AN INVESTIGATION OF THE FACTORS TO BE CONSIDERED IN PLANNING COMMUNICATIONS SATELLITE SYSTEMS

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PREFACE

Ever since the orbiting of artificial earth satellites became a possibility, engineers and scientists in the field of communications have dreamed of their providing, on a reliable basis, the means for near real time communications over long distances. This technique appeared to have certain advantages. It could provide communications between any two stations which had the satellite within mutual visibility; it would not be dependent on the ionosphere for long distance communications; frequency interference problems would be greatly reduced because essentially line of sight frequencies could be used; and since the technique was not dependent to any great extent on cable systems, mass communications to about any place on the globe and, on a very reliable basis, appeared to be a distinct possibility.

After the first successful launching of an artificial satellite (Sputnik I), interest in the use of satellites for communications purposes became intensified, and a series of communications-type satellites were built and successfully placed in orbit around the earth. These were mostly research vehicles used to test techniques of stabilization, active circuitry, power generation and storage devices, antennas, tracking devices and other associated systems. Additional communications satellite projects were started by the Department of

Defense but later canceled for various technical reasons. Although the communications satellite technique has been demonstrated in a variety of ways, no operational communications satellite system has yet been developed and those systems under development now are not expected to become operational for some years.

Within the past year the author of this thesis was assigned, among other things, to be the focal point within the Directorate of Command Control and Communications of the Headquarters, United States Air Force, for the communications satellite projects of the Department of the Air Force. While he was serving in this capacity, the Department of the Air Force was assigned the responsibility for the developing and launching of the space elements of a new Department of Defense communications satellite program. This program is to consist of two space systems -- a medium altitude random polar orbit system of over twenty satellites between 6,000 and 12,000 miles altitude, and a high capacity, synchronous orbit system. Following the establishment of this program, the author performed a rather cursory examination of the communications coverage and operational problems which might be expected from the proposed medium altitude random polar orbit system. The results seemed to indicate that a system using inclined or equatorial orbits, with satellites uniformly spaced, would have decided advantages over the system proposed by the Department of Defense. It was therefore decided to publish the findings and circulate the study to provoke thought and comment. The paper, "AN ARGUMENT FOR

EQUALLY SPACED MEDIUM ALTITUDE COMMUNICATIONS SATELLITES" was prepared and circulated among the elements of the Air Staff and the Air Force Systems Command.

While preparing this paper it was found that very little information was available on the many factors which should go into the planning for communications satellite systems. The little information which was available appeared to only exist in fragments. As the result the author was motivated to investigate as many of the pertinent factors as possible and prepare the results of the findings as this thesis.

Acknowledgment and appreciation is hereby expressed to Dr. Herbert L. Jones for the encouragement, guidance and assistance which he furnished as chairman of the committee and the generous contribution of his time in helping with the many problems and details associated with the Doctorate program of the candidate. Similar appreciation is also expressed for the guidance furnished by Dr. Robert MacVicar, the Dean of the Graduate School and the following members of the committee: Professors H. E. Harrington, F. C. Todd, C. F. Cameron, W. J. Bentley.

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CHAPTER I

BACKGROUND INFORMATION

Once the successful orbiting of artificial earth satellites had been demonstrated, much attention was given to their use for providing the means for reliable point-to-point communications. Communications via satellite relay have several important advantages over other more conventional means of communication. Among these advantages are:

1. Increased propagation reliability through the use of line of sight communications which are less subject to the effects of changes in the ionosphere.

2. Probably less vulnerable to physical damage by an enemy.

3. Fewer switching and relay requirements because most of the communications will be direct from user to user through a single satellite relay.

4. A very large area of coverage on the surface of the earth is possible with satellites.

5. Not limited to readily accessible areas. Satellite systems can provide communications to any place on the surface of the earth and space where suitable receiving equipment is available and the receiving power levels are high enough for the particular receivers.

Although most of the development on communications satellites and launch into space effort has been on non-synchronous orbit, relatively low altitude vehicles, many space authorities believe that the big economic and political payoff will result from these satellites when they are placed in synchronous 24-hour orbits. These 22,000 mile high satellites will remain stationary with respect to any point on the surface of the earth and because of this will eliminate or reduce doppler shift problems, simplify operational problems because no antenna tracking and handover requirements between stations will exist, and permit very large area coverage from a single satellite. An English scientist and author[1], foresaw the day when stationary satellites would make television available to everyone on earth. He declared that

> Of all the applications of astronautics during the coming decade, I think the communications satellite the most important...it is now widely conceded that this may be the only way of establishing a truly global TV service. The political, commercial, and cultural implications of this, however, do not seem so thoroughly appreciated... The printed word plays only a small part in this battle for the minds of largely illiterate populations, and even radio is limited in range and impact. But when line-of-sight TV transmissions become possible from satellites directly overhead, the propaganda effect may be decisive ...It may well determine whether Russian or English is the main language of the future.

---

[1] Staff Report of the Select Committee on Astronautics and Space Exploration, 86th Congress, "The Next Ten Years in Space," 1959 - 1969.

History of Communications Satellites

In the early 50's, when it appeared that the launching of artificial earth satellites would occur within the next few years, forward looking communicators were seriously discussing their use for communications purposes. Sputnik I was successfully launched into earth orbit in October, 1957. This successful event appeared to stimulate an increased emphasis on the uses of space within the United States. The Select Committee on Astronautics and Space Exploration was created by House Resolution 496, March 5, 1958, 85th Congress, and The National Aeronautics and Space Administration (NASA) and the Advanced Research Projects Agency (ARPA) of the Department of Defense were formed in 1958. Most of the early communications satellite projects were carried out under ARPA order. The ARPA operated on a project basis in three categories:[2]

1. Research not identified with a specific military requirement.

2. Research which related to the primary functions of two or more military services.

3. Research which, for other reasons, was better handled by an agency other than one of the military services.

NASA was formed, among other things, to be responsible for the development of the peaceful purposes of space.

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[2] DOD Document "The Advanced Research Projects Agency," August, 1960.

Shortly after the Advanced Research Project Agency was formed, Major General Bestic, the USAF Director of Telecommunications, made a speech to members of that organization and pointed out the advantages which could be expected from the development of communications satellites. He also urged them to take vigorous action to develop reliable communications satellites.

The initial communications satellite program which was established by the ARPA consisted of the following:

1. A delayed repeater satellite named COURIER under Army technical management by ARPA order. Later the entire project was completely transferred to the Army in 1960.

2. A UHF polar orbit satellite called STEER proposed by the Air Force.

3. A microwave polar orbit satellite called TACKLE. This satellite was planned to be a research and development program for a follow-on project known as DECREE.

4. A synchronous orbit 24-hour satellite called DECREE. This DECREE program was later canceled as an ARPA project and transferred to the Army and was renamed the ADVENT program.

One of the COURIER packages was placed in orbit and lasted about 18 days. Budget limitations and other considerations led to the cancellation of the STEER and TACKLE programs and the termination of the COURIER program in 1960.

The initial budget estimates for the ADVENT program were much lower than the actual costs and additional money was needed. In addition to the fund problems, the ADVENT program suffered from many technical and management problems on the communications package, spacecraft, and booster. These many problems caused the Director of Research and Engineering, the Honorable Harold Brown, to conduct a study and reappraise the complete ADVENT program. As the result of this study, Secretary McNamara on May 23, 1962, and Dr. Harold Brown, Director of Research and Engineering for the Department of Defense, on May 26, 1962, issued directives, which, among other things, canceled the ADVENT program and established two new programs: A medium altitude randomly spaced polar orbit communications satellite program, and a synchronous orbit communications satellite program. The Army was made responsible for the development of the surface based stations and the Air Force was made responsible for the development, orbit control, and launch of all space elements of both systems. Both of these systems are currently being slipped from their originally expected operational dates because funds for them have not as yet been released.

The Bell Telephone Laboratories communications satellite "TELSTAR" was launched on June 7, 1962, by NASA. The TELSTAR was designed for real time relay of wide band communications. It could relay television information or the band could be subdivided into individual band segments to permit large individual channel handling capability to permit multiple simultaneous communications. This first TELSTAR launch was highly successful and it appears that all of the TELSTAR experiment objectives were met. These were:

1. To test an orbital broadband communications satellite.

2. To test the reliability of electronics equipment in the satellite under the stress of launch and the environment of space.

3. To provide additional knowledge on satellite tracking techniques.

4. To obtain measurements of radiation levels in space.

5. To test associated ground station equipment and operating techniques.

Primary power limitations permitted the TELSTAR to be used for only about 100 minutes each day but even with this limitation the TELSTAR was able to dramatically demonstrate its capability to relay video information by providing the means for transatlantic relay of television. TELSTAR failed after a few months in orbit. It is believed that the increased radiation in space brought about by the high altitude nuclear tests in the summer of 1962 probably caused its failure.

The NASA-sponsored communications satellites RELAY and SYNCOM were launched within the past few months. The operation of RELAY was not very successful. Problems with the power supply caused it to be quite ineffective. The SYNCOM satellite which was built by the Hughes Corporation was to be launched into synchronous orbit. It was tracked during the boost phase of flight but the beacon ceased to transmit after the injection motor was turned on. The general opinion as to its fate is that the injection motor probably exploded when it was turned on.

In addition to the history on active communications satellites, there has been some effort to relay information via passive reflectors. This thesis will deal primarily with the active systems. It should be noted, however, that the moon has been used as a reflector of signals and communications via moon scatter are being conducted on a routine basis. These communications are limited to low information rates and require high transmitter power and very high gain antenna systems. The Echo-type satellites which are metallic coated spheres have also been used for communications relay. In addition, attempts have been made to place several billion dipoles in orbit but the launch attempts failed. This so-called Westford project is somewhat dormant at the moment -- probably as the result of objections to the use of these dipoles in space by the international scientific community, particularly radio astronomers.

CHAPTER II

A DESCRIPTION OF THE PROBLEM

From the previously furnished background information, it can be seen that there has been considerable effort in the communications satellite field with only limited results. Probably over a billion dollars have gone into programs which were canceled before they got off the ground or achieved only limited success. Although many of these programs were announced as canceled for budgetary reasons, it appears that also coupled with these reasons are factors which would have made the programs technically impossible. Space communications relay systems will be difficult to achieve. Limited booster capability requires that the satellites be of low mass and yet be capable of operating effectively over great distances within a unique environment. Because replacement or repair in space will be difficult and costly, reliability will be extremely important. As evidenced by the cancellation of many programs and the progress on those programs which were permitted to proceed, there is some question as to the adequacy of the technical planning which has been and is now being done on the various satellite communications systems.

Because of the newness of the communications satellite technique, there is a dearth of information which can be used for planning purposes. The information which is available is scattered and many times incomplete. Because of this, it was felt that there was a need for a study of the various factors involved in communications satellite planning. Because of this need, this thesis is directed toward that objective. The thesis will deal primarily with the problems associated with the geometry of satellite deployment and many of the factors which affect the electrical or radio frequency part of the system.

In the chapter on the geometry of satellite deployment, the earth's distance coverage by single satellites will be determined; the earth's area coverage by a single satellite on a single satellite pass and the instantaneous surface area covered by a single communications satellite will be calculated and the method of calculation discussed. Satellite periods at different altitudes and different orbital configurations will be shown.

Factors which affect the operation and design of the communications satellite will be investigated. System loss or gain, atmospheric attenuation, stabilization of the satellite, primary power supplies and other items which are pertinent to the problem will be considered.

CHAPTER III

SATELLITE DEPLOYMENT

The deployment possibilities for communications satellites are without limit. Certain geometries of satellite deployment appear to have more desirable advantages than the others for certain applications. The lower altitude satellites have the advantage of being able to communicate with ground stations with less power and therefore can be made lighter. However, under these same conditions the lower altitude satellites cause greater operational problems because of antenna tracking and handover requirements. In addition, the lower altitude satellites are in the earth's shadow for longer periods of time and therefore require more solar cell and power storage capacity. Then, too, the earth distance covered by a satellite is a function of the satellite's altitude.

Earth Distance Covered by a Single Communications Satellite

In determining the number of satellites required for this purpose, two approaches to the problem may be taken. The number of satellites may be selected and the maximum ground distance between ground stations can then be determined. Alternatively the distance between ground stations can be selected and the altitude and number of satellites required to satisfy this requirement can then be determined. The former method was chosen. The first approach then is to determine the distance coverage over the surface of the earth for a single satellite at different altitudes and then choose a logical altitude for use in an example situation.

First, consider the earth with a mean radius of 6370 KM or 3440 NM. Using these figures the mean circumference of the earth is calculated to be about 40, 000 KM or 21, 600 NM.

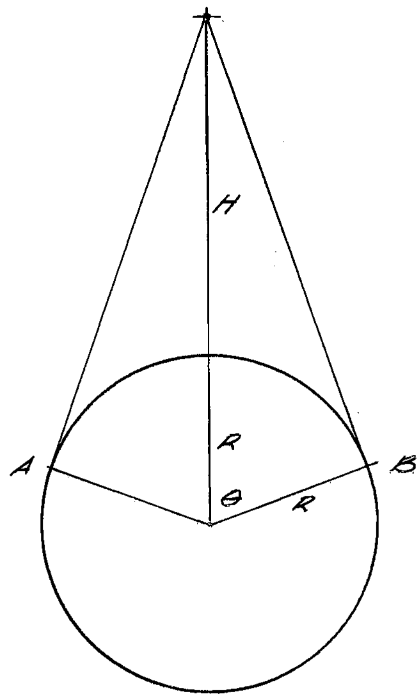
Next, the distance coverage over the surface of the earth for a satellite at various altitudes is computed (see Fig. 1). Line of sight coverage in KM is

$$\frac{2(\cos^{-1}\frac{R}{R+H})}{360}\,(40,000)\ . \tag{1}$$

See Table I for coverage computed for altitudes of various earth radii.

Earth's Area Coverage by a Single Satellite on a Single Orbital Pass

From Table I the earth distance coverage can be used to calculate the area coverage. By revolving the curve represented by the earth distance around the X axis (see Fig. 2) the area covered by one pass of the satellite can be calculated.



$$\theta = \cos^{-1}\left(\frac{R}{R + H}\right) ; \quad \frac{2\theta}{360}\,(40,000) = \frac{\left(\cos^{-1}\frac{R}{R + H}\right)}{180}\,(40,000)$$

kilometers = earth's distance AB.

Fig. 1 Calculation of Earth Distance Covered by a Single Satellite

TABLE I

EARTH'S SURFACE DISTANCE COVERED BY SATELLITES AT DIFFERENT ALTITUDES

Satellites Altitude						
Earth's Radii	KM	NM	$\frac{R}{R \& H}$	θ Degrees	Earth Surface Distance (KM)	Earth Surface Distance in Nautical Miles
1	6,370	3,400	.5 (1/2)	60	13,333	7,200
2	12,740	6,880	.3333 (1/3)	70.4	15,645	8,448
3	19,110	10,320	.25 (1/4)	75.4	16,755	9,048
4	25,480	13,760	.2 (1/5)	78.3	17,400	9,396
5	31,850	17,200	.1666 (1/6)	80.3	17,844	9,636

Note: Circumference of the earth is approximately 40,000 KM or 21,600 NM.

Referring to Fig. 2: For rotation of the arc length -a to +a of $x^2 + y^2 = R^2$: (2)

1. $S = 2\pi \int_{-a}^{a} y \, ds$

2. and $ds^2 = dx^2 + dy^2$; $ds = \left[1 + (\frac{dy}{dx})^2\right]^{\frac{1}{2}} dx$

3. and $y = (R^2 - X^2)^{\frac{1}{2}}$.

4. Differentiating $X^2 + y^2 = R^2$; $\frac{dy}{dx} = \frac{-x}{y}$.

Substituting 3 in 4; the result in 2; and this result and 3, in 1,

$$S = 2\pi R \int_{-a}^{a} (R^2 - X^2)^{\frac{1}{2}} (R^2 - X^2)^{-\frac{1}{2}} dx;$$

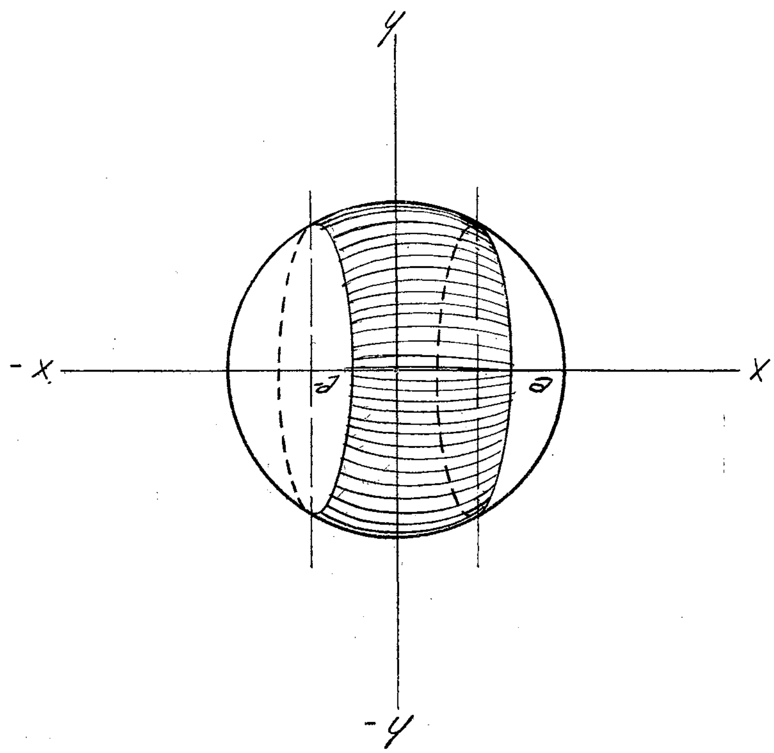


Fig. 2 Earth's Area Covered by a Single Satellite in One Pass

Integrating the above,

$$S = 2\pi R \left[X\right]_{-a}^{a} .$$

Evaluating,

$$S = 2\pi R\ (2a) = 4\pi Ra. \tag{3}$$

Thus, the equation for determining the area coverage by a single satellite during one orbital pass has been developed. The results of the calculations of earth coverage during a single orbital pass of a communications satellite from 1 to 5 earth radii in altitude are shown in Table II.

TABLE II

SATELLITE EARTH SURFACE COVERAGE

Altitude (Earth Radii)	Kilometers^2	Nautical Miles^2	% of Earth's Surface
1	442×10^6	128×10^6	86.6
2	481×10^6	139.6×10^6	94.3
3	494×10^6	143×10^6	96.8
4	500×10^6	145×10^6	98.0
5	503×10^6	146×10^6	98.6

The preceding formula can be used to determine the percentage of the earth surface covered by a single orbital pass.

Since the surface of the earth covered by the satellite orbital pass is $4\pi Ra$, then

$$\frac{4\pi Ra}{4\pi R^2} \times 100 = \text{percent of earth surface covered}$$

$$= \frac{a}{R} \times 100 \tag{4}$$

referring to Fig. 3.

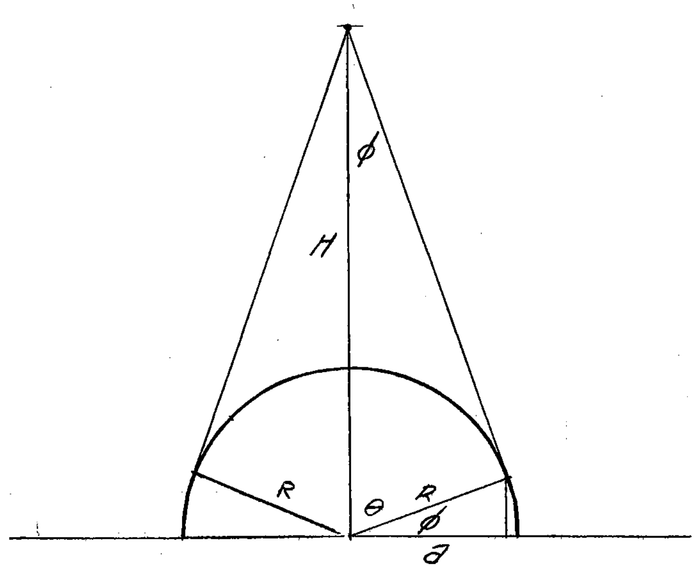


Fig. 3 Calculation of Percentage of Earth's Surface Covered by Single Satellite Pass

$$\cos \emptyset = \frac{a}{R} \quad \text{and} \sin \emptyset = \frac{R}{R + H} = \frac{R}{nR} = \frac{1}{n} .$$

Where R + H = nR

$$\cos^2\emptyset + \sin^2\emptyset = 1$$

$$\cos^2\emptyset = 1 - \sin^2\emptyset$$

$$\cos \emptyset = \sqrt{1 - \sin^2\emptyset} = \sqrt{1 - (\frac{1}{n})^2}$$

$$= \sqrt{1 - \frac{1}{n^2}} = \frac{1}{n}\sqrt{n^2 - 1}$$

$$\therefore \cos \emptyset = \frac{1}{n}\sqrt{n^2 - 1} = \frac{a}{R}$$

and for altitude above surface in earth's radii

For:

One radius; n = 2

Two radii; n = 3

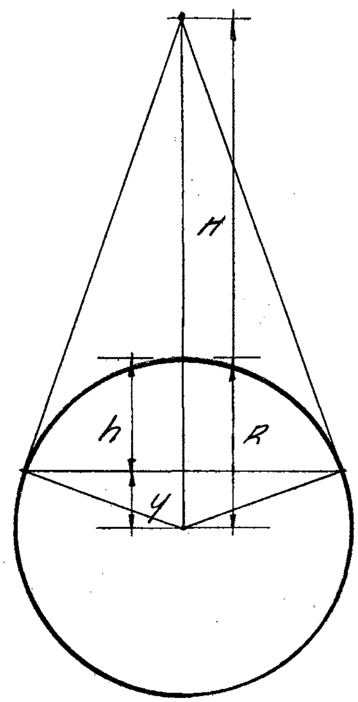
Three . . .

and substituting the correct values in the preceding equation the percentage of earth's surface covered by a single satellite pass for different satellite altitudes is shown in the fourth column of Table II.

Instantaneous Surface Area Covered by a Single Communications Satellite

To complete the picture of the earth coverage by a single communication satellite at any point in time, the area covered is now computed.

Consider the example shown in Fig's. 4 and 5. Figure 4 shows the diameter or chord length of the small circle resulting from the intersection of the cone of coverage from the satellite and the method for calculating the distance of this chord from the center of the earth. The difference between this distance and the radius of the earth represents the height of the spherical segment created by the intersection of the sphere and the cone. Referring to Fig. 5, the surface of curvature of the spherical zone is determined by rotation of the ARC length around the X axis.



For Satellite Altitude of
2 earth radii

$\phi = 70.4^{\circ}$

$h = R - Y$

$= 6370 - 2137 \text{ km} = 4233$

$Y = R \cos \theta$

$= .3354 \times 6370 \text{ km}$

$= 2137 \text{ km}$

Fig. 4 Calculation of Height of Spherical Segment

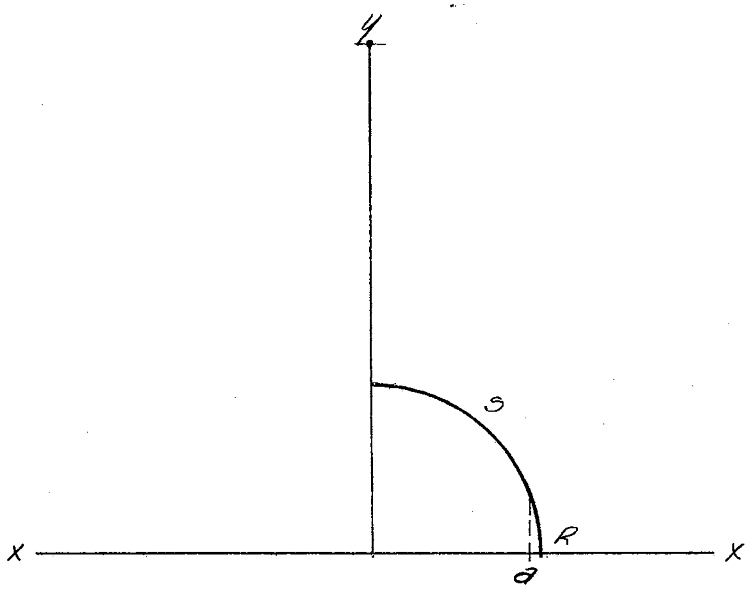


Fig. 5 Calculation, Instantaneous Earth Area Coverage by Communications Satellite

For rotation around the x axis

(1) $S_x = 2\pi \int_a^R y \, ds$

(2) and $dx^2 = dx^2 + dy^2$; $ds = (dx^2 + dy^2)^{-\frac{1}{2}} = \left[1 + (\frac{dy}{dx})^2\right]^{-\frac{1}{2}} dx$

(3) and $y = (R^2 - x^2)^{\frac{1}{2}}$.

(4) Differentiating $x^2 + y^2 = R^2$; $\frac{dy}{dx} = \frac{x}{y}$.

Substituting (4) in (2); and substituting (2) and (3), in (1):

$$S_x = 2\pi R \int_a^R (R^2 - x^2)^{\frac{1}{2}} (R^2 - x^2)^{-\frac{1}{2}} dx .$$

Integrating:

$$S_x = 2\pi R \,[X]_a^R = 2\pi R\,(R - a) = 2\pi R\,(6370 - 2137)$$

$$= 2\pi R\,(4,233) = 2\pi\,(6370)\,(4,233) = 2\pi\; 26,964,210$$

$$= 169,431,524 \text{ sq. km.} \qquad (5)$$

A fair share of the earth's surface.

Satellite Period

In order to get a feel for the time coverage by a single satellite in circular orbit, the period of the satellite orbit must be known.

1. Since $\frac{mv^2}{R} = \frac{Gm\,M}{R^2}$

   Where:

   M = Mass of earth

   R = Total distance from center of the earth

   G = Gravitational constant

   V = Velocity of the satellite

   m = Mass of satellite

2. and from 1 we obtain:

$$V = \sqrt{\frac{GM}{R}}$$

   Since $T = \frac{\text{Distance}}{\text{Velocity}}$

   and distance $= 2\pi R$

3. Then $T = \dfrac{2\pi R}{\sqrt{\frac{GM}{R}}} = \dfrac{2\pi R^{3/2}}{\sqrt{GM}}$

Set $R = R_e + H$

Where:

R_e = Radius of earth

and H = Altitude of satellite in earth radii

then $$T = \frac{2\pi(R_e + H)^{3/2}}{\sqrt{GM}}$$

$$= \frac{2\pi}{\sqrt{GM}}(R_e + H)^{3/2} = \frac{2\pi R_e^{3/2}}{\sqrt{GM}}\left[1 + \frac{H}{R_e}\right]^{3/2}$$

Lumping $\frac{2\pi R_e^{3/2}}{GM}$ into a constant:

$$\text{Then } T = 84.4\left[1 + \frac{H}{R_e}\right]^{3/2} \text{ Minutes} \quad (6)$$

The time calculations are included in Table III.

TABLE III

ORBIT PERIODS

Satellite Altitude (Radii)	Orbit Time for $\frac{H}{R_e}$ (Minutes)	(Hours)
1	237	3.95
2	438	7.3
3	675	11.25
4	943	15.73
5	1240	20.67

Example: Twelve Evenly Spaced Satellites in Polar Orbit

Using the calculated information, the data for a satellite altitude of two earth radii (6880 NM) is selected for use in justifying an example of coverage to be expected from a system of twelve satellites, equally spaced in three polar orbital planes, equally displaced around the earth. The satellites of each orbital plane are in the same relative position in each plane. In showing the coverage the particular example chosen represents the worst case; where a satellite from each plane is over the equator and consequently they are at greatest maximum distance from each other. First consider (Fig. 6) the four satellites of a single orbital plane.

From Table I, note that the ground distance covered by a single satellite at 2 earth radii is 8,448 NM or slightly less than half of the circumference of the earth. By construction using the satellite altitude of two earth radii, and four satellites equally displaced in the polar orbit, the expected earth's coverage by the satellites is shown in Fig. 6. From the figure it can be seen that there are four rather large areas of two station coverage, four lesser areas of single station coverage and a still less area of no coverage.

Next consider the increased earth coverage which results by adding the coverage afforded by the satellites in the two remaining orbits for the case where the satellites in the three planes are in the same relative position. In this case, six satellites are equally spaced in relation to

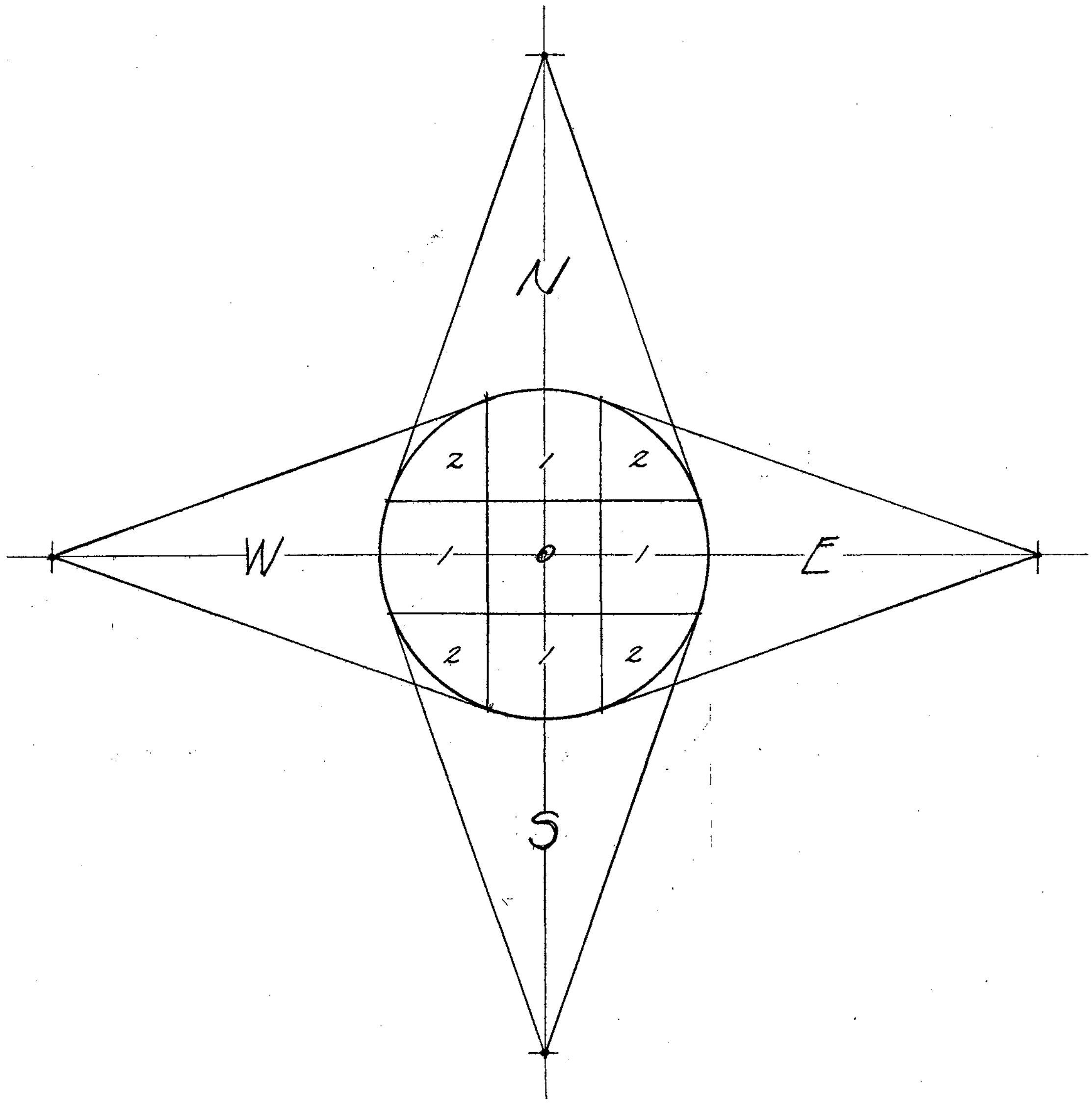


Fig. 6 Earth Coverage by Four Satellites in Same Polar Orbit (Altitude Equal to 2 Earth Radii)

each other over the equator (Fig. 7) and three satellites over each pole, giving triple coverage in the polar areas. The number of satellites covering each area is indicated on the drawing. The drawing shows a view of the system looking down at one of the poles.

The projection of the satellites' coverage over the polar areas on the plane of the paper is shown by the circle of lesser diameter. See

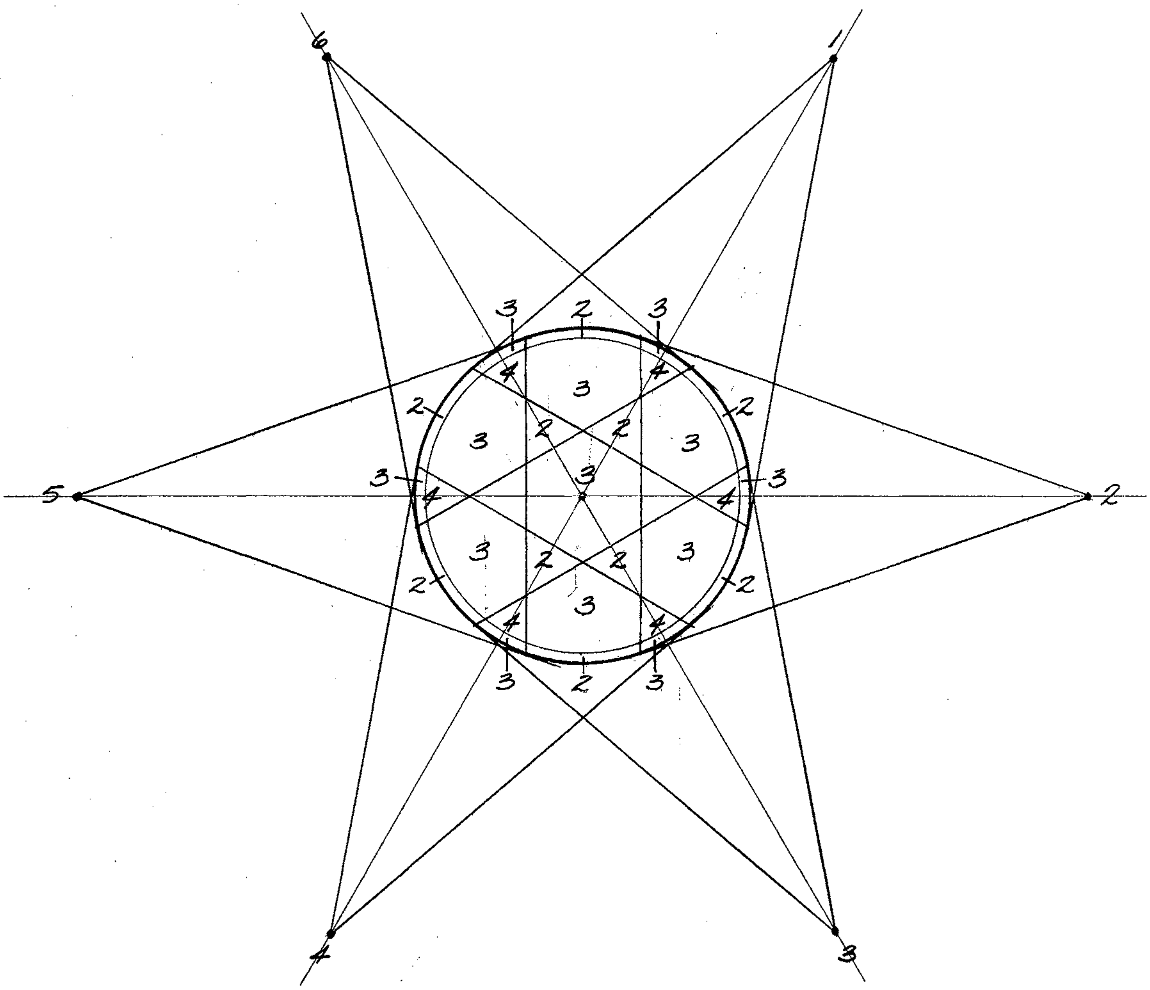
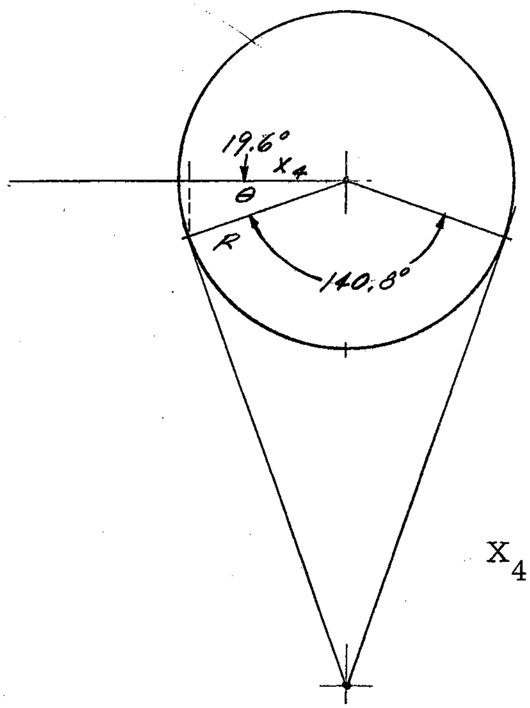


Fig. 7 Earth Coverage by System of 12 Satellites (4 Satellites per Orbit, 3 Orbits) (Altitude of Satellites is 2 Earth Radii)

Fig. 8 for the method of determining the extent of the projected area.

Now the development of the satellites' coverage is continued by determining hemispherical coverage of the earth when viewed from a point directly over the equator. Figure 9 shows the additional coverage over that of Fig. 6, which results by introduction of the two additional satellites which are over the equator and add to the coverage of the



$$X_4 = R \cos \theta \quad \text{and} \quad \theta = 19.6^{\circ}$$

$$X_4 = .9420\,R$$

Fig. 8 Projection of Single Satellite Coverage on Plane Located on Diameter of Earth and Normal to Satellite Coverage Cone (Altitude of Satellite equal to 2 Earth Radii)

hemisphere. These two additional satellites could be considered to be the satellites numbered 3 and 4 in Fig. 7. See Fig. 10 for method of determining the projection of the satellite coverage by 3 and 4 on the plane of the paper as shown in Fig. 8. The satellite coverage for each area is indicated on Fig. 9. Note however that the triple coverage by the three satellites over each pole is not indicated since this is redundant coverage by satellites at about the same point in space.

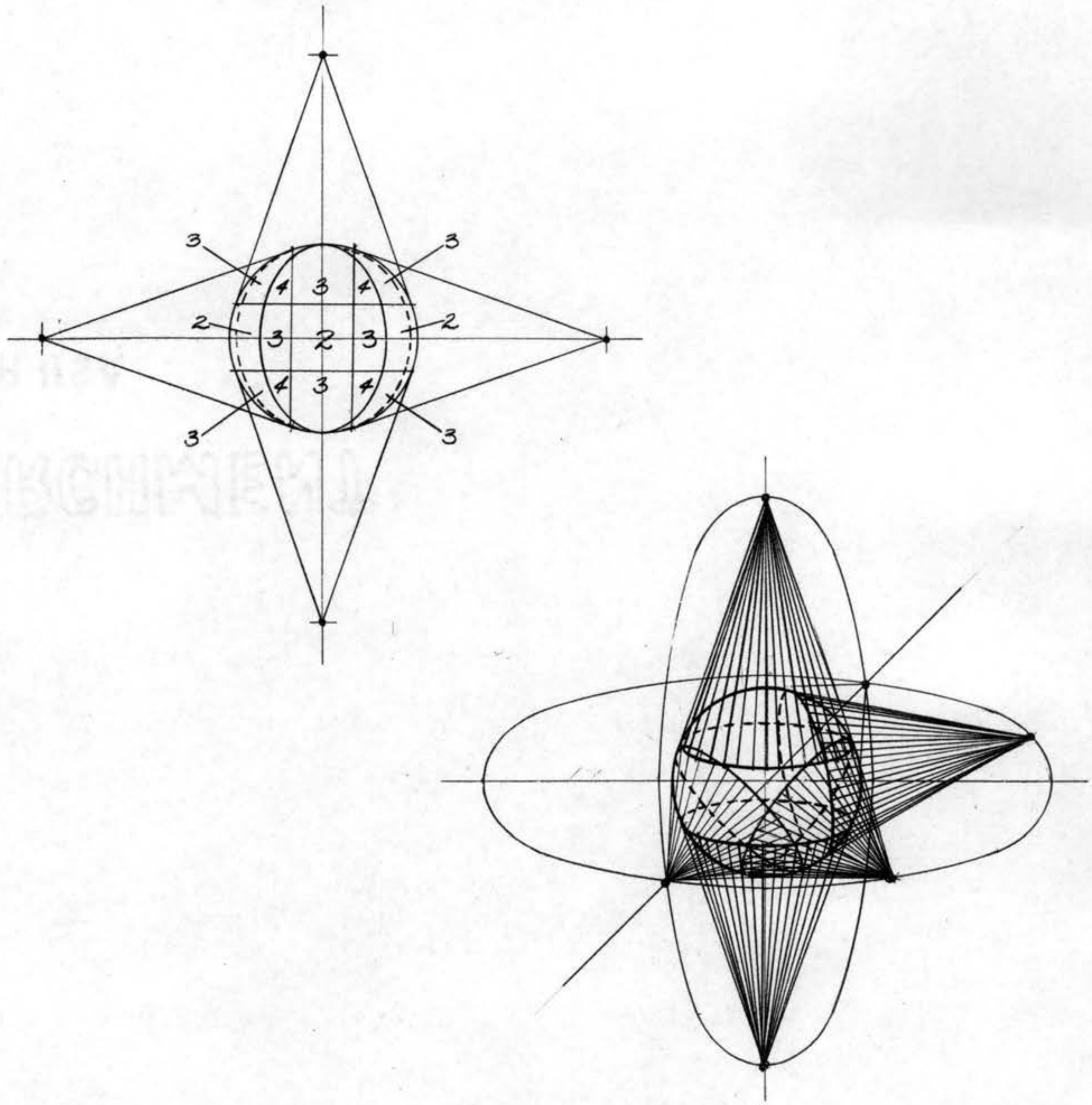
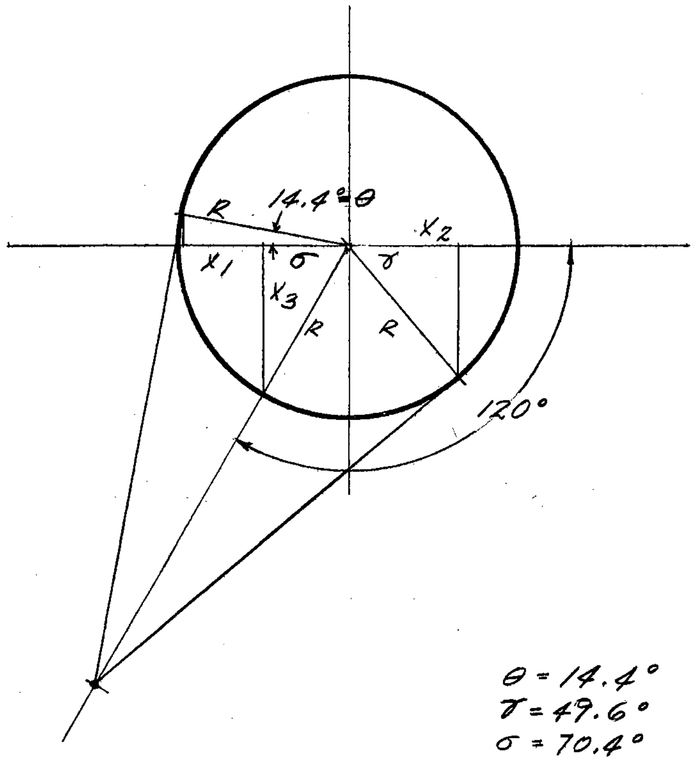


Fig. 9 Hemisphere Coverage by One Polar Orbit (4 Satellites) with Two Additional Satellites in the Hemisphere Equatorial Plane and also in Polar Orbit



$$X_1 = R\cos\theta = R\cos 14.40^{\circ} = .9685\,R$$

$$X_2 = R\cos\gamma = R\cos 49.6^{\circ} = .6481\,R$$

$$X_3 = R\cos\sigma = R\cos 70.4^{\circ} = .3354\,R$$

Fig. 10 Projection of Satellite Coverage on a Plane at 120° from the Axis of the Cone (Altitude of Satellite is 2 Earth Radii)

Now for the purposes of discussion, consider the ground station coverage in terms of the earth's central angle. Antenna elevation above the horizon for acquisition or handover will be assumed to be 10 degrees. From Fig. 11 can be seen the results of calculation which show that the earth's central angle coverage will be about 120 degrees or 1/3 of the satellite period in polar orbit. This figure also represents 1/3 of the earth's rotation or 8 hours of coverage in the equatorial plane. The distance between ground stations and the period of satellite mutual acquisition are interdependent. As the ground station separation distance is decreased the mutual acquisition coverage time is increased. Detailed consideration of these factors is not within the scope of this thesis.

General Considerations for Other Systems of Non-Synchronous Satellites

From the discussion of the system of twelve satellites, it appears that a system using all satellites in a polar orbit gives far greater coverage of the arctic regions with what appears to be an unnecessarily large amount of redundant coverage. Here are points on the earth's surface where every satellite must pass during each orbit. Since most of the point-to-point communications take place in the temperate zones nearer the equator, it appears that a system of inclined orbital planes may give more even coverage.

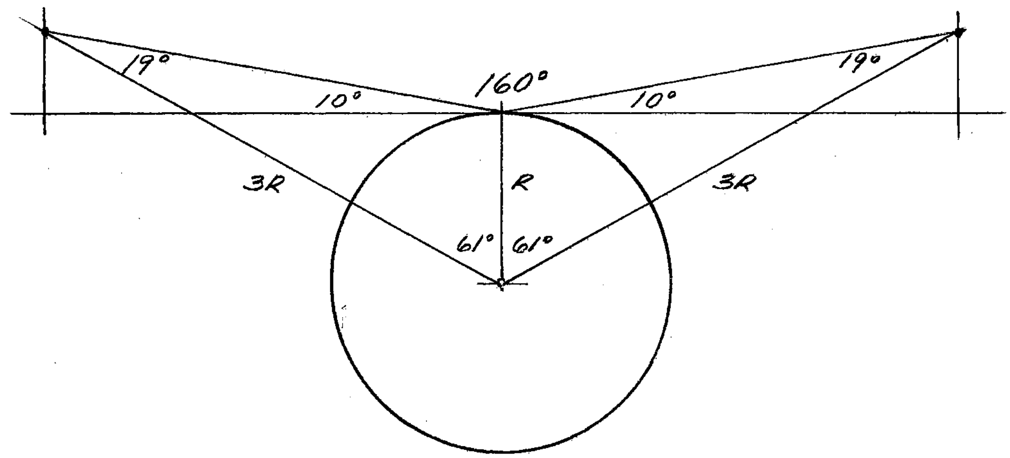


Fig. 11 Central Angle Coverage Represented by a Single Ground Station (Ant. Elevation Angle 10°)

We might even consider for study a system using satellites in an equatorial plane and a separate polar orbit of three or four satellites to satisfy the needed coverage for the very small areas at the poles.

For example, using 21 satellites three could be in polar orbit and 18 in equatorial orbit. (See Fig. 12.) This means we could have a satellite in the equatorial plane spaced every 20 degrees. Considering that each satellite at two earth radii altitude sees a distance of 8,448 miles at any point in time and 94.3% of earth surface during a single pass, coverage is formidable.

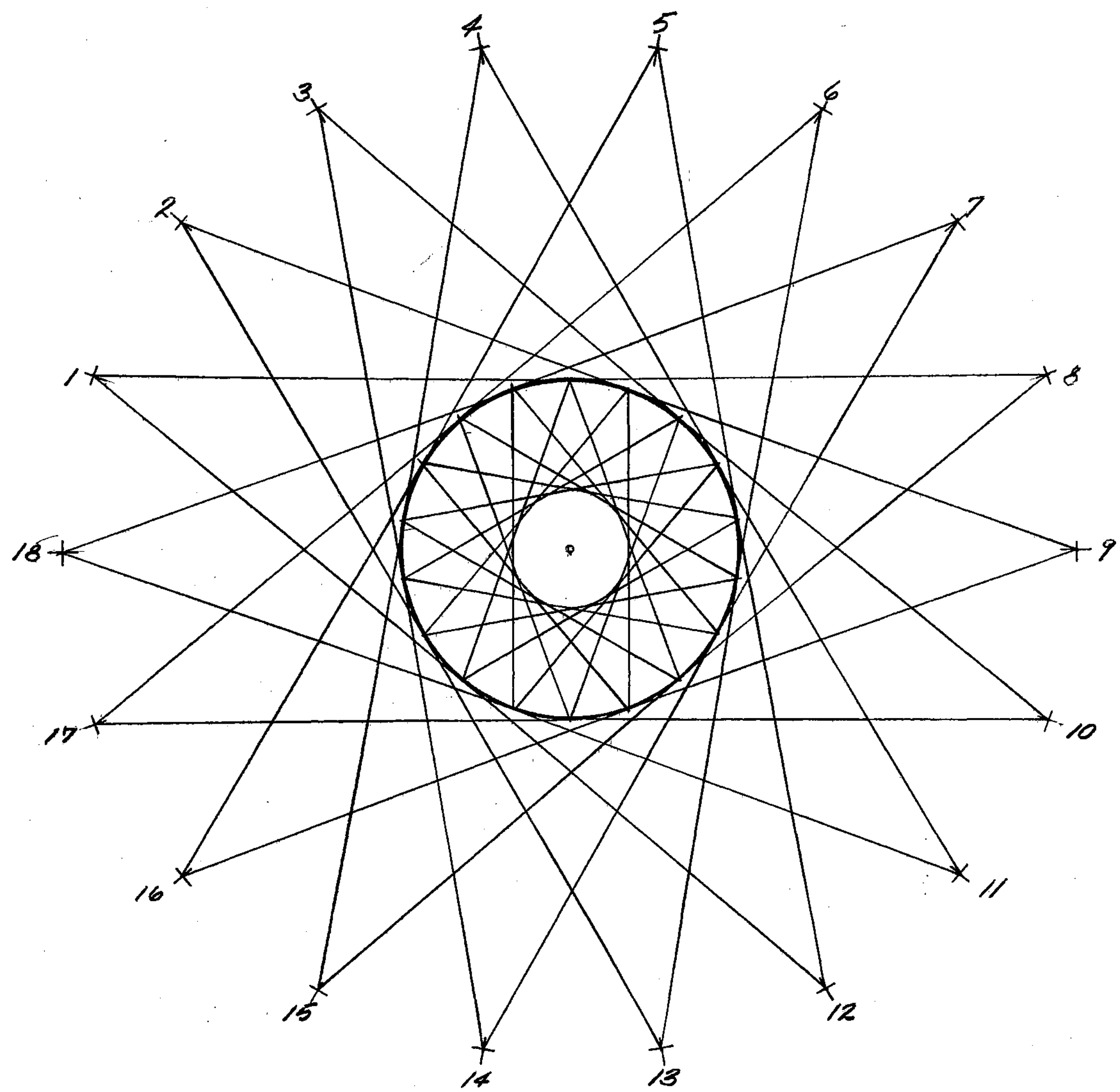


Fig. 12 A System of 18 Satellites in Equatorial Orbit

The Synchronous Orbit Communications Satellite

The synchronous orbit satellite is a satellite which is in a 24-hour orbit and therefore has the same orbital period as the earth. Such a satellite if launched in the equatorial plane will appear to remain stationary with respect to a point directly beneath it on the surface of the earth. If the satellite is launched in an inclined plane, it will appear to make a figure eight over the surface of the earth with the center of the figure eight always passing over the same point on the equator. The 24-hour satellite has several important advantages and a few disadvantages.

In this system one or more satellites are used as relays to provide continuous real-time communications among a number of ground stations. The geometry of what might be considered to be a typical system is shown in Fig. 13.

The advantages of the synchronous communications satellite appear to be:

1. It can provide wide coverage multiple user service.
2. Microwave frequencies can be used to provide reliable propagation and relieve some of the traffic congestion on the lower frequencies.
3. High capacity wide band traffic could be accommodated.
4. Little or no ground tracking will be required.
5. Survivability of the ground stations can be improved because fixed antennas are more easily hardened against nuclear attack.
6. Little or no doppler shift problems will exist.

The disadvantages appear to be:

1. Sophisticated launching techniques will be required.
2. High power boosters will be required.

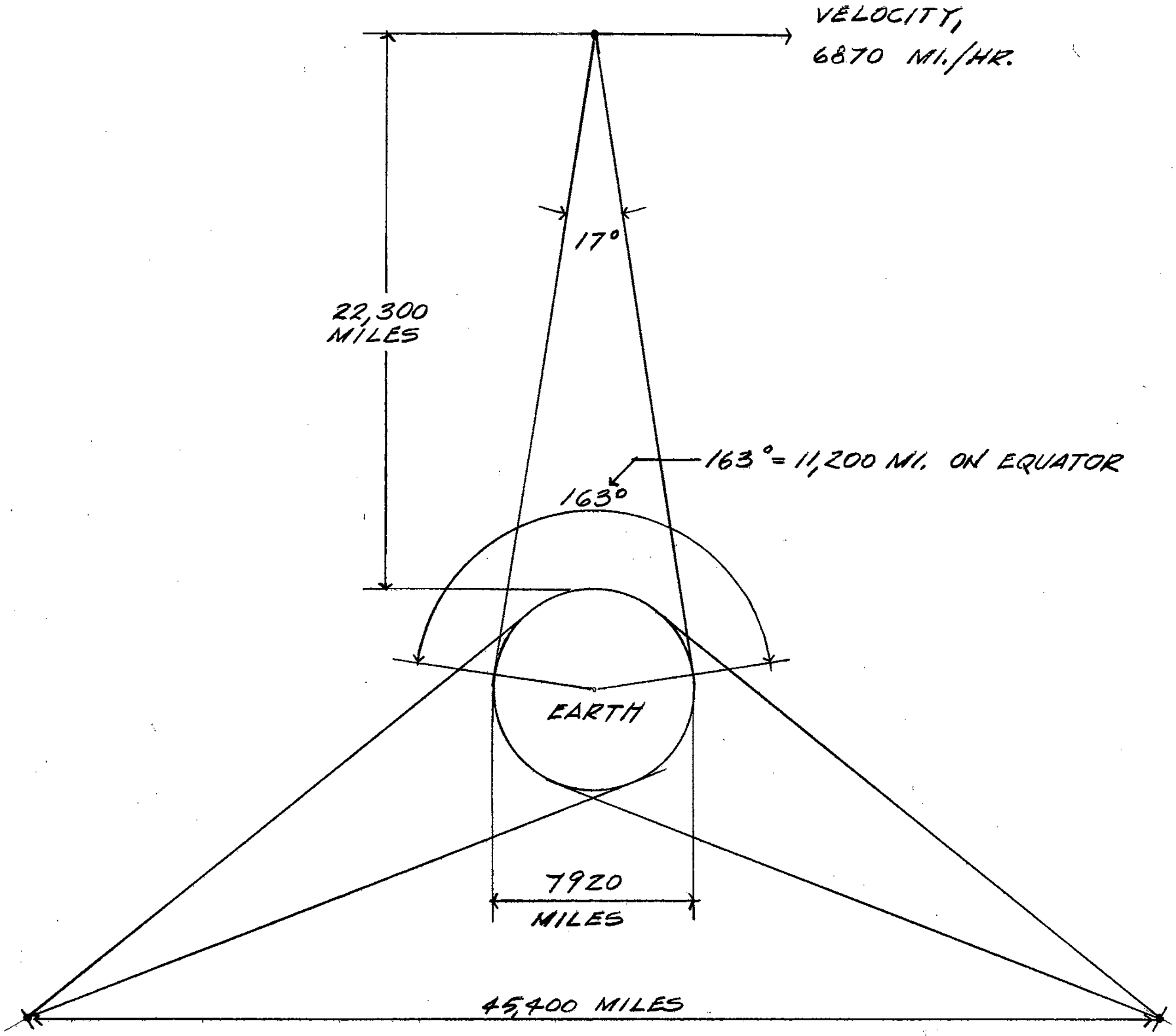


Fig. 13 Typical Synchronous Satellite Deployment

3. Attitude stabilization and station-keeping capability would be required.

4. Higher power will be required because of the long distance involved.

CHAPTER IV

DETERMINATION OF COMMUNICATIONS SATELLITE POPULATION FOR NON-SYNCHRONOUS RANDOM ORBIT SATELLITE SYSTEMS

In planning the number of communications satellites which will be required to provide a desired reliability of communications between two separated earth based stations a certain distance apart, the probability of communications between the two stations must be calculated for various numbers of satellites. The author of this paper has previously discussed the desirability of using uniformly spaced satellites within orbits which are inclined in a manner which will best serve the areas on the surface of the earth of greatest interest.[1]

In the above referenced document, the earth's area and distance coverage of a single satellite is a function of the altitude of the satellite. The earth's mean radius being about 6370 kilometers, any satellite above two earth radii would see considerably more than one third of the earth's circumference at any one point in time. The earth's circumference is about 40,000 kilometers. This means that any satellite

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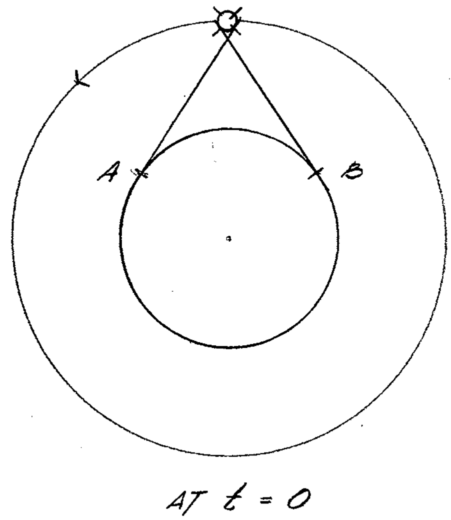
[1]USAF document "An Argument for Equally Spaced Medium Altitude Communications Satellites," by Lt. Col. Charles C. Mack.

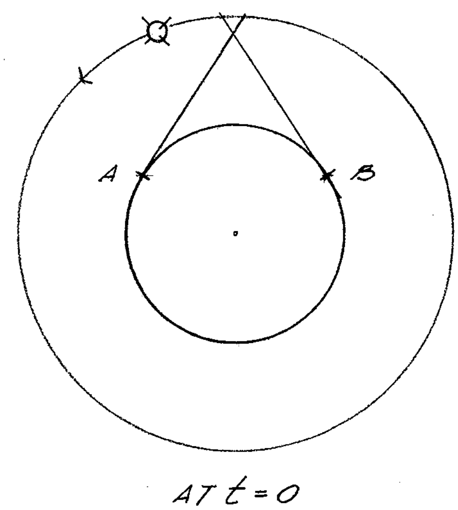
in a low inclination orbit will give considerable area coverage both north and south of the equator. In fact, there will be times when polar coverage will be furnished. Obviously the time of mutual visibility between two ground stations on the surface of the earth is a function of the distance between those stations which are using the satellite as a radio relay. For example, consider the two extremes assuming that the communications satellite passes directly over the two ground stations. One extreme is when the two ground stations are located a distance apart which approximates the distance which the satellite sees over the surface of the earth. The other extreme is when the two ground stations are located immediately adjacent to each other and approximating zero distance apart.

In the first extreme example, where the ground stations are located about as far apart as the satellite seeing distance, the ground stations would have the satellite within mutual visibility only for an instant before the communications satellite passes from the view of one of the ground stations. (See Fig. 14.) In the figure, stations A and B at a particular point in time (t = 0) have the communications satellite within mutual visibility. A moment later (t = 0+) the satellite has passed out of the field of view of one of the ground stations which in this case is station B and has moved on well within the field of view of station A.

In the other extreme, when the two ground stations are located adjacent to each other, both stations will have mutual visibility of the communications satellite as long as the communications satellite is



a.



b.

Fig. 14 Mutual Visibility of a Communications Satellite when the Ground Stations are Located a Maximum Distance Apart

within the field of view of either satellite. This mutual visibility time will be the time that the satellite has taken to travel the number of degrees of the earth's central angle which is represented by the ground coverage distance.

Since the earth's surface distance coverage is limited to one half of the circumference of the earth when the satellite is at infinite altitude and over one third of the circumference of the earth is covered at a satellite altitude of over two earth radii, it becomes obvious that the earth coverage distance increases very little as the altitude of the satellite is increased beyond about two radii. Increases in altitude will only dilute the received power on the surface of the earth and therefore require either greater power in the satellite or greatly complicate the ground stations. Increases in altitude will, however, have the advantage of increasing the orbital period with a corresponding reduction in the frequency of the satellite acquisitions and handovers by the ground stations.

Figure 14 illustrates the mutual visibility at ground stations with capability of zero look angles. Actually in a practical situation the minimum look angle would approximate about 10 degrees. Terrain features and atmospheric attenuation of the signal at low angles would prohibit zero look angles in most cases. Figure 15 illustrates a more practical situation where look angles above zero are used.

Keeping the foregoing discussion in mind and realizing that the sub-satellite area of mutual visibility shown in Fig. 16 represents the mutual visibility area in terms of the earth's surface, the probability of two ground stations seeing the same satellite is:

$$p = \frac{\text{Area of mutual visibility}}{\text{Area of the earth}} \tag{7}$$

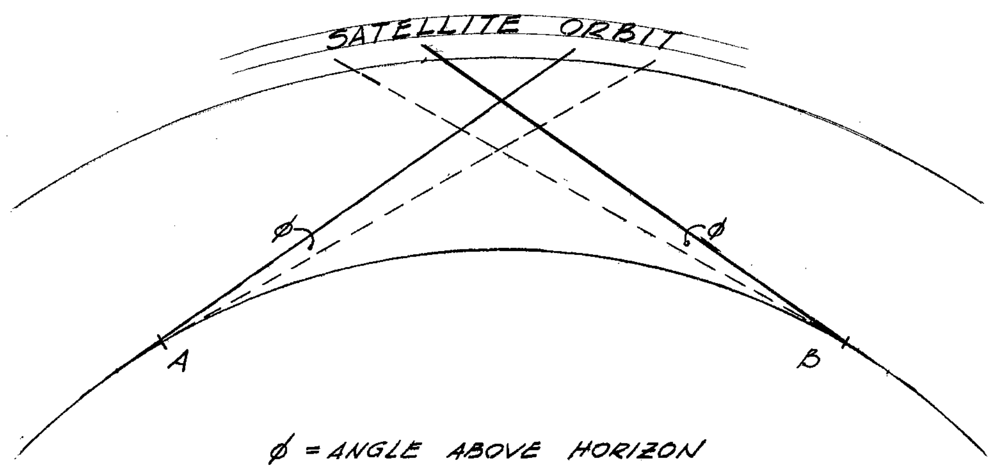


Fig. 15 Satellite Mutual Visibility Area where Look Angles are Above Ten Degrees

and therefore:

$(1 - p)$ = The probability of the same two ground stations not seeing the satellite. (8)

Assuming n satellites in random distribution at a particular altitude, then:

$(1 - p)^n$ = Probability of the ground stations not seeing n satellites. (9)

And from the above:

$1 - (1 - p)^n$ = Probability of the two ground stations seeing a minimum of one out of n satellites. (10)

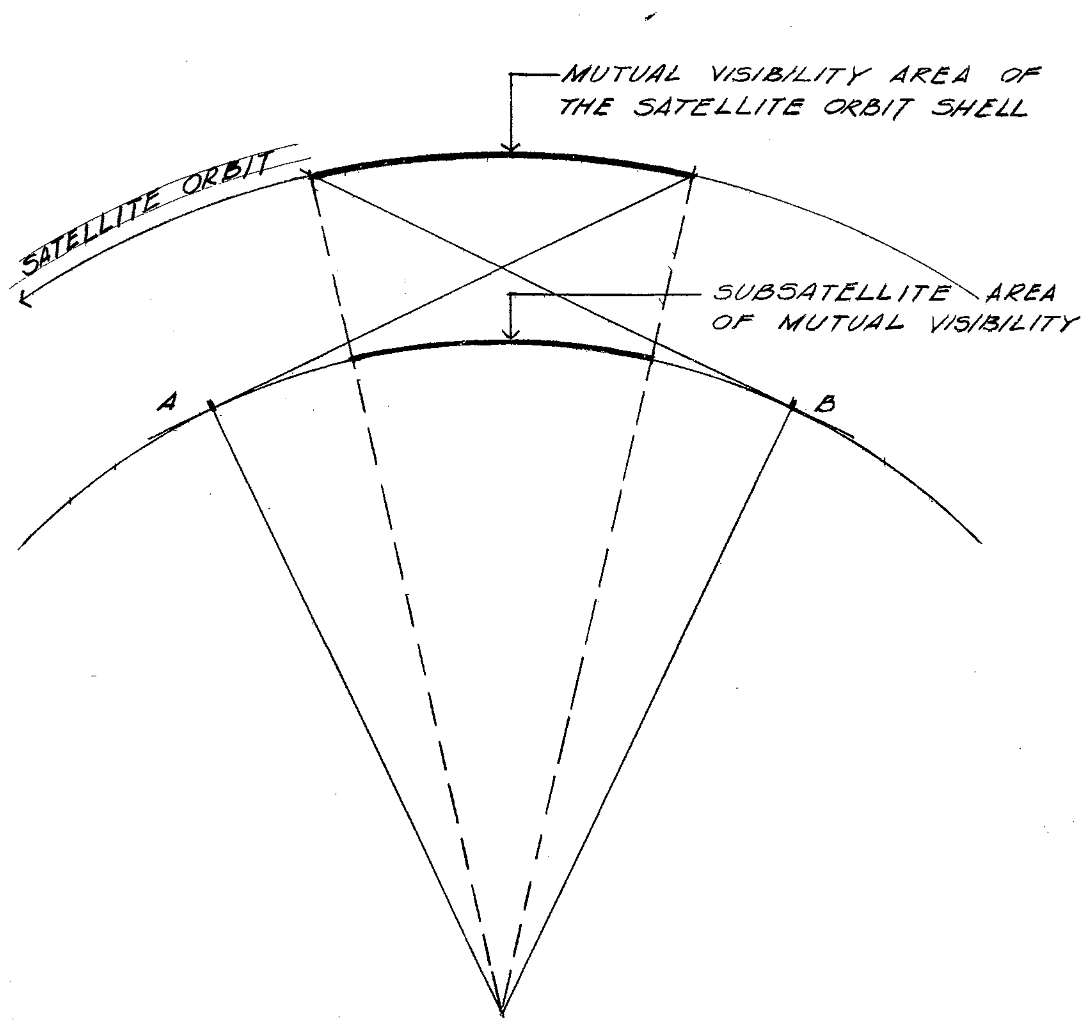


Fig. 16 Subsatellite Area of Mutual Visibility

Using the preceding equation the probability of two selected ground stations being able to communicate with each other can be determined when the number of satellites in random distribution and the mutual visibility area are known. The mutual visibility area is a function of the distance between two ground stations and the altitude of the satellites.

The main problem now is to develop a means for calculation of the overlapping cones of coverage area which is shown on Fig. 16. There appears to be a choice of two approaches to the calculation of the overlapping areas of coverage. Looking outward from the surface of the earth toward the celestial sphere one can imagine a surface in space which would be represented by the orbits of the communications satellites rotating about the earth in a random manner all of which are approximately at the same altitude above the surface of the earth. Two ground stations located a definite distance apart on the surface of the earth are able to scan from horizon to horizon toward the surface previously mentioned. The coverage on this surface from the ground stations overlap and the amount of overlap depends on the distance between the ground stations and the altitude of the satellite orbit. Figure 17 shows a drawing of a spherical shell which would be developed or represented by many periods of revolution from a number of randomly distributed satellites at approximately the same altitude above the earth. Equation 10 may be solved either in terms of the earth's surface or in terms of the surface represented by the satellite orbits. In any case the results will be identical. For purposes of this discussion it might be easier to solve the equation in terms of the satellite orbital shell as shown in Fig. 17. In this figure the overlapping zone of mutual visibility is the area C, D, E, and F. But since this zone of mutual visibility is formed by the intersection of small circles of the satellite shell, the area is difficult to calculate exactly. However, by dividing this

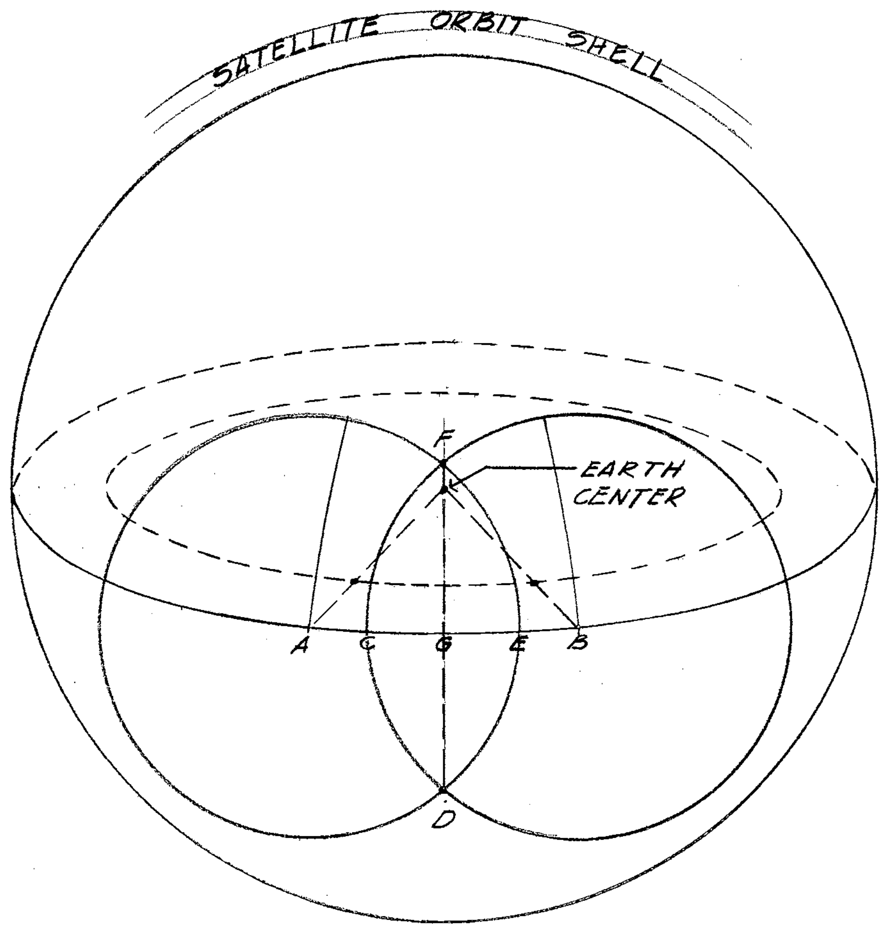


Fig. 17 Determination of the Overlapping Zone of Coverage of an Orbital Shell

area into four equal areas through the use of great circles between FD and AB and then adding a great circle between D and E, a spherical triangle is obtained which will permit a close approximation of the area to be calculated.

In order to calculate the area of the spherical triangle, first the semi-excess of the spherical triangle is determined in the usual manner. Since the area of a spherical triangle is equal to the number of right angles contained in the semi-excess multiplied by the area of a great circle, the area being found by dividing the semi-excess by 90 degrees and multiplying the result by πR^2. By multiplying the result thus obtained by four, the overlapping zone of coverage from the two ground stations is determined.

Using the computer, the series of calculations were made to determine the number of communications satellites which would be required for various orbital altitudes to obtain a probability of 99.99 percent when the earth's central angle distance between ground stations is 40 degrees and the look angle limit is 10 degrees to the horizontal. See Fig. 18 for the results.

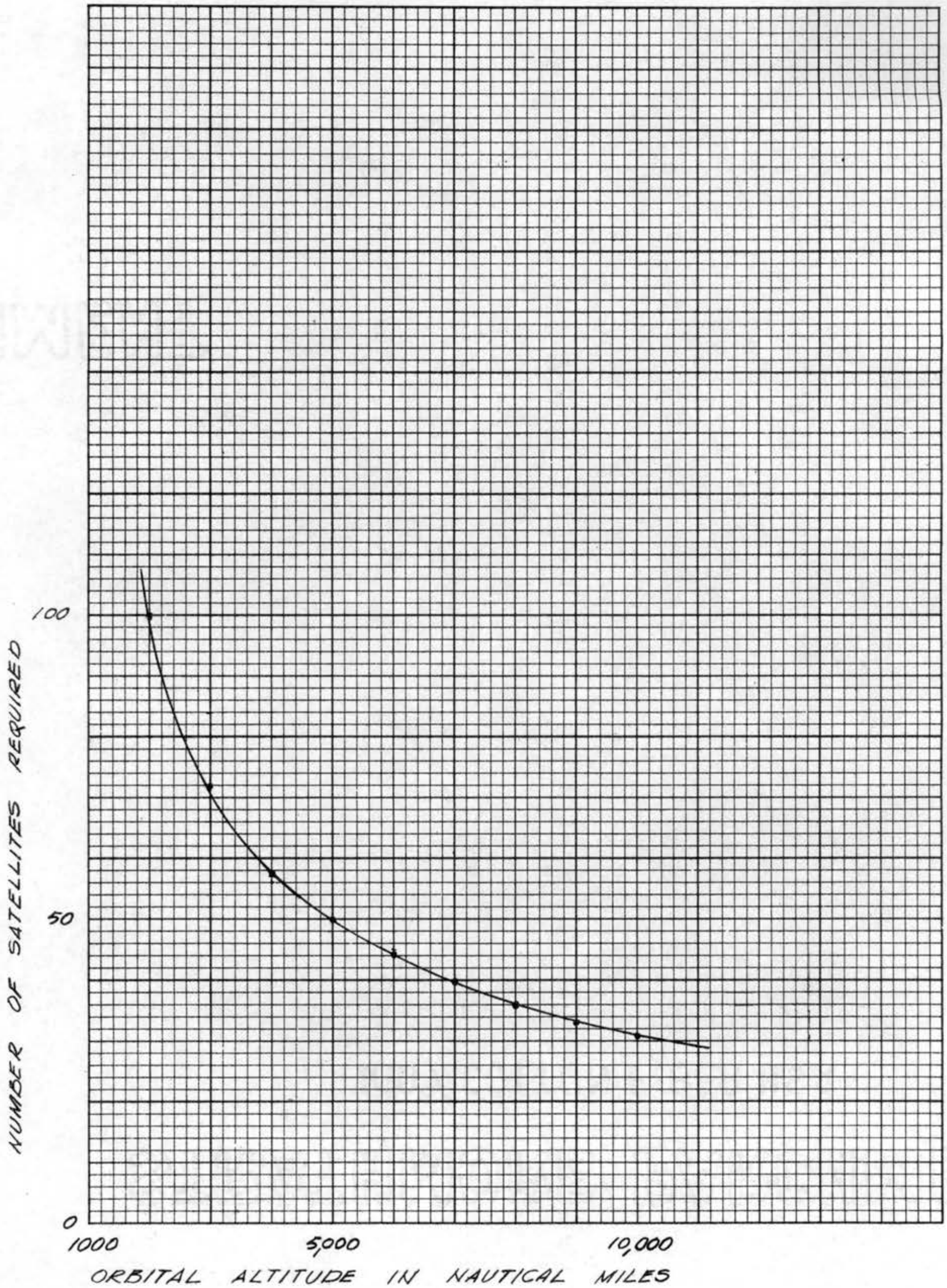


Fig. 18 Satellite Population as a Function of Altitude for Ground Station

CHAPTER V

PROPULSION SYSTEMS AND THEIR CAPABILITIES

In order to properly plan a communications satellite system, it appears that a knowledge of the current and near term future capability is necessary. At this time the propulsion capability is the major limiting factor in determining what kind of communications satellite system is possible.

Because of the economic limits and the cost of the development of new types of propulsion vehicles, a major effort is being instituted to standardize as much as possible. The current capability and the near term capability can therefore be categorized. These categories can be divided into separate groupings by the number of stages that might be employed, the type of propellant used, the size of the load which might be carried and the possible schedule of development. Table IV describes the several general types of vehicles which are in existence or may be available within the near future.

From the information in Table IV, it appears that operational systems can be placed into satisfactory orbits only with the last three categories of launch vehicles. Synchronous systems with usable band-width will probably utilize the Saturn type vehicles. For very broad band systems capable of relaying television bandwidths, the advanced type Saturn and even more capable and further advanced vehicles would seem to be desirable.

TABLE IV

CURRENT AND FUTURE ROCKET VEHICLE CAPABILITIES

Class		First Flight	Take-Off Wt. lbs.	No. of Stages	Payload in 300 Mi. Orbit lbs.
I	Scout	1960	30,000	3-4	200
II	IRBM Type	1958	120,000	2-3	500 - 1,000
III	ICBM Type	1959	240,000	2-4	2,000 - 10,000
IV	Saturn	1963	1-2.5 million	3-4	25,000 - 100,000
V	Saturn Advanced	1966	8-10 million	3-5	Above 300,000

There are several types of propulsion systems in current usage and there are others which are in the research category. These types include liquid and solid-propellant rockets, nuclear-fission rockets, radioactive-isotope decay devices, several different types of electrically powered rockets, together with devices using solar energy for propulsion and stored gas. The stored gas devices will probably be limited to use for attitude control of synchronous type satellites. Ion propulsion systems will probably be used for station keeping, attitude control and interplanetary flight. It might be well to list here the characteristics of the performance of the various types of propulsion systems.

TABLE V

PERFORMANCE CHARACTERISTICS OF
ROCKET ENGINE PERFORMANCE

Rocket Motor Type	Specific Impulse	Thrust to Wt. Ratio	Status
Standard Chemical Propellant	200 - 300	10^{-2}-10^{2}	Well Developed
High Energy Chemical	300 - 500	10^{-2}-10^{2}	Current Development
Fission	500 - 1,100	10^{-2}-10	Current Research
Electrical (Ion or Plasma)	5,000 - 25,000	10^{-5}-10^{-3}	Current Research
Solar Heating	400 - 700	10^{-3}-10^{-2}	

Intensive efforts on rocket motor development in the United States date back to only late 1957. There have been rather rapid strides since that date. It seems reasonable to assume that industry and research organizations were slower during their initial buildup and that therefore the developments in the future will be somewhat accelerated. By the end of the decade the improvements of rocket motor capability should be very gratifying to the planners and builders of space systems.

Liquid and solid propellant chemical rockets and fission propulsive devices are capable of high accelerations with high thrust to engine weight ratio, but they have a low specific impulse and therefore require a large amount of fuel. The electrical type rockets, on the other hand, have a high specific impulse, but require heavy power machinery, are complex, convert only a small amount of their available energy into propulsive force and they give only slow accelerations. It appears that the real need toward improvement of the electrical type propulsion is for the development of a light-weight, high-level power source of high reliability. It appears that this requirement might be satisfied by the development of a light-weight nuclear-type power supply.

Although it appears that the higher power rocket propulsion systems would be the most desirable for communications satellite systems, the relative cost of the various type propulsion systems makes it necessary that the appropriate propulsion vehicle be selected for the particular operational system. For example, the cost for the vehicle and vehicle launch of the Atlas-Agena rocket combination is about eight million dollars. The probability of successful launch factor is .5 which means that twice as many vehicles and satellites must be developed and built as the number of satellites that will be required in orbit. As a comparison the author estimates that the cost of an advanced Saturn type rocket such as the Saturn C-5 will be somewhere between 30 and 50 million dollars per launch. It must be borne in mind that the foregoing costs include the cost of the actual launch which includes missile tracking, repair of the launch facilities after launch, preparation of the facilities and other such items. The probability of the successful launch of the advanced Saturn will initially be somewhat lower than that previously quoted for the Atlas-Agena combination.

In considering the economics of a communications satellite system, one must consider other parts of the system to arrive at an appropriate decision. For example, a high powered communications satellite in synchronous orbit may be hundreds of times as expensive as a simple light weight similar device. However, the high powered satellite may be capable of many times the bandwidth with a greatly increased channel capability, including television. The high power satellite may also be able to communicate with thousands of ground stations which can be kept relatively simple because of the greater received power and consequently simple and less expensive antennas can be employed. Such stations on the ground would be relatively inexpensive when compared with the complex antennas and stations which would be required for communications through very low power satellite relays. It appears then that it would be more desirable to use a high power satellite in orbit with the total bandwidth being supplied with equipments which individually cover only part of the band of operation. This will permit the satellite to continue to function on the remaining part of the band segment in the event one of the individual equipments fail. Operation of a high power satellite in this fashion will permit low power stations on the ground where the greatest number of stations in the system will be located, and thus reduce the overall cost of the system and possibly improve its operation at the same time.

In order to provide a feel for the costs of the ground stations, the author estimates that a sophisticated complex station for operation with synchronous very low power satellites might be upwards of one million dollars each. A simple ground station for operation with a high power communications satellite would probably cost less than $50,000. For example, with the satellite power increased 100 times, the antenna gain requirements of the receiving antenna could be reduced by 20 db. A very great reduction in size and cost would result. In addition, the operating staff requirements for the simple station would be greatly reduced.

CHAPTER VI

POWER SOURCES

Power sources are obviously necessary for all communications satellites. In the discussion of power sources, consideration will be given only to basic power sources and power conversion devices. This type of information will be necessary for general over-all planning and understanding of the problem. Detailed design information such as power sequencing operations switching, selection of switching devices and other such information will not be included.

Environmental Considerations

High-energy radiation, meteorites, lack of gravity, and heat transfer appear to be the main considerations. Severe mechanical shocks, acceleration and duration of operation are also factors which can be considered under the environment problems even though strictly speaking they are not environmental factors. The space borne power systems must withstand greater acceleration forces for part of the time than their counterparts on the surface of the earth. They must also withstand greater mechanical shocks for a part of the flight and must operate for a long time without attendance. Separation of liquid and gas in a zero acceleration environment can be a problem and high energy particle radiation may affect the semi-conductor devices aboard the spacecraft. Meteoroids can cause erosion and can even penetrate the spacecraft itself. Much work has been and is now being done on the study of the micrometeorites and their effects on spacecraft. Dr. F. C. Todd of the Oklahoma State University, as well as others, is now undertaking extensive research under government contract on these problems. Even though the equivalent heat sink temperature of deep space is only 3.5 degrees K, the waste heat radiator may be the heaviest part of the system and severe heat transfer problems will probably be encountered especially where high-power nuclear power sources are used. Ozone and dissociated gases can have deleterious effects up to nearly 100 miles but since the communications satellite vehicle will only briefly pass through this region, no real problem from these sources is expected. Following, in Table VI, are the environments expected, their extent and possible deleterious effects.

Chemical Power Units

Chemically powered turboalternator positive displacement units, and batteries including fuel cells fall into this category. In general, chemically powered sources such as the turboalternator displacement units are essentially energy sources and cannot serve as prime power sources for extended periods of time. They will not be further discussed in this thesis because they appear to be impractical for communications satellite use. The discussion will therefore center around batteries and fuel cells as far as the above category is concerned.

TABLE VI

POWER SYSTEM ENVIRONMENTAL EFFECTS

Environments	Conditions	Possible Deleterious Effect
Pressure	Less than 10^{-9} mb at 500 miles	Heat transfer by radiation; thermionic emission; requirement for hermetic sealing
Radiation - (Corpuscular and Electromagnetic)	X-rays, high energy protons and electrons, cosmic rays	Damage to solid state devices, sputtering
Gravity	Weightlessness	Difficulty in separating liquid from vapor, no convection currents
Vibration and Acceleration	Up to 50g for short periods	Mechanical failure
Meteoroids	Uncertain spatial density; specific gravity may vary from 5 to 5×10^{-2}	Abrasion and/or puncture

Silver-zinc cells can deliver, under ideal conditions, as high as 80 watt-hr/lb, and about 2 watt-hr/in.^{3}; silver-cadmium batteries about 30 watt-hr/lb and a similar specific volume. Hydrogen-oxygen fuel cells can deliver about 200 to 300 watt-hr/lb including storage of the feed gases (or about 1 kw-hr/lb of feed gases alone). The fuel cell itself, operating at 240 degrees centigrade and 800 psi, exclusive of the feed gases and their containers, could deliver about 10 kw/ft^3 and 55 watts per lb. However two factors should be considered with regard to batteries. First, the nominal watt-hour per lb figures are for specific drain rates and temperatures. Figures 19 and 20 show the performance of typical zinc-silver oxide and nickel-cadmium batteries as a function of temperature and drain rate. High drain rate and/or low temperature adversely affect specific energy and in addition the regulation becomes poorer. For accurate information regarding the particular batteries, the planner or user should check with the specific manufacturer of the unit. The second point regarding batteries is that there is a definite limit to the performance of electrochemical systems. For example, for the hydrogen-oxygen fuel cell, it is approximately 1700 watt-hr/lb of feed gases.

When using batteries as secondary power sources, methods of supplying primary power for recharging are necessary. Table VII indicates the secondary battery characteristics of general type batteries which are readily available. These characteristics can be used for planning purposes.

The hydrogen-oxygen fuel cell appears to have great potential and there appears to be considerable research currently in progress to exploit their potential and improve their performance. The author is particularly aware that Dr. W. L. Hughes and Professor C. M. Summers of Oklahoma State University are currently conducting

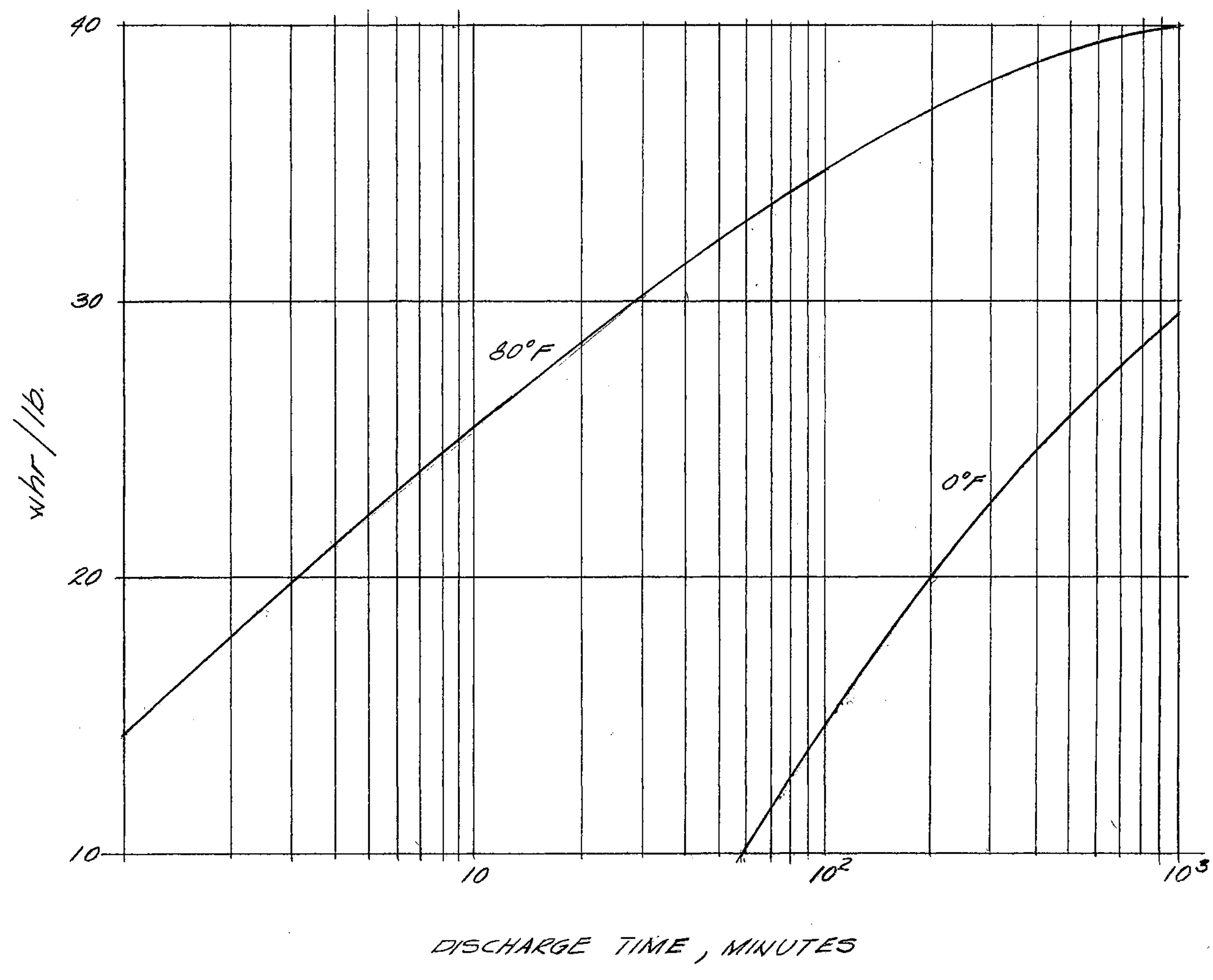


Fig. 19 Zinc-Silver Cell Performance Characteristics

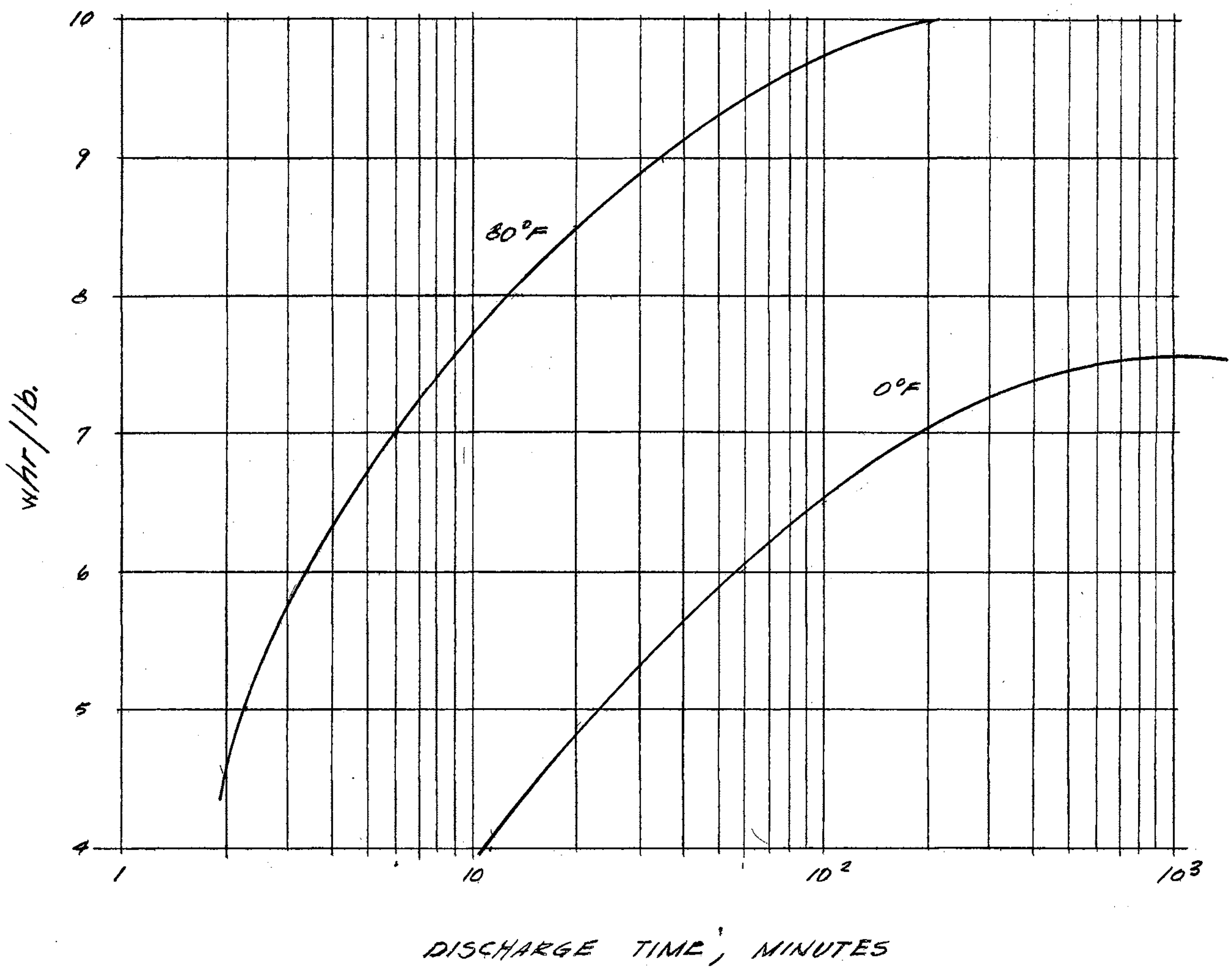


Fig. 20 Sealed Nickel-Cadmium Cell
Performance Characteristics

TABLE VII

SECONDARY BATTERY PERFORMANCE

				Light drain under good conditions				
Type	Anode	Electro-lyte	Cathode	Voltage/ Cell	Watt-hr/lb	Watt-hr/in^3	Cycle Life	Charge Retention - %
Lead Acid	Pb	H_2SO_4	$Pb0_2$	2	20	2	Few Hundred	6-30 drop/mo
Nickel-Cadmium	Cd	KOH	$Ni(0H)_2$	1.2	12	0.9	Several Thousand	20-40 drop/ year
Zinc-Silver	Zn	KOH	Ag0	1.5	40-60	3.3	Few Hundred	20-40 drop/ year
Cadmium-Silver	Cd	KOH	Ag0	1.1	30	3	Several Thousand	

research on hydrogen oxygen fuel cells.

Solar Powered Sources

As of the time of this writing, the only solar powered satellites used were powered with silicon solar cells. Under optimum conditions nearly all of the power developed by the cell will be available to the external load. This is in contradistinction to the usual power generator where only half of the power appears externally. With the currently used solar cell, about .48 volts at about 2.0×10^{-2} amps are produced at full sunlight with a cell area of 2 by $\frac{1}{2}$ cm. Larger cell areas are generally less efficient because of higher internal resistance. At room temperature the efficiency of the commonly used cells is about 10 to 12 percent. Efficiency of the solar cells falls off rather evenly with increase in temperature. A 200° F. increase drops the efficiency by about one half. Cost of the solar cells is now about 100 dollars per watt.

High energy particles tend to damage the solar cells and decrease their output as time goes on. More damage will obviously result to the cells when the satellite traverses the radiation belts with greater damage per unit of time when the satellite is in equatorial orbit than for the polar orbit. Table VIII shows the estimated times before the output decreases by 25 percent for various orbits.

TABLE VIII

SOLAR CELL ENVIRONMENTAL DAMAGE

Solar Cell Damage Time for 25 Percent Decrease in Output	
2,000 mile equatorial orbit	3 months
2,000 mile polar orbit	18 months
1,000 mile polar orbit	10 years

Nuclear Power Systems

The current nuclear reactors appear to be isotope-powered units and reactor-powered units. Isotopes appear to have the advantage of containing no critical mass and little external radiation hazard for selected isotopes.

There are two types of isotope-powered supplies -- those occurring from the fissioning of the nuclear fuel and those which must produce by pile irradiation of a nonfissionable material. Polonium 210, which is an artificially produced alpha emitter, has the advantage of producing very little external radiation hazard, while at the same time furnishing a high specific power. Table IX provides some information which might be used for planning and development purposes.

TABLE IX

CHARACTERISTICS OF SOME ARTIFICIAL ISOTOPES

Isotope	Fuel Form	Specific Watts/cm	Power Watts/g	Half life	Total radiation (γ + Neutrons) 1 yd. from a 100 watt thermal source, rem/hr
Po^{210}	Po	90	9.6	138 days	2.6×10^{-2}
Cm^{244}	Cm	20	2.9	18 years	1.3×10^{-2}
Pu^{238}	PuC	7	0.55	86 years	0.8×10^{-3}

Snap III-B, a prototype polonium powered thermocouple power source, has been built. This unit has the following characteristics:

4-3/4 inches diameter

5-1/2 inches high

4 lb. over-all

1700 curies using 0.4 g of Po^{210}

Output 3.3 watts to a 1.7 ohm load

Shell temperature 99 degrees centigrade.

There are a number of difficulties with isotope power supplies. Among them are: the usable isotopes are expensive and scarce; the isotopes are extremely dangerous to human beings if inhaled or ingested; and the problem of disposing of excess heat. Nevertheless these power supplies are extremely rugged with reasonably long life and are relatively insensitive to environmental radiation damage. If sufficient quantities of the needed isotopes can be obtained, these power supplies would be desirable for many uses.

Contamination hazard is also present with the use of reactor-type nuclear power supplies. The long-lived fission products which are produced by a reactor-type supply from a reactor which produces 100 Mw of heat over a year's time, would be equivalent to those released by a megaton atomic burst. For space flight, reactors must be compact to minimize shielding. The minimum size reactor is determined by criticality and therefore it seems that the smallest sizes would weigh somewhere between 70 and 100 lbs. It should be possible to obtain 1 to 10 thermal Mw per cubic foot with a mean reactor density of 200 lbs. per cubic foot. From the information available which does not have any military classification, it appears that high power nuclear power converters could be built within the next few years which could have the capability of .5 to several megawatts and weigh in the order of 2,000 to 3,000 lbs. Such power supplies could be used with high power communications satellites and deployed at some distance away from the main communications package to reduce the shielding requirements. Interconnecting cables or rods could be used to transfer the power to the communications package. Many problems relative to reactor power supplies must still be resolved. There will be heat transfer and conversion problems as well as others but the author (based on his research of the available information) feels that high power multi-kilowatt to megawatt range power supplies which will be suitable for space use can be available before the end of the decade. He feels that power supplies with limited shielding can be built to weigh less than 3,000 lbs. for electrical output power of 1 megawatt or less.

CHAPTER VII

SYSTEM LOSSES ASSOCIATED WITH PROPAGATION

In planning communications satellite systems the losses which will be introduced in the system must be calculated. The scope of this thesis does not permit a rigorous examination of the losses which might be expected in the large number of combinations of available equipment. The discussion in this particular chapter will be limited to the approach to the problem and methods for determining over-all attenuation associated with signal propagation.

Space Spreading Effect

Most of the losses in the system will be experienced by the so-called "space spreading effect". This is the continuing dilution of the power caused by the expanding area of the spherical wave as it propagates radially from the antenna. The line of sight energy, which is transmitted, moves outward from the transmitting antenna. This energy as it moves outward must be distributed over the expanding area with the power density per unit area decreasing with increasing distance. This can be expressed as:

$$P_r = \frac{P_t}{A} \quad . \tag{11}$$

P_r is the power received at some distance from the transmitting antenna.

P_t is the power radiated by the transmitting antenna.

A is the area of the spherical wave at the particular distance from the transmitting antenna.

Since the area of a sphere is $4\pi r^2$, the above equation becomes:

$$P_r = \frac{P_t}{4\pi r^2} \quad . \tag{12}$$

From the preceding equation it can then be seen that the actual power which is received by the receiving antenna system is the product of the effective area and the unit area power of the transmitted wave when it arrives in the region of the receiving antenna. Assuming an isotropic source, the received signal now becomes:

$$P_r = \frac{P_t}{4\pi r^2} \text{ (the effective area of the receiving antenna) .} \tag{13}$$

With antenna gain of G_t at the transmitter, the resulting equation now becomes:

$$P_r = \frac{P_t G_t A_r}{4\pi r^2} \tag{14}$$

where A_r is the effective area of the receiving antenna.

To assist in planning systems, the line of sight ranges for three orbits from the zenith to the horizontal are shown in Fig. 21; the free space transmission loss as related to frequency and distance is shown on the normagram of Fig. 22; and the free space losses as related to

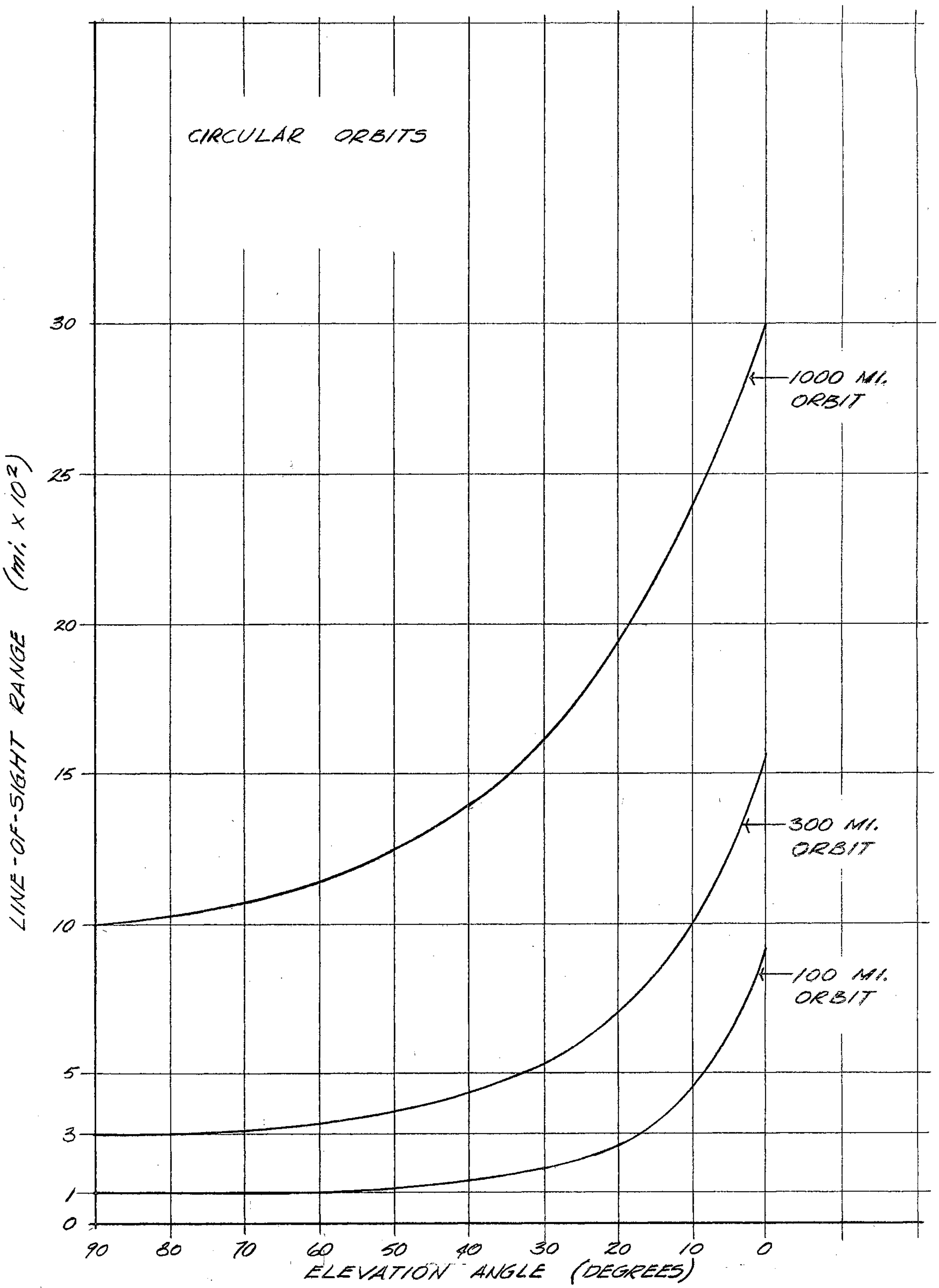


Fig. 21 Line of Sight Range vs Elevation Angle

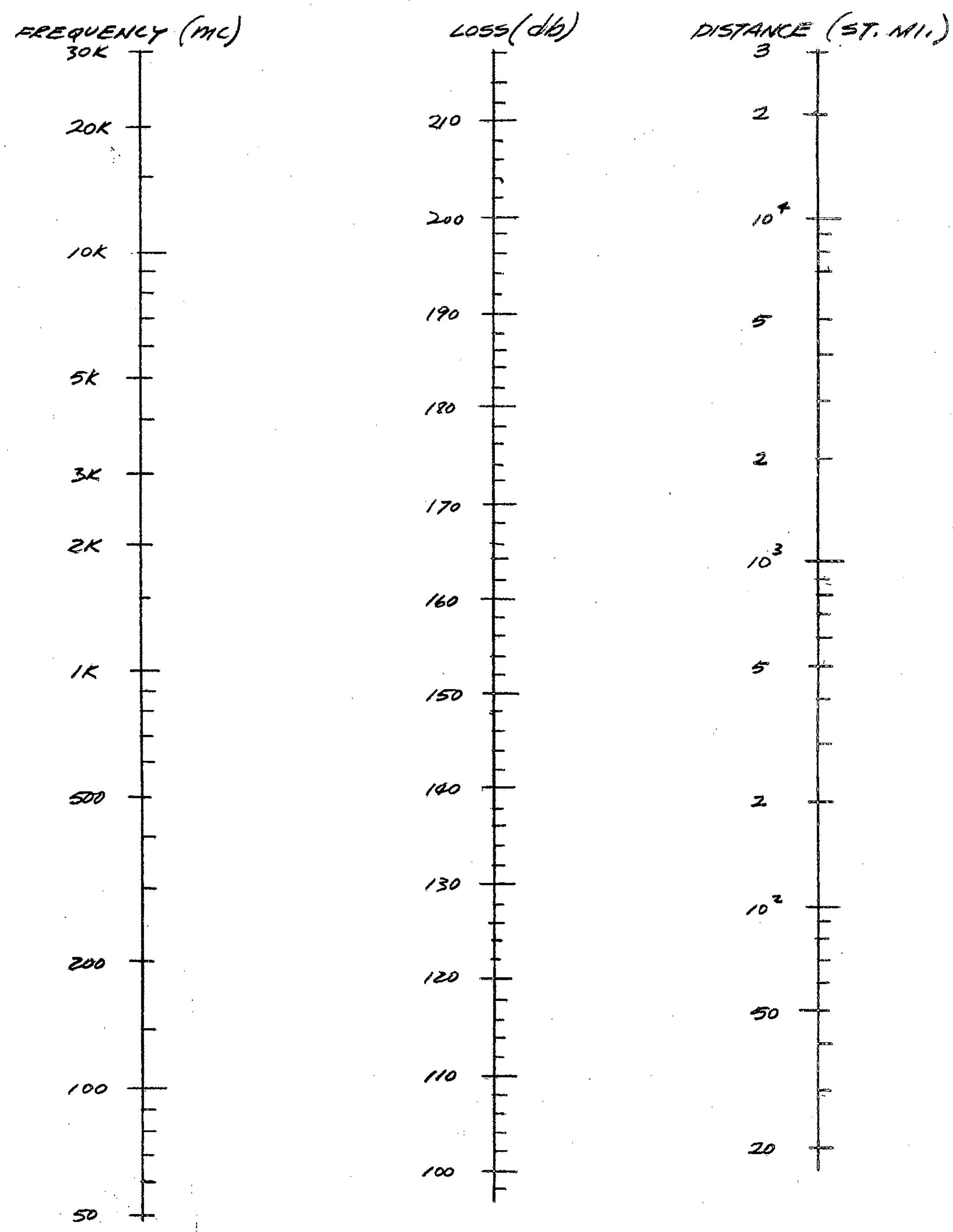


Fig. 22 Free Space Transmission Attenuation (Isotropic Antennas)

frequency are plotted for three representative orbits on Fig. 23. The information presented is for isotropic antennas. This will permit ease of system planning since the antenna gains in db can simply be added appropriately.

Ground Station to Satellite Range

To calculate the losses in the system which are caused by the space spreading effect, the maximum distance between the ground station and the satellite for the lowest antenna elevation angle to be used must be calculated. Assuming zero angle for the elevation of the ground antenna, Fig. 24 shows the method of arriving at the calculations. For the right triangle shown:

$$(R + h)^2 = R^2 + S^2$$

which reduces to

$$S = \sqrt{h(2R + h)} \qquad (15)$$

where S is the maximum slant range between the satellite and the ground station.

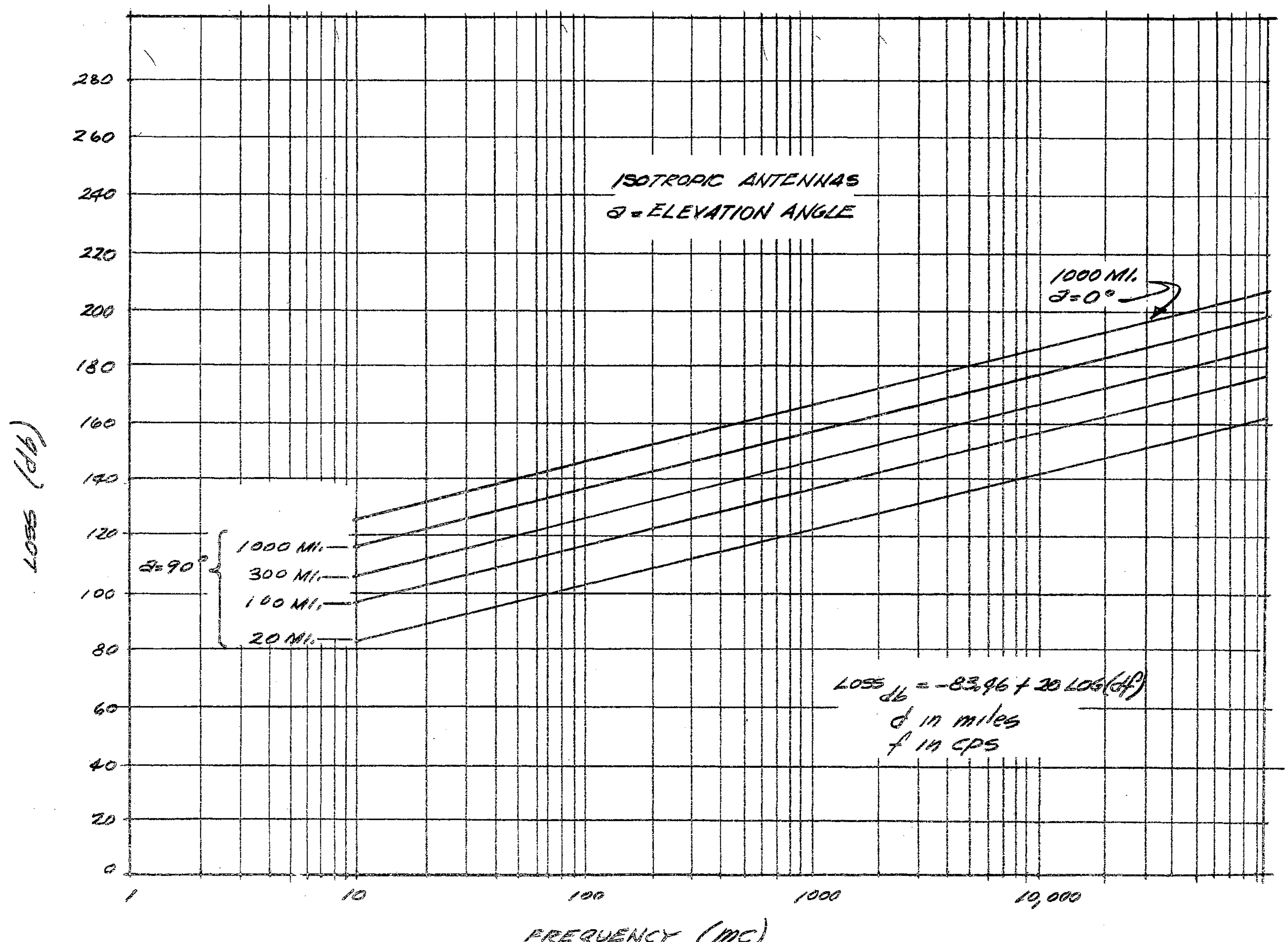
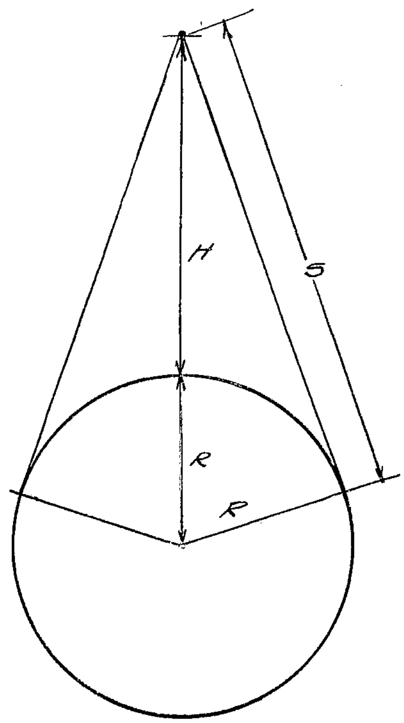


Fig. 23 The Attenuation Effects of Free Space vs Frequency



$$S = \sqrt{H(2R+H)}$$

Fig. 24 Calculation of Maximum Slant Range

Atmospheric Attenuation

Certain components of the atmosphere attenuate the radio frequency signal as it passes through it. The degree of attenuation varies with the frequency and the angle of the line of communication with the horizontal. It should be immediately obvious that more atmosphere is intervening at lower angles than at the higher angles. At 23,000 Mc/sec the major contributor to attenuation is water vapor and at 60,000 Mc/sec it is oxygen which produces the greatest attenuation. The one way attenuation of the atmosphere as a function of frequency is shown in Fig. 25.[1] Note that above 10,000 Mc/sec the attenuation due to atmospheric effects rises sharply. From the information thus available, it appears that operating frequencies for relay by communications satellites should probably be limited to below this figure.

Ionospheric Attenuation

When an electromagnetic wave passes through an ionized medium, losses due to energy absorption occur. These losses result from the electrons being accelerated by the electromagnetic fields as the wave passes through the ionized region. The net result is that there is a transfer of energy to the ionized medium from the radio wave. It can be seen that this transfer of energy will be dependent on the electron density (N), distance through which the electromagnetic wave must pass where the ionized region is involved, and the collision frequency ($\frac{\nu}{2\pi}$). When the frequency (f) of the propagating wave is much greater than the electron collision frequency and the earth's magnetic field effects are neglected, the attenuation (A) can be expressed as follows:

$$A = \frac{e^2}{2\pi c m f^2} \int_{S_1}^{S_2} N\nu \, ds \qquad (16)$$

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[1] Koelle, H., Handbook of Astronautical Engineering, McGraw-Hill Book Company, 1961, 1st ed., pp. 16-18.

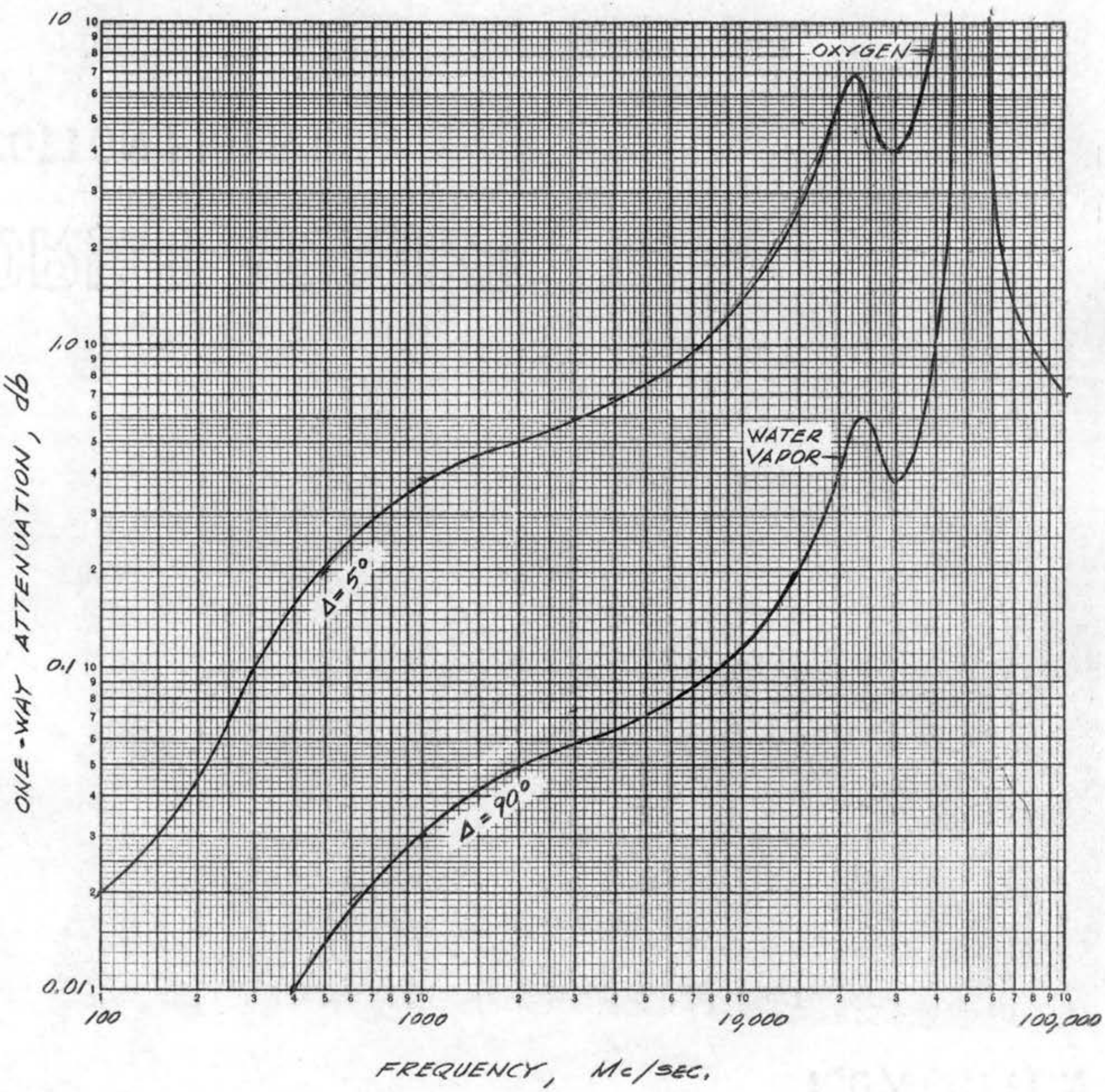

Fig. 25 Atmospheric Absorption vs Frequency, Standard Atmosphere

where e = charge of electron

m = mass of an electron

c = velocity of light

S_1, S_2 = path limits

A = loss in db .

The previous expression indicates that the attenuation decreases quite rapidly with frequency. Limited experimental measurements have confirmed the validity of the method of calculation. Figure 26 shows the absorption in db for 10, 20, 40, and 60 Mc/sec signals in passing through the ionosphere at angles from the horizontal to the zenith.

From the figure it appears that the effects of the ionospheric absorption will be negligible at the frequencies which will probably be used for communications satellite purposes. Considering the atmospheric attenuation previously discussed, the ionospheric absorption and the noise problems which will be discussed in a later chapter, it appears that a radio window exists which will permit communications via communications satellite. This window extends from above 60 megacycles to below 10,000 megacycles, approximately.

Faraday Rotation

An electromagnetic wave, when passing through a magneto-ionic medium such as the earth's atmosphere, splits up into two elliptically polarized waves which travel different paths at slightly different

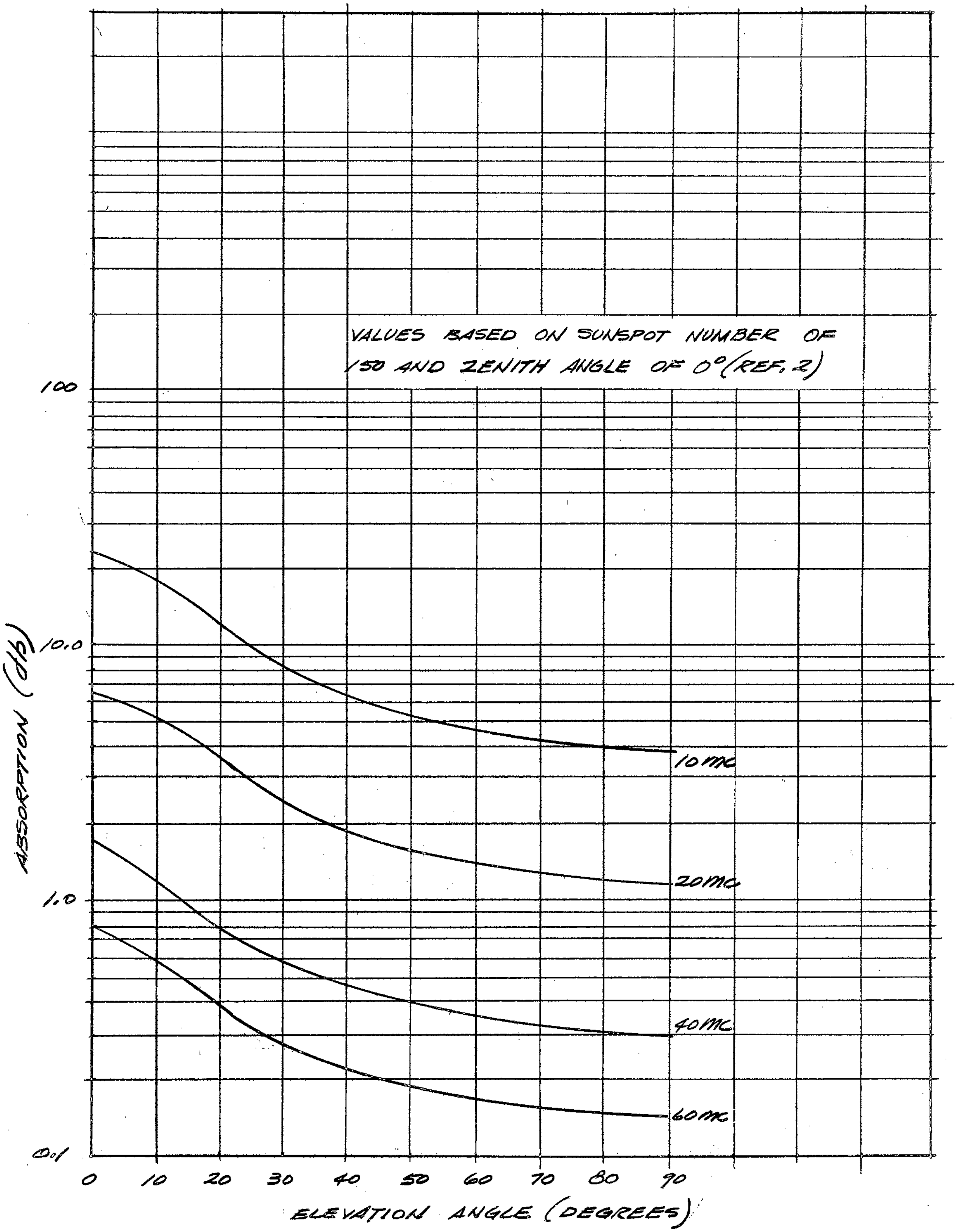

Fig. 26 Ionospheric Absorption vs Elevation Angle for Several Frequencies

velocities with slightly different indices of refraction. At the exit side of the magneto-ionic medium, the two waves will recombine so that the initially linearly polarized wave is now rotated so that its plane of polarization is then at an angle $\emptyset$ with the original wave.

The expression which gives the angle of rotation is:

$$\emptyset = \frac{e^3}{2\pi m^2 c^2 f^2} H \cos\theta \sec\delta \int_{h_1}^{h_2} N\, dh$$

where

e	=	charge of an electron
m	=	mass of the electron
c	=	velocity of light
f	=	propagating frequency
H	=	magnitude of the earth's magnetic field
θ	=	angle between H and the direction of propagation
δ	=	angle between the direction of propagation and the zenith
N	=	number of electrons per cubic cm between h_1 and h_2.

From available fragments of data and extrapolation of the moon radar data at 120 megacycles and by using the $1/f_2$ dependence, curves were plotted and are shown on Fig. 27. The curves shown on the graph represent the upper and lower averages of Faraday rotation.

With the receiving antenna at the same polarization of the initially propagated wave, there will be a decoupling effect which has been determined by others to follow the relationship:

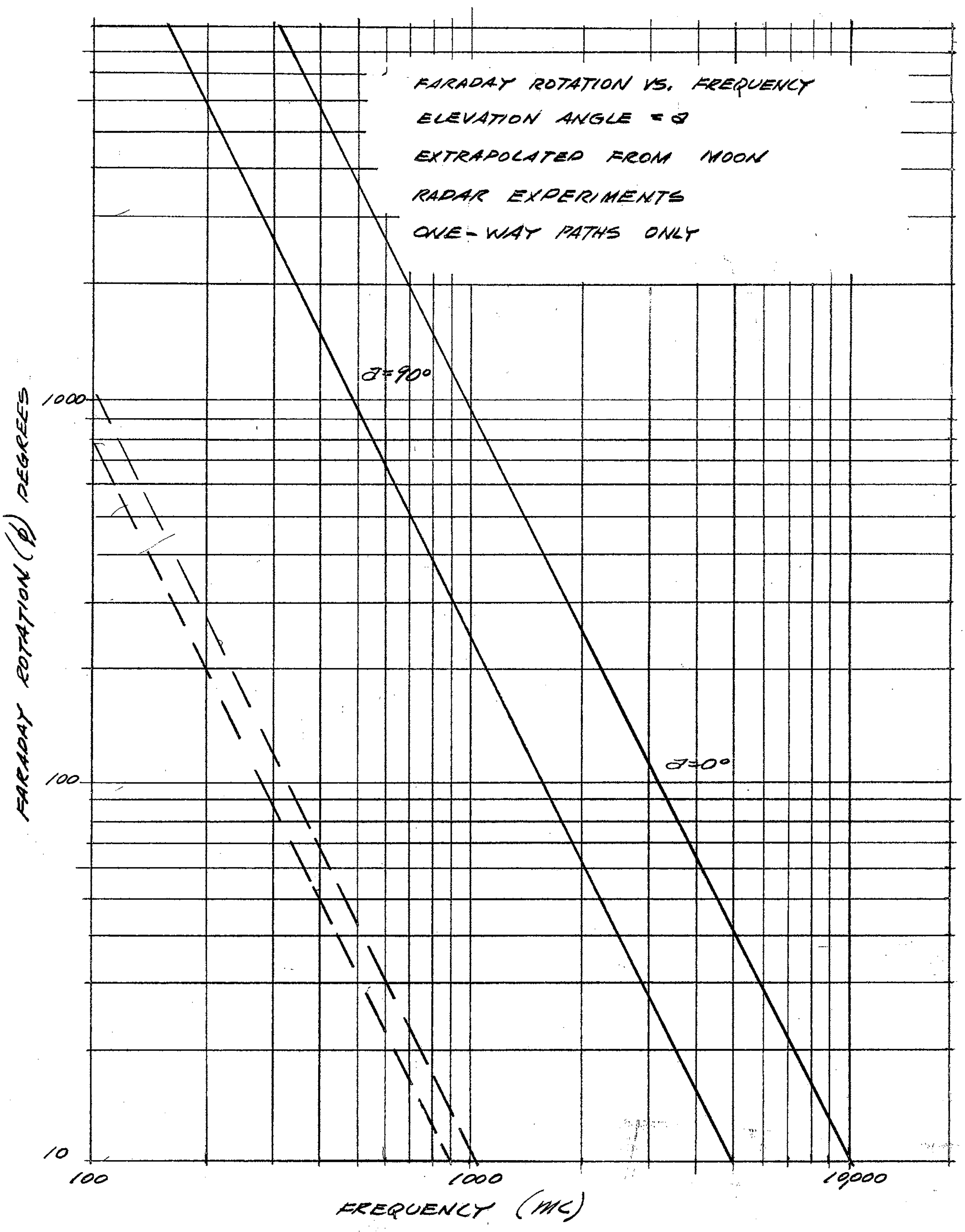

Fig. 27 Faraday Rotation

$$\text{Attenuation} = -20 \log \cos \phi \text{ in db} .$$

From this, it can be seen that 90 degree shifts will cause an attenuation of the signal of about 20 db. As the space vehicle is continually in motion with respect to a point on the surface of the earth (except for synchronous satellites) the values of θ, δ, H and N are changing continually and consequently causing changes in the degree of Faraday rotation. To minimize the effects of the Faraday rotation circular polarization should be used. The loss of a circular polarized antenna over linear polarized antennas with maximum coupling is about 3 db .

Other Attenuation Associated with Ionization

There are other sources of attenuation of the propagated wave. These are the high radiation belts, high altitude nuclear bursts, high sunspot activity and emissions from the sun of highly ionized plasma. Of particular interest are those sources of attenuation which result from nuclear explosions. It is known that the high altitude nuclear tests have increased the trapped radiation in the radiation belts and it is also believed by many that the difficulties experienced with TELSTAR and other satellites after the nuclear tests were caused by the high altitude explosions.

Sum Total of the Attenuation Effects

The attenuation losses due to atmospheric losses and the attenuation by free space and the ionosphere are all plotted on the graph of Fig. 28. By adding the 3 db loss from circular polarized antennas this effect could be included. Notice that there is a "window" which exists between frequencies of about 60 megacycles and 10, 000 megacycles. This chart should serve as one of the keys to communications satellite planning or other space-earth communication links.

Signal Strengths Required

Based on the preceding attenuation information and making allowances for fading factors which might be encountered, the typical signals which might be expected for various power levels and orbits are shown graphically for 100 and 1, 000 mile orbits in Fig's. 29 and 30. The information plotted was based on the following assumptions:

1. Isotropic radiators and receiving antennas.
2. Use of circular polarization.
3. Atmospheric losses of about 3 db at low angles.
4. Negligible ionospheric losses.
5. Fading allowances as follows:

Orbit Altitude (miles)	Elevation Angle (degrees)	Fading Allowance db
20	90	0
	0	15
100	90	10
	0	25
300	90	15
	0	30
1000	90	15
	0	30

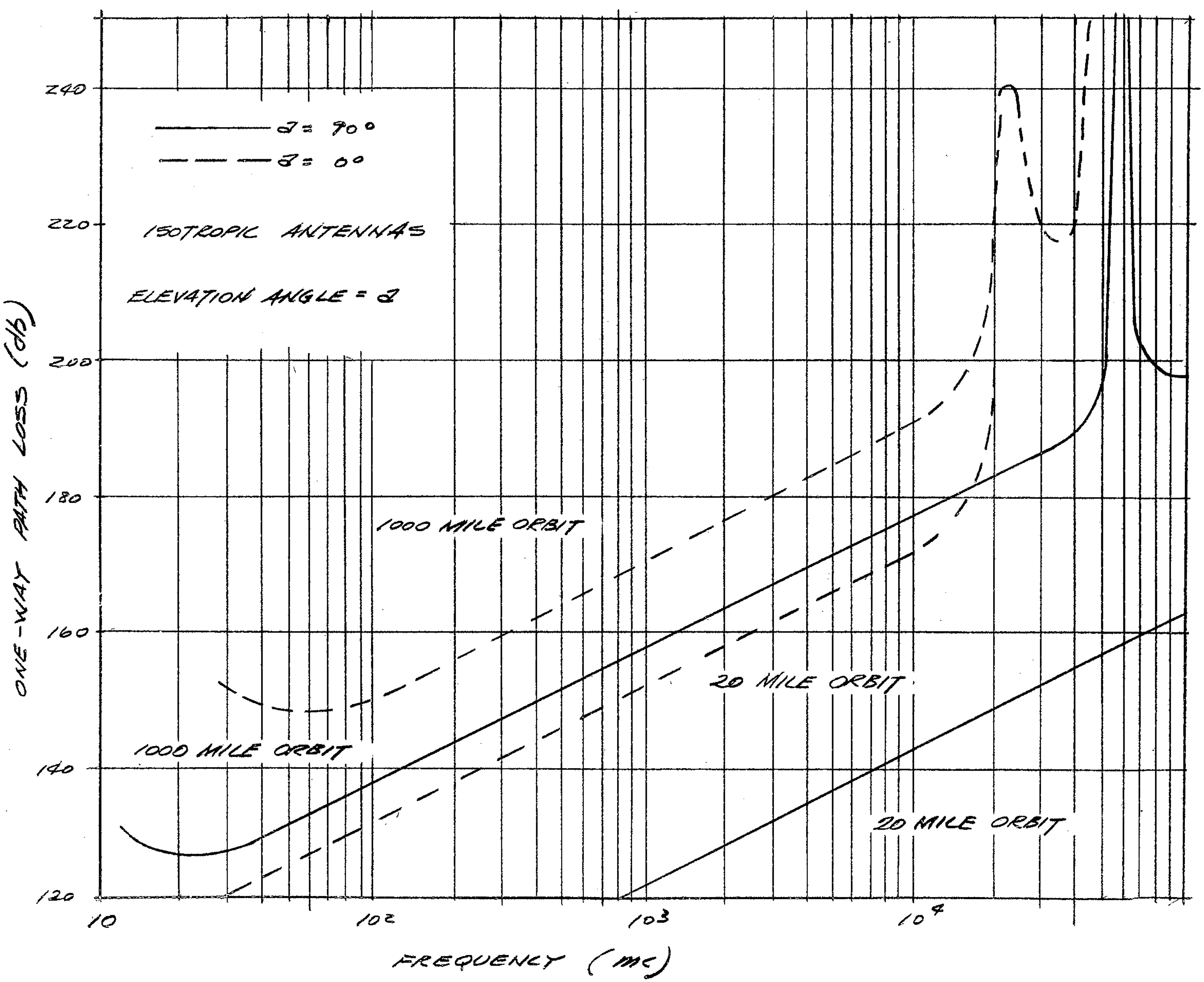

Fig. 28 Total Attenuation vs Frequency

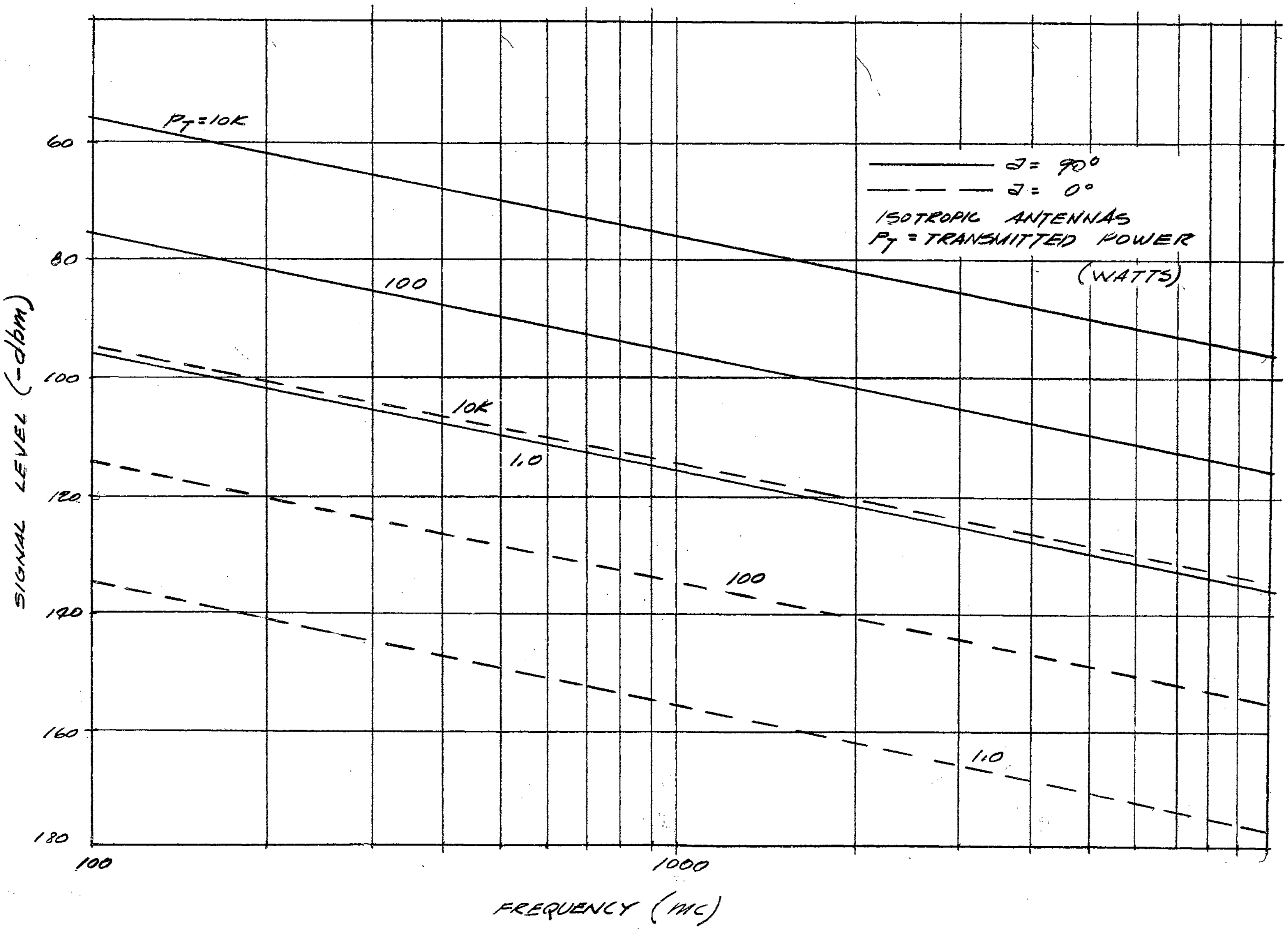

Fig. 29 Approximate Signal Strength at 100 Mile Communications Satellite Orbit

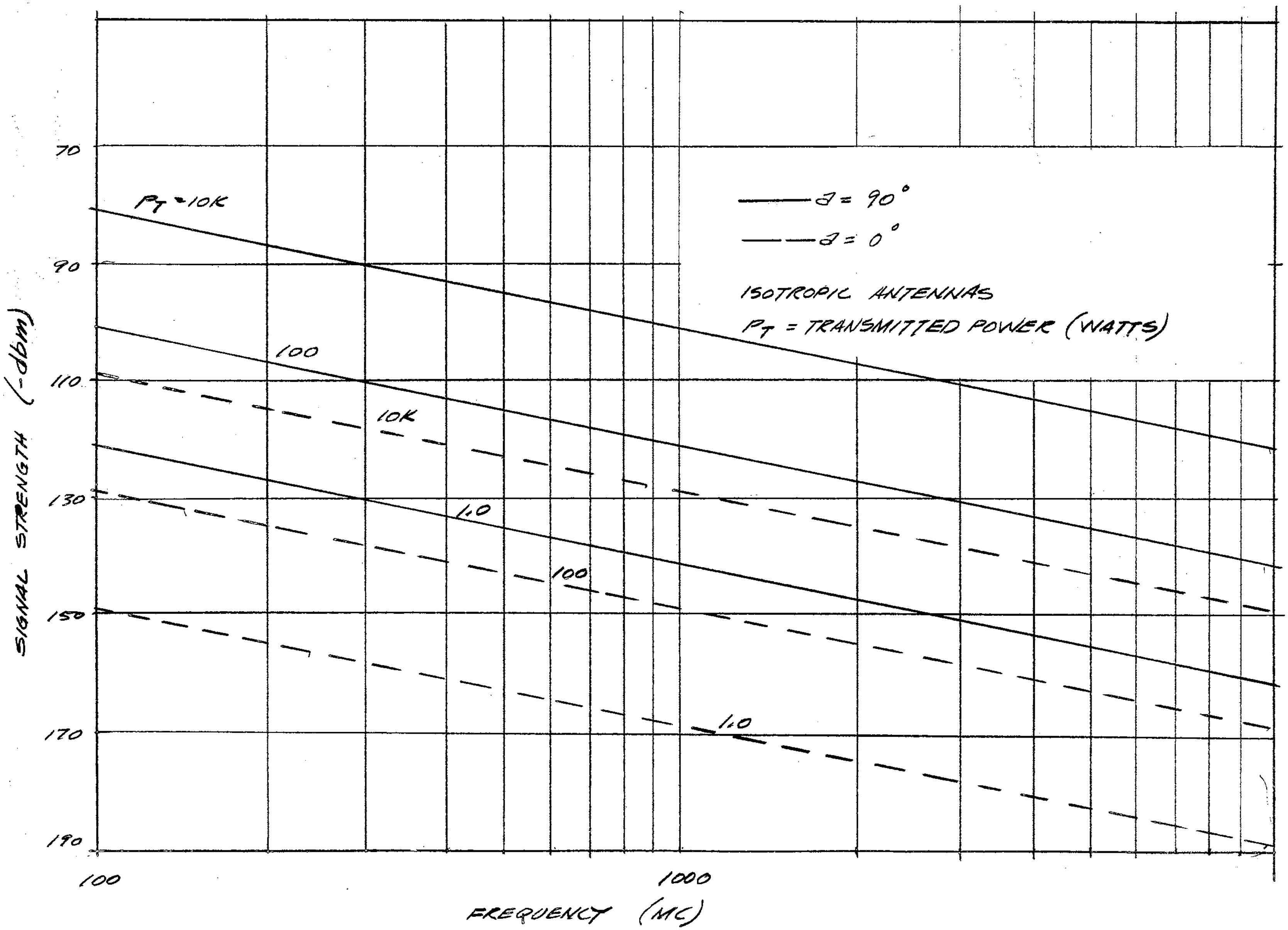

Fig. 30 Approximate Signal Strength at 1,000 Mile Communications Satellite Orbit

CHAPTER VIII

REFRACTION CONSIDERATIONS

Whenever an electromagnetic wave passes through the boundary between media of different index of refraction, the direction of propagation will change. Within the atmosphere there are no well defined boundaries but nevertheless models have been used which are idealized by considering the atmosphere to be made up of a series of concentric shells. Based on these models and supplemental experimental information, equations have been developed to estimate the expected elevation angle refractive error. Likewise for the elevation angle refractive error for an electromagnetic wave passing through the ionosphere, an expression has been derived for determining the elevation angle refractive error for the wave passing through the magneto-ionic medium.

Atmospheric

Based on calculations and experimental data, curves have been plotted for determining the elevation angle error due to refraction of the electromagnetic wave in passing through the atmosphere. This information is shown on Fig. 31.

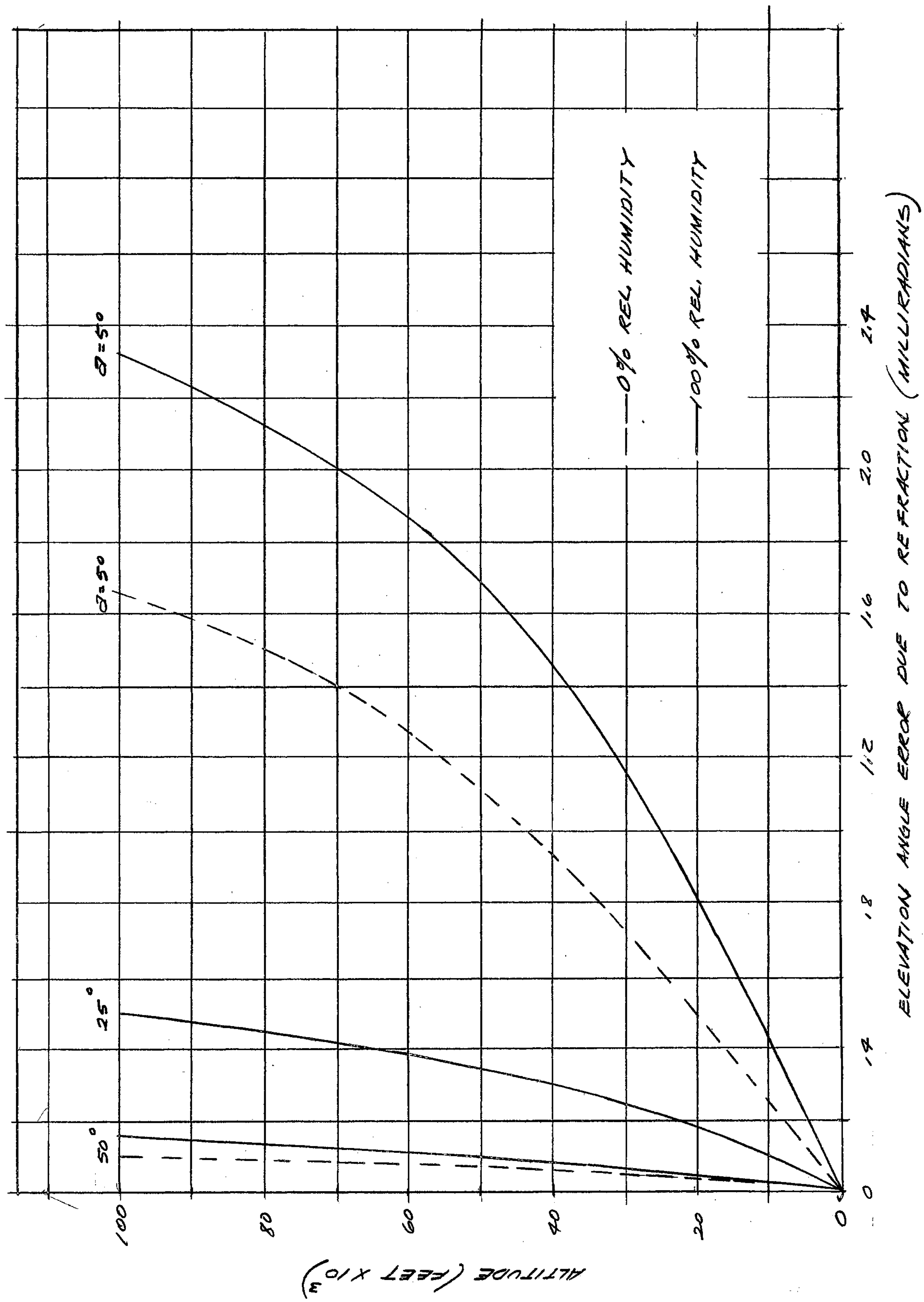

Fig. 31 Tropospheric Refraction Angular Elevation Error for Altitudes to 100,000 Feet

Ionospheric

The ion density and the gyromagnetic frequency of electrons and the collision frequency of the electrons appear to be the determining factors in the magnitude of the index of refraction. By neglecting the magnetic field of the earth and the electron collisions, the refractive index may be approximated by

$$n = \sqrt{\frac{1 - 4\pi Ne^2}{m\omega^2}}$$

where e and m are the charge and mass of the electron and the angular frequency of the wave is ω .

From the preceding equation, one can see that as the frequency is increased and the electron density is decreased the index of refraction approaches free space conditions with the result that there is less elevation angle error. Figures 32 and 33 provide information as to the elevation angle error which one might expect for several different orbital altitudes.

Total Refraction Considerations

By consulting the aforementioned Figures on tropospheric and ionospheric elevation angle errors, it becomes evident that the errors at the higher frequencies are quite small and will be considerably less than one degree. These errors should present no real problem, since the beam widths of the high gain antennas are not small enough to

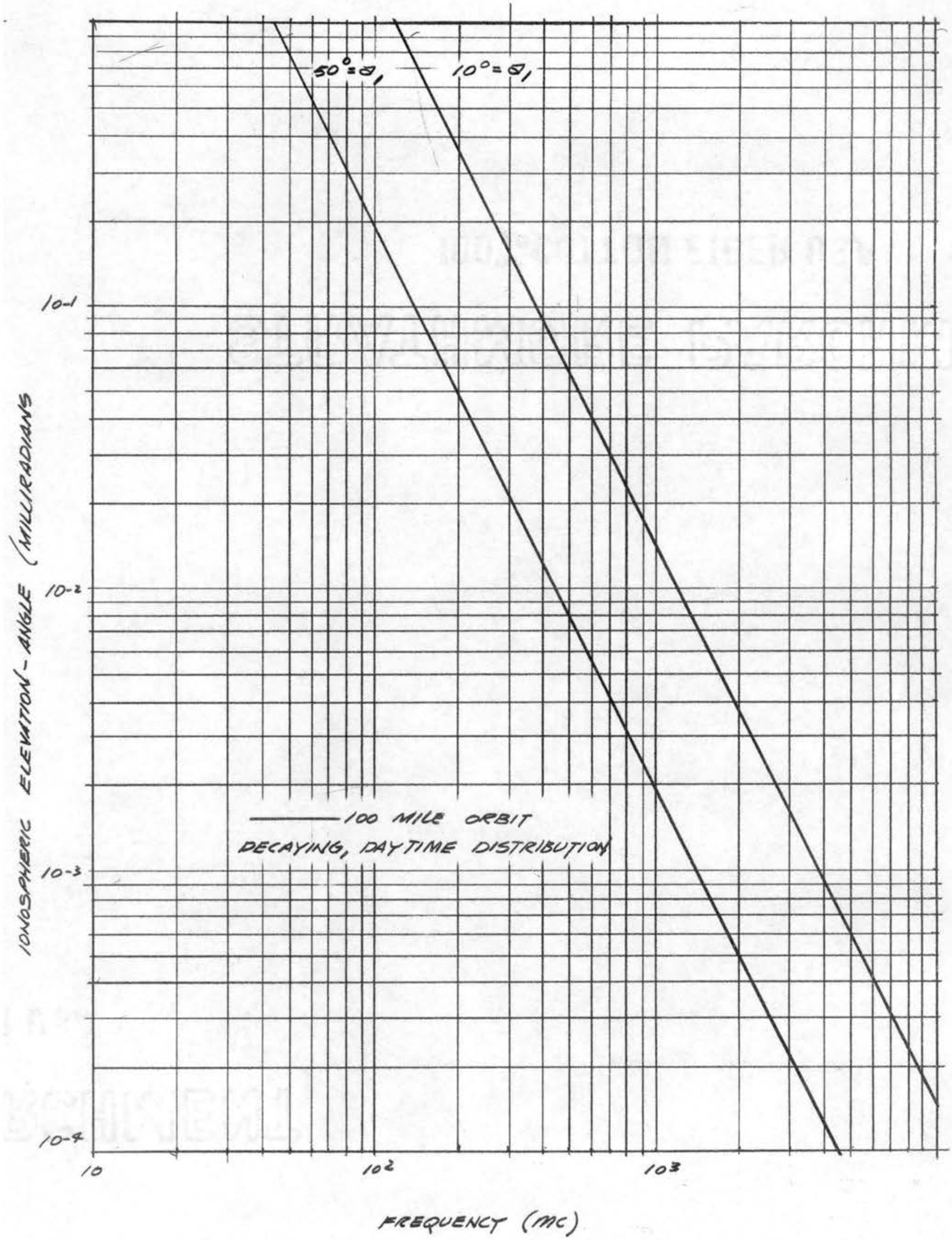

Fig. 32 Elevation Angle Error for Electromagnetic Waves Passing Through the Ionosphere (100 Mile Orbit)

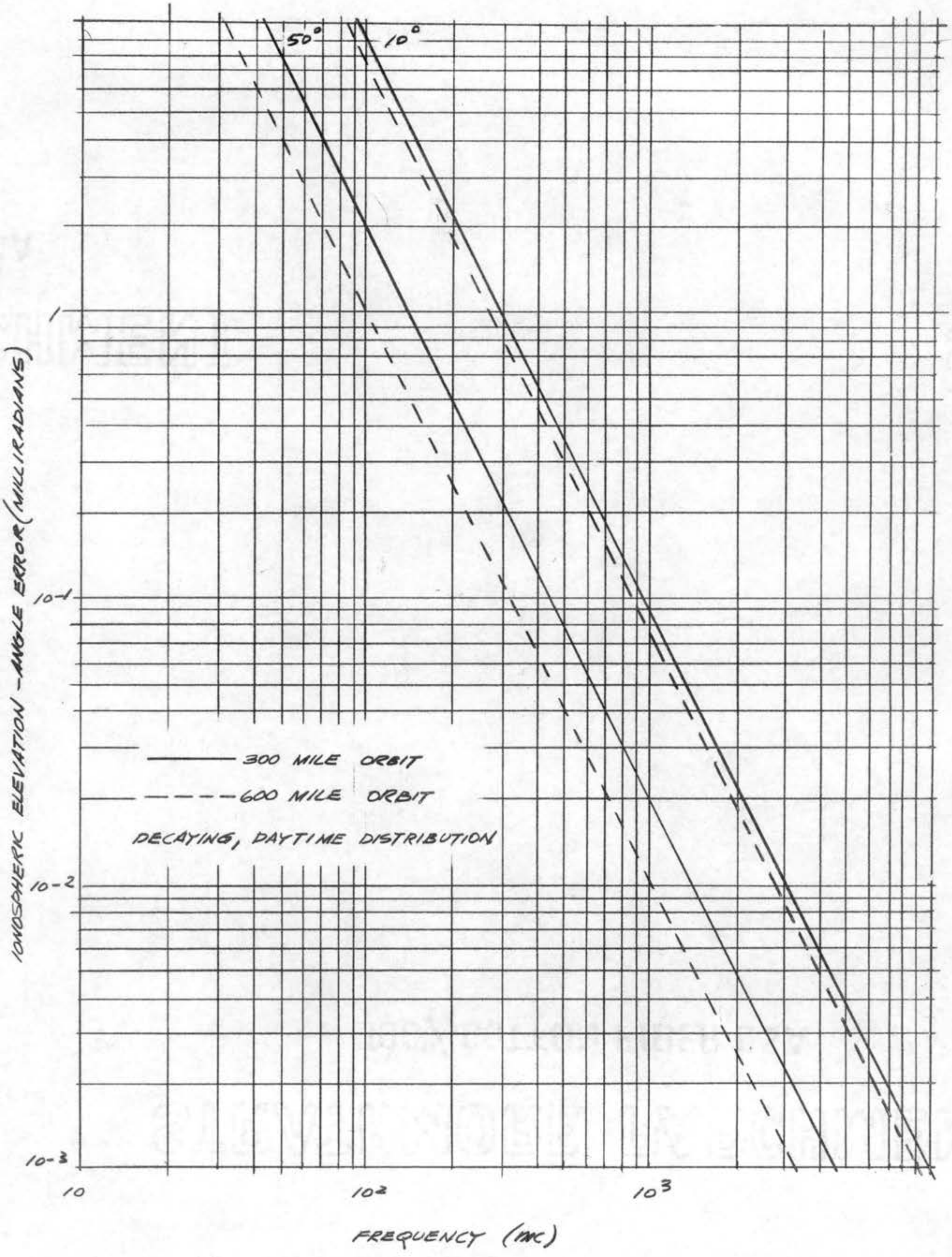

Fig. 33 Elevation Angle Error for Electromagnetic Waves Passing Through the Ionosphere (300 and 600 Mile Orbits)

present any real problem in acquisition and handover of communications satellites by the ground stations.

CHAPTER IX

NOISE AND SIGNAL EFFECTS

Ambient noise external to the receiver and the noise within the receiver appear to be the main factors which determine the minimum detectible signal. This noise external to the receiver and present and near term future capabilities is shown graphically on Fig. 34. The information shown is:

1. Atmospheric Noise
2. Cosmic Noise
3. Solar Noise
4. Some Galactic Point-Source Noise
5. Oxygen and Water-Vapor Noise
6. Current and Near Term Future Receiver Capabilities.

Atmospheric Noise

Measurements of atmospheric noise have been made by many researchers. Most of the atmospheric noise is caused by thunderstorm activity. Extensive information on this noise source has been obtained by Dr. H. L. Jones at Oklahoma State University during his many years of research in this area. The results of his findings appear to indicate

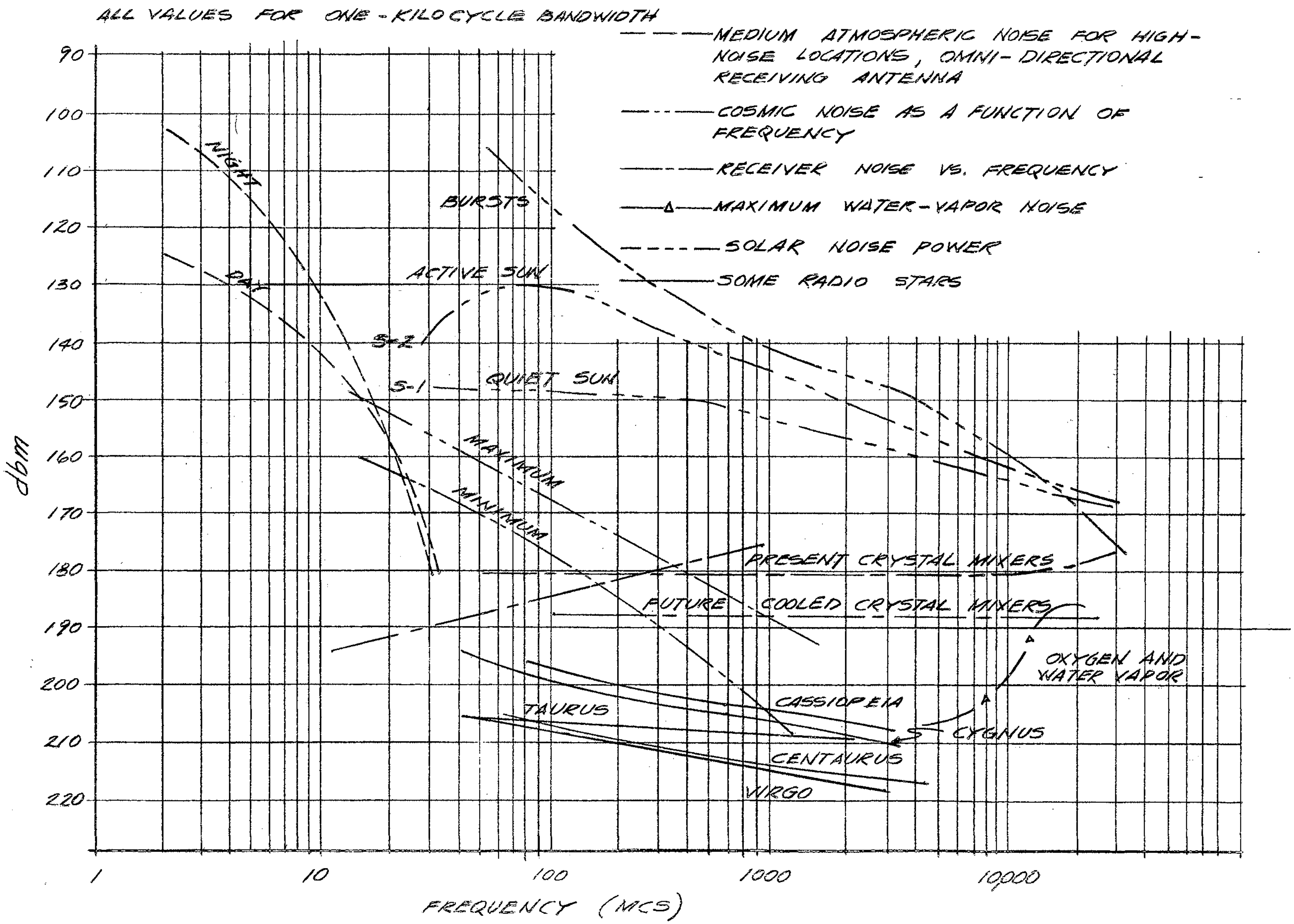

Fig. 34 Ambient Noise

that most of the energy from the lightning strokes is in that part of the radio frequency spectrum below 60 kilocycles. The noise levels from this source at frequencies of interest for communications satellite use appear to be negligible.

Cosmic Noise

There has been much information obtained on cosmic noise by many research organizations throughout the world. Maximum and minimum representative curves are shown on the Figure. At the frequencies of current interest for satellite communications, the cosmic noise does not appear to present any serious problem.

Solar Noise

Solar noise appears to over-ride most of the other noise. This is to be expected from such a high temperature radiating source and the distances involved. From the noise measurements taken, the radio emissive diameter of the sun is greater than the visual diameter.

Galactic Point-Source Noise

Same galactic point-source information was presented on the chart to obtain a feel for the extent of the noise from those sources. It should be noted that noise levels are below the objectional noise levels and therefore should present no problem.

Oxygen and Water Vapor Noise

There is some absorption of the electromagnetic energy by these elements which implies mutual electromagnetic coupling. The energy levels tend to stabilize with the signal energy being replaced with thermal noise.

Detector and Preamplifier

The noise levels of detectors and preamplifiers are of prime importance in the operation of a communications link and the lowest noise levels possible are desirable. Curves of the sensitivity of some of the detectors which might be used are shown for comparison purposes.

Measurement of Radio Noise Levels

The method of measurement of the radio noise levels as used by the National Bureau of Standards is described so that the problem might be better understood. A series of receivers are used to cover a number of frequency regions of interest. The detector output is averaged over several minutes by means of a resistance-capacitance circuit. The averaged detector output voltage operates the automatic gain control circuit and a recording meter in such a way as to provide approximately equal decibel intervals on the meter scale with practically any desired dynamic range.

The basic calibration of the equipment is by means of a noise diode. This provides a reference level of available power per unit bandwidth from individual dummy antennas that have been adjusted to match the antenna impedance at each operating frequency. Levels above and below the reference level are calibrated by means of a signal generator through the same dummy antenna. The losses in the antenna are taken into account and the recorded noise level is then expressed in terms of the antenna external noise figure, f_a, which has been defined by Norton[1] as

$$f_a = \frac{p_n}{kt_o b} \tag{17}$$

where

p_n = average noise power in watts available from an equivalent lossless antenna.

$kt_o b$ = noise power in watts available from a passive resistance at reference temperature.

For received noise such as galactic noise having the same amplitude-time distribution as the noise from the calibrating diode, f_a is evaluated as follows:

For equal detector voltages, the noise power, p_a, from the actual antenna (as contrasted to a lossless antenna) is equal to the noise power, p_d, from the calibrating noise diode (since the dummy antenna has been adjusted to the antenna impedance), and this power is given by

$$p_a = p_d = \frac{e i_d r_d b}{2} + kt_d b, \tag{18}$$

[1] K. A. Norton, Reports of Ad Hoc Committee on Radio Propagation Factors, May 26, 1949, and July 7, 1950.

where

e = electronic charge = 1.6018×10^{-19} coulomb

i_d = diode current in amperes

r_d = diode load resistance in ohms

t_d = temperature of diode load resistance in degrees Kelvin.

Taking $t_d = t_o$ (18) becomes

$$p_a = p_d \cong (20 i_d r_d + 1)\, k t_o b \,. \tag{18a}$$

Expressing the losses in the antenna circuit by a noise figure f_c, the effective noise figure, f_{ac}, for the antenna and antenna losses (taking $t_c = t_o$) is

$$f_{ac} = f_a - 1 + f_c \,. \tag{19}$$

As the power output of a network is equal to the product of its power gain, noise figure, and $kt_o b$, then

$$p_a = \frac{(f_a - 1 + f_c)\, k t_o b}{fc} \,. \tag{20}$$

Solving (18a) and (20)

$$f_a = 20 i_d r_d f_c + 1 \,. \tag{21}$$

Atmospheric radio noise from the antenna and thermal noise from the calibrating noise diode do not have the same voltage distributions with time, and thus for the same average power will not produce the same average output from a linear detector. As f_a is by definition proportional to average power, it is necessary to take into account the effect of the detector in evaluating the measurements.

Landon[2] has shown that the instantaneous voltage of thermal or fluctuation noise follows a normal distribution. Thus in the output of any linear network containing fluctuation noise, the probability that the noise voltage will be between v and v+dv is given by

$$dp = \frac{1}{e\sqrt{2\pi}} \exp\left(\frac{-v^2}{2e^2}\right) dv, \tag{22}$$

where e is the rms noise and voltage. When the noise is fed to a linear detector that follows the envelope of the noise, the instantaneous output of the detector will follow a Rayleigh distribution. The probability that the instantaneous output voltage lies between a and a + da is

$$dp_a = \frac{a}{e^2} \exp\left(\frac{-a^2}{2e^2}\right) da \;. \tag{23}$$

The average detector output voltage, $\bar{a}$ that would be measured with a long time constant record is the integral form $a = 0$ to ∞ of adp_a . Thus

$$\bar{a} = \int_0^\infty \frac{a^2}{e^2} \exp\left(\frac{-a^2}{2e^2}\right) da = \sqrt{\pi/2} = 1.253e \;. \tag{24}$$

If the recorder were calibrated by substituting a sine-wave signal for the thermal noise and adjusting its intensity to give the same recorder deflection, the recorder voltages would be

$$a_s = e_s \sqrt{2} = 1.414\, e_s \;, \tag{25}$$

[2] V. D. Landon, "Distribution of Amplitude with Time in Fluctuation Noise," <u>Pro. IRE</u> 29, 50 (1949).

where

a_s = detector output voltage with sine-wave input

e_s = rms voltage of sine-wave input to detector.

As the sine-wave input was adjusted so that $a_s = \bar{a}$, the rms noise voltage is

$$e = 1.129\, e_s \quad . \tag{26}$$

Atmospheric noise does not lend itself readily to mathematical analysis because it does not follow any specified distribution. Therefore, it is necessary to use experimental methods to determine the effect of the detector characteristics on the recorded noise. Jansky has stated that, from his experimental measurements, the ratio of average to effective voltage is 0.85 for thermal noise and 0.55 to 0.8 for atmospheric. From his circuit diagram, it appears that he was measuring the average value of the noise envelope instead of the average value of the noise. The latter was defined by Jansky to be the average of the instantaneous noise values without regard to sign. It was shown theoretically in (26) that when thermal noise is measured with a linear detector, calibrated with a sine-wave signal, the radio of rms sine-wave voltage to the rms noise voltage is 0.886. This theoretical factor apparently corresponds to Jansky's experimental value of 0.85 for thermal noise, and thus the ratio of rms sine-wave voltage to rms atmospheric noise voltage has been assumed to fall between Jansky's measured values of 0.55 to 0.8. Using the geometric mean of these

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[3]K. G. Jansky, "An Experimental Investigation of the Characteristics of Certain Types of Noise," Pro. IRE 27, 763 (1939).

two limiting values, the rms atmospheric noise voltage will be 1.51 e_s, where e_s is the rms sine-wave carrier voltage used to calibrate the linear detector.

Recent measurements at CRPL, using both a square law and linear detector, have indicated that the above ratio is 3.1 on 50 kc and 1.55 on 2.5 Mc. However, because of the limited number of CRPL observations, Jansky's ratio of 1.51 has been used in analyzing all of the atmospheric-noise data for this thesis.

As calibration of the CRPL noise recorders is by means of a noise diode instead of a sine-wave signal, it is necessary to combine Jansky's factor of 1.51 with the factor obtained from (26), and thus the average atmospheric noise power from the actual antenna is given by

$$p_a = \left(\frac{1.51}{1.129}\right)^2 p_d = 1.79\ p_d\ , \tag{27}$$

and for the atmospheric noise powers greater than about 10 kt_ob ,

$$f_a = 35.8\ i_d r_d f_c\ . \tag{28}$$

Thus, F_a , in decibels, above kt_ob, is

$$F_a = I_d + R_d + F_c + 15.54, \tag{29}$$

where

$I_d = 10 \log_{10} i_d$, $R_d = 10 \log_{10} r_d$, $F_c = 10 \log_{10} f_c$, provided the resulting value of F_a is greater than 10.

Measurements of radio noise levels at Tatsfield, England, were made by means of the Thomas method of subjective observations in which an operator adjusts the level of a low-speed radio-telegraph signal relative to the received noise so that about 95 percent copy is obtained. The noise is received on a short vertical antenna, and the data are reported in terms of the minimum required signal field strength, f_m, expressed in microvolts per meter. Taking the CCIR required signal-to-noise ratio of -7 db for a bandwidth of 6 kc for 90 percent intelligibility of 8 baud, low grade, A1 telegraphy, the noise field strength, f_n, in microvolts per meter, is

$$f_n = 2.24 f_m \quad . \tag{30}$$

The lossless-antenna noise power, in watts, is

$$p_n = \frac{f_n^2 h_e^2}{4 r_a} (10^{-12}) \tag{31}$$

where

h_e = effective height of the antenna in meters = 1/2 actual height, h .

r_a = radiation resistance in ohms = $40\pi^2 (h/\lambda)^2$.

From (30) and (31), and taking into account the ground-wave antenna power gain, g_m, relative to a short vertical monopole, f_a for the Tatsfield measurements is

$$f_a = (1.305)(10^{-18}) \frac{f_m^2 \lambda^2}{\pi^2 kt_o} g_m = (2.871)(10^6) \frac{f_m^2}{f_{Mc}^2} g_m \qquad (32)$$

where

f_m = required signal field strength in microvolts per meter

f_{Mc} = frequency in megacycles.

Thus F_a in decibels above $kt_o b$, is

$$F_a = F_m + G_m - 20 \log_{10} f_{Mc} + 64.76 , \qquad (33)$$

where

F_m = $20 \log_{10} f_m$

G_m = $10 \log_{10} g_m$.

The tangential sensitivity of a receiver is generally measured by applying a pulse moldulated RF signal to the input of the receiver and observing the output of the intermediate frequency output signal on an oscilloscope. The amplitude of the RF signal is then adjusted until the top of the noise output of the receiver is at the same level as the bottom of the noise which is modulating the RF pulse. The tangential sensitivity then is

$$S_1 = 4 \times 7.1 \, FB_e \times 10^{-21} \text{ watts}$$

$$= -165 + 10 \log F + 10 \, B_e \text{ dm} \qquad (34)$$

where

$$B_e = \sqrt{2B_r B_v - B_v^2}$$

and

B_r = RF Bandwidth

B_v = Video or IF Bandwidth .

Considering an ideal receiver where the output signal to noise ratio is equal to the input signal to noise ratio, the tangential sensitivity under these conditions as obtained by the preceding Equation is plotted in Fig. 35. To obtain the signal input required for a desired signal to noise ratio output, the noise figure of the actual receiver used must be added to the tangential sensitivity obtained from the graph.

Antenna Noise

The problem on antenna noise is not included specifically in the scope of this thesis. There is already much information available on this subject from numerous sources. It should be noted, however, that the radiation received by the receiving equipment consists of the propagated signal in the vicinity of the receiving antenna times the gain of the antenna, the noise picked up by the antenna times the antenna gain, and the noise of the antenna itself.

Modulation

Basically there are two types of signals. These are discrete and

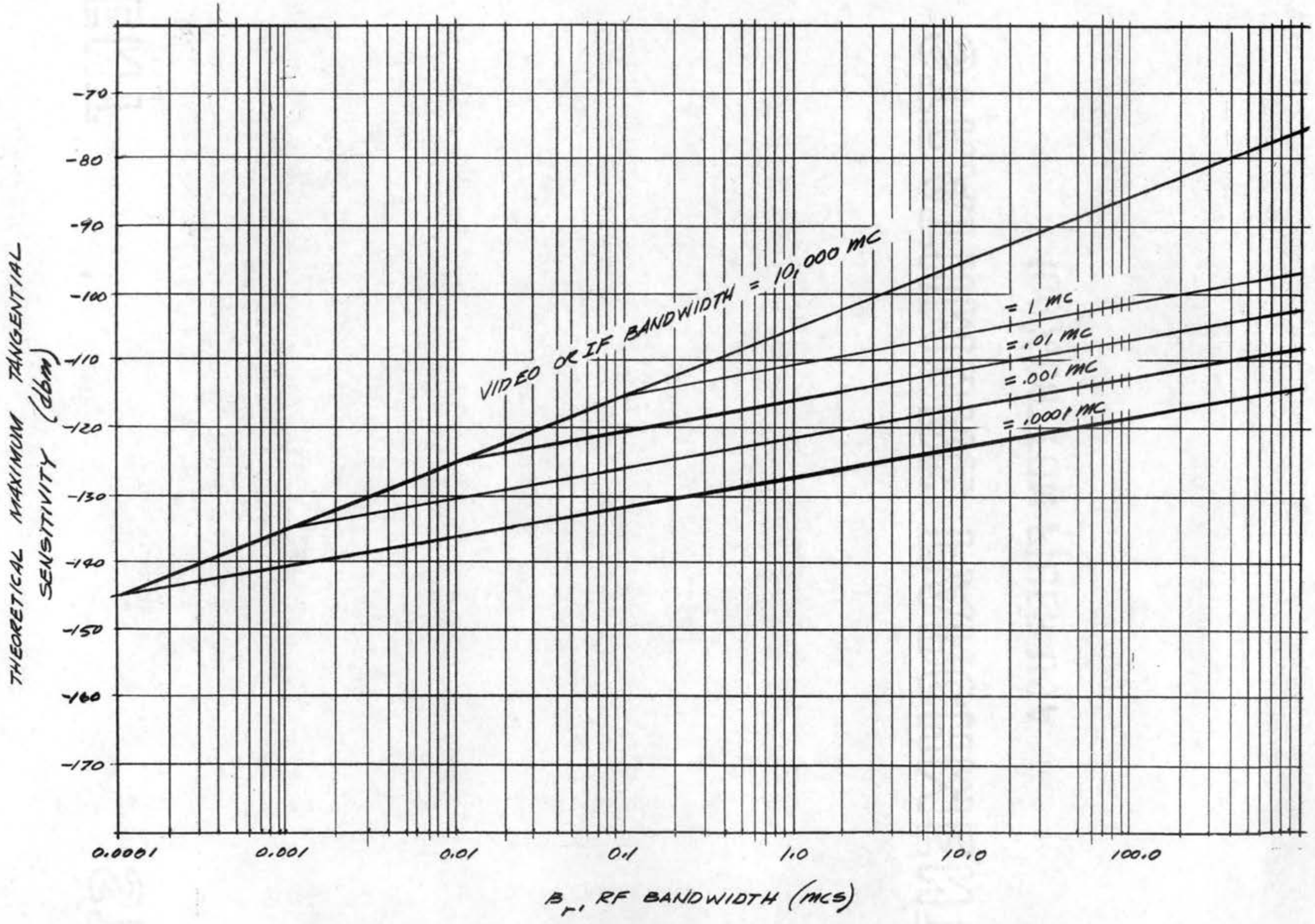

Fig. 35 Theoretical Tangential Sensitivity vs RF Bandwidth

continuous types. Examples of discrete type signals are pulse code, pulse amplitude and pulse frequency coding. Voice, Morse, and two-tone teletype signals are examples of continuous signals. With amplitude modulation, the bandpass of the transmitted signal is twice the modulation frequency. For example, for a standard voice modulation frequency of 3,000 cps, the bandpass signal would be 6,000 cps. See Fig. 36 for modulation envelopes of some of the types of signals. Single sideband uses one half of the bandpass used with similar AM modulating frequencies.

The bandpass which will be needed to pass a pulse is not easily precisely determined. The general equation for normally satisfactory pulse fidelity is

$$\text{Bandpass} = 2 \times \frac{1}{\text{pulse time}} \ \text{cps} \ .$$

The bandpass receiver requirements will affect the noise figures of the system and therefore the determination of the bandpass requirements in the over-all planning procedure will be necessary.

Signal Strength

As previously shown, the path length between a communications satellite and the ground station varies to a great extent from the horizontal to the vertical. Figure 37 will give the reader an idea of the difference in distances involved. Figures 38 and 39 show the orbital signals which can be expected. By combining the previously discussed propagation losses and the free space losses for a 300 mile orbit

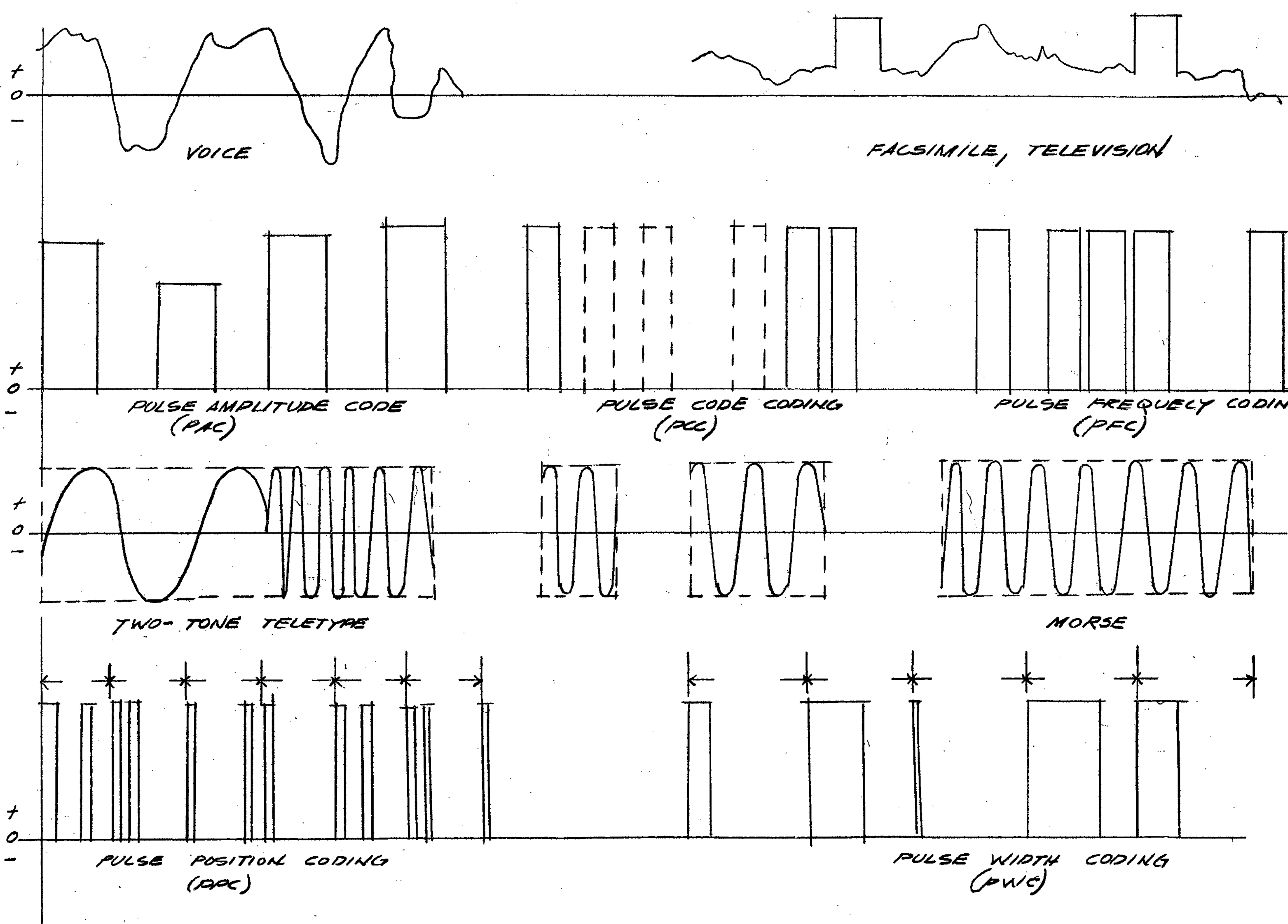

Fig. 36 Modulation Envelopes of Some Typical Signals

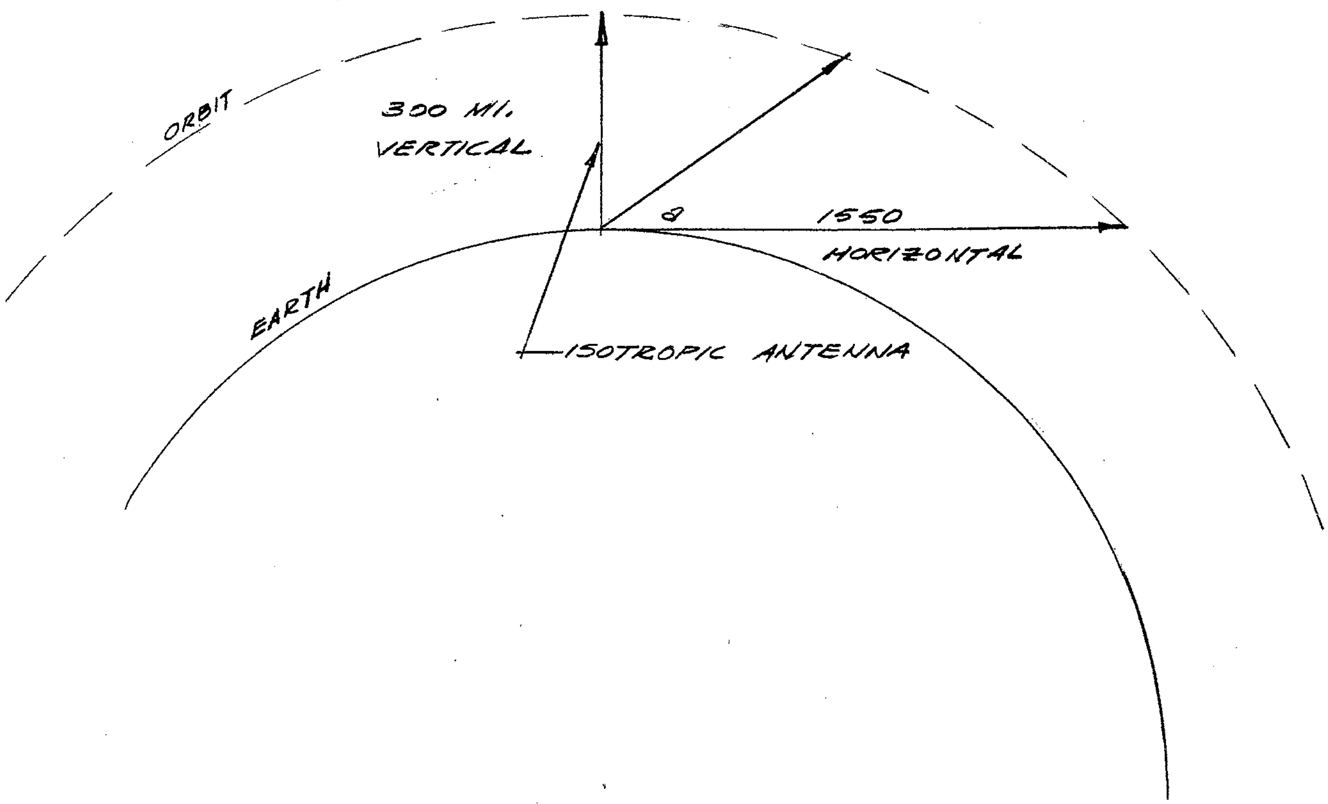

Fig. 37 An Example of the Difference in Horizontal and Vertical Signal Path Lengths

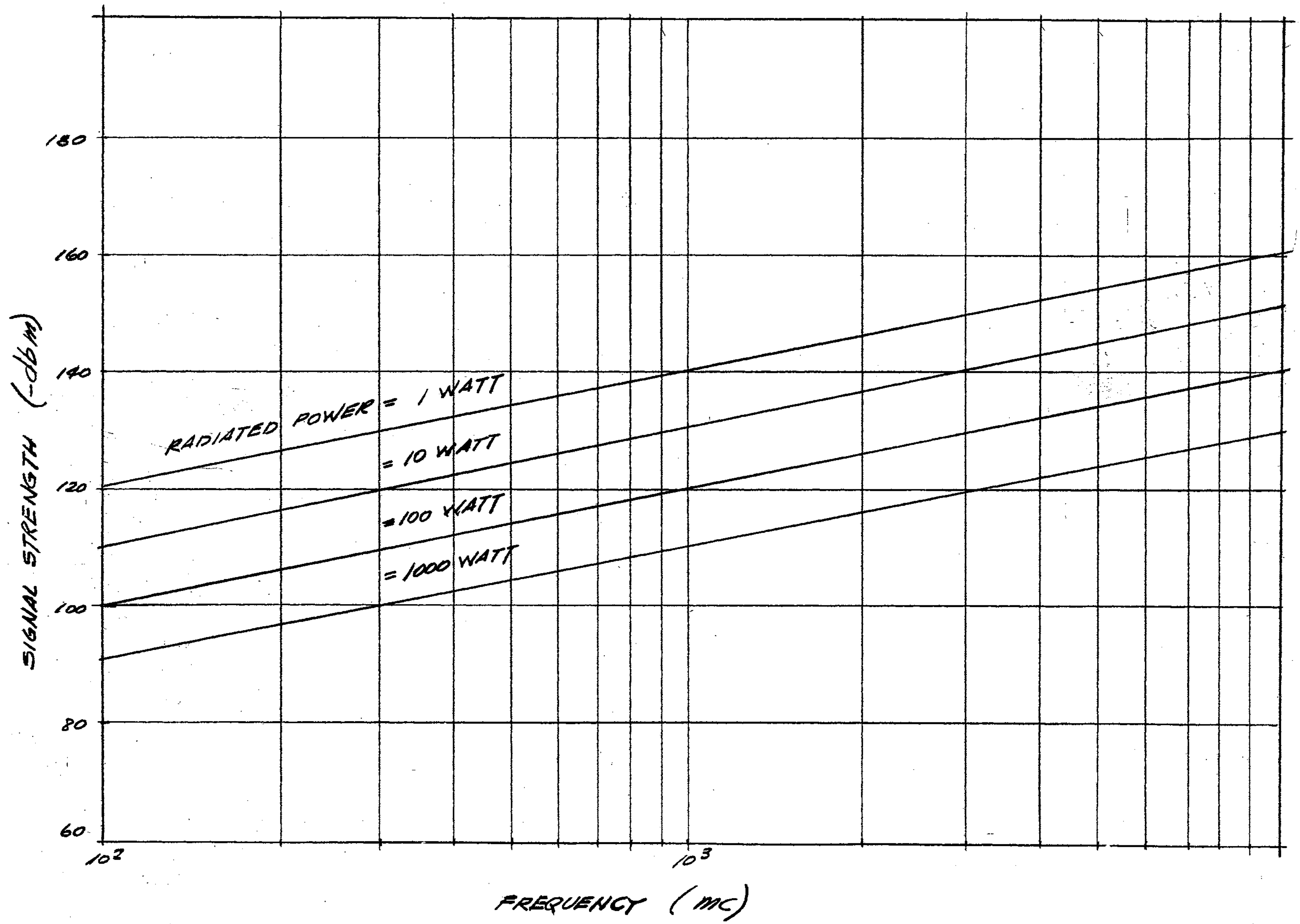

Fig. 38 Orbital Signal Intensity with Isotropic Antennas, and for 300 Mile Orbit at Extreme Slant Range

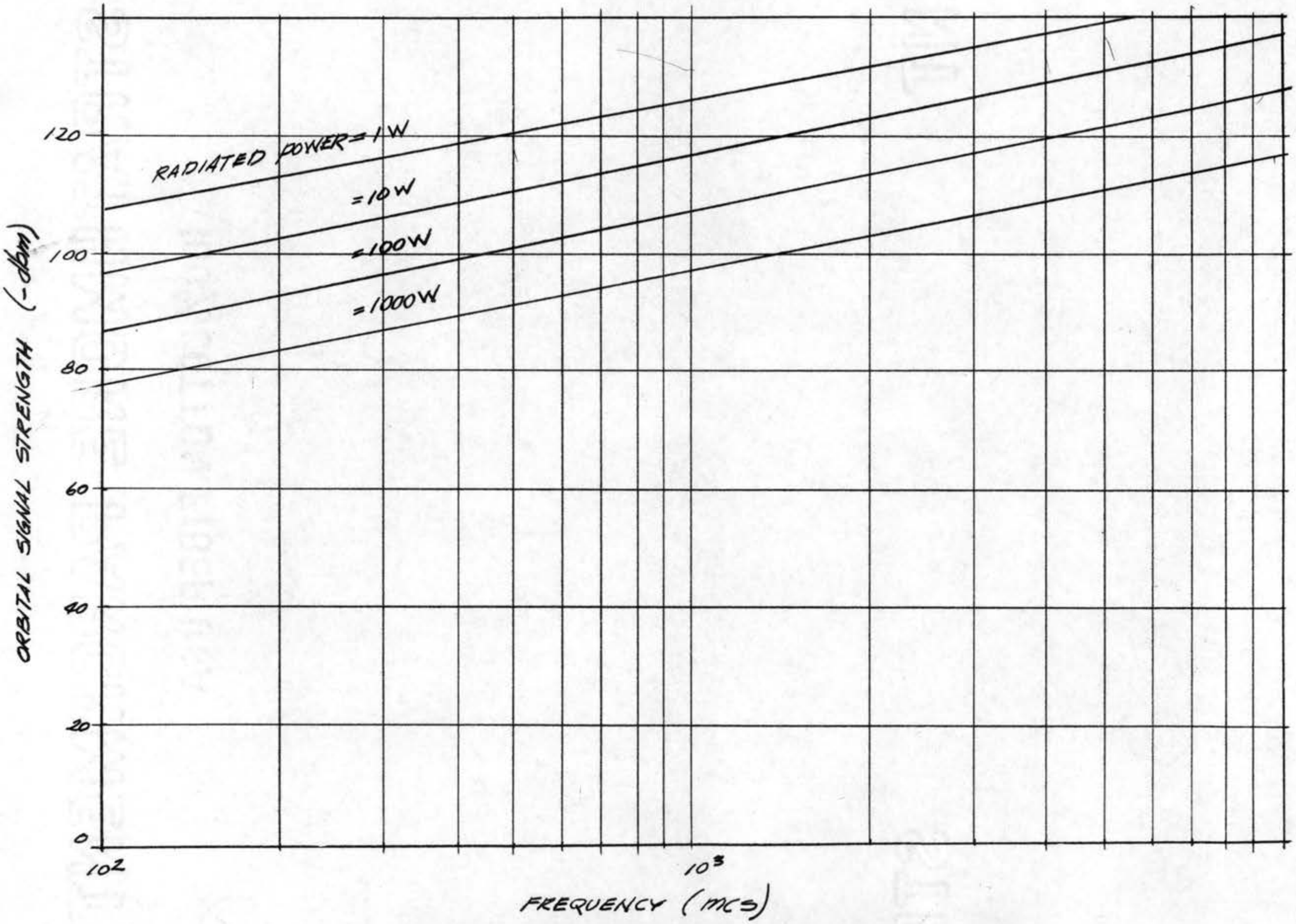

Fig. 39 Orbital Signal Intensity with Isotropic Antennas and 300 Mile Orbit at Shortest Slant Range (Vertical)

situation, the expected signal strengths are shown on the just previously mentioned Figures. These Figures do not consider additional losses which might be expected from effects such as multipathing, scattering, etc. It appears that -12 to -18 db should be added to get more realistic information.

Signal Preamplifiers and Detectors

For a fixed bandwidth the sensitivity of a receiver is, in a large measure, determined by the preamplifier stages or, in the absence of preamplifier stages, by the detector itself. Most of the noise which is seen in the output of a receiver is that noise which is generated in the first stage. The reason for this is that the noise signal from this stage undergoes amplification in the succeeding stages and thus causes it to be the dominant source of noise which ultimately composes the noise part of the output signal of the receiver. Because of the distances involved in communications via communications satellites, the other propagation medium attenuations and the limited weights of the payloads which can be placed in orbit, the signal at the receiver will generally be rather weak. Assuming that the signal is sufficiently above the noise level, which also reaches the antenna terminals of the receiver, to produce a usable signal at the output of an ideal receiver it is obvious that any departure from the ideal receiver will limit the signal to noise ratio in the output and cause a degradation of the system. Many of the noises which reach the antenna of the receiver are beyond the control of man and, as such, there is little that can be done about it. Doubtlessly there will be improvements in antenna systems of the future which will reduce the noise produced in the antenna itself. But it appears that the greatest improvements in reducing the noise have been in the early stages of the receivers used. There is a continuing effort going on at many research activities to improve detectors and amplifiers still further. Because of the frequent improvements in this field, it will be impossible to provide information on the various devices which can be considered up to date. The latest information on these devices should be obtained when any actual planning is to be done. Figure 40 shows the noise temperature of the various devices, and Fig. 41 shows the possible frequency ranges of the various devices.

From the information on these two Figures, it should be noted that the MAVAR or parametric amplifier appears to have the greatest promise for operation at the upper frequency limits. Low noise temperature MASERS require cooling. It appears that the parametric amplifier will approach, if not exceed, the capabilities of the MASER without any cooling required.

Noise in Receivers

The previous part of the chapter dealt primarily with the noise external to the receiver and methods of calibrating the noise recording equipment. Now the receiver noise problem will be discussed.

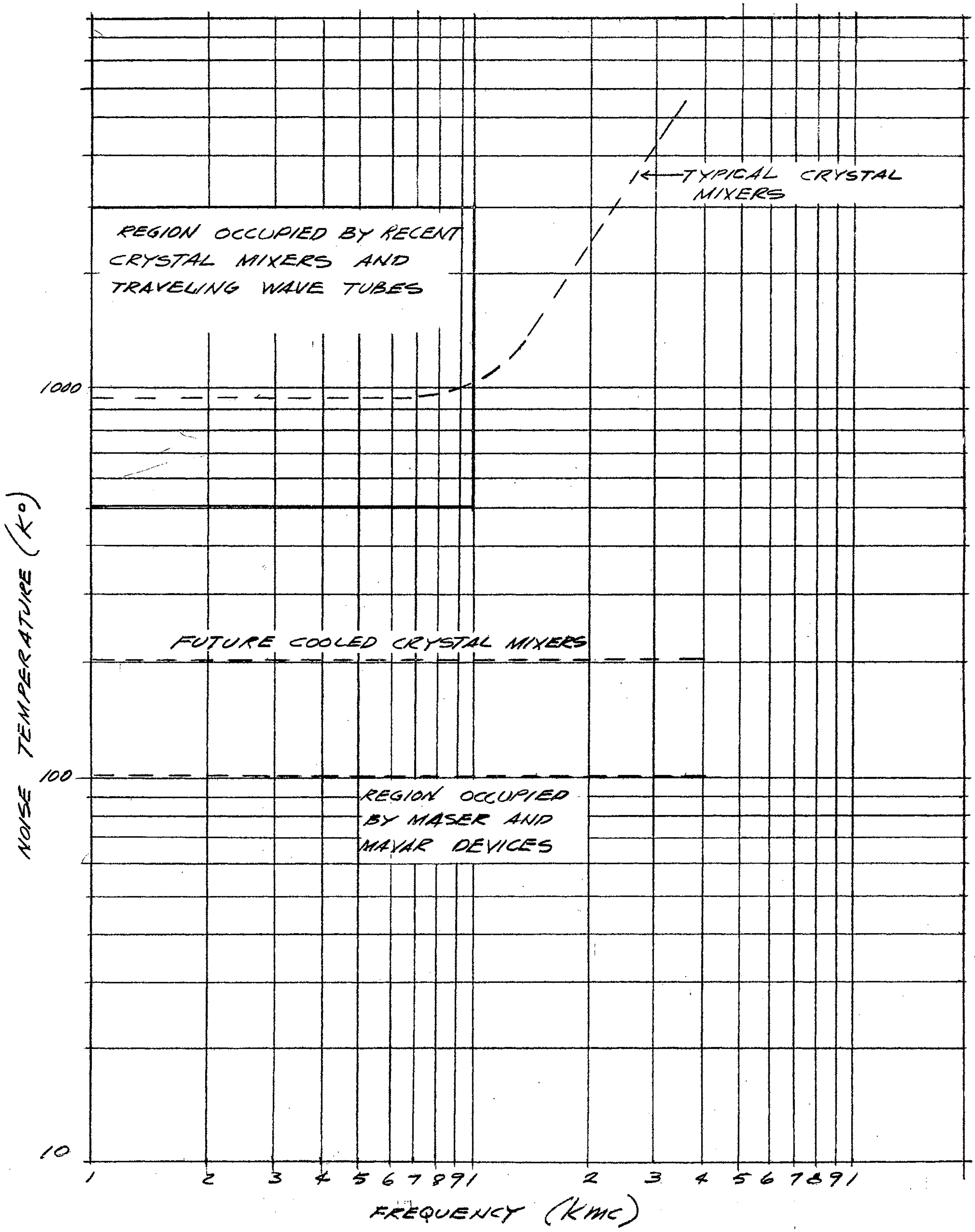

Fig. 40 Noise Temperature vs Frequency for Some Preamplifier and Detector Devices

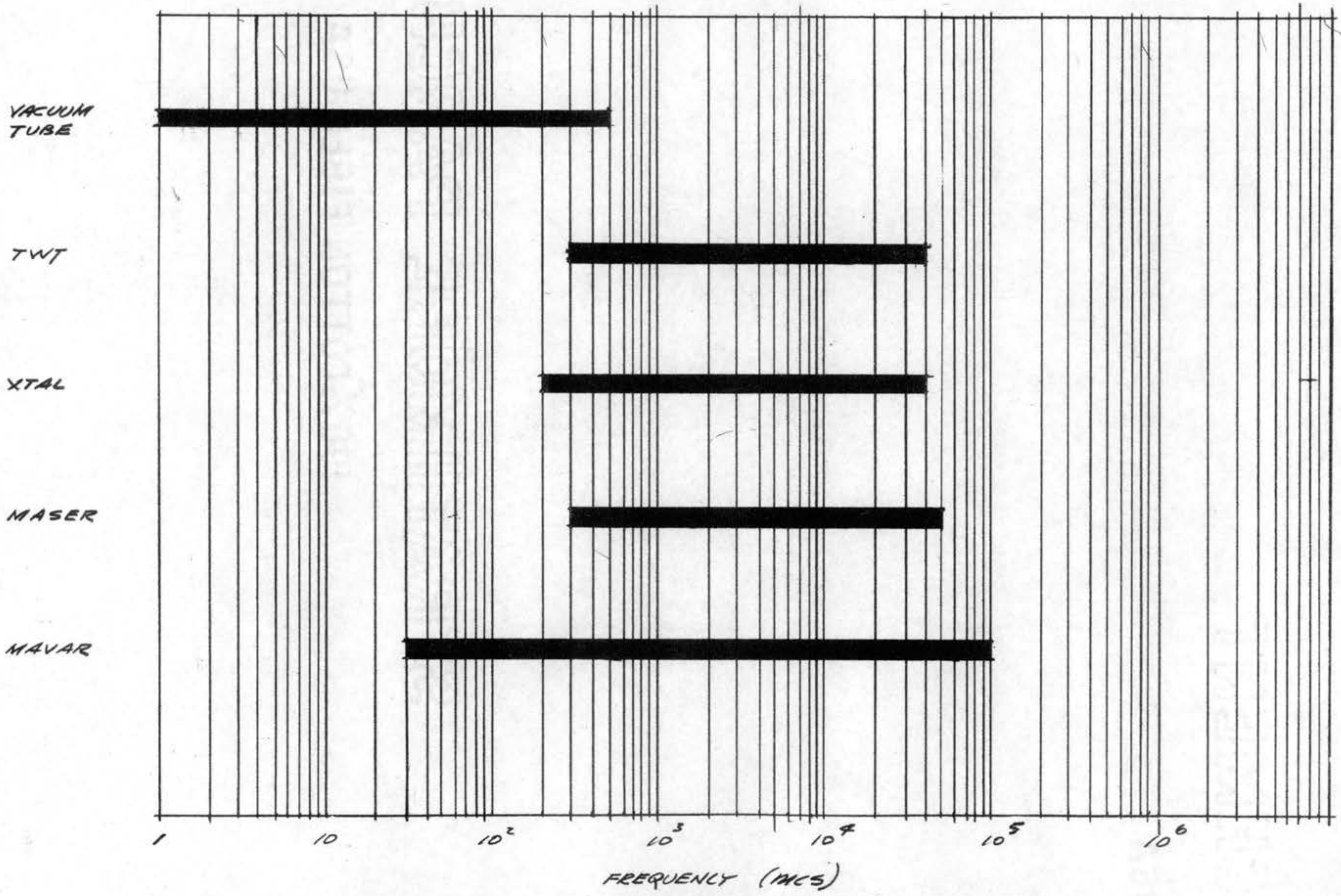

Fig. 41 Preamplifier and Detector Possible Frequency Operating Ranges

The noise figure of any active network can be expressed by

$$F = \frac{S_1/N_1}{S_0/N_0} \tag{35}$$

where F is the noise figure, S_1/N_1 is the available signal power to the available noise power ratio at the input of the active network, and S_0/N_0 is a similar output ratio of the same network. In the case of the receiver this noise figure is usually applied to the RF stages up to but not including the first detector. In any case most of the noise which is produced in the first stage is further amplified by the succeeding stages and therefore the noise of the first amplifier stage becomes usually the limiting noise factor. The generally-used expression for determining the input noise power is

$$N_1 = KTB_e = 4 \times 10^{-21} B_e \text{ watts} \tag{36}$$

where

K = 1.38×10^{-23} Joules per ^{o}K

T = absolute temperature of signal source

B_e = equivalent bandwidth of network in cps .

By defining the particular value of output signal to noise ratio which is necessary for the particular system, the minimum input signal to satisfy this condition can now be calculated:

$$S_1 = N_1 F (S_0/N_0) = 4F B_e(S_0/N_0) \times 10^{-21} \text{ watts.} \tag{37}$$

CHAPTER X

CONCLUSIONS AND RECOMMENDATIONS

The object of this thesis is to research and investigate a number of the more important factors which should be considered in planning communications satellite systems. In so doing, an attempt was made to present the information in a manner which would permit the factors to be considered in an orderly, logical arrangement that would make it more usable for future planners. Chapters were also arranged so that the more fundamental information occurred in the earlier chapters and was followed in the latter chapters by the items which required a knowledge of this background information for an understanding of the problems.

As the result of the investigation, the following conclusions and recommendations are presented.

Conclusions

1. When non-synchronous communications are used, there is very little increase in earth surface distance coverage with increase in satellite altitude above two earth radii.

2. During one orbital pass, there is very little increase in the earth's surface area coverage as orbital altitude is increased above two

earth radii.

3. Increases in orbital altitude up to synchronous altitude increases the orbit period substantially.

4. Low inclination orbits appear to be less wasteful and reduce redundant coverage.

5. Uniformly spaced non-synchronous satellites reduce the operational complexity through fewer satellite handovers and acquisitions over short time periods.

6. Uniformly spaced non-synchronous satellites are less wasteful of satellite coverage by reducing or eliminating redundant coverage.

7. Synchronous communications satellites, by remaining substantially fixed with respect to a point or general area of the surface of the earth, provide the following advantages over other conventional means of communications.

They can provide wide coverage multiple-user service.

Microwave frequencies can be used to provide reliable propagation.

High-capacity, wide-band traffic could be accommodated.

Little or no ground tracking would be required.

Fixed antennas could be used thereby reducing the construction and operating costs.

Fixed antennas would improve the survivability from nuclear attack because they are more easily hardened than rotatable antennas.

Little or no doppler shift problems would be encountered.

8. The disadvantages of synchronous communications satellites appear to be:

Sophisticated launching techniques would be required.

High power boosters will be required.

Attitude stabilization and station capability would be required.

High power capability will be necessary because of the long distances involved.

9. The probability of two ground stations being able to communicate with a satellite of a randomly distributed system can be calculated.

10. With a particular system of randomly distributed satellites, the probability of communications between two stations on the surface of the earth increases as the distance between the two ground stations is decreased.

11. Propulsion systems are currently available which have the capability of placing simple communications satellites in non-synchronous orbits.

12. Extremely simple, research type, synchronous satellites might be placed into orbit with current technology capability.

13. Heavy high powered communications satellites probably weighing as much as 15,000 lbs. could be placed into synchronous orbit before the end of the decade.

14. High powered synchronous communications satellites have great political and military value because of the possibility of being able to communicate with large masses of the world's population.

15. Nuclear power sources will eventually be required in space communications to supply the large amounts of power needed for high power communications satellites.

16. Ion propulsion systems appear to offer a great potential for use in station keeping and attitude control of communications satellites.

17. Of the energy storage devices considered, the hydrogen-oxygen fuel cell seems to offer the greatest promise for application to future requirements.

18. Nuclear power sources have the advantage of providing power to the satellite communications package without regard to orbital position (shadow or sunlight).

19. Because of the large number of ground stations which would be used in a synchronous orbit communications satellite system, it appears that it might be more economical to plan for high power communications satellites to permit the ground stations to be kept simple and relatively inexpensive.

20. Because high power communications satellites will permit ground stations to operate with relatively simple antennas, they could be used to provide communications to mobile stations such as airplanes and ships.

21. The satellite environment must be considered in planning communications satellite systems. Radiation belt conditions, meteoroids, orbital inclination and altitude should be evaluated.

22. System losses in the propagation medium and equipment can be determined accurately enough for planning purposes.

23. Faraday rotation will occur and should be taken into account in systems planning.

24. Losses are greater for horizontal paths than for the vertical.

25. Refraction in the atmosphere and the ionosphere does not appear to present any serious problem.

26. A window in the frequency spectrum, which permits less signal attenuation and noise, exists between about 90 and 10,000 megacycles.

27. Parametric amplifiers and MASERS appear to offer the best signal to noise ratio capabilities.

Recommendations

1. That the Department of Defense start a project to coordinate the efforts of all engineering areas involved with communications satellites to place a high power communications satellite in synchronous orbit as soon as the high power boosters become available. The national prestige gained from being first in this type of venture would be tremendous.

2. Studies be continued to confirm the probability calculations of the author of this thesis. This could be accomplished by generating random numbers and creating a mathematical model of a system for sample results.

3. Continue and possibly accelerate research and development of nuclear power sources for use in communications satellites.

4. Continue and possibly accelerate research and development efforts on hydrogen-oxygen fuel cells.

5. Accelerate research and development on ion-propulsion systems to be used in communications satellite station keeping and attitude control systems.

6. Economic studies should be undertaken to determine the costs of a large number of complicated ground stations operating with simple low power communications vs simple ground stations operating with high powered expensive communications satellites.

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