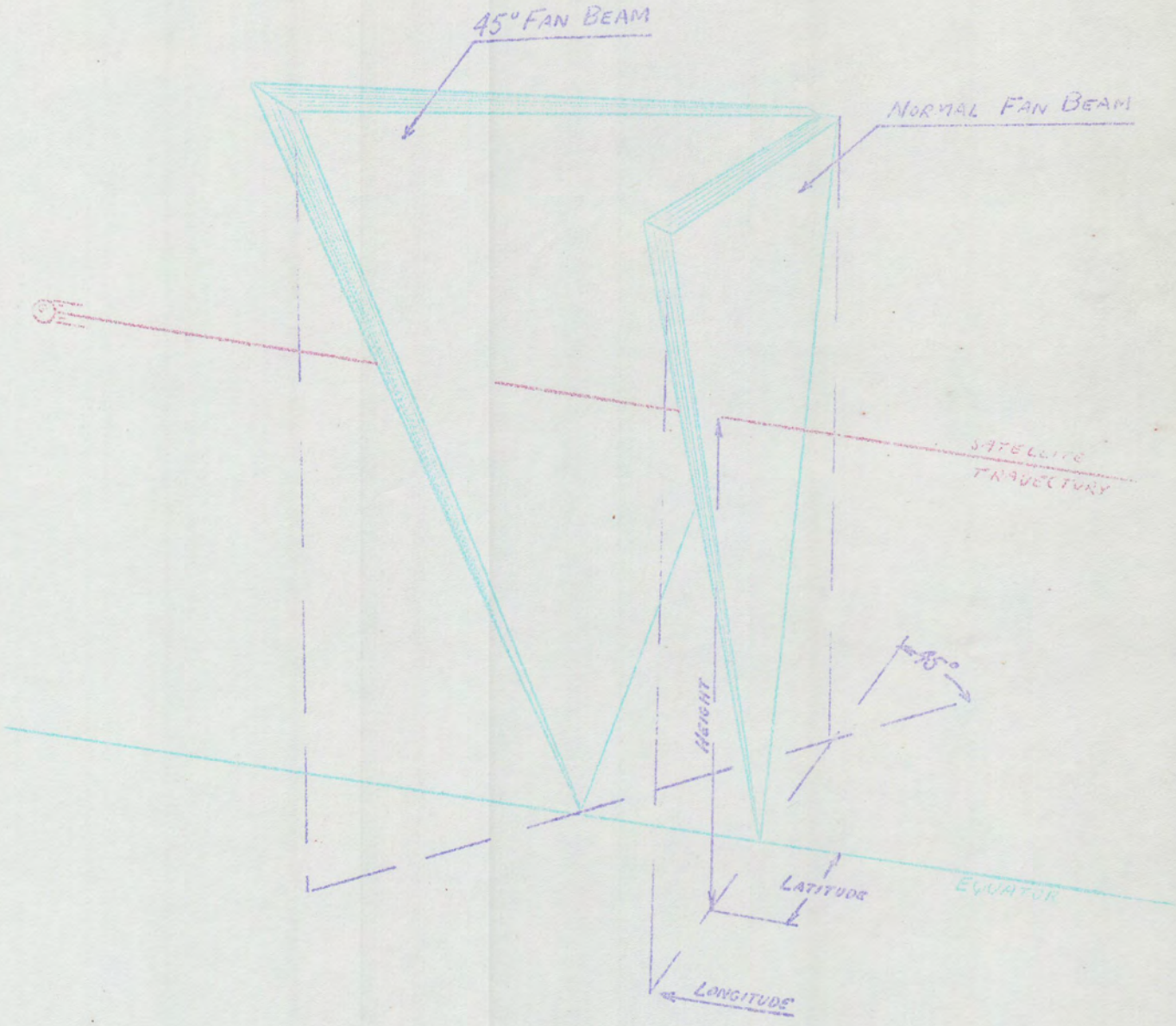

Figure I - Perspective View of Normal and 45° Fan Beam Intercept Pattern

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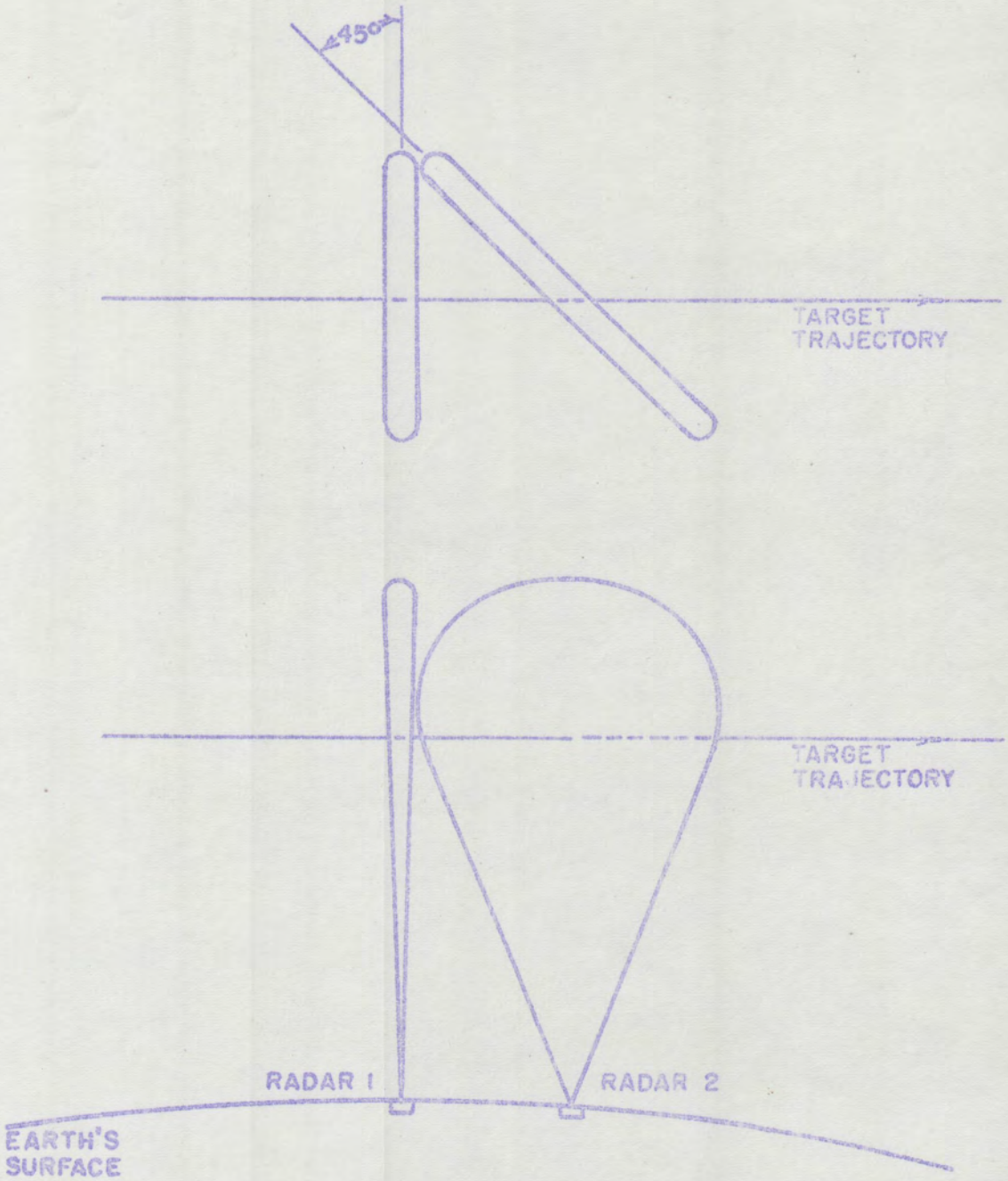

Figure 2 - Orthographic Projections (plan view above, profile below) of the V-Beam Intercept Radiation Patterns

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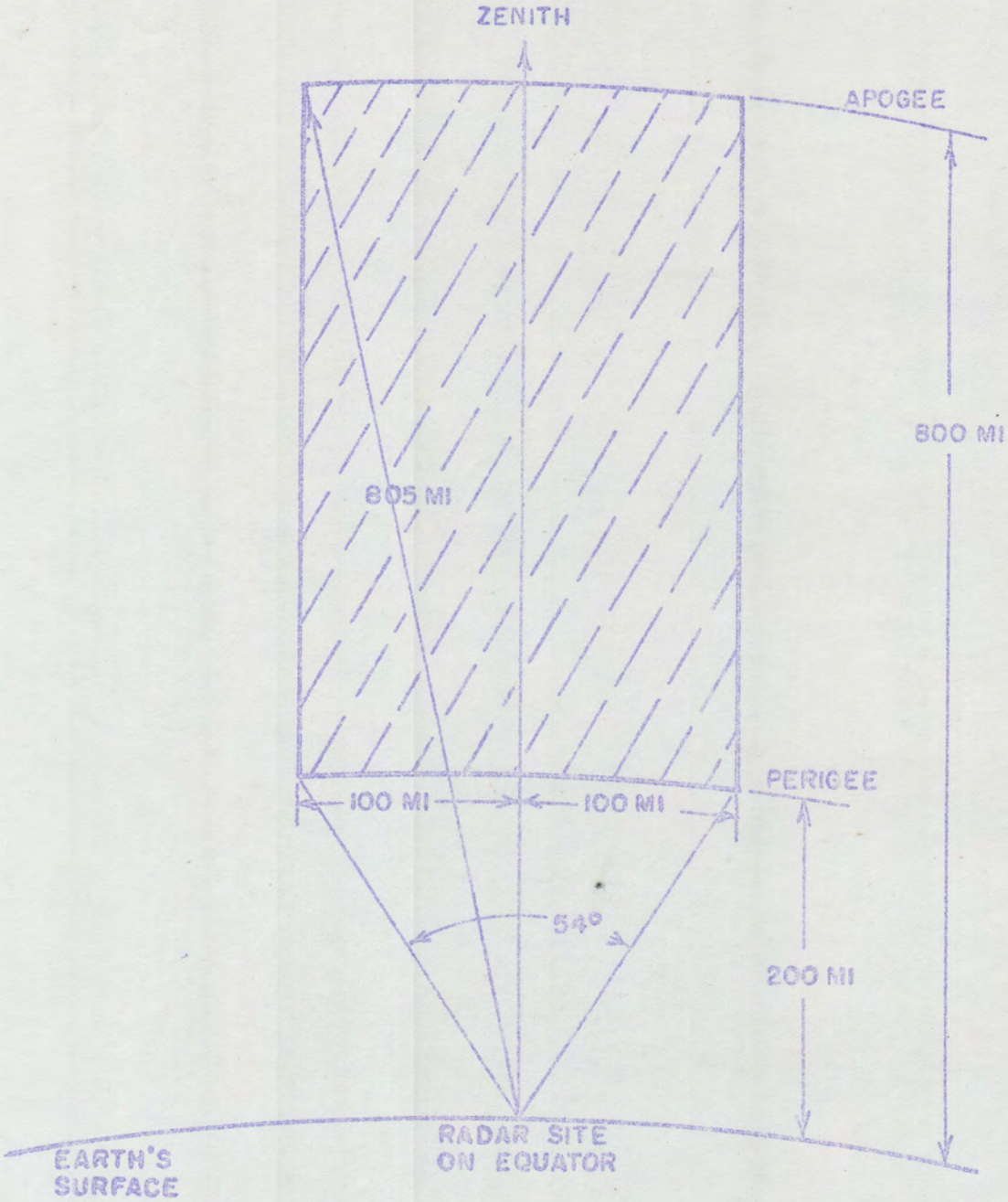

Figure 3 - Cross Section of Anticipated Orbital Paths, Showing Altitude Variations Due To Recessing Elliptical Orbit and Inclination to an Equatorial Plane, and Maximum Anticipated Slant Range Requirement

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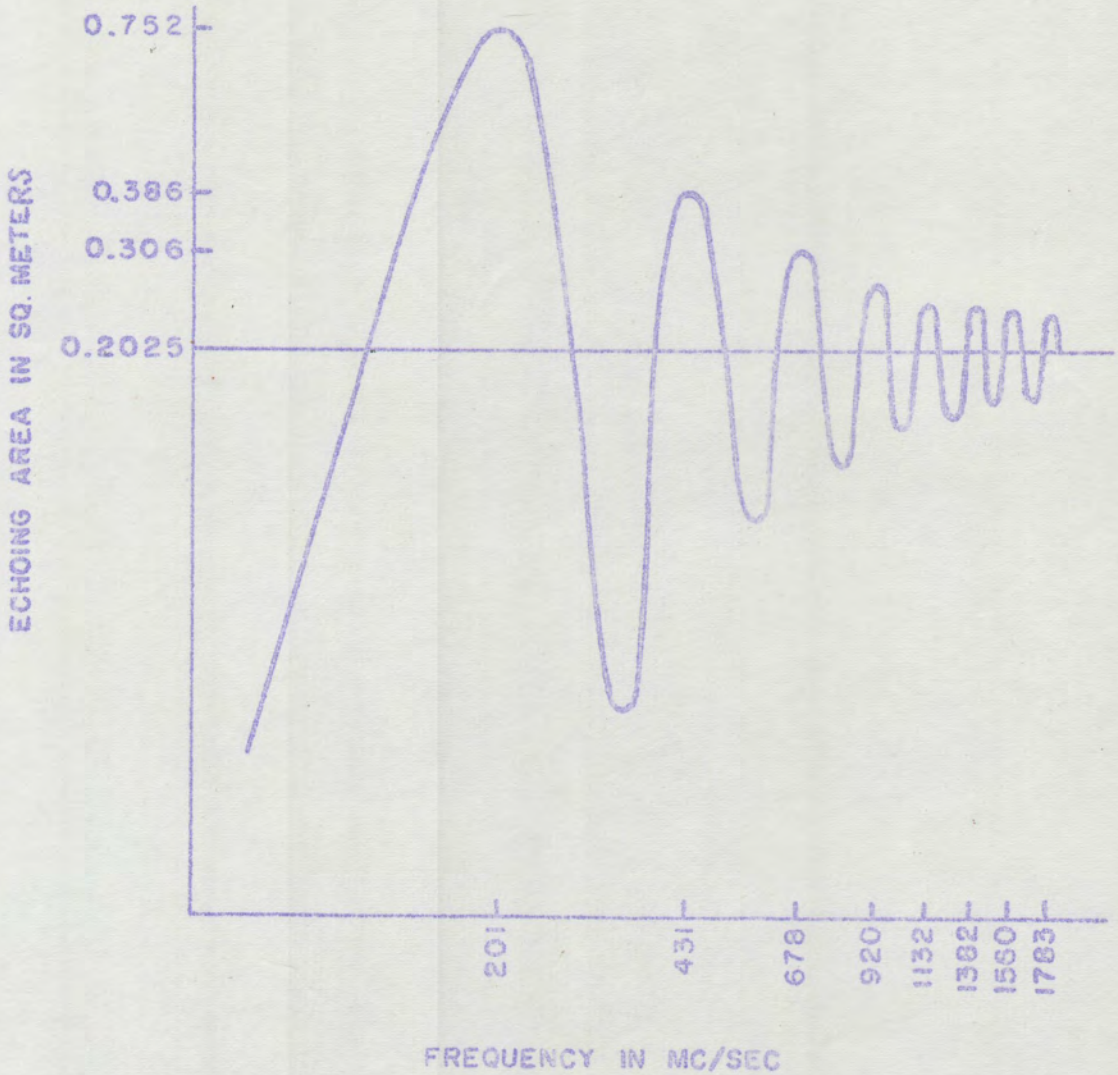

Figure 4 - Backscattering on Radar Cross Sectional Areas for a Metallic Sphere of 20" Diameter, showing the Resonance Maxima plotted vs. Frequency

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

1. Tracking Systems

c. Tracking Systems Requiring Electronic Equipment in the Satellite

(1) "Radiv" Tracking System

[redacted]

II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I

1. Tracking Systems.

c. Tracking Systems Requiring Electronic Equipment in Satellite

(1) RADIV System

Prepared by: Radar Division

SYNOPSIS:

A detailed design proposal is presented for a tracking interferometry system (RADIV) (see Figure 1), requiring a minimum of electronic equipment in the satellite, which will give accurate position and velocity information over a possible 800 mile section of the orbit and provide instantaneous pointing to an optical system (and later telemetering equipment) over a possible 2400 mile sector of orbit.

Multiple station considerations, including recommended sites, survey requirements, acquisition features, are also given.

Time and manpower estimates are included.

1. GENERAL SYSTEM DESCRIPTION

The proposal for a satellite tracking system using a VHF transmitter mounted in the satellite, known as the RADIV System, provides for position determination to an accuracy of $\pm$ 900 feet, by triangulation between two locations. Velocity determination to 25 feet per second may be accomplished if position data is smoothed on a least-square basis prior to velocity determination.

The RADIV System provides for continuous tracking of the satellite over a sector of the orbit from 320 to 800 miles, dependent upon the satellite altitude. It also provides the means for directing an accurate optical system over a possible 2400 mile sector of the orbit.

The RADIV System will require a satellite weight of 6 pounds overall, in order to provide for a transmitter, power supply, and antenna. The installation contemplated will provide for mounting these

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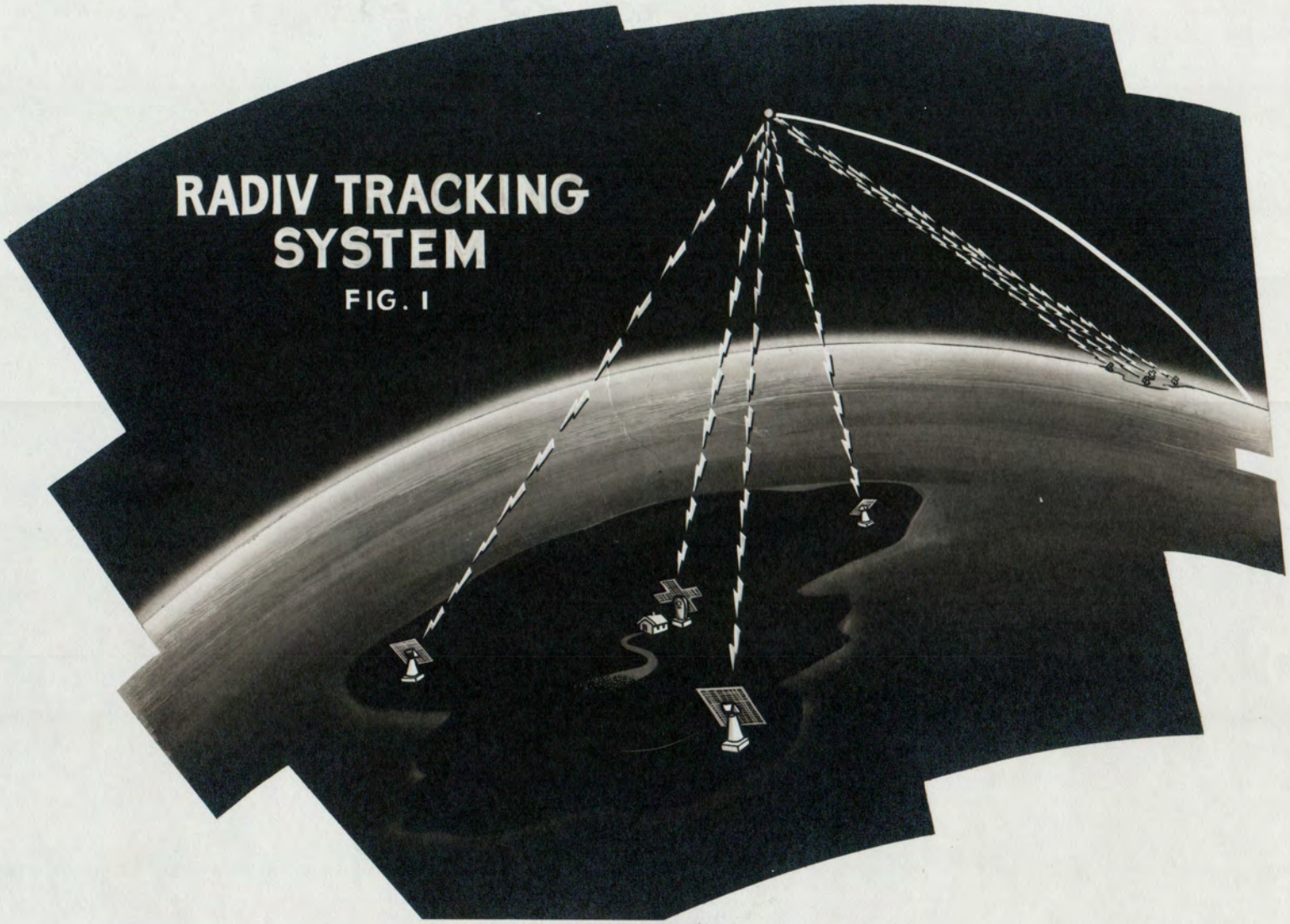
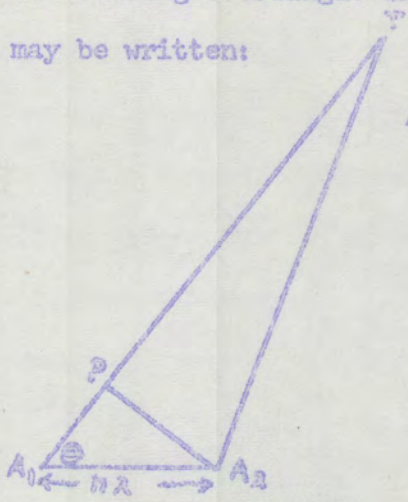


components inside a 20-inch diameter sphere. The transmitter frequency has been tentatively established as 125 MC because of the availability of lightweight, efficient components, though it is recognized that the exact frequency allocated may be somewhat different due to interference or conflict with existing uses of this band. The transmitter will consist of a single stage crystal controlled oscillator which will feed a dipole-like antenna so that the radiated power is approximately 2.5 milliwatts cw. Life of the transmitter, based upon one and one-half pounds of batteries will be approximately 15-20 days.

The basic system used for accurate angle tracking is that of phase comparison or radio interferometry. The angle of arrival of a radio signal from the satellite is determined by measuring the difference in length of the radio paths from the satellite to each of two antennas located a known distance apart.

For example, let T in the figure below represent the satellite transmitter, A1 and A2 the receiving antennas spaced a distance of n wavelengths. If T is far distant from A1 and A2, then the triangle A1PA2 may be considered right triangle and the electrical path length difference, A1P may be written:

$$A_1P = \beta n \lambda \cos\theta$$
$$= 2\pi n \cos\theta$$



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A characteristic of this method of measurement is that ambiguities exist between each 360 degrees of electrical phase difference measured since for a given phase angle measured $0 \leq \phi_e < 2\pi$. No information is available to distinguish between ϕ_e and 360 degrees plus ϕ_e. We may write an expression showing this relationship between total phase difference and measured phase difference as follows:

$$\phi_e + K 2\pi = 2n\pi \cos \theta$$

Here K is an integer which lies between -n and n, that is, there are n values of θ that correspond to a given ϕ_e between 0 and 90 degrees.

Thus it may be seen that this method of measurement requires a supervisory coarse angle measuring system which defines which lobe of the phase pattern, i.e., which value of K, corresponds to the measured ϕ_e.

For the case of the RADIV System, the baseline length between phase comparison antenna is chosen as 50 wavelengths. Thus, a coarse angle measurement to an accuracy of $\pm$ 0.5 degrees may be shown to be sufficient to resolve the ambiguities of the phase measurement.

Figure 2 shows a diagram of the layout of the RADIV System. At each location there are two pairs of phase comparison antennas seperated by 50 wavelengths and a master tracking antenna to which the three phase comparison antennas are slaved. The coarse angle tracking system acquires the satellite and tracks in azimuth and elevation by means of lobe-switching to an accuracy of $\pm$ .5 degrees. At each location there are two phase comparison receivers and a tracking receiver, together with the associated data converters and servos which derive two sets of electrical phase angles and values of K. This information is sufficient

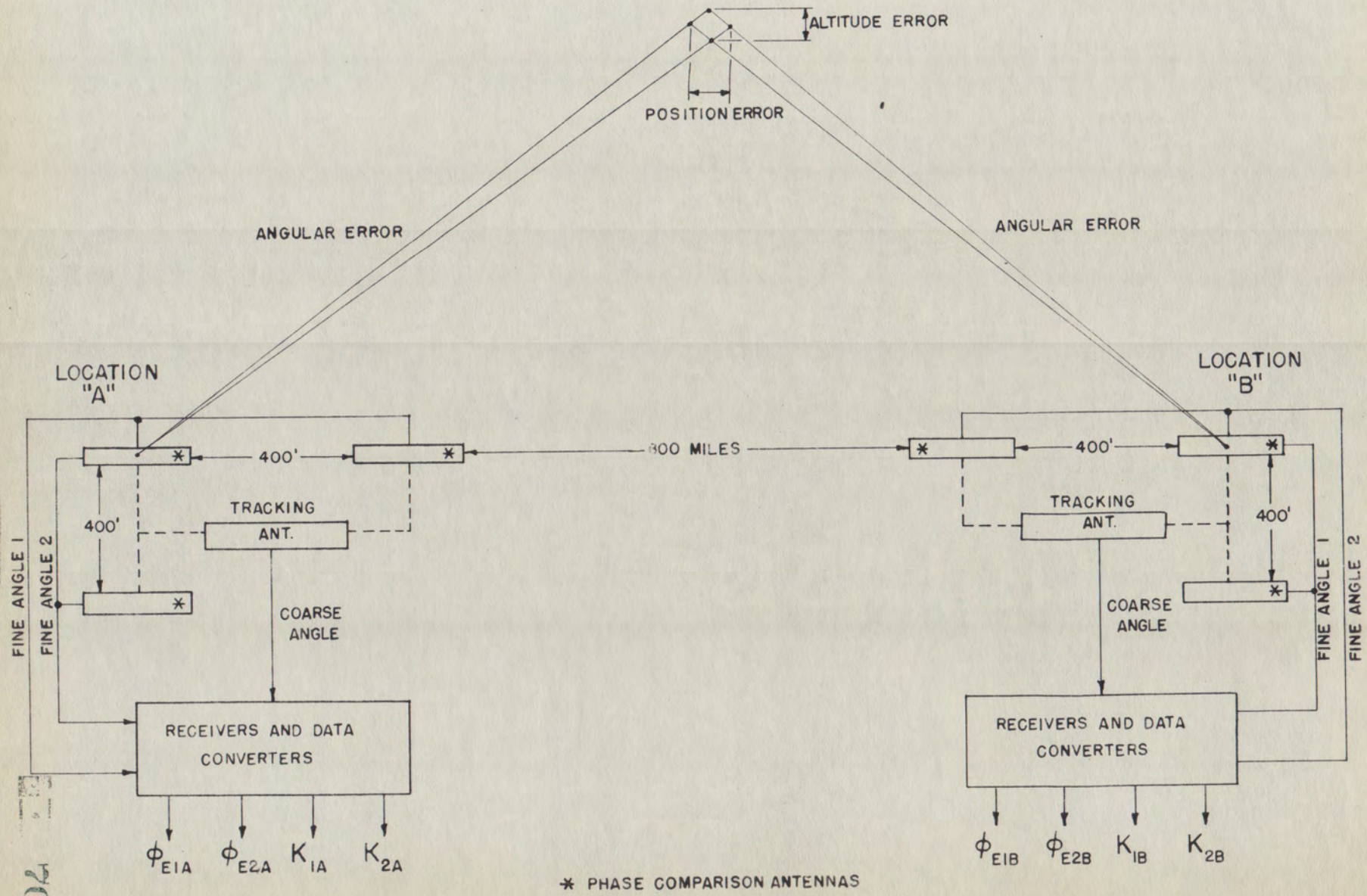

FIGURE 2 LAYOUT OF RADIV SYSTEM, BLOCK DIAGRAM

to compute the direction angles in two planes.

Position determination is accomplished by triangulation with the angle data from two locations separated by approximately 800 miles. It is contemplated that this computation be accomplished at a remote central computer. Here the angle data from the stations will be received, corrected for systematic errors such as parallax, difference in time of measurement due to different path lengths, etc., and the position data computed. This position data would then be smoothed and the velocity computed from position and time.

The tracking range of the system is approximately 1400 miles and the detection range is approximately 3000 miles, which is in excess of limitations of line-of-sight. There are, however, a number of errors inherent in the system, most of which are a function of elevation angle; consequently the system range is limited because of inaccuracies at low angles. It has been established that the system will not accurately track at lower angles than 15 degrees. An analysis of system errors and their effect upon accuracy is contained in Appendix I to this section.

2. DETAILED SYSTEM DESCRIPTION - GROUND SYSTEM

a. Coarse Angle Tracking System.

(1) General Description.

The coarse angle tracking is accomplished in two antennas, one for azimuth tracking and one for elevation tracking. Each antenna has a beamwidth of $20^\circ \times 80^\circ$, the narrow beam being in the tracking dimension. Mechanical lobe switching is used to generate the tracking error signal. The output of the receiver is a square wave at the lobing frequency. This square wave is compared in a phase detector

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to a reference voltage generated from the lobe switching mechanism. The phase between the receiver output and the reference voltage determines the direction in which the antenna must move to reduce the error signal to zero. For initial detection, the bandwidth of the receiver is 20 KC. After the AFC locks on the received signal, receiver bandwidth is reduced to 2 KC. This will result in considerably tighter tracking.

A block diagram of the azimuth tracking system is shown in Figure 3. The elevation tracking system is identical. A detailed description of the tracking antennas is contained in Appendix 2.

b. Range Considerations.

For a tracking accuracy of $\pm$ 0.5 degrees, the range of the system will be approximately 1400 miles. The calculations made to arrive at this figure are given in Appendix 3.

c. Fine Angle Tracking System.

(1) General Description.

The fine angle tracking system is similar to the Single-axis Phase-Comparison Angle-Tracking System developed at NRL*. A block diagram of the system is shown in Figure 4. A detailed description of the phase comparison antennas is contained in Appendix 4.

The signals received at each of the 3 phase comparison antennas are fed into 3 separate preamplifiers. These amplifiers have low gain and a bandwidth of 10 MC to minimize phase drift. The outputs of the amplifiers are fed into 3 separate mixers. Two of the signals are beat with a 125.500 MC local oscillator, while the third, the common

* NRL Report 4393, September 2, 1954

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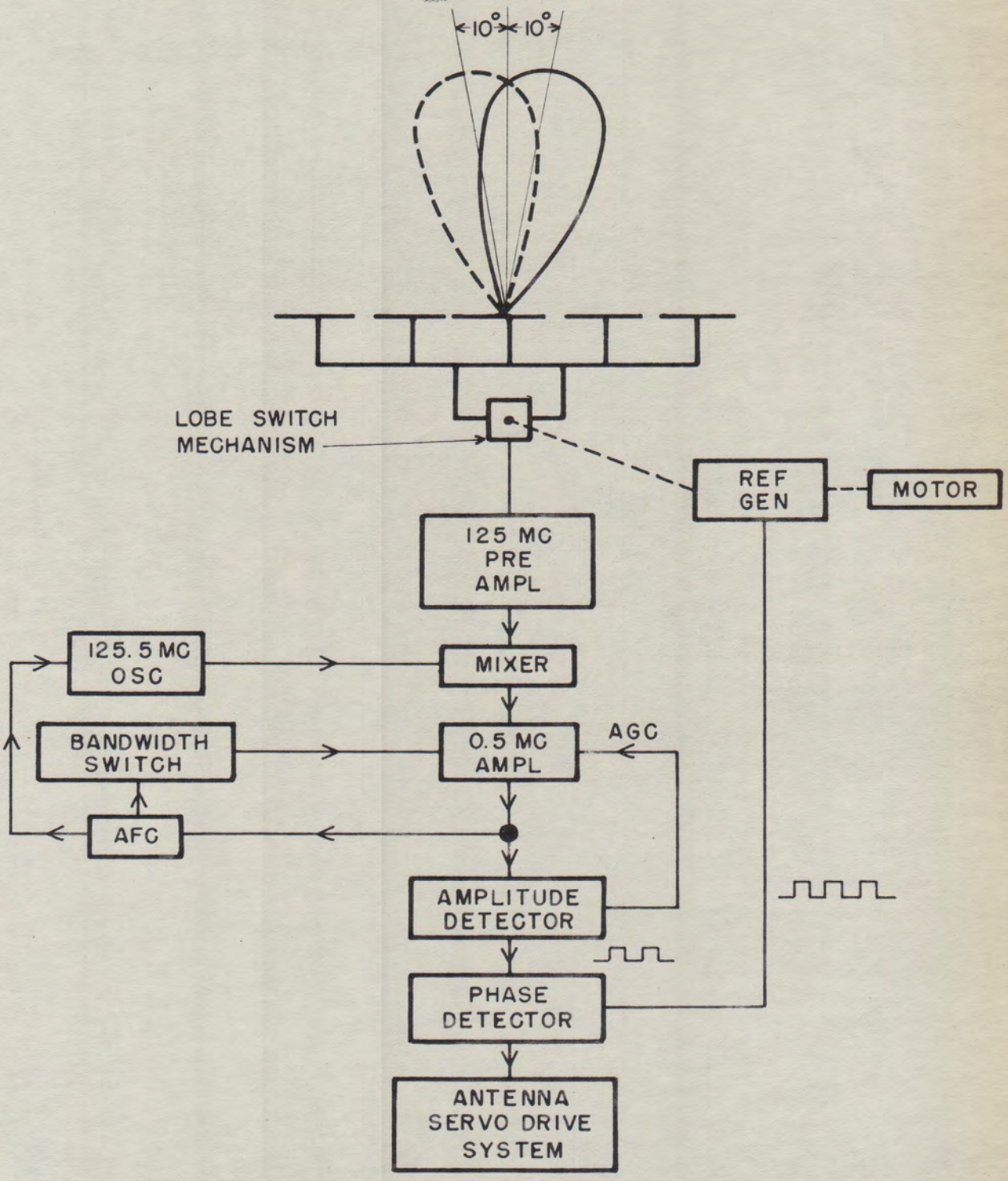
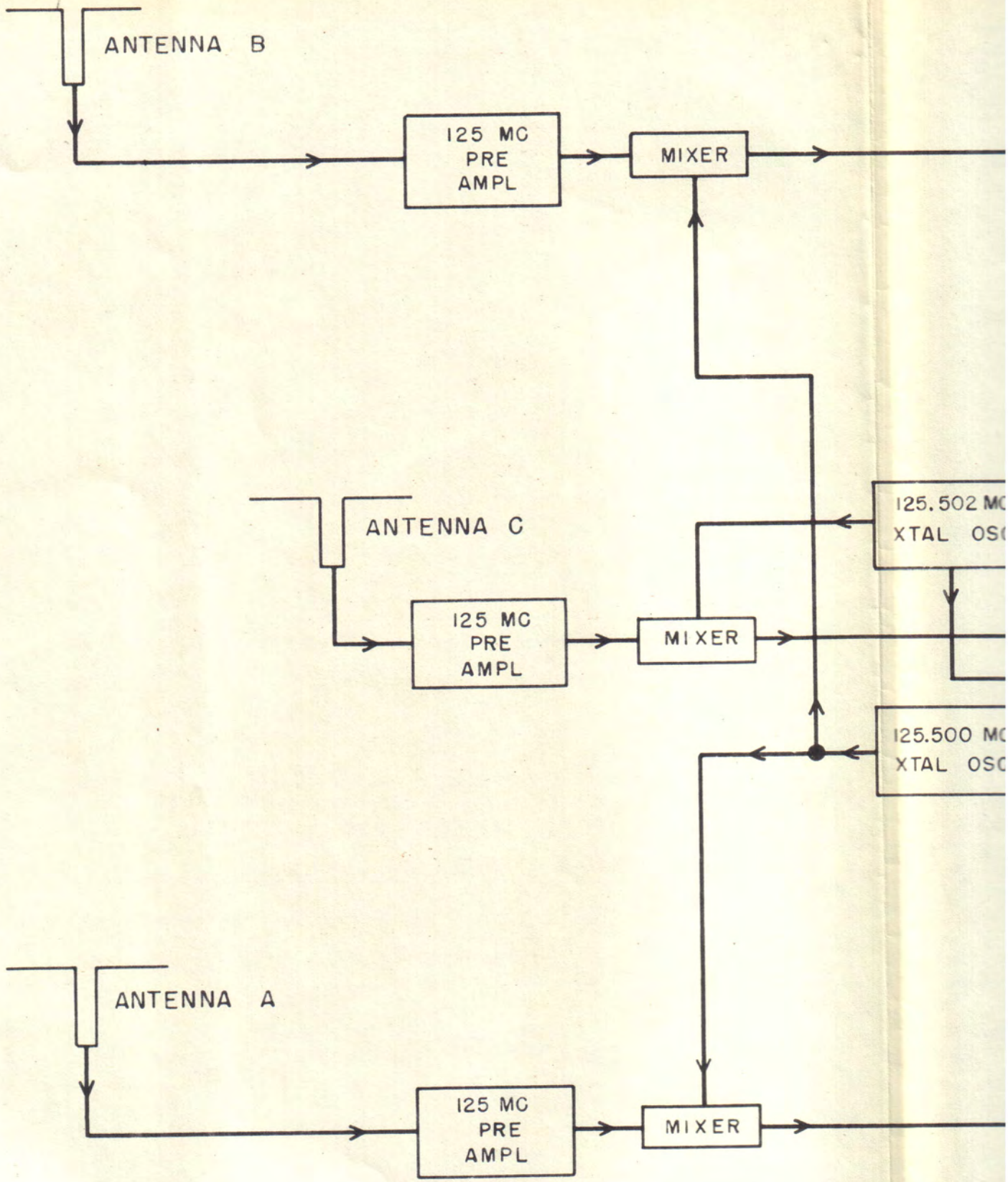


FIGURE 3. AZIMUTH TRACKING SYSTEM-
BLOCK DIAGRAM


ANTENNA B
125 MC PRE AMPL
MIXER
ANTENNA C
125 MC PRE AMPL
MIXER
125.502 MC XTAL OSC
125.500 MC XTAL OSC
ANTENNA A
125 MC PRE AMPL
MIXER

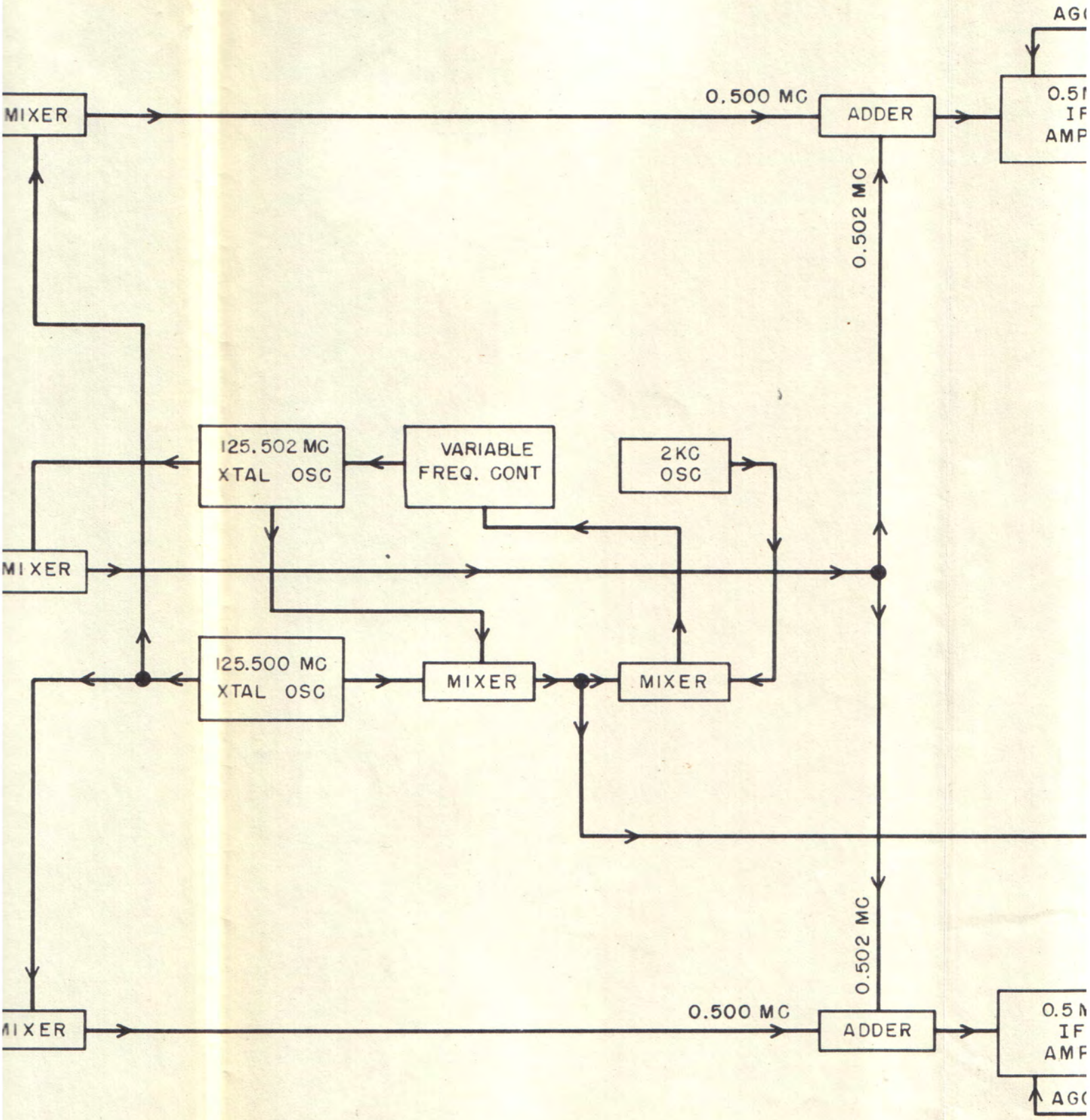
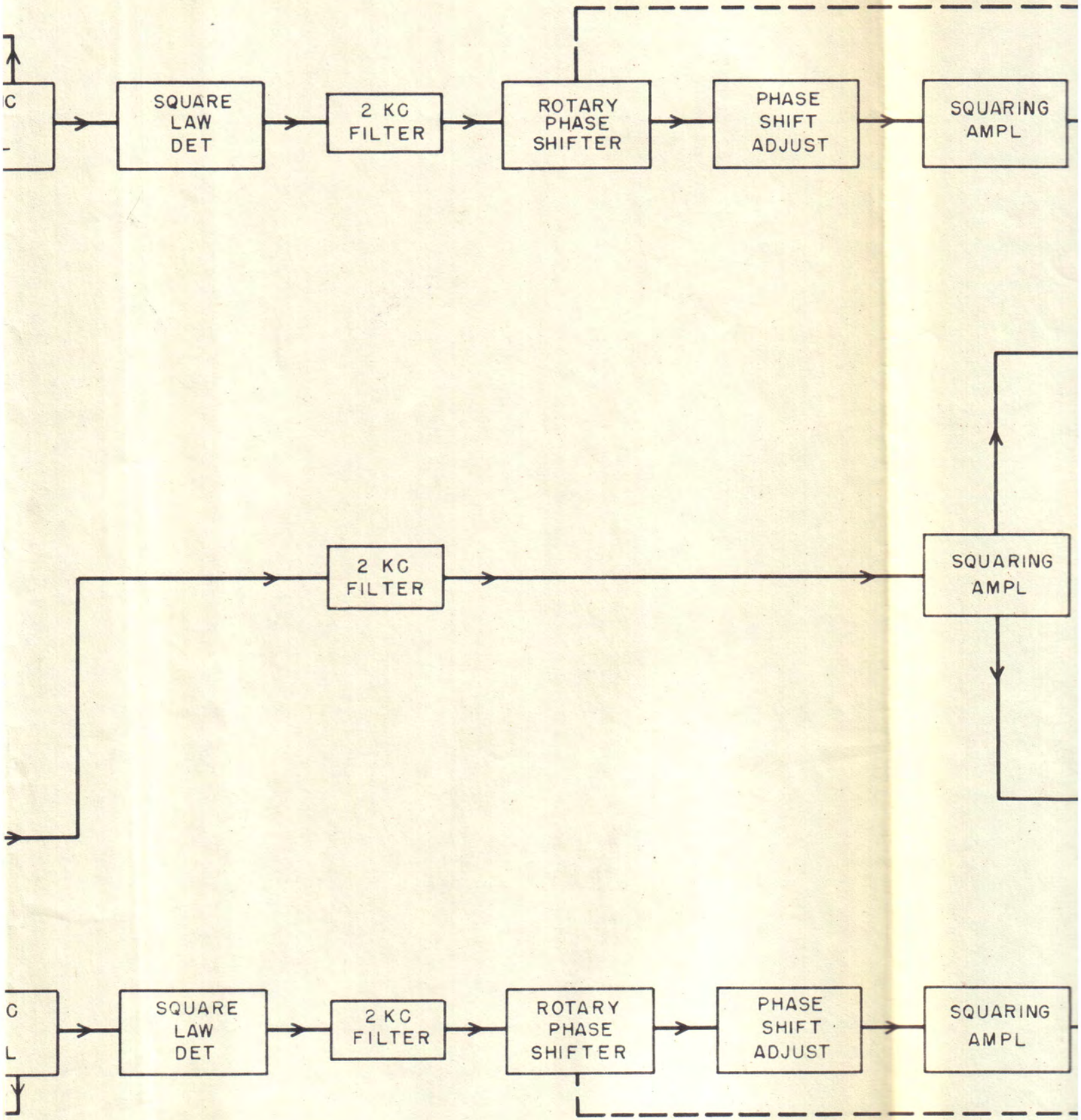
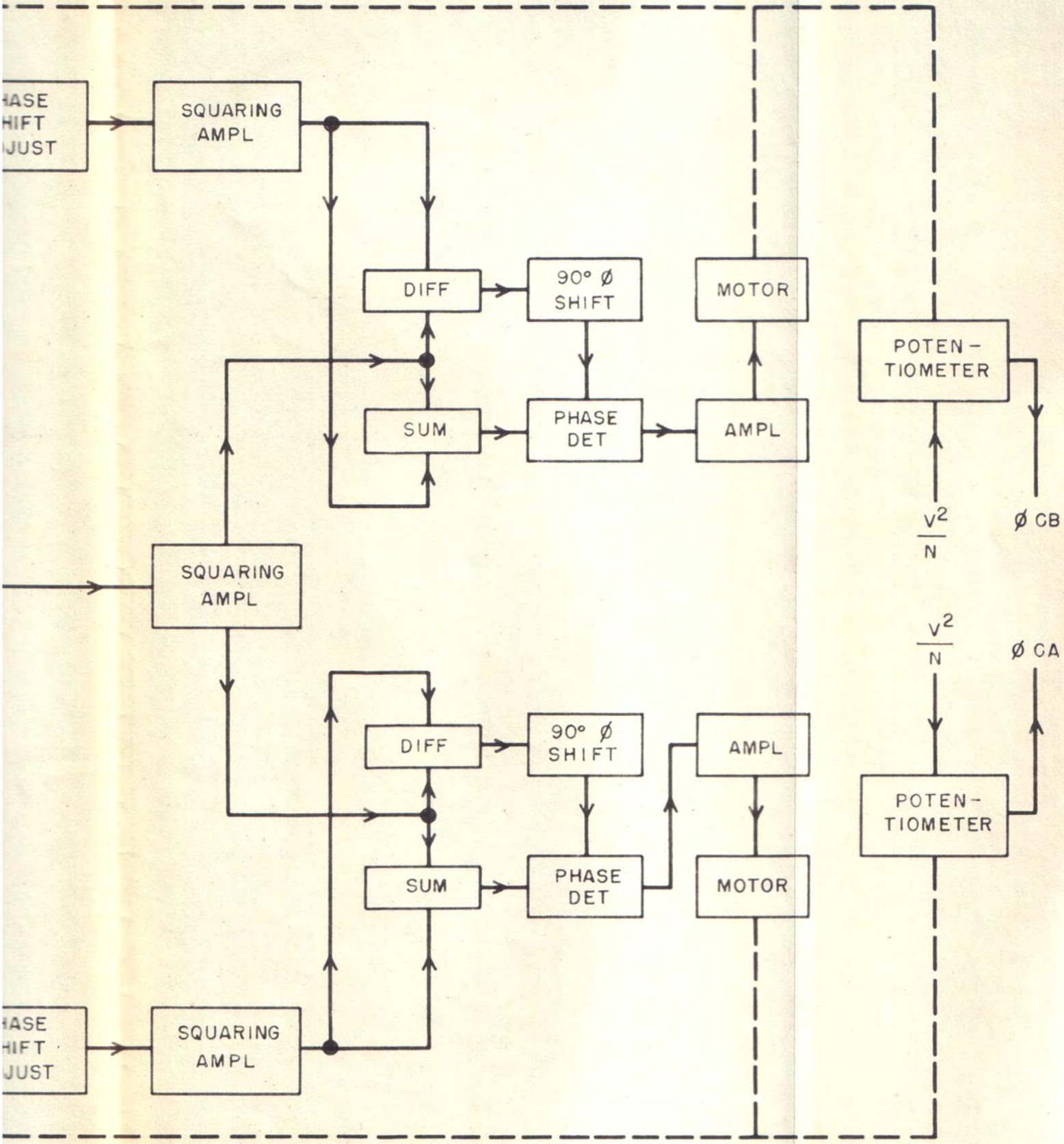


FIGURE 4. PHASE



COMPARISON SYSTEM, BLOCK DIAGRAM


HASE
HIFT
JUST
SQUARING AMPL
DIFF
90° Ø SHIFT
MOTOR
POTEN-TIOMETER
SUM
PHASE DET
AMPL
V²/N
Ø CB
SQUARING AMPL
V²/N
Ø CA
DIFF
90° Ø SHIFT
AMPL
POTEN-TIOMETER
SUM
PHASE DET
MOTOR
HASE
HIFT
JUST
SQUARING AMPL

-6

phase comparison antenna, is beat with a 125.502 MC local oscillator. The two local oscillators are maintained at a 2 KC separation by comparing their beat note with a 2 KC standard. Three IF signals result from beating the two local oscillators with the three antenna signals, two at 0.500 MC and one at 0.502 MC. The 0.502 MC signal is split and added to each of the 0.500 MC signals. These sum signals are amplified in two high gain IF amplifiers, each having a bandwidth of approximately 20 KC. Using a common amplifier for each pair of signals minimizes the differential phase shift between signals which are to be phase compared. The output of each IF amplifier is fed into a square law detector. The phase between the resulting 2 KC beat note and the 2 KC signal resulting from the beating of the two local oscillators, is the same as the phase existing between the RF signals received at the antennas.

The output of each square law detector is passed through a filter, a rotary phase shifter of the capacitor type, an adjustable phase shift and a squaring amplifier which converts the 2 KC sine wave into a 2 KC square wave. The reference 2 KC signal resulting from the beating of the two local oscillators, is also converted to a square wave. The reference is split and then added to and subtracted from each of the other 2 KC signals. The two sets of sum and difference signals are then compared in phase detectors. The output of the phase detector, which is proportional to the difference signal, turns the rotary phase shifter through an amplifier and motor. The phase shifter rotates until the difference signal is reduced to zero. The amount of rotation is proportional to the phase difference between the 2 KC ref-

[redacted]

erence and the beat note out of the square law detectors. The shaft position of the phase shifter is converted to a voltage by means of a linear potentiometer. The output voltage of the potentiometer will vary linearly from 0 to a maximum as the RF phase difference between antennas varies from 0 to 2 π, and will be direct by proportional to ϕe.

(2) Accuracy and Range.

The phase comparison system can measure the electrical phase angle between two antennas to within approximately 3.5 degrees at a range of 3000 miles. The calculations made to determine this range are given in Appendix 5. Since the phase comparison system is not required to track smoothly at ranges greater than 1200 miles, there is a considerable safety factor. This is the maximum distance of the satellite from one location when it is being simultaneously tracked from two locations for triangulation purposes.

(3) Siting of Phase Comparison Antennas.

The siting of the antennas to minimize the effect of ground reflections is extremely important to the accuracy of the system. If possible, the antennas should be located behind screening crests, either nature or man made. Also the ground within 50 to 100 feet of the antennas should be of poor conductivity.

(4) Calibration of Phase Comparison Antennas.

The problem of proper calibration of the phase comparison antennas can be divided in two parts. One is the actual orientation of the line between phase centers and the other is the measurement of the distance in wavelengths between phase centers. If the antennas and psdestals are accurately made, the position of the phase

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centers with respect to a given point on the pedestal should not vary by more than 1/8" between pedestals. The actual position of the phase center need not be known with this accuracy. The orientation of the line between phase centers then reduces to a problem of positioning the given points on the pedestals along a known line. The distance in wavelengths between phase centers can be determined electrically by means of a transmitter located along the line of phase centers. Since considerable power is available from such a transmitter, the electrical angle between antennas can be measured to considerably better than one electrical degree.

d. Local Data Processing.

The tracking radar at each location will provide course azimuth and elevation angle data for positioning the three antennas used in the phase comparison system. This information will be transmitted by a synchro system. In addition, sine-cosine potentiometers mounted on the tracking radar pedestal and driven by the azimuth and elevation axis will provide d.c. voltage data to a coordinate converter.

The output from these potentiometers will be used in the coordinate converter, as shown in Figure 5, to calculate $V^2 \cos \theta_{1A}$ and $V^2 \cos \theta_{2A}$, assuming that the tracking radar is oriented in azimuth with the base line formed by taking one pair of the three phase comparison antennas. (See Figure 6).

The phase comparison antennas will be used in pairs, as explained in paragraph 2b, to obtain the electrical phase angle difference ϕ_e in terms of voltage and shaft position. The value of ϕ_e will vary from 0 to 2π. Simple analogue computers will combine $V^2 \cos \theta_{1A}$

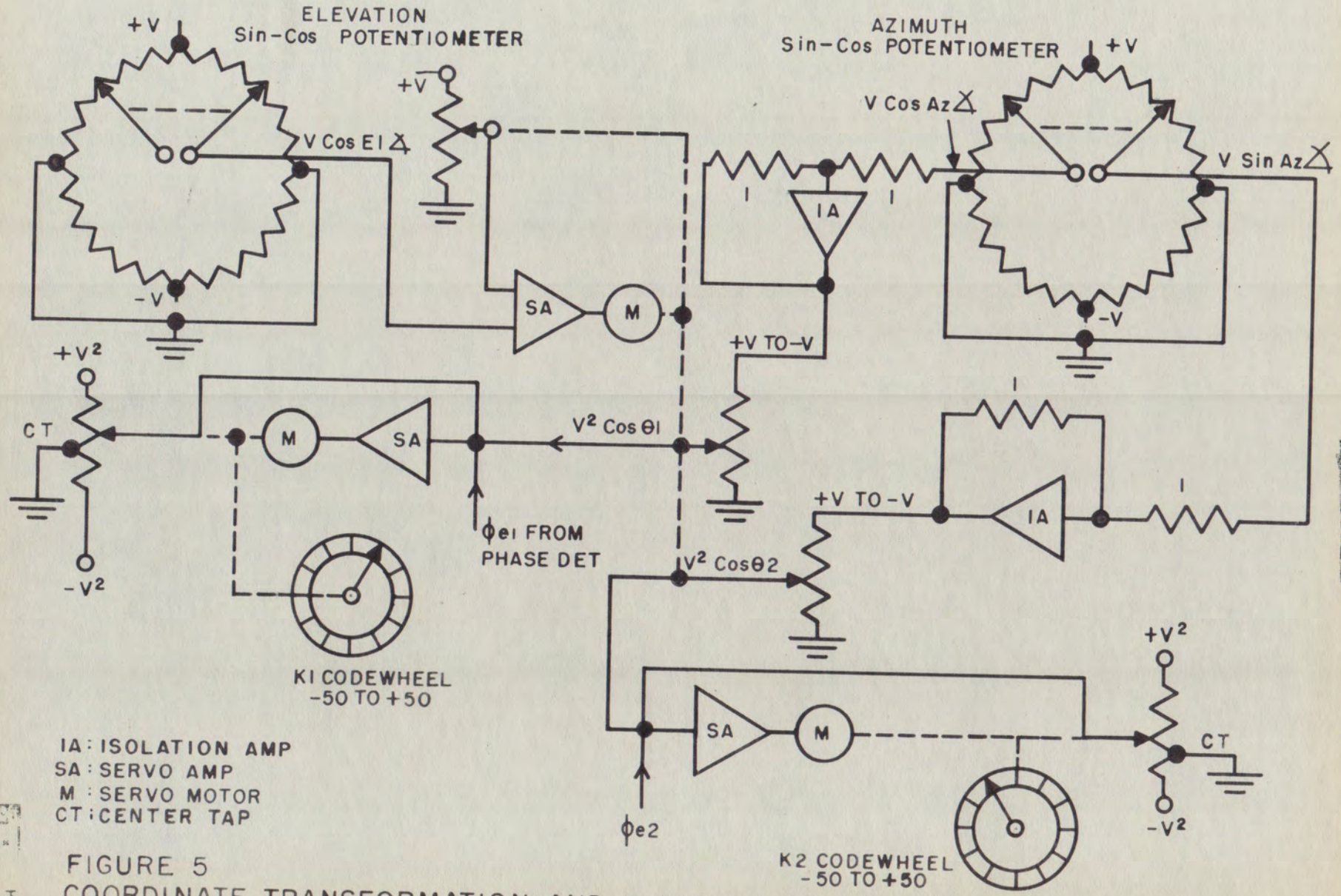
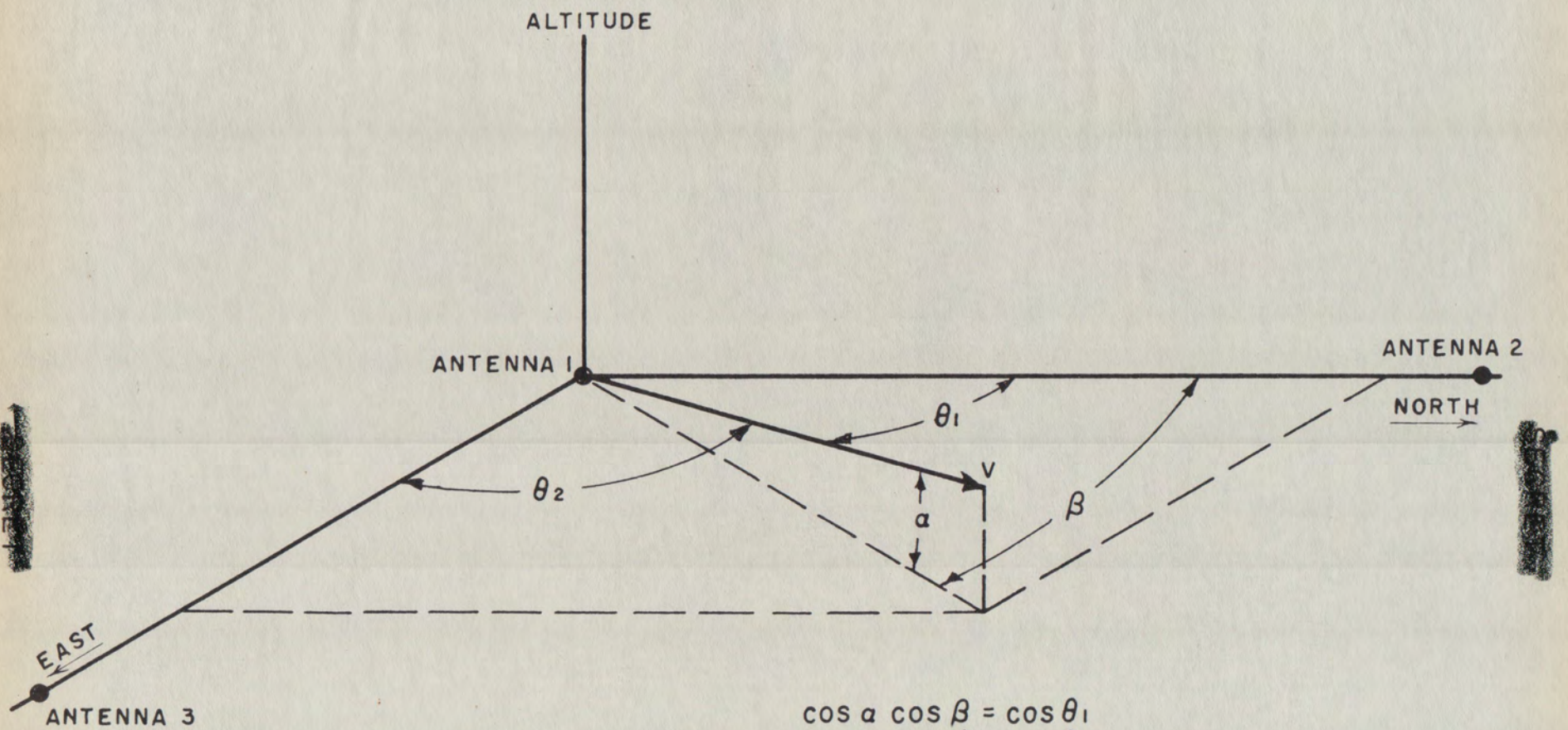


FIGURE 5

FIGURE 5
COORDINATE TRANSFORMATION AND K SERVOS REQUIRED AT EACH LOCATION
SIMPLIFIED SCHEMATIC DIAGRAM



$$\cos\alpha \cos\beta = \cos\theta_1$$
$$\cos\alpha \sin\beta = \cos\theta_2$$

FIGURE 6 SINGLE LOCATION ORIENTATION AND COORDINATE TRANSFORMATION

and ϕ_{e1A} and V^2 Cos Θ_{2A} and ϕ_{e2A} to solve the following equations:

$$\phi_{E1A} - 2\pi M \cos \Theta_{1A} + 2\pi K_{1A} = 0 \qquad (1)$$

$$\phi_{E2A} - 2\pi M \cos \Theta_{2A} + 2\pi K_{2A} = 0 \qquad (2)$$

for K_{1A} and K_{2A} to the nearest integers. Knowing the K's accurately to the nearest integer and the ϕe's to $\pm$ 3.5 electrical degrees, a digital computer could re-solve equations 1 and 2 for Θ_{1A} and Θ_{2A} with an rms error of $\pm$ .3 angular mils. (See discussion of accuracy, Appendix 1).

It is proposed to perform the final calculations at a centrally located computing site. This site could utilize any one of several commercially available general purpose digital computers.

ϕe can vary from 0 to 360 degrees and is accurate to $\pm$ 3.5 degrees. If ϕe were quantized into 512 parts and the quantizing error of $\frac{360}{512}$ = .703 degrees is combined in an rms manner with the expected noise error of 3.5 degrees, the resulting error will be 3.57 degrees or a negligible increase over the expected noise error. The value of K can vary from -50 to + 50 and must be accurate to the nearest integer. In binary form K can be represented by 6 bits plus 1 bit for sign. Thus the accuracy of the system can be preserved if nine bits of information are used to represent ϕe and seven bits are used for K.

The following table illustrates the amount of data available at each location:

Quantity	No. of bits required
K_{1A}	7
K_{2A}	7
ϕe_{1A}	9

Quantity	No. of bits required
ϕe_{2A}	9
Total	32

The K servos and the ϕe servos will drive code wheels. These code wheels are commercially available and can be sampled many times per second. Refer to Figure 7. The code wheels could be sampled simultaneously at periodic intervals and the results stored in registers. A programming device such as a diode matrix or a ganged stepping switch could be used to read out the stored data in samples of five bits at a time and these samples could be made to drive a teletype tape punching machine. These machines can operate at ten five-character words per second giving the system a capacity of 50 bits per second. If the code wheels were sampled at one second intervals, the data would be accumulated at a rate of 32 bits per second, which is compatible with the capacity of the tape punch machine. Thus, the tape would provide a permanent record of the data for future reference and at the same time could be used to drive a one channel teletype transmitter to relay the information to the central computer.

Because each location only measures the direction of the satellite in terms of two angles, two locations are required to triangulate and determine the position of the target. See Figure 8. As will be shown later, it is proposed that for an equatorial orbit, a station should consist of two locations, approximately 500 to 800 miles apart and that the RADIV System should consist of two stations. For an inclinted orbit, the situation is slightly more complex, but for any one

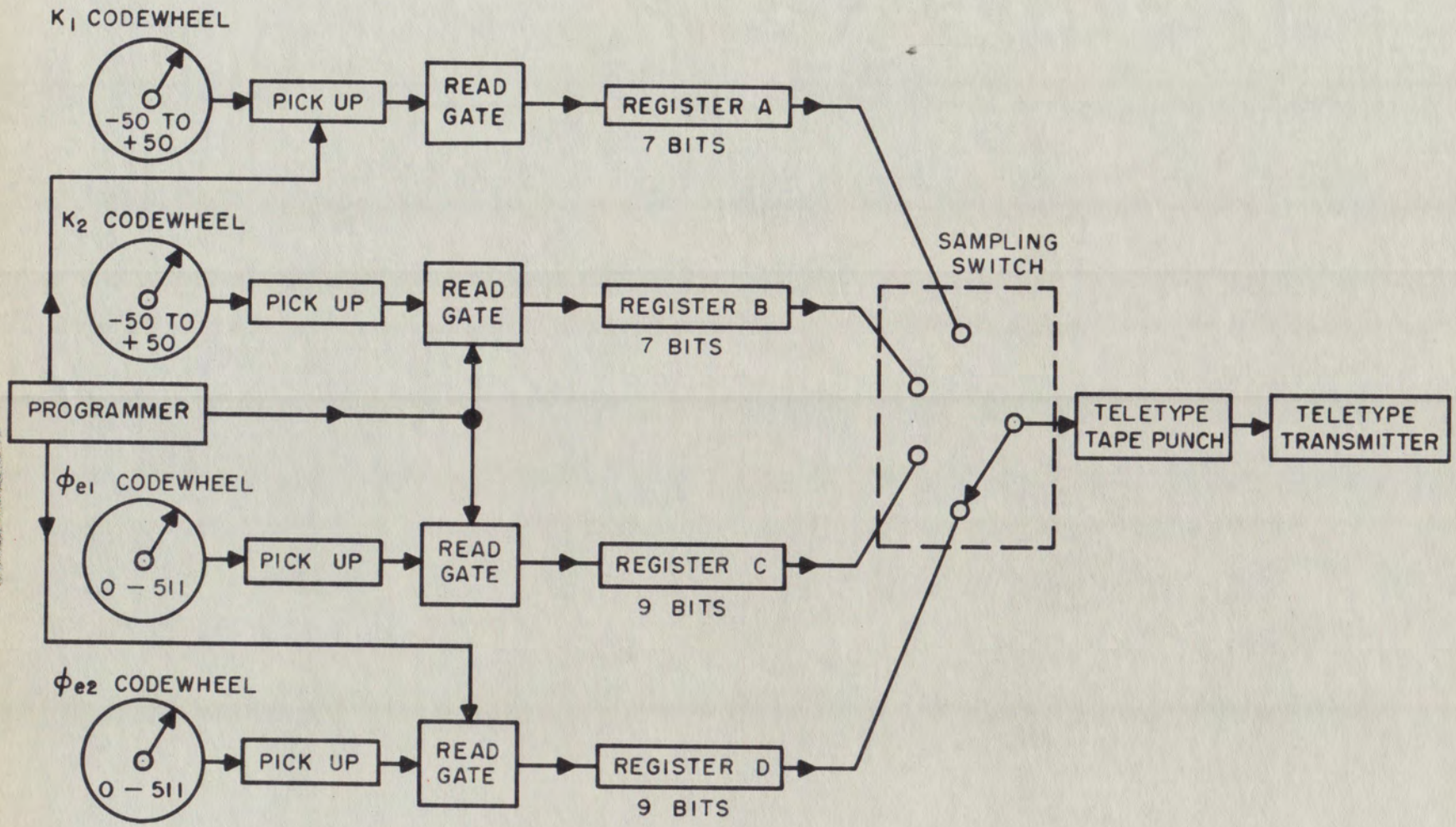


FIGURE 7
METHOD OF DATA SAMPLING & TRANSMISSION AT EACH LOCATION, SIMPLIFIED BLOCK DIAGRAM

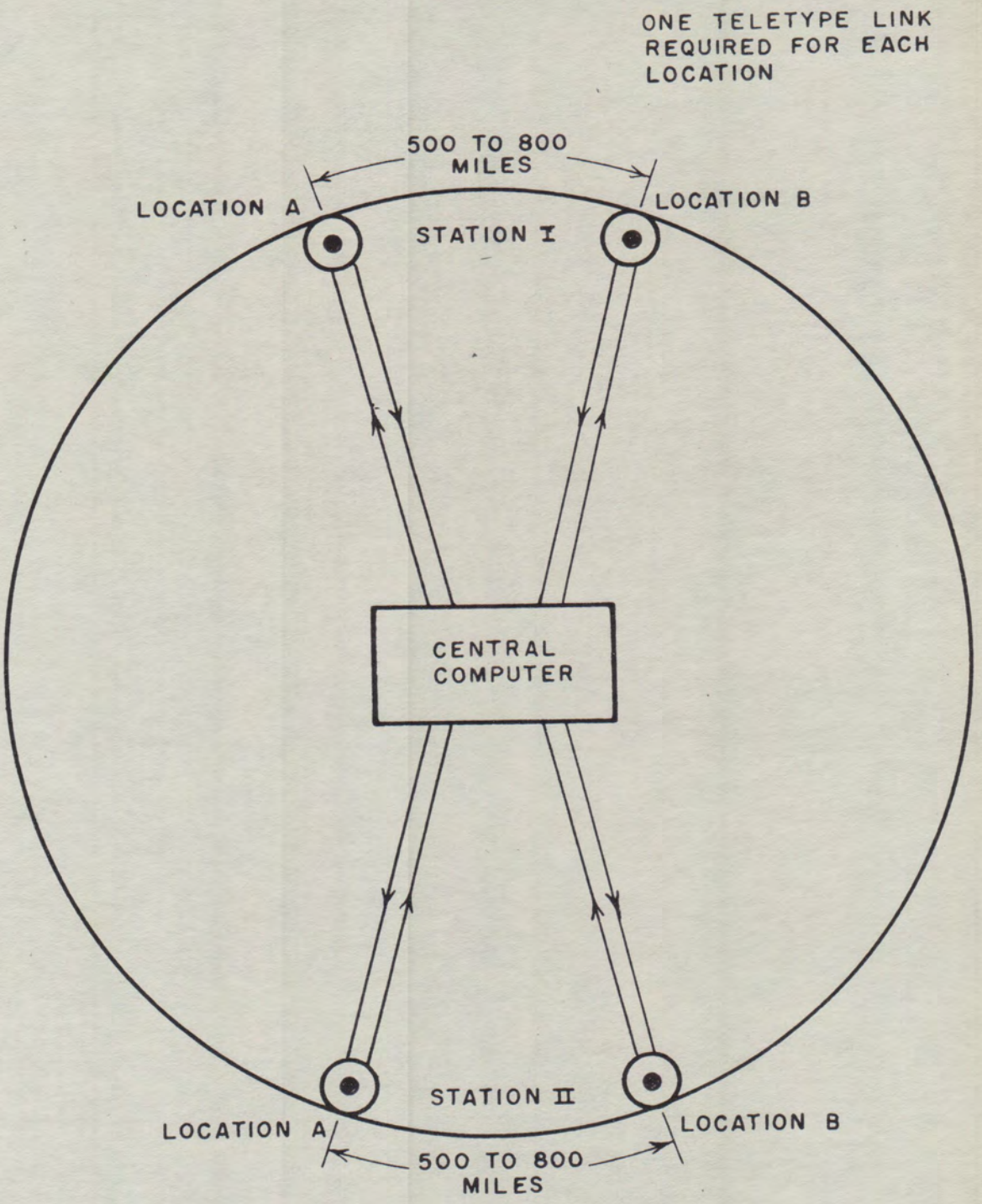


FIGURE 8. TWO STATION—FOUR LOCATION SYSTEM WITH CENTRAL COMPUTER

3-11

trip around the earth, there will only be two stations, each consisting of two effective locations. In order to eliminate errors in the triangulation, the data from one station must be sampled simultaneously at each location. It is proposed to accomplish this by having a 100 kilo cycle/second crystal controlled clock at each location. These clocks will be synchronized weeks in advance of proposed flights by an expert with the aid of various time signals available throughout the world. These clocks can be made accurate and stable to one part in one million. Synchronization between any two locations and the central computer can be obtained by transmitting a train of pulses one second apart from location A to location B, this train to start as soon as tracking has become stable at station I. This train of pulses would be derived by dividing down the clock frequency standard. At station II, the received train of pulses from station I is passed through a switch and retransmitted to the central computer. This switch is normally open and is closed when the operator at location B determines that tracking is satisfactory at location B. Thus the central computer receives a train of pulses one second apart when location A and B are both tracking. The occurrence of this train signals the central computer to start reading data from station I. Thus the basic accuracy of the system depends only on the accuracy of the local clock at each point and the variables of transmission time do not enter into the problem. A more sophisticated system of coding may be arranged, but the basic principles are valid.

3. DETAILED SYSTEM DESCRIPTION - AIR SYSTEM

a. General

The RADIV System provides for fabricating the satellite in the form of a 20-inch dielectric sphere coated with a suspension of titanium dioxide to provide good optical visibility and reduce ultra-violet decomposition of the polyester resins, within which a 125 MC transmitter, power supply and antenna are inclosed. The antenna will be an equivalent dipole structure along perpendicular to the axis of spin. The transmitter and power supply will be housed in an hermetically sealed and heat shielded metallic cylinder, mounted rigidly on the axis of spin. The weight of the whole satellite has been estimated to be a minimum of approximately 6 pounds, distributed as follows:

Batteries	1½ pounds
Transmitter and Housing	½ pound
Antenna and Structural Members	1½ pounds
Fibreglass Shell	2½ pounds

b. Transmitter

The transmitter contained in the satellite must be designed primarily for minimum power drain. System considerations dictate that minimum power delivered to the antenna is 25 milliwatts. The frequency chosen of 125 MC limits the oscillator to a sub-miniature tube, one of several which are standardized and available in quantity, or the experi-

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3-13

mental PNIP Intrinsic Barrier transistor which would require approximately one year development effort. From the sub-miniature tubes, types 5677, 5971, or 6611, the following characteristics obtain:

Output Power	30 Mw
Filament and Plate Input Power	150 Mw
Overall Efficiency	20 %
Life per Pound of Batteries	300 hours maximum
Tube Life	500 hours

The PNIP transistors, after a years development effort, should produce the following characteristics:

Output Power	35 Mw
Input Power	120 Mw
Overall Efficiency	25 %
Life per Pound of Batteries	375 hours maximum
Transistor Life	Unlimited

It is felt that if batteries are used which will weigh 1½ pounds total, it is best at this stage to use the sub-miniature tube, since there are no development problems involved, and that tube life should exceed battery life. These tubes have been given a preliminary mechanical test to include 1000 G steady acceleration from various directions and have shown no failures. Figures on efficiency are the results of laboratory tests on a few samples of tubes.

The oscillator circuit will be a simple crystal controlled one-stage oscillator, from which we expect frequency stability of 1 part

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in 10^5 if the temperature swing inside the transmitter housing is between 0 and 10 degrees C. This temperature control will be effected by heat shielding in the transmitter housing in order to retain heat from the tube.

As an alternate recommendation for the transmitter in the event that solar cells are available for the power source, the PNIP transistor would be used because of its longer life, and lower operating voltage.

c. Power Source

The power source recommended, which is presently available and offers greatest efficiency per pound, is the High-rate Zinc-Silver Oxide cell, which is available in 1½ volt increments for the "A" batteries and 30 volt increments for the "B" batteries. Each type of these cells comes in a variety of sizes, so that there should be no problem in selecting those to give the required voltage and maximum possible life per pound. 1½ pounds weight are allocated in the RADIV System for the batteries. At the maximum life, this corresponds to 67 watt-hours total. These batteries maintain good regulation of voltage for temperatures between 0 and 120 degrees F. This temperature range should be suitable, provided the temperature control cited above can be met.

The solar cells may be available in time to use on this development. This is considered to be an alternate recommendation to batteries with the provision that these cells would be used with

the transistors rather than the sub-miniature tubes. These cells in their present form could supply some 40 volts from 125 individual cells and is estimated to weigh 3/4 of a pound. An additional 3/4 of a pound for a storage cell to enable the satellite to transmit in darkness would provide a long-life power supply for the transmitter. For details of such type power supplies reference is made to paragraph II.3.d. Power Sources.

4. MULTIPLE STATION CONSIDERATIONS:

a. Site location for 2 orbits.

(1) General Considerations:

A site location consists of three phase comparison antennas for fine tracking, and a course tracking radar to resolve ambiguities. Since altitude information is obtained by triangulation, two locations are required, and therefore two locations comprise a station. In general, the locations will be situated 800 statute miles apart, thus permitting triangulation for up to 960 miles of the orbit when it is at 800 miles altitude. There is a restriction on the length of tracking data, due to the earth's curvature. The course tracking antennas cannot be pointed less than 15° with the horizontal because of ground reflections. This combination of effects is shown in Figure 9

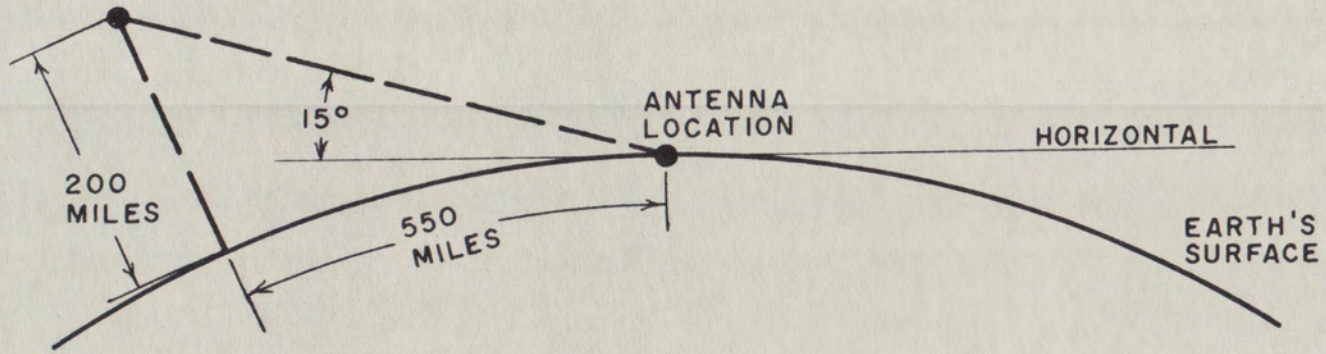

FIGURE 9. TRACKING DISTANCE FOR LOW ALTITUDE

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and means that when the satellite is at perigee, i.e., 200 miles altitude, and the antenna is pointed at 15°, the satellite cannot be seen further than approximately 550 miles away from the location of the antenna. This distance is measured on the earth itself and is not the slant range. Reference Figure 10. The graphs show the angle of elevation for an antenna, when it is looking at the target for 200 and 800 miles altitude and the target is directly over another location, as a function of the distance between locations. It can therefore be concluded that the triangulation can occur for all stations, as planned, when the satellite is at 800 miles, but that the locations have to be situated closer than 562 miles for the satellite at 200 miles in order to obtain triangulation for the entire distance between locations.

It is necessary to obtain information about the orbit at a minimum of two points, in order to accurately predict the future path and it is also desirable to situate these points as far apart as possible. As a consequence of the above requirement, two stations are suggested for each orbit.

The locations specified are only approximate and the final decision as to the permanent location will depend on the general topography, climate, logistics, diplomatic considerations, and such environmental factors that will preclude the success of the radar installation.

It is also suggested that optical tracking equipment be slaved to the coarse tracking radar to receive positioning information. Consideration has been given to the fact that optical seeing ability is very poor in the tropics, but as one place is about as bad as

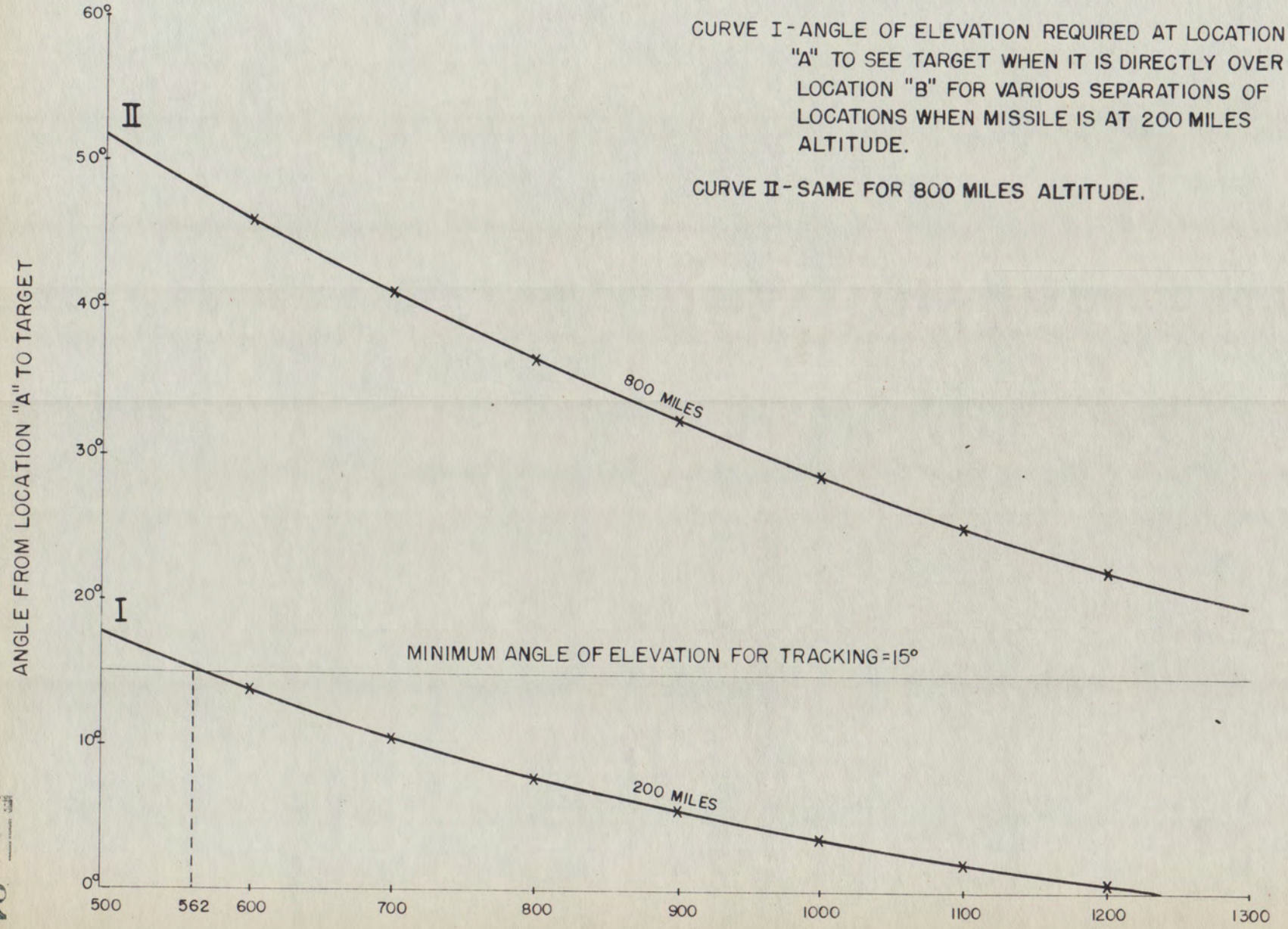

FIGURE 10. DISTANCE IN MILES FROM LOCATION "A" TO LOCATION "B"

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another, and the optical equipment is readily portable, the locations have been determined from the radar point of view.

It is considered mandatory that the satellite be tracked as soon after launch as possible, and hence a station is located to the east of the launching site.

(2) Equatorial Orbit.

Present information indicates that if an equatorial orbit is chosen, the launching site will be located in the Gilbert Islands (Brit.) in the Pacific Ocean. The satellite will pass over every point on the equator 16 times during the day and the perigee and apogee will fall back along the equator $22\frac{1}{2}^{\circ}$ in each revolution because of the earth's own rotation. There will be other perturbations in the orbit, but these will be relatively small and the actual computations will be left to other sources.

In view of the foregoing, several of the small islands near the Gilberts suggest themselves as possible location sites. Tarawa (Brit.) and Baker (U.S.A.), which are separated by approximately 738 miles, would form a good station. If Baker and Jarvis (U.S.A.) are chosen, the amount of triangulation is reduced to about 500 miles and there will be no triangulation when the satellite is at 200 miles altitude, because the distance between locations is approximately 1135 miles.

The second station, situated about 7000 miles away, would consist of a location at Chavis Island in the Galapagos Islands (Ecuador) and a location at Salinas (Ecuador) on the coast, or Quito (Ecuador) which is inland at an altitude of about 10,000 feet. The dis-

18 tance from Chavis Island to Salinas is 656 miles and from Chavis Island to Quito is 818 miles.

The triangulation area in the latter case is reduced by about 150 miles because of the greater distance between Chavis Island and Quito. In all of the triangulation considerations, 800 miles is considered as the maximum reliable range for stated accuracies, but it should be understood that position data is obtained before and after the triangulation period for about 550-800 miles, depending upon the altitude of the satellite. Reference Figure 11. Using position data, angular velocity and acceleration, and Newton's laws of gravitation, additional useful information about the orbit can be obtained. This means that for two stations, good position data will be received for about 20% of the orbit from electronic means alone.

Consideration has been given to placing the second station about 180° away from the launching site or approximately on the west coast of Africa. Unfortunately, this would necessitate the placing of a location about 500 miles inland in French Equatorial Africa, or else placing a location at about 5° N latitude in order to remain on the coast. A similar situation exists on the east coast of South America and on the east coast of Africa. Any location in the East Indies would be less than 100° away from the launch site, and therefore would not be as desirable as the Chavis Island - Salinas location. Other locations along the equator do not appear feasible. See Figure 12.

(3) <u>30° N-S Orbit.</u>

Another proposed orbit which has been under consideration is that which would result if the satellite were launched at Pat-

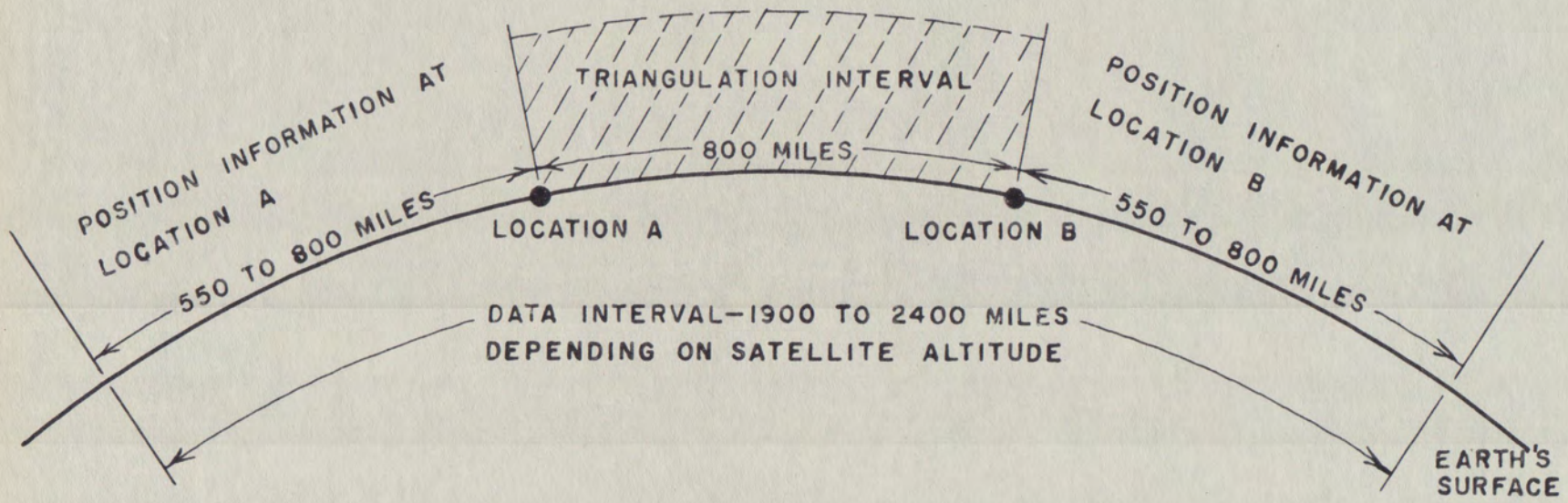

FIGURE 11 RANGE COVERAGE FROM ONE STATION

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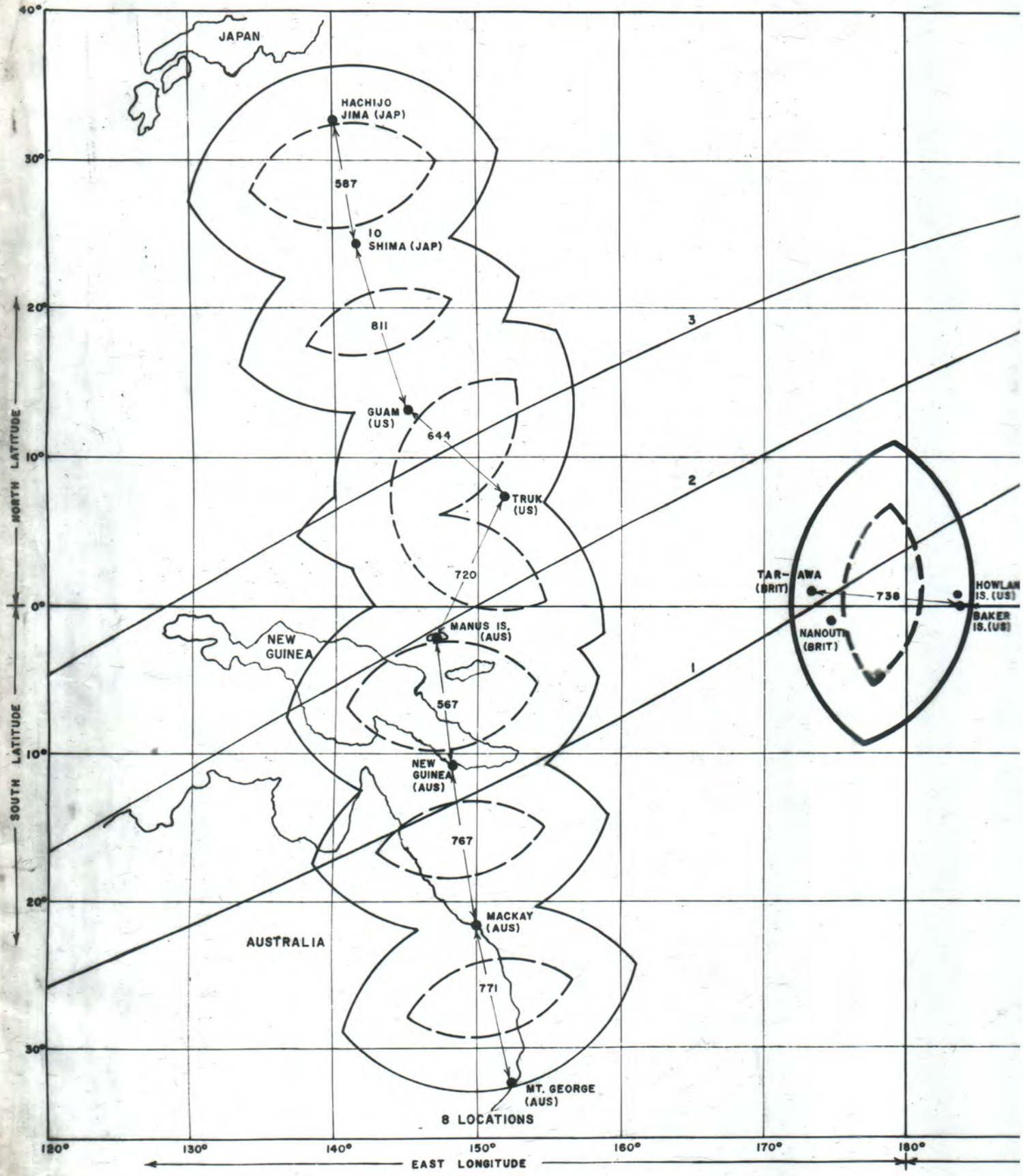
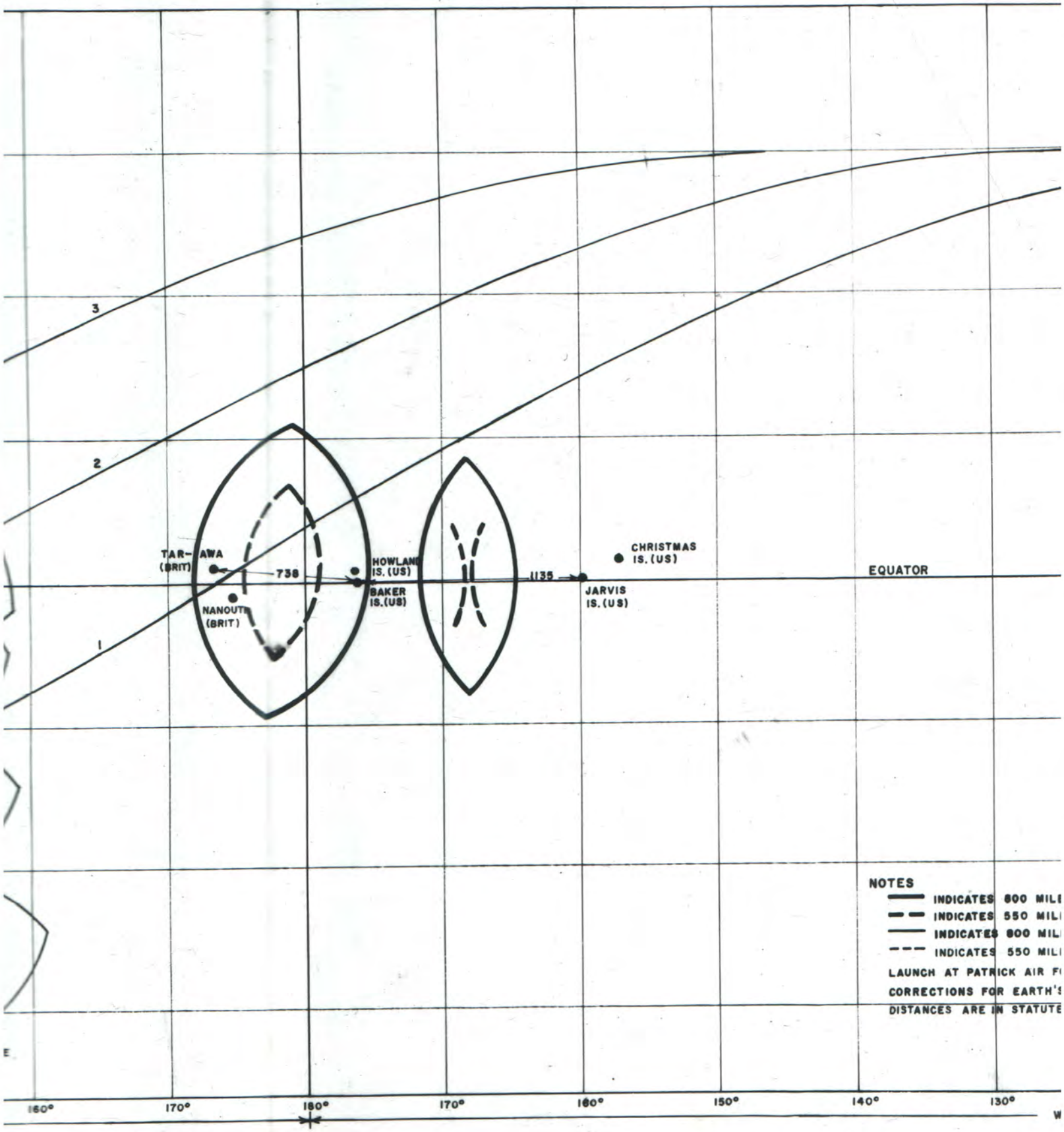
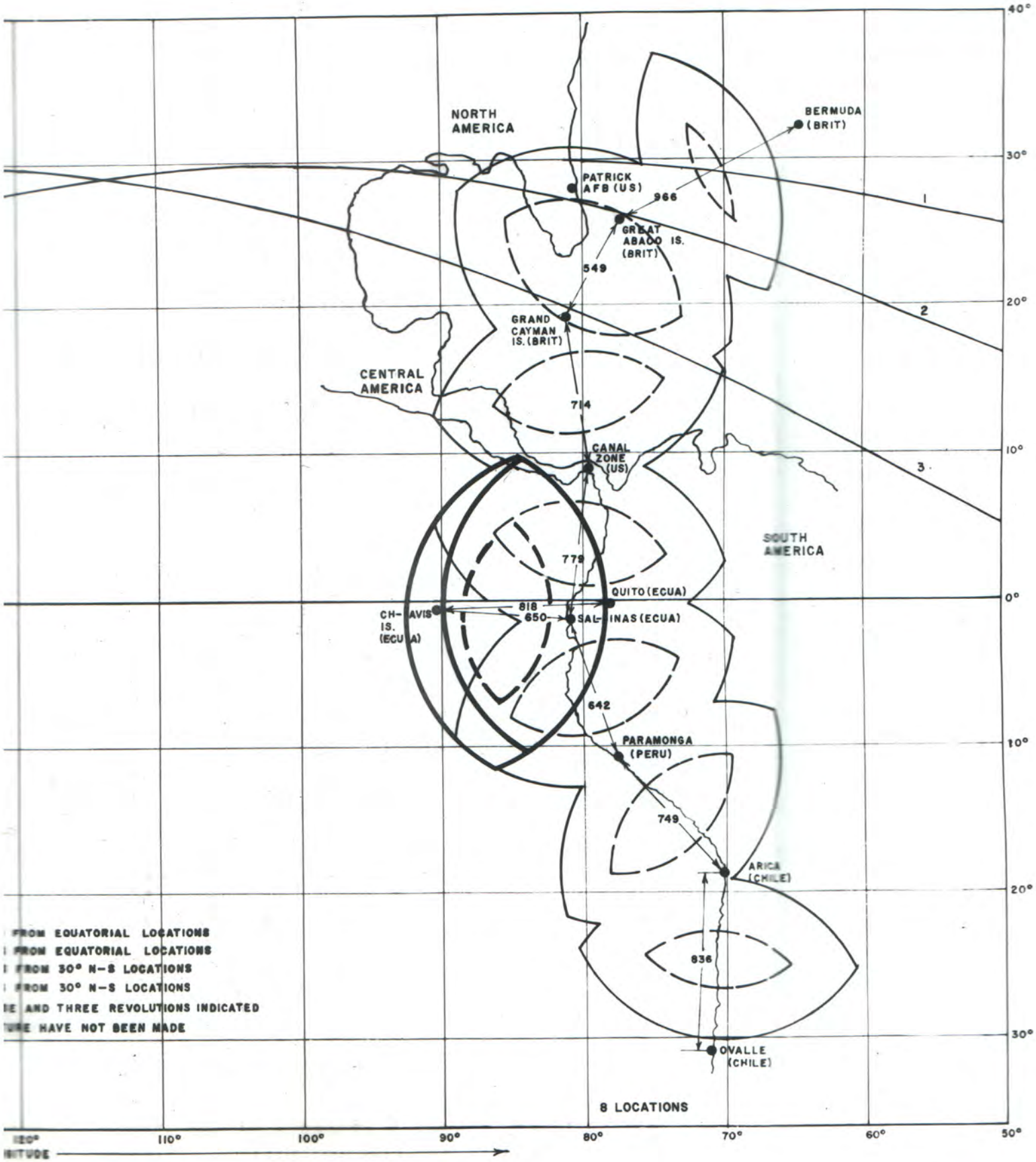


FIGURE 12 PICTORIAL PRESENTATION OF SITE LOCATIONS FOR TWO ORBITS


3
2
1
TAR-AWA
(BRIT)
NANOUTI
(BRIT)
736
HOWLAND
IS. (US)
BAKER
IS. (US)
1135
JARVIS
IS. (US)
CHRISTMAS
IS. (US)
EQUATOR
NOTES
INDICATES 800 MILE
INDICATES 550 MILI
INDICATES 800 MIL
INDICATES 550 MILI
LAUNCH AT PATRICK AIR F
CORRECTIONS FOR EARTH'S
DISTANCES ARE IN STATUTE
160°
170°
180°
170°
160°
150°
140°
130°

BITS


NORTH AMERICA
BERMUDA (BRIT)
PATRICK AFB (US)
966
GREAT ABACO IS. (BRIT)
549
GRAND CAYMAN IS. (BRIT)
CENTRAL AMERICA
714
CANAL ZONE (US)
SOUTH AMERICA
779
QUITO (ECUA)
818
650
SAL- INAS (ECUA)
CH- AVIS IS. (ECU A)
642
PARAMONGA (PERU)
749
ARICA (CHILE)
836
OVALLE (CHILE)
8 LOCATIONS
1
2
3
40°
30°
20°
10°
0°
10°
20°
30°
110°
100°
90°
80°
70°
60°
50°
FROM EQUATORIAL LOCATIONS
FROM EQUATORIAL LOCATIONS
FROM 30° N-S LOCATIONS
FROM 30° N-S LOCATIONS
AND THREE REVOLUTIONS INDICATED
HAVE NOT BEEN MADE

[redacted]

[redacted]

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tick Air Force Base (U.S.A.) approximately 30° N Latitude, 80° W Longitude. The satellite would then oscillate between 30° N latitude and 30° S latitude and because of perturbations would in time pass over every point between these two latitudes. Therefore, in order to ensure that the satellite be seen on every revolution, it would be necessary to place, on a particular longitude, a series of locations stretching 4000 miles from 30° N to 30° S latitude. It is also possible to obtain complete coverage by placing locations from the equator to 30° N in two places, separated by 180°. This would necessitate more stations because of two extra end points, besides being difficult to find land bases with the proper spacing. Fortunately, it is possible to place such a series of locations from Bermuda (Brit.) to Ovalle (Chile) and obtain data on each revolution. Additional information can be obtained from a series of locations stretching from Hachijo Jima (Jap.), an island near Japan, to Mount George (Aus.) a location on the east coast of Australia. The distance between these two belts is about 135° or about 9300 miles. In each series there are 8 locations necessary to cover the band so that triangulation is obtained twice on each revolution.

Figure 8 is a pictorial presentation of the location of the radars for both the equatorial and 30° N-S orbits. The large circles indicate 800 mile range and the circles within the large circles indicate the 550 mile maximum range for 200 miles altitude. The triangulation area is indicated by heavy lines. Parts of the first three orbits after launch have been drawn on the chart. Because of the earth's rotation, the satellite does not return to the launching point, but slips back 22½°. Hence, the satellite will pass over the launching

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point once during the first day. Because of other perturbations, it may not return to the exact launching point for many days. However, for a location on Great Abaco Island (Brit.) in the Bahamas, the satellite would be seen no less than 6 times a day. Certain stations located near the 30° N or S parallel might see the missile 8 times a day. For stations near the 20° N or S parallel, the rate of seeing the missile would be 4-6 times a day, whereas for stations between 10° N and 10° S the missile would be seen 4 times a day. For a table of latitude crossings of the satellite on a particular longitude and formulas to be used in computing these latitudes, see Appendix 6.

The reason for the higher incidence of visibility at the higher latitudes is because the projection of the orbit on the surface of the earth is approximately a sine wave, and the sine wave changes less rapidly and hence is flatter at the higher latitudes. This means that by judiciously placing the locations in the higher latitudes, a sufficient number of orbits can be plotted to give satisfactory prediction data and a large financial saving could be effected. This would especially be true if the tracking system and satellite had a long life, in the order of months, during which time accurate data could be gathered.

b. Survey Considerations.

The accuracy of the entire system depends upon the accuracy with which long distances can be measured between locations and also upon the precision with which the phase comparison antennas can be located. In general, accuracies have been achieved, by taking measurements over extended periods, that are far greater than will be required to locate

[redacted]

the radar sites and to orient the antennas. However, these measurements have, in general, been made on large land masses and in territories that have been previously surveyed, at least to a third order of accuracy. It is anticipated that there will be many places that are void of bench marks and other well known reference points. For example, it was stated on the map* that was used for Tarawa that the position of the islands was in accordance with the best available data, but that the true position may vary by as much as 10 miles.

However, it was stated in Breed and Hosmer, "Higher Surveying", that latitude may be determined by astronomical methods to a probable error of 0".10 of arc, which corresponds to 10 feet on the earth's surface. The observing period necessary to obtain this accuracy is one night. Longitude can be determined by comparing local sidereal time with a known standard, e.g., W.W.V. The longitude can be determined nearly as accurately as the latitude.

The location of the site can also be computed by triangulation methods from known points. Over large distances the differences between the values obtained by the astronomical method and the triangulation method is of the order of 200 to 300 feet. This is known as the station error and is caused by the earth's deviation from a geoid. However, this station error can be practically eliminated. It is to be noted that 300 feet in 800 miles is an error of the order of 1/14,000, which is better than a second order survey.

Using the same reference, it was stated that the lengths of baselines, of the order of several kilometers, could be measured to

* World Aeronautical Chart 7/48 Map #872 - Tarawa Atoll

[redacted]

better than $1/10^6$. The accuracy required in measuring the length of the baseline between phase comparison antennas is 1/10" in 400 feet or $\frac{1}{50,000}$. This appears feasible in light of the previous statement. Angular measure can have a precision of 0.2 seconds of arc, and thus the orientation of the antenna should present no problem, but will require great care in mounting and stabilizing.

c. Acquisition.

(1) General Considerations.

The coarse tracking radar consists of two crossed rectangular arrays 5' wide and 25' long, situated so that one tracks in azimuth and the other tracks in elevation. The beamwidth corresponding to the five foot dimension is 80° and that corresponding to the twenty-five foot dimension is 20°.

In the search phase, the antennas will look along a $5-10^\circ$ angle of elevation. The lobe switching mechanism will be in operation. When the target is intercepted by the radar, it will first be tracked manually and when centered in both beams, the automatic tracking units will be turned on.

(2) Equatorial Orbit.

Along an equatorial orbit, it would not be necessary for either antenna to scan for search purposes. The azimuth antenna centered on the equator will cover either side of the equator for 20°. The elevation antenna centered on the equator will overlap 40° on either side. If the antennas are now pointed in the direction of expected arrival (due West) of the satellite, at a $5-10^\circ$ elevation angle, then the area of interception at 1000 miles will be a vertical square 700

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miles on a side.

If the launching is within 5° of the equator, the satellite will pass through this 700 mile square. It is therefore concluded that each station on an equatorial orbit will acquire the satellite on each revolution without scanning of the antenna.

(3) 30° N-S Orbit.

In order to acquire the satellite on each of the 4 passes that occur during the day, the antennas will have to scan 180°. However, with prediction of the orbit available, it will be possible to orient the antenna to within $\pm 20^{\circ}$ of the expected path. This will permit acquisition without scanning of the antenna as in the equatorial orbit.

d. Interference Considerations.

The frequency chosen for this system is approximately 125 mc/s. Because of the high communications activity in this region, it is highly probable that a good deal of interference will result from external sources. Consideration must be given to the choosing of an operating frequency in this range that will not be affected by television or short wave broadcasts. It is believed that such a frequency can be found without too much difficulty.

5. PROGRAM CONSIDERATIONS

One of the governing factors in the consideration of possible electromagnetic tracking systems for this application was the time required for development, installation and test. Assuming a target date of 30 September 1957, and that an authorization to proceed is received by 1 November 1955, some twenty-three months are available for design

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of the equipment, its fabrication, installation and test.

Assuming that a contract can be let on 1 January 1956, it is believed that the first location can be installed and tested in eighteen months from the award of a contract, or by 30 June 1957, and the remaining locations by 30 September 1957.

It will be recalled that for an equatorial orbit, two stations or four locations are proposed. Development cost of the first station (two locations) is estimated at $1,000,000.00 and fabrication cost for each succeeding station $500,000.00. Thus for an equatorial orbit, a total of $1,500,000.00 is estimated to be required.

For a 30° orbit, eight stations consisting of sixteen locations are proposed. Estimated costs here are again $1,000,000.00 for the development, installation and test of the first station, and $500,000.00 for each succeeding location, or a total of $4,500,000.00.

Manpower estimates for an equatorial orbit are FY 56, 500 man-days; FY 57, 2000 man days; FY 58, 500 man days. Travel costs are estimated at $6000.00.

Man power estimates for a 30° orbit are: FY 56, 500 man days; FY 57, 4000 man days; FY 58, 1000 man days. Travel costs are estimated at $15,000.00.

It should be noted that the above estimates do not include logistical considerations such as housing, administrative, communications, etc., nor do they include manpower estimates for operation.

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APPENDIX I

The following chart is a list of the errors considered in connection with the RADIV System.

The errors were divided into two categories, systematic errors that are correctable, and random errors that are essentially uncorrectable. It is assumed that the rms value of the systematic errors can be corrected to 10% of the uncorrected value. The final system error is then computed as the rms of the corrected systematic errors and the random errors. For the case where $\theta = 15$ degrees, the system rms angular error is .83 mils, and for the 45° case the rms angular error is .30 mils.

When the satellite is 400 miles above the base line between locations A and B, and θ at each location is 45 degrees, the rms position error, assuming an angular rms error of .3 mils in each θ, can be determined as follows:

$$\text{RMS error} = \pm \frac{400}{\sin \theta} \times \frac{\Delta\theta}{1000} = \frac{400 \times .3 \cdot 2 \cdot 5280}{1000}$$

$$= \pm 900 \text{ feet}$$

[redacted]

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	Systematic & Correctable		Random	
	15°	45°	15°	45°
Parallax Bias	.02	.05 Mils		
Doppler	.095	.018		
Transmitter Shift	.036	.01		
Transmission Delay	Neg.	No error		
Time	Neg.	Neg.	Neg.	Neg.
Station Survey			.07	.07
Local Survey			.04	.015
Noise			.79	.294
Orientation	Neg.	Neg.	Neg.	Neg.
Propagation	2.30	.35		
Computation	Neg.	Neg.		
Data Transmission	Neg.	Neg.		
R.M.S. Error	2.31	.36	.795	.295
Corrected	.23	.04		

RMS of Corrected Systematic & Random

	15°	45°
RMS	.83	.30

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1. Parallax Error.

The equation for calculating the angle θ from the measured value of ϕ_e and K, assumes that the target is far enough away that the rays from two antennas are essentially parallel. This assumption introduces a small parallax error which may be derived as follows: (Refer to Figure 1-1).

In determining θ from a measurement of phase difference between points B and D, the assumption is made that the actual path length difference is ℓ. This results in an error in path length of $\Delta\ell$. The error in θ can be estimated as follows:

$$\Delta\ell \cong \alpha \times S \sin\theta$$

This corresponds to a phase angle error of

$$\Delta\phi_e \cong \Delta\ell \times \frac{2\pi}{\lambda} \cong \alpha \cdot S \sin\theta \cdot \frac{2\pi}{\lambda}$$

In calculating θ from ϕ the following relationship is used:

$$\phi_e = \ell \cdot \frac{2\pi}{\lambda} = S \cdot \cos\theta \times \frac{2\pi}{\lambda}$$

$$\cos\theta = \frac{\lambda\phi_e}{2\pi S}$$

taking a total derivative

$$-\sin\theta\, d\theta = d\,\frac{\phi_e \times \lambda}{2\pi S} \cong \alpha S \sin\theta \cdot \frac{2\pi}{\lambda} \cdot \frac{\lambda}{2\pi S}$$

$$-\sin\theta\, d\theta \cong \alpha \sin\theta$$

or $$d\theta \cong -\alpha$$

The position error corresponding to an error in θ can be determined as follows:

$$X = (R+\ell)\cos\theta = (R + S\cos\theta)\cos\theta$$
$$= R\cos\theta + S\cos^2\theta$$

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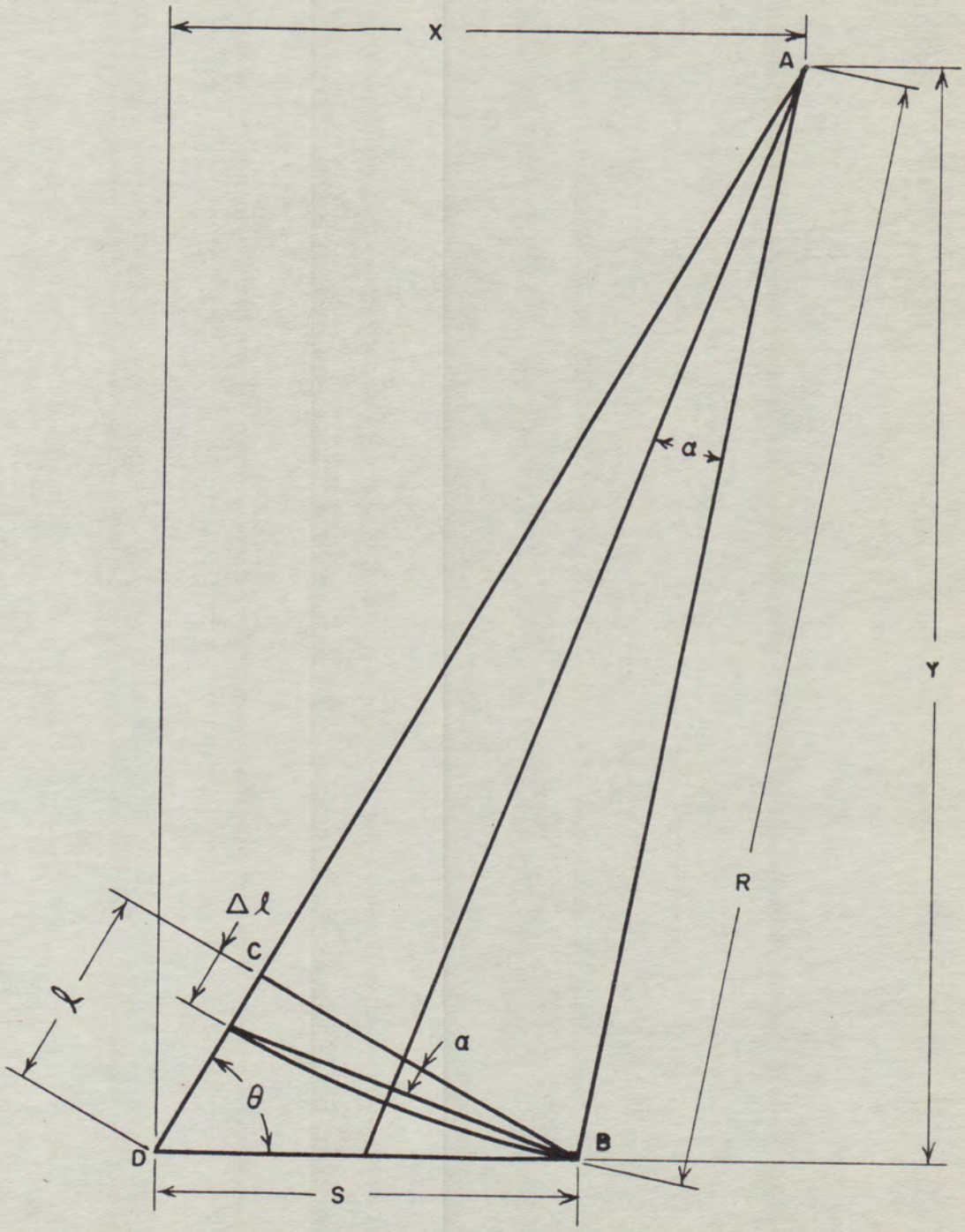

FIGURE I-I PARALLAX ERROR

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But $S \cos^2\theta << R \cos\theta$

$\therefore \quad X \cong R \cos\theta$

$dX \cong R\, d(\cos\theta) \cong R\alpha \sin\theta$

Since $\alpha \cong \dfrac{S \sin\theta}{2R}$

$$dX \cong \Delta X \cong \frac{S}{2} \sin^2\theta$$

at 45° $\Delta X = \dfrac{400}{2} \times 1 = 200$ feet

at 15° $\Delta X = 12$ feet

$$y = (R+\ell) \sin\theta = (R + S\cos\theta)\sin\theta$$

$$S \sin\theta \cos\theta << R \sin\theta$$

$$y \cong R \sin\theta$$

$$dy \cong R \cos\theta\, d\theta$$

But $d\theta \cong -\alpha$

$\therefore \quad dy \cong -R\alpha \cos\theta$

and $\alpha \cong \dfrac{S \sin\theta}{2R}$

$$\therefore \quad dy \cong \Delta y \cong -\frac{S \sin\theta \cos\theta}{2}$$

at 15° $\Delta y = \dfrac{400 \times 0.258 \times .966}{2} = 45.2$ feet

at 45° Δy is a maximum and equals $\dfrac{S}{4} = 100$ feet.

This error does not vary with the contemplated variation in range of the target and may be compensated for exactly by programming the central computer to solve for the intersection of four hyperbolas

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instead of the intersection of four cones. The hyperbolic solution is straightforward and has been described in various reports on Loran.

2. Doppler Frequency Shift.

Doppler shift $= f_d = \frac{v}{\lambda} \cos\theta$, where $v =$ satellite velocity

$$\cos\theta = \frac{\phi_e \lambda}{2\pi S} - \frac{K\lambda}{S}$$

$$= \lambda\left[\frac{\phi_e}{2\pi S} - \frac{K}{S}\right]$$

Assume $\phi_e = 0$

and take total derivative

$$-\sin\theta\, d\theta = d\lambda\left(\frac{-K}{S}\right)$$

$$\text{or} \quad \Delta\theta \cong \frac{\Delta\lambda K}{S \sin\theta}$$

$$\text{Var. in } \lambda = \frac{f_d \cdot \lambda}{f_0} = \frac{\frac{v}{\lambda}\cos\theta \times \lambda}{f_0}$$

WHERE f_0 = TRANSMITTER FREQUENCY.

$$\text{"} = \frac{5280 \times 4.894 \times \cos\theta}{125 \times 10^6}$$

$$\text{"} = .0002067 \cos\theta$$

at $\theta = 45°$ $K = 35$

$$\Delta\theta \cong \frac{v\cos\theta}{f_0} \times \frac{K}{S \times \sin\theta} = \frac{.0002067 \times .707 \times 35 \times 1000}{400 \times .707}$$

$$\Delta\theta \cong .018 \text{ Mils}$$

at $\theta = 15°$ $K = 49$

$$\Delta\theta = \frac{.0002067 \times .9659 \times 49 \times 1000}{400 \times .2588}$$

$$\Delta\theta = .095 \text{ Mils}$$

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3. Transmitter Frequency Shift.

It is estimated that the transmitter oscillator is stable to 10 parts in one million.

$\therefore$ the variation in $\lambda = \frac{\Delta f_0}{f_0} \times \lambda$

where Δf_0 = Change in transmitter frequency

$$\Delta\theta \cong \frac{\Delta\lambda K}{S \cdot \sin\theta}$$

$$\text{or} \quad \Delta\theta = \frac{\frac{\Delta f_0 \lambda}{f_0} K}{S \cdot \sin\theta}$$

at $\theta = 45°$ $\Delta\theta = .01$ Mils

at $\theta = 15°$ $\Delta\theta = .036$ Mils.

4. Transmission Delay.

At θ = 45 degrees, the path length to the two locations A and B is the same. Therefore, the error is zero.

At θ = 15 degrees, the path lengths may be different by 270 miles. This corresponds to a time difference of .00145 seconds. This corresponds to a space error of 36 feet. This may be considered negligible.

5. Station Survey.

As previously discussed in paragraph 4b of the basic report, the distance between locations can be determined to within $\pm$ 300 feet. This corresponds to an error of $\frac{300}{800 \times 5280} = 7 \times 10^{-5}$, which is equivalent to an angular error of .07 Mils.

6. Local Survey.

As previously stated in paragraph 4b of the basic report, the

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expected accuracy of the base line between a pair of phase comparison antennas is one part in one hundred thousand.

This corresponds to an error in ϕe

$$\Delta \phi e = n \times \frac{1}{10^5} \times 360 = .18 \text{ degrees}$$

The error in θ is then

$$\Delta \theta = \frac{\Psi}{360} \times \Delta \phi e$$ - where Ψ is the phase pattern lobe width.

At 45 degrees $\Delta \theta = \frac{1.7 \times .18 \times 17.8}{360} = .015$ Mils.

At 15 degrees $\Delta \theta = \frac{2.68 \times .18 \times 17.8}{360} = .04$ Mils.

7. Noise.

A signal to noise ratio of 24 db is assumed for the system. This corresponds to a minimum detectable $\phi e = \tan^{-1} \frac{1}{15.8} = 3.5$ degrees.

An error of 3.5 degrees in ϕe corresponds to

$\Delta \theta = \frac{\phi e}{360} \times \Psi$, where Ψ is the phase pattern lobe width

$$\Delta \theta = \frac{1.7 \times 3.5}{360} \times 17.8 = .294 \text{ Mils at } \theta = 45^\circ$$

$$\Delta \theta = \frac{2.68 \times 3.5 \times 17.8}{360} = .790 \text{ Mils at } \theta = 15^\circ$$

8. Propagation.

Refraction causes significant errors in propagation of radio waves. This refraction may be conveniently divided into atmospheric (tropospheric) and ionospheric phenomena. Atmospheric phenomena is a function of air density, pressure and humidity, and is to be considered correctable*. The worst atmospheric case, ducting, is not serious above 10°, therefore Lunchbox may disregard this difficult problem.

Ionospheric refraction is more difficult to correct for than

* Volume 13, Rad Lab - Chapter 3, pp 181-293

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atmospheric refraction because the parameters affecting ionospheric refraction are more difficult to measure. In addition, parallax errors are introduced because the refracting layers in the ionosphere are many miles above the earth's surface. Project Rand has made certain studies of ionospheric refraction and presents a method for computing total refraction. These data and methods are used in the sequel.

The following table gives the total refraction to be expected in a typical situation:

LAYER	Refraction angle at 125 MC			
	15° Elevation	30° Elevation	45° Elevation	60° Elevation
E	.18 min	.03 min	.01 min	.004 min
F_1	.62 "	.17 "	.06 "	.024 "
F_2	3.25 "	1.11 "	.38 "	.173 "
G	2.88 "	1.15 "	.46 "	.211 "
Total	6.93 "	2.46 "	.91 "	.412 "
	(2.04 mils)	(.725 mils)	(.268 mils)	(.122 mils)

The above table takes no account of diurnal variations, but does give an overall idea of the error to be corrected.

The Bureau of Standards publishes compendiums of world wide ionospheric data, giving hourly average ionospheric conditions at sounding stations for one month periods. These data are particularly valuable, since they give an indication of the time factor involved. The tables following cover the period of most rapid ionospheric change, giving average ionosphere data at San Francisco and Canal Zone for May 1954*. (See following page).

[redacted]

[redacted]

B-33

Angular error is computed for an elevation angle of 15° for 125 MC*.

SAN FRANCISCO

Time	Layer	Layer Height Km	Refraction Angle Minutes	Mils
0400	E	Not formed	Not formed	
	E_s	100	.28	.084
	F_1	Not formed	Not formed	
	F_2	280	.38	.114
	G	600	27.61	8.27
		TOTAL	28.27	8.47
0600	E	130	.03	.009
	E_s	100	.28	.084
	F_1	250	.29	.088
	F_2	310	.35	.105
	G	600	30.00	9.1
		TOTAL	30.95	9.4

* Ionospheric Data CR PL-F120, August 1954. Bureau Standards, Central Radio Propagation Laboratory, Boulder, Colorado

[redacted]

[redacted]

B-34

CANAL ZONE

Time	Layer	Layer Height Km	Refraction Angle Minutes	Mils
0600	E	Not formed	Not formed	
	E_s	100	.15	.045
	F_1	Not formed	Not formed	
	F_2	250	.36	.108
	G	600	31.46	9.43
		TOTAL	31.97	9.58
0800	E	110	.07	.021
	E_s	100	.31	.093
	F_1	220	.04	.012
	F_2	300	.62	.186
	G	600	27.60	8.26
		TOTAL	28.64	8.58

At San Francisco, the total error increased an average of .8 mil during the early morning, whereas at Canal Zone, it decreased 1.11 mil during the corresponding period. Now the E_s or E sporadic layer typically forms in about fifteen minutes and may or may not occur at a given time. (There is some statistical data on this also). Thus, for the above tables, it may be necessary to correct for $\pm$ 0.'3 = .1 mils in a few minutes time.

[redacted]

B-35

Also in these two tables, it was assumed that the E_s layer formed at 100 KM and had typically a semi-thickness of 15 KM and that the G layer formed at 600 KM and had a semi-thickness of 150 KM. The changes in refractive index for these two layers was solely on the basis of measured critical frequency.

The E_s layer assumptions are fairly good; however, the G layer is said to vary from 400-700 KM in height. This height is difficult to measure because of the effects of lower layers. A continuous ionospheric sounding station should be able to provide accurate ionospheric information up to and possibly including the F_2 layer and these lower layer errors may be considered resolvable.* Therefore, an ionospheric sounding station should be included at each antenna location.

The G layer is shown to have a higher electron density (based on the measured critical frequency) in the Bureau of Standards Report than the Rand report. This may be due to better measurements of both F_2 and G layers, since the publication of the Rand Report, a period of seven years.

The following table was computed to study the parallax problem as a function of angle of elevation, height of the G layer and height of the orbit. It is assumed the G layer, in all cases, has a thickness of 150 KM and a critical frequency of 30 MC. A similar, but much less severe, problem occurs for the lower layers, particularly F_2. The G layer is studied as being by far the worst case.

* The cost of an ionosphere sounding station is estimated at $1000.00. Present sets sweep the frequency range in 10 seconds.

[redacted]

B-36

Elevation Angle	G layer Height KM	Refraction Angle	
		Minutes	Mils
15°	400	44.8	13.44
	700	22.5	6.75
30°	400	12.9	3.87
	700	9.1	2.73
45°	400	6.3	1.91
	700	3.9	1.17
78°	400	0.56	.168
	700	0.52	.156

The above table was used to compute the table below for a station with the following conditions and assumptions:

The Station consists of two locations, A and B, separated by 12 degrees, or about 1300 KM. All orbits lie in a plane containing A, B, and the earth's center. Radius of the earth is 4000 miles. The errors at location A are calculated for angles of elevation of 15°, 30°, 45°. The angles of elevation at location B are assumed to be 78°, or greater, and that the refractive errors at 78°, or greater, may be neglected in computing the approximate errors at location A.

Note: There is very likely some parallax effect due to the thickness of the refraction layer, the actual amount of which is a function of the layer profile. The optical analog is

[redacted]

[redacted]

B-37

G Layer Height	Orbit Height	Elevation Angle	Refraction Angle in Mils	Parallax Correction Factor	Radar Angular Error
400 KM	200 Mi.	Orbit below G layer			0
	500 Mi.	15°	13.44	≥.5	≥6.72
		30°	3.87	≥.5	≥1.93
		45°	1.91	≥.5	≥0.96
	800 Mi.	15°	13.44	≈.33	≈4.43
		30°	3.87	≈.33	≈1.28
		45°	1.91	≈.33	≈.63
700 KM	200 Mi.	Orbit below G layer			0
	500 Mi.	15°	6.75	≈.13	≈.88
		30°	2.73	≈.13	≈.36
		45°	1.17	≈.13	≈.15
	800 Mi.	15°	6.75	≈.42	≈2.84
		30°	2.73	≈.42	≈1.15
		45°	1.17	≈.42	≈.49

The greatest refraction error encountered is caused by the G layer. This error varies with the height of the G layer, the angle of elevation, and the height of the satellite. When the satellite altitude is 500 miles and the height of the G layer is 600 KM, the radar angular error will be 2.3 mils at an elevation of 15°. A satellite altitude of 500 miles and a G layer height of 600 KM represents the most probable conditions. At higher elevation angles, the error is much less, being 0.35 mils at 45°.

It is assumed that the errors due to refraction in the atmos-

[redacted]

B-38

phere and the ionosphere may be reduced to 10% of the uncorrected value.

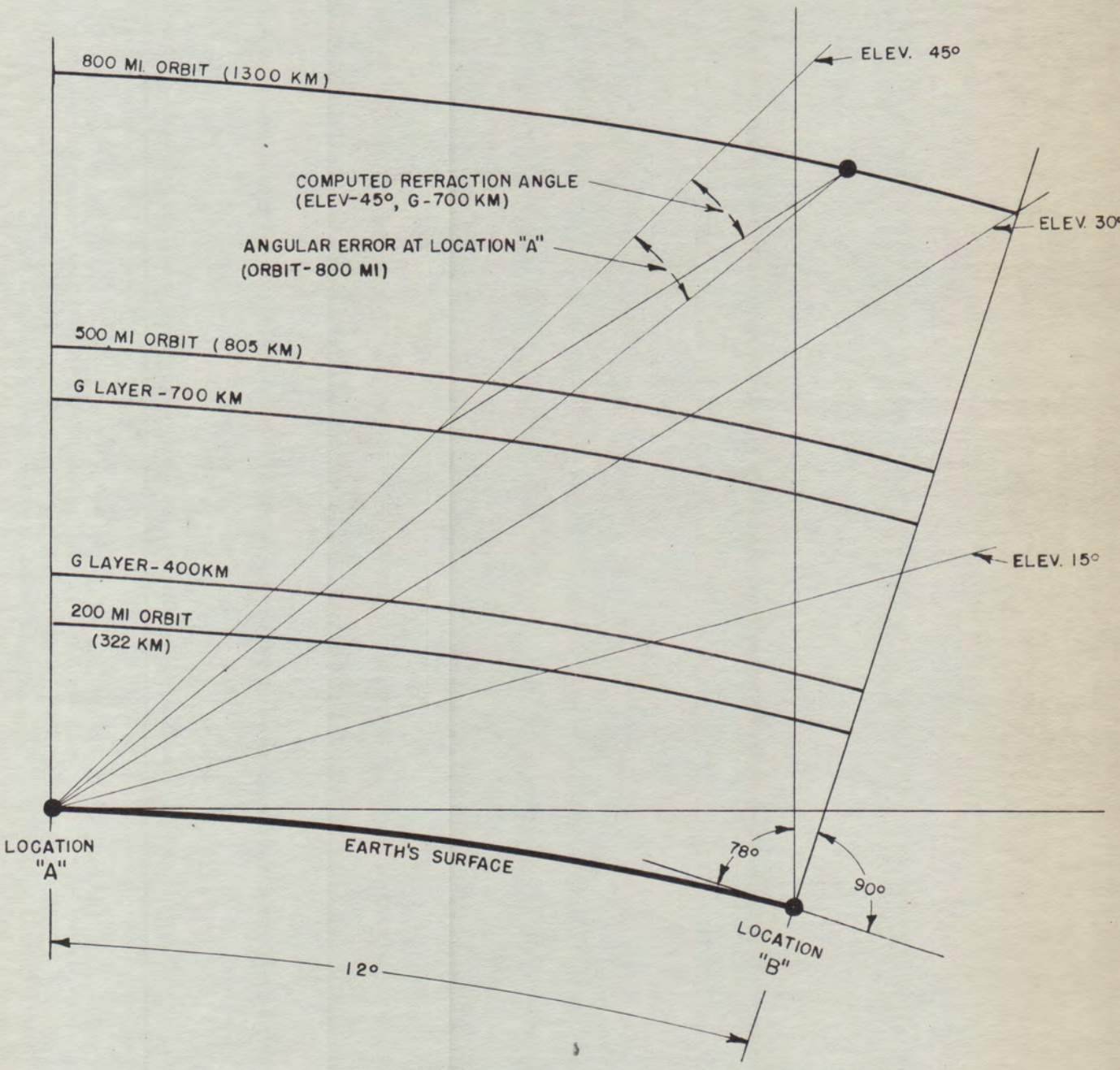

FIGURE 1-2. EFFECT OF REFRACTION ON RADAR ANGULAR ACCURACY

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APPENDIX 2

TRACKING ANTENNA

The tracking antenna shall consist of individual lobe switched elevation and azimuth antennas. Each antenna will be separately fed and shall consist of a broadside array of five dipoles. The reflector will be of lightweight mesh or expanded metal construction. In a preferred embodiment (See Figure 2-1), the two antennas will be incorporated into a single unit (in cruciform shape) of overall dimensions 25' x 25', and mounted on a single pedestal of the type used for the Instrumentation Radar. If this is not feasible, separate, individually mounted antennas can be used. In the actual antenna, the dipoles may be tilted at 45° to the major axes of the antenna.

Characteristics of Individual Elevation and Azimuth Antennas:

Aperture:	25' x 5'
Beamwidth:	20° x 78°
Crossover:	-3 db
Weight of Integrated Unit	200 lbs.

Pedestal:

Weight:	8000 lbs.
Dimensions:	10' high ≠ cement block x 80" diameter base
Accuracy:	0.1 mil

[redacted]

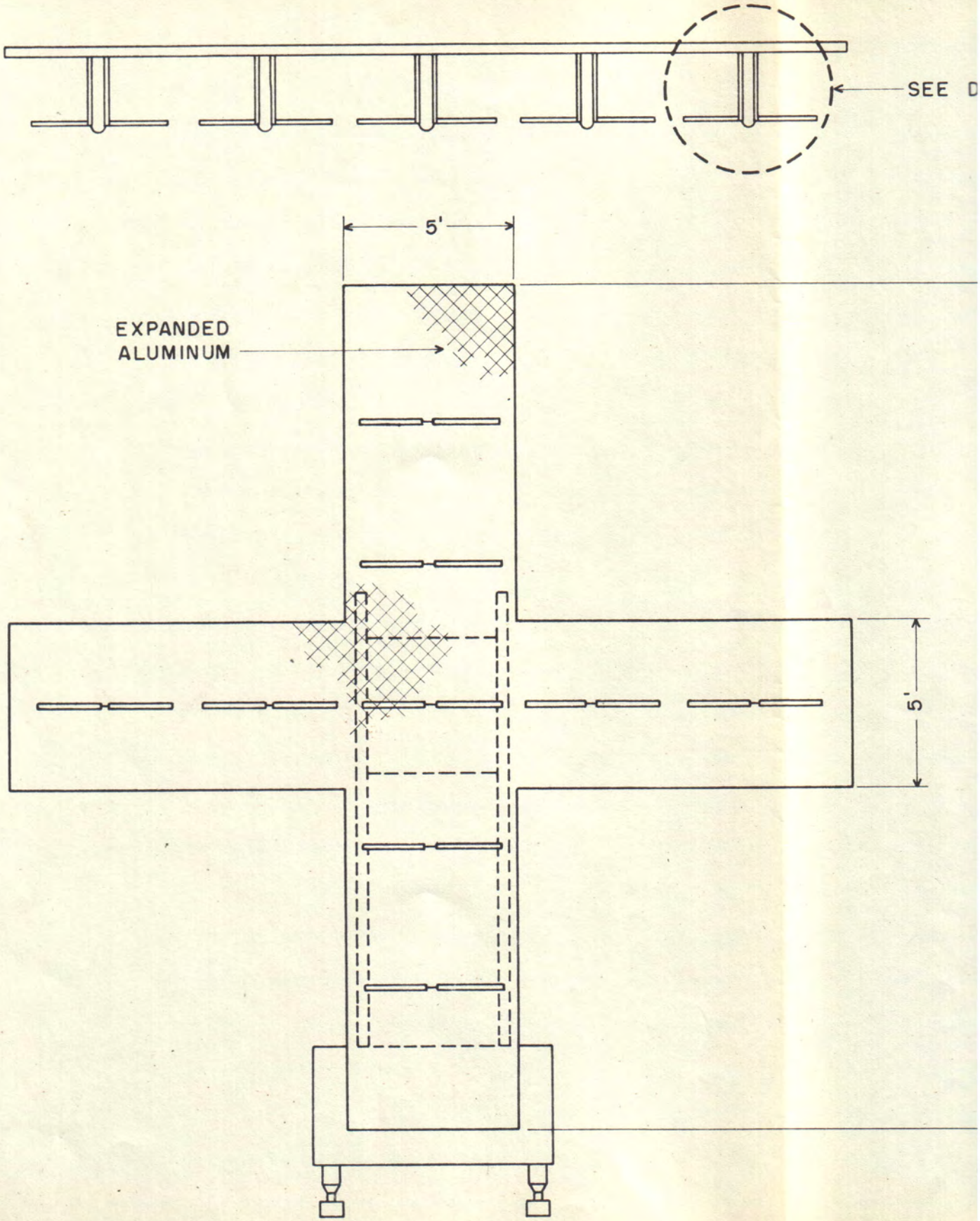
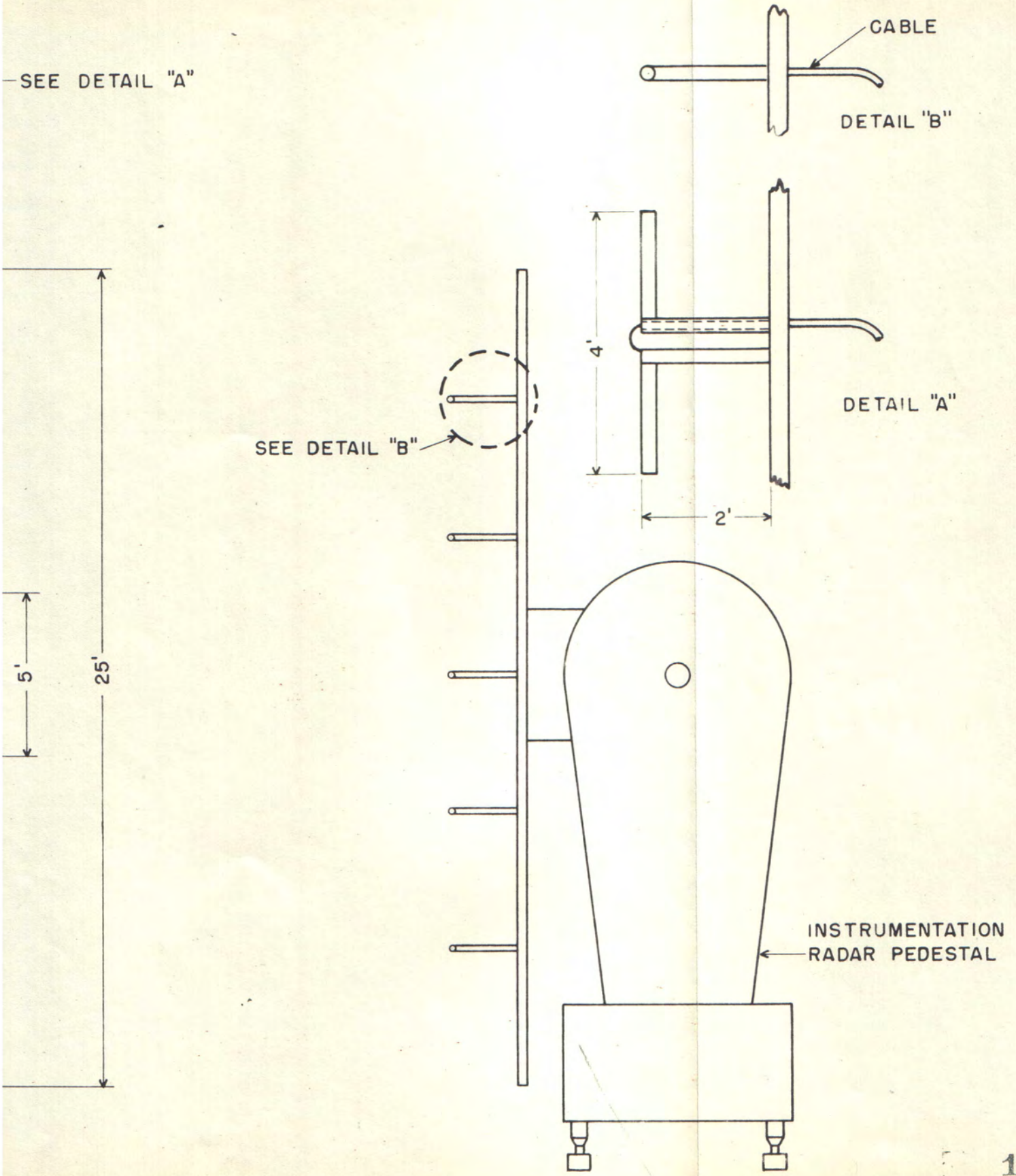


FIGURE 2-1. TRACKING ANTENNA


SEE DETAIL "A"
CABLE
DETAIL "B"
4'
DETAIL "A"
SEE DETAIL "B"
2'
5'
25'
INSTRUMENTATION
RADAR PEDESTAL

[redacted]

B-40

APPENDIX 3

CALCULATION OF COARSE TRACKING RADAR RANGE

The radar parameters pertinent to range performance are:

a. Effective antenna collecting area at crossover - 50 ft.

b. Slope at crossover - 6% per degree.

c. Pedestal accuracy - $\pm$ 2 mils.

d. Receiver noise factor - 4 db.

e. Receiver IF bandwidth - 20 KC.

f. Receiver post detection bandwidth - 1 cps.

The pertinent transmitter characteristics are:

a. Transmitter power output - 25 milliwatts.

b. Radiation efficiency - 10%.

c. Polarization scintillation - 50%.

d. Amplitude Scintillation - 50%.

e. Transmitter stability - $10/10^6$ or 1250 cps at 125 mc.

f. Maximum Doppler Shift - 3330 cps at 1250 mc.

The receiver IF bandwidth is determined from the maximum variation in transmitter frequency, allowing a factor of safety of 2

$$\Delta f = 4\ (1250 + 3330) \cong 20KC$$

Due to scintillation, the average power radiated will be reduced by

$$0.5 \times 0.5 \times 0.5 = 0.125$$

Since the gain of a dipole over an isotropic radiator is 1.64, and considering the radiation efficiency, the average radiated power will be

$$P_r = 25 \times 10^{-3} \times 0.125 \times 0.10 \times 1.64 = 0.513 \times 10^{-3} \text{ watts.}$$

The total rms tracking error is to be $\pm$ 8 mils. Since the pedestal error is $\pm$ 2 mils, the error due to noise can be no greater than:

$$e_n = \sqrt{8^2 - 2^2} = 7.75 \text{ Mils} = 0.44°$$

[redacted]

[redacted]

This means that for a target which is 7.75 mils off the crossover, the error signal must be equal to noise.

The signal to noise ratio at the tracking circuits can be related to the signal to noise ratio of the RF as follows:*

$$\frac{V_s}{V_n} = M \frac{E_s^2}{E_o^2} \sqrt{\frac{1}{B_2(B_1 - 0.5\ B_2)}}$$

where $\frac{V_s}{V_n}$ = signal to noise ratio at tracking circuits = 1

E_s = RMS signal voltage at input to RF amplifier

E_o = RMS noise voltage per unit of bandwidth at input to RF amplifier.

m = modulation factor of signal voltage

$E_n = E_o\sqrt{B_1}$ = RMS noise voltage at input to RF amplifier

B_1 = Pre-detection receiver bandwidth = 20 KC

B_2 = Post detection receiver bandwidth = 1 cps

Then, since $B_1 \gg B_2$

$$\frac{V_s}{V_n} = M \frac{E_s^2}{E_N^2} B_1 \sqrt{\frac{1}{B_2(B_1 - 0.5\ B_2)}} \approx \frac{M E_s^2}{E_o^2} \sqrt{\frac{B_1}{B_2}}$$

$$\frac{E_s^2}{E_N^2} = \frac{1}{M} \times \frac{V_s}{V_N} \sqrt{\frac{B_2}{B_1}}$$

$\frac{E_s^2}{E_N^2}$ = Signal to noise power ratio at the input to the RF amplifier

For a target at crossover, the modulation factor is:

$$M = \frac{dE}{d\theta} \times \Delta\theta \times \frac{E_M}{E_C}$$

$\frac{dE}{d\theta}$ = Slope of voltage antenna pattern at crossover 6% per degree

* Callender, M. Z., "Thermal Noise Output of AM Receivers". Wireless Engineer 25 (No. 303) 305-399, 1948.

[redacted]

[redacted]

B-42

$\Delta\theta$ = Angle of target off crossover = 0.44°

E_M = Signal at maximum of antenna pattern

E_C = Signal voltage at crossover = 0.707 Em

Then

$$\frac{E_S^2}{E_N^2} = \frac{P_S}{P_N} = \frac{1}{\frac{dE}{d\theta} \times \Delta\theta \times \sqrt{2}} \sqrt{\frac{B_2}{B_1}}$$

$$= 0.190$$

$$Ps = 0.190\ Pn = 0.190 \times KT\ B_1\ \overline{NF}$$

$$= 0.190 \times 4.1 \times 10^{-21} \times 20{,}000 \times 2.5$$

$$= 39 \times 10^{-18} \text{ watts}$$

The tracking range can then be determined from the following:

$$Ps = \frac{P_t\ A_r}{4\pi R^2}$$

where P_t = Radiated power from transmitter = 0.513×10^{-3} watts.

A_r = Effective antenna collecting area = 50 ft^2

R = Range to transmitter

[redacted]

[redacted]

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APPENDIX 4

PHASE COMPARISON ANTENNAS

The phase comparison antenna system will consist of three separate broadside arrays mounted on pedestals similar to those used on the AN/MPQ-12. (See Figure 4-1). Each array will consist of 9 dipoles arranged in a 3 x 3 square and separated by about five-eighths of a wavelength. The 16' x 16' reflector will be of lightweight mesh or expanded metal construction. The dipoles may possibly be oriented at 45° to the reflector edges.

Antenna

Aperture:	16' x 16'
Beamwidth:	30° x 30°
Weight:	200 lbs - 240 lbs

Pedestal

Weight:	600 lbs
Dimension:	12' x 4' diameter base
Accuracy:	0.5 mil

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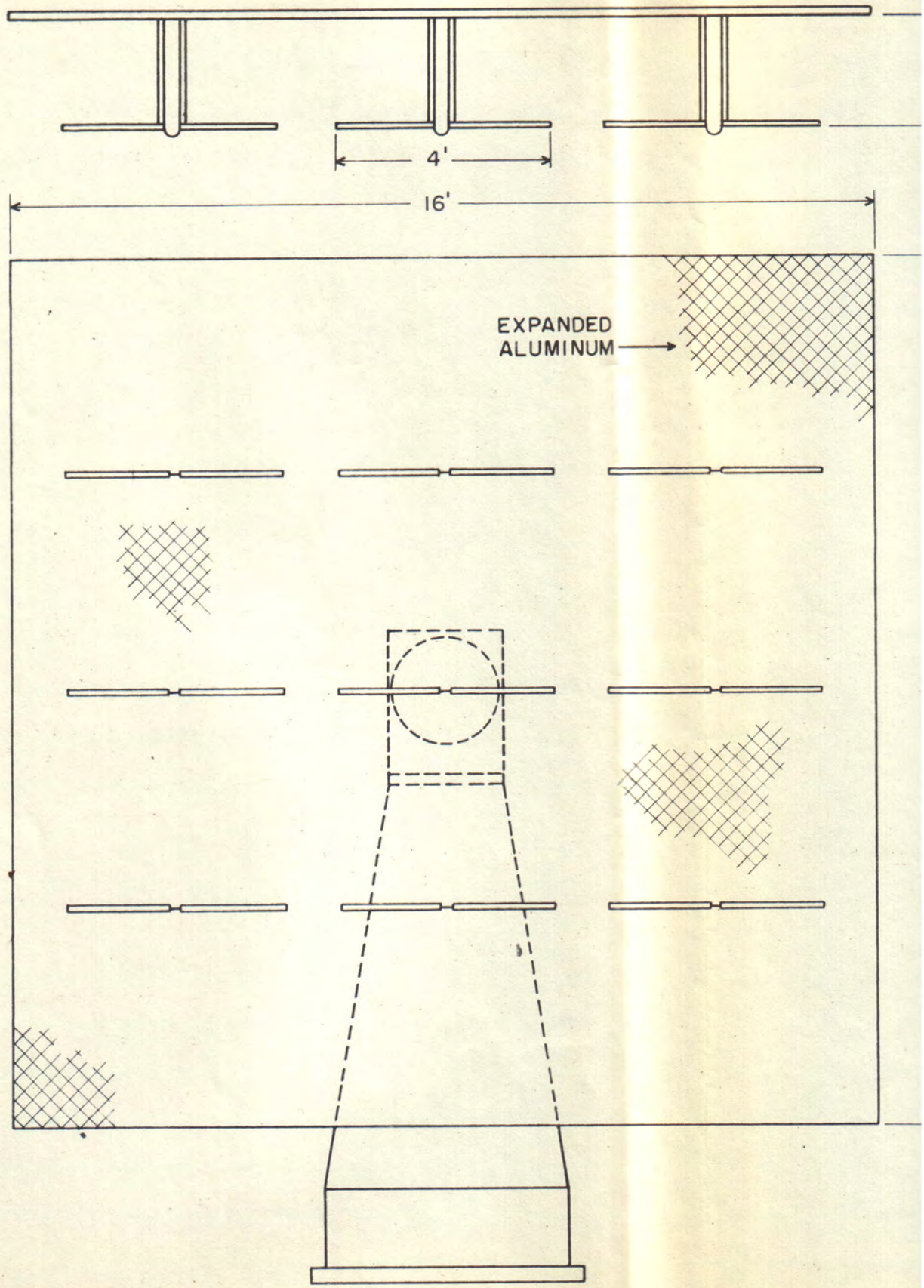
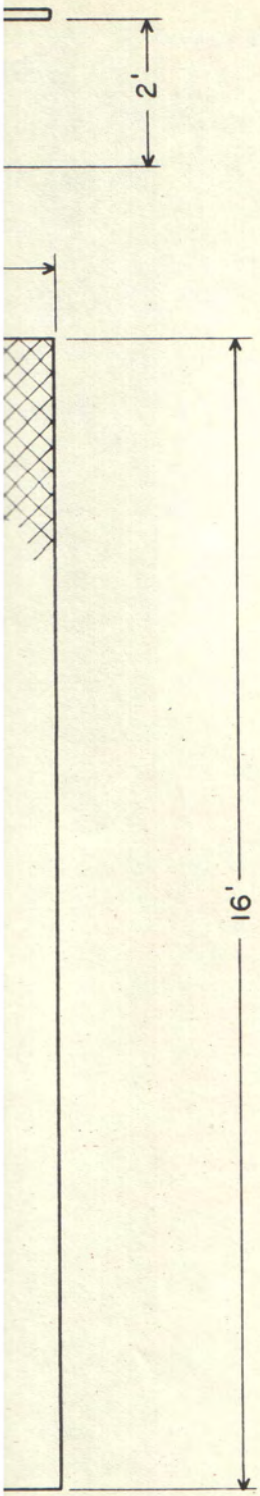
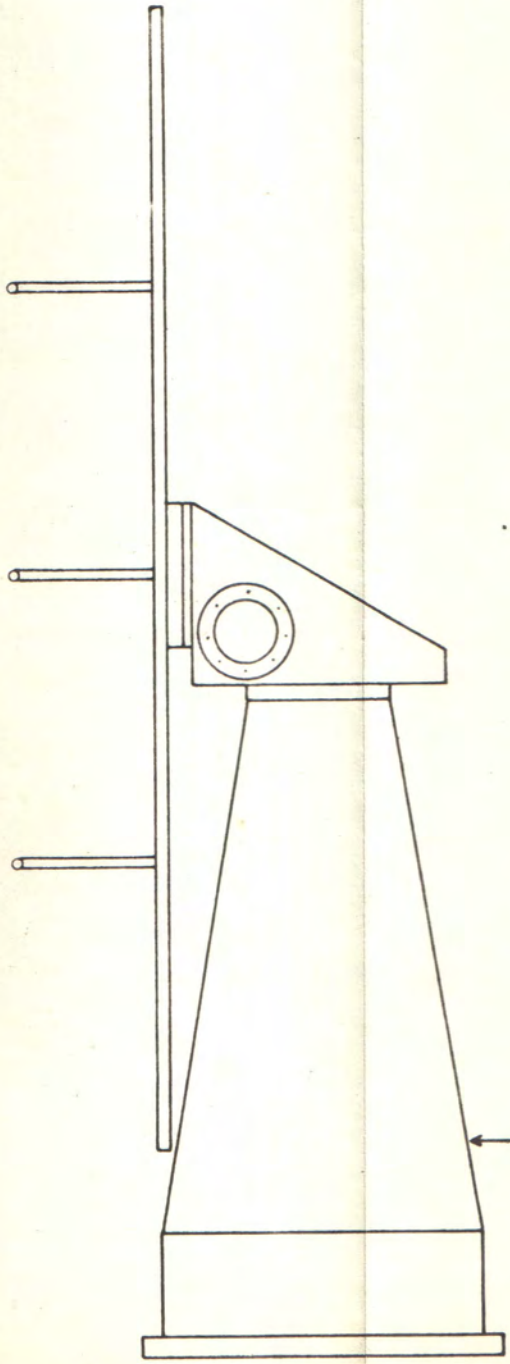


FIGURE 4-1. PHASE COMPARISON ANTENNA





[redacted]

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APPENDIX 5

CALCULATION OF RANGE OF FINE ANGLE TRACKING SYSTEM

Parameters pertinent to calculation of tracking range are:

a. Effective antenna collecting area - 200 ft^2

b. Receiver Noise Factor - 4 db

c. Receiver IF bandwidth - 20 KC

d. Receiver post detection bandwidth - 1 cps

e. Average radiated power - 0.513×10^{-3} watts

In order to measure electrical phase to within 3.5°, the signal to noise ratio at the phase detector should be at least 24 db. The signal to noise ratio at the phase detector is related to the signal to noise ratio in the IF as follows:

$$(S/N)_{PD} = \left(\frac{S}{N}\right)^2_{IF} \times \frac{\Delta f}{\Delta F}$$

where Δf = IF bandwidth

ΔF = Post detection bandwidth

$$(S/N)_{IF} = \sqrt{(S/N)_{PD} \times \frac{\Delta F}{\Delta f}} = \sqrt{256 \times \frac{2}{20{,}000}}$$

$$= 0.16$$

$$P_{SIF} = (S/N)_{IF} \times KT\Delta f \times \overline{NF}$$

$$= 0.16 \times 4.1 \times 10^{-21} \times 20{,}000 \times 2.5$$

$$= 0.16 \times 1.05 \times 10^{-18} = 32.8 \times 10^{-18} \text{ WATTS}$$

The tracking range can be found from the following:

$$R = \sqrt{\frac{P_t \times A_r}{P_{SIF} \times 4\pi}}$$

Where P_t = Average radiated power

[redacted]

[redacted]

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A_r = Effective antenna collecting area

$$R = \frac{0.513 \times 10^{-3} \times 200}{32.8 \times 10^{-18} \times 4\pi} = 3000 \text{ MILES.}$$

[redacted]

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APPENDIX 6

DETERMINATION OF NUMBER OF TIMES A TARGET PASSES OVER A GIVEN STATION

Assumptions:

1. Orbit is confined between 30° N and 30° S.
2. Launch is made at 80° W longitude 30° N latitude ≈ (Patrick Air Force Base)
3. No perturbation other than the earth's rotation influences the satellite's orbit.
3. 90 minutes is period of orbit.

Case I. Determination of crossing points at 80° W longitude.

The first crossing will be at the launch point. On the second crossing, however, the earth has moved 22.5 degrees plus the distance it moves in the time it takes the satellite to traverse the extra 22.5° or (22.5) (.0625) = 1.4°.

Therefore the general formula for determination of the angle on the nth crossing is

$$\theta = (n - 1)\ (1.0625)\ (22.5)$$

The latitude of the crossing is 30 cos θ, since the projection of the orbit on the earth is approximately a cosine wave. See Table I.

Case II. Determination of the latitudes of crossing on any meridian other than one on which the satellite will reach the 30° peak.

It is necessary to determine the equivalent number of degrees that the satellite has traversed since the longitude that had a 30° peak.

$$X = (80^\circ\ W - \text{longitude}) \bmod 22.5$$

$$\theta = X + 0.0625\ X = 1.0625X$$

The second term accounts for the additional rotation of the earth. In

[redacted]

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figuring the number of degrees between 80° W longitude and the desired longitude one must travel from West to East.

Finally

$$\text{Latitude} = 30 \cos \theta$$

If cos θ is negative, this indicates South latitude.

Example: Find the latitude of first crossing at 140° E longitude.

$$\text{Latitude} = 30 \cos \theta$$

$$\theta = 1.0625\ X$$

$$X = (80^\circ\ W - 140^\circ\ E) \bmod 22.5 = 17.5^\circ$$

$$\theta = (1.0625)\ (17.5) = 18.6^\circ$$

$$\text{Latitude} = 30 \cos \theta = 30\ (.9485) = 28.5^\circ\ N$$

To find the latitude of the nth crossing on a particular longitude, the following formula is used:

$$\text{Latitude} = (30 \cos \theta)^\circ \quad \text{where}$$

$$\theta = 1.0625\ X$$

$$X = \left\{(80^\circ\ W - \text{longitude}) \bmod (22.5)\right\} + (n-1)\ (22.5)$$

where n is the number of the crossing desired on the particular longitude corresponding to the nth crossing on the 80° W meridian. See Table II.

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TABLE I

LATITUDES OF CROSSING ON 80° W LONGITUDE

Orbit	Degrees to Left of Launch of Peak	Amount of Additional Rotation at 80°	θ	Cos θ	Latitude
Launch Crossing 1st	0	0	0	1	30° N
2nd	225	1.4°	23.9	.9142	27.4° N
3rd	45	2.8	47.8	.6717	20.2° N
4th	67.5	4.2	71.7	.3130	9.4° N
5th	90	5.6	95.6	-.0941	2.8° S
6th	112.5	7.0	119.5	-.4924	14.8° S
7th	135.0	8.4	143.4	-.8026	24.1° S
8th	157.5	9.8	167.3	-.9759	29.3° S
9th	180.0	11.3	191.3	-.0808	29.4° S
10th	202.5	12.7	215.2	-.8175	24.5° S
11th	225.0	14.1	239.1	-.5145	15.4° S
12th	247.5	15.5	263.0	-.1219	3.7° S
13th	270.0	16.9	286.9	-.2907	8.7° N
14th	292.5	18.3	310.8	.6539	19.6° N
15th	315.0	19.7	334.7	.9040	27.1° N
16th	337.5	21.1	358.6	.9997	29.991° N

[redacted]

[redacted]

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TABLE II

LATITUDES OF CROSSING ON 140° E LONGITUDE

Orbit	X	θ	Cos θ	30 Cos θ = Latitude
Crossing				
1st	19.5	20.7	9354	28.1° N
2nd	42.0	44.6	7121	21.4° N
3rd	64.5	68.5	3665	11.0° N
4th	87.0	92.4	-.0419	1.3° S
5th	109.5	116.3	- 4431	13.3° S
6th	132.0	140.3	- 7694	23.1° S
7th	154.5	164.2	- 9622	28.9° S
8th	177	188.1	- 9900	29.7° S
9th	199.5	212.0	- 8481	25.4° S
10th	222	235.9	- 5606	16.8° S
11th	244.5	259.8	- 1771	5.3° S
12th	267	283.7	2368	7.0° N
13th	289.5	307.6	6102	18.3° N
14th	312	331.5	8789	26.4° N
15th	334.5	366.6	9934	29.7° N
16th	357	379.3	.9438	28.3° N

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

1. Tracking Systems

c. Tracking Systems Requiring Electronic Equipment in the Satellite

(2) A Method Utilizing Doppler's Principle

[redacted]

[redacted]

II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE 1

1. Tracking Systems

c. Tracking Systems Requiring Electronic Equipment in the Satellite.

(2) A Method utilizing Doppler's Principle.

Prepared by: Components Department

SYNOPSIS: The shift in the frequency due to the Doppler effect of a signal, emitted from a stable transmitter located in the satellite, extends a possibility to very accurately compute the velocity of the object and at the same time determine its path, if only the frequency of the signal as received by a number of stations on the ground is being measured. Time did not permit final analysis of the subject in general form, but the results obtained for a simplified case are so encouraging that it is deemed worthwhile to call attention to this method. According to the preliminary results, it should be entirely feasible to determine the velocity of the satellite to within 20 feet per second, a figure which will not change materially, if the most general case is considered. In the following, the general operating principle will be explained by means of a simplified example and a possible set-up for the receiving stations discussed.

1 1. If a signal of frequency f_o is emitted from a source, in motion relative to a receiving station, the frequency as measured at this station is different from f_o because of the Doppler effect. The magnitude of the difference depends upon the component of the velocity of the moving transmitter in the direction of the radius vector from the receiving station to the object.

This effect extends the possibility of determining the velocity and the trajectory of the signal source, if the general law governing the motion and the frequency of the signal as measured at the receiving station as a function of time are known. The accuracy obtainable by this method depends upon the accuracy of the measurement of the signal frequency at the receiving station and upon the stability of the emitted frequency. An additional source of error, common to all methods using

[redacted]

an electromagnetic signal propagated over long distances, the erratic changes in the refractive index of the atmosphere, however, will set the ultimate limit.

2. To illustrate the general principle of this method it shall be assumed, for simplicity, that the object carrying the transmitter is moving along a straight line with constant velocity. Hence, the law of motion is $\vec{v} = \text{const}$. In Fig. 1, A is the position of the receiving station, b the trajectory of the object, and r(t) the radius vector from the receiving station to the object at the time t. Hence B is the position of the transmitter at a certain time t, C at the time t_o.

2 At t_o the

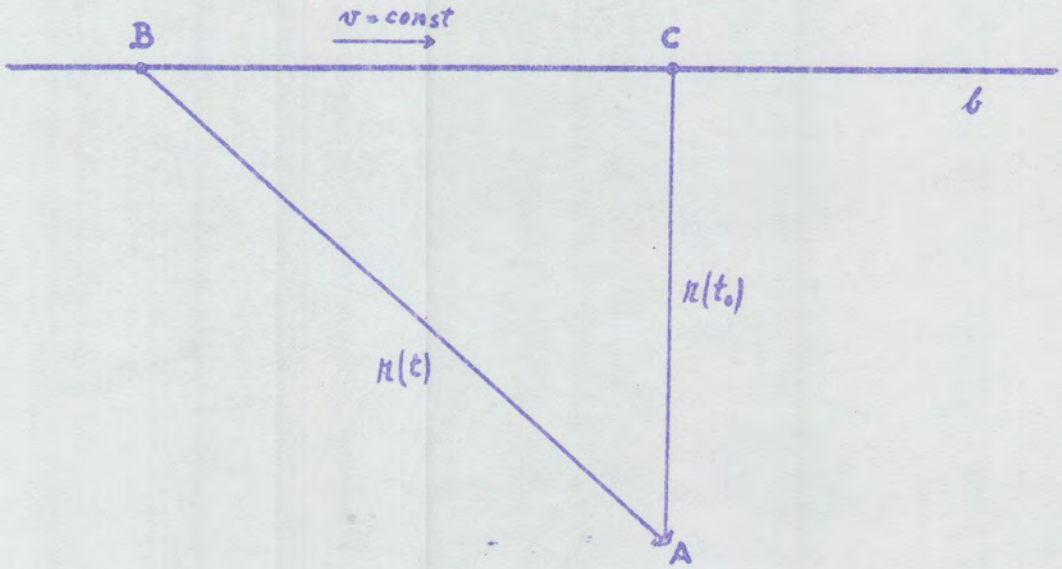

Fig. 1

radius vector from A is orthogonal to the trajectory, and the frequency of the received signal is exactly equal to the emitted frequency f_0. The frequency f of the received signal at an arbitrary time t is easily found to be

$$f = f_0\left(1 - \frac{v}{c}\,\frac{t-t_0}{\sqrt{\frac{a^2}{v^2} + (t-t_0)^2}}\right) \tag{1}$$

where f_0 is the transmitter frequency, v the velocity of the object, c the velocity of propagation of the signal and a the length of the radius vector at $t = t_0$. The general form of this function is represented in Fig. 2:

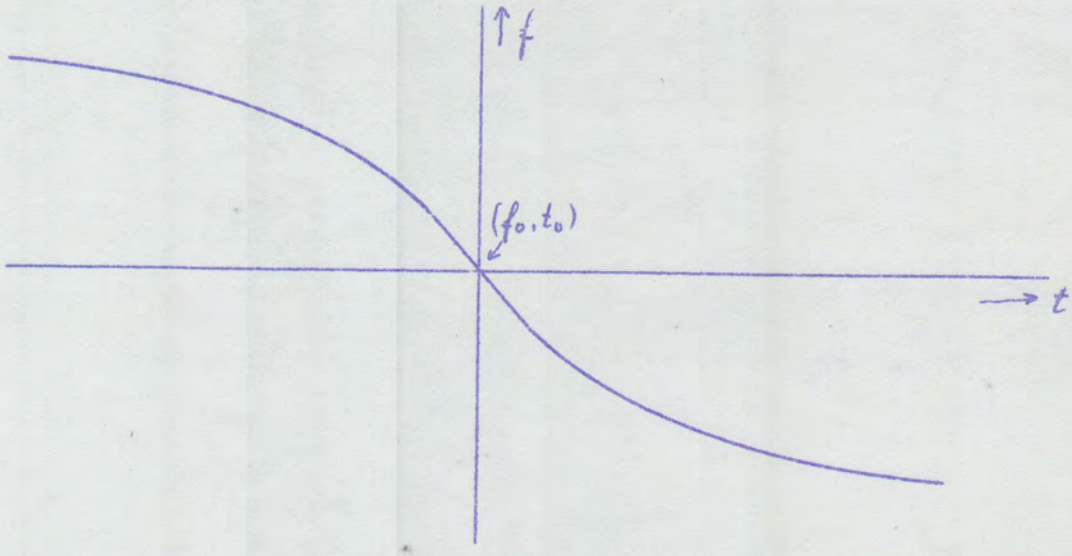

Fig. 2

If the velocity of propagation c is assumed to be known, equation (1) contains four unknowns v, a, t_0 and f_0 which can be computed if f is being measured for a number of different values of t.

From (1) follows

$$\frac{a^2}{v^2(t_1-t_0)^2} = \frac{v^2}{c^2}\,\frac{f_0^2}{(f_1-f_0)^2} - 1 \tag{2}$$

and

$$\frac{a^2}{v^2(t_2-t_0)^2} = \frac{v^2}{c^2}\frac{f_0^2}{(f_2-f_0)^2} - 1 \tag{3}$$

if the frequency of the received signal is f_1 at t_1, and f_2 at t_2. From (2) and (3) one finds

$$v^2 = \frac{c^2}{f_0^2}\frac{(t_1-t_0)^2 - (t_2-t_0)^2}{\left(\frac{t_1-t_0}{f_1-f_0}\right)^2 - \left(\frac{t_2-t_0}{f_2-f_0}\right)^2} \tag{4}$$

and

$$a^2 = v^2(t_2-t_0)^2\left(\frac{v^2}{c^2}\frac{f_0^2}{(f_2-f_0)^2} - 1\right) \tag{5}$$

4 Hence, if f_0 and t_0 were known, two pairs of values (f_1, t_1) and (f_2, t_2) would be sufficient to obtain v and a. It can be shown that f_0 and t_0 have to be known only approximately, if the arithmetic mean is taken from values of v and a, computed from two sets of pairs (f_1, t_1), (f_2, t_2) and (f_1', t_1'), (f_2', t_2') which are symmetric to the actual values of f_0 and t_0. This will be the case if v and a are being determined by averaging over the results, obtained from a great number of measured values of the right and at the left of (f_0, t_0). For this, f_0 and t_0 can be found from the graphical representation of the measurements (Fig. 2).

However f_0 and t_0 can also be found analytically if four pairs (f_i, t_i) are taken into account. By inserting the expressions (4) and (5)

[redacted]

into the formulas

$$\frac{a^2}{v^2(t_i-t_0)^2} = \frac{v^2}{c^2}\frac{f_0^2}{(f_i-f_0)^2} - 1 \qquad (i=3,4) \tag{6}$$

two relations between f_0, t_0 and the measured values (f_i, t_i) ($i = 1, 2, 3, 4$) can be obtained

$$\frac{(t_3-t_0)^{-2} - (t_1-t_0)^{-2}}{(f_3-f_0)^{-2} - (f_1-f_0)^{-2}} = \frac{(t_2-t_0)^{-2} - (t_1-t_0)^{-2}}{(f_2-f_0)^{-2} - (f_1-f_0)^{-2}}$$

$$\frac{(t_4-t_0)^{-2} - (t_1-t_0)^{-2}}{(f_4-f_0)^{-2} - (f_1-f_0)^{-2}} = \frac{(t_2-t_0)^{-2} - (t_1-t_0)^{-2}}{(f_2-f_0)^{-2} - (f_1-f_0)^{-2}} \tag{7}$$

5

These equations are of the third degree in both unknowns and are best solved by successive approximation.

The final values f_0 and t_0 found in this manner can then be inserted into (4) and (5) to compute v and a. If more than four points of the curve Fig. 2 are available, a least square method can be employed to reduce the errors.

This procedure is particularly useful if one portion of the curve Fig. 2 shall be evaluated independently from the rest of the measurements.

3. Four measurements of frequency and time at a single observation station are therefore necessary and sufficient in the present case, to

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determine the velocity v of a moving transmitter, the emitted frequency f_0, the shortest distance "a" between the trajectory and the observation station, and the time t_0, when the transmitter passes the station. To determine the position of the trajectory in space, however, simultaneous measurements at three stations, the relative positions of which are known, are in general necessary. If in Fig. 3, A_1, A_2, A_3 are the three observation stations with the respective coordinates $(X_1, 0, 0)$, $(0, 0, 0)$,

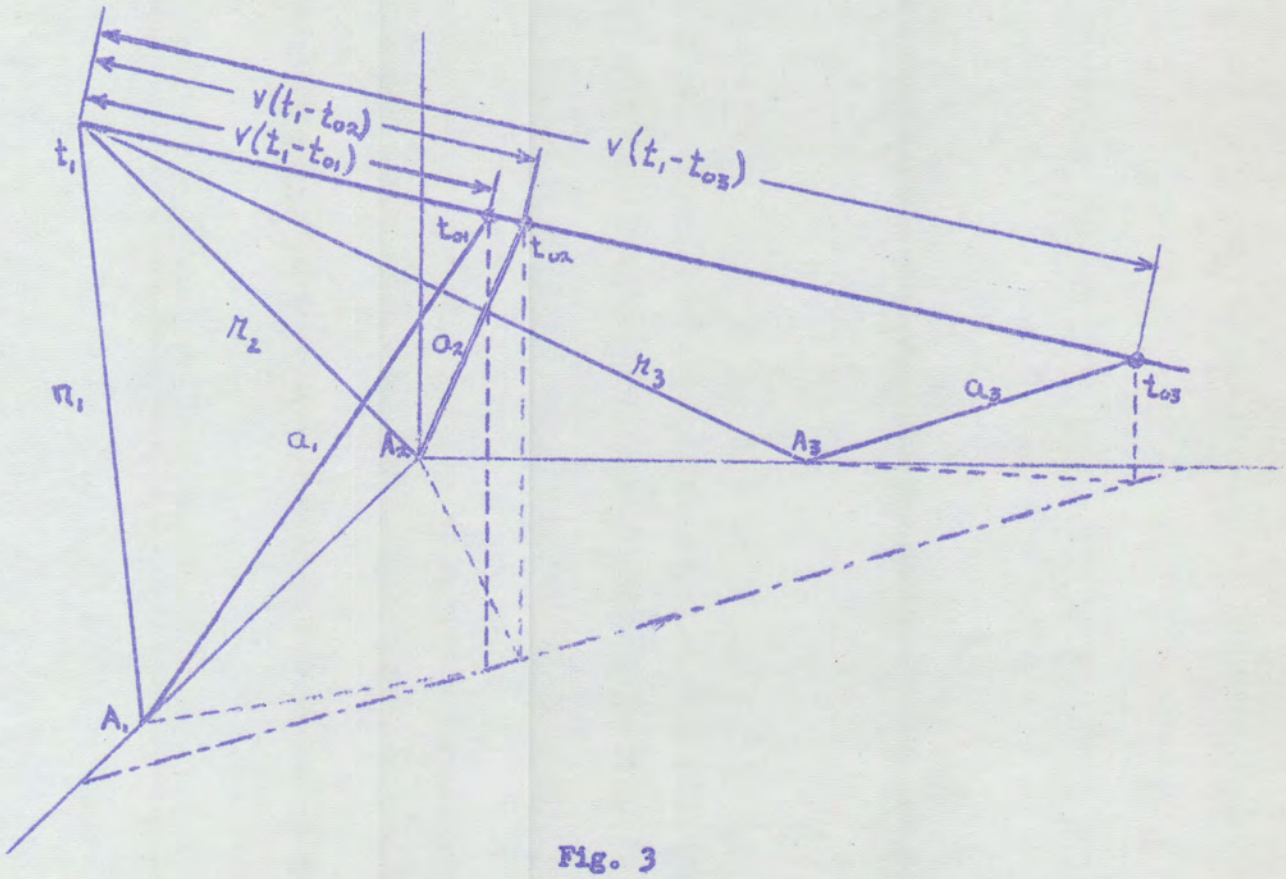

Fig. 3

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$(0, Y_3, 0)$, one can find immediately three simultaneous equations

$$
\begin{aligned}
(X - X_1)^2 + Y^2 + Z^2 &= r_1^2 = a_1^2 + v^2(t - t_{01})^2 \\
X^2 + Y^2 + Z^2 &= r_2^2 = a_2^2 + v^2(t - t_{02})^2 \\
X^2 + (Y - Y_3)^2 + Z^2 &= r_3^2 = a_3^2 + v^2(t - t_{03})^2
\end{aligned}
\tag{8}
$$

for the coordinates (X(t), Y(t), Z(t)) of the transmitter as a function of time. The solutions of (8) are obviously

$$
\begin{aligned}
X(t) &= \frac{X_1^2 + r_2^2(t) - r_1^2(t)}{2X_1} \\
Y(t) &= \frac{Y_3^2 + r_2^2(t) - r_3^2(t)}{2Y_3} \\
Z^2(t) &= r_2^2(t) - X^2(t) - Y^2(t)
\end{aligned}
\tag{9}
$$

4. The relative error involved in the determination of v from measurements at one station is given by

$$
\begin{aligned}
\left(\frac{\Delta v}{v}\right)^2 = \left(\frac{\Delta c}{c}\right)^2 &+ \left(\frac{\Delta f_0}{f_0}\right)^2 + \\
&+ \left(\frac{1}{2}\,\frac{\Delta\left[(t_1 - t_0)^2 - (t_2 - t_0)^2\right]}{(t_1 - t_0)^2 - (t_2 - t_0)^2}\right)^2 + \\
&+ \left(\frac{1}{2}\,\frac{\Delta\left[\left(\frac{t_1 - t_0}{f_1 - f_0}\right)^2 - \left(\frac{t_2 - t_0}{f_2 - f_0}\right)^2\right]}{\left(\frac{t_1 - t_0}{f_1 - f_0}\right)^2 - \left(\frac{t_2 - t_0}{f_2 - f_0}\right)^2}\right)^2
\end{aligned}
\tag{10}
$$

An analysis of this expression, which follows from (4), shows that only the last term is significant, if the error in time measurements is not larger than one millisecond. The first term is interesting since it indicates what variations in the refractive index of the atmosphere are important for the result.

$$n c = c_0$$

$$\frac{\Delta^2 c}{c^2} = \frac{\Delta^2 n}{n^2} + \frac{\Delta^2 c_0}{c_0} \tag{11}$$

If n changes from $n = 1.00000$ to $n = 1.00029$ as the signal penetrates the atmosphere, $\frac{\Delta n}{n}$ is in the order of 3×10^{-4}. Hence, only if $\frac{\Delta v}{v}$ shall be smaller than that, the systematic change in refractive index with altitude has to be taken into consideration. Any erratic changes, layers of temperature inversion and the like will have hardly any effect unless small angles of incidence are being used. In the present application, angles smaller than 30° would not be anticipated.

The fourth term of (10), carried out, has the following form

$$\frac{\left(\frac{(t_1-t_0)^2}{(f_1-f_0)^3} - \frac{(t_2-t_0)^2}{(f_2-f_0)^3}\right)^2 \Delta^2 f_0 + \left(\frac{(t_1-t_0)^4}{(f_1-f_0)^6} + \frac{(t_2-t_0)^4}{(f_2-f_0)^6}\right)^2 \Delta^2 f}{\left[\left(\frac{t_1-t_0}{f_1-f_0}\right)^2 - \frac{(t_2-t_0)^2}{(f_2-f_0)^2}\right]^2} \tag{12}$$

if again the terms accounting for the errors in time measurements are

[redacted]

neglected, and $\Delta^2 f = \Delta^2 f_1 = \Delta^2 f_2$. As an example it has been assumed that f_1 has been measured when the target is seen under an angle of 30° with the surface of the earth, f_2 when the angle is 60°, f_3 and f_4 somewhere in between. The speed was assumed 5 mi/sec. In that case

$$\frac{\Delta v}{v} = 1 \times 10^{5} \frac{\Delta f}{f} \tag{13}$$

has been found, a figure which is true whether the object passes the station at a distance of 200 mi or 800 mi. Hence, if the frequency can be measured with an accuracy of 2 parts in 10^8,

9

$$\frac{\Delta v}{v} = 2 \times 10^{-3} \tag{14}$$

A similar consideration for the errors in the determination of the shortest distance "a" gives for the same example

$$\frac{\Delta a}{a} = 10^{-2} \tag{15}$$

This applies for the evaluation of only four measurements. In reality, however, ca 130 measured points are available for evaluation, if $a \leqq 180$ mi and more than 500 if $a \leqq 800$ mi. Application of the least square method to that many points will eliminate practically any random error. For instance the mean errors of the results, obtained by using 8 measured values are already down to

$$\frac{\Delta v}{v} = 0.1\%$$
$$\frac{\Delta a}{a} = 0.5\% \tag{16}$$

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The errors in the determination of the position of the object as a function of time, from measurements at three stations can easily be found from (9) to

$$\Delta^2 X = \frac{1}{4}\left(1 - \frac{h_2^2 - h_1^2}{X_1^2}\right)^2 \Delta^2 X_1 + \frac{a_1^2}{X_1^2}\Delta^2 a_1 + \frac{a_2^2}{X_1^2}\Delta^2 a_2$$

$$\Delta^2 Y = \frac{1}{4}\left(1 - \frac{h_2^2 - h_3^2}{Y_3^2}\right)^2 \Delta^2 Y_3 + \frac{a_1^2}{Y_3^2}\Delta^2 a_1 + \frac{a_3^2}{Y_3^2}\Delta^2 a_2 \tag{17}$$

$$\Delta^2 Z = \frac{a_2^2}{Z^2}\Delta^2 a_2 + \frac{X_1^2}{Z^2}\Delta^2 X + \frac{Y_3^2}{Z^2}\Delta^2 Y.$$

5. So far it has been assumed that the trajectory of the object is a straight line and that $v \equiv$ const. In the case of a satellite to the earth, however, the trajectory is an ellipse and the velocity depends upon the position. Here again, the general law of motion is known and it will be possible to derive similar relations between the frequency of the received signal and the parameters of interest. Even though the mathematical expressions will be more complicated, the basic accuracy of the results is in the same order of magnitude. This has been proven in an analysis of the relations for a circular trajectory, which, however, shall not be presented at the present time.

6. It can be concluded that the measurements of the Doppler shift of a signal emitted from a body whose motion is governed by a known law allows very accurate determination of the velocity of that body and

at the same time to define the position of the trajectory within fair limits.

7. If this method should be applied to the determination of the velocity and the path of an artificial satellite, it would be necessary to equip the satellite with a transmitter, stable to within one part in 10^8 during the time of the measurements at one set of stations, and capable of delivering around 5 milliwatts minimum of radiated power. Reference is made to Section II, 3.e. "Frequency Control Devices" for details concerning the feasibility of such a device.

11 8. It would further be necessary to set up a number of receiving stations which are capable of measuring the frequency of a weak signal accurately to within a few parts in 10^8.

In the following, one possible set-up is described which would be suitable for this application and allow measurement of the signal up to distances of around 1400 mi.

(1) Although this method does not utilise any angular information and hence could use in principle an antenna with radial symmetry, a high gain antenna having 30 to 50 square feet collecting area would have to be employed in order to assure good reception of the signal up to large distances. This antenna can either be slaved to a tracking antenna or have its own tracking mechanism. One of the antennas as for instance proposed for use in connection with RADIV

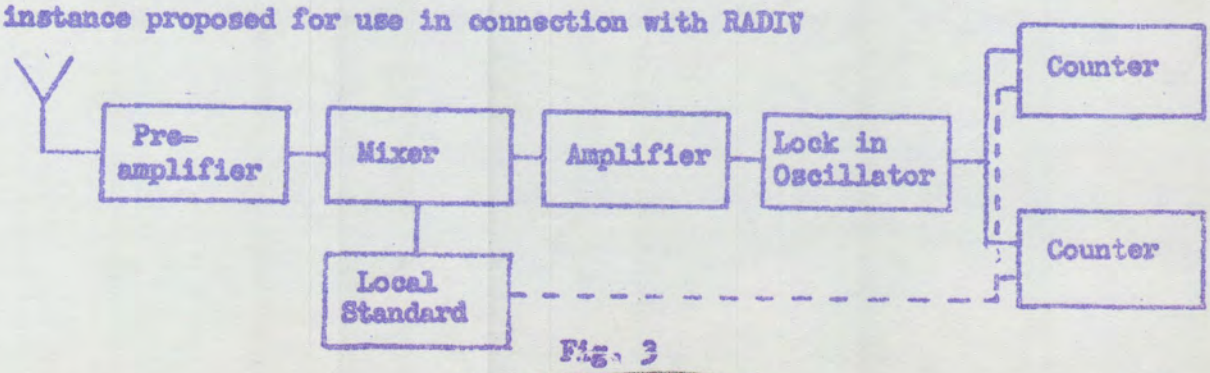

Fig. 3

system (see Section II, 1.c.(1)) would be adequate for this application.

(2) The signal received by the antenna would be preamplified and mixed with the signal of the local standard. The resultant IF signal having a frequency of around 2 mc would then be amplified to ca 1 volt in a high gain amplifier with a band width of 5 to 10 kc and fed into a lock-in oscillator. This device serves as a variable frequency filter with a bandwidth of a small fraction of one cycle and at the same time eliminates any amplitude modulation of the incoming signal. (Reference is made to Contract No. NObsr-64245, Index No. NE-020237, nb Task No. 11.)

The output signal of the lock-in oscillator has the same frequency as the incoming signal, but a greatly improved signal to noise ratio and could thus be directly applied to a frequency counting device. Two
12 frequency counters could be used, arranged such that one is counting while the other prints. The sampling period would be one second whereby the local frequency standard would provide the time base.

(3) If each count is associated with the time bisecting the counting period this method would allow measurement of the frequency of the incoming signal to better than two parts in 10^8.

(4) One observation site may consist of three receiving stations arranged in a triangle and approximately 200 mi apart from each other.

The number of observation sites to be set up would depend upon the orbit (equatorial or other) and upon the desired degree of coverage.

(5) The power consumption of the equipment at one receiving station would probably not exceed two kilowatts. A line voltage of 110 volt ac would be sufficient.

(6) Radio communication between the three stations of one site and between the various sites would be necessary for the purpose of synchronization of the frequency standards and for the exchange of information.

9. (1) The reduction of the data as obtained from the various stations could be done with the help of an electronic computer at some convenient time after the measurements have been taken.

(2) The research and development required in connection with the transmitter in the satellite is covered in section 3.e.

(3) Practically no developmental work would be required for the electronic equipment to be used at the ground stations, since all of the components are commercially available or can be furnished under existing Government contracts.

(4) The developmental work on the receiving antennas would be
13 the same as described under Section II, 1.c.(1) for the antennas, proposed in connection with the RADIV tracking system.

(5) The total cost of the electronic equipment for a set of three observation stations would be in the order of $50,000.

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

2. Communications Systems for Data Transmission, Correlation and Processing

a. Communications Requirements

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I

2. Communications Systems for Data Transmission, Correlation & Processing

a. Communication Requirements

Prepared by: Communication Department

SYNOPSIS: Existing communication equipment, or communication equipment which will be available within two years, will fulfil communication requirements needed for proposed tracking systems. The cost for these equipments both in time and money will depend upon the possible integration of this system into existing Army Service Communication systems.

1. Introduction.

Communications required with proposed systems will be very extensive, both because of the wide geographical limits involved and because of the possibilities of communication being the first use which may be made of the satellite.

1. The immediate communication requirements will be in conjunction with the tracking instrumentation. Next all scientific tests will require communications between the various geographical stations not only in connection with the inter-station coordination, but between two or more stations of individual radar sets which may be involved in a triangulation type of tracking.

The Signal Corps has had extensive experience with this type of communications for missile ranges. The most analogous system of this type is that which the Signal Corps has provided in connection with the White Sands Proving Ground. Next the Signal Corps had the responsibility for the initial programming of a communication system for what is now known as the Long Range Air Force Guided Missile Range located at Coco, Florida. Also data transmission terminals have been developed and are entering pro-

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duction for use in data transmission systems for the anti-aircraft defense systems conducted under Signal Corps project 414.

In this the contemplated system the Signal Corps Engineering Laboratories will act as a consulting communication service with the expectation that the established Signal Corps operational units such as the Army Communication Service Division will operate the required communication system with personnel trained possibly at the White Sands Proving Ground. Also wherever possible existing global communication systems of all services will be utilized.

In accordance with this policy of attempting to use existing communication systems the selection of observation sites will take into account the availability of existing communication centers as well as the other
2. necessary items such as optical observatories.

Also existing transmissions such as those emanating from the Bureau of Standards Station WWV, various navigation stations, as well as both the American and Foreign stations such as the Navy low-frequency station at Jim Creek, Washington State and the British low-frequency timing standard station will be utilized as much as possible in connection with timing functions required in the network.

While this policy will be pursued as closely as the techniques involved allow, it is possible that as this project continues the accuracy required may be such as to require a data type of communication much more precise than the conventional systems provide. This has been found to be true at the Air Force Long Range Missile station at Coco, Florida. At this range, for instance, the Air Force has had to resort to a multimillion dollar submarine cable for communication purposes. The wide geographical distributions of the contemplated satellite system which encompasses the

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whole earth around the equator and ranges of ±30 degrees latitude, though, will require radio communication. The vagaries of radio transmission propagation may force the use of new propagation techniques or modifications of existing equipment as the tracking instrumentation is improved.

2. Immediate Communications System Requirements.

a. Communication between observatory stations. This type of communication can be obtained by the use of high-frequency radio teletype transmissions. The details as to the number of site locations, types of equipment, and the other logistics will depend upon the locations relative to existing world-wide networks. The majority of the communications, though, will be of the usual command, and housekeeping variety, and even the data obtained by optical or electromagnetic means will be communicatable on teletype speeds available in existing conventional radio sets and conventional transducer equipment. The data involved will be transmitted to a computing center, and the results re-transmitted by the same method.

b. Communication for radar tracking system RADIV proposed by SCEL. The communication involved in this type of radar tracking system will be met by standardized teletype equipment in conjunction with conventional radio and wire equipment.

If radar shaft position data is eventually required to be transmitted directly, Coordinate Data Sets AN/TSQ-7 and/or AN/TSQ-8 are scheduled for production. These data sets were engineered for use with existing equipment and provide methods for correcting for transmission introduced errors of parallax and earth curvature. Although these data equipments are usable with conventional high-frequency radio sets the accuracy required is limited by propagation phase changes occurring for sky-wave radio

transmission paths. Such propagation phase changes may be in the order of 2-5 milliseconds while the AN/TSQ coordinate data sets depend upon a time-division method of multiplexing which provides a form of pulse modulation in which basic time slots for individual pulses are only 1-1/3 milliseconds duration. The terminals themselves, though, provide accuracy of transmission over a distortionless medium of one part in 1000 within a range of ±50 volts or ±100 volts of DC analog voltages.

An important advantage of the use of existing teletype circuits is the fact that security equipment is also available for encrypting such communications. While the data transmissions may not be worthwhile encrypting from a political viewpoint, it is felt that the command and other transmissions which may provide the enemy with information as to the status of the research, should be encrypted with the highest security devices possible. This requirement in itself necessitates the use of conventional teletype methods.

4.

c. Timing Signals. The precise timing required at the various observatories as well as at the central computing station can be obtained by placing a highly stable electromagnetic frequency oscillator device at each station, and periodically calibrating this device by resorting to existing world-wide precise timing transmissions. This would merely require radio receivers at each of the stations. These techniques are well established.

The accuracies available from molecular resonances established at such places as the Bureau of Standards are currently being incorporated as a reference for such timing transmissions.

It is to be emphasized that the various geographical stations established for this system will, over a period of time (weeks), by listening

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to these various frequency standard transmissions be able to compensate and correct for the propagation difficulties and errors introduced. Establishment of such stations for an extended period of time therefore essentially provides accuracies at each establishment comparable, if not exactly equal, to the accuracies available at the locations of the primary standards. Even without benefit of the molecular resonance phenomena, stabilities of one part in 10^8 are immediately available.

3. Time and Funding Requirements.

5.

a. The communication equipment required is readily available, either in the operating units of Army Communication Division, or in depot stocks. This is based upon the use of standard Signal Corps teletypewriter and high-frequency radio equipment.

b. It is expected that the Signal Corps' Army Communication Service Division will have to be augmented by additional service personnel in order to man the communication stations, at least where additional stations are required beyond those already in the Army Communication Global network. Personnel of this type have standard MOS numbers, but it may be desirable, in order to keep the additional personnel required to a minimum, to consider the communication personnel requirements in conjunction with the requirements of other operations of the tracking equipment.

c. Wherever a new station must be established, a minimum amount of additional communication equipment will be required. This minimum requirement will be at least a Radio Set AN/GRC-26, or its equivalent. This unit currently costs between $15,000 and $20,000. It is complete not only with radio and teletypewriter equipment, but also includes a small truck-transportable, shelter and primary power unit. The additional equipment required will include a low-frequency receiver and timing equipment. This

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equipment will cost approximately $5000.00 per station.

Each station established outisde the existing Army Communication Global Network will therefore require a minimum of $25,000.00 worth of communication and timing standard equipment.

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

2. Communications Systems for Data Transmission, Correlation and Processing

b. Data Processing

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I
2. Communications Systems for Data Transmission, Correlation, and Data Processing
b. Data Processing

Prepared by: Physical Sciences Division

SYNOPSIS: Data processing in the satellite program falls under two categories:

a. Furnishing of sufficiently precise predicted data to observers to enable them to position equipment having a narrow angle of view for the purpose of obtaining more accurate position data.

b. Deducing from all obtained observational and telemetered data scientifically valuable results. A discussion of computer requirements in the system is presented together with possible computers which may be available for performing required functions.

1. Discussion:

Two objectives of the satellite program appear to be obtainable if careful consideration is given to the orbit of the satellite. These objectives are:

a. Furnishing of sufficiently accurate predicted data to observers to enable them to position equipment having a narrow angle of view for the purpose of obtaining more accurate position data.

b. Deducing from all obtained observational and telemetered data results of scientific importance.

Prediction of future position of the satellite must be accomplished on the freshest data available in order to insure maximum tracking capability. This is especially important if narrow angle of view device are utilized for obtaining information, since prediction errors larger than half the angle of view jeopardize tracking completely. This sort of prediction calculation would consist of

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1. Discussion: b.(Contd)

a discrete number of computations (depending on the number of observing sites) for each passage around the earth. Thus each observing site would receive a fresh position prediction every 1½ hours or less. Only a significant meteoric collision should cause tracking to degrade or fail with such fresh data always available. (See Appendices). At any one time the position will be computed from an instantaneous ellipse which is tangent to the variable conic in which the satellite is actually travelling.

If a RADIV system of tracking is used, a communications system with a capacity of 32-50 bits of information per second per channel is required. For a single RADIV station (consisting of two sites 800 miles apart) communication links between sites as well as to a central computer from each site are required. All
2. RADIV data, as well as optical and telemetered data should be recorded on site by magnetic recording devices, paper tape punching equipment, or IBM card punches.

The raw data obtained by radars will be transmitted to a central computer which will perform the necessary coordinate conversions and perform the calculations necessary to determine future position. These positions will then be transmitted to all observation stations together with timing information for their use in acquiring the satellite.

The computer to be used in Phase I of the IGY program must be a presently available computer. There are several general

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1. Discussion: b. (Contd)

purpose high speed computers whose characteristics are such as to permit integration into the system with few modifications and additional equipment to make possible compatibility with the communication equipment. Two machines which are government owned and which are similar in design are FLAC and DYSEAC. The DYSEAC is presently under Signal Corps control and is transportable. It is housed in vans and may relatively easily be moved to any part of the United States. It is presently in operation at White Sands Proving Ground and if an orbit of 30° to 45° inclination is chosen this site would well be a suitable one for the satellite program. The FLAC is located at Patrick Air Force Base, Florida, and is also situated satisfactorily for the satellite program. Another machine of the same family is the MIDAC at the University of Michigan, Ann Arbor, Michigan. It is proposed that attempts be made to determine whether these machines can be made available for this program. It is proposed to use two computers for the purpose of providing prediction information so that a breakdown will not cause the flow of prediction information to be interrupted at any time.

Other computers which may be available on a rental basis are the IBM 701 at IBM World Headquarters and the UNIVAC at Sperry Rand, New York City. The rental costs on these machines are $300 per hour. In addition it is proposed to determine the availablity of the Lincoln Laboratory AN/FSQ-7 which was designed with reliability as the prime consideration. This equipment was

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1. Discussion: b. (Contd)

designed for real time operation and might be the most suitable from an over all view point.

The problem of reducing and analyzing all data for scientific purposes may be handled in delayed time. For these purposes it may be possible to enlist the services of the National Bureau of Standards. If the computational facilities of the Signal Corps Engineering Laboratories are increased to the extent of acquisition of an ERA 1103 computer it will be quite possible to perform all the delayed scientific computations at SCEL.

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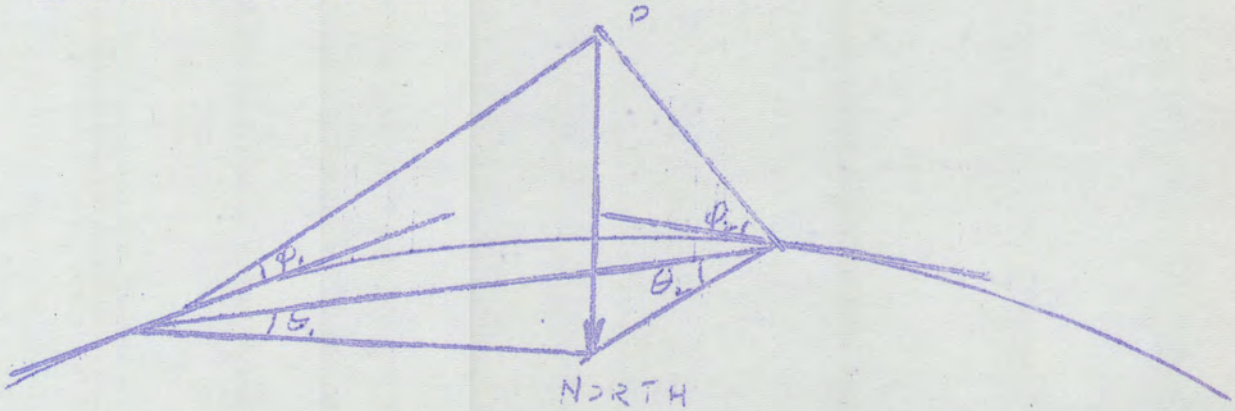
APPENDIX A

DISCUSSION OF COORDINATE SYSTEMS AND REAL TIME COMPUTATION PROBLEM

Assume two direction finders determine θ and ϕ -each in their local coordinate systems where the tangent plane yields $\phi = 0$ and θ is azimuthal direction measured from true north.



The coordinates θ_1 and θ_2 define two planes in space, and hence their intersection must be a straight line on which the point must lie. Either coordinate will now define a conical surface which will intersect the line at the point P. Thus we can see that an over-determined solution is possible since only one ϕ coordinate is absolutely necessary.

In order to trigonometrically solve even the single case ignoring the redundary in data it is necessary to transform the measured angles into a coordinate representation which is free of the local restraint. The question arises as to which coordinate frame we should choose.

We can choose a frame fixed with respect to the earth and with the origin at the earth's center.

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We can choose alternately a frame fixed with respect to the distant stars and with the origin at the earth's center

Several considerations favor the latter choice. First of all, the satellite motion is such as to ignore the rotation of the earth's and except for the advance of perigee due to the oblateness of the earth and more minor effects due to drag the elliptical orbit should tend to remain constant with respect to the distant stars. It should be noted that computation carried out in such a reference plane will serve no purpose unless transformation to a local coordinate system is effected. However all transformation from the fixed-in-space coordinate system to the local coordinated systems are identical in form and if the reference direction in longitude in the astronomical frame is made to coincide with the Greenwich meridan at an arbitrary zero time then the geographical coordinates of different observing points are the only constants which differ in the transformation to all local frames.

Let us consider that the tracking system gives us a set of points $P_i, P_2, \ldots, P_j$ at times $T_i, T_2, \ldots, T_j$. How do we best use this set of observational data after transformation into the astronomical frame so as to permit future position prediction? In the RADIV system this set of points may consist of 40 points separated by one second in observation time or approximately $\frac{40}{60 \times 90} \times 100\% = 1\%$ of the orbit. Each point has equal precision of measurement and should be weighted accor accordingly. Actually it would be much simpler if we were given only one point P and the vector velocity of the satellite at this point. Then we could consider all known acceleration and numerically integrate the motion to obtain the displacement from P. However we present a

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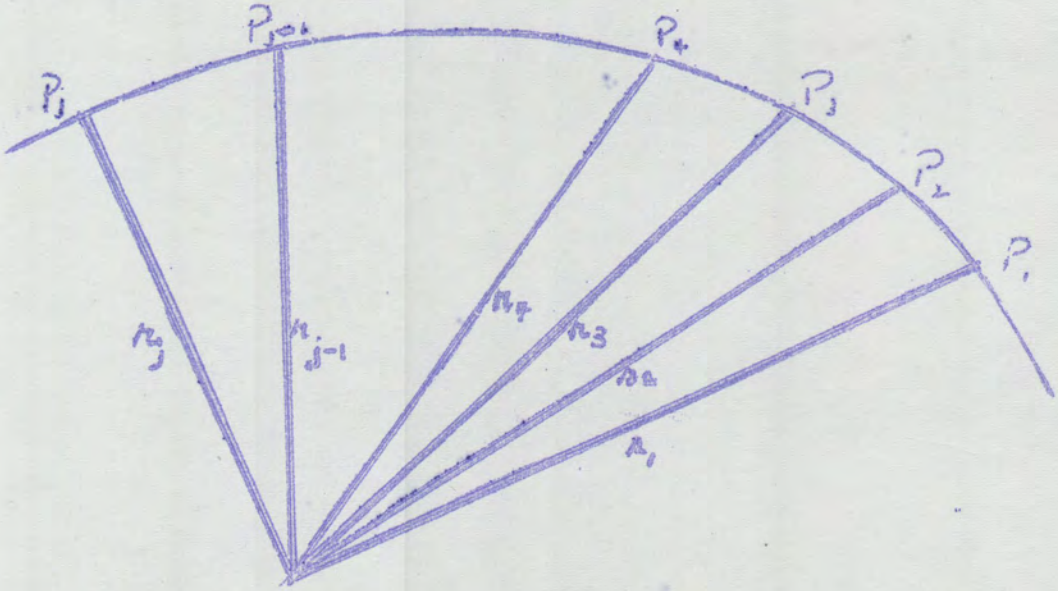
method for giving rather equal weight to the 40 points.



One of Kepler's Laws states that in elliptical motion of a planet, equal areas are swept out by the planet in equal times. A set of 40 points at 40 equally spaced times would enable 39 equal area segments to be computed. The mean of these 39 points would give equal weight to each observation and enable the determination of a mean areal velocity for the section of track. The angular momentum $\underline{l}$ of the satellite is given by

$$l = |r \times \mu \dot{r}| = 2\mu\dot{A}$$

Thus if we know the satellite mass, and the areal velocity $\dot{A}$ we know an important constant of the motion, the angular momentum.

We now compute the total energy. The kinetic energy

$$T = 1/2 \frac{l^2}{\mu r^2}$$

is computed for each point P. The potential energy is

$$V = \frac{G(\mu M)}{r}$$

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where G = gravitational constant
μ = mass of satellite
M = mass of earth.

The total energy is $E = T + V$. We compute a mean E from the 40 observations giving equal weight to each.

The ellipticity of the orbit is then given by

$$e = \sqrt{1 + \frac{2E\ell^2}{K^2 \mu}}$$

where $K = G\mu M$. We now have another parameter of the motion. Furthermore the major axis is given by

$$a = \frac{K}{2E}$$

The period of the orbit is given approximately by

$$\tau = \frac{2\pi\ell^3}{K^2\mu}(1 - e^2)^{-3/2}$$

The total mean of the ellipse may be obtained from

$$A = \dot{A}\tau.$$

These quantities will permit us to predict the ephemesis.

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APPENDIX B

INACCURACIES IN ORBITAL ELEMENTS RESULTING FROM INACCURACIES IN OBSERVED QUANTITIES

Type of Observation Station / Quantity	One Station Radar & One Cinetheodolite or Three Station Radars or Two Cinetheodolites		Two Stations Highly Accurate Visual Observations (Photograph with long focus camera against dark background.	
	Cinetheodolite Acc. 0.2' error in arc Radar Range Acc. 300 ft.		Accuracy 0.01" error in arc	
Time observed	1 sec.	1 min.	1 sec.	1 min.
Height	300 ft. = 0.02%		5 ft. = 0.0003%	
Latitude of Station	60 ft. = 0.5"		1 ft. = 0.01"	
Longitude of Station	75 ft. = 0.6"		1 ft. = 0.01"	
Time	0.001 Sec.		0.001 Sec.	
Velocity	0.062 m/sec. = 1.3%	0.001 m/sec = 0.02%	0.001 m/sec = 0.02%	0.00002 m/sec = 0.0003%
Distance from earth's center	54 mi = 1.3%	1 mi = 0.02%	1 mi = 0.02%	80 ft. = 0.0003%
Eccentricity	0.018	0.0003	0.0003	0.000005
Inclination	0.2°	0.003°	0.003°	0.00005°
Mean Motion	2% = $2X10^{-6}$ sec	0.03% = $3X10^{-8}$ sec	0.03% = $3X10^{-8}$ sec	0.0005% = $5X10^{-10}$ sec
Period	2% = 1.0 sec	0.03% = 2 sec	0.03% = 2 sec	0.0005% = 0.03 sec
Perigee	100 mi = 50%	2 mi = 0.5%	2 mi = 0.5%	0.03 mi = 0.001%

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

3. Materials, Electronic Parts and Subsystems

Summary to a,b,c,d and e.

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE 1
3. Materials, Electronic Parts, Subsystems

Prepared by: Components Department

SYNOPSIS: Proposed participation of the Components Department in the fields of Power Sources, Precision Oscillators, Materials and Parts Specifications, Fabrication and Environmental Testing.

1. The proposed participation of the Components Department in the satellite program is presented here in three parts. The first part contains a list of subsystems, components, materials and services which this Department is prepared to furnish in an extended program including many launchings, etc. The second part constitutes a fairly definite plan which has been cast so as to fit the present plans of the Department of Ordnance, and which would have to be, and could be modified to fit any other specific satellite plan. The third part, which is separate from the first two, consists of the assembly of the several Branch reports on their respective specialties. (see 3a, b, c, d, and e).

2. In a long range program the Components Department is prepared to make contributions in the following fields:

a. Power Sources.

b. High Precision Oscillators.

c. Materials and Electronic Parts.

3. Power Sources.

Various sources of electrical power may be presently supplied for a satellite with a minimum of R&D effort. Among these, the most promising are:

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a. Silver peroxide alkali zinc batteries, which can deliver, at a low drain rate, a total energy of 55 watt-hrs per lb at 32°F, and of 50 watt-hrs per lb at 0°F or 80°F, at approximately 1.5 volt per cell. The only work required will be some experimentation with sealed containers designed to withstand the small internal vapor pressure of the electrolyte against the external vacuum.

b. Silicon Solar batteries which, in accordance to recent statements received from BTL, could be supplied as an added task under one of SCEL contracts, and have the following technical characteristics: An efficiency of at least 8% of incident sunlight, so that cells distributed symetrically around a sphere which is illuminated two-thirds of the time would deliver an average power of 1 milliwatt cm^{-2} at the expense of a weight, including housing, of .8 gr cm^{-2}. This power figure makes allowance for a storage battery acting as a sump and for several diodes to protect the battery from discharge into the non-illuminated cells, with an overall efficiency of 65%. At the present time, a battery of the Ni Cd type appears most promising for this, and a study is being initiated to determine how many 100 minute cycles of charge and discharge may be reasonably expected from it.

c. Thermoelectric generators should be mentioned here, because they constitute a potentially reliable, if inefficient, source of power based on an old established principle. Thermoelectric generators for a satellite can be divided into two classes, those which derive their source of heat from the sun, and those which derive it from a nuclear source, and constitute thus a form of primary battery because they have a predetermined life. They can also be classified in accordance with the particular pair

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of metallic alloys utilized to generate a thermo-electric voltage.

Two classes of thermoelectric systems may be mentioned here, those based on a Zinc Antimony alloy, and those based on the Chromel-Constantan couple.

The first system has the relative advantage of higher efficiency than the second, but the disadvantage of involving very brittle metallic compounds, and a considerable R&D effort appears required here, to determine whether their assembly can be protected sufficiently against shock. This R&D effort is not believed advisable at this time.

The Chromel-Constantan couple can deliver a voltage of .01 volt per junction with a temperature differential of 155°C. This temperature differential is believed to be easily obtainable during the satellite's "daytime" by placing the proper finish on various surface segments, a point which will be discussed under "materials". On the other hand, some 2000 couples would be required to deliver an open-circuit voltage of 20 volts, which would be reduced, under load, to the 15 odd volts which are desired, as will be seen in the subsequent discussion of precision oscillators. A further disadvantage of the solar thermopile lies in the necessity of having rectifiers which protect the storage battery from discharge during the satellite "nighttime".

An interesting possibility is offered by the combination of a Nuclear heat source in which Polorium 210 is the fuel, and a thermopile. With the latter based on the Chromel-Constantan system, a total energy delivery of 500 watt hours per lb, with an exponential voltage decay, has been obtained in one experimental unit. This system appears promising for

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future applications, and will be further explored.

4. High Precision Oscillators.

The design of extremely light directly quartz crystal controlled precision oscillators up to 150 Mc operating with a very small power is essentially an engineering problem requiring a minimum of R&D work, as most of the ground work has been done already in connection with several Signal Corps projects. The frequency range just mentioned can be extended by frequency multiplication.

Temperature stability constitutes an important aspect of this work, and a temperature stability of the order of $\pm .01^{\circ}C$ is required, if a frequency stability of the order of 1 p. in 10^9 is to be met. Placing the temperature control point at the melting point of a substance such as water-ice or Glauber's salt has yielded very promising results, and studies aimed at the application of this principle for the stabilization of the temperature of a satellite oscillator constitute a natural extension of this department's effort in this field. Another approach will be considered which consists in placing the precision oscillator inside a well heated insulated chamber with a well defined thermal conductivity to the satellite shell, so that the dissipation of a fairly definite wattage inside this chamber will determine a fairly definite temperature for it with very small day to night fluctuations.

Voltage stability constitutes another important aspect of this work. In many practical circuits a fractional voltage variation ε has been observed to cause a fractional frequency variation of the order of $10^{-6}\varepsilon$. This can be translated immediately in terms of a battery

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requirement in which a voltage variation of less than .1% in ten minutes could constitute a minimum requirement for the determination, by means of the Doppler frequency shift, of the variation of the radial velocity of the satellite with respect to a single observing station, during one pass over this station, with an accuracy of 1 ft sec^{-1}. (1 p. in 10^9 of the speed of eight). This voltage stability aspect will be especially scrutinized when a suitable satellite battery is selected.

An initial study of possible circuits has indicated that 15 volts would constitute the minimum practical energizing potential, albeit a higher voltage would be desirable if practicable.

A brief survey of the ultra-miniaturization required for satellite oscillators and emitters has indicated that, in the 125 Mc region, and in the absence of temperature and voltage stabilization, an oscillator with a single transistor will consume the order of .15 watt, and have a stability of 50 ppm. Conversely, a four transistor oscillator with a .4 watt consumption, could have a stability of the order of 10^{-9} in ten minutes, provided the temperature and voltage requirements outlined above are met.

In the frequency range of 250 Mc, it is too early to predict whether a PNIP transistor or a subminiature tube such as #5677 will be best for efficient operation. At the present time the best available PNIP transistors are marginal in this range, but the latest information received on experimental PNIP's indicated that they are strong runner-ups in this frequency range.

To sum up the situation with regards to precision oscillators, most of the work anticipated for satellite oscillators will be of a

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development and engineering character, with a minimum of research work. The same remarks apply a fortiori to the precision oscillators which may be required by the listening and tracking stations on the ground.

5. Materials.

Consulting services on materials, as well as fabricating and testing services will be supplied by this department in a general satellite program along the four following lines:

a. Based on available knowledge of invironmental conditions of radiation, temperature, and mechanical shock at launching, this Department will specify what materials, especially non-metallic materials, will have the desired mechanical and electrical properties for adequate performance and life.

b. Experimentation will be initiated aimed at obtaining maximum light reflecting properties for a given weight of material. Two systems are contemplated at this time. In one system, white foaming polymers will be formed once the satellite is in its orbit, so that the latter will assume the shape of an enlarged sphere (or disc!) having good diffuse reflectivity. In the other system, large, thin reflecting sheets of metal will be unfolded, after launching, and will remain unfolded by virtue of the mutual electrostatic repulsion induced by a small radioactive charge in the satellite. Both systems are contemplated for satellites having a perigee higher than 300 miles above the earth, and for which the braking action of the exosphere will be negligibly small, and will not be one of the parameters for which exact measurements will be sought.

c. Experimentation has been initiated to develop different

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kinds of surface finishes with selective absorption and reflexion properties in the solar spectral region, which has its peak at around .5 micron wavelength and in the thermal region, which has its peak at 10 micron wavelength for a body at room temperature. Several years ago, tests undertaken at SCEL in this domain showed that a copper cylinder painted with ordinary black enamel and placed in a partially evacuated glass tube will assume a temperature of 85°C when placed in sunlight, while in a similar experiment in which Tellurium black, which is transparent in the thermal region, is substituted for enamel black, the temperature rose to 128°C. A greater deal of experience has been acquired by and for the meteorologists with converse substances such as gypsum which are reflecting or diffusing in the solar spectral region, but emissive in the thermal region. The application of this experience will permit to engineer the thermal properties of the skin of a satellite, either to obtain large temperature differential for thermopile generators, or in order to produce predictable day and night cycles of skin temperature for the sake of prediction and control of the thermal behavior of important internal electronic components.

d. The simulation of the near vacuum, and of the solar as well as blackbody radiative cycle of night and day, which constitute the most important environmental conditions of the satellite, can be created in the large test chambers of this Department. These tests will be essential in order to determine what special surface finishes have the desired radiative properties and how the temperature of critical elements within the satellite can be stabilized satisfactorily.

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6. A specific program has been studied and discussed, predicated on the following premises, which were established on the basis of currently available information on the Ordnance plans for a satellite.

a. The satellite will have a total weight of 5 lbs., and a diameter of 20 inches; it will have an initial rotational speed of up to 1800 rpm, and will be subjected to accelerations of up to 250 g's at launching; its orbit will be equatorial and have a perigee height of 186 miles, and an apogee height of 820 miles.

b. The essential function of this initial satellite will be to be seen (rather than be heard!), and any emitting electronic equipment installed in it will be for the primary purpose of helping the initial optical search for it, first 1500 miles East of its launching point, then from near the launching point after the satellite has completed its first orbital motion, 101 min. later.

The plan proposed for the Component Department participation in this program consists of the following three points:

(1) Structural and Thermal Engineering

(2) Power Source

(3) Precision Oscillator and Transmitter

(1) Structural and Thermal Engineering. A mock-up of a proposed satellite will be built along the lines indicated in the accompanying drawing.

One purpose of the structure shown is to reduce the weight of the visible structure to the minimum consistent with negligible change in the optical appearance of the satellite, in order to spare from

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$1\frac{1}{2}$ to 2 lbs. of weight for electronic equipment. This model will be tested for strength against an 1800 rpm rotation and for resistance against a 250 g shock in the shock and vibration section of this Department.

Another purpose of the structure shown is to provide thermal time constants which are predetermined by the specific heat of certain central assemblies, and by the heat conductivity of the struts connecting these central assemblies to the outside. For instance, if a water-ice mixture is relied on to "clamp" the temperature of a precision oscillator at the triple point of water for most of the time, it may be beneficial to have relatively large temperature fluctuations reach this oscillator. On the other hand, if no liquid to solid phase is utilized, the best possible thermal insulation will serve to minimize the temperature fluctuation of the oscillator.

These considerations are complicated further by the 6% variation of the solar constant due to the eccentricity of the earth's orbit, and by the variation of the night and day periods due to the perigee motion caused by the earth's ellipticity, and due to the angle of the ecliptic.

A third purpose of the structure shown is to carry the reflecting surfaces of the satellite in the form of thin metallic (e.g. .008" Al) sectors which are riveted to the rib structure, and to carry as well temperature determining surfaces with the spectrally selective finishes which are now being studied. It is anticipated that the surface will have

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to be tailored thermally by a process of trial and error, until the simulated conditions produced in the test chambers of the Component Department determine a suitable temperature range for the precision oscillators.

A fourth purpose of the structure is to provide mounting space and attachments for four silicon solar cells, disposed as shown with their centers at the apexes of a regular tetrahedron. It can be verified easily that with this symetrical arrangement, the ratio of maximum to minimum exposure is only 1.41 ($=\sqrt{2}$) whereas it would be 1.95 ($=\frac{9\sqrt{3}}{8}$) if six cells arranged like the faces of a cube were utilized, and 1.41 again if eight cells were disposed like the faces of a regular octahedron.

The fifth purpose of the structure is to provide a flush mounted antenna, which consists of an insulated metallic strip having the shape of a trumcated sector extending between the elevation angles 25° and 65° and over 120° of azimuth, with the rotational axis of the sphere being taken as the polar geometrical axis. Thus the axis of the radiating dipole formed by this flush antenna and the rest of the sphere will be at an angle of nearly 45° from the axis of rotation of the sphere, following a suggestion from the Radar Division.

(2) Power Sources. It is proposed to furnish for the first satellite a power plant consisting of four clusters of silicon cells, several diodes and a Nickel Cadmium battery. The silicon cells will have a total effective area of 400 cm^2 and an efficiency of 8%. On the basis of a solar constant of 1.94 cal cm^{-2} min^{-1}, of a ratio of intercepted cross-section of solar radiation over total sphere area of 25%, of 65% of daylight

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(63 to 68 min. out of a 101 min. period) and of 70% overall efficiency of the storage battery and protective diodes, a power of over .4 watts should be available for the operation of the precision oscillator and transmitter. This power can be available at a voltage of 16 volts and a current of 25 milliamps; alternately, a voltage of a few times this volume, with a corresponding smaller current could be obtained.

Four dry disc diodes will serve to protect the battery from discharge into the non-illuminated silicon cells.

The battery will be an 8 oz. Nickel Cadmium battery, with a storage capacity of 6 watt-hrs. which will insure operation of the oscillator and transmitters for an initial critical period of several hours, even if the silicon cells were to malfunction due to some unforseen circumstance.

(3) Precision Oscillator. It is proposed to incorporate in the satellite, a precision 4- transistor oscillator weighing 10 oz. having a total power consumption of .4 watt, and capable of causing the antenna to radiate 5 milliwatt of c.w. 132 mc. power.

The 132 mc. frequency represents a compromise between the 50 mcs initially planned by the Components Department, and the 250 mcs initially required by the Radar Division. It is of sufficiently short wavelength to permit a direction finding set to track the satellite with a precision of $\pm 1^{\circ}$, and transmit this information by means of servoes to the tracking theodolites, in case the latter require this initial information. It will permit the Radar Division to follow through with their plans for precision interferometric tracking. It will also permit the direct

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utilization of the overtone of a thickness shear mode Quartz crystal, without having to resort to frequency multiplication.

The temperature of the crystal and its associated circuit elements will be either "clamped", for as large a fraction of the time as can be engineered, at the melting point of a substance with a large heat of melting, or stabilized by means of thermal insulation. It is anticipated that one of these methods will provide an oscillator stability of the order of 1 part in 10^9 over a period of 10 min. This stability will be aimed at the measurement of the radial velocity change of the satellite with respect to a ground listening station over which it passes, with an accuracy of 1 ft sec^{-1}. It has been established by a theoretical study that knowledge of the frequency vs. time curve of the observed signals as received from only three ground stations is sufficient to establish the satellite trajectory, and it is proposed to develop this promising method which has the large advantage of requiring no tracking system, but only fixed listening stations with antennae having limited directivity. Reference is made to the Frequency Control Branch write-up for a more detailed description of this system.

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PROPOSED SATELLITE CONSTRUCTION

1. Reflecting Aluminum Surface
2. Twelve Aluminum Ribs
3. Solar Cells
4. Flush Mounted - 125 mc. Antenna
5. Thermal Conduction Members
6. Thrust Members
7. Aluminum Girdles
8. Battery and Oscillator Housing

Total Weight: 5 Pounds - - - - - Diameter: 20 Inches

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

3. Materials, Electronic Parts and Subsystems

a. Materials

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11. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE 1
3. Materials, Electronic Parts, Subsystems
a. Materials

Prepared by: Components Department

SYNOPSIS: The following factors of the service conditions and the environment control the selection of materials to be used in the satellite: temperature, solar x-rays, ultraviolet light, cosmic dust, high vacuum and stresses set up by mechanical shock.

If the outer surface of the satellite is made of aluminum 1/32 inch thick, the x-rays, ultraviolet light and cosmic dust will be effectively excluded from the interior, and no damage will therefore be expected to materials within the structure from these environmental factors. The temperature and vacuum will therefore be the only important factors which must be considered when selecting materials for use within the sphere. The high vacuum which is present will limit the choice of materials to those which are composed of substances which will not vaporize under these conditions. Since the temperature inside the satellite can be controlled to some extent by utilization of the radiant heat from the sun and the earth, the choice of materials will be made in accordance with the operating temperature range so established. Based on these temperature requirements, assuming that the proper design has been selected, the following materials may be utilized within the satellite, if they do not contain constituents which will be vaporized under vacuum:

TABLE I

Temperature Assumed Within the Satellite	Recommended Materials
At -100°C (-148°F)	Ceramics; glass; all metals & alloys; thin Oxide Coatings as electrical insulation
At -80°C (-112°F)	All metals & alloys; ceramics; glass; thin oxide coatings as insulation
At -60°C (-76°F)	All metals & alloys; glass ceramic; polyethylene; Teflon; epoxy resins; and laminated thermosetting resins.
At -40°C (-40°F)	Polyethylene; Teflon; epoxy resins; phenolic

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Temperature Assumed Within the Satellite	Recommended Materials
	resins; melamine resins; polystyrene; laminated thermosetting resins; polyesters; ceramics; glass; all metals & alloys.
-20°(-4°F) to 100°C(212°F)	Polyethylene; Teflon; epoxy resins; polystyrene; polyesters; phenolic resins; ceramics; glass; metals & alloys; laminated thermosetting resins; & melamine resins.

Many materials used on the external surface of the satellite may be expected to be damaged by ultraviolet light, solar x-rays, temperature effects or by volatilization under vacuum. Metals, alloys, asbestos, mica and ceramics are recommended as materials to be utilized on the outer surface of the satellite. Although plastics are not recommended for use on the outer surface, should it be necessary to utilize materials, such as polyester-fiberglas, caution should be exercised to specify that the resins be light and x-ray stabilized and that they contain no constituents which will volatilize under vacuum. The sandblasting effect of cosmic dust (micrometeorites) would be expected to cause the hazing of any unprotected optical surfaces exposed to the external environment. Damage from meteors is expected to be extremely unlikely.

The stresses resulting from the mechanical shock conditions can be readily accomodated by many existing alloys and metals.

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a. Environmental Factors and their Effects on Materials.

(1) Temperature

The temperature of the surface of the satellite can be controlled within certain limits by a proper choice of the materials composing it. However, temperatures as low as -70°C or -80°C are anticipated, and therefore, to provide a margin of safety, materials listed in Table I include those suitable for use at -100°C. At temperatures of this order, plastic insulation can crack and fall off metallic conductors because of brittleness and stresses set up by differential thermal expansions and contractions. Reference is made to Table I for a suitable choice of materials for use under these conditions.

The internal temperature of the satellite can be controlled by balancing the radiant heat from the shell with that dissipated by the transmitter and other equipment. Since this temperature will be higher than that of the surface a wide scope of selection of materials is possible as lised in Table I.

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(2) Vacuum (Pressure)

According to published data the vacuum (pressure) at the level expected to be traversed by the satellite will be in excess of 10^{-10} mm of Hg. (2)

The above vacuums are in general lower than any obtainable with ordinary laboratory equipment, which range in the order of 10^{-5} to 10^{-8} mm. of Hg (24) (4) (3). For this reason it was necessary to extrapolate existing data to determine if certain metals or plastics would vaporize under these extremely high vacuums.

The rate of evaporation was calculated in accordance with Dushman (4) by the formula: $\log W = C - 0.5 \log T - \frac{B}{T}$, Where W = rate of evaporation in grams per sq. cm. per sec. and B & C are constants. It was found that the rates of evaporation of metals were not sufficient to cause significant losses by vaporization. Plastics in general were found, by extrapolation of vacuum metallizing data for resins, to be non-volatile at these vacuums. Plastics may, however, contain volatile constituents, such as plasticizers, anti-

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oxidants or light stabilizers, which could vaporize, under vacuum, and would therefore result in damage to these materials. Materials containing such volatile constituents should be avoided.

(3) X-rays

Soft x-ray intensities of the order of 0.1 erg-cm^{-2}-sec^{-1} from a quiet sun & as much as 1 to 2 ergs-cm^{-2}-sec^{-1} from active sun were observed by rocket measurements (1). Solar x-ray data obtained from the Meteorological Branch, SCEL were as follows:

Wave Length Angstroms	Intensity
5 to 30	3×10^{-3} erg/cm^2/sec
7 to 10	10^{-4} to 10^{-3} erg/cm^2/sec
8 to 18	0.6 erg/cm^2/sec
10 to 60	1.0 erg/cm^2/sec
Total x-ray radiation for all wavelengths =	2 ergs/cm^2/sec.

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Table II shows the stability of plastics under irradiation. (12)

A calculation of the irradiation over an entire year of one square centimeter of the outer-surface of the satellite was made utilizing the above figure of 2 ergs/cm^2/sec. Assuming that one gram of a given material were irradiated under one square centimeter of surface then:

$$\frac{(2)\ (\text{ergs})}{(\text{cm}^2)\ (\text{sec})} \times \frac{(\text{cm}^2)}{(1)\ (\text{gm})} = \frac{2\ \text{ergs}}{(\text{sec})\ (\text{gm})}$$

1 rad = 100 ergs per gram

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$$\text{Number of rads per year} = \frac{(2)(\text{gm})(\text{ergs})(365)(24)(3600)(\text{sec})}{(100)(\text{ergs})\ (\text{sec})(\text{gm})}$$

$$= 635{,}000 \text{ rads per year.}$$

By comparison of this value of 635,000 rads, with the values of greater than 10^9 rads for 50% decrease in properties in Table II, it is concluded that damage to plastics from this intensity of x-rays is not extreme. It must be noted, however, that actual measured data for solar x-ray intensities apparently do not extend beyond 250 miles, which is the limiting height reached by the WAC Corporal rocket (14) (18). The orbit of the satellite is estimated to have a perigee of 200 miles & a apogee of 800 miles. Apparently no measured x-ray data exist for space between 250 & 800 miles from the earth. For this reason, caution should be exercised in utilizing plastics on the outer surface of the satellite, since they might be damaged by the unknown x-ray intensities between 250 & 800 miles.

6

The aluminum sphere, assuming a thickness of 1/32 inch, is an effective shield against the x-ray intensities listed above. This was determined by calculations utilizing the following equation ($\frac{25}{A}$):

$$I = I_o e^{-Mx}$$

Where I = intensity at depth X; I_o = intensity at surface; $\frac{M}{P}$ = mass absorption coeffient; x = depth of penetration in centimeters; P = density of material. Various values of $\frac{M}{P}$ are given ($\frac{25}{A}$) for aluminum as:

2A	M/P
44.6	850

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2A	M/E
11.88	500
9.87	330
8.32	3700
6.97	2800

Since the aluminum sphere is an effective barrier against the expected x-ray intensities, many plastics may be safely used within the satellite without being subject to damage from irradiation.

For the reason noted above, plastics are not recommended for use on the outer surface of the satellite. If, however, it should be found necessary to use plastic compositions, such as, polyester fiberglas on the outer surface, specifications for such materials should include a requirement that they be stabilized against damage by x-rays.

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TABLE II

STABILITY OF PLASTICS UNDER IRRADIATION

Irradiation to Produce 50% Decrease -(10^9 rads)

MATERIAL	Tensile Strength	Impact Strength	Shear Strength	Elongation
Aniline formaldehyde	10	10	10	10
Araldite, Type B	-	-	>4	-
Asbestos fabric phenolic	>4	>30	>4	>4
Asbestos fiber phenolic	>4	>4	>4	>4
Butacite (polyvinyl butyral)	0.6	-	-	0.6
Casein	0.4	0.1	0.4	0.4
Catalin (phenolic)	1.0	1.0	1.0	2.0
Cellulose Acetate Butyrate	0.07	0.05	0.07	0.05
Cellulose Acetate	0.2	0.07	0.2	0.07
Cellulose Nitrate	0.1	0.1	0.1	0.04
Cellulose Propionate	0.04	0.04	0.04	0.04
CR-39 (polydiglycol carbonate)	3.0	8.0	8.0	7.0
Duralon (furan	>3.0	>3.0	-	>3.0
Ethocel (ethyl cellulose)	0.05	0.03	0.07	0.02
Fluorothene	0.10	0.05	0.2	0.05
Geon 2046 (polyvinyl chloride)	4.0	-	-	0.4
Haveg 41 (phenolic)	>3.0	>3.0	-	>3.0
Karbate (phenolic)	>3.0	>30.0	>3.0	>3.0
Linen Fabric phenolic	0.4	0.02	-	0.02
Lucite (methylmethacrylate)	0.05	0.1	0.1	0.05
Melmac 592 (amino)	-	-	>4	-
Micarta (phenolic laminate)	0.5	0.1	-	0.1
Mylar	2.0	-	-	1.0
Nylon	>10.0	0.05	>10.0	0.1
Paper Base Phenolic	0.2	2.0	2.0	0.2
Plicotuf (Bungs Blend)	>10.0	2.0	>10.0	2.0
Plaskon alkud (polyester)	>4.0	>4.0	>4.0	>4.0

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STABILITY OF PLASTICS UNDER IRRADIATION (Cont'd)

Irradiation to Produce 50% Decrease - (10^9 rads)

MATERIAL	Tensile Strength	Impact Strength	Shear Strength	Elongation
Polyalphamethyl styrene	-	-	0.4	-
Polyethylene	>10.0	0.5	>10.0	0.5
Polystyrenes (Amphenol) Styrons 637,671 & 411 C)	>30.0	>30.0	>30.0	>30.0
Polyvinyl carbazol	>10.0	>10.0	>10.0	>19.0
Polyvinyl formal	10.0	-	-	10.0
Royalite (styrene copolymer)	>10.0	-	>10.0	1.0
Saran B-115 (Polyvinylidine chloride)	1.0	0.5	1.0	0.5
Selectron 5038 (polyester)	>10.0	3.0	>10.0	0.1
Silicone-varnished Glass Cloth	-	-	4.0	-
Styron 475 (polystyrene)	>30.0	0.4	>30.0	0.04
Teflon	0.07	0.1	0.1	0.003
Triallyl Cyanurate	-	-	>10.0	-
Vinyl Chloride Acetate Polymer	5.0	10.0	5.0	5.0

NOTES: S=Small; I=Increases; D=Decreases; Plaskon, Karbate, Haveg 41, Duralon & Melmac 592 contain mineral fillers.

9

(4) Ultraviolet Light

The following data on ultraviolet light intensities were obtained from the Meteorlogical Branch:

Range	Intensity
2000 to 3000 A°	0.8 watt/M.^{2}/100 A°
1150 to 1350 A°	1 to 80 ergs/cm^2/sec.

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These values are quite high when compared to the ultraviolet light intensities in this wavelength range on the ground, which are taken as zero. (8)

Ultraviolet light can cause photodegradation of plastics. (9) (10) (26) (27). Damage from this source may cause undesirable changes in the properties of the plastics. For this reason plastics are not recommended for use on the outer surface of the satellite. If it should be necessary to use plastic compositions, such as polyester-fiberglas on the outer surface, then specifications for such materials should include a requirement that the resins be light stabilized for the wavelengths and intensities listed above.

The aluminum sphere is an effective barrier against the entrance of ultraviolet light into the satellite. No damage is therefore to be expected to materials within the satellite from ultraviolet light, and any compositions may be utilized inside the sphere without danger from possible photodegradation.

(5) Ionosphere (Ionized atmosphere)

The F 2 layer extends from 155 to 250 miles from the earth. No damage to materials is anticipated in F 2 layer because of the very few ionized particles in the high vacuum which is present.

(6) Cosmic Rays

The Meteorological Branch, SCEL has provided the information that the cosmic ray intensity will be in the order of 1/3 particle per square centimeter per second.

The incident radiation is in the form of protons from outer

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space. These protons disintegrate atoms forming protons, neutrons and mesons. The protons and neutrons so formed will disintegrate more atoms. This will occur for about one more step when the total energy is dissipated. This destruction of atoms is not considered to be serious because of the low intensity given above, and the fact that the number of atoms is enormous, 6.06×10^{23} per gram atom. No serious damage is therefore to be expected from cosmic rays.

(7) Meteorites and Cosmic Dust (Micrometeorites)

The data shown in Table III on meteors and meteorites were determined by the Radio Propagation Laboratory at Stanford University

11 (25).

TABLE III

	Mass Gms.	Visual Magnitude	Radius	No. of this Mass swept up by earth each day
Particles Pass thru atmosphere and fall to ground.	10^4	-12.5	8 cm	10
Particles totally disintegrated in upper atmosphere	10^3	-10.0	4 cm	10^2
	10^2	- 7.5	2	10^3
	10	- 5.0	0.8	10^4
	1	- 2.5	0.4	10^5
	10^{-1}	0	0.2	10^6
	10^{-2}	2.5	0.08	10^7
	10^{-3}	5.0	0.04	10^8
	10^{-4}	7.5	0.02	10^9
	10^{-5}	10.0	0.008	10^{10}
Micrometeorites	10^{-6}	12.5	40 Microns	?
	10^{-7}	15.0	20 "	
	10^{-8}	17.5	8 "	
	10^{-9}	20.0	4 "	Total for this group estimated as high as 10^{20}
	10^{-10}	22.5	2 "	
	10^{-11}	25.0	0.8 "	
	10^{-12}	27.5	0.4 "	
Particles removed from the solar system by radiation pressure	10^{-13}	30.0	0.2 "	---

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The probability of puncture of the satellite by meteorites has been variously calculated by the personnel of the Meteorological Branch as from 1 in 200 days to 1 in 10 years.

Ley (21) has made a similar calculation for 1000 square feet area and aluminum 0.04 inch thick. His estimate is that puncture would occur once in two years and four months. Aluminum this thick would be punctured by any particle down to visual magnitude 12. It is to be noted that this is for an area much larger than that of the proposed satellite.

It is therefore concluded that damage to be anticipated from meteorites and cosmic dust is very slight. The sand blasting effect of the cosmic dust would be expected to haze unprotected optical surfaces exposed to the atmosphere.

12 (8) Atmospheric Composition

Above 200 miles the atmosphere consists of nascent oxygen (O) and nitrogen (N) and some biatomic nitrogen (N_2). Because of the high vacuum present it is not anticipated that these substances will cause any appreciable damage. (2) (22) (23)

(9) Stresses Resulting from Mechanical Shock

The most severe mechanical shock conditions resulting in stresses are expected to be of short duration and will have a magnitude of 250 G. Most alloys and metals, if properly designed, will be capable of withstanding the above stresses without failure.

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(10) The materials and recommendations in the synopsis, above, were selected on the basis of the considerations under Text subheadings 1 to 9. Additional information regarding materials, which may be suitable for use on the satellite, may be obtained from the literature by or pertaining to the following: NRL Reports (7) (22); Wentworth Institute, Boston (13); WAC Corporal Missile (14) (18); Project Atlas (16); Moby Dick Transmitter (15); and the Rocket Panel (28).

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b. Allocation of Time and Funds.

Since materials exist which meet the present requirements of the satellite, no allocation of man hours and funds is required at this time for development. Should the requirements be changed at some future date it may be necessary to request an allocation of time and funds to develop new materials.

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PROPOSALS TO SATELLITE PROGRAM

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II. Signal Corps Program Proposals to Phase I

3. Materials, Electronic Parts and Subsystems

b. Absorbing and Emitting Surfaces

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I
3. Materials, Electronic Parts, Subsystems
b. Absorbing and Emitting Surfaces

Prepared by: Components Department

SYNOPSIS: A program is proposed for a theoretical and experimental study of the temperatures which the surface and the interior of a satellite will assume during its flight as a consequence of radiation absorbed and emitted; the investigation will include primarily a study of the absorptivity and emmissivity in the visible and infrared region of a variety of materials with defined surface treatments.

1. Among other problems it is of importance to investigate the temperatures which the surface and the interior of the satellite will assume during its flight as a consequence of the radiation from the sun. It is quite obvious that the temperature range involved will play an important role in the proper performance of the satellite body itself and also will govern, to a certain extent, the performance of the incorporated scientific equipment. From the standpoint of equipment performance it appears desirable to keep temperature fluctuations low. However, the intentional creation of large temperature differences at certain areas of the satellite surface may be very desirable as a means of producing thermoelectric energy. The temperatures of various parts of the satellite will depend greatly on the materials used in the construction and the optical properties of the surfaces. A compilation of pertinent data is not readily available.

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2. It is proposed to undertake an investigation of the temperatures to be expected in the satellite considering a variety of materials, surface treatments and geometrical arrangements. Such a study will include (a) investigation of the absorption and emission characteristics of available materials including coatings with high absorptivity in the visible range and low emissivity in the far infrared region and vice versa (b) experimental determination of the temperatures obtained with selected materials and defined surface preparation whereby the experimental conditions are being chosen as to simulate the conditions on the satellite (c) comparison of the experimental results with theoretical calculations based on data from the literature or values measured during the course of this investigation (d) theoretical and experimental determination of optimal conditions for achieving small temperature differences within the shell and in the interior, finally (e) theoretical and experimental determination of optimal conditions for achieving large temperature differences between selected surface elements of the shell.

3. It is believed that the results of such an investigation could be of great value in the design of the proposed satellite. In addition the results would be of general interest for scientific measurements in high altitudes or in artic areas as well as for the utilization of solar energy.

4. It is estimated that an effort of 500 man days and funds in the amount of $4000 for instrumentation and materials will be required to carry the suggested program to final conclusions.

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PROPOSALS TO SATELLITE PROGRAM

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3. Materials, Electronic Parts and Subsystems

c. Electronic Parts

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE 1
3. Materials, Electronic Parts, Subsystems
c. Electronic Parts

Prepared by: Components Department

SYNOPSIS: Based on the extensive background of the Signal Corps in the research, development, application and standardization of passive electronic parts on overall analysis has been made of the resistive, capacitive and inductive parts concerned with this program. The Signal Corps experience with electronic parts has covered the general environmental conditions anticipated during launching and free flight including extensive high shock and vibration. In addition the Signal Corps is participating extensively in the development and application of passive electronic parts for current guided missiles and high velocity projectiles. In connection with these programs the overall aspects of electronic part reliability have been treated in detail thus providing extensive background to select parts for this project. The passive electronic parts that would be needed for either free running or crystal control CW transmitters or auxiliary electronic instrumentations can be selected from designs now available with characteristics tailored to meet the specific circuit demands. The materials used in these electronic parts would be consistent with the general materials recommendations contained in section 3a on materials and would withstand the anticipated environment.

1. Resistive Components. Based on anticipated environmental conditions such as temperature, pressure, shock, vibration, and radiation, available resistors of the precision wire wound variety, composition, or power wire wound types offer adequate resistance range, power handling capacity and suitable temperature coefficient of resistance to meet circuit needs. Specifically resistive parts have been characterized up to vibrations of 2000 cycles and 10 G's in all possible positions both with and without

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auxiliary mounting supports. This program has yielded reliable techniques for securing parts to withstand anticipated vibration during launching or free flight. Resistive elements have likewise been subjected to high shock testing in projectiles to accelerations above 10,000 G's and techniques are available to mount parts to withstand this environment. Final size will be dependent upon the specific resistance and power required. Units with high orders of reliability can be selected provided the ultimate in miniaturized parts are avoided.

2. Capacitive Components. The wide variety of capacitor designs available indicate that units to meet the overall missile environments can be selected without difficulty. This includes such items as precision fixed glass capacitors for radio frequency applications, fixed ceramic capacitors for low capacity coupling and bypass applications, hermetically sealed organic dialectric capacitors for high capacity bypass applications. As noted in the resistive component discussion above capacitive components have also been subjected to extensive high vibration and high acceleration analysis and techniques have been developed for the reliable use of parts within the above referred environments. Electrolytic units would probably not be needed and could not safely be used. Capacitors have been tested and have withstood the anticipated environmental conditions of temperature, shock, and vibration.

3. Inductive Components. Available inductive component designs should fill general circuit needs with only tailoring of electrical characteristics. Precision fixed inductors are available having temperature coefficient of inductance with 30 parts per million over antici-

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pated temperature range of operation. Other inductive items such as fixed chokes and transformers can be constructed using available design information. Similarly inductive components have been characterized over wide shock and vibration ranges and techniques are available to the Signal Corps to reliably use inductive parts in these environments.

4. General. In addition to above referenced specific electronic parts the Signal Corps has extensive background in electromechanical parts, printed circuit parts and transmission line parts which may be required for the missile instrumentation. All passive electronic parts referenced above have been fully characterized against the complete range of military environments. In addition to vibration and shock these include thermal shock, low temperature exposure, low pressure operation, life testing, humidity testing, etc. The selection of any parts for use in the missile would be based upon this complete background in the research, development and application of parts now possessed by the Signal Corps.

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PROPOSALS TO SATELLITE PROGRAM

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3. Materials, Electronic Parts and Subsystems

d. Power Sources

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II Signal Corps Program Proposals to Phase 1
3. Materials, Electronic Parts, Subsystems
d. Power Sources

Prepared by: Components Department

SYNOPSIS: If electronic equipment is required within the satellite, as an aid to identification and tracking, then a power source will be required. The following energy conversion devices were investigated, as possible power supplies: electrochemical (batteries), photovoltaic cells (solar converter), thermoelectric, nuclear, electromechanical conversion and self contained fuel supply systems.

Power requirements for Phase I may range from 100 MW to 1.0 W at from 15-45 volts for continuous operation from several hours to one year.

To satisfy these requirements the following systems are recommended:

(a) Batteries:

For relatively short life operation the electrochemical system has a distinct advantage. Zinc silver peroxide batteries are capable of providing up to 50 watt hours/lb. Other advantages of this particular battery are: excellent voltage regulation, immediate availability from several manufacturers, satisfactory operation under anticipated environmental conditions, form factor can be readily adapted to conform to satellite requirements, and this system has been proven by actual use in numerous applications.

(b) Solar Converters

For intermittent operation of electronic equipment, that is during the period of time when satellite is in the light region of its orbit 63-68 minutes out of 101 minutes, the solar converter or photovoltaic cells offer a great deal of promise. This system has a conversion efficiency of 8% and is capable of providing, at the present date, 112 watts/sq meter of surface area.

(c) Solar Converters and Ni Cd Batteries

Continuous operation can be achieved by use of a rechargeable storage battery of the Nickel Cadmium system, floating across a charging system consisting of solar

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converters. This system would be so designed that the small storage battery would provide the required power while satellite is in the dark area of its orbit, approximately 33-38 minutes. During the operation in the light region of orbit, approximately 63-68 minutes, solar converters would convert the sun's energy to electrical energy to charge the battery and also to provide the power for the electronic equipment.

The following tables illustrate the characteristics and capabilities of the various systems.

Zinc Silver Oxide

Temperature	80°F	32°F	0°F	-20°F	-40°F
Watt hrs/lb	50	55.5	50.7	21.2	9.2
Watt hrs/in^3	2.4	2.66	2.4	1.29	.575
Max. volt/cell	1.55	1.51	1.45	1.41	1.27

10 volts not 20 volts

Nickel Cadmium

Temperature	60°F - 100°F	32°F
Watt hrs/lb	10	9
Watt hrs/in^3	1.1	1.0
Max. volt/cell	1.25	1.20
Min. No. of cycles	1286	1286

Solar Converters
based on 1400 watts/sq meter of available Solar Energy

Efficiency	Max. Watt per unit of area	Max. load voltage per solar cell	Operating Temperature
8.0%	112 w/sq meter .0112 w/sq centimeter	0.33V	-20°C to +80°C

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1. DISCUSSION

a. Batteries: The maximum watt hrs/lb and per unit of volume obtainable from presently available electrochemical systems is provided by the zinc silver oxide system (AgO-KOH-Zn). This system can provide at low drains up to 50 watt hrs/lb. See chart page 2. At low temperatures the capacity of the battery will fall off and, if possible, some means of insulation will be required. At high temperatures, if no long activated stand periods before use are involved, the capacity will be approximately the same as at normal temperatures. Another advantage of this system is its excellent voltage regulation. Under specific discharge conditions (constant temperature and drain) the voltage regulation over majority of discharge period will be approximately 2-3%.

By using a new construction technique it is believed that a rechargeable sealed battery of this electrochemical system can be made. The number of charging cycles are unknown, but they will in all probability be considerably less than for the Nickel Cadmium System. Because of limited cycle life and the difficulty with gassing during charge the rechargeable zinc silver oxide battery does not offer, at this time, the promise that the sealed Nickel Cadmium Battery has for a rechargeable application. Although this system can provide only about 1/5 the capacity of the zinc silver oxide system for straight battery operation the particular advantages of this system will become apparent when used as a storage battery in conjunction with the solar energy converters. It is in fact the only storage battery which has been proven by test to be capable of being operated on discharge as well as charge in a completely sealed condition. Its characteristics will ideally match those of the

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solar converters which are discussed later in this report.

For illustrative purposes the requirement of 20 volts at 400 mw will be discussed. To satisfy this requirement a battery consisting of 16 cells with a capacity of 0.15 to 0.3 AH and weighing .3 to .6 lbs would appear adequate. This battery, about one-half charged, will be connected in parallel with the solar converter and the load. During the 33-38 minute period of darkness about 16 to 8% of the energy available in the battery will be discharged into the load. During the 63-68 minute period of sunlight, the solar converter will feed into the load and at
4 the same time recharge the storage battery.

During the period of darkness the terminal voltage of the battery will gradually decrease by approximately 6.5% to 4.0%, depending on the size of the battery. During the sunlight period the average voltage will be higher, with instantaneous values increasing by 6.5% to 4.0%. The result is a total voltage variation of approximately ±5 to ±3% over the complete cycle. If required, the voltage across the load could be held slightly more constant, namely, within ±3.5 to ±2% by using a series resistance during the battery charging periods, which can be shorted out automatically during discharge. Voltage regulation could also be improved by using cells with higher capacities. Tests showed that a voltage regulation of ±0.5% for a ten minute period during charge and ±0.7% for a 10 minute period during discharge was possible for a 0.19 AH battery. Further tests are being conducted to determine how large a battery is required to obtain a voltage regulation of ±0.2% for a ten minute period, which is required for the stable frequency oscillator during Doppler measurements.

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The cycle of discharging and charging the storage battery will continue periodically and when working at the proper point of the solar converter, I-V characteristic, the storage battery will operate continuously at the same flat portion of its charge and discharge characteristic thus assuring very good voltage regulation.

Though no final data concerning the cycle life of sealed nickel cadmium cells are available, it is conservatively estimated that a minimum of 90 days operation will be obtained. This is equivalent to approximately 1286 complete revolutions of the satellite.

Batteries of the above electrochemical systems can easily withstand shocks and accelerations of 250 g's and spins of 3600 RPM.

5 b. Energy conversions systems based on use of self-contained fuel supply other than nuclear type:

Energy conversion systems employing engines and generators, which would have a poor watt per pound ratio and low overall efficiency for the small amount of power required for the satellite are not considered practical for this application.

c. Photovoltaic Effect (Solar Converters):

The system that appears most promising at this time is the use of silicon photovoltaic cells, recently developed by Bell Telephone Laboratories, as solar energy converters. This system will provide power continuously during daylight operation and is not limited to a definite useful life such as batteries. If used in conjunction with a hermetically sealed, rechargeable nickel cadmium battery, dark side operation of the electronic equipment will be possible, thus providing continuous operation.

The data and conclusions on Photovoltaic Effect that follow are the results of joint efforts of the Electron Devices Division and the

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Components Department. The Electron Devices Division is currently engaged in a research and development program for the investigation of different materials for solar converter cells and is also conducting experimental tests on various silicon solar cells. Close contact is being maintained with commercial laboratories engaged in solar converter research and production.

With 1400 watts per square meter of solar energy available in outer space, silicon solar converters having a conversion efficiency of 8% would provide 112 watts/sq meter or 11.2 milliwatts/sq centimeter. Maximum efficiency is obtained when the load voltage is approximately 2/3 of the open circuit voltage or .33 volts per cell. Other materials for solar converters such as Gallium Arsenide, Iridium Phosphide and Cadmium Telluride are currently being investigated at R.C.A. Laboratories, Princeton, New Jersey under a Signal Corps contract. Initial results indicate an efficiency of 5% can be realized and improvements may be forthcoming as this work continues.

Assuming a satellite with a 20 inch diameter and four (4) groups of solar converter cells arranged on its surface in such a manner that regardless of spin or tumbling some clusters are always illuminated, the following approximate figures are given to show the number of cells, areas and weights involved for different power levels.

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Table I

Power Output			Cells		Total Weight in Grams Includes Housing		
Volts	Current MA	Power MW	Number of Cells per Cluster	Area per Cell sqcm	Total Number of Cells	Total Area sqcm	Total Weight In Grams Including Housing
45	2	90	150	.059	600	35.4	52.4
* 1.2	125	150	4	3.68	16	59	87.6
**20	20	400	60	.6	240	144	213.4
***20	7.5	150	60	.223	240	53.5	80

Note: * These values of voltage and power are for equipment using a sub-miniature vacuum tube as employed in the Navy Minitrack System.

** Power requirements for stable frequency oscillator for Doppler measurements and tracking.

*** Power requirements for oscillator for tracking only.

The weight figures are based on a weight of 2.33 grams per cc of silicon and a cell thickness of 1/16 inch. The weight of the housing is estimated to be three times the weight of the silicon. All values given are for operation in the light region and do not include power for charging a battery. If dark side operation is required, the values of power cell area and weight will be approximately doubled.

Connecting the four groups of cells in parallel to the load may introduce losses due to the active group discharging into the inactive or partially inactive groups of cells if the dark impedances of the inactive groups are not high enough. This can be overcome by placing a semi-conductor diode in the output lead of each group of cells to prevent this type

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of discharge. Some additional cells in each group may be required to compensate for the forward voltage drop of the diode.

The area per cell shown in Table I are based on power requirements and in some cases the small areas required may not be feasible due to limitations in construction and cutting techniques. This may result in excess power for certain designs. The latest information available from Bell Telephone Laboratories indicates that it may be possible to fabricate cells with a minimum cross sectional area of 0.25 sq. cm. A cell of smaller area has been fabricated experimentally, but it is thought that for this application and possible large quantities of cells that may be required a more
8 practical minimum cell area would be 0.25 sq. cm.

Another possible method of multicell fabrication would be to cut a single silicon wafer of 1.25" diameter into 16 pie shaped segments or cells of 0.5 sq. cm. This would result in utilization of practically all the wafer area for cells and lends itself to a compact arrangement for electrical connections and housing. A number of these wafers would be arranged in clusters and this number depends on the power requirements of the electronic equipment.

Factors which must be considered in the design of this type of conversion system and which will require further investigation are: (1) Type of plastic required for housing of silicon cells. The plastic material must, in addition to being extremely light, have properties that will permit maximum transparency of solar energy at any angle of incidence, and should not deteriorate with age, X-ray, or Gamma radiation, and must withstand shocks of 250 g's and centrifugal forces of 900 g's. (2) Effect of cosmic dust on the overall performance of the solar converter. Particles

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of this dust may hit the cells with sufficient force to cause pitting of the plastic and cells, reducing the effective area and thus the amount of power than can be converted. If, due to the static charge on the sphere, the dust particles adhere and form a dust film, this will also reduce the amount of solar energy available for conversion. (3) Effect of X-ray and Gamma radiation on silicon junctions.

d. Thermoelectric Effect (thermocouples).

The use of thermocouples for the conversion of solar energy to electrical energy has some possibilities and will depend chiefly on the temperature gradient that can be established between the hot and cold junctions. The most efficient thermocouple materials investigated to date are the alloys Zn Sb + (Sn, Ag Bi) vs 91 Bi + 9 Sb (melts at 260°C) or vs Constantan. The approximate efficiencies are determined chiefly by the temperature gradient and are listed below:

Temperature Gradient	Thermocouple	Efficiency
100°C	Zn Sb(Sn Ag Bi) vs 91 Bi + 9 Sb	1.5
200°C	" "	3.5
300°C	" vs Constantan	4.0
400°C	" "	5.0
400°C	Chromel P vs Constantan	0.5

For a temperature gradient of 400°C, the alloys Zn Sb (Sn, Ag, Bi) vs Constantan are ten (10) times more efficient than Chromel P vs Constantan (the best commercially available material). The alloys Zn Sb (Sn, Ag, Bi) and 91 Bi + 9 Sb have, however, the inherent disadvantage of being brittle and cannot be drawn into thin wires as the less efficient, more ductile alloys, thus necessitating their being cast into rather bulky forms in

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order to obtain reasonable mechanical strength. The 91 Bi + 9 Sb alloy melts at 260°C and limits its use in normal applications. If it is possible to maintain the hot junction at 200°C and reduce the cold junction below the usual ambient temperature encountered in normal generator design, higher efficiencies with this alloy may be possible. It should be remembered, however, that only about 5% of the heat at the hot junction will be converted to electrical energy while the balance will be transmitted by conduction to the emitter. Greater emitter area will therefore be required to maintain a suitable temperature gradient between junctions. One proposed method to obtain suitable temperature gradients in the sphere is to treat the various surface segments to make some absorbent to infrared while others reflect infrared. These surfaces will be insulated from one another; some will act as receivers (dark absorbing surface) and emitters (light reflecting surfaces). The size and number of these areas depends on the required power input to the thermocouples, and on the amount of heat to be dissipated as losses at the cold junction. In general this area will have to be vastly greater than the hot junction area. The following calculations are based on the assumption that a voltage of 20V at 0.150W (7.5 ma) or 20V at 0.4W (20 ma) are required and a temperature gradient of 300°C is feasible. The maximum open circuit voltage obtainable with one of the most efficient couples at this temperature gradient will be approximately 0.080 volts/couple. Maximum power is obtainable when R internal = R external. Therefore, 1/2 of the open circuit voltage will be across the load, thus, 21V/.040V = 525 couples. One (1) additional volt may be required to compensate for the ohmic drop in the series connection. The design of the couples required to produce the desired voltage and power

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creates problems, due to the fact that there will be a minimum cross section that can be cast with these brittle, high efficiency alloys for mechanical reasons and this cross section will be greater than that required for the power. The weight of the 525 couples with necessary mounting details will be prohibitive for only 0.15 or 0.4 watt output. Considering the fact that these thermocouples are essentially a low voltage, heavy current device, it may be more practical to produce a lower voltage by thermocouples and use a transistorized power converter to obtain the desired high voltages. This may reduce the overall reliability of the system because of additional components, but it would most certainly reduce the weight. It is estimated that with the differential temperature of 300°C, 75 thermocouples will deliver approximately 3 volts at .266 amperes or .8 watts. With only a 50% efficient transistor converter, 0.4 watts at 20 volts can be obtained. Assuming an efficiency of 5% for the thermocouples, an input of 16 watts is required. Because of the losses in the transfer of heat to the hot junctions approximately 32 watts of incident solar energy are required, therefore, $\frac{32W}{1400 \text{ W/sq meter}} \approx .023$ sq meters of receiver are required. It is estimated that the total weight of the thermocouples would be approximately 1000 gr. Thermocouples may be used in conjunction with batteries the same as solar converters. In this event a correspondingly greater load will be placed on the thermocouples to charge the battery during light area operation and a diode must be used to prevent discharge of the battery into the couples when in the dark side, thus necessitating a greater surface area and weight than calculated above.

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e. Thermoelectric Effect and Nuclear Heat:

An energy conversion system employing radioactive Polonium 210 as a source of heat and thermocouples has been investigated at the Mound Laboratory of Monsanto Chemical Company. According to experimental data given in a report, a watt hrs. per pound ratio of 500 is possible using this system. This figure indicates that it is 10 times greater than the best electrochemical system. An experimental unit using 146 curies of Po210 and 40 thermocouples of Chromel P vs. Constantan weighed 31 grams and delivered 25 ma at 37.6 millivolts (9.4 milliwatts) at maximum power for 138 days which is the half life of Po210. This system offers the advantage of being rather independent of external temperatures and is an entirely self-contained source of power. One disadvantage is the low efficiency, 0.2% of the present experimental unit. This means that consideration will have to be given as to how the waste heat liberated from the nuclear source will effect other equipment in the sphere. The use of more efficient thermocouples should improve the watt hrs. per pound ratio while reducing the quantity of waste heat.

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f. Nuclear Heat for Thermionic Emission:

The conversion of heat to electrical energy by thermionic emission has been accomplished in the past, but very little experimental data is available for use in comparing this system with others. Experiments are presently being conducted to procure data for use in arriving at comparative figures for this system.

g. Electromechanical System Operating on a Temperature Differential:

The Longine Wittnauer Watch Company has developed a system used to operate their "Atmos" motor clock. This system consists of a thin

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diaphragmed bellows filled with a chemical substance that will exert pressure at elevated temperatures. This pressure changes with the ambient temperature operating the bellows which in turn winds a clock spring. The manufacturers claim that 1°C temperature change results in a potential of 0.713 foot lbs. On the basis of published spring data, it would require 885 cu. in. of steel spring to store 1 watt hour or 0.35 watt hours/lb. Liquid springs in form of hydraulic systems are more efficient, here 0.88 watt hrs/lb can be realized. To utilize this system it would require establishing a suitable temperature gradient, a means of converting the differential temperature to potential energy in the spring, and the use of a system of gears to drive a small generator capable of providing the required output power. The overall efficiency of this system would be very low, involve the use of many moving parts, and would result in questionable reliability under the conditions required for the satellite.

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h. Availability

Batteries:

The zinc silver oxide batteries are immediately available from several sources. It is anticipated that only a slight amount of development work at a nominal cost will be required to modify the form factor of these batteries for use in subject application.

The sealed Nickel Cadmium batteries discussed in this report are currently manufactured in Europe. There are, however, several American companies that are producing sealed Nickel Cadmium cells, and it is felt that they are quite capable of providing the required batteries. No further development work on this system will be required. Therefore, it is anticipated that several hundred units could be procured for research and

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development purposes at a cost of several thousand dollars.

Solar Converters:

There seems to be a fair degree of assurance that a solar power pack could be made available with a one year development program. The Signal Corps Engineering Laboratories are currently engaged in contractual research effort on photovoltaic solar energy converters and silicon junction devices with Bell Telephone Laboratories and R.C.A. Laboratories. New ideas and techniques which will result from this program could be applied to the subject project. In view of the importance and promise that Solar converters have, it is recommended that the existing contracts be modified and extended to call for the development of a satisfactory Solar Energy Power Supply.

14

Thermoelectric Devices:

If these devices are to be used in the satellite program some research and development work will be required to investigate techniques for casting sufficient quantities of the more efficient but brittle thermocouple alloys and to establish methods of mounting these couples in the satellite to protect them from the adverse effect of spin and shock. The development of transistorized power supplies for converting low potential high current to high voltage low current and of coating materials that will be absorbent or reflective to infrared in order to help establish a satisfactory temperature gradient for the thermocouples will also be required. It is estimated that this work can be accomplished at a cost of one year in time and $35,000.00.

Nuclear Heat Sources:

These types of heat sources offer a great deal of promise for use with thermocouples or thermionic emission converters. Before they can

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be used, however, research and development should be conducted toward improving the overall efficiency of these systems and reduce the cost. It is anticipated that within one (1) year at a cost of $40,000.00 Nuclear Heat Source power packs could be made available.

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PROPOSALS TO SATELLITE PROGRAM

Volume III

II. Signal Corps Program Proposals to Phase I

3. Materials, Electronic Parts and Subsystems

e. Frequency Control Devices

[redacted]

II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE 1
3. Materials, Electronic Parts and Subsystems
e. Frequency Control Devices

Prepared by: Components Department

SYNOPSIS: If it should be decided to equip the satellite with a transmitter, it will be necessary to control its frequency within certain limits in order to be able to reduce the required bandwidth of the receiving systems. For most of these systems a moderate stability of the signal frequency, say in the order of 50 parts per million will probably be sufficient to serve this purpose. If, however, it should be decided to utilize the frequency shift due to the Doppler effect for the determination of the velocity and the trajectory of the object a stability of better than one part in 10^8 would be required. In the following section two possible transmitter designs are discussed as examples in order to illustrate the feasibility of frequency control for the present application.

1. The Signal Corps Engineering Laboratories have been actively engaged in the field of frequency control for the past twenty years and have accumulated considerable experience in the design of oscillators, the frequency of which is controlled by means of electro-mechanical resonators. The state of the art at the present time is such that practically any frequency in the range from audio frequencies on up to the very high frequencies can be controlled within very close tolerances by a suitable chosen electro-mechanical resonator and, in recent years, appreciable progress has even been made toward precision frequency control in the ultra high frequency range by using atomic or molecular resonances.

The choice of a particular frequency for a transmitter to be located in the satellite depends upon a number of facts. The propagation characteristic of the ionosphere sets the lower limit around 50 mc.

The efficiency of the transmitting antenna and the use of the receiving antennas make it desirable to go as high in frequency as possible, and a compromise has to be made between these requirements and the requirements of weight, power consumption and reliability of the transmitter. If the frequency of the transmitter should be controlled by a quarts crystal resonator, and no frequency multiplication be employed, the upper limit at the present time for efficient operation of an overtone crystal unit would be around 135 mc. It has been indicated by FCC representatives that there are some open bands at 132 mc which could be made available if need be.

Antenna experts have indicated that at 132 mc an efficiency of at least 10% could be expected for the transmitting antenna in the satellite and a collecting area of 30 to 50 square feet for the receiving antenna would be sufficient to assure adequate reception of the signal up to a distance of 2000 mi. if around 5 mw of r-f power is emitted from the satellite antenna. Based upon this information, an operating frequency in the range of 132 mc has been assumed for the following examples, which are representative for the capabilities of crystal controlled transmitters, suitable for use in the artificial satellite program.

2. If only a moderate stability of the transmitter frequency is required a single stage crystal controlled oscillator, capable of delivering 25 to 50 mw of r-f power into an optimum load would be required in order to obtain 2.5 to 5 mw of radiated power out of an antenna with an efficiency of around 10%. For several reasons, mainly because of the lower power consumption and the longer useful life, it would be desirable to use a transistor in preference to a vacuum tube for the oscillator.

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The most suitable transistor design for this application would be the diffused base PNP transistor, having an α-cutoff of 400 mc. Transistors of this type have been developed at Bell Telephone Laboratories and can be made available in the near future. The crystal unit would be an AT-cut quartz crystal with a fundamental frequency of 26.4 mc, operating at the fifth harmonic at 132 mc.

The frequency of this single stage transmitter would be stable to within 50 parts per million as long as the ambient temperature is within the range of -20°C to +40°C. At lower temperatures the operation of the batteries, which may be used as a power supply, and at higher temperatures the operation of the transistor become questionable. The DC power consumption of the device would be in the order of 150 to 200 mw, depending upon the required r-f output, the operating voltage between 10 and 30 volts.

The total weight of the single stage crystal controlled oscillator would be in the order of 4-6 ounces, depending upon the amount of radiation shielding required to maintain the temperature within the specified limits.

3. (1) If, however, a power consumption in the order of 400 mw and an approximate weight of 8 ounces can be tolerated a signal source stable to better than one part in 10^8 could be installed in the satellite. This would extend the possibility to utilize the frequency shift due to the Doppler effect for the determination of the velocity and the trajectory of the satellite, by a method which promises to be exceptionally accurate, since only frequency measurements are required.

To assure good frequency stability a multiple stage, crystal controlled transmitter employing transistors would be used. It would

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consist of a crystal oscillator, a driver stage and a power amplifier. The crystal oscillator stage would be operating at a relatively low level and capable of providing two to five milliwatts of power to the driver stage, the driver and power amplifier then have to provide the additional power gain. The most suitable transistor design would again be the diffused base PNP transistors. As mentioned before, the total DC power consumption of this device would be in the order of 400 mw, the operating voltage between 10 and 30 volts.

(2) The crystal unit to be employed in the oscillator stage would be a carefully selected CR-54 type unit with a fundamental frequency of 26.4 mc, operating at the fifth harmonic at 132 mc. Tests have shown that units of this type are capable of withstanding considerably higher accelerations than the ones to be encountered during the launching of the satellite, and that only a very slight shift of the operating frequency is caused by the shock. Since no frequency measurements of the emitted signal are anticipated during the launching, only the stability of the transmitter frequency after the launching is important for the present application. Tests have shown that the frequency of a conventional test oscillator is stable to within 5 parts in 10^9 for a period of half an hour if controlled by a crystal unit of the type intended to be used, temperature controlled in a water bath at room temperature. The crystal units for this test have been picked at random from the stock and have undergone severe shock tests prior to the experiment described.

(3) From this test and from previous experience it can be expected that the stability of the multiple stage transistorized transmitter

will definitely be better than one part in 10^8 for any ten minute period, the time required for the measurements at one particular observation site during any one passage of the satellite, provided, however, that the temperature of the crystal unit is constant to within 0.05°C during this period.

(4) If the temperature of the crystal unit should be kept constant without the use of a power consuming thermostatic device, it is desirable to make the temperature cycling of the satellite shell as small as possible so that only a moderate amount of thermal insulation will be required. This can be accomplished by making the energy absorbed in the visible range small in comparison with the absorbed thermal energy, in other words, by making the ratio A_V/A_I as small as possible. For example, if $A_I \approx 5\ A_V$, the maximum temperature change of the hull between day and night would be in the order of 35°C. The average temperature around -50°C.

A considerably higher average temperature is required, however, for efficient operation of conventional type storage batteries, which might be used for the power supply, perhaps in connection with solar batteries. It would therefore be necessary to enclose the batteries together with the transmitter in a common housing, and to utilize the 400 milliwatt, dissipated by the transmitter, to raise the temperature inside the housing to some convenient value.

For this a radiation shield consisting of 5 layers of aluminum sandwiched between layers of coarse fiberglass fabric would, for instance, be sufficient to establish a temperature of 300°K within the shielded area and to maintain this temperature to better than 0.05° for any period of 10 minutes, if the figures of the previously cited example are being used. This of course applies only for c-w operation of the transmitter

and only as long as the supply voltage is constant to within 0.2%.

4. The electrical connections from the transmitter to the antenna and from the storage battery to the solar batteries could be made by thin metal foils, incorporated in the plastic supporting structure, to keep the associated thermal conduction to a minimum.

5. The total weight of the transmitter, the housing and the radiation shield would be in the order of 8 ounces, not counting the weight of the power supply.

6. (1) The design and development of the crystal unit, the transistorized crystal controlled transmitter and the radiation shield could be made within the Components Department and would require approximately three man years.

(2) Funds required for the purchase of material and test equipment would probably not exceed $30,000.00.

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PROPOSALS TO SATELLITE PROGRAM

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II. Signal Corps Program Proposals to Phase I

3. Materials, Electronic Parts and Subsystems

f. Electron Devices

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I
3. Materials, Electronic Parts, Subsystems
f. Electron Devices

Prepared by: Electron Devices Division

SYNOPSIS: The status and availability of transistors and subminiature tubes has been examined for application in transmitters to be carried by the satellite. Technical details concerning their operation and recommendations formulated to make them available in time for the first flight. The status of various power tubes for proposed ground tracking radars is examined and those tubes which would be available within a year are indicated.

1. INTRODUCTION:

The Electron Devices Division has advised the systems Divisions of SCEL on the availability of tubes, transistors, and solar power sources required in their proposed tracking equipment. Since the choice of tracking methods must depend to a great extent on the availability of electron devices, the ensuing paragraphs will attempt to indicate the present status of these devices, the possibilities of achieving a satisfactory reliable device within the prescribed time and the program required to insure such a device.

The conclusions arrived at have been the result of consultations among cognizant personnel within the Electron Devices Division as well as commercial laboratories. The types of electron devices discussed are transistors, subminiature tubes, solar cell power sources, and power transmitting tubes. In the cases of all but the solar cell, the comments are directed toward satisfying specific system proposals proposed by other divisions of SCEL.

2. DISCUSSION:

a. Transistors. It is self evident that transistors would be the

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ideal device for the low power transmitter in the satellite. For one of the proposals a frequency in the 100-135 mc frequency range was chosen for the crystal controlled transmitter. Further, the Radar Division system requires a 35 m.w. transistor output into a matched load and the Frequency Control Branch requirements are for 50 m.w. With this information in mind, in addition to the environmental problems, the following facts and conclusions can be stated.

(1) No transistors exist at this time which would meet the environmental and electrical characteristics required.

(2) Transistors have been fabricated in the Bell Telephone Laboratories under SCEL contract which will operate at 200 mc with a few mw of power output. Further fabrication of such transistors will be done in the development laboratory and includes adapting new research techniques as well as the difficult packaging problem. Suitable 25-35 m.w. transistors, 50-100 in number, should be available within a year. Further, a one year's development program directed towards satisfying the 50 m.w. requirement should prove adequate.

(3) The Electron Devices Division has obtained verbal assurance of a qualified contractor's interest in performing the necessary development. Such a development would be entirely apart from the present SCEL contract in view of the important timing factor and the necessity of a well directed effort in order to insure success.

(4) The transistor will be of the 2039 diffused base PNP type with an $\alpha \cong .95$, $f_\alpha \cong 5 \times 10^8$, $C \cong 20$ uuf, $r_b \cong 150\,\Omega$ and R series $\cong 150\,\Omega$. The unit will operate at 40-50 volts and an efficiency of 25% was considered a minimum by BTL. The frequency stability required

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by the Frequency Control Branch and Radar Division of 50 parts in 10^6 and 1 part in 10^5 respectively is expected to be governed by the temperature control required by the crystal rather than the transistor. This assumption is predicated on the fact that the operating frequency is approximately one-fifth of the high frequency cutoff ($f\alpha = 5x10^8$). The validity of this assumption however should be checked experimentally as soon as possible in view of the limited knowledge available about the behavior of transistors at these frequencies. Frequency Control Branch is also considering a system which would require a stability of 1 part in 10^8. The frequency-temperature stability characteristics of the transistor should be examined experimentally before any commitment is made to such a system.

Two precautions should be emphasized concerning the maximum operating temperature and the need for a heat sink to conduct away the heat generated by the transistor. First, the efficiency of the transistor will drop off rapidly above 65°C and operation above this temperature is not recommended. Second, the transistor shell must be attached to an adequate heat sink to conduct the heat away.

b. Subminiature Tubes. There are no performance data available on subminiature tubes suitable for this application operating in the 100-135 mc range. The guided missile types all have heavy filaments and require large filament powers eliminating them for the satellite application. In order to determine the potentialities of some of the more promising subminiature types a number of them were evaluated for performance characteristics in a 125 mc oscillator. Measurements were made of power output, plate current and grid current at various values of grid resis-

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tance and plate voltage. Plate circuit efficiencies were computed for each set of conditions. Vibration and acceleration tests were performed to determine any possible deficiencies of this type of tube.

The results of the testing indicate the suitability of only two types from the group tested; i.e. tube type 5971 and tube type 5677. A third type 6611 gave mixed results in the three tubes tested and is therefore difficult to evaluate. The characteristics of the 5791 are as follows:

Plate supply	70 volts
Filament current	80 ma.
Filament voltage	1.3-1.2 volts
I_b	1.3 ma.
R_g	16,000 Ω
Power output	47 mw.
Plate circuit efficiency	52%
Overall (plate and filament) efficiency	24.6%

The oscillator characteristics for the 5677 are as follows:

Plate supply	80 volts
Filament current	60 ma.
Filament voltage	1.3-1.2 volts
I_b	1.9 ma.
R_g	20,000 Ω
Power output	43 m.w.
Plate circuit efficiency	28%
Overall (plate and filament) efficiency	19%

The 6611 is a recently developed pentode RF amplifier which was used in

this connection as a triode connected oscillator. The first tube gave excellent results under the following conditions:

E_{bb}	70 volts
R_g	40,000 Ω
I_b	1.3 ma.
I_g	150 microamps
Power output	33 m.w.
Plate efficiency	36.3%
Overall efficiency (Plate and filament)	28.4%

Of the two additional 6611 tubes examined one had runaway characteristics and the other much poorer efficiency. It would seem that with some additional development work on the 6611 tube it might be possible to obtain a subminiature tube which could compete with the transistors efficiency. There is some uncertainty as to whether this could be accomplished in one year.

The life of the 5677 and the 5971 is estimated at 500 hours under the operating conditions cited.

The preliminary vibration tests indicate resonance of the filament at frequencies above 1000 cycles. Acceleration tests indicate that this type of tube will withstand 1000 g.

c. Power Tubes. The Radar Division is considering a number of ground tracking systems operating in various parts of the 100-6000 mc frequency region. The tubes discussed below include those which have been considered by the Radar Division as well as some additional tubes which may perform better or be more available.

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In the 100-500 mc frequency range, two Eimac tubes, 4 w 20,000 A and 3 w 10,000 A, should prove satisfactory with the first requiring approximately 90 day delivery and the second a 6 month delivery. The 4 w 20,000 A is a tetrode rated for 1.5 megawatts output at a 0.007 duty cycle and capable of long pulse usage. A number of these tubes can be operated together to give power levels of 12 megawatts. The 3 w 10,000 A is a triode with a power output of 1 megawatt at a 0.001 duty cycle and also capable of being stacked to achieve power levels of 10 megawatts. An RCA tube, A 2342, operating up to 200 mc is rated for 1.7 megawatt output at a 0.09 duty cycle when plate pulsed and 1 megawatt at 0.06 duty cycle when grid pulsed. These latter tubes are made in development tube shop and procurement would take a year.

Other tubes in this frequency range include the 6448 RCA beam power tube which will operate up to 900 mc and is comparable to the 4 w 20,000 A although it is rated more conservatively than the Eimac tube.
6 In addition RCA could probably assemble A 2346 tubes within the year. This tube is capable of 10 megawatts output at 0.01 duty cycle when plate pulsed. The tube has never been fabricated before but on the basis of our knowledge of the problem it should come very close to making the 1 year deadline.

The Radar Division has indicated an interest in the 3000 mc, 1200 mc and 5,000 mc frequency bands where power outputs of 10, 2 and 3 megawatts respectively would be required. The experimental QK 170 magnetron operating in the 3000 mc region with a power output of 10 megawatts would not be available in one year. The availability of this tube is complicated by the long pulse duration required in order to keep

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the receiver bandwidth to within 100 Kc. The QK 170 never achieved 10 megawatts at 10 microsecond pulse width. At 1200 mc the QK 264 magnetron should be satisfactory for 2 megawatts output and has been produced in preproduction lots. However, this tube has never operated at more than 5 microseconds pulse lengths and should therefore not be considered if 10 microsecond pulses are required. Samples of a high power klystron developed for the Air Force at General Electric operating in this frequency band have been delivered but not enough information is known about this tube to pass judgement on its availability within the year. The SAC 42 Sperry klystron with an output of 3 megawatts would appear to meet Radar Division requirements for a 5000 mc system. Tubes are available and should meet a 10 microsecond pulse duration requirement.

7 d. Solar Power Source. The Electron Devices Division is engaged in a research and development program to exploit large area junctions for photovoltaic conversion of solar energy. This work is done in conjunction with Components Division, Power Sources Branch and a coordinated proposal for the satellite application is presented under III d. It appears that suitable solar cells may be available for the first launching.

3. CONCLUSIONS:

As a result of the above the following conclusions and recommendations are submitted. Transistors satisfying the 35 m.w. output should be available in time for the first satellite. The 50 m.w. transistor must be considered a little less probable although still fairly certain. In order to improve the probability it is recommended that as soon as assignment of responsibility is made, a 1 year contract be

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placed for these transistors.

In the event that it is decided to utilize a transmitter in the satellite, it would be desirable to do additional development work on the 6611 type subminiature. This can not be counted on, however, with high probability for the first flight. The other subminiature tubes appear suitable for this application but their overall efficiency is below that anticipated from the transistor and they are limited in life.

The general area covered by power tubes does not require any conclusions or recommendations beyond what appears in the text.

The Signal Corps Engineering Laboratories is currently engaged in contractual research effort on photovoltaic solar energy converters (solar cells). New ideas and techniques which will be discovered on this program could be applied to the subject project resulting in the most up to date system available for satellite application.

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PROPOSALS TO SATELLITE PROGRAM

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II. Signal Corps Program Proposals to Phase I

4. Utilization of Satellites of Phase I for Signal Corps R&D Purposes.

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II. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE I
4. Utilization of Satellite for Signal Corps R&D Purposes

Prepared by: Physical Sciences Division

SYNOPSIS: The Signal Corps Engineering Laboratories propose to instrument one or more research satellites of the 30-40 pound class as part of their R&D programs in communications and meteorology. Experiments in earth cloud-cover, ionospheric absorption, solar ultraviolet, and x-rays are planned. The satellite program and the SCEL program in upper atmosphere rocket research are visualized as being complementary.

Fundamental to the Signal Corps use of the satellite as a research tool is the idea that the primary research use of the object should be to carry out those experiments unique to a satellite. Experiments that are equally well or better done by means of rocket techniques should receive a lower priority in the satellite program.

The primary fields of activity for the SCEL research use of the satellite are believed to be:

a. Solar activity

1 b. Ionosphere and electromagnetic propagation

c. Meteorology

Within these fields lie many research problems and activities that would make large contributions to the Signal Corps missions in communications and meteorology.

To avoid confusion, the following definitions are made:

a. A satellite without instrumentation is defined as a satellite without means of telemetering information to a ground station. It is consistent with one proposal for an inert IGY-satellite or a satellite with a simple beacon (transmitter) with no telemetering. The total

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4. Utilization of Satellite for Signal Corps R&D Purposes (Cont'd)

weight at altitude is expected to be 10 lbs.

b. A satellite with limited instrumentation is defined as a satellite containing a basic transmitter and telemetering system capable of sending information from a limited number of end-instruments to the ground. It is consistent with the one proposal for an IGY-satellite in which the total weight at altitude is expected to be 30 to 40 pounds.

c. The long-range program is defined as that research program resulting in a satellite in which almost unlimited instrumentation can be placed and from which highly complex measurements will be possible. Control of aspect and recovery of instrumentation should be anticipated.

2 1. Use of Satellite without Instrumentation

Data Obtainable:

a. Drag data and atmospheric densities

b. Earth-figure and other geodesic parameters

By providing the optical and electronic tracking data necessary for the operational use of the satellite, the SCEL will be able to make drag, and thus air-density, measurements. The quality of these results will depend on the accuracies of the tracking systems. However, the accuracies estimated as possible with the planned tracking systems will yield data of limited value, although of supplementary interest. Only average values will be obtained because of the need for large number of orbits; therefore, the shape of the object is not too important.

Earth-figure and other geodesic qualities also may be derived from the tracking data. The usefulness of these data will depend on the

4. Utilization of Satellite for Signal Corps R&D Purposes (Cont'd)

accuracy of the tracking and other data. Detailed and somewhat extensive study will be required to determine the extent to which significant data can be obtained.

2. Use of Satellite with Limited Instrumentation

Data Obtainable:

a. Solar Activity

(1) Ultraviolet radiation

(2) Soft and hard x-rays

(3) Correlation with other solar and geophysical phenomena

(4) Magnetic field

(5) Airglow and auroral radiations

(6) Cosmic rays

3 b. Ionosphere and Propagation of Electromagnetic Waves

(1) Ionospheric densities

(2) Ionospheric absorption

(3) Correlation with other solar and geophysical phenomena

(4) Solar activity

c. Meteorology

(1) Distribution and movement of clouds

(2) Solar activity

d. Temperature and other properties of the satellite that reflect its environment.

Each of these parameters may be measured with a minimum of instrumentation, not requiring recovery, and being capable of adaptation to a

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4. Utilization of Satellite for Signal Corps R&D Purposes (Cont'd)

relatively simple telemetering system.

The basic instrumentation required would be a transmitter and telemetering system that would be mutually compatible with the electronic tracking system(s) used.

Since the data obtainable in these three fields of activity are specifically relevant to the Signal Corps Engineering Laboratories missions in meteorology and communications, as demonstrated by the fact that the Laboratories already have active external and internal research groups and projects in these fields, particularly in its rocket research of the upper atmosphere and ionospheric research groups, the Signal Corps Engineering Laboratories specially propose to instrument one or more satellites as shown in Table 1.

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4. Utilization of Satellite for Signal Corps R&D Purposes (Cont'd)

TABLE 1

Signal Corps Engineering Laboratories Instrumentation of a Research Satellite

Parameter	Instrument	Weight	Life
Cloud Distribution	Infra-red detectors plus filters	6 to 8 lbs.	2 weeks
Ultraviolet radiation	Photon counters plus filters	2 lbs.	3 to 4 weeks
Soft x-rays (100-400A)	Photon counters plus absorbers	2 lbs.	3 to 4 weeks
Ionospheric Absorption	Transmitter in satellite sweeping the 15-50 mc band	5 lbs.	1 week
Temperatures on satellite	Bead thermistors	1/4 lb.	1 to 2 months
Transmitter-telemetering	Subminiatures and batteries; crystal-controlled	4 to 8 lbs.	1 to 2 months
Keying Beacon	Subminiature	1 to 2 lbs.	1 to 2 months
Structure		15 lbs.	
	Total Weight:	35 to 43 lbs.	

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PROPOSALS TO SATELLITE PROGRAM

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III. Signal Corps Program Proposals to Phase II (Long Range Program) Brief Outlines.

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III. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE II
(Long Range Program)

Prepared by: Communications Department

SYNOPSIS: The presence of a predictable satellite at a relative short distance (200 to 20,000 miles) from the earth's surface offers a research basis for new long-distance, especially trans-oceanic, radio communication systems which will be usable both by the military and the domestic economy. The contemplated satellite program is therefore a potentially new means for strategic military communications.

7.

1. Long Term Communication Research.

a. Introduction.

A satellite that is only 200 to 800, (or 20,000 miles if possible) from the surface of the earth provides a new means for radio-transmission between various points of the earth by using the satellite as a re-transmission or repeater point. It offers possibilities from several viewpoints. These depend upon whether the model is a passive device or an active device in the sense that it does or does not have a receiver and transmitter located within it, or whether it does or does not have means within it for controlling its orientation by radio from the earth.

The possibilities can be summarized by making reference to all the previous discussions of electromagnetic tracking methods for both the passive and the active satellite. In other words, the same electronic equipment complexity in relation to the same frequency ranges discussed for radar also apply to the use of the satellite for radio communication.

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The essential change is that intelligences of various forms will be modulated on to the transmissions and demodulated at the receiver.

The first application of such a communication system after the satellite is first established and located, could be its use as a media for transmission of the required communication between certain of the geographical stations having the satellite under observation. Extensions of such communication-means to both military and the domestic economy falls into the realm of operational planning.

Many optimistic operational applications of such a new system could be envisioned, but the statements here will be confined to indications of new technical aspects. Foremost among these are the use of frequencies in the upper UHF and microwave and possibly eventually in the
8. millimeter regions at distances heretofore not possible.

Certain limitations will be inherent in such a system in which the relay point is only radio-observable between any two points during certain portions of its orbit. Modern techniques, though, provide relatively simple storage means whereby tremendous amounts of intelligence can be stored and then with the use of equally tremendous bandwidths available will allow high-speed transmissions during the period in which communication is possible.

The strategic aspects of such a system from an anti-jam viewpoint as well as its commercial use by commercial communication concerns are very apparent provided an enemy cannot observe the satellite.

This whole system is definitely in the research category though. Much could be said regarding the technical pitfalls. The most notable trouble is the fact that existing system such as the moon-radar, Diana, of the Signal Corps make use of extremely narrow receiver bandwidths in

the order of 50 cps to obtain the necessary signal-to-noise ratio. These narrow bandwidths are inconsistent with the above mentioned storage and burst systems, but not impossible if high-power efficient transmitter and antenna components at the higher microwave frequencies become available as expected.

It is to be noted that this satellite is entirely different in a radio transmission respect from the moon. The moon has been considered for communication purposes for many years, particularly since the Signal
9. Corps' radar contact with it. The outstanding disadvantage in addition to the high radio power requirement of the moon for such purposes is the fact that the long distance (approx. 500,000 km) to the moon requires a total transmission time between the earth and return of 2.4 seconds. This delay-time together with the tremendous radio-frequency powers, and equipment sensitivity required confine contemplated transmissions to the above discussed extremely narrow-receiver-bands with the result that the intelligence limit and the speed of transmission expected via the moon was at best in the order of 1 teletype word per second.

As satellites are launched which are much larger in size than the contemplated initial ones, though, the communication equipment may be correspondingly reduced in size.

b. Communications via a small (20" diameter sphere) satellite 200 to 800 miles from Earth.

As mentioned in the Introduction to this section, radio communication could be established via the initial satellite which is planned to have a diameter of 20" and to encircle the Earth at a distance of 200 to 800 miles from the surface. This particular satellite, though, has definite disadvantages from a practical viewpoint. It requires tremendous installa-

tions on the Earth of a size and complexity comparable if not larger than the existing moon-radar Diana establishment. The useful distance from a line-of-sight viewpoint would be confined to approximately 2464 miles maximum when the satellite is midway between two points separated by that distance along the circumference of the earth. The average distance available, though, would be much smaller. Also the time during which communications via this satellite route would be possible between any two points would
10. be extremely limited because of the small period during which the satellite may be seen simultaneously from any two points on the earth's surface.

Because the initial satellite is so small a relatively large amount of radio frequency transmitted power would be required with highly directive antennas. Even at the 200 mile distance, for instance, if both the transmitter and receiver had paraboidal antennas and 15 kilowatts at 700 megacycles were concentrated into a radiation beam of 1 degree between half power points, the signal-to-noise ratio at best would be 57 db for a 1 megacycle receiver bandwidth.

Radio communication via the 20" satellite, while within the realm of probability is therefore not very feasible when also considered together with the limited area on earth between which this communication is achievable. The large monetary investment required in communication stations would not be warranted in competition with existing systems. Also the limited area as well as distance over which communication is possible is not sufficiently great to make it particularly attractive for the greatest need in radio communication today, namely a means for communicating video intelligence across the Atlantic and Pacific oceans within the northern hemisphere.

c. Communication via larger satellite(s) spaced many thousands of miles from earth.

One of the outstanding long-distance communication problems in the world today is that of trans-oceanic communication. The American Telephone and Telegraph Co. and the British Post Office are cooperating in the construction of a 36 channel two-way submarine telephone cable across the Atlantic to cost in the neighborhood of $35,000,000.00. At this rate a video channel requiring 5 mcs of frequency for approximately 25 times as much bandwidth as a submarine telephone cable could conceivably cost an amount of money which would make the $35,000,000.00 look small by comparison. Also the feat is not technically feasible by known conventional means.

11. It is therefore interesting to consider what type of solution this satellite program may provide to this trans-oceanic communication problem. If the satellite were launched primarily for this purpose, two types of satellites could be used. Each has its advantages and disadvantages.

The one which would appear the most useful would be a system consisting of several spheres so located that one of them is always within sight simultaneously from the transmitting and receiving locations. In order to be so visible from most inhabited latitudes these satellites must be located above the equator at a height of approximately 2000 miles.

Another system would use a satellite having a plane mirror. The plane mirror has the disadvantage that the satellite would not stay fixed relative to the surface of the earth. The perturbations caused by the moon and the sun would require that the mirror be automatically positioned through some form of radio control. Although the actual orientation of the mirror could be accomplished by adjustment of a moving mass this would require primary power as well as a radio remote-control-system. A big advantage though, would be that the radio-frequency power required on the

ground is much more obtainable then that required by the spherical satellite. A 100' mirror, for instance, would only require a transmitter power in the order of 10 kws at low microwave frequencies (10 cmλ).

A passive repeater consisting of spherical satellites also has several advantages as well as disadvantages. A passive repeater with a spherical antenna would have no limitation in the number of two-way channels which could be obtained between various points on the surface of the earth and at various wavelengths. As isotropic radiators they are therefore the most successful type of repeaters. Also no controlled orientation is required such as would be necessary with a mirror-type satellite.

12. The earth transmitter would require 10 to 100 kws of directed power, though, even with 100 to 1000 feet diameter spheres unless a sufficient number of spheres are available. Dr. J. R. Pierce of Bell Telephone Laboratory in the February 1955 issue of Jet Propulsion magazine, for instance, indicates that one sphere would have to be 22,000 miles above the earth, while ten spheres each 100' in diameter circling the earth above the equator would only have to be 2200 miles. The path length would then only be about 1/10th that required for one sphere at 2200 miles, and the power required would be only around 100 kws which seems quite feasible. Dr. Pierce also estimates that if an active satellite repeater was provided with 30 mws output only 100 watts would be required on the earth.

A particularly intriguing application of such a system to the military is in the field of anti-jamming. This can be viewed from two incompatible operational viewpoints though. One is that the enemy would find such a system relatively easy to jam because of its availability to line-of-sight from earth. The other viewpoint bases anti-jam ability on the

same premise, but in the reverse sense, viz. during certain portions of its orbit the satellite will not always be observable to enemy-controlled transmitters provided friendly nations hold a sufficiently large portion of the earth's surface. This assumes that the satellite is placed upon an orbit that allows this operational condition in which we can see it, but the enemy cannot.

Many other technical details bearing on the radio communication research problem could be presented here in detail, but for the purpose of this report it is felt sufficient only to mention them. The Signal Corps has become aware of these, primarily, through the investigation of the possibilities of moon-relay communications.

13. These include, in addition to the astronomical aspects of providing directional antennas which must be steerable, on both the transmitting and receiving stations, many other radio propagation effects. The most obvious is the Doppler effect due to a change of relative velocity between the earth and the satellite which introduces a Doppler shift in frequency between the transmitted and received wave. The double Doppler shift experienced as a result of going and returning from the satellite, relative to the carrier frequencies which may be utilized, will be of a sufficiently small percentage depending upon height of the satellite that the receiver bandwidth may accomodate it, although at the expense of signal-to-noise ratio.

Cosmic noise at the carrier frequencies contemplated is normally not the limiting factor in detection of weak radio signals. At times, though, particularly during sun spot eruptions such noise becomes many times greater in magnitude than radio signals. It is conceivable that the satellite due to its exposure to intense ultra-violet radiation may provide

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cosmic noise in the system which could be objectionable, although such a condition is not expected.

Ionization factors at the frequencies contemplated, while they may not be expected, nevertheless may be experienced. The so-called "Luxembourg" effect in which Ionospheric (ionized layer above the earth) distortion phenomena is caused by large concentrations of radiated power in narrow beams may be noticeable. Also Ionospheric noise may modulate the transmitted waves. Such information is not available. Information of this type may therefore be a scientific contribution to this work.

A relatively large proportion of the distances of the transmission-paths even at the high latitudes contemplated may be within the
14. earth's atmosphere and the ionosphere layers. In addition to the above mentioned ionospheric effects we may therefore expect the usual reflection effects which arise when narrow beams are used. These effects will be particularly apparent when the paths run tangential to the earth's surface. Two types of refraction effects can be expected. One would be a predictable bending of the beam and the other a random bending caused by weather variations of the index of refraction. This can be as much as $1/2^{\circ}$ in the vertical and $2/10^{\circ}$ in the horizontal.

The absorption effects of molecular resonances as well as such things as water vapor are well known as providing very high attenuation in the millimeter regions. This attenuation may therefore eliminate use of frequencies this high for this purpose.

Additional factors which may arise and require consideration involve perturbations of the satellite itself which, while predictable to a certain extent, may provide unexpected difficulties.

Satellite refracted and/or reflected noise may also introduce ad-

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ditional noise in the system. The reflection coefficient provided for good radio refraction may also increase sun-radiated noise above the thermal noise relative to the ambient temperature of the earth. Normally this should not be important since the reflected noise would normally be below the thermal noise of the earth. However, measurements of solar activity have indicated instances in which the noise increases in the order of 60 db above normal from moon-reflected noise which normally is in the order of 50 db below thermal noise of the earth. Also, the surface of the satellite may experience temperature variations ranging from 100 or more degrees centigrade, and can provide in itself a thermal noise. It is conceivable that this may reach a level equivalent to the thermal noise from the earth.

It appears reasonable then that with the instruments available, or becoming available, that it is possible to achieve a long-distance type of communication such as over trans-oceanic circuits by use of a satellite repeater.

It is difficult to provide firm funding and time estimates for such a program, though, because of its research nature. It is felt, that at least a small effort should be continued in radio-communication research by the Signal Corps in close conjunction with the rocket people with the object of investigating all possibilities of using space satellites as a radio repeater communication station. It is conceivable that the long hoped-for inter-continental television networks, for instance, may be achieved by use of such radio-repeater systems.

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III. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE II
(Long Range Program)

Prepared by: Physical Sciences Division

In addition to the data and techniques planned for the Phase I program, it is visualized that the long-range program will permit more complex instrumentation, longer life, and, ultimately, recovery. Within this frame-work, the following experiments and instrumentations would be possible:

a. Orientation and control of the satellite

b. Television

c. Photography

d. Optical telescopes

e. Radio relay

Such instrumentation would permit the broadest research use of a satellite.

Proposal: The Signal Corps Engineering Laboratories propose to continue their research of Phase I into Phase II, depending on the experience gained in Phase I to guide the activities and developments of Phase II.

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III. SIGNAL CORPS PROGRAM PROPOSALS TO PHASE II (Long Range Program)
3. Materials, Electronic Parts, Subsystems
f. Electron Devices

Prepared by: Electron Devices Division

1. INTRODUCTION:

The following discussion pertains to the use of electron devices in future flights of the satellite which may have less stringent weight requirements and would therefore make possible certain devices not considered for the first flight both for the satellite and its supporting ground instrumentation. Furthermore, the additional time should make it possible to use items in development which could not be considered for Phase I. It is anticipated that later flights will attempt to transmit more information than the first satellite, imposing new requirements on the electron devices. In predicting the probable status of the devices, a period of from two to two and one-half years is considered to be available for their development.

2. DISCUSSION:

Status of Electron Devices. It is anticipated that with the present program properly directed, transistors would be available at 250 mc with power outputs of 100 m.w. with 30% efficiency minimum. Pencil triodes can be made available for this application at frequencies up to 1700 mc by relatively simple modifications of present tubes. The operating voltages, power output and filament drain will have to be lowered and the tubes and cavities ruggedized. RCA is of the opinion that this can be done given a little more than a year. These tubes would be capable of being modulated.

Changes can be foreseen in the power tube field below 1000 mc in

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view of considerable service interest in this region as a result of recent studies. A stabilotron should be available in two years for the 1200 mc band with an output of 2 megawatts. Power klystrons at this frequency and at 3000 mc will almost certainly be available at much higher power (10 Mw) levels.

In the event that a solar cell is not available for the first satellite, it should certainly be available for a later one with many of its characteristics improved.

3. RECOMMENDATIONS:

It should be emphasized however, that in many of the devices indicated above as much of the two to two and one-half year period as possible should be set aside to attain the desired characteristics important to this specific application. It is recognized that this timing does not permit a determination of the reliability of the newer devices other than by intuitive prediction. For a venture of this significance, a performance test period for determining the equipment reliability would certainly be preferred to a "last minute" assembly of new and untried devices and hopeful wishes that they will perform as planned.

From an applications viewpoint, it is urged that specific requirements be stated as early as possible, and that adequate bench testing time be allocated where untried electron devices are being incorporated in equipment.

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