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ABSTRACT

The function, design, and ground statict requirements of the AMSAT-OSCAR ncn-ccmmercial space satellite system are described, and various experiments designed for direct student participation in space communications are provided in this text primarily written for the science educator. Six chapters include: (1) an introduction to general satellite crbit problems and sclutions, (2) a number of tracking methods for ccmputing the positicns of satellites, (3) onboard systems common to most satellites, (4) a complete description of each of the satellites currently in orbit or planned for the near future, (5) specifics fcr ground stations, and (6) four complete student satellite experiments, 15 detailed prcject outlines, and 19 suggestions for tcpics and activities. Appendices include listings of equipment manufacturers and publishers, as well as a glossary of terms. (Author/RAO)

USING SATELLITES IN THE CLASSROOM: A GUIDE FOR SCIENCE EDUCATORS

Martin R. Davidoff, Ph.D.

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A GUIDE FOR SCIENCE EDUCATORS

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This book is dedicated to

the volunteers around the world who helped the OSCAR program fly.

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CONVERSION FACTORS

Length $1.000 \text{ m} = 3.281 \text{ feet} = 6.214 \times 10^{-4}$ mile (statute)
$= 5.396 \times 10^{-4}$ mile (nautical)

$1.000°$ of arc at surface of earth = 111.2 km

Mass $1.000 \text{ kg} = 6.852 \times 10^{-2}$ slugs

Force $1.000 \text{ N} = .2248$ pounds

1.000 kg (force) $\longleftrightarrow$ 2.205 pounds (at surface of earth)

CONSTANTS

R Mean radius of earth: 6.371×10^{6} m (3,959 st. mile)

R_{eq} Equatorial radius of earth: 6.378×10^{6} m (3,963 st. mile)

M Mass of earth: 5.98×10^{24} kg (4.10×10^{23} slugs)

G Universal Gravitational Constant:

$$6.67 \times 10^{-11} \frac{m^3}{kg\text{-}s^2} \; ; \; \left(3.44 \times 10^{-8} \frac{ft^3}{slug\text{-}s^2}\right)$$

GM $$3.986 \times 10^{14} \frac{m^3}{s^2} \; ; \; \left(1.408 \times 10^{16} \frac{ft^3}{s^2}\right)$$

P_o Solar constant: 1.38 kwatt/m^2

σ Stefan-Boltzmann constant:

$$5.67 \times 10^{-8} \frac{\text{joules}}{K^4 m^2 s}$$

Abbreviations

K = Kelvin
kg = kilogram
m = meter
N = Newton
s = second

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PREFACE

TO THE SCIENCE EDUCATOR:

The primary objectives of this book are:

1. to show science educators how space satellites can be used to
   - a) present normal course content in a new and intrinsically interesting format and/or
   - b) introduce students to space science;
2. to provide educators with complete information needed to assemble a satellite ground station suitable for "live" classroom demonstrations and "hands-on" "real-time" student laboratory exercises;
3. to provide science educators with background material on satellite systems.

This text treats the use of a new and exciting tool for science instruction -- multi-million dollar satellites which are directly available to faculty and students without charge for use in the classroom. Note that this text is not concerned with instructional TV or radio via satellite.

INTRODUCTION

The major practical development of the space age has been earth satellites. In the brief period since the launch of Sputnik I (Oct. 4, 1957) space technology has evolved to the point where today, the majority of international telecommunications is handled by satellite [1]. Satellites are also used to provide daily data for weather forecasting and environmental monitoring as well as for natural resource assessment, navigation, and instructional TV broadcasting. Satellite parking problems have already developed in some regions of space due to the increasing number of satellites in certain desirable orbits [2].

While a few science educators pioneered in using early satellites in the classroom, educational applications of satellites never became widespread. Perhaps the main reason for this is that educational applications have always been a distinctly secondary consideration in the development of most satellites. Therefore, system design, scheduling, etc., have not been arranged to facilitate educational use. Meanwhile, improved technology and the large amount of data being transmitted to earth have led satellite communications system designers to employ higher and higher frequencies and use involved modulation schemes -- greatly complicating ground station requirements. We are all familiar with the pictures of big "dishes" and rooms full of electronic equipment. Using commercial and scientific satellites in the classroom was actually much easier in the "good old days" when satellite systems were far less complex.

In 1969 work began on a series of long-lifetime non-commercial satellites designed to facilitate educational applications. The first satellite in this series, AMSAT-OSCAR 6, was launched Oct. 15, 1972 and served until June 1977. Two satellites in this series, AMSAT-OSCARs 7 and 8, are currently operating. Built primarily by volunteers from Australia, Japan, Canada, the United States, and West Germany these satellites were launched by the United States National Aeronautics and Space Administration (NASA) to provide "an opportunity for radio operators in the developing countries to become involved directly in space communications and, for the first time, for educators to have available a real satellite for direct student participation in and observations of space communications" [3]. OSCAR is an acronym for Orbiting Satellite Carrying Amateur Radio, and AMSAT stands for the Radio Amateur Satellite Corporation, a non-commercial organization composed of individuals devoted to building satellites for educational, public service, and scientific purposes [4]. Construction has begun on a new advanced series of spacecraft designed for much higher orbits. These high altitude models are referred to as Phase III satellites while AMSAT-OSCARs 6, 7, and 8 are called Phase II satellites.

A group of Russian scientists and academicians formally announced in July 1977 that work on a series of satellites similar to those in the AMSAT-OSCAR Phase II program was underway. The first Russian RS satellite (RS-1) will probably be launched shortly and two or three additional spacecraft are promised for 1978. The reader should keep in mind that, although this text only provides data on RS-1 there may be a number of similar RS spacecraft in orbit and available for use as this is being read.

It is hoped that this book, in conjunction with the AMSAT-OSCAR and Russian RS spacecraft will enable science educators to introduce into their curricula demonstrations and student laboratory exercises involving direct use of these satellites. Since each institution will (1) have available different types of equipment and (2) be interested in different educational objectives, this text is designed to provide comprehensive background information enabling science educators to formulate student laboratory experiments and/or demonstrations which complement their curricula objectives and which, when possible, use equipment already on hand.

An attempt has been made to balance the presentation, giving (1) specific information on AMSAT-OSCAR 7 and 8 and RS-1 so educators can develop programs around them and (2) presenting general information applicable to all satellite systems so that this book will also prove useful when future satellites in the AMSAT-OSCAR and Russian RS series become available.

This text was primarily written for the science educator. However, a number of study aids have been incorporated so that instructors may assign specific sections to students. The MKS system of units is used exclusively in the body of the text.

Chapter I introduces the general satellite orbit problem and, using an intuitive and non-rigorous approach, discusses how it is solved and the general characteristics of the resulting motion. The aim is to provide the reader with an overview of the important properties of satellite orbits, an ability to visualize the motion, and access to the equations needed to compute particular orbital parameters. In Chapter II a number of tracking methods, based on the developments of Chapter I, for computing the positions of current and future AMSAT and Russian satellites are presented. Chapter II has been designed so that it is completely self-contained and can be read out of context. In Chapter III, important onboard systems common to most satellites are discussed. A complete description of each of the satellites currently in orbit, or planned for the near future, is given in Chapter IV. In Chapter V, the specifics of ground station assembly are presented. It is hoped that readers will be pleasantly surprised to find out how simply a ground station can be constructed -- the ubiquitous "short wave" receiver and a little "know how" can make a number of experiments and demonstrations possible. Chapter VI includes (1) four complete student Satellite Experiments (SEs), (2) fifteen detailed Satellite Project Outlines (SPs), and nineteen brief Satellite Miscellaneous Suggestions for topics and activities (SMs) focusing on satellites. A list of addresses of the organizations, manufacturers and publishers mentioned in the text is contained in Appendix A.

Comments on this text are helpful and always appreciated by this author.

References

Introduction

1. B. I. Edelson, "Global Satellite Communications," Scientific American, Vol. 236, no. 2, Feb., 1977, pp. 58-68, 73.

2. W. L. Morgan, "Satellite Utilization of the Geosynchronous Orbit," COMSAT Technical Review, Vol. 6, no. 1, 1976, pp. 195-205.

3. Space Science Involvement, published by the American Radio Relay League, Inc., Newington, Conn. 06111, (1974).

4. What is AMSAT?, available from AMSAT, Box 27, Washington, D. C. 20044.

AMSAT Phase III-A satellite is shown with Karl Meinzer (left), designer of many of the subsystems, and Jan King (right), spacecraft project manager. (Photograph by Richard Daniels).

CHAPTER I

EARTH SATELLITES: ORBITS AND TRACKING

In order to work with satellites, we have to know where they are going to be at any given time so we begin this book by examining satellite orbits. The objectives of this chapter are:

1. to introduce the satellite-orbit problem,
2. to provide the reader with an overview of the important parameters of satellite motion and an ability to visualize satellite motion,
3. to summarize the important equations needed to compute orbital parameters so that these equations will be easily accessible when needed.

The reader who, at this point, is primarily interested in simple step-by-step techniques for tracking AMSAT-OSCAR and Russian RS satellites can skip directly to Chapter II. Many of the key equations presented in this chapter are derived in Chapter VI, section 2.

1.1 BACKGROUND

The satellite-orbit problem (determining the position of a satellite as a function of time and/or finding its path in space) is essentially the same whether we are studying the motion of the planets around the sun, the moon around the earth, or an artificial satellite revolving around either. The similarity is a consequence of the nature of the forces on the orbiting body when no propulsion systems are in use. Our understanding of our own solar system is based on the work of Kepler who, in the early 17th century, using the extensive and highly accurate data on planetary motion compiled by Tycho Brahe, discovered some remarkable properties of planetary motion:

I. Each planet moves around the sun in an ellipse, with the sun at one focus (motion lies in a plane);

II. The radius vector from the sun to a planet sweeps out equal areas in equal intervals of time;

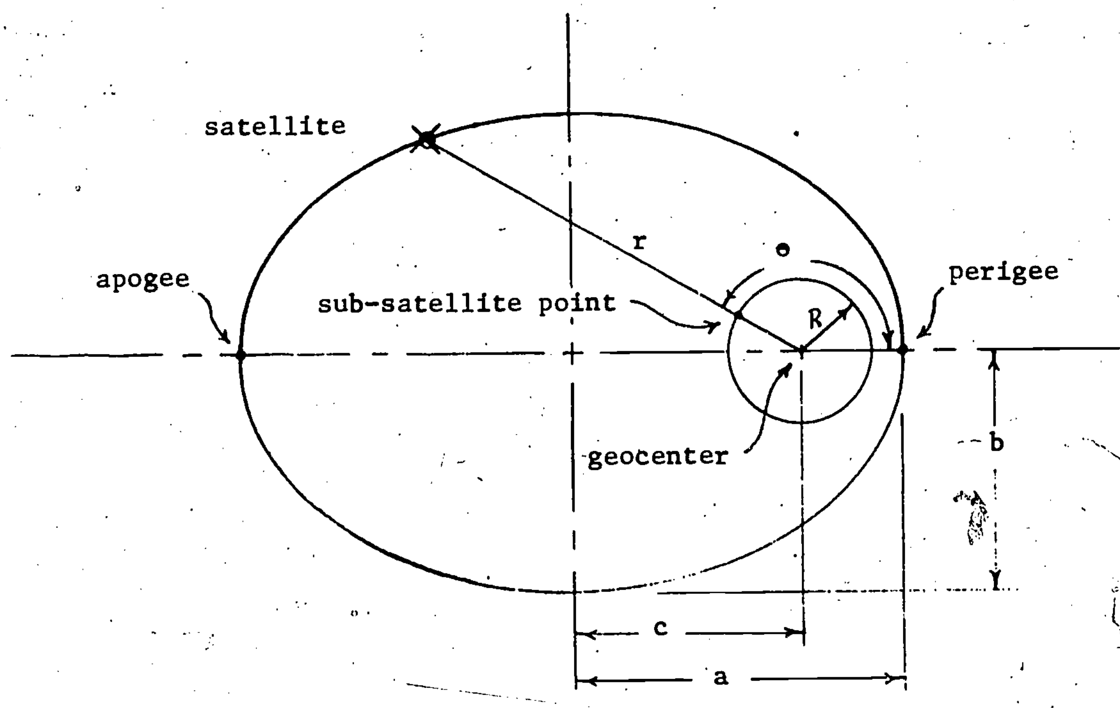
III. The ratio of the square of the period (T) to the cube of the semimajor axis (a) is the same for all planets in our solar system.

These three properties, known as Kepler's laws, summarize observations; they say nothing about the forces governing planetary motion. It remained for Newton to deduce the characteristics of the force that would yield Kepler's laws. Newton showed that the second law would result if the planets were being acted on by an attractive force always directed at a fixed central point -- the sun (central force). To satisfy the first law this force would have to vary as the inverse square of the distance between planet and sun ($1/r^2$). Finally, if Kepler's third law was to hold, the force would have to be proportional to the mass of the planet. Actually, Newton went a lot further; he assumed that not only does the sun attract the planets in this manner, but that every mass (m_1) attracts every other mass (m_2) with a force directed along the line joining the two masses and having a magnitude (F) given by

$$F = \frac{Gm_1m_2}{r^2} \qquad \text{(Universal Law of Gravitation)} \tag{1.1}$$

where G is the Universal Gravitational Constant.

Figure 1.1, showing a typical earth satellite orbit, establishes the terminology that we will be using. As per Kepler's observations, the orbit is shown as an ellipse and confined to a plane. It takes two independent parameters to describe the size and shape of an ellipse. For example, one could specify: major and minor axes, apogee and perigee distance, apogee distance and eccentricity, etc.



a semimajor axis
b semiminor axis
e eccentricity = $[1-(b/a)^2]^{.5}$; $0 \leq e < 1$; (circle: e = 0)
c distance between center of ellipse and focal point = ae

R mean radius of earth
r,θ polar coordinates of satellite; θ (the true anomaly) is measured from perigee

geocenter: position of center of mass of earth
sub-satellite point: point where r intersects surface of earth
altitude (height): h = r-R
apogee: point on orbital ellipse where r is a maximum
perigee: point on orbital ellipse where r is a minimum

r_a apogee distance = a(1+e)
h_a apogee altitude = $r_a - R$
r_p perigee distance = a(1-e)
h_p perigee altitude = $r_p - R$

Figure 1.1. Geometry of the orbital ellipse for earth satellite.

Problem 1.1.

A satellite is in an earth orbit with an apogee distance (r_a) of 6.64R and a perigee distance (r_p) of 1.23R. Specify the orbit in terms of the semi-major axis (a) and eccentricity (e).

Answer: Referring to Figure 1.1, $r_a + r_p = 2a$. From $r_a = a(1+e)$ and $r_p = a(1-e)$ we obtain $r_a - r_p = 2ae$. Therefore, $e = (r_a - r_p)/(r_a + r_p)$. The major axis of the specified orbit is 3.93R and the eccentricity is .668.

If the major and minor axes of an ellipse are equal, the ellipse is a circle. Since the circular orbit (e=0) is just a special case of the elliptical orbit, the most general approach to the satellite-orbit problem would be to study elliptical orbits. However, we sometimes work with circular orbits directly when this simplifies the analysis. At times the term ellipse is used when the circular orbit is being excluded but this ambiguity should not be troublesome as the meaning is usually clear from the context.

Our approach to the satellite-orbit problem involves a number of discrete steps:

1. The path of the satellite in space is determined,
2. The path of the sub-satellite point on the surface of a static (non-rotating) earth is computed,
3. The earth's motion about its axis is taken into account,
4. Effects of the earth's motion about the sun are considered,
5. From any point on the surface of the earth, the distance to the satellite and the azimuth and elevation angles of the satellite are computed at any point in time.

1.2 SATELLITE PATH IN SPACE

To determine the path of a satellite in space (1) a number of simplifying assumptions are made about the forces acting on the satellite and other aspects of the problem, care being taken to keep the most prominent determinants of the motion intact; (2) the simplified model is solved; (3) corrections to the solution, accounting for the initial simplifications, are added. We start by listing the assumptions usually employed:

1. The earth is considered stationary and a coordinate system is chosen with an origin at its center of mass (geocenter);
2. The earth and satellite are represented by point masses, M and m, located at their respective centers of mass;
3. The satellite is subject to only one force, an attractive force directed at the geocenter whose magnitude varies as the square of the inverse distance separating satellite and geocenter ($1/r^2$).

Detailed solutions to this problem are given in many introductory physics texts [1,2]. Some of the important results follow.

1. Certain initial conditions, namely the velocity and position of the satellite at burnout (the instant the propulsion system is turned off), produce elliptical orbits ($0 \leq e < 1$). Other initial conditions produce hyperbolic ($e > 1$) or parabolic ($e = 1$) orbits which we will not be discussing.
2. For a certain subset of the set of initial conditions resulting in elliptical orbits, the ellipse degenerates (simplifies) into a circle ($e = 0$).
3. The satellite orbit lies in a plane which always contains the geocenter. The orientation of this plane remains fixed in space after being determined by the initial conditions.
4. The period (T) of the satellite and the semimajor axis (a) of the orbit are related by the equation

   $$T^2 = \frac{4\pi^2}{GM} a^3 \qquad (1.2)$$

   where M is the mass of the earth and G is the Universal Gravitational Constant. Note that the period of an artificial satellite orbiting the earth depends only on the semimajor axis of its orbit.
5. The magnitude of the satellites total velocity (v) is given by

   $$v^2 = GM \left(\frac{2}{r} - \frac{1}{a} \right) \qquad (1.3)$$

   where r is the distance between satellite and geocenter. Note that the range of velocities is bounded -- the maximum velocity occurs at perigee (when the satellite is closest to earth) and the minimum velocity occurs at apogee (when the satellite is farthest from earth).

Problem 1.2.

Consider circular orbits.

1. Plot T (period in minutes) vs. altitude (from 0 to 50,000 km). Don't forget that altitude is measured from the surface of the earth while "a" is measured from the geocenter. How would the interpretation of this plot be changed for an elliptical orbit?

2. Plot v (speed in m/s) vs. altitude (from 0 to 50,000 km). How would the interpretation of this plot be changed for elliptical orbits?

Answer: See Figure 1.2.

1. Using the appropriate constants, Eq. 1.2 can be written:
$T = 5.24 \times 10^{-9} a^{3/2}$ *(T in minutes, a in meters). A plot of this relation gives the period of elliptical orbits having the indicated semimajor axis. For circular orbits the semimajor axis is equal to the radius.*

2. For circular orbits, $r = a$, Eq. 1.3 simplifies to $v^2 = GM/r$ (v is a constant). The magnitude of the velocity of a satellite in an elliptical orbit varies. Figure 1.2 yields the velocity at the two points on the path where $r = a$. Eq. 1.3 will give the velocity at any point on the path.

Some corrections to the satellite-orbit problem which can be taken into account are:

1. In the two body (earth, satellite) problem, the stationary point is the center of mass of the system, not the geocenter. The mass of the earth is so much greater than the mass of artificial satellites that this correction is negligible for the applications discussed in this text.

2. Treating the earth as a point mass involves the assumption that the shape and the distribution of mass in the earth are spherically symmetrical. Taking into account the actual asymmetry of the earth (most notably the bulge at the equator) produces additional central force terms acting on the satellite. These forces vary as higher orders of 1/r (i.e., $1/r^3$, $1/r^4$, etc.). The main effects of these additional terms can be visualized as (1) causing the major axis of the orbital ellipse to rotate slowly (precess) in the plane of the satellite and (2) causing the plane of the satellite to rotate (regress) about the rotational axis of the earth. These effects are readily observed and are discussed in several places in this text.

3. The satellite is affected by a number of other forces in addition to gravitational attraction by earth. For example: gravitational

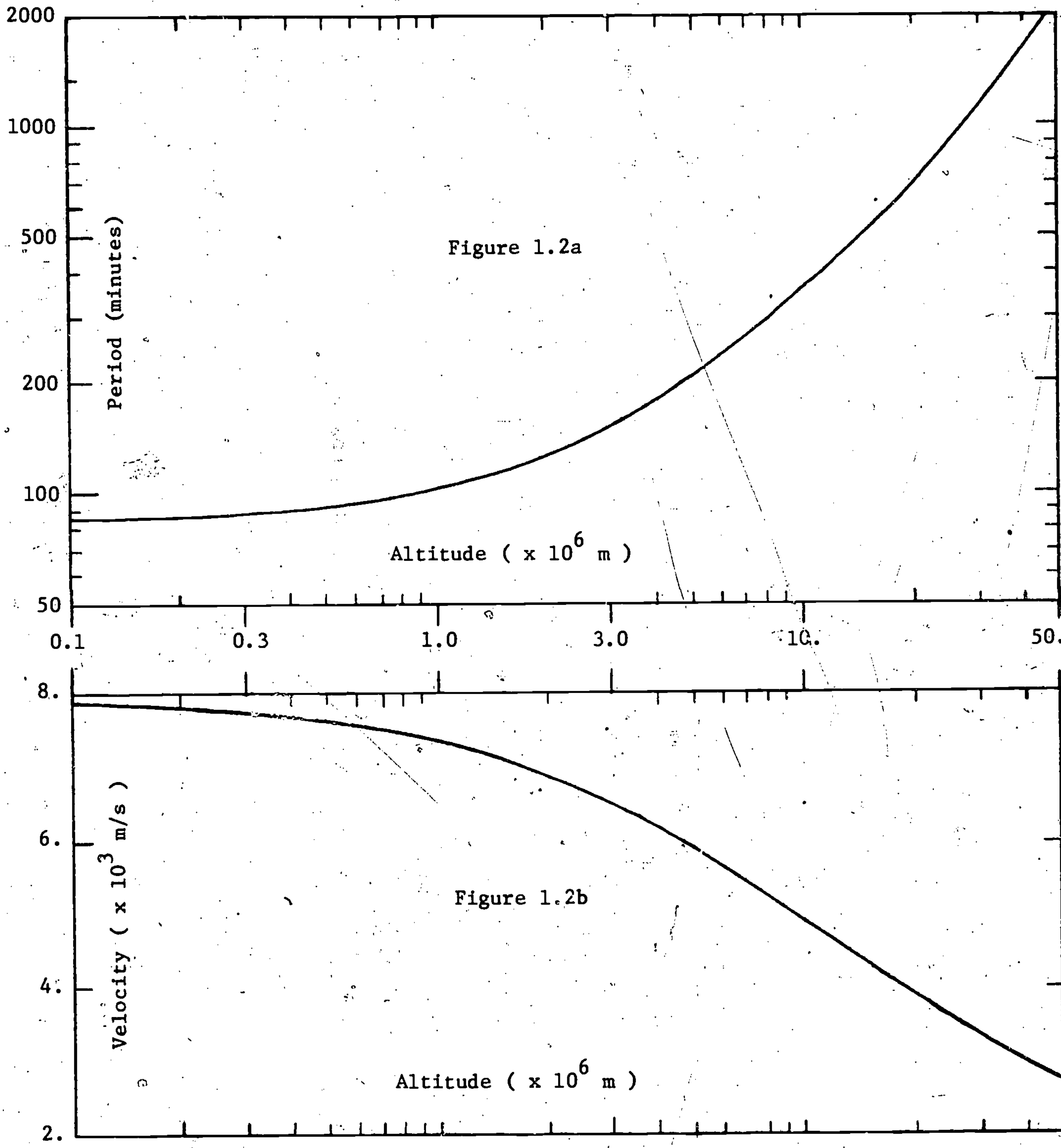


Figure 1.2. a. Satellite period vs. altitude for circular orbit.
b. Satellite velocity vs. altitude for circular orbit.

attraction by the sun, moom, and other planets; friction due to the atmosphere (atmospheric drag), radiation pressure due to the sun, etc. We turn now to the effects of some of these forces.

At low altitudes the most prominent perturbation is atmospheric drag. Let us consider the effect of drag in two cases: (1) elliptical orbits with high apogee and low perigee and (2), low altitude circular orbits. In the first case drag acts mainly near perigee, reducing the satellite velocity, and causing the altitude at the following apogee to be lowered (perigee altitude remains nearly constant). Atmospheric drag therefore tends to reduce the eccentricity of elliptical orbits (makes them more circular) by lowering the apogee. In the second case drag is of consequence during the entire orbit. It causes the satellite to spiral in toward the earth with an increasing velocity. A satellite's lifetime in space (before burning up upon reentry) depends upon the initial orbit, the geometry and mass of the spacecraft, and the composition of the earth's ionosphere. The lifetime of satellites similar in geometry and mass to AMSAT-OSCARs 7 and 8 can be roughly estimated from Figure 1.3 [3].

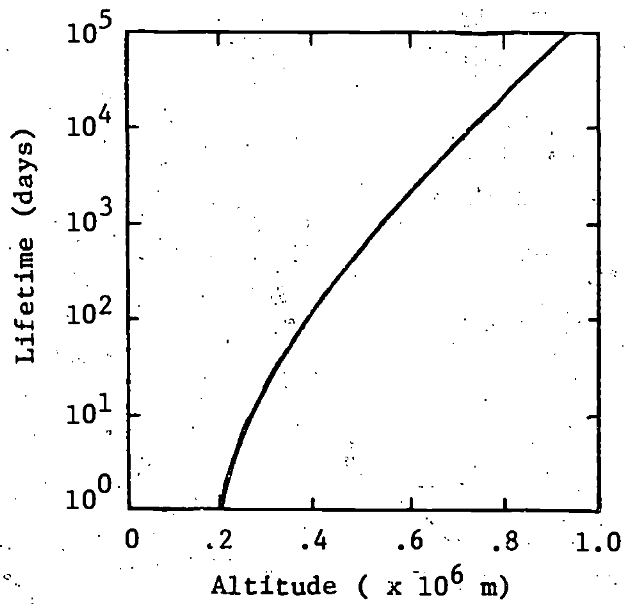


Figure 1.3.

Satellite lifetime for circular orbit and satellite geometry and mass similar to AMSAT-OSCAR 7 and 8.

The altitudes of AMSAT-OSCAR spacecraft are greater than 850 km so lifetime in orbit should be greater than the lifetime of the onboard electronic subsystems.

Effects on the orbit due to gravitational attraction by the sun and moon are most prominent when a satellite's apogee distance is large. The sun and moon will therefore have a significant long-term effect on the orbit of AMSAT Phase III satellites. The casual user need not worry about this problem but AMSAT scientists must investigate this perturbation in detail to insure that the orbit chosen is stable. Instabilities due to resonant perturbations are capable of causing the loss of the satellite within months.

Now that the motion of the satellite in space has been described, we turn to the problem of determining the path of the sub-satellite point on the surface of the earth.

1.3 SUB-SATELLITE PATH ON EARTH (GROUND TRACK)

The rotational axis of the earth (N-S axis) provides a unique reference line through the geocenter which intersects the surface of the earth at two points designated the north (N) and south (S) geographic poles. The intersection of any plane containing the geocenter and the surface of the earth is called a great circle. One such great circle is the equator which is formed from the equatorial plane, the plane containing the geocenter which is perpendicular to the N-S axis. The set of great circles containing the N-S axis are of special interest. Each is divided into two meridians (half circles), connecting north and south poles.

Treating the earth as a sphere, points on the surface are specified by two angular coordinates, latitude and longitude. As an example, the angles used to specify the latitude and longitude of Washington, D.C. are shown in Figure 1.4.

Latitude. Given any point on the surface of the earth, the latitude is determined by (1) drawing a line from the given point to the geocenter, (2) dropping a perpendicular from the given point to the N-S axis, and measuring the included angle. A more colloquial, but equivalent, definition for latitude is: the angle between the line drawn from the given point to the geocenter and the equatorial plane. To prevent ambiguity a suffix is appended to the latitude to indicate whether the given point is in the northern or southern hemispheres. The set of all points having a given latitude lies on a plane perpendicular to the N-S axis. The set of all points having the latitude 0°, known as the equator, has already been mentioned.

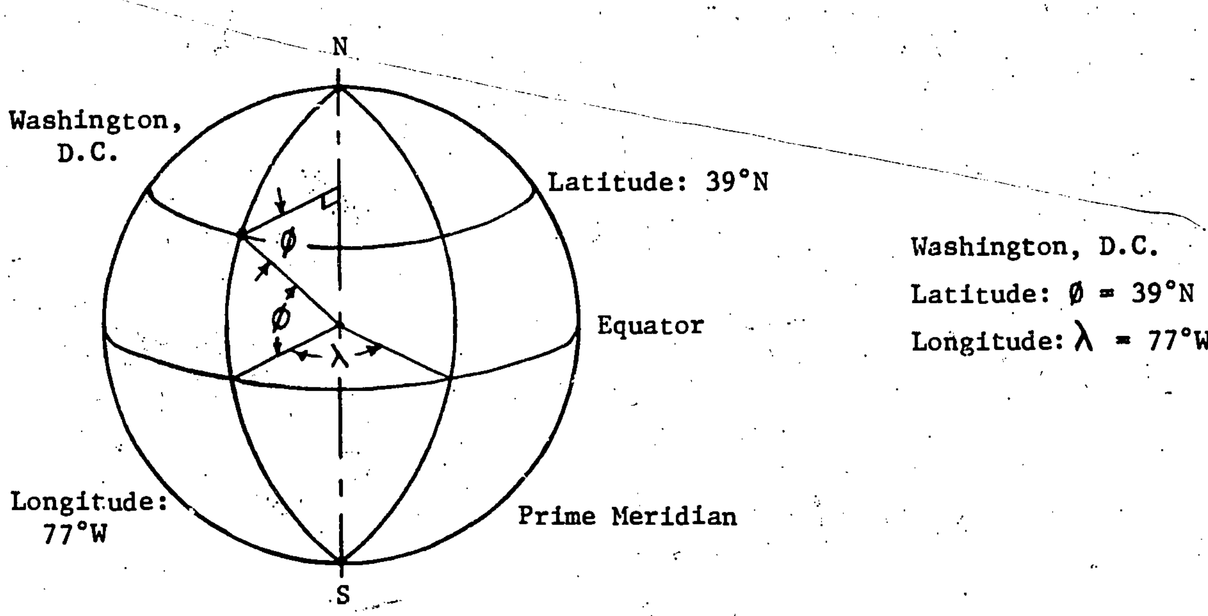


Figure 1.4. Location of Washington, D.C.

Longitude. All points on a given meridian are assigned the same longitude. To specify longitude one chooses a reference or prime meridian (the original site of the Royal Greenwich Observatory in England is used). The longitude of a given point is then obtained by measuring the angle between the lines joining the geocenter to (1) the point where the equator and prime meridian intersect and (2) the point where the equator and the meridian containing the given point intersect. For convenience, longitude is given a suffix, E or W, to designate whether one is measuring the angle east or west of the prime meridian.

As the earth rotates on its axis and revolves around the sun, the orientation of both the plane containing the equator (equatorial plane) and, to a first approximation, the plane containing the satellite (orbital plane) remain fixed in space (fixed relative to the "fixed stars"). Figure 1.5 shows how the orbital plane and equatorial plane are related. The line of intersection of the two planes is called the line of nodes. The relative orientation of these two planes is very important to satellite users. It is partially specified by giving the inclination. The inclination, (i), is the angle between the line joining the geocenter and north pole and the line through the geocenter perpendicular to the orbital plane (to avoid ambiguity the half line in direction of advance of a right-hand screw following satellite motion is used). An equivalent definition of the inclination -- the angle between the equator and sub-satellite path as the satellite enters the northern hemisphere -- is shown in Figure 1.6.

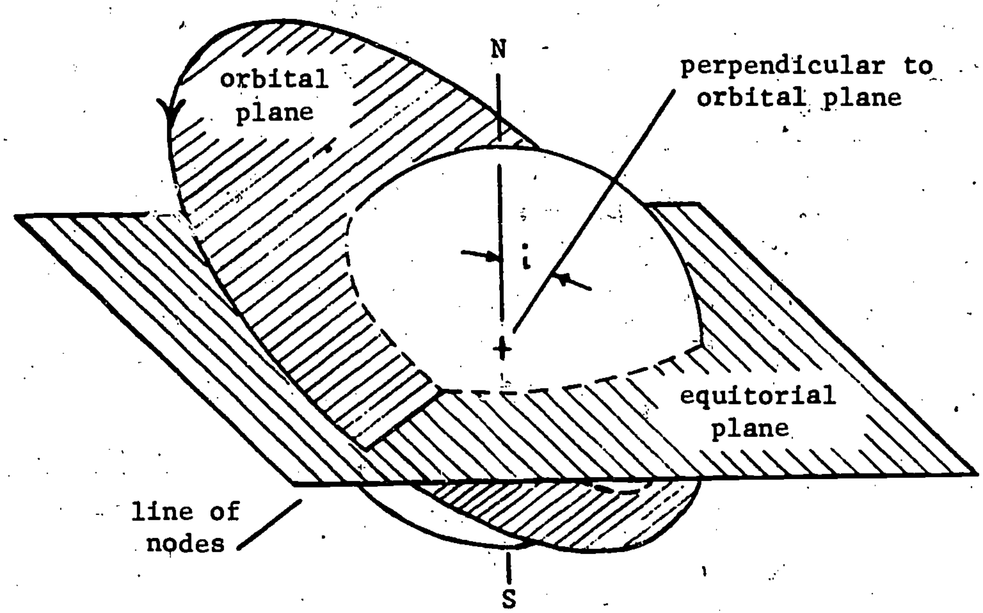


Figure 1.5. Orbital plane of satellite and equitorial plane of earth.

The inclination can vary from 0° to 180°. To first order, none of the perturbations (corrections to the simplified model) cause the inclination to change, but higher order effects result in small oscillations about an

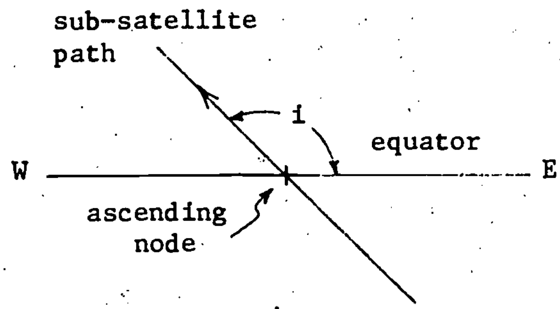


Figure 1.6. Inclination angle (i) of orbital plane.

average value. Diagrams showing orbits having inclinations of 0°, 90°, and 135° are shown in Figure 1.7. A quick analysis of these three cases yields the following information. When the inclination is 0°, the satellite will always be directly above the equator. When the inclination is 90°, the satellite passes over the north pole and over the south pole once each orbit and over the equator twice, once heading north and once heading south.

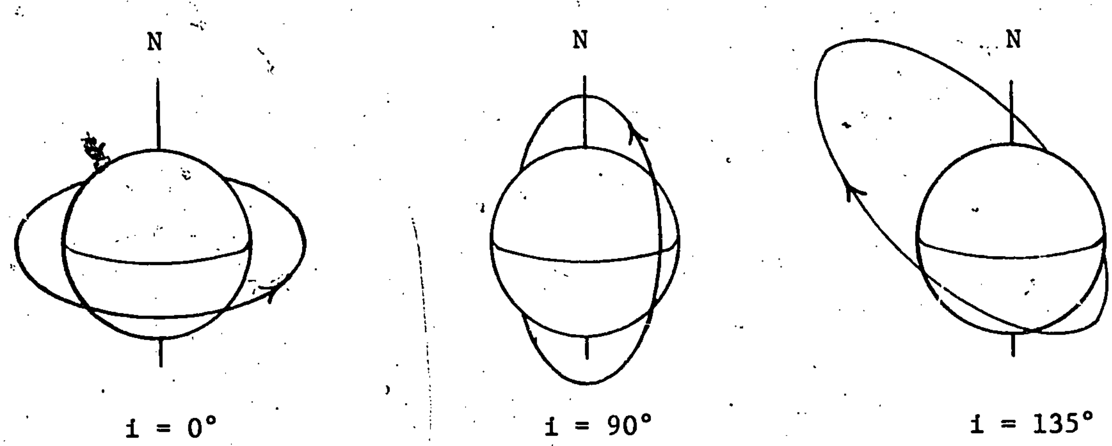


Figure 1.7. Satellite orbits with inclination angles of 0°, 90°, 135°.

Orbits are sometines classified as being polar (near polar) when their inclination is 90° (near 90°) or equatorial (near equatorial) when their inclination is 0° (near 0° or 180°). Finally, for other values of inclination, 135° for example, we see that the satellite still passes over the equator twice each orbit but it never crosses above the north or south poles. The maximum latitude ($\emptyset_{max}$), north or south, that the sub-satellite point will reach equals: (1) the inclination when the inclination is between 0° and 90°; (2) 180° less the inclination when the inclination is between 90° and 180°. This can be seen from Figure 1.8, it's proved in Problem 1.5.

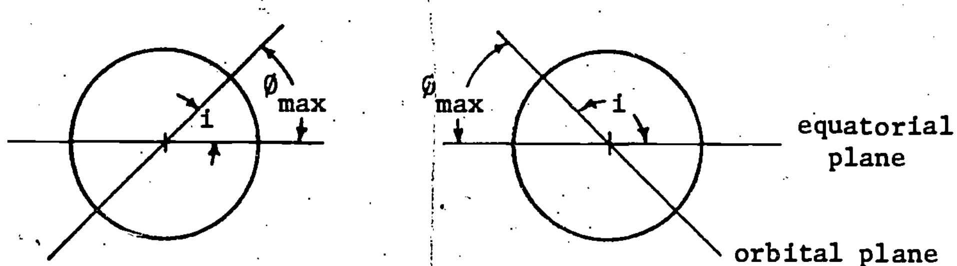


Figure 1.8. Relation between maximum latitude ($\emptyset_{max}$) of sub-satellite point and the inclination angle (i). Cross section taken through geocenter perpendicular to orbital and equatorial planes.

The position of the satellite when it crosses above the equator from southern to northern hemispheres is known as the <u>ascending node</u>. The position of the satellite when it crosses above the equator from northern to southern hemispheres is called the <u>descending node</u>. Referring to Figure 1.5 the nodes are the two points where the satellite orbit intersects the equatorial plane. The <u>line of nodes</u> (the line of intersection of the orbital and equatorial planes) contains these two points. Every satellite orbit, circular or elliptical, for which the inclination is non-zero, has two nodes. Later, taking into account the rotation of the earth, we'll see that there is one very special case for which the longitudes of the two nodes coincide.

When working with elliptical orbits, it is important to specify an additional parameter which locates the perigee in the orbital plane. This is usually done in terms of the angle -- called the <u>argument of perigee</u> ($\omega_{\bullet}$) -- between the lines joining the geocenter to (1) the ascending node and (2) the perigee (see Figure 1.12). Since the argument of perigee changes with time it is computed between an ascending node and the following perigee and labeled according to the time of node. When discussing circular orbits, such as those of AMSAT-OSCAR 7 and 8, this parameter can be ignored. However, future OSCAR satellites may be placed in highly elliptical orbits making it necessary to take the argument of perigee into account. Elliptical orbits are discussed in section 1.5.

In the simplified case of a non-rotating earth if we knew that a satellite in a circular orbit having a period of 115 minutes had an ascending node at, for example, longitude 66° W at 09:09 local time, it would again have ascending nodes every 115 minutes at 66° W (i.e., at 11:04, 12:59, 14:54, etc.). If the same satellite passed over Washington, D.C. at 09:22 (13 minutes after the ascending node), it would again pass over Washington, D.C. at 11:17, 13:12, 15:07, etc. (i.e. 13 minutes after each node or every 115 minutes).

Now, to take the rotation of the earth into account, we recall that the orientation of the orbital plane of the satellite and the equatorial plane of earth are fixed in space, it is just our vantage point that is changing. The earth rotates approximately 360° each day, or 15° per hour (counter-clockwise as seen by an observer above the north pole). Referring to our previous example, in 115 minutes the earth would rotate about 29°. If an ascending node occurred at 09:09 at 66° W, ascending nodes would again occur at 11:04, 12:59, 14:54, etc. but they would not all be at 66° W. Since the earth would have rotated about 29° during each complete orbit, each successive node would be 29° farther west, i.e. the ascending nodes in our example, based on a node at (09:09, 66° W) would be (11:04, 95° W), (12:59, 124° W), (14:54, 153° W), etc. Continuing with the example, 13 minutes after each node, the satellite would reach latitude 38° N (the latitude of Washington, D.C.) but on each successive orbit it would be about 29° farther west. Referring to a map, at 11:17 the sub-satellite point would actually be near Denver; and at 13:12 about 1,000 miles due west of San Francisco.

The angular rotation of the earth during one complete orbit (the increment) is an important parameter to satellite users. The increment, (I), given in degrees west per orbit, equals the change in longitude between two successive ascending nodes. The increment can be estimated from the satellite's orbital period (T)

$$(1.4) \qquad I[^\circ\ \text{west/orbit}] = \frac{T\ [\text{min/orbit}]}{1440\ [\text{min/day}]}\ 360\ [^\circ\ \text{west/day}] = T/4$$

where units are specified within the brackets. Consider the example we used earlier where T = 115 minutes: I = 115/4 = 28.75 °west/orbit.

This estimate of the increment has neglected two factors -- (1) the earth's rotation about the sun and (2) the regression of the orbital plane caused by the departure of the earth from spherical symmetry -- which can be important when following a satellite over a number of revolutions. We now look at how these two factors change the approximation to the increment given by Eq. 1.4.

The motion of the earth about the sun causes the <u>solar day</u> (the time for one revolution of the earth with respect to the sun) and the <u>sidereal day</u> (the time for one revolution of the earth with respect to the fixed stars) to differ. The solar day contains 1440 minutes (by definition), the sidereal day 1,436.07 minutes. Eq. 1.4 should, therefore, be modified by replacing 1,440 minutes by 1,436.07 minutes. This amounts to a correction of roughly .27%. Taking the motion of the earth around the sun into account, therefore increases the estimate of the increment given in Eq. 1.4 by about .27%.

Finally, the regression of the orbital plane about the rotational axis of the earth causes another correction to the increment. The magnitude of this effect on circular orbits ($\frac{d\Omega}{dt}$ in °east/day) is given, to first order, by

(1.5) $$\frac{d\Omega}{dt}[°\text{east/day}] = -9.95 \left(\frac{R_{eq}}{r}\right)^{3.5} \cos(i) \qquad \left(\begin{array}{c}\text{circular earth}\\ \text{orbits only}\end{array}\right)$$

where R_{eq}, the equatorial radius of the earth, is 6,382 km [4].

Problem 1.3.

1. Find the regression of AMSAT-OSCAR 7 (circular orbit with $i = 101.7^0$, altitude = 1.45×10^6 m).

2. Estimate the error which would result from using Eq. 1.4 without any corrections in this case.

Answer:

1. Altitude = 1.45×10^6 m, $r = 7.82 \times 10^6$ m, $\frac{d\Omega}{dt} = .99^0$ east/day. A positive regression must be subtracted from Eq. 1.4.

2. The two corrections are nearly equal in magnitude and opposite in sense. In this case using Eq. 1.4 without any correction would result in an error of less than .01%.

When the two corrections to the increment which we have been discussing "cancel out", as in Problem 1.3, we call the orbit "sun synchronous". Sun synchronous orbits have a property which is often convenient for many missions -- if the satellite passes within range between, for example, 9 am and 11 am local standard time soon after launch it will continue to pass nearby between 9 am and 11 am for year after year. Sun synchronous orbits also maximize exposure to sunlight (when the point of injection into orbit is chosen properly) -- a factor which is important when solar cells are being used to power the satellite.

In practice, when a satellite is first launched Eq. 1.4 can be used to estimate increments. The true increment is usually obtained by averaging observations over a long time interval, not by computing corrections.

Once the time and longitude of one ascending node are known, all future ascending nodes (time and longitude) can be computed using the period and increment. However, errors will accumulate and the accuracy of any predictions is only as good as one's data. Restricting our attention to circular orbits for a moment, a descending node always occurs T/2 minutes after each ascending node and at a longitude 180° + (increment/2) further west of the ascending node.

If the satellite period is an exact divisor of 24 hours, the satellite will pass over the same position on earth at the same time each day. For example, a period of 8 hours (480 Minutes) yields an increment of 120°. 24 hours (3 complete revolutions) after passing over a given point on the surface of the earth, the satellite will be at the same point in its orbit and the earth will have rotated exactly 360° placing the satellite back over the initial point. Satellites having a period of 24 hours are of special interest. From Eq. 1.2 we find that such a period occurs for satellites in circular orbits having a radius of about 4.21×10^7 m (synchronous orbit) and for satellites in elliptical orbits having a semimajor axis of the same amount. A 24 hour satellite placed in a circular equatorial orbit (0° inclination), will appear to remain stationary (stationary orbit) over a particular site on the equator. If a satellite with a 24 hour period is placed in a circular orbit with an inclination other than 0°, it will have two nodes. For such a satellite the positions of the sub-satellite points at the ascending and descending nodes will coincide and the sub-satellite path (the ground track) will look like a symmetrical figure eight. See Figure 1.9.

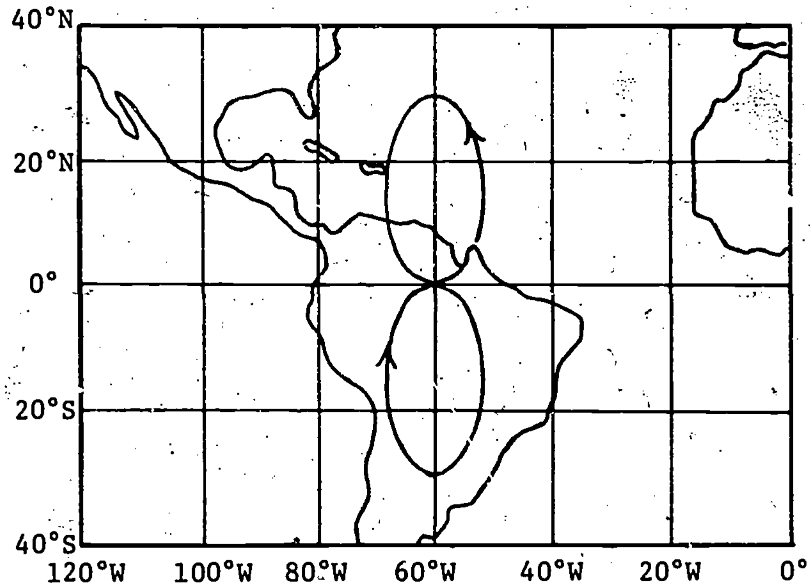


Figure 1.9. Subsatellite path for satellite with 24 hour period, circular orbit, and inclination of 30°.

Note that it is impossible to have a stationary satellite above Washington, D.C. or any other city not on the equator. This results from the basic fact that the satellite orbital plane must contain the geocenter.

We now turn to the problem of determining the position of the sub-satellite point at time t after an ascending node at time t_o. The problem is mainly one of spherical trigonometry. Full details of the solution for elliptical and circular orbits are presented in Chapter VI, section 2.

The latitude, $\phi(t)$, and longitude, $\lambda(t)$, of the sub-satellite point as a function of time are described by Eqs. 1.6 and 1.7 where the sign convention adopted designates: North latitudes and East longitudes as positive;

positive; South latitudes and West longitudes as negative -- angles are given in degrees; time in minutes.

- i = inclination of orbit
- T = period
- $\mathcal{t}$ = clock time
- t_o = time at ascending node
- t = elapsed time from ascending node = $\mathcal{t} - t_o$
- $\omega(t)$ = angular position of satellite in orbital plane measured from ascending node
- λ_o = longitude of sub-satellite point at ascending node

(1.6) $\phi(t) = \arcsin(\sin\omega(t)\sin i)$ (Latitude)

(1.7) $\lambda(t) = \lambda_o - \frac{t}{4} - (-1)^{n_2+n_3} \arccos\left(\frac{\cos\omega(t)}{\cos\phi(t)}\right)$ (Longitude)

$$n_2 = \begin{Bmatrix} 0 & 90° \leq i \leq 180° \\ 1 & 0° \leq i < 90° \end{Bmatrix}; \quad n_3 = \begin{Bmatrix} 0 & \phi(t) \geq 0° & (\text{Northern Hemisphere}) \\ 1 & \phi(t) < 0° & (\text{Southern Hemisphere}) \end{Bmatrix}$$

In Eq. 1.7 the $t/4$ term results from the rotation of the earth about its axis at the rate of .25 degrees per minute while the arccos term results from a static problem in spherical trigonometry. The n_2 and n_3 terms incorporate our sign convention. The angular velocity of a satellite in a circular orbit is constant so $\omega(t) = kt$. To obtain k consider one complete orbit: $\omega = 360°$, $t = T$. Therefore, $k = 360°/T$ and the $\omega(t)$ term in Eqs. 1.6 and 1.7 can be replaced by

(1.8) $\omega(t) = 360°t/T$ (circular orbits only).

Eqs. 1.6, 1.7, and 1.8 enable us to plot ground tracks for satellites in circular orbits. In chapter II we will present a number of fast and simple methods for applying these equations. Note that the ground track of a satellite in a circular orbit can be obtained if one knows four parameters: i, t_o, λ_o, and T. There are other sets of four independent orbital parameters (often called orbital elements) which can be used for tracking [5]. This particular set was chosen because it is convenient for the applications discussed in this book. In section 1.5 we will generalize this discussion to elliptical orbits. Eqs. 1.6 and 1.7 will still apply. However, Eq. 1.8 will need to be revised.

Problem 1.4.
Use Eqs. 1.6, 1.7 and 1.8 to prepare a "tracking table" for OSCAR 7 showing the ground track latitude and longitude every four minutes starting at the ascending node and continuing until the satellite enters the southern hemisphere. Let i = 102°, T = 115 minutes, $\mathcal{t}_o$ = 0, and λ_o = 0.

Answer: See Table 2.4.

Problem 1.5.
Prove that the northernmost latitude that a satellite can reach is given by:
1. *the inclination (i) when i is less than 90°*
2. *180° - i when i is between 90° and 180°.*

Answer: Starting from Eq. 1.6 we compute $d\phi/d\omega$ and set it equal to zero:

$$\frac{d\phi}{d\omega} = \sin(i)\,\frac{\cos(\omega)}{\cos(\phi)} = 0.$$

For $i \neq 0$ this equation will only be satisfied when $\cos\omega = 0$, in which case $\sin\phi_{max} = \pm \sin i$.

1.4 AZIMUTH, ELEVATION, COVERAGE

A ground station using a directional antenna needs to know where to point it. The most widely used coordinates for this purpose are azimuth (direction in a plane tangent to the earth at the ground station measured with respect to true north) and elevation (the angle above this plane). The physical situation suggests a unique solution if the positions (latitude and longitude) of the ground station and sub-satellite point, and the height of the satellite, are known. The problem can be divided into two parts: (1) a problem in spherical trigonometry on the surface of the earth of finding the azimuth from one point to a second and the surface distance between the two points and (2) a problem in plane trigonometry of finding the elevation angle from ground station to satellite and line of sight distance (slant range) between ground station and satellite. Part (1) is a standard problem in spherical trigonometry and navigation. Although we will not solve this problem, the solutions are included here (Eqs. 1.9 and 1.10) for reference.

ϕ_1, λ_1	latitude and longitude of point 1 (ground station)
ϕ_2, λ_2	latitude and longitude of point 2 (sub-satellite point)
γ	surface distance in degrees of arc (1 degree of arc corresponds to 1.112×10^5 m on surface of earth)
s	surface distance in meters; $s = R\gamma$ (γ in radians)
β	azimuth (east or west of north) of point 2 as seen from point 1.

(1.9) $$\cos\gamma = \sin\phi_1 \sin\phi_2 + \cos\phi_1 \cos\phi_2 \cos(\lambda_1 - \lambda_2)$$

(1.10) $$\sin\beta = \cos\phi_2 \csc\gamma \sin(\lambda_1 - \lambda_2)$$

Approximate solutions for azimuth and terrestrial surface distance having the accuracy we need can often be read off a globe, or an azimuthal equidistant projection map if one is available for the ground station position. Part (2) is solved in section 6.2. The results are usually presented in terms of surface distance (s) in meters between sub-satellite point and ground station

$$\tan\epsilon = \frac{(R+h)\cos(s/R) - R}{(R+h)\sin(s/R)} \qquad (1.11) \quad \text{(elevation angle)}$$

$$\ell = [(R+h)^2 + R^2 - 2R(R+h)\cos(s/R)]^{1/2} \qquad (1.12a) \quad \text{(slant range)}$$

where the following notation has been used: R = radius of earth, h = instantaneous altitude of satellite, ℓ = line-of-sight distance between satellite and ground station (slant range), ϵ = elevation angle, and s/R is in radians. Slant range can also be expressed in terms of the latitude and longitude of the ground station and the sub-satellite point.

$$\ell = [(R+h)^2 + R^2 - 2R(R+h)[\sin\phi_1 \sin\phi_2 + \cos\phi_1 \cos\phi_2 \cos(\lambda_1-\lambda_2)]]^{1/2} \qquad (1.12b)$$

Note that elevation angle and slant range depend only on the height of the satellite and the surface distance between sub-satellite point and ground station. Eqs. 1.11 and 1.12 are valid for elliptical as well as circular or orbits.

Problem 1.6.
Use Eq. 1.11 and Eq. 1.12 to plot (1) elevation angle vs. surface distance and (2), line-of-sight distance vs. surface distance for a satellite in a circular orbit at a height of 1.45 x 10^6 m (AMSAT-OSCAR 7).

Answer: See Figure 2.2 for (1).

We turn now to the question of coverage -- when will a ground station be able to hear radio signals from the satellite? Because the radio frequencies used in conjunction with most satellites normally propagate over line-of-sight paths, we will consider a communications satellite to be within range whenever the elevation angle at the ground station is greater than zero. However, depending upon the actual propagation conditions, communication could begin when the satellite is below the local horizon, or communication might not be possible until the satellite is well above the local horizon.

The locus of all lines through the satellite and tangent to the earth at a specific instant of time forms a cone. The intersection of this cone with the surface of the earth is a circle whose center lies on the line through the satellite and sub-satellite point. Any ground station inside the circle has access to the satellite. Any two suitably equipped ground stations inside the circle can communicate via the satellite. The maximum terrestrial distance (between ground station and sub-satellite point) at which one can hear signals from the satellite is s_o (see Figure 1.10). The maximum surface distance over which communication is possible is $2s_o$ (see Figure 1.10, stations B and C, for example). Since line AC is tangent to the earth, triangle AOC is a right triangle, $\cos\beta = R/(R+h)$, and s_o is given by $R\beta$. Therefore

$$2s_o = 2R \arccos[R/(R+h)] \qquad (1.13) \quad \text{(maximum communication distance)}$$

Eq. 1.13 is plotted in Figure 1.11.

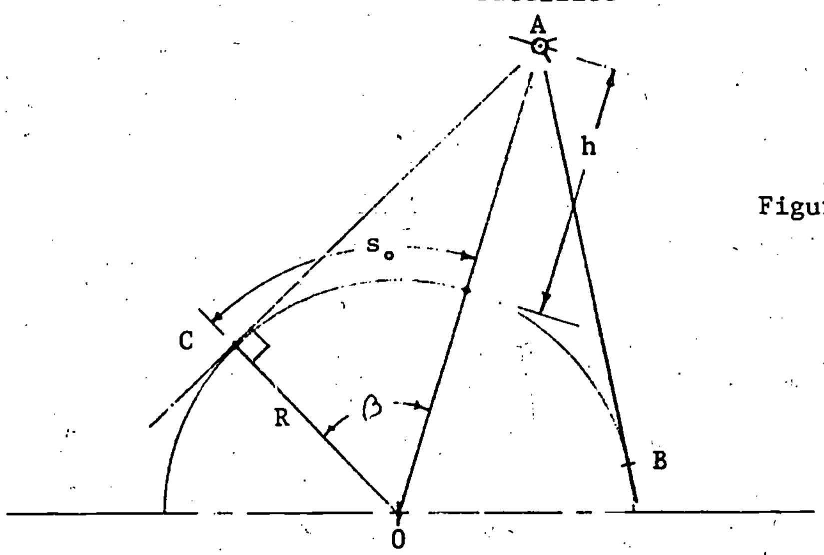


Figure 1.10. Satellite coverage cone.

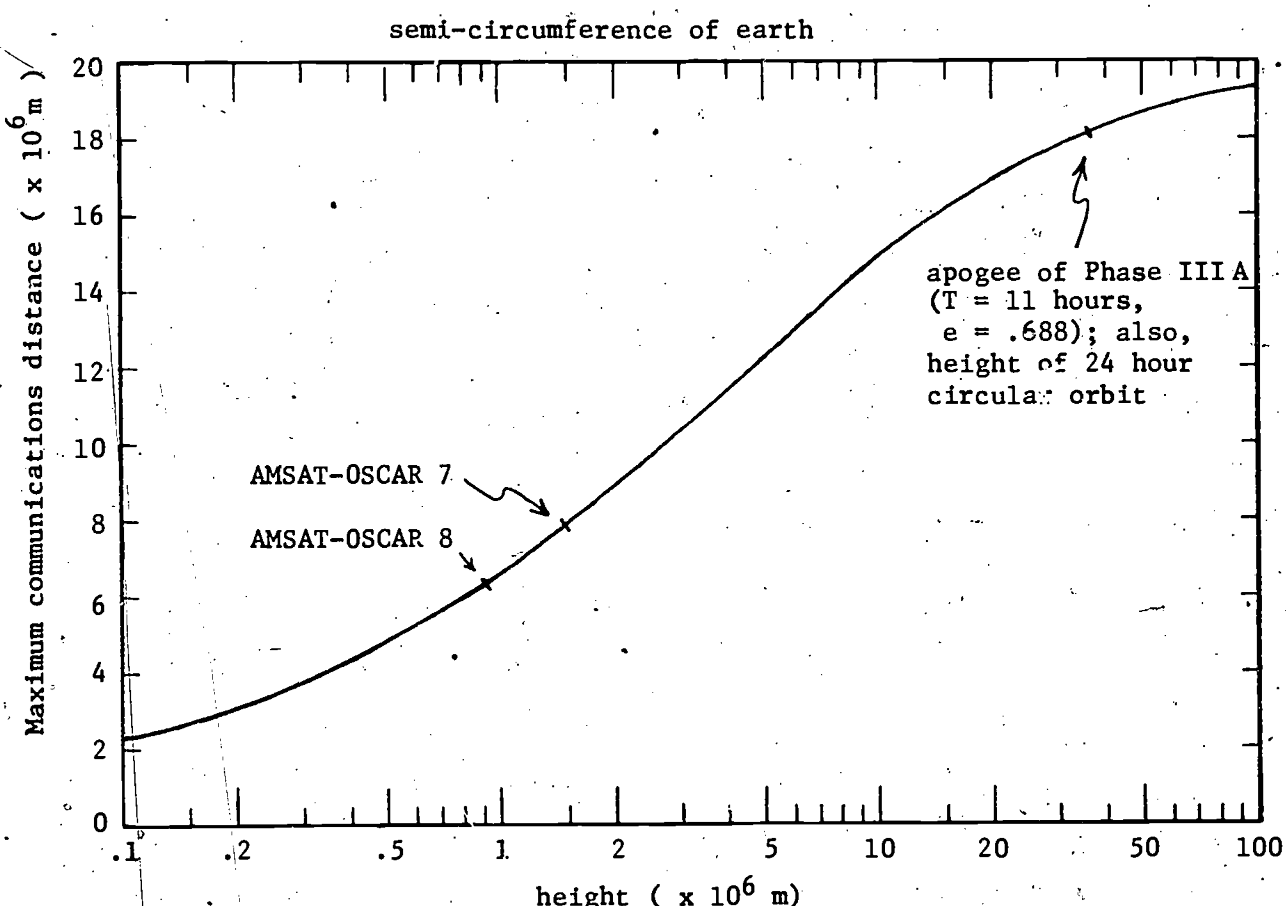


Figure 1.11. Maximum communications distance vs. instantaneous satellite altitude.

1.5 ELLIPTICAL ORBITS: SELECTED TOPICS

In this section we will discuss a number of topics related to tracking satellites in elliptical orbits. The topics covered include:

1. Motion of satellite in the orbital plane,
2. Motion of the orbital plane about the earth,
3. The ground track,
4. Azimuth, Elevation, Coverage.

Since the objective of this chapter is to provide an overview we will primarily point out important considerations and summarize results. Relevant derivations have been included in Chapter VI, section 2. Simple mechanical tracking methods based on the equations in this section are presented at the end of Chapter II.

Motion of satellite in the orbital plane. The formula for an ellipse in polar coordinates is (see Figure 1.1)

(1.14) $$r(\theta) = \frac{a(1-e^2)}{1 + e\cos(\theta)}$$ (r, θ are polar coordinates of satellite with θ measured from perigee)

where a is the semi-major axis and e is the eccentricity of the ellipse. Using Kepler's third law, we can obtain the relation between θ and time since perigee passage (t) (see Chapter VI, section 2 for derivation)

(1.15) $$t = \frac{T}{2\pi}\left[\, E(\theta) - e \sin E(\theta) \,\right] \; ; \; E(\theta) = \arcsin\left(\frac{(1-e^2)^{.5} \sin\theta}{1 + e\cos\theta}\right).$$

Eqs. 1.14 and 1.15 can be used to locate the satellite in the orbital plane at a specified time. Eq. 1.15 can be regarded as the generalization of Eq. 1.8. The quantity $E(\theta)$ in Eq. 1.15 is known as the eccentric anomaly and it should be expressed in radians for calculations.

When discussing elliptical orbits, one must specify the argument of perigee ω_0 (the angle that locates the perigee in the orbital plane with respect to the line of nodes) -- see Figure 1.12. As mentioned in section 1.2, the argument of perigee is not necessarily constant, the equatorial bulge of the earth causes the perigee to precess in the orbital plane. To first order the precession, in degrees per day, is given by

(1.16) $$\frac{d\omega_0}{dt} = 4.97 \left(\frac{R_{eq}}{a}\right)^{3.5} \frac{(5\cos^2 i - 1)}{(1-e^2)^2}$$ (earth orbits only [6]).

When $i = 63.4°$ the argument of perigee is constant. The position of the perigee rotates in the same direction as the satellite when $i < 63.4°$ and in the opposite direction when $i > 63.4°$. Eq. 1.16 should be used to modify Eq. 1.15. However, as long as the change in ω_0 over an orbit is small, (for our purposes generally less than 1°), it's simpler to treat ω_0 as a constant during the orbit and to increment it at the end of each orbit.

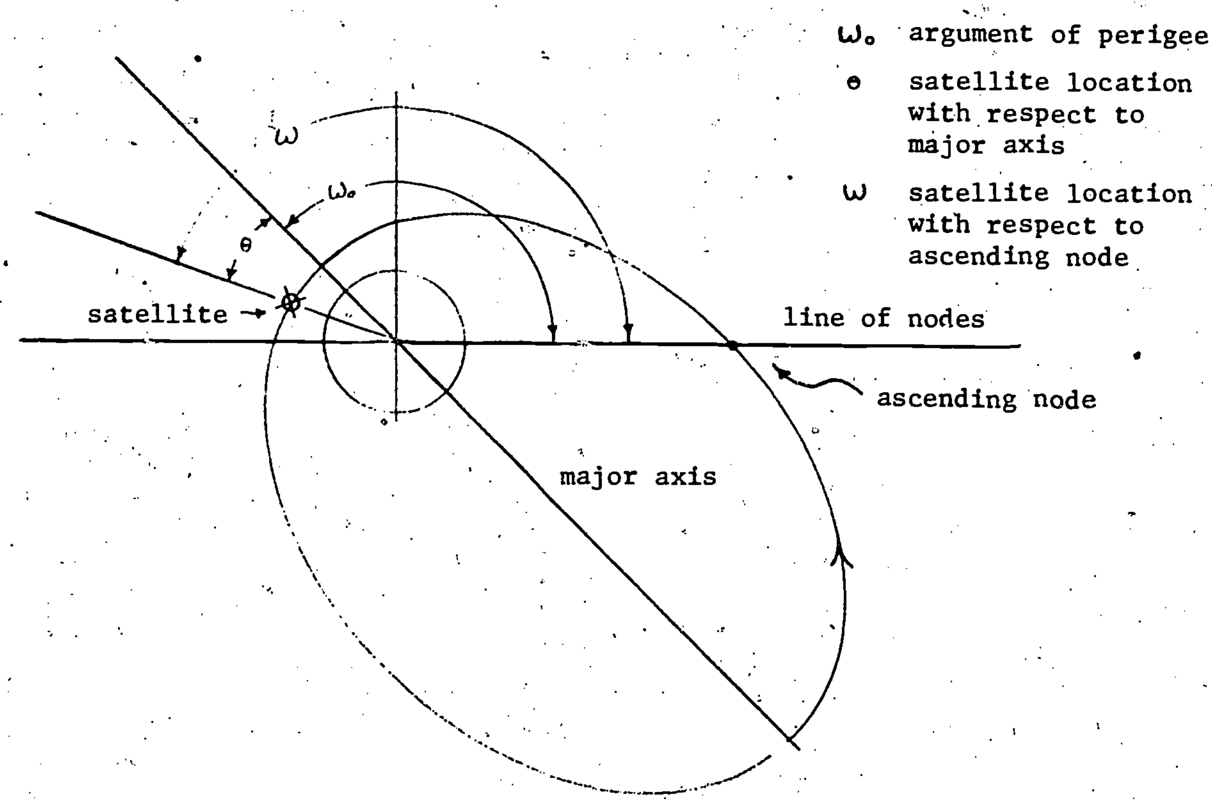


Figure 1.12. Coordinate system in orbital plane.

Problem 1.7.
Find the precession rate of the argument of perigee for a satellite in the following orbit: $i = 101.7^0$, $e = .688$, $a = 3.93R$. If the argument of perigee begins at 315^0, what will it be at the end of 30 days? one year?

Answer: 311.7^0 (30 days), 275^0 (one year).

Problem 1.8. Plot $d\omega_o/dt$ (Eq. 1.16) for $56^0 \leq i \leq 80^0$ using 2 degree increments for each of the following cases:

1. *e = .705, a = 4.17R (period = 12 hours)*
2. *e = .614, a = 3.18R (period = 8 hours)*
3. *e = .533, a = 2.63R (period = 6 hours)*

These three orbits have been considered for the AMSAT Phase III program.

Motion of the orbital plane about the earth. Earlier in this chapter we discussed how the earth's asymmetrical distribution of mass causes the orbital plane of the satellite to regress about the rotational axis of the earth. To first order, the regression for elliptical orbits in degrees per day is given by

$$(1.17) \qquad \frac{d\Omega}{dt} = -9.95 \left(\frac{R_{eq}}{a}\right)^{3.5} \frac{\cos i}{(1-e^2)^2} \qquad [7]. \quad \text{(earth orbits only)}$$

Eq. 1.5, the regression rate for circular orbits is just a special case of Eq. 1.17.

The ground track. The latitude and longitude of the sub-satellite point, $\phi(t)$ and $\lambda(t)$, for a satellite in an elliptical orbit can be obtained, as a function of time, from Eqs. 1.6 and 1.7 with equation 1.15 used to relate θ to t. A set of six independent parameters (orbital elements) is needed to specify an elliptical orbit. The six used here, i, λ_o, ω_o, T, t_o, and e, are referred to as classical or Keplerian elements.

Educators interested in using a satellite in an elliptical orbit for many of the experiments in Chapter VI may find it easier to work with the following set of orbital elements:

- ϕ_a latitude of sub-satellite point at apogee,
- λ_a longitude of sub-satellite point at apogee,
- t_a time at apogee,
- i inclination,
- e eccentricity,
- T period.

It is also convenient to measure angles in the orbital plane from apogee so we introduce $\theta' = \theta - 180°$ and $\omega_o' = \omega_o + 180°$ (the argument of apogee). As an example, $\phi(t)$ and $\lambda(t)$ are now presented for a single revolution of the satellite in terms of the new set of orbital elements. Details of the derivation are contained in Chapter VI, section 2, and a numerical example is presented at the end of Chapter II.

One begins by solving Eq. 1.15 in graphical or tabular form for $0° \leq \theta \leq 360°$ yielding the positon of the satellite in the orbital plane as a function of time for one half period before and after apogee ($-180° \leq \theta \leq 180°$). Note that when θ is between 180° and 540° 2π must be added to E(θ) in Eq. 1.15. Figure 1.13 shows θ as a function of t for an orbit where T = 11 hours and e = .688. Next we solve Eqs. 1.18 and 1.19 for the argument of apogee and the longitude at ascending node. Recall that our sign convention designated N latitudes and E longitudes as positive, S latitudes and W longitudes as negative.

(1.18) $\omega_\circ' = n_1\pi + (-1)^{n_1} \arcsin\left(\frac{\sin \phi_a}{\sin i}\right)$ (argument of apogee)

$$n_1 = \begin{cases} 0 & \text{satellite headed north at apogee} \\ 1 & \text{satellite headed south at apogee} \end{cases}$$

(1.19) $\lambda_\circ = \lambda_a + (-1)^{n_2} \arccos\left(\frac{\cos \omega_\circ'}{\cos \phi_a}\right)$ (longitude at ascending node)

$$n_2 = \begin{cases} 0 & 90^\circ \leq i \leq 180^\circ \\ 1 & 0^\circ \leq i < 90^\circ \end{cases}$$

Finally, using convenient increments in either time or θ' we solve Eqs. 1.20 and 1.21 for $\phi(t)$ and $\lambda(t)$.

(1.20) $\phi(t) = \arcsin[\sin(i)\ \sin(\theta'(t)+\omega_\circ')]$

(1.21) $\lambda(t) = \lambda_\circ - (-1)^{n_2+n_3} \arccos\left(\frac{\cos(\theta'(t)-\omega_\circ')}{\cos \phi(t)}\right) - \frac{t}{4}$

$$n_3 = \begin{cases} 0 & \text{when } \phi(t) \geq 0^\circ \text{ (Northern hemisphere)} \\ 1 & \text{when } \phi(t) < 0^\circ \text{ (Southern hemisphere)} \end{cases}$$

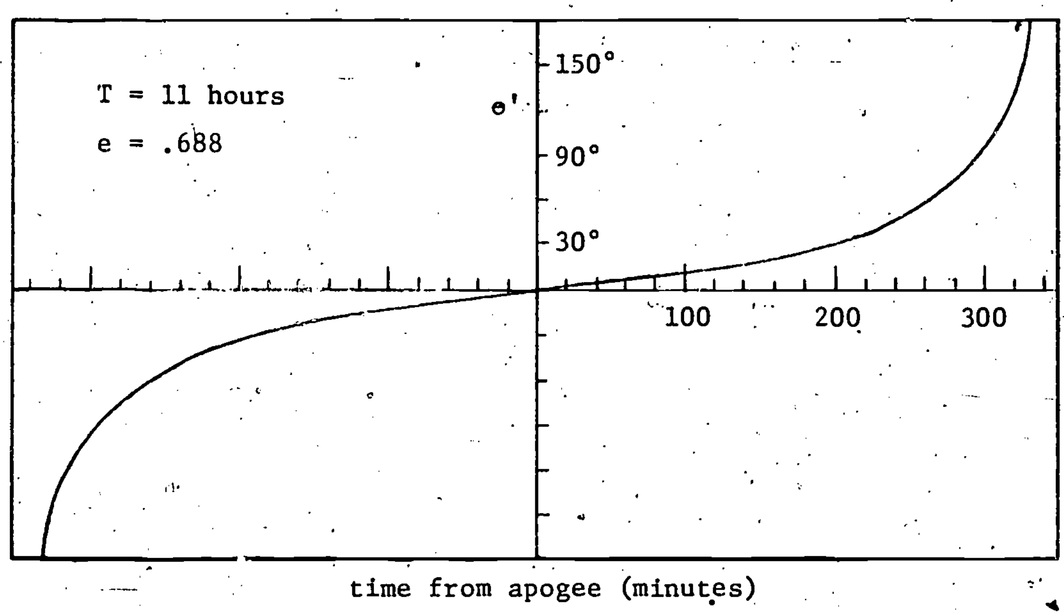


Figure 1.13. Position of satellite in orbital plane (measured from apogee) vs. time from apogee.

<u>Azimuth</u>, <u>Elevation</u>, <u>Coverage</u>. As discussed in section 1.4, the azimuth and elevation of a satellite from a ground station can be computed at any instant if the latitude and longitude of the sub-satellite point and the height of the satellite are known. Therefore, Eqs. 1.9, 1.10, and 1.11 can be used to compute azimuth and elevation for a satellite in an elliptical orbit as long as the height, h, is treated as a variable which must be computed at each point on the orbit. This is easily accomplished using Eq. 1.14. The maximum terrestrial communications distance can be computed using Eq. 1.13 after Eq. 1.14 is used to obtain the altitude.

<u>Problem 1.8.</u>
1. Derive an expression for θ (the polar angle in the satellite plane measured from perigee) as a function of s (the terrestrial distance between sub-satellite point and ground station), e (eccentricity), and a (semi-major axis).
2. Assume that e = .688 and that a = 3.93R. For what values of θ will the maximum terrestrial communication distance be 4,000 km, 4,500 km, 5,000 km, 6,000 km, 7,000 km, 8,000 km, 9,000 km, and 10,000 km?

<u>Answer</u>:

1. $$\theta = \arccos\left(\frac{1}{e} \left(\frac{a}{R}(1-e^2)\cos\left(\frac{s}{R}\right) - 1 \right) \right)$$

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6. See 4.

7. See 4.

CHAPTER II

TRACKING AMSAT-OSCAR SATELLITES

The objectives of this chapter include:

1. Introducing a number of tracking techniques which can be used in conjunction with AMSAT Phase II (OSCAR 7 and 8) and Phase III satellites, and the Soviet RS system;
2. Demonstrating how these techniques can be used to predict the position of the satellite and times when signals from the satellite can be received;
3. Presenting information on the construction of orbit calculators;
4. Discussing the relative advantages and disadvantages of various tracking techniques.

2.1 INTRODUCTION

This is a "how to do it" chapter. It discusses a number of widely used methods for tracking AMSAT-OSCARs 7 and 8, and the Soviet RS satellites. The equations on which the methods are based were included in Chapter I. This chapter has been designed so that it can be read without first reading Chapter I by those who are primarily interested in the practical aspects of tracking.

In this section we discuss some basic information common to all tracking methods. The point on the surface of the earth directly below a satellite is known as the sub-satellite point. The path that the subsatellite point traces out on the surface of the earth is known as the ground track. A portion of the ground track for three consecutive orbits of AMSAT-OSCAR 8 is shown in Figure 2.1. We use the term ground station to refer to any fixed or mobile station on or near the surface of the earth set up to receive radio signals from satellites. We say that a satellite is in range of a ground station when radio signals from the satellite can be received. This generally occurs when the satellite is above one's local horizon. For satellites in circular (or nearly circular) orbits the satellite will be in range when the distance between the sub-satellite point and the ground station is less than some critical value which we call the maximum access distance. The maximum access distance for AMSAT-OSCAR 7 is 3,950 km; for AMSAT-OSCAR 8, 3,250 km; and for RS 1 it will probably be 3,140 km (assumes 860 km altitude). AMSAT Phase III satellites will usually have a much greater access distance but we can't specify it with a single number since it varies along the orbit. Usually, the closer a ground station is to the sub-satellite point, the stronger the received signals will be. Most users will find that they are in range of OSCARs 7 and 8 and RS 1 for three passes each morning and three passes each evening (for each satellite). Most passes have a duration of about ten to twenty minutes. Phase III satellites will be in range of stations in the northern hemisphere for about 13 hours each day during the first few years in orbit.

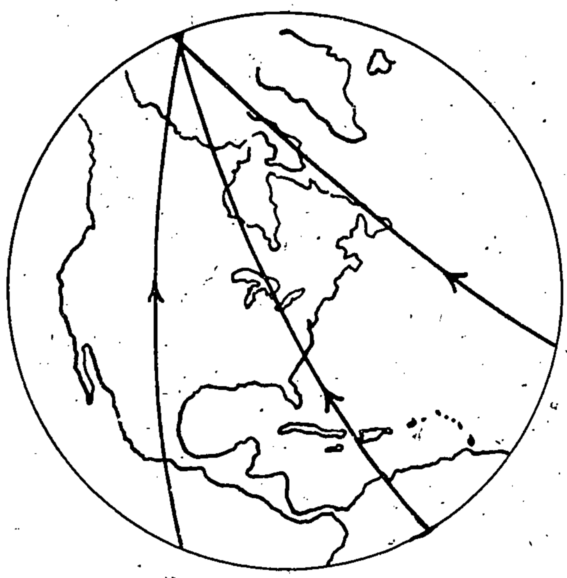


Figure 2.1 The ground track for three consecutive orbits of AMSAT-OSCAR 8.

Satellite tracking data is presented in terms of a 24 hour clock based on Universal Coordinated Time (UTC). Table 2.1 will enable ground stations in the contiguous United States to convert from UTC to local standard or daylight time.

Time zone	EST	EDT	CST	CDT	MST	MDT	PST	PDT		
Time difference	5	4	6	5	7	6	8	7		

To convert from UTC to ____________ (time zone) subtract ____________ (time difference) hours.

To convert from ____________ (time zone) to UTC add ____________ (time difference) hours.

Table 2.1 Time conversion chart.

Problem 2.1

Make the following time conversions (use 24 hour clock for all answers).

a) 2:45 pm PST June 21 = __________ PST June __________.

b) 9:13 UTC May 11 = __________ CDT May __________.

c) 22:37 EST Oct. 13 = __________ UTC Oct. __________.

Answer: a) 14:45 PST, June 21; b) 04:13 CDT, May 11; c) 03:37 UTC, Oct. 14

The first and simplest tracking method we will be discussing is called the Selected Cities Method. While the Selected Cities Method may be adequate the first few times one listens for the satellite, it has a number of shortcomings which limit its usefulness. If at this time you are primarily interested in the Selected Cities Method, skip directly to section 2.2. If you are interested in one of the more informative methods continue with this section.

As the satellite moves in space, the sub-satellite point moves over the surface of the earth. In order to compute when a specific ground station will be able to receive radio signals from the satellite, one has to trace out the ground track on a map or globe and note when the sub-satellite point will be in range of the ground station. A number of simple graphical techniques for accomplishing this are described in sections 2.3 - 2.8. Sections 2.3 -2.7 apply to circular orbits like those of OSCARs 7 and 8, and RS 1. Techniques to be used with elliptical orbits like those planned for Phase III satellites will be covered in section 2.8.

We now concentrate on tracking methods for OSCARs 7 and 8, and RS 1. Each method can be thought of as consisting of two parts: (1) the path of the sub-satellite point is drawn for the orbit of interest and (2) an acquisition "circle" is drawn around one's ground station. The ground track must be drawn for each orbit. The acquisition "circle" need only be drawn once for each satellite. Whenever the sub-satellite point is inside the acquisition "circle", the ground station will be able to receive radio signals from the satellite. The term "circle" has been put in quotes because most map projections distort distances and a circle on the surface of the globe often does not look like a circle on these maps. Later in this chapter we see the odd shapes taken by acquisition "circles" and discuss the advantages of tracking methods based on different map projections. It should be noted that, on the stereographic map discussed in section 2.3, acquisition circles are true circles. Ground stations with directional antennas also need information on where to point their antennas. We discuss how to obtain this information later in the chapter.

We turn first to tracing out the ground track for a single orbit. Orbits are arbitrarily said to start at the point where the sub-satellite point crosses the equator headed north (called the <u>ascending node</u>). Our problem is to locate the ascending node and then plot the ground track for the remainder of the orbit. The first ascending node each UTC day is called the <u>reference node</u>.

Problem 2.2

An ascending node for OSCAR 7 occurs at (02:10 UTC, 80° W). Use a globe to determine whether a ground station in Miami, Florida will be able to hear radio signals from the satellite as the ascending node occurs. How about a station in San Francisco?

Answer: Looking at a globe one can see that the terrestrial distance between the sub-satellite point at latitude 0°, longitude 80° West, and Miami is considerably less than 3,950 km. The distance between the node and San Francisco is considerably greater than the access distance. Therefore, at 02:10 UTC, the satellite is in range of Miami and out of range of San Francisco.

Commonly used methods of obtaining data on the time and longitude of ascending nodes for satellites of interest are:

1. A computer generated ORBIT CALENDAR listing all ascending nodes for a year can be purchased from S. Reymann, P. O. Box 374, San Dimas, Calif. 91773; (1978 price: $5.00).
2. Members of AMSAT receive listings of reference nodes and other relevant information on the OSCAR satellite program via the AMSAT Newsletter which is published four times per year. Individual membership is $10 per year, institutional membership is $20 per year. (AMSAT, Box 27, Washington, D.C. 20044).
3. Reference nodes for the coming month are published in each issue of QST. QST is published by the ARRL monthly -- see Appendix A for address.

Ascending nodes for a particular satellite occur at specific intervals called the period. Due to the rotation of the earth the longitude of each ascending node is a specific distance further west (the increment) than the preceding one. For example, OSCAR 7's period is 115 minutes and its increment is about 29° (west per orbit). If OSCAR 7 has a reference node at (01:18 UTC, 68°W) the next ascending node will occur at (03:13 UTC, 97°W) and the next at (05:08 UTC, 126°W) etc. So we see that these numbers -- 115 minutes for the period and 29° west/orbit for the increment -- can be used to forecast future ascending nodes for OSCAR 7 when one ascending node is known. The numbers we've just used are approximate and errors will be cumulative so they should not be employed to forecast more than one day (about 12 revolutions) in advance.

Problem 2.3

Specify all OSCAR 7 ascending nodes during the UTC day if the reference node is (01:18 UTC, 68°W) using approximate values for the period (115 minutes) and increment (29° west/orbit).

Answer:

Orbit	Time (UTC)	Longitude
1	01:18	68°W
2	03:13	97°W
3	05:08	126°W
4	07:03	155°W

Orbit	*Time (UTC)*	*Longitude*
5	*08:58*	*184°W*
6	*10:53*	*213°W*
7	*12:48*	*242°W*
8	*14:43*	*271°W*
9	*16:38*	*300°W*
10	*18:33*	*329°W*
11	*20:28*	*358°W*
12	*22:23*	*27°W*
1 (next day)	*00:18*	*56°W*

Data for the period and increment of the satellites currently in orbit is presented in Table 2.2. Since OSCAR 7 has been in orbit a number of years we can describe its orbit relatively precisely. OSCAR 8 has only been in orbit a few weeks as this is written so our data is not as precise. Space has been left so the reader can add values for RS 1 and other satellites as they are launched. Additional orbital parameters are listed in Table 2.2 for reference although we will not be using them at this time.

Satellite	Period (minutes)	Increment (°W/orbit)	Inclination	Mean Altitude	Date*
AMSAT-OSCAR 7	114.945	28.737	101.7°	1,460 km	1/78
AMSAT-OSCAR 8	103.23	25.81	98.99°	910 km	3/78
Soviet RS 1					
[ ]					
[ ]					

Table 2.2 Summary of orbital parameters for low altitude satellites of interest. (*) Date when parameters were computed.

Problem 2.4

Recompute the answers to Problem 2.3 using the more accurate OSCAR 7 data in Table 2.2. What are the errors in time and longitude of the ascending node during orbit 12 which result from using the approximations: period = 115 minutes, increment = 29° west/orbit ?

Having seen how ascending nodes can be determined once one node is known we now turn to the problem of tracing out the path of the sub-satellite point for the orbit following the node. Table 2.4, which was obtained using Eqs. 1.6 and 1.7, enables us to compute the position of the sub-satellite point off the equator. In order to use it, we have to know the coordinates of the most recent ascending node.

Time after ascending node (minutes)	AMSAT-OSCAR 7 Lat. (°N)	AMSAT-OSCAR 7 Long. (°W)	AMSAT-OSCAR 8 Lat. (°N)	AMSAT-OSCAR 8 Long. (°W)	Soviet RS-1 Lat. (°N)	Soviet RS-1 Long. (°W)
0	0.0	0.0	0.0	0.0	0.0	0.0
2	6.1	1.8	6.9	1.6	7.0	359.5
4	12.3	3.6	13.8	3.2	14.0	359.0
6	18.4	5.4	20.7	4.9	21.0	358.4
8	24.5	7.4	27.5	6.7	27.9	357.7
10	30.6	9.5	34.4	8.7	34.9	356.9
12	36.7	11.9	41.2	11.0	41.8	355.8
14	42.7	14.5	48.0	13.6	48.8	354.3
16	48.7	17.6	54.8	16.9	55.6	352.1
18	54.6	21.5	61.4	21.4	62.4	348.9
20	60.4	26.4	67.9	28.0	69.1	343.4
22	66.0	33.2	74.0	39.0	75.3	333.1
24	71.2	43.4	79.0	60.7	80.4	309.7
25.5					82.0	276.4
25.8			81.0	96.4		
26	75.5	59.9	80.9	100.8	81.8	264.0
28	78.1	85.8	78.2	137.7	78.1	228.9
28.7	78.3	97.2				
30	77.7	116.3	72.9	156.6	72.3	213.6
32	74.5	139.6	66.7	166.5	65.8	206.2
34	69.9	154.1	60.2	172.5	59.0	202.1
36	64.6	163.2	53.5	176.7	52.2	199.4
38	58.9	169.4	46.7	179.8	45.3	197.7
40	53.1	174.0	39.9	182.4	38.4	196.4
42	47.1	177.6	33.1	184.6	31.4	195.4
44	41.1	180.6	26.2	186.5	24.4	194.7
46	35.1	183.1	19.3	188.3	17.5	194.0
48	29.0	185.4	12.4	190.0	10.5	193.5
50	22.9	187.5	5.6	191.6	3.5	193.0
51					0.0	192.8
51.6			0.0	192.9		
52	16.8	189.4				
54	10.6	191.3				
56	4.5	193.1				
57.5	0.0	194.4				

Table 2.4 Data for plotting ground tracks based on following orbital parameters:

	Period	Inclination
AMSAT-OSCAR 7	114.945 min.	101.7°
AMSAT-OSCAR 8	103.23 min.	98.99°
Soviet RS-1	102. min.	82.°

Continuing with the example used in Problem 2.2 where an ascending node for OSCAR 7 occurred at (02:10, 80°W). The columns labeled OSCAR 7 in Table 2.4 reveal that at 02:12 (2 minutes later ascending node) the sub-satellite point will be at latitude 6.1°N and longitude 81.8°W (1.8° further west); at 02:14 it will be at 12.3°N, 83.6°W (3.6° further west); etc.

Problem 2.5

1. *Use Table 2.4 to construct a chart for OSCAR 7 for the sub-satellite path following an ascending node at (02:10, 80°W). Include time (UTC); latitude, longitude, time (local). Plot these points on a map or globe and connect them with a smooth curve. Locate your ground station on the map.*

2. *At what time (UTC) will the satellite be closest to your ground station? This time is called TCA (time of closest approach).*

3. *What is the TCA in local time?*

4. *Will the pass be within range of your ground station?*

5. *At what time (UTC) will the satellite come into range? This time is called AOS (acquisition of signal).*

6. *At what time (UTC) will the satellite pass out of range? This time is called LOS (loss of signal).*

Note: 5 and 6 should only be attempted if a globe is available for plotting the sub-satellite path and drawing the acquisition circle (radius = 3,950 km for OSCAR 7).

Answer.

1.

Time (UTC)	*latitude*	*longitude*	*time (local)*
02:10	0.0	80.0°W	
02:12	6.1°N	81.8°W	
02:14	12.3°N	83.6°W	
02:16	18.4°N	85.5°W	
02:18	24.5°N	87.4°W	
02:20	30.6°N	89.6°W	
02:22	36.6°N	91.9°W	
.	.	.	
.	.	.	
.	.	.	

2. *Looking at the plot of the sub-satellite path, pick out the point which is closest to your ground station. This point will fall between two plotted points. Since the points were plotted at two-minute intervals, you should be able to use "eyeball interpolation" to estimate TCA to better than one minute.*

3. *To convert from UTC to local time see Table 2.1.*

4. *The satellite will be within range at TCA if the surface distance between the sub-satellite point and the ground station is less than 3,950 km (OSCAR 7) at this time. This distance can be estimated using a map or globe -- remember, we are interested in whether the distance is greater or less than 3,950 km, not the actual value. Stations in the 48 contiguous states and most of Alaska will be in range of this pass at TCA.*

5. and 6. To calculate AOS and LOS we have to draw an acquisition circle around the ground station with a radius of 3,950 km. This is easily done on a globe using a protractor. Methods for drawing acquisition circles on various map projections are discussed later in the chapter. The time marks on the sub-satellite path can be used to estimate AOS and LOS (the points where the sub-satellite point enters and leaves the acqusition circle). Other methods of finding AOS and LOS are covered later in the chapter.

Refering to Table 2.4, we see that each orbit has two nodes (points where sub-satellite point crosses equator; latitude = 0); one as the orbit begins with the sub-satellite point entering the northern hemisphere (ascending node), and a second one midway through the orbit as the sub-satellite point enters the southern hemisphere (descending node).

Problem 2.6

Use Table 2.4 to answer the following questions.

1. *How far north does each satellite go?*
2. *If an OSCAR 8 ascending node occurs at 09:13 UTC, when will the next descending node occur?*
3. *If an OSCAR 7 ascending node occurs at (09:13 UTC, 170°W) what will be the time and longitude of the next descending node?*
4. *If an OSCAR 8 ascending node occurs at (09:13 UTC, 170°W) give the time and position of the sub-satellite point when it is next headed due west.*

Answer:

1. *OSCAR 7: 78.3°N; OSCAR 8: 81.0°N; RS 1: 82°N (pre-launch data)*
2. *10:04.6 UTC. For circular orbits nodes are separated in time by one half the period.*
3. *10:10.5 UTC, 004.4°W (364.4°W = 004.4°W). For circular orbits the ascending node and descending node are separated in longitude by 180° plus one half the increment.*
4. *As OSCAR 8 approaches 81°N, its north-south velocity component passes through zero. When the north-south velocity is zero, the sub-satellite point is moving due west. The time is midway between ascending and descending nodes or 25.8 minutes after the ascending node. The time is therefore 9:38.8 and the position, from Table 2.4, is 81°N, 266.5°W.*

Acquisition Circles, Azimuth and Elevation. Having covered the first part of the satellite tracking problem -- tracing out the sub-satellite path during an orbit -- we turn to the second part -- acquisition circles, azimuth, and elevation data. Information on the position of the satellite with respect to one's ground station is usually most convenient in terms of elevation (angle above the horizontal plane) and azimuth (angle in the horizontal plane with respect to true north). If one is using an omnidirectional antenna (no direction favored) or an antenna which can not be aimed then one can generally ignore elevation and azimuth data -- only an acquisition circle is needed. If one is using a broadly directive antenna array crude pointing data becomes necessary. If one is using a highly directive array precise pointing information is required.

The elevation angle of the satellite and the concept of acquisition circle are closely related -- the acquisition circle is simply the locus of all sub-satellite points for which the satellite will have an elevation of 0°. This value was arbitrarily chosen because radio signals from the satellite are generally restricted to line-of-sight paths and, if one's ground station is set up in relatively level terrain, the satellite will be line-of-sight when its elevation angle is above 0°. The relation between the elevation angle and the terrestrial distance between ground station and sub-satellite point is shown in Figure 2.2 for OSCAR 7 and OSCAR 8. Note that for a particular satellite in a circular orbit elevation angle only depends on terrestrial distance. The basic relationship for elevation angle, Eq. 1.11 applies to all satellites (circular and elliptical orbits).

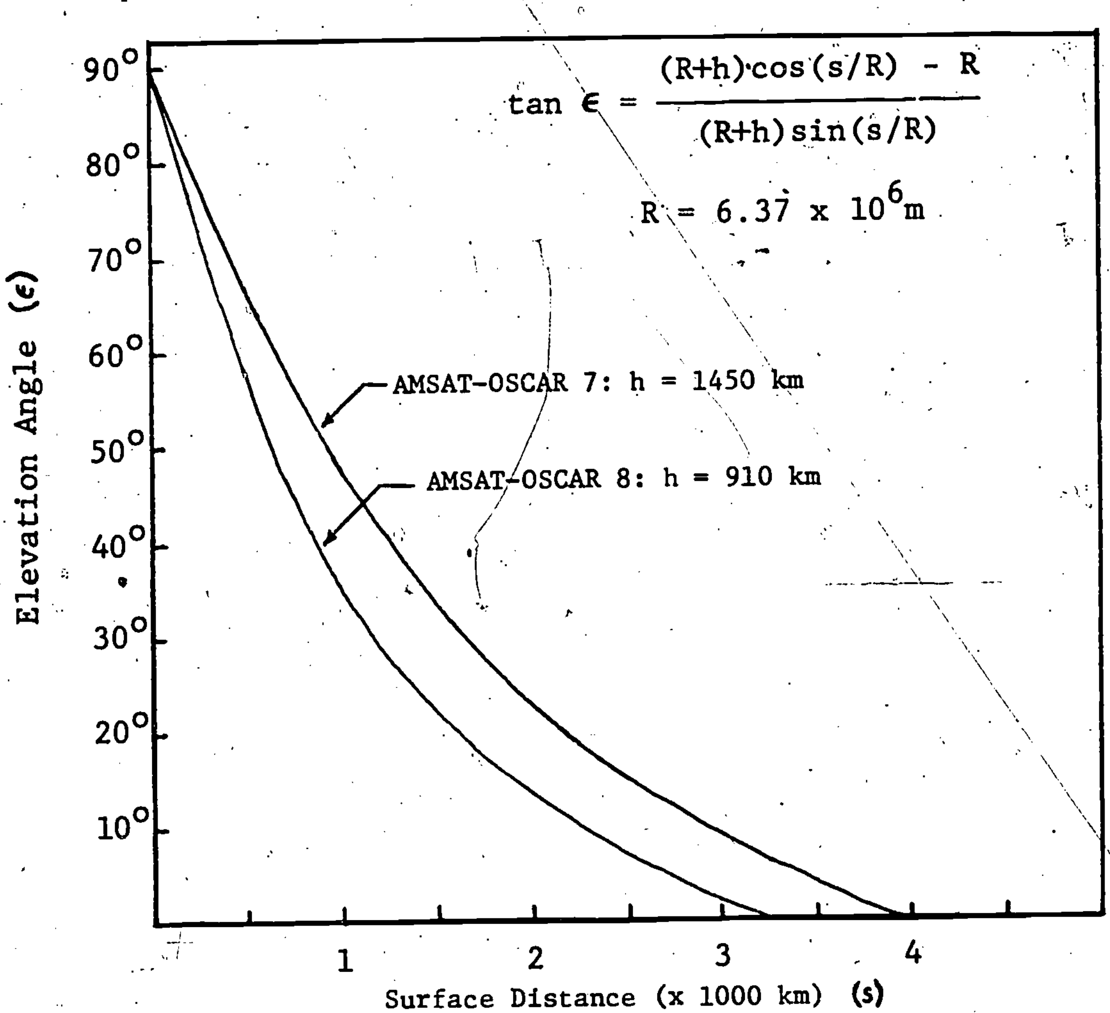


Figure 2.2 Elevation angle vs. surface distance between ground station and sub-satellite point for OSCAR 7 and 8.

Problem 2.7

Refer to Figure 2.2 and estimate the terrestrial distances corresponding to elevation angles of 0°, 30°, 60°, and 90° for OSCAR 7 and OSCAR 8.

Answer: See *Table* 2.3.

Since we already know how to find the position of the sub-satellite point at any time, the elevation-azimuth problem reduces to: given two points on the surface of the earth, find the terrestrial distance between them and the azimuth of the second with respect to the first. This is a fundamental problem in spherical trigonometry, perhaps more familiar in navigation. The solution to the problem in terms of the latitude and longitude of the two points (sub-satellite point and location of ground station) was given in Chapter I, Eq. 1.9 and Eq. 1.10. However, information on bearing and terrestrial distance having the accuracy we need for tracking can usually be obtained using simpler methods based on maps and globes as we soon see.

The map based tracking methods discussed later in this chapter will suggest that you draw an acquisition circle around your ground station for each satellite. This is just an iso-elevation circle for a 0° elevation. Additional iso-elevation circles at 30° and 60° may also be useful. Radii values for the various circles are given in Table 2.3 which was obtained from Eq. 1.11.

Satellite	mean altitude	Elevation Angle (*1) 0°	15°	30°	45°	60°	75°
OSCAR 7	1460 km	3950 km	2580 km	1690 km	1099 km	667 km	316 km
OSCAR 8	910 km	3220 km	1925 km	1193 km	754 km	451 km	212 km
RS 1 (*2)	860 km	3138 km	1854 km	1141 km	719 km	429 km	202 km

Table 2.3 Terrestrial distances (radii) for iso-elevation circles. Data obtained by inverting Eq. 1.11:

$$s = R[\pi/2 - \epsilon - \arcsin(R\cos\epsilon/(R+h))]$$

Term in [] expressed in radians; R = mean radius of earth; h = mean altitude; s = terrestrial distance; ϵ = elevation.
(*1) acquisition circle.
(*2) tentative data.

On maps we will be plotting azimuth every 45°. This is adequate with a broadly directional antenna. Again, if one is using an omnidirectional antenna, this data can be omitted and if one is using a highly directional antenna, the same approach can be used to plot azimuth lines at closer intervals. For most map projections (not including stereographic or azimuthal equidistant projections centered on ones ground station) data on surface distance and azimuth is most easily transferred from a globe to the map point by point. This can be a very tedius procedure but azimuth lines only have to be drawn once for your ground station. Acquisition and iso-elevation curves only have to be drawn once for each satellite.

Our next step is to choose one of the plotting methods described in sections 2.3 - 2.7 which are designed to eliminate the time-consuming calculations needed to plot the ground track for a particular orbit point by point. But before turning to these methods, we pause to look at the simplest (and least informative) tracking method -- Selected Cities.

2.2 METHOD I: SELECTED CITIES

Tracking information in the Selected Cities format indicates the times at which the satellite of interest passes near the selected cities. To use information in this format, one picks a nearby selected city and starts listening about 15 minutes before the time indicated. Data in this format usually takes the form shown in Table 2.5. The first column gives the total number of complete revolutions of the satellite since launch. It is often useful for reference. The second column contains the UTC date.

		Ascending Node			TCA
Orbit	Date (UTC)	Time (UTC)	Longitude	City	Time (UTC)
12253	Sept. 9	00:52.4	64.5°W	New York	01:06
12254	Sept. 9	02:47.4	93.3°W	Denver	03:01
12255	Sept. 9	04:42.4	122.0°W	Fairbanks	05:04
12256	Sept. 9	06:37.4	150.8°W	Honolulu	06:44

Table 2.5 Tracking data using Selected Cities format. AMSAT-OSCAR 6 September 9, 1975.

For example, OSCAR 6 orbit 13253 September 9, 1975, actually occurs during the evening of September 8, local time for station in the eastern United States. The next two columns contain data on ascending nodes -- information which can be ignored when using the Selected Cities method. The final two columns contain the selected cities and times when the satellite is closest to them (time of closest approach or TCA). If you live in one of the selected cities, you can expect to begin hearing satellite signals (acquisition of satellite -- AOS) about ten minutes before the satellite passes overhead and continue receiving them until about ten minutes after the time listed (loss of satellite -- LOS). The pass duration will be about 15 to 20 minutes for a ground station in the selected city. If you do not live in a selected city, just pick the closest one and start listening about 15 minutes before the time indicated. The tracking methods discussed later in this chapter enable one to make more accurate predictions of when to listen -- using these alternate methods most stations can predict AOS, TCA, and LOS to better than one minute.

Problem 2.8

1. *Use Table 2.5 to predict the time and date of (a) a pass over NYC in EST; (b) a pass over Denver in MST.*
2. *Use Table 2.5 to predict when a ground station in Chicago will be able to receive signals from the satellite.*

Answer:

1. *NYC pass: 8 Sept. 1975, 7:56 - 8:16 pm EST.*
   Denver pass: 8 Sept. 1975, 7:51 - 8:11 pm MST.
2. *Chicago is approximately midway between Denver and NYC. A ground station in Chicago should be able to receive satellite signals during both the Denver and NYC passes -- 01:06 UTC and 03:01 UTC. The ground station should actually start listening about 15 minutes earlier so as not to miss AOS. A demonstration is always more effective when you tune around for a few minutes and there are absolutely no signals present and then, all of a sudden, signals from the satellite begin to appear.*

Tracking information in the Selected Cities format is available from: OSCAR Educational Programs Office, American Radio Relay League, 225 Main Street, Newington, Conn. 06111.

If you have not already done so please read pages 2-3 to 2-11 before continuing.

2.3 METHOD II: NORTH POLE PROJECTION MAP

More satellite ground stations use tracking aids based on polar projection maps than any other method. Two types of polar maps are of special interest: (1) Azimuthal Equidistant Polar Projection (AEPP) charts and (2) Stereographic Polar Projection (SPP) charts. Latitude lines on both of these charts are concentric circles centered on the pole. AEPP charts are characterized by equally spaced latitude lines while SPP charts are characterized by increasing spacing between latitude lines as one moves further from the pole. SPP charts have certain advantages but, because they're more readily available, AEPP charts have been used and described more extensively [1-5]. Sources for both types of maps are listed at the end of this section.

Sub-satellite paths. The first requirement of any tracking method is that it show sub-satellite paths. If Table 2.4 is used to plot ground tracks of a specific satellite for a number of orbits on either AEPP or SPP charts, one observes that every ground track has the same shape. This makes it possible to construct an "orbit calculator" by drawing a single ground track on a transparent overlay (ground track overlay) using Table 2.4. The overlay is then repositioned for each orbit by rotating it until the ascending node on the overlay coincides with the actual ascending node on the map. Figure 2.3a and 2.3b contain drawings which can be used to construct a simple orbit calculator for OSCAR 7, OSCAR 8, or RS 1. To assemble the calculator, trace the ground track overlay for the satellite of interest (Figure 2.3b) on a piece of transparent plastic. Then place the overlay on the map and insert a pin through the pivot point on the overlay and the north pole on the map. To preview a specific orbit, rotate the overlay until the ascending node on the overlay and the position of the node on the map are aligned. One can then see exactly where the sub-satellite point will be 2, 4, 6, 8,... minutes past the ascending node.

The exact same procedure works with an SPP map as shown in Figure 2.4 where we have drawn a single ground track for OSCAR 8 on the map. The ground track can be traced on a transparent overlay to preview other orbits. In sum, techniques for using AEPP and SPP charts to observe ground tracks are identical. Techniques for plotting acquisition and iso-elevation circles (iso-distance for circular orbits) about ones ground station are, however, different.

<u>Acquisition and elevation circles</u>. Plotting acquisition and elevation circles on an AEPP chart is done by the tedious process of transfering them point by point from a globe to the map as outlined in section 2.1. "Circles" scaled to the map in Figure 2.3a for ground stations at various latitudes are presented in Figure 2.3b where acquisition and elevation information is for OSCAR 7.

All circles on the globe are also circles on the SPP chart. This greatly simplifies drawing acquisiton and elevation circles on these maps. To draw a circle about ones ground station (latitude = $\emptyset_g$; longitude = λ_g) on the SPP chart note that the center of the circle does <u>not</u> coincide with the ground station, although both lie along the same longitude line, and then follow the following steps.

1. Select the radius distance and transform it into degrees of arc on the surface of the earth using: 1.000° of arc = 111.2 km. This gives $\emptyset^*$
2. Compute $\emptyset_g+\emptyset^*$ and $\emptyset_g-\emptyset^*$ and then plot these two points along longitude λ_g.
3. Bisect the line joining the two points found in step (2) -- this gives the center of the circle.
4. We now have the center of the circle (step 3) and two points lying on it so it is easily drawn.

As an example the acquisition circle for OSCAR 8 has been drawn in Figure 2.4 about a ground station in Washington, D. C. (39°N, 77°W). The calculation is outlined below following the procedure just described.

1. radial distance = 3,220 km; $\emptyset^*$ = 28.96° (about 29°)
2. 68°N, 10°N
3. and 4. See Figure 2.4.

<u>Map sources: AEPP charts</u>. Commercial tracking calculators of this type measuring about 30 cm in diameter are available from Ham Radio Magazine (the <u>Satellabe</u> at $7.00) and from the ARRL (the OSCARLOCATOR at $1.00). Addresses are in Appendix A and prices are subject to change. A more

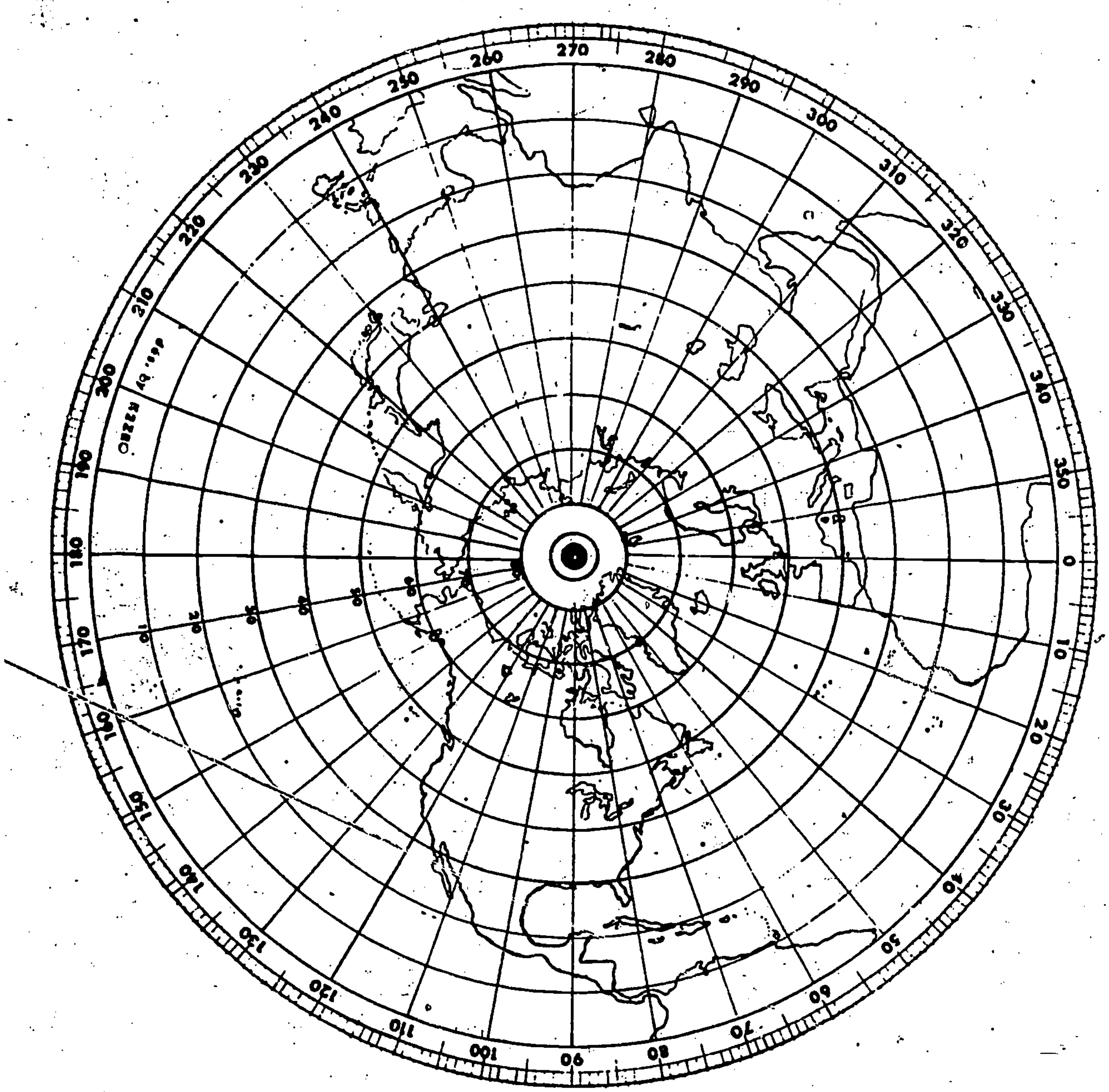


Figure 2.3a Azimuthal equidistant projection map centered on north pole for use in constructing tracking calculator.

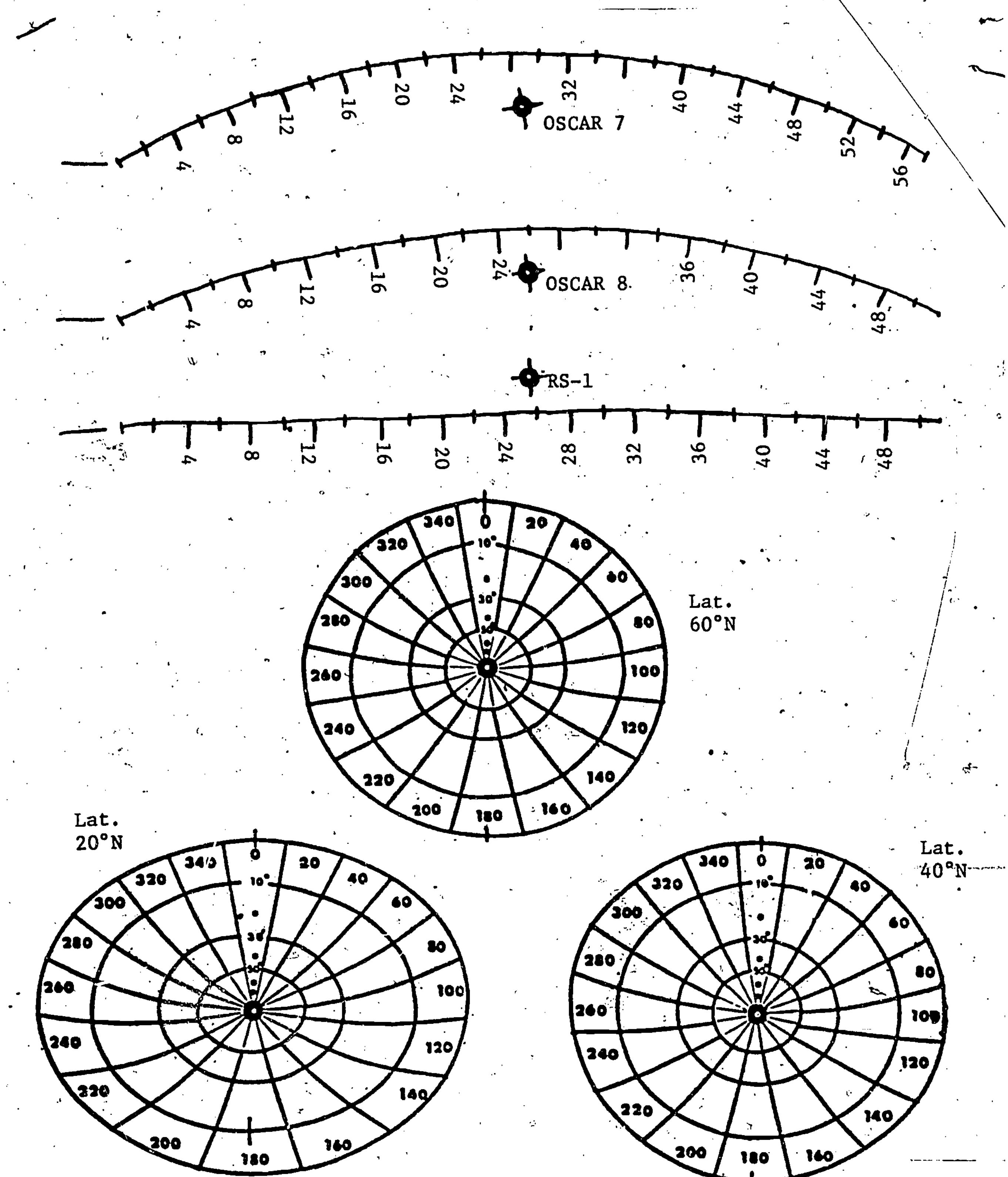


Figure 2.3b Orbit overlays for OSCAR 7, OSCAR 8, and RS-1, to be used with map of Figure 2.3a. Sample acquisition, elevation circles for OSCAR 7 also shown.

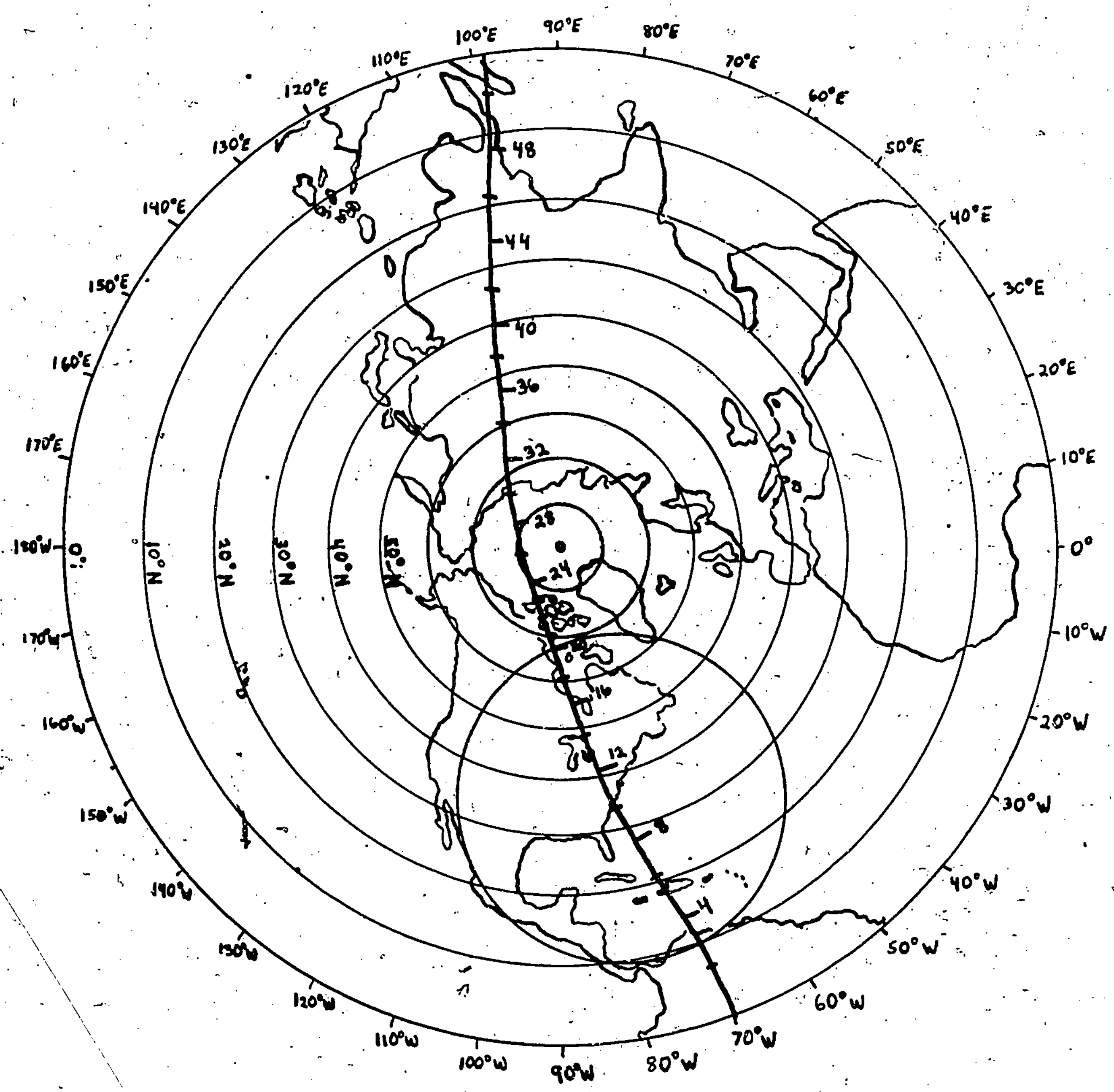


Figure 2.4 Stereographic polar projection map with acquisition circle for OSCAR 8 drawn about Washington, D.C. Sample OSCAR 8 ground track shown.

accurate calculator can be constructed from a larger map and a transparent plexiglass overlay. A good map (60 cm in diameter) is available to educators without charge from: APT Coordinator, U. S. Department of Commerce, NOAA, National Environmental Satellite Center, Suitland, Md., 20233. Request "APT plotting board". A useable AEPP tracking calculator can be constructed from polar graph paper as illustrated in section 2.8.

Map sources; SPP charts. A highly accurate SPP chart is available from Department of Commerce; Distribution Division (C-44); National Ocean Survey; Riverdale, Md. 20840. It is titled "U. S. A. F. Physical-Political Chart of the World". Chart GH-2A (60 cm diameter) costs .50¢. Chart GH-2 is twice the size but otherwise identical ($1.00).

You can easily draw your own SPP chart using the following formula for constructing latitude lines concentric with the pole:

$$s = k \tan[(90° - \emptyset)/2]$$

where s = distance between pole and latitude line;
$\emptyset$ = latitude (when using north pole as center northern latitudes are (+), southern latitudes are (-));
k = arbitrary constant which adjusts overall size.

2.4 METHOD III: MERCATOR MAP

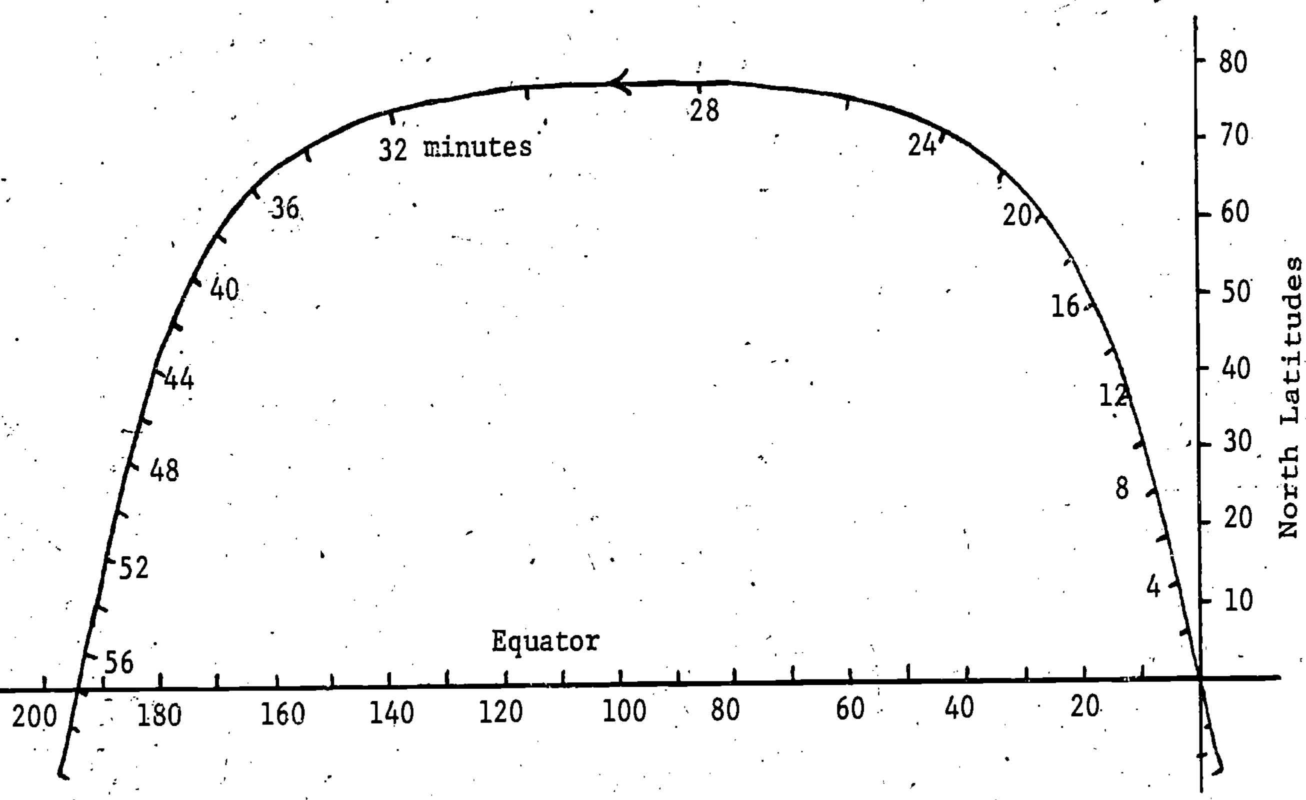
Mercator maps are widely available, familiar to most students and well suited to satellite tracking [5]. If Table 2.4 is used to plot a number of orbits on this type of map, one will notice that every ground track has the same shape. This makes it again possible to construct an orbit calculator by drawing a single ground track on a transparent overlay (ground track overlay) and repositioning the overlay on the map for each orbit. Figure 2.5a shows an orbit overlay for a calculator based on a mercator map. To use this type of calculator one sets the equator on the overlay to coincide with the equator on the map and aligns the reference ascending node on the overlay with the correct position on the map. One can then see precisely where the satellite will be 2, 4, 6, 8, ... minutes after the ascending node.

The second requirement of a plotting method is that it provide an acquisition circle and information on elevation and azimuth. Techniques for obtaining this information from a globe and presenting it on any type of map were discussed in section 2.1. Figure 2.5b shows the results for a ground station at latitude 32°N.

2.5 METHOD IV: AZIMUTHAL EQUIDISTANT PROJECTION MAP (non-polar)

The azimuthal equidistant projection (AEP) map can be constructed around any central city. On this type of map straight line azimuths radiate out from the central city and the loci of points at a given distance from the city form concentric circles. Although the AEP map is relatively unfamiliar to students, it is very useful for tracking a specific satellite in a circular orbit [6]. However, these maps are only available for certain cities and, if you are located more than 150 km from such a city, using an AEP map causes significant tracking errors. In addition, a separate calculator must be drawn for each satellite.

If one uses Table 2.4 to plot a number of orbits on an AEP map one quickly notices that sub-satellite paths do <u>not</u> have the same shape. Therefore, one cannot construct an overlay showing a single ground track which can be repositioned for each orbit. However, one can construct an orbit calculator in the following manner. (An OSCAR 7 AEP tracking calculator is shown in Figure 2.6). Use Table 2.4 to plot orbits for the satellite of interest with ascending nodes at 0°, 20°, 40°, 60°, 80°, 100°, and 120° directly on the AEP map using a distinctively colored pen. Use a second colored pen to draw the orbits of descending nodes occurring at 50°, 70°, 90°, 110°, 130°, and 150°. (The nodes specified include orbits available to east coast U.S. stations for near polar, low-altitude satellites. Central and western ground stations should begin and end further west). Use Table 2.4 to label latitude lines on the map in terms of minutes from equatorial crossing -- the closest minute is sufficient. Now affix a sheet of clear plastic over the map. To preview a specific orbit, for example one having an ascending node at 66° W longitude, one notes that this is between the orbits shown at 60°W and 80°W. Using a felt-tipped marker, the orbit is sketched in using "eyeball interpolation". A damp tissue serves to erase the orbit.



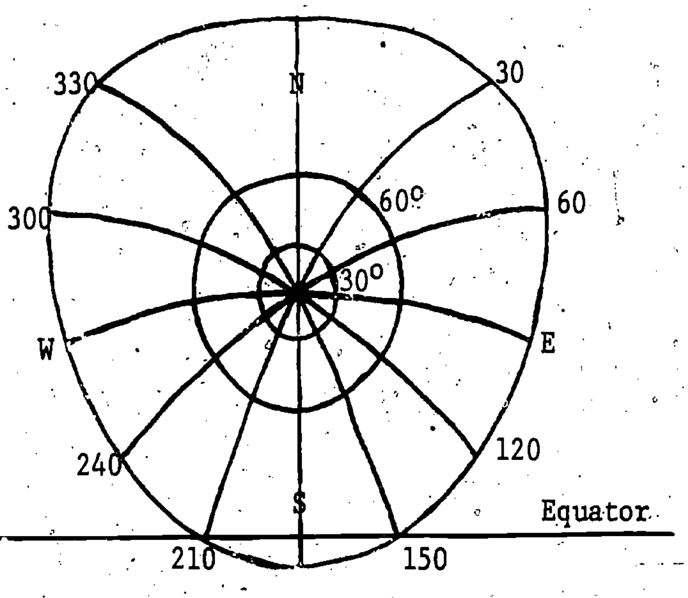
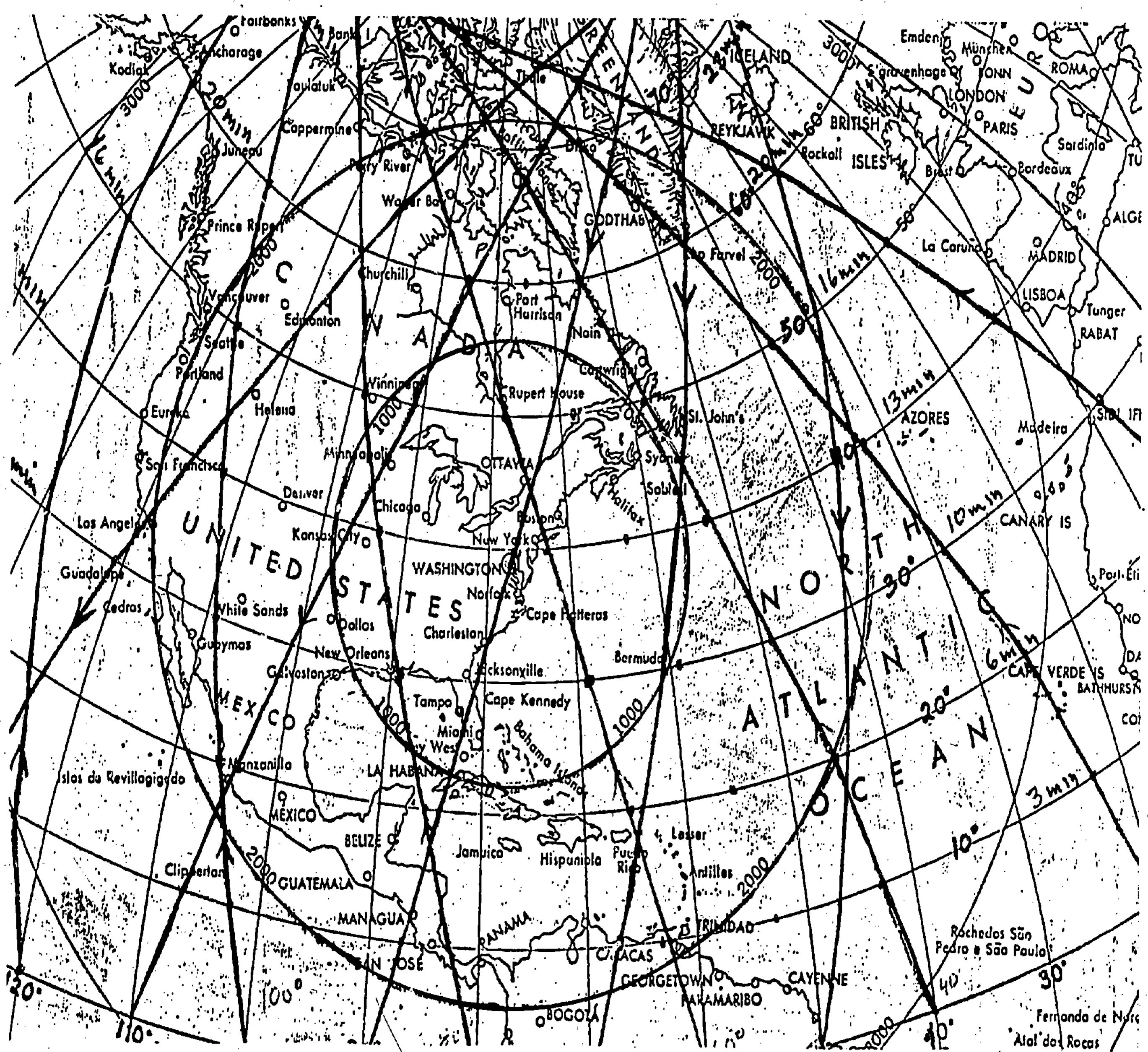


Figure 2.5a AMSAT-OSCAR 7 orbit overlay for Mercator type map.

Figure 2.5b Acquisition, elevation, and azimuth data for Latitude 32°N (same scale as Figure 2.5a).



Fairbanks
Anchorage
Kodiak
Coppermine
Juneau
Prince Rupert
Vancouver
Edmonton
Seattle
Portland
Eureka
Helena
San Francisco
Los Angeles
Guadalupe
Cedros
Denver
Kansas City
White Sands
Dallas
Guaymas
New Orleans
Galveston
MEXICO
Manzanillo
Islas de Revillagigedo
MEXICO
Clipperton
GUATEMALA
MANAGUA
SAN JOSE
BELIZE
Churchill
Winnipeg
Minneapolis
Chicago
OTTAWA
Boston
New York
WASHINGTON
Norfolk
Cape Hatteras
Charleston
Jacksonville
Tampa
Cape Kennedy
Miami
Key West
Bahama Islands
LA HABANA
Jamaica
Hispaniola
Puerto Rico
Lesser
Antilles
TRINIDAD
PANAMA
CARACAS
BOGOTA
GEORGETOWN
PARAMARIBO
CAYENNE
Port Harrison
Rupert House
Nain
Cartwright
St. John's
Sydney
Sable I
Halifax
Bermuda
Thule
Godthab
Water Bay
Perry River
Baffin Island
ICELAND
REYKJAVIK
Rockall
BRITISH
ISLES
Brest
Bordeaux
LONDON
PARIS
Emden
München
BONN
's-gravenhage
ROMA
Sardinia
La Coruña
MADRID
LISBOA
Tanger
RABAT
AZORES
Madeira
CANARY IS
CAPE VERDE IS
BATHHURST
Rochedos São Pedro e São Paulo
Fernando de Noronha
Atol das Rocas
C A N A D A
U N I T E D S T A T E S
M E X I C O
N O R T H A T L A N T I C O C E A N
G R E E N L A N D
1000
2000
3000
20 min
16 min
13 min
10 min
6 min
3 min
60°
50°
40°
30°
20°
10°
100°
110°
120°

The second requirement of a plotting method is that it provide an acquisition circle and information on elevation and azimuth. This is already built into the AEP chart. Azimuths and distances from the central city on the AEP map are true and elevation and acquisition "circles" are actual circles at the distances specified in Table 2.3.

Problem 2.9

Can an AEP map be constructed so that all ground tracks will have the same shape?

Answer: Yes, but only AEP maps centered on the North or South pole have this property -- see section 2.3.

2.6 METHOD V: GLOBE

A globe does away with the distortions of two dimensional maps and provides a clear picture of the sub-satellite path when Table 2.4 is used to plot an orbit. However, for mechanical reasons, orbit calculators based on a globe are not as convenient to use as those based on maps. A globe tracking calculator for OSCAR 7, for example, can be constructed by drawing a latitude line on a globe at 78°N and using Table 2.4 to label existing latitude lines as to time from equator crossing. Some simple methods for previewing an orbit are:

1. Tape a piece of string to the ascending node and route it tangent to the circle at 78°N (97.2° west of ascending node) and then down to the descending node (194.4° west of ascending node);

2. Bend a wire coat hanger into a semicircle matching the globe equator then twist the ends slightly so that when one end of the wire is set at the ascending node the wire will align with the reference points at 78°N (97.2° further west) and at the descending node (194.4° further west).

3. If transparent overlay hemispheres matching the globe are available then the actual ground track for a single orbit can be drawn on the transparent overlay using Table 2.4 and repositioned for each pass. Hubbard Scientific Co. (2855 Shermer Rd., Northbrook, Ill. 60062) has an inexpensive 20 cm diameter globe which comes with two transparent overlay hemispheres (about $8 in 1978). The National Geographic Society (17th and M Streets, N.W., Washington, D.C. 20036) has a 30 cm diameter political globe with transparent overlay (about $29 in 1978).

The first two globe methods are approximate. The third is more accurate but users almost unanimously report that the map based methods are more convenient and accurate.

2.7 METHOD VI: COMPUTER

Since the map and globe methods are sometimes used mainly to provide information on (1) surface distance to sub-satellite point, (2) azimuth, and (3) elevation, a number of satellite users have decided to compute these three parameters directly and thereby eliminate the middle step -- plotting the sub-satellite path on a map or globe. This can be done on a small programable hand calculator. A large computer speeds up the process and provides printout in a more useful format. The information needed to write the computer programs is contained in Eqs. 1.6, 1.7, 1.8, 1.9 and 1.10 and in Table 2.2. Rather than provide a printout for each actual orbit -- the amount of paper soon becomes overwhelming -- the usual practice is to produce "look up tables" for each satellite for ascending nodes at 1° intervals along the equator. The tables will depend on the location of the ground station. A sample printout for Baltimore, Md. is shown in Figure 2.7. When the ascending node is 53.6°W for example, the table for 54°W could be used with very little error. If greater precision is required one could interpolate between the tables for 53°W and 54°W or even rerun the program for 53.6°W. Information in this format is especially suitable if one desires to automate the antenna aiming mechanism or to do mathematical simulation studies of the orbital characteristics of a particular satellite.

OSCAR 7 TRANSIT CHART FOR ASCENDING NODE AT 63 DEG WEST
CALCULATED FOR BALT MD LONGITUDE -76.60 LATITUDE 39.35

MIN AFTER NODE	AZIMUTH	ELEVATION	SLANT RANGE (m)
2.0	159.1	0.6	4476.1
3.0	159.1	4.0	4121.5
4.0	159.1	7.7	3768.5
5.0	159.1	11.9	3418.8
6.0	159.1	16.6	3074.9
7.0	159.0	22.1	2740.1
8.0	158.9	28.7	2419.6
9.0	158.6	36.7	2120.8
10.0	158.1	46.7	1855.7
11.0	156.9	59.1	1641.5
12.0	152.8	74.2	1500.9
13.0	43.7	87.4	1455.8
14.0	348.3	72.0	1514.9
15.0	345.0	57.3	1667.0
16.0	344.0	45.2	1889.2
17.0	343.6	35.5	2159.5
18.0	343.4	27.7	2461.3
19.0	343.3	21.3	2783.6
20.0	343.4	15.9	3119.2
21.0	343.4	11.3	3463.3
22.0	343.5	7.2	3812.7
23.0	343.7	3.6	4165.0
24.0	343.8	0.2	4518.6

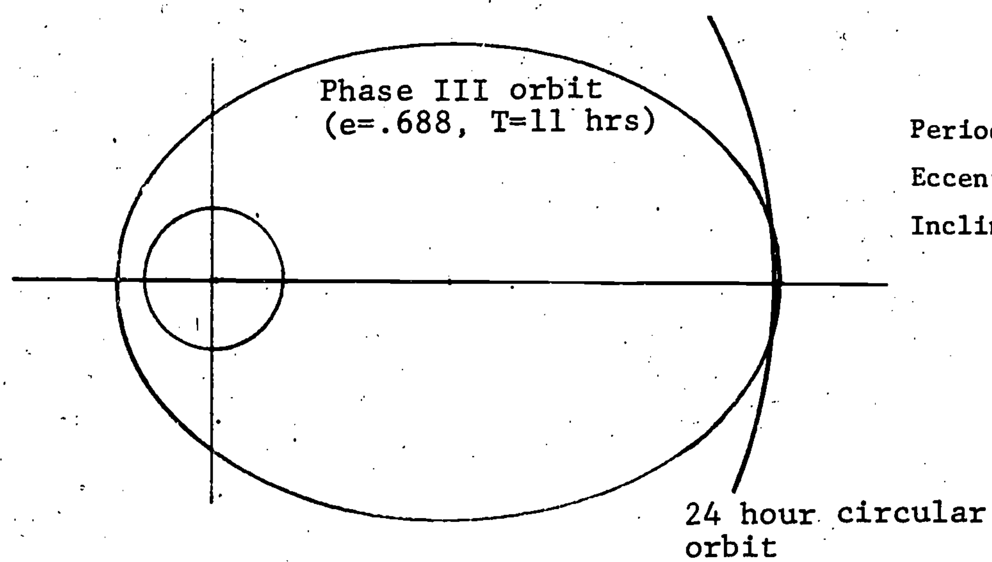
Figure 2.7. Page from computer generated "look up tables" for AMSAT-OSCAR 7.

2.8 ELLIPTICAL ORBITS AND AMSAT PHASE III SATELLITES

An orbit calculator for a satellite in an elliptical orbit must take into account two parameters -- apogee position and altitude -- which a calculator for a satellite in a circular orbit can ignore. Before proceeding with this section, the reader should be familiar with the tracking methods for circular orbits, especially those discussed in conjunction with north pole projection maps (section 2.3).

The apogee of a satellite is the point of greatest distance from the earth. In order to plot the ground track for a satellite in an elliptical orbit, we will assume that the latitude of the sub-satellite point at apogee (latitude of apogee) is known. For a specific latitude of apogee all ground tracks of a given satellite will have the same shape when plotted on a polar projection map, Mercator map or globe. For clarity we temporarily restrict ourselves to techniques based on a polar projection map. If the latitude at apogee is constant for a specific satellite, we can make a single transparent ground track overlay to be used with the polar projection map. The overlay can be rotated to show ground tracks for all orbits as was done with circular orbits. The latitude of apogee does not generally remain constant for satellites in elliptic orbits. As an example, one Phase III orbit under consideration (period = 11 hours, eccentricity = .688, inclination = 102°) has a rate of change of latitude of apogee of slightly less than 4° per month. A single ground track overlay based on the mean value for the latitude of apogee for a 30 day period will give relatively accurate data. So, to make a calculator for this orbit, we use a polar projection map and change the ground track overlay every thirty days. A set of tables using appropriate increments in latitude of apogee, each similar to Table 2.4, will be published soon after a Phase III spacecraft is in orbit. These tables will enable one to construct accurate ground track overlays using any type of map or globe.

One possible orbit for a Phase III satellite is described in Figure 2.8. Ground track data for this orbit is presented in Table 2.6.



Period: 11 hours

Eccentricity: 0.688

Inclination: 101.7°

Figure 2.8 Possible Phase III orbit (approximately to scale).

θ' (°)	t (min)	r(θ') (km)	h(θ') (km)	s(θ') (km)	Argument of Apogee 90° Lat.	90° Long.	60° Lat.	60° Long.	30° Lat.	30° Long.
-180	-330.0	7811	1440	3930	-78.3	262.5	-58.0	262.5	-29.3	262.5
-170	-327.5	7860	1490	3990	-74.7	220.9	-67.0	252.1	-39.0	258.9
-160	-325.0	8008	1640	4140	-67.0	200.4	-74.7	231.7	-48.6	254.4
-150	-322.3	8262	1890	4400	-58.0	190.0	-78.3	190.0	-58.0	247.9
-140	-319.5	8634	2260	4720	-48.6	183.5	-74.7	148.2	-67.0	237.5
-130	-316.3	9142	2770	5090	-39.0	178.8	-67.0	127.6	-74.7	216.8
-120	-312.7	9810	3440	5500	-29.3	174.9	-58.0	115.8	-78.3	173.8
-110	-308.5	10673	4300	5930	-19.6	171.3	-48.6	110.0	-74.7	132.8
-100	-303.4	11777	5410	6370	-9.8	167.9	-39.0	104.9	-67.0	111.7
-90	-297.2	13184	6810	6790	0.0	164.3	-29.3	100.3	-58.0	100.3
-80	-289.2	14973	8600	7210	9.8	160.3	-19.6	95.9	-48.6	92.6
-70	-278.8	17241	10870	7600	19.6	155.5	-9.8	91.1	-39.0	86.0
-60	-264.8	20098	13730	7950	29.3	149.5	0.0	85.6	-29.3	79.6
-50	-245.6	23638	17270	8270	39.0	141.7	9.8	78.7	-19.6	72.3
-40	-218.9	27876	21510	8540	48.6	131.1	19.6	69.8	-9.8	63.4
-30	-182.0	32620	26250	8750	58.0	116.1	29.3	58.2	0.0	52.2
-20	-132.5	37298	30930	8910	67.0	94.0	39.0	42.8	9.8	37.7
-10	-70.4	40888	34520	9010	74.7	58.6	48.6	23.4	19.6	20.1
0	0.0	42258	35890	9040	78.3	0.0	58.0	0.0	29.3	0.0
10	70.4	40888	34520	9010	74.7	-58.6	67.0	-27.4	39.0	-20.6
20	132.5	37298	30930	8910	67.0	-94.0	74.7	-62.7	48.6	-40.0
30	182.0	32620	26250	8750	58.0	-116.1	78.3	-116.2	58.0	-58.2
40	218.9	27876	21510	8540	48.6	-131.1	74.7	-166.4	67.0	-77.1
50	245.6	23638	17270	8270	39.0	-141.7	67.0	-192.9	74.7	-103.7
60	264.8	20098	13730	7950	29.3	-149.5	58.0	-207.5	78.3	-149.5
70	278.8	17241	10870	7600	19.6	-155.5	48.6	-216.8	74.7	-194.0
80	289.2	14973	8600	7210	9.8	-160.3	39.0	-223.3	67.0	-216.5
90	297.2	13184	6810	6790	0.0	-164.3	29.3	-228.3	58.0	-228.3
100	303.4	11777	5410	6370	-9.8	-167.9	19.6	-232.3	48.6	-235.8
110	308.5	10673	4300	5930	-19.6	-171.3	9.8	-235.7	39.0	-240.8
120	312.7	9810	3440	5500	-29.3	-174.9	0.0	-238.9	29.3	-244.8
130	316.3	9142	2770	5090	-39.0	-178.8	-9.8	-241.8	19.6	-248.2
140	319.5	8634	2260	4720	-48.6	-183.5	-19.6	-244.8	9.8	-251.2
150	322.3	8262	1890	4400	-58.0	-190.0	-29.3	-247.9	0.0	-253.8
160	325.0	8008	1640	4140	-67.0	-200.4	-39.0	-251.6	-9.8	-256.7
170	327.5	7860	1490	3990	-74.7	-220.9	-48.6	-256.1	-19.6	-259.4
180	330.0	7811	1440	3930	-78.3	-262.5	-58.0	-262.5	-29.3	-262.5
position on orbit	time from apogee	radial distance	height	access range						

R_e (radius of earth) = 6371 km

a (semi-major axis) = 3.93 R_e

r_a (apogee radius) = 6.64 R_e = 42300 km

r_p (perigee radius) = 1.23 R_e = 7820 km

Table 2.6 Data for preparing AMSAT-OSCAR Phase III orbit overlays for orbit of Figure 2.8 (T = 11 hours, e = .688, i = 101.7°). Ground tracks for arguments of apogee of 30°, 60° and 90° included.

Ground track overlays for latitudes of apogee of 78°N (argument of apogee = 90°) and 58°N (argument of apogee = 60°) are shown in Figure 2.9b. The overlays are scaled to polar graph paper (Figure 2.9a). Notice the tick marks on the ground track which indicate time from apogee. In a working orbit calculator the ground track would actually be drawn on a transparent overlay which could be rotated.

To prepare your own orbit calculator, obtain a polar projection map (AEPP or SPP) and draw constant distance circles about your ground station as shown in Figure 2.9a using 4, 4.5, 5, 6, 7, 8, and 9 x 10^6m radii. This part of the calculator is never changed. A new transparent ground track overlay will generally need to be drawn every thirty days (data for drawing the overlay will be supplied by AMSAT). To use the calculator set the overlay so that the position of the longitude of apogee lines up with the true apogee longitude published for the orbit. Time, latitude and longitude of apogee, and possibly other orbital data for each orbit will be distributed by AMSAT once a satellite is launched. The time marks on the ground track overlay show where the sub-satellite point will be at various times before and after apogee. A simple but useful ground track calculator can be constructed using polar graph paper instead of the polar projection map as in Figure 2.9. We turn now to the problem of determining when the satellite will be within range.

Determining when the satellite will be within range is a special case of the more general problem of obtaining the elevation angle from one's ground station to the satellite. We say that the satellite is in range when this elevation angle is greater than zero. The elevation angle of the satellite depends on two factors: (1) the surface distance between ground station and sub-satellite point and (2) the altitude of the satellite. For a satellite in a circular orbit the altitude is constant so we can plot elevation angle vs. surface distance as in Figure 2.2. The surface distance for zero elevation is then taken as the radius of the acquisition circle. However, with elliptical orbits, one does not have a constant acquisition circle because the altitude is changing. For a specific satellite, altitude is a function only of time from apogee. As a result, one can plot elevation angle vs. (1) surface distance and (2) time from apogee, parameters which can be read directly from the ground track calculator. In Figure 2.10 we present the satellite elevation angle vs. surface distance and time from apogee for the possible Phase III orbit described in Figure 2.8 and Table 2.6. Figure 2.10 was constructed from Eqs. 1.11, 1.14 and 1.15.

Figure 2.10 may be used in conjunction with the orbital calculator to determine the elevation angle to the satellite and azimuth of the sub-satellite at a specific time.

1. Use ground track plotter (Figure 2.9) to:
   a. locate position of, and azimuth towards, sub-satellite point,
   b. estimate surface distance between ground station and sub-satellite point,
   c. obtain time from apogee;
2. use graph (Figure 2.10) to obtain elevation angle.

That's all there is to it!

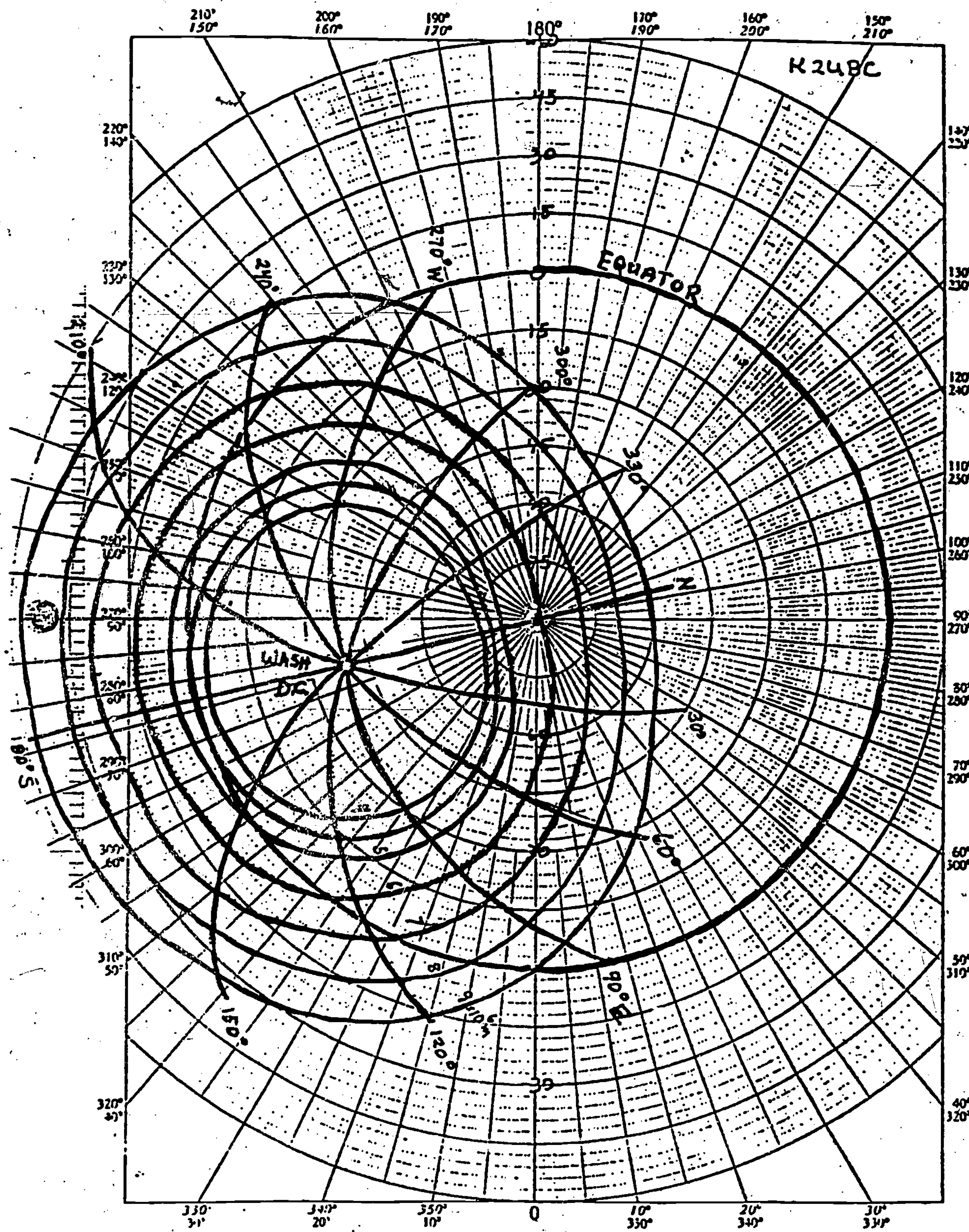


Figure 2.9a. Azimuthal equidistant polar projection map with range circles drawn about Washington, D.C.

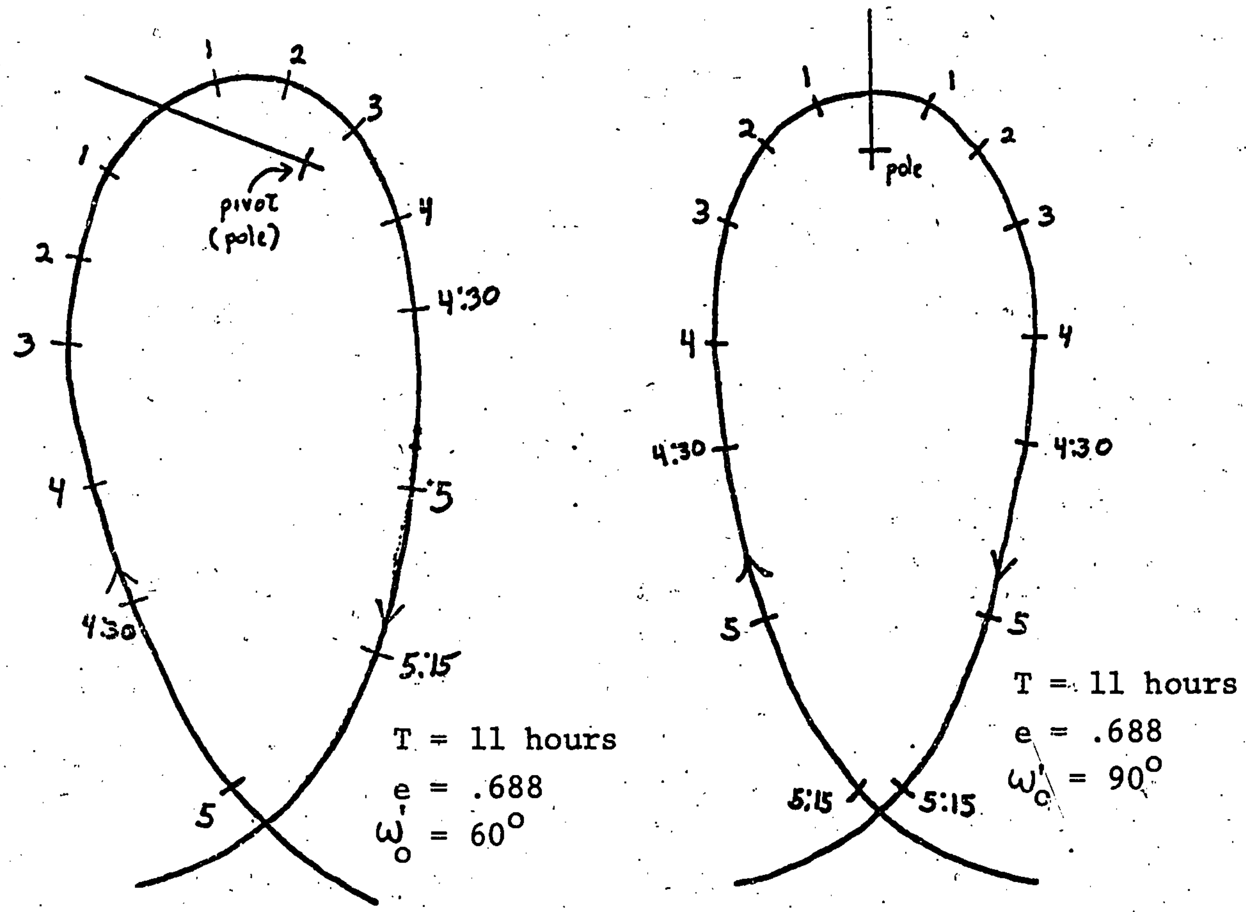


Figure 2.9b Orbit overlays for use with Figure 2.9a (Phase III orbit described in Figure 2.8 and Table 2.6). Overlays for arguments of apogee of 60° and 90°.

A modification of the technique just described makes it possible to determine when the satellite will be in range without using the graph of Figure 2.10. The method involves color coding (1) the ground track overlay and (2) the constant distance circles drawn about one's ground station. By using the standard radio component color code an integer can be associated with each color. The color code of Table 2.7 is useful for the orbit we've been working with (Figure 2.8). First, we draw constant distance circles on the map using the color code. Next, the orbit is divided into segments and the color code is used to indicate the minimum communications distance during each segment. To preview an orbit the overlay is positioned as before. When the integer associated with the ground track overlay color is greater than, or equal to, the integer associated with the range circle color, the satellite is in range. AOS and LOS times can therefore be read directly off the plotter without referring to the graph of Figure 2.10 or the key of Table 2.7.

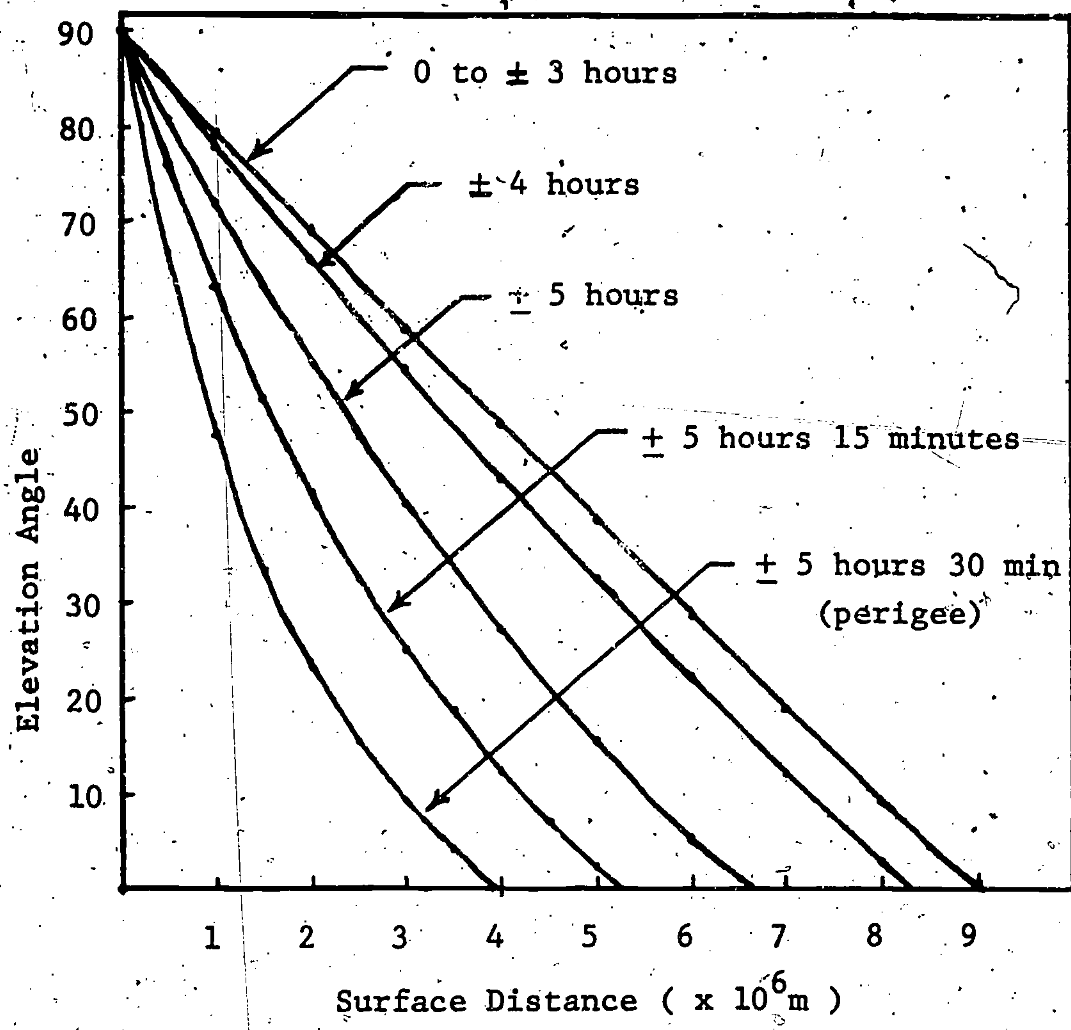


Figure 2.10 Elevation angle vs. surface distance and time from apogee for Phase III orbit of Figure 2.8.

Time From Apogee (absolute value) less than	Minimum Communication Range	Color Code
71 minutes	9,000. km	blue
260 minutes	8,000. km	green
293 minutes	7,000. km	yellow
308 minutes	6,000. km	orange
317 minutes	5,000. km	red
322 minutes	4,500. km	brown
327 minutes	4,000. km	black

Table 2.7

Color code used with AMSAT Phase III orbit calculator for simplifying prediction of AOS and LOS times. (Orbit of Figure 2.8).

2.9 COMMENTS

This chapter has outlined practical tracking methods suitable for (1) satellites in circular orbits (sections 2.3 - 2.7) such as OSCAR 7, OSCAR 8, and RS 1 and (2) satellites in elliptical orbits (section 2.8) as proposed for AMSAT Phase III.

Although the techniques have been illustrated with specific satellites they can be used with any spacecraft if information in the form of Table 2.4 or Table 2.6 is available or by computing equivalent information from the orbital elements.

The Selected Cities method is only suitable for casual listening since it provides almost no information on the ground track. The map based methods all work well -- orbital calculators using them are simple to construct and use and they provide relatively accurate results. The globe and computer based methods are most suitable for certain special applications. Nothing beats a globe for helping students visualize satellite motion in three dimensions.

For most tracking applications I prefer the method based on the north pole stereographic projection map. It is suitable for circular and elliptical orbits. Special circumstances or projects may make one of the other techniques more convenient.

The Orbital Calendar, mentioned in section 2.1, with data for all ascending nodes during the year is the most convenient source for this information. It enables one to plan demonstrations and laboratories far in advance without having to pause for calculations -- assuming that the date of switching from daylight to standard time is not varied. According to the Uniform Time Act of 1966 daylight savings time is observed for six months each year extending from the last Sunday of April to the last Sunday of October. Arizona, Hawaii and Michigan do not conform.

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Chapter II

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CHAPTER III

SATELLITE SYSTEMS

The objectives of this chapter include:

1. Providing a general introduction to scientific and communications satellites;
2. Discussing the functions and design options for each satellite subsystem;
3. Describing the subsystem design approaches selected by AMSAT and the tradeoffs involved.

3.1 SATELLITE SYSTEMS: OVERVIEW

A satellite is a complex collection of hardware. When designing communications and scientific satellites it's usually convenient to think of them as being composed of a standard set of subsystems each with a specific task [1]. See Table 3.1. One then analyzes and optimizes each of the subsystems. There are a number of design objectives which apply to almost all spacecraft subsystems: minimizing weight and cost, maximizing reliability and performance, and insuring compatability. Since these objectives frequently conflict it is important to constantly keep in mind how each subsystem impacts on the others so that effective tradeoffs can be made. For example, if state-of-the-art battery lifetime is of the order of five years and power available from the solar panels is expected to drop to a marginal level after about the same period of time, it would be foolish to invest a great deal of expense or effort trying to design other systems to last much longer. So, the first step in satellite design is system engineering -- specifying the major overall system parameters. These include: mission subsystem requirements, size and weight limits for the spacecraft, projected orbit, minimal design lifetime which all subsystems will be expected to attain, and the funds available for the project.

Next, all the subsystems are designed, constructed, tested and refined. Electronic subsystems usually are built in a number of versions: an engineering development model, a flight prototype, and the flight unit which uses the highest reliability components available. All the subsystems are then integrated into the spacecraft which is then checked for proper operation and intersystem compatability.

After passing the preliminary operational tests the satellite is subjected to a series of stress-tests and operational-checks -- the spacecraft is stressed under harsh conditions and its operation completely checked. The object of the series of tests and checks is to try and insure that any potential problems will turn up while the satellite is still on the ground where repairs are relatively easy. The procedure is based on the failure curves applicable to many electrical and mechanical components --

starting off relatively high the curves quickly level off at a considerably lower value.

The stresses include: a burn-in period for electronics systems during which electrical parameters and temperatures are similar to those expected in space but with the system at atmospheric pressure; environmental tests which involve operating the spacecraft in a vacuum chamber -- with temperature extremes considerably more severe than those expected in space (for example, -20°C and +60°C); and a vibration test to insure that the satellite will make it through the launch. The objectives of the vacuum test include (1) checking for material sublimation, which could result in the contamination of spacecraft subsystems, and corona discharge, and (2) verifying the predicted thermal behavior in the absence of convective heat flow.

Now we briefly look at the construction of a satellite from a project management perspective. The procedure can be roughly divided into six stages: (1) preliminary design, (2) system specification, (3) subsystem design and fabrication, (4) integration and testing, (5) launch operations, and (6) information dissemination and post launch management. The time frame for these activities is roughly outlined in Figure 3.1. The preliminary design stage involves feasibility studies related to new approaches to satellite design. State-of-the-art advances in electronics, cost reductions in components, launch access to unusual orbits, sources of financial support, etc. continually open up new design options. In a long term program where many satellites are being constructed feasibility studies are going on continually. At some point in time the decision is made to construct a satellite. A set of system specifications must then be agreed upon and the subsystem requirements defined. Subsystems are then designed, built and tested. With AMSAT satellites subsystem construction is by small volunteer groups, usually from a number of countries. Next, the subsystems are integrated into a spacecraft and the stress-tests and operational checks are performed. When the satellite is performing satisfactorily it is transported to the launch range, attached to the launch vehicle, and tested one last time. The project doesn't end with the launch. Information must be disseminated to users, command stations must be available when and where needed, and data on spacecraft operation must be collected to assist in the design of future spacecraft. The entire procedure, from system specification to launch, can take anywhere from nine months to five years depending on the complexity of the spacecraft and the available personpower.

Problem 3.1.

How many interfaces are there between n subsystems?

Answer: Since order is not important, this is a problem in combinations. There are n(n-1)/2 possible combinations of two items taken from a group of n.

The remainder of this chapter presents an overview of satellite subsystems focusing on systems from Table 3.1 which have immediate or potential educational applications. For each subsystem, we look at some of the methods which can be used to accomplish the system objectives,

Subsystem	Function	OSCAR Series Equipment
Attitude-control	To sense and modify satellite orientation	Phase II: magnets; Phase III: sun and earth sensors, solid propellent spin motors, torquing coils
Communication	To receive uplink commands and to transmit downlink telemetry	command receivers, transmitters (beacons)
Computer (housekeeping)	To coordinate and control other subsystems; provides memory, computation capability	digital logic, microprocessor
Energy-supply	To provide power for all onboard subsystems	batteries, solar cells, conditioning electronics
Telemetry	To measure operating status of onboard subsystems	electronic sensors, telemetry encoders
Environment control	To regulate temperature levels, provide electromagnetic shielding	mechanical design, thermal coatings
Guidance-and-control	To interface computer with other subsystems	hardwired electronics
Mission unique equipment	To accomplish mission objectives	transponders, scientific and educational instruments
Propulsion	To provide thrust for orbit changes	Phase II: none; Phase III: kick motor, ignition system
Structure	To provide support and packaging function, thermal control, protect modules from stress of launch	mechanical structure

Table 3.1. Satellite subsystems: emphasis OSCAR.

consider the important tradeoffs involved, and discuss the particular approaches selected for AMSAT spacecraft. In Chapter IV satellites currently in orbit and available for use or planned for the near future are described in detail. Ground station equipment needed to monitor the OSCAR communications links is discussed in Chapter V. Suggested classroom experiments based on information in this chapter are presented in Chapter VI.

3.2 COMMUNICATIONS, MISSION AND ENGINEERING SUBSYSTEMS

The communications subsystem provides a direct link to the satellite, enabling us to (1) observe what is happening inside the spacecraft as it happens (via telemetry) and (2) to make changes in the control logic aboard the spacecraft. The immediacy of this link generates a great deal of student interest and excitment. While it is possible to provide students with telemetry information in a "non real-time" format -- i.e., a cassette recording of satellite signals made by someone else or a decoded and printed table of telemetry -- this approach is a pale second insofar as generating interest is concerned.

There are three communications links involving the OSCAR satellites of interest to educators: (1) downlinks (signals generated aboard the satellite which are transmitted to earth -- beacons), (2) uplinks (signals generated by ground stations and directed at the satellite for control purposes -- command signals), and (3) communications or broadcast links (signals generated by ground stations and retransmitted by the satellite). See Figure 3.2. We now turn to the functions performed by beacons, command links, and transponders (devices which receive and retransmit signals).

Beacons: Function

The beacons aboard the OSCAR satellites serve a number of functions -- (1) in the telemetry mode they convey information about onboard satellite systems (for example: solar cell panel currents, temperatures at various points, storage battery condition, etc.); (2) in the Codestore mode they can be used for a special form of delayed communication; (3) in either the telemetry or Codestore mode they can be used for Doppler studies (see Chapter VI, experiment SE 2), propagation measurements, and as a reference signal of known characteristics. Beacon functions are listed in Table 3.2.

1. Telemetry
    a. Morse code
    b. Radioteletype (RTTY)
    c. Advanced encoding techniques
2. Communications
    a. Codestore
3. Miscellaneous
    a. Doppler studies
    b. Propagation measurements
    c. Reference signal

Table 3.2. Beacon functions. Each OSCAR satellite includes some, but not necessarily all, of the telemetry options listed.

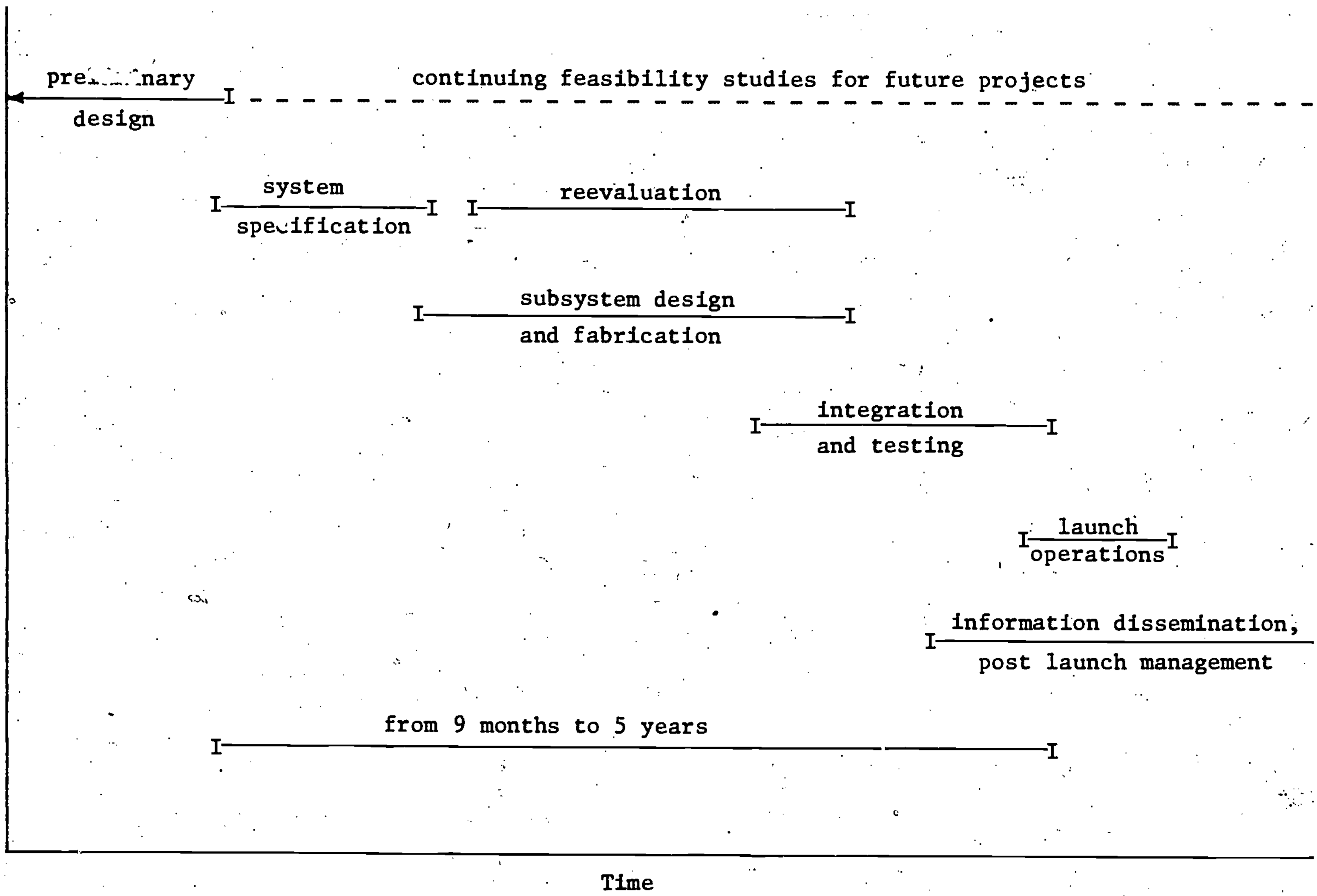


Figure 3.1. Timeframe for stages of satellite construction.

Beacon telemetry. From the users point of view, each telemetry mode can be characterized by (1) the capacity (the amount of information that can be transmitted in a given time interval) and (2) the complexity of the decoding equipment required at the ground station. There is, to a certain extent, a tradeoff between these two factors as summarized in Table 3.3.

Telemetry Encoding Method	Relative Ground Station Complexity	Telemetry Capacity
Morse code	low	low
Radioteletype	moderate	moderate
Advanced encoding techniques	high	high

Table 3.3. Telemetry encoding methods and the relative data capacity and ground station complexity associated with each.

We now look at each of the beacon operating modes in greater detail.

Morse code telemetry: The Morse code telemetry system is one of the ingenious features that make the OSCAR series of satellites so valuable to educators [2]. In the Morse code telemetry mode information on satellite systems is sent in the form of Morse code using a numbers only format, usually at either 25 or 50 numbers/minute (about 10 or 20 words/minute). Because of the regularity and redundancy of the Morse code numbers (see chart, section 6.4) students can learn to decode Morse code telemetry after a short period of practice. A multispeed tape recorder (record at high speed, play back at low speed) makes decoding even easier. Because AMSAT put the telemetry information processing equipment aboard the satellite, ground stations do not need any specialized decoding electronics. The information capacity of this mode is inherently limited in that any attempt to speed up the Morse code transmission would interfere with the ability of untrained users to decode it without special equipment.

Radioteletype telemetry: Standard radioteletype equipment is readily available, often at low cost through the used equipment market (see Chapter 5). AMSAT therefore felt that information in this format could provide data at a moderate speed to a large number of users. Radioteletype telemetry is most useful to the advanced experimenter, to stations managing satellites, and to the scientists designing and building future spacecraft.

Advanced encoding techniques: A number of advanced techniques for encoding telemetry, including various forms of pulse code modulation (PCM) can be used to transmit large amounts of data in a short period of time. One PCM encoding scheme known as ASCII (American Standard Code for Information Interchange) is especially useful because it enables ground stations to handle the telemetry on standard data processing equipment with a minimal amount of interfacing hardware or software. In the past these

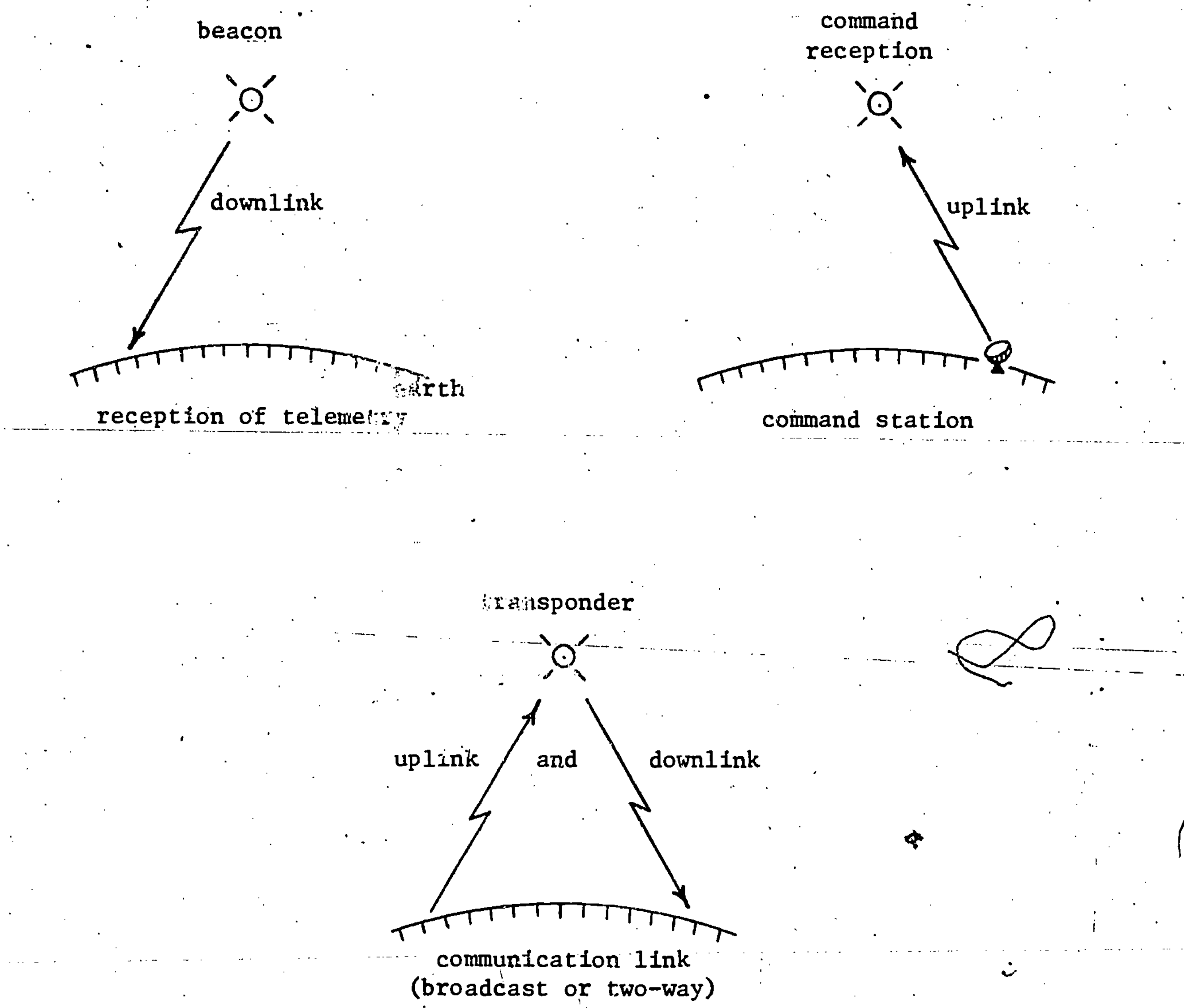


Figure 3.2. Radio links involving OSCAR satellites.

advanced modes have not been used on AMSAT satellite beacons because of the specialized equipment needed on the satellite and at the ground station. However, recent advances in microprocessor technology make it likely that these modes will replace radioteletype telemetry on future (post 1978) AMSAT spacecraft. Phase III satellites will have onboard computers which can be instructed from the ground to use ASCII, Morse or other codes. Present plans are to simultaneously use Morse code telemetry on one or more educational beacons and ASCII or some other code on an engineering beacon. The performance of telemetry formats has been operationally tested by generating a signal on the ground and relaying it through transponders aboard operating spacecraft.

Beacon communication mode (Codestore). The beacon Codestore mode relies on a digital memory system aboard the satellite which can be loaded by suitably licensed and equipped ground stations for later rebroadcast in the form of Morse or other codes. The system has proved very useful for disseminating information to the world wide network of OSCAR users and command stations.

Beacon: Miscellaneous Functions. In either the telemetry or Codestore modes a beacon, with a well characterized intensity and frequency, can serve a number of useful functions. For example, it can be used for Doppler shift studies, propagation measurements, and testing ground based receiving equipment. In addition, stations communicating via a satellite transponder can optimize their uplink power levels so that the strength of their downlink signal, relative to the beacon, is at the desired level.

Beacons: Design

Engineering beacon power levels are chosen to provide adequate signal to noise ratios at well equipped ground stations. Overkill (too much power) only serves to (1) decrease power available for other satellite subsystems -- especially the transponder, (2) reduce reliability, (3) lead to potential compatability problems with other spacecraft electronics systems. Educational beacons generally require a higher power level so that they can be used in conjunction with relatively simple ground station equipment. Since beacons are often used for Doppler studies, frequency stability over a wide range of temperatures and battery conditions is an important requirement. Although some saving in satellite complexity and weight can be achieved by using one power amplifier common to both the beacon and transponder this approach has undesirable consequences -- beacon power output is no longer constant and system reliability is degraded. If possible, telemetry systems should include provisions for monitoring beacon power output. Beacon reliability is usually enhanced by including redundant systems, often at different frequencies. Beacon frequencies are usually set just outside of the transponder downlink passband -- a location convenient to both users (the same ground station receiving system can be used for both downlinks) and to spacecraft designers (the same satellite antenna can serve both systems). As with all spacecraft subsystems, a high power efficiency is very important.

Command Links

The OSCAR satellites have been designed so that cooperating ground stations with the necessary equipment can command them from one operating mode to another. The ability to command the satellites is both a necessity and a convenience. It is a necessity because AMSAT must be able to turn off a malfunctioning transmitter since it could conceivably cause harmful radio interference to important services worldwide. The ability to command satellites greatly enhances their utility. Subsystems which are not working properly can be turned off, operating schedules can be adjusted to suit changing needs of users, telemetry modes can be switched when required, etc. Intensive commanding of the AMSAT-OSCAR 6 spacecraft was probably a significant factor in its serving for approximately 4.5 years even though the design lifetime was only one year. Command stations are constructed and manned by dedicated volunteers. Command frequencies, access coding, and formats are considered confidential. However, they are available to responsible stations for projects coordinated with AMSAT. To date, command stations have operated in more than 8 countries.

Although the OSCAR operating schedules have been designed to facilitate general educational use, educators may find that a special event or demonstration necessitates a special operating feature which is not scheduled to be on. If this happens, the desired change can usually be arranged by contacting the OSCAR operations office as described later in this section under Operating Schedules.

Transponders: Function

A transponder is a device which receives signals in a narrow slice of the radio frequency spectrum, amplifies them, translates (shifts) their frequency, and then transmits them. The transponders aboard AMSAT satellites are the primary mission unique subsystem. To transmit signals to the satellite, one needs a government issued license. In the United States. these are Amateur Radio Service licenses issued by the Federal Communications Commission to individuals who pass appropriate exams. Note that the applications in this book stress receiving only experiments which do not require any license. However, the ability to transmit does make a larger variety of experiments possible. Students can legally use the transponder if they are being supervised by an instructor who is licensed. The translators currently in orbit are linear in the sense that any type of signal put in (single sideband, FM, CW, AM, facsimile, slow scan television, etc.) comes out the same except for the shift in frequency and the great amplification -- of the order of 10^{12} (120 dB). Because all stations using the transponder must share the limited power available, the general use of modes such as FM and AM with carrier, which are inefficient with respect to power consumption, is discouraged. Listening to the transponders one often hears stations discussing new equipment being tested or experiments in progress such as bouncing radio signals off the moon. In addition to two-way communications, the transponders are also used at certain times for special one-way educational broadcasts. These broadcasts are especially well suited to demonstrations where the group can be greeted by name and the satellite and its path can be described via s[illegible]llite. In addition, one-way broadcasts of general interest to satellite users are currently scheduled on reference orbits (the first orbit each UTC day) on OSCAR 8.

Transponders: Design

Transponder design is, in many ways, similar to HF (High Frequency) receiver design. For a low orbiting satellite, input signals are often of the order of 10^{-12} watts and the output level is of the order of 1 watt. However, the satellite output is at radio frequencies while the receiver output is at an audio frequency. The normal convention is to specify a transponder by first giving the approximate input frequency and then the output frequency. For example, a 146/29 MHz transponder will have an input frequency passband centered near 146 MHz and an output frequency passband centered near 29 MHz. Often, wavelength is used instead of frequency and the same transponder would be referred to as a 2m/10m unit. A block diagram of a simple transponder is shown in Figure 3.3. For a

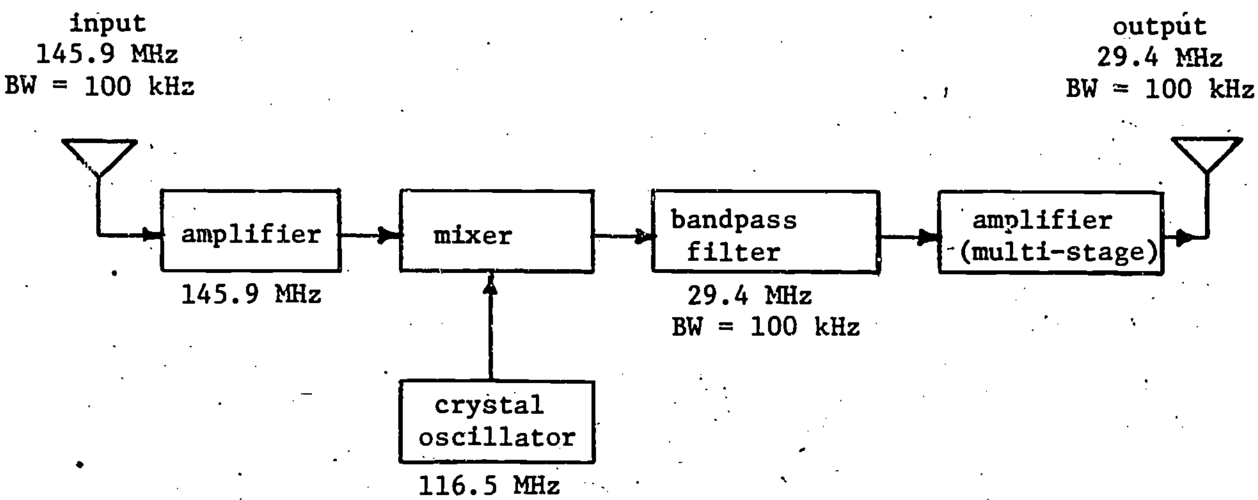


Figure 3.3. Block diagram of a simple 2m/10m linear transponder: input passband 145.850 - 145.950 MHz, output passband 29.350 - 29.450 MHz.

number of reasons, flight model transponders are more complex than the one shown. As with receiver design, considerations related to band pass filter availability, image response, and required overall gain often lead to a multiconversion approach.

<u>Linear vs. nonlinear transponders</u>. If a transponder only (1) shifts the frequency of all incoming signals by a certain fixed amount and (2) amplifies them, it is called a <u>linear transponder</u>. Such a transponder will accept input signals of any mode and transmit them in the same mode -- on a different frequency and at a higher power level. Undesired sum and difference frequencies resulting from input signals should be down by 30 dB or more.

One way to build linear transponders is to use linear amplifiers and mixers for all stages. ~~However, linear~~ amplifiers are not very efficient, a fact which creates serious problems aboard a spacecraft. A special technique for constructing high efficiency linear transponders was developed by Dr. Karl Meinzer to overcome this problem [3]. Although a number of stages in the transponder developed by Dr. Meinzer do not operate in a linear mode the overall transponder is a linear device. His technique is usually employed when transponder output power requirements rise above about 4 watts.

Inverting vs. non-inverting transponders. In any multiconversion mixing scheme the various local oscillator frequencies can each be above or below the incoming frequency. If the local oscillator frequencies are chosen so that signals entering a linear transponder are inverted before being retransmitted we have an inverting transponder. Such a transponder will change upper sideband signals into lower sideband (and vice-versa) and transpose relative mark-space placement in RTTY, etc. An important advantage of an inverting transponder is that Doppler shifts on the uplink and downlink are in opposite directions. As a result they will, to a limited extent, cancel. Using the 146/29 MHz link combination Doppler is not serious so transponders using this frequency combination have been non-inverting. Transponders using higher frequencies are usually inverting.

Power - Bandwidth - Frequencies. The power, bandwidth, and frequencies of a transponder must be compatable -- i.e., when the transponder is fully loaded with equal strength signals each should provide an adequate signal to noise ratio at the ground. It is difficult to select appropriate values with the needed accuracy on a purely theoretical basis. However, experience with a number of satellites has provided AMSAT with a great deal of practical data from which it is possible to accurately scale to different orbits, bandwidths, power levels, frequencies, and antenna systems using the "channel transmission equations" [4] and to different modulation schemes by employing information theory concepts [5]. Linear amplifiers are characterized by their Peak Envelope Power (PEP) output -- the maximum instantaneous power output which can be produced without exceeding a specified distortion level.

In general, low-altitude (800 to 1,600 km) satellites using passive magnetic stablization and roughly omni-directional antennas (over at least a hemisphere) can provide reasonable downlink performance with from 1 to 4 watts PEP at frequencies between 29 and 435 MHz using a 100 kHz transponder. A high-altitude (35,000 km) spin-stabilized satellite employing modest (7-10 dB_I) gain antennas should be able to provide acceptable performance with 35 watts PEP using a 100 kHz transponder downlink at 146 or 435 MHz. Path loss and satellite antenna pattern constraints (see Chapter 6, Project SP 14) strongly favor 146 MHz for the downlink and 435 MHz for the uplink. The difficulty of employing a gain antenna on the satellite at 29 MHz makes this frequency unsuitable for the downlink aboard a high-altitude spacecraft.

Dynamic range. The dynamic range problem for transponders is quite different than for HF receivers. At first glance it may seem that satellite transponders pose a simpler problem. After all, an HF receiver is designed to handle signals differing by as much as 100 dB while a low-altitude satellite will only encounter signals in its passband differing by perhaps 35 dB. Good HF receivers solve the problem by filtering out all but the desired signal before employing significant gain. A satellite, however, has to accommodate all users simultaneously. The maximum overall gain is therefore determined by the strongest signal in the passband. Considering the state-of-the-art in transponder design and available power budgets aboard the spacecraft it seems that an effective dynamic range between 20 and 25 dB is about the most that can be currently obtained. If the AMSAT satellite program were to continue to emphasize low-altitude satellites this problem would merit a great deal of attention. However, the emphasis will be shifting to higher altitude satellites where 20 dB dynamic range should be adequate.

The practical implications of the dynamic range limitations are: If the strongest signal in the passband is driving the satellite transponder to full power output than stations down more than about 22 dB from this level will not be heard on the downlink, even though these weaker stations might be perfectly readable if the strong signal were not present. Any attempt to remedy this problem by changing the transfer characteristic of the transponder from linear to logarithmic would cause intermodulation problems between stations using the transponder which would serve to decrease the dynamic range.

Transponder design is an interesting area for innovative developmental work. Topics deserving attention include (1) increasing effective dynamic range by using channelized transponders (linear and non-linear) where each channel has its own automatic gain control, (2) evaluating the potential utility of limiting type repeaters suitable for FM voice and/or digital signals.

Redundancy. Since the transponder is the primary mission subsystem reliability is extremely important. One way of enhancing reliability is to include two transponders on each spacecraft -- if one fails, the other will be available full time. Rather than use identical units it is usually advantageous to work with different frequency combinations. This enables AMSAT to acquire practical data on the performance of different link frequencies and to allow for the fact that during the period of time between satellite conception and the later part of the satellites useful life (a period on the order of seven years for AMSAT-OSCAR 6) there may be drastic changes in the availability of equipment to users. For example, in 1972 when planning for AMSAT-OSCAR 7 began, 432 MHz power amplifiers were not being produced commercially for the amateur market. However, theoretical predictions of link performance led AMSAT to include a 432/146 MHz transponder on this spacecraft (in addition to a 146/29 MHz unit). In 1978 a large number of commercial amplifiers from at least six manufacturers at power levels ranging from 10 to 1,000 watts are available. The excellent performance of the 432/146 MHz unit and the increased equipment availability have led AMSAT to schedule this transponder for operation 67% of the time.

Telemetry Systems

The telemetry system gathers information about all onboard subsystems and encodes the data in a format suitable for downlinking (engineering subsystem) and then transmits it via the spacecraft beacons (communications subsystem). In this section we look at engineering aspects of the telemetry subsystem. A block diagram of a typical telemetry encoding system is shown in Figure 3.4. Each parameter of interest aboard the spacecraft is monitored by a sensor and associated electronics having a voltage output with an appropriate transfer characteristic. The sensor signal passes through a variable gain amplifier and into an analog-to-digital converter. The digital output is then changed into Morse code, RTTY or some other format for transmission via a beacon. Phase II AMSAT satellites have used hardwired logic to convert the output of the analog-to-digital converter to Morse code or RTTY. Phase III satellites will probably employ software in the resident computer for this purpose. Telemetry control logic (either

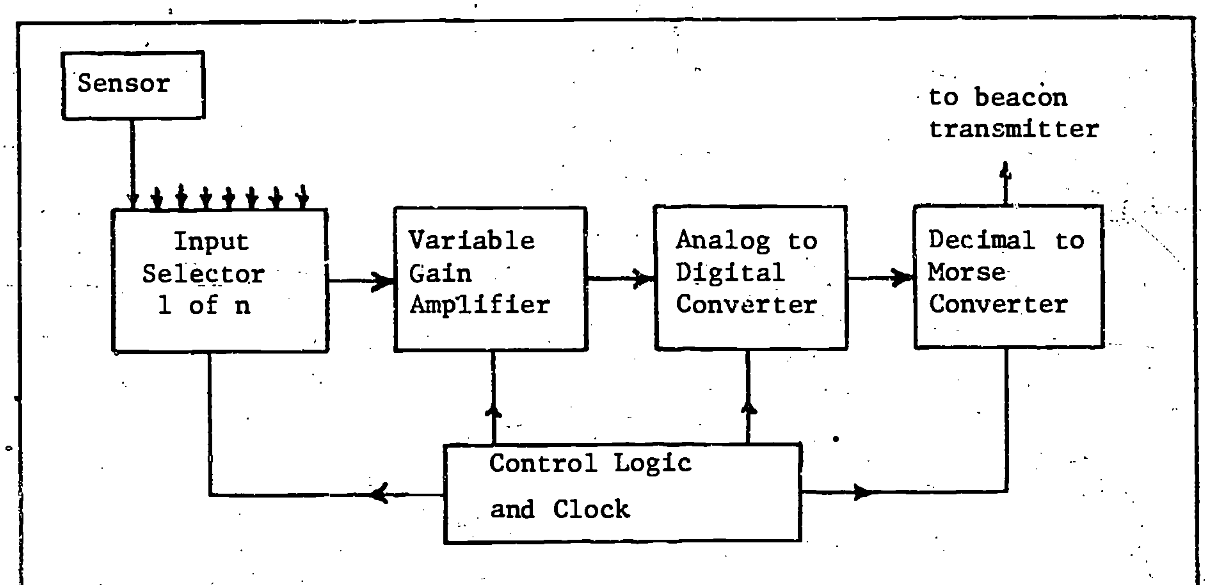


Figure 3.4. AMSAT-OSCAR Morse code telemetry encoder.

hardwired or software) takes care of selecting the proper input sensor, choosing the appropriate amplifier gain, and other bookkeeping chores.

The sensors are usually sampled sequentially (serial mode) and the measurements are transmitted as they are made. Some satellites can be commanded to stick with a particular sensor (dwell mode) so that short term changes can be studied. The capabilities of each satellite are described in Chapter IV. In general, the control logic on future satellites will be handled by software which can be programmed from the ground providing AMSAT with a great deal of flexibility in telemetry encoding.

<u>Morse code format</u>. The Morse code telemetry systems aboard OSCAR satellites have a number of features in common. The parameters being measured are sampled in a fixed serial mode. One complete series of measurements is called a <u>frame</u>. The beginning and end of each frame are marked by a distinctive signal -- the letters HI in Morse code (·•·•· ··) are used on OSCAR 7 and OSCAR 8. Each measurement consists of three integers called a <u>channel</u>. To interpret a channel we need to (1) identify the parameter being monitored and (2) obtain a raw data measurement which can be converted into a meaningful value. AMSAT uses the first integer in the channel for parameter identification. When the number of channels is small (OSCAR 8 has six channels) a single digit can uniquely identify the parameter being measured. When the number of channels is large [illegible] user must also attend to the order in which the channels are being [illegible] in order to identify the parameter being measured. As an example [illegible] telemetry frame for an imaginary satellite might consist of nine channels as shown in the top row of Figure 3.5. Channel identity information, shown in the bottom row of Figure 3.5, takes into account the order in which the channels are transmitted to uniquely label them. The last two digits in each channel

<u>Raw data</u>	(begin) HI	142	116	178	239	202	216	392	352	365	HI	(end)
<u>Channel ID</u>		1A	1B	1C	2A	2B	2C	3A	3B	3C		

Figure 3.5. A Morse code telemetry frame with nine channels. The top row is the actual data as received. The bottom row assigns a unique label to each channel. Channel 1A is the first one received, channel 1B is the second, 1C is the third, 2A is the fourth, etc. The data is sometimes written in the form of a 3 x 3 matrix in which case the ID integer is a line number and the ID letter is a column label.

encode the information of interest. To decode a telemetry channel one refers to the specific satellite in Chapter IV to determine the parameter measured by the given channel and to obtain the simple algebraic equation relating the last two digits in the channel to the quantity being measured. For example, the spacecraft description for the imaginary satellite which transmitted the data in Figure 3.5 might inform us that channel 1A is total solar panel current and that multiplying the significant digits (42) by thirty will yield the current in milliamperes (1,260 ma).

<u>Advanced telemetry formats</u>. Radioteletype and other advanced telemetry systems aboard OSCAR spacecraft can generally be operated in a serial mode, a dwell mode, or a combination of the two where important data are frequently sampled. With these modes each channel can include the information needed to uniquely identify the parameter being measured. So, unlike the Morse code telemetry system, knowledge of the location of a

channel within a frame is not very important to users. Since these modes transmit data at a relatively high speed more accurate data can be sent down to earth in a reasonable period of time by using three digits to encode measured levels. Decoding information for each satellite is included in Chapter IV.

Operating Schedules: frequencies, telemetry modes, etc.

Every OSCAR satellite has a number of beacons and transponders operating at various frequencies and in different modes. In addition to keeping track of when a given satellite will be within range users must therefore be aware of the frequencies and operating modes scheduled. The latest schedule information available at presstime for each satellite is included in Chapter IV. Up-to-date scheduling information is available from: OSCAR Educational Programs, ARRL, 225 Main St., Newington, Conn. 06111. In addition, scheduling information is included in the yearly AMSAT Orbital Calendar (see Section 2.1) and monthly in QST magazine. Educators can usually arrange for special changes in operating schedules by contacting the ARRL educational programs office. Requests should be submitted as far in advance as possible so that the changes can be publicized so as not to inconvenience other users.

3.3 STRUCTURAL, ATTITUDE-CONTROL, PROPULSION, AND ENERGY-SUPPLY SUBSYSTEMS

Structural Subsystem

The structural subsystem -- the box that holds it all together -- serves a number of functions including support of antennas, solar cells, and internal electronics; protection of onboard subsystems from the environment during launch and while in space; and conduction of heat into and out of the satellite interior. Structural design (size, shape, materials) is influenced by launch-vehicle constraints and by function. AMSAT low-altitude satellites have generally been in the 20-30 kg range. Phase III high-altitude spacecraft, with their own kick motors and fuel, will probably weigh in at close to 70 kg at launch. Insofar as possible, AMSAT satellite structures are fabricated from sheet aluminum to minimize machining operations. The prominent features one observes when looking at a satellite (satellites currently in orbit and under construction are shown in Chapter IV) are (1) an attach fitting used to mount the satellite on the launch vehicle, (2) antennas for the various radio links, (3) solar cells, (4) heat radiative coatings designed to achieve a desired equilibrium temperature aboard the spacecraft and, for Phase III, (5) the nozzle of the apogee kick motor. For a discussion of how satellite shape effects equilibrium temperature and overall solar cell efficiency see Chapter VI, experiment SE 3 and project SP 7.

Attitude-Control Subsystem

Attitude control of a satellite (control of its orientation in space) can provide distinct advantages with respect to antenna gain, solar cell efficiency, thermal control and operation of scientific instruments.

Attitude-control subsystems vary greatly in complexity. A simple system might consist of a bar magnet which tends to align itself parallel to the earth's magnetic field while a complex system might employ cold gas jets, solid rockets, and inertia wheels, all operating under the control of a computer and in conjunction with a sophisticated system of sensors. Complex attitude-control systems can be used to provide three axis stabilization or to point a particular satellite axis toward the earth, toward any fixed direction in inertial space (with respect to the fixed stars), or along the earth's magnetic field. Single axis orientation is usually achieved by spinning the spacecraft about its major axis. Attitude-control systems are classified as active or passive. Passive systems do not require any power or sensor signals for their operation. As a result they are usually simpler and more reliable. However, they are also less accurate and flexible. Some of the attitude-control systems in general use are described below [6].

Mass Expulsors. Devices of this type are based on the rocket principle. They are classified as active and are relatively complex. Examples are cold gas jets, solid propellent rockets and ion thrust engines. Mass expulsors are often used to spin a satellite around its principal axis. The resulting angular momentum of the satellite is then parallel to the principal axis. The principal axis therefore tends to maintain a fixed direction in inertial space as a result of angular momentum conservation.

Angular Momentum Reservoirs. This category includes devices based on the inertia (fly) wheel principle. Assume that a satellite contains a fly wheel as part of a dc motor which can be powered up from the satellites power supply by ground command. If the angular momentum of the fly wheel is changed then the angular momentum of the rest of the satellite must change in an equal and opposite direction. Systems of this type are classified as active.

Moment-Of-Inertia Changers. The spin rate of a satellite can be changed by deploying booms. The booms change the satellite's moment of inertia and the spin rate therefore changes in order to conserve angular momentum. These systems are classified as active.

Environmental-Force Couplers. The satellite is coupled to (affected by) its environment in a number of ways. In the two body central-force model of Chapter I, we discussed how the satellite and earth were first treated as point masses at their respective centers of mass. We went on to discuss how the departure of the earth from spherical symmetry caused readily observable perturbations of the satellite's path. The departure of the satellite's mass distribution from spherical symmetry likewise causes readily observable effects. An analysis of the mass distribution in the satellite defines a specific axis which tends to line up with the geocenter as a result of the earth's gravity gradient. (See note under Energy Absorbers). Gravity gradient devices can be used to point a specific satellite axis towards the geocenter. However, anyone who's been on a sailboat knows that gravity can produce two stable states. Another environmental factor which can be tapped for attitude control purposes is the earth's magnetic field. A strong bar magnet carried by the satellite will tend to align itself parallel to the direction of this field. One characteristic of the earth's magnetic field, the dip angle, is shown in Figure 3.6

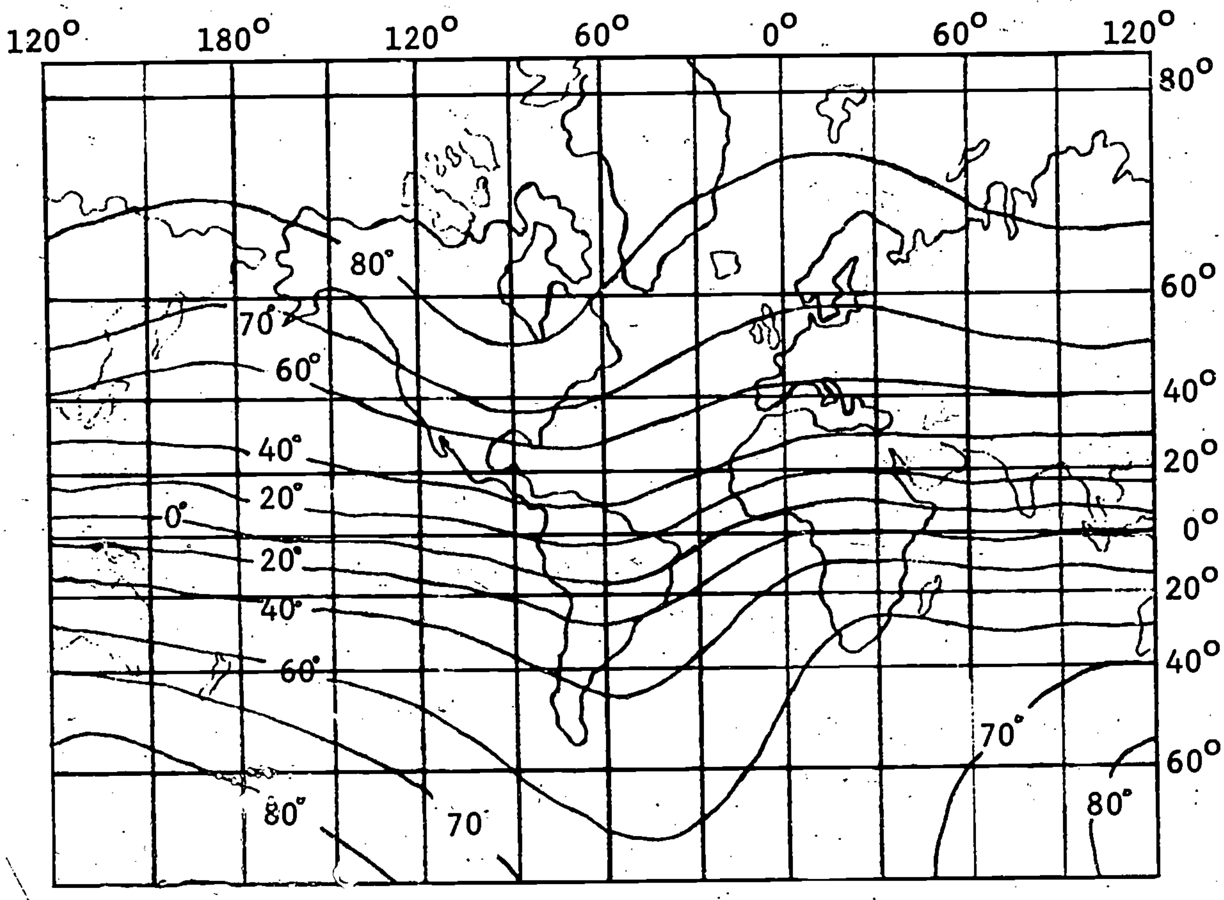


Figure 3. Inclination (dip) angle of earth's magnetic field (1960).

Magnetic and gravity based environmental couplers are passive systems which are simple and reliable. Note that even if a satellite designer does not employ environmental coupling for attitude control, these forces are always present and their affect on the satellites attitude must be taken into account.

Energy Absorbers. Energy absorbers or dampers are used to convert undesired motional energy to heat. They are used in conjunction with many of the previously mentioned devices. For example, if dissipative forces did not exist, gravity gradient forces would cause the satellites principal axis to swing pendulum-like about the local vertical instead of pointing towards the geocenter and a bar magnet carried on a satellite would oscillate about the local magnetic field direction instead of lining up parallel to it. Dampers are passive devices. They may consist of springs, viscous fluids, or hysterisis rods (eddy-current brakes).

Practical Attitude Control. Let's look at some of the trade-offs involved in choosing an attitude control system for low-altitude satellites such as AMSAT-OSCARs 6, 7 and 8. If stabilization systems were not employed on these satellites, isotropic satellite antennas would be desirable. The mechanical complexity of the antenna system would increase and power levels on all uplinks and downlinks would need to be raised to provide the desired signal levels at the satellite and at earth. Higher power aboard the satellite means more complex transmitters, larger power supplies, more solar cells and batteries or less operating time -- generally more weight and complexity. In addition, efficient illumination of the solar cells calls for matching the physical structure of the satellite to the attitude-control system. If no attitude control is used, then a spherical distribution of cells is most efficient. An attitude-control system is clearly desirable if its requirements in terms of system complexity, weight, etc. are small compared to the savings it provides. The system chosen for AMSAT-OSCARs 6, 7 and 8 was a passive one -- bar magnets were mounted in the satellite to align a specific axis along the earth's local magnetic field. Permalloy hysterisis damping rods, mounted perpendicular to the primary magnet, are used to reduce spin and small oscillations. Spin about the bar magnet axis is undesirable because it causes radio link fades as the relative orientations of the spacecraft and ground station antennas change. The principal axis of the satellite rotates 720° in inertial space during each orbit. The system has been very satisfactory.

High-altitude Phase III satellites will require a more sophisticated approach to attitude control. With Phase III AMSAT will, for the first time, utilize an active stabilization system. These spacecraft will be spun about their principal axis which will be in the orbital plane and aligned so that it points toward the geocenter at apogee. When sun angles are very poor, a slightly different orientation will have to be used. Initial spin up will probably be accomplished by small solid propellent rockets. Torquing coils aboard the spacecraft will then enable command stations to change the spin rate and spin axis orientation. Torquing coils operate on the same physical principle as bar magnets. However, they can be turned on and off (pulsed) at will. Antennas aboard Phase III satellites will be carefully designed to prevent spin modulation of uplink and downlink signals.

Propulsion Subsystem

The simplist type of space propulsion system consists of a small solid propellent rocket which, once ignited, burns until the fuel is exhausted. Rockets of this type are often used to boost a satellite from a near earth orbit into an elliptical orbit with an apogee close to synchronous altitude (35,000 km) or to shift a satellite from such an elliptical orbit into a circular orbit near synchronous altitude. Such rockets are known as "apogee kick motors" or "kick motors". AMSAT plans to employ kick motors for the first time aboard the Phase III satellite series. The motor, in conjunction with a sophisticated system of sun and earth sensors, will enable AMSAT to either (1) raise the apogee, (2) change the orbital inclination, or (3) raise the apogee and change the inclination of the orbit. Since the amount of energy available is limited its allocation will be carefully evaluated to select the optimal final orbit achievable from a given injection orbit. Kick motors are potentially very dangerous devices and their use and handling must conform to rigid safety procedures.

Energy-Supply Subsystem

Communications satellites can be classified as active or passive. An example of a passive satellite is a big balloon (Echo I, launched August 12, 1960, was 30 meters in diameter when fully inflated) coated with conductive material which reflects radio signals. When used as a passive reflector such satellites do not need any electronic components nor any power source. (Actually some electronic equipment and a power source may be associated with the inflation equipment or included for other purposes). While such a satellite is appealingly simple, the radio power it reflects back to earth is down by a factor of ten million (70 dB) compared to the signal transmitted by a transponder aboard an active satellite (assuming equal uplink signal strength and a comparison made on the basis of equal satellite mass in the 50 kg range) [7]. Ground station antenna and power requirements for use with passive satellite systems are therefore prohibitively large and expensive.

An active satellite (one with a transponder) needs power. We now survey the power systems used in active satellites. The energy source supplying power to the satellite should be reliable, efficient and low in cost and it should have a long life. By efficient we mean: the available electrical power to weight ratio and the available electrical power to waste heat ratio should be large. We examine three energy sources which have been extensively studied: chemical, nuclear and solar [8].

Chemical Power Sources. Chemical power sources include primary cells, secondary cells and fuel cells. Early satellites were flown with primary cells. When the cells ran down the satellite "died". Sputnik I, Explorer I, and the early OSCAR satellites were powered by primary cells. Satellites of this type usually had lifetimes of a few weeks, although Explorer I with its low power transmitters (about 70 mW total) ran almost 4 months on Hg batteries. These early experimental satellites demonstrated the feasibility of using satellites for communications, scientific exploration, etc. and thereby provided the impetus for the development of longer lived power systems. Today, batteries (secondary cells in this case) are mainly used to store energy aboard satellites; they are no longer employed as a primary power source. Nickel-Cadmium batteries are almost universally used. However, workers at COMSAT Laboratories have recently developed sealed Nickel-Hydrogen batteries which operate at energy densities of up to 75 Wh/kg (as compared to about 15 Wh/kg for Nickel-Cadmium batteries). If the life-time of these cells lives up to expectations they will probably be used extensively in the future. The longest lived OSCAR satellite (to date) was AMSAT-OSCAR 6 whose life (4.5 years) was terminated by battery failure. Another chemical power system, the fuel cell, has been used as a source of energy on manned space missions such as Apollo and Gemini which require large amounts of power over a relatively short time span. The development of fuel cells is currently attracting a great deal of attention.

Nuclear Power Sources. One nuclear power source to be flight tested is the radioisotopic-thermoelectric power plant. In devices of this type, heat from decaying radioisotopes is converted directly to electricity by thermoelectric couples. Some U. S. transit navigation satellites have employed generators of this type -- the Snap 3B and Snap 9A (25 watts). These generators have a high available power to weight ratio but they generate large amounts of waste heat and have a high cost per watt due to the fuel. They are most useful when solar cells are unsuitable -- orbits inside the Van Allen Belts or deep space missions where solar intensity is greatly reduced (such as Pioneer and Viking) -- or when very large amounts of power are required. Nuclear power sources constitute a very serious safety hazard in case of accidental reentry of the satellite as occurred in early 1978 with a Russian spacecraft. AMSAT is not considering nuclear power systems.

Solar Power Sources. The third power source we consider is solar. Solar cell technology has evolved to the point where solar cells power the great majority of satellites in orbit. However, solar cells have a number of undesirable features. They compete for mounting space on the outer surface of the satellite with antennas and heat radiating coatings. They are subject to degradation, especially when the satellite orbit passes through the Van Allen radiation belts (roughly, altitudes between 1,600 and 8,000 kms). They work most efficiently below 0°C while the electronics systems aboard the satellite are usually designed for a 20°C environment. They call for attitude control which often conflicts with other mission objectives. Their power output decreases with distance from the sun making them unsuitable for missions to the outer planets and beyond. Finally, they do not produce any output when eclipsed from the sun. But, power sources using solar cells to produce electrical energy and secondary cells to store energy are by far the simplest for long lifetime satellites, they are comparatively low in cost for the power produced, they generate little waste heat, and power systems using them have an acceptable available electric power to weight ratio. When satellite power needs are greater than about 50 watts, the use of solar cells usually requires the satellite structure to include paddles or panels (paddles which can be oriented toward the sun) since the body of the satellite may not have sufficient surface area to mount all the cells required. A one meter square solar panel oriented perpendicular to the sun-panel line will intercept about 1,380 watts of solar energy (panel-sun distance assumed equal to earth-sun distance). New solar cells are typically 10.5 to 11.0 percent efficient. The efficiency decreases with time; the exact dependence is a function of the cells environment. New cells cost about $20,000. per square meter at the present time. Cells donated to AMSAT often are only 8 percent efficient. A great deal of development work is currently underway in solar cell technology. The aims are to increase efficiency and reduce costs. Workers at COMSAT Laboratories have developed a 13% efficient "violet" cell (1972) and a 15% efficient "non-reflective" cell (1974). Solar cells mounted on a satellite are usually protected by glass cover slides to reduce the rate of degradation due to radiation damage. The glass cover slides reduce the efficiency of the cells -- the thicker the slide the greater the reduction.

Practical Energy Subsystems. The typical AMSAT satellite energy system consists of a source, a storage device, and conditioning equipment as shown in Figure 3.7. The source consists of silicon solar cells.

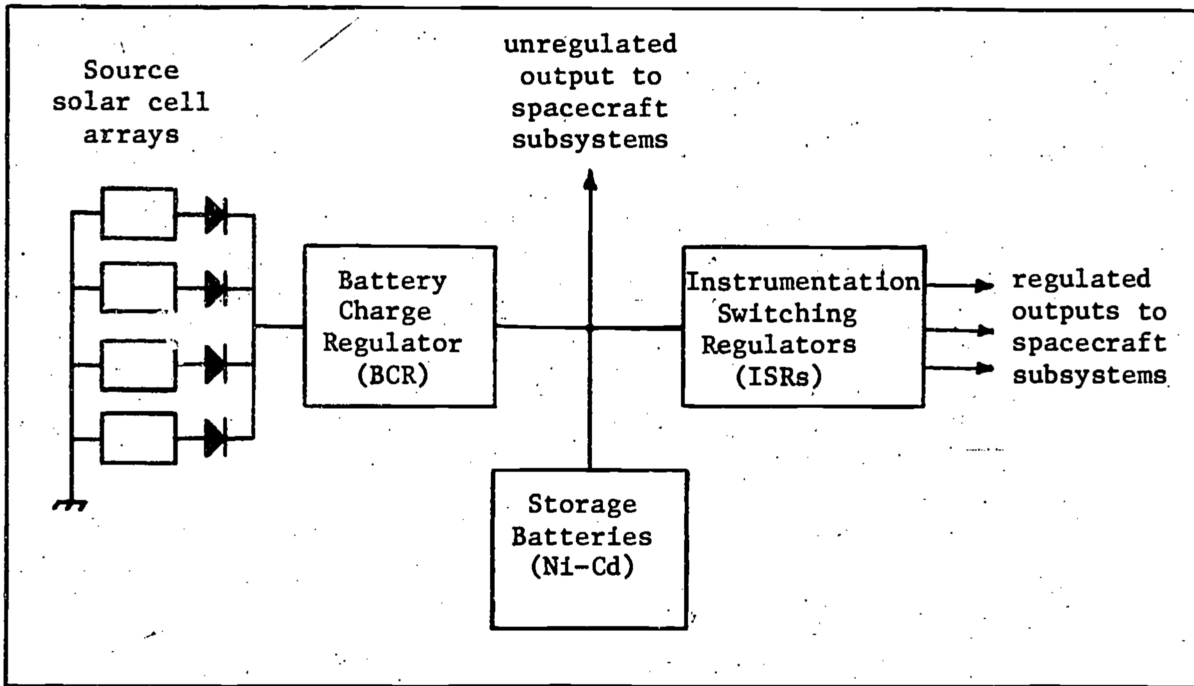


Figure 3.7. Long-lifetime communications satellite Energy Subsystem.

A storage unit is needed because of eclipses (satellite in earth's shadow) and the varying nature of the load. Nickel-Cadmium secondary cells are currently being used on AMSAT satellites. Power conditioning equipment typically includes a battery charge regulator (BCR) and an instrument switching regulator (ISR) providing dc to dc conversion with changes of voltage, regulation and protection. Because failures in the energy subsystem could be catastrophic (insofar as the mission is concerned) special attention is paid to insuring continuity of operation. Battery charge regulators and instrument switching regulators usually are constructed as redundant twin units with switchover controlled both automatically and via ground command. Solar cell strings are isolated by diodes so that a failure in one string will lower total capacity but will not otherwise effect spacecraft operation. These diodes also prevent current from flowing in the reverse direction through cell strings on the dark side of the satellite.

When the energy supply subsystem provides sufficient energy to operate the satellite on a continuous basis we say it has a positive power budget. If some satellite subsystems must be periodically turned off to enable the storage batteries to recharge we say the spacecraft has a negative power budget. Detailed calculations showing how to estimate the amount of power a solar cell array can provide are included in Chapter VI, experiment SE 3. Ideally, a satellite is designed with a beginning of life power sufficiently high so that, at the end of design life, a positive power budget still exists.

References

Chapter III

1. W. R. Corliss, Scientific Satellites (NASA SP-133), National Aeronautics and Space Administration, Washington, D. C., (1967), p.78.

2. P. I. Klein, J. Goode, P. Hammer and D. Bellair, "Spacecraft Telemetry Systems for the Developing Nations," presented at IEEE National Telemetering Conference, 14 April, 1971. (Reprints available from AMSAT).

3. K. Meinzer, "Lineare Nachrichtensatellitentransponder durch nichtlineare Signalzerlegung", (Linear Communications Satellite Transponder Using Non-linear Signal Splitting), Doctoral Dissertation, Marburg University, Germany, 1974.

4. P. L. Bargellini, "Radio Communications From Early Days to Satellites," Radio Science, Vol. 10, no. 7, July (1975), pp. 719-726.

5. B. P. Lathi, Communications Systems, John Wiley & Sons, N. Y. (1968).

6. W. R. Corliss, Scientific Satellites (NASA SP-133), National Aeronautics and Space Administration, Washington, D. C., (1967), section 9.7.

7. G. E. Mueller and E. R. Spangler, Communications Satellites, John Wiley & Sons, N. Y. (1964), p. 12.

8. W. R. Corliss, Scientific Satellites (NASA SP-133), National Aeronautics and Space Administration, Washington, D. C., (1967), section 9.5.

CHAPTER IV

SATELLITE DESCRIPTIONS

The primary objective of this chapter is to:

1. Describe the AMSAT and Russian educational satellites currently in orbit or planned for the near future.

This chapter has been organized so that each satellite is treated in a separate section using the format outlined in Table 4.1. The scheme used to number figures and tables treats each section as a separate entity. This was done to simplify future additions or deletions of spacec: t descriptions. Before turning to the satellites currently available or planned for the near future we briefly look at the history of the OSCAR satellite program.

Eight amateur radio satellites have been launched (as of March 20, 1978), two are currently active, and six (three AMSAT, three USSR) are probably in the development stage. The program started nearly twenty years ago with a few dedicated individuals working on a task that must have often seemed impossible. Today thousands of radio amateurs and educators all over the world regularly use AMSAT-OSCAR satellites. It's conceivable that in the near future hundreds of thousands of radio amateurs will be using satellites for a large percentage of their long distance communications and that satellite ground stations will be a common fixture in engineering, physics and electronics educational facilities.

The history of the amateur satellite program involves people as well as technical accomplishments. The real story can't be summed up in a few short paragraphs -- the interested reader should refer to the articles which have appeared in <u>QST</u> magazine over the years. A complete bibliography of <u>QST</u> satellite articles is available from the ARRL (see Appendix A). Selected technical aspects of the OSCAR satellites are summarized in Table 4.2.

Note: The spacecraft descriptions in this chapter assume a basic familiarity with satellite subsystems (see Chapter III).

SPACECRAFT NAME

GENERAL

1.1 Series
1.2 Launch: date, vehicle, agency, site
1.3 Orbital Parameters: general designation, period, apogee altitude, perigee altitude, inclination, eccentricity, longitude increment, maximum access distance (terrestrial distance between sub-satellite point and ground station; satellite at apogee, elevation angle of 0°)
1.4 Ground Track Data
1.5 Operating Schedule
1.6 Construction Credits: project management, spacecraft subsystems
1.7 Primary references

SPACECRAFT DESCRIPTION

2.1 Physical Structure: shape, mass
2.2 System Integration: block diagram

SUBSYSTEM DESCRIPTION

3.1 Beacons: frequencies, power levels, telemetry format, notes
3.2 Telemetry: formats available; description of each format including: decoding information, sample data, etc.
3.3 Command System
3.4 Transponders: for each transponder specify -- type, uplink passband, downlink passband, translation equation, output power, uplink EIRP, delay time, etc.
3.5 Attitude Stabilization: primary control, secondary control, damping
3.6 Antennas: description, polarization table
3.7 Energy-Supply and Power Conditioning: solar cell characteristics and configuration, storage battery, switching requlators, etc.
3.8 Propulsion System
3.9 Integrated Housekeeping Unit (IHU)

Table 4.1. Format used for technical descriptions of satellites. Background information on satellite subsystems is in Chapter III.

Satellite; Launch Date	Operating Life	Number of Transponders	Composite Transponder Bandwidth	Peak Transmitter Power	Highest Frequency Employed	Number of Beacons	Apogee (km)
OSCAR 1 Dec. 12, 1961	21 days	0	0	0.1 W	144 MHz	1	471
OSCAR 2 June 2, 1962	19 days	0	0	0.1 W	144 MHz	1	391
OSCAR 3 March 9, 1965	transponder, 18 days; beacon, several months	1	50 kHz	1. W	145 MHz	2	941
OSCAR 4 Dec. 21, 1965	85 days	1	10 kHz	3. W	432 MHz	1	33,600
OSCAR 5 Jan. 23, 1970	52 days	0	0	0.2 W	144 MHz	2	1,480
OSCAR 6 Oct. 15, 1972	4.5 years	1	100 kHz	1.5 W	435 MHz	2	1,460
OSCAR 7 Nov. 15, 1974	operational as of 3/78	2	150 kHz	8. W	2304 MHz	4	1,460
OSCAR 8 March 5, 1978	operational as of 3/78	2	200 kHz	1.5 W	435 MHz	2	912
RS-1		2		1.5 W	146 MHz		860

Notes: OSCAR 1 -- First satellite built by Radio Amateurs.
OSCAR 3 -- First transponder on amateur satellite.
OSCAR 4 -- Launch vehicle malfunctioned, satellite did not attain desired orbit. First solar powered amateur spacecraft.
OSCAR 5 -- First amateur satellite which could be controlled from the ground.
RS-1 -- First Russian amateur satellite

Table 4.2. A brief summary of Radio Amateur satellites as of publication date.

SPACECRAFT NAME: AMSAT-OSCAR 7

GENERAL

1.1 Series: AMSAT Phase II (low-altitude, long-lifetime)

1.2 Launch
- Date: November 15, 1974
- Vehicle: Two-stage Delta 2310
- Agency: U.S. National Aeronautics and Space Administration
- Site: NASA Western Test Range; Lompoc, California (Vandenberg Air Force Base)

1.3 Orbital Parameters
- General designation: low-altitude, near-circular, sun-synchronous
- Period: 114.945 minutes/orbit (1978)
- Altitude: 1,460 km; eccentricity = .001
- Longitude increment: 28.737°west/orbit (1978)
- Inclination: 101.7°
- Maximum access distance: 3,950. km

1.4 Ground Track Data: Data for drawing ground tracks is contained in Chapter II, Table 2.4.

1.5 Operating Schedule

Users should check QST or with ARRL (see Appendix A) to obtain latest schedule. In general, Modes B and C (2 m downlink) will be in use more than 50 percent of the time. On Wednesdays (UTC) the transponders are reserved for special experiments coordinated in advance with AMSAT. On Mondays (UTC) transponder users are requested to limit themselves to 10 watts EIRP.

1.6 Construction Credits
- Project management: AMSAT U.S.
- Spacecraft subsystems: Designed and built by groups in Australia, Canada, United States, West Germany.

1.7 Primary Reference

J. Kasser and J. A. King, "OSCAR 7 and Its Capabilities," QST, Vol. LVIII, no. 2, Feb. 1974, pp. 56-60.

SPACECRAFT DESCRIPTION

2.1 Physical Structure
- Shape: Right octahedral (8-sided) solid as shown in Figure 1.
- Mass: 28.9 kg

2.2 System Integration
- Block diagram: See Figure 2.

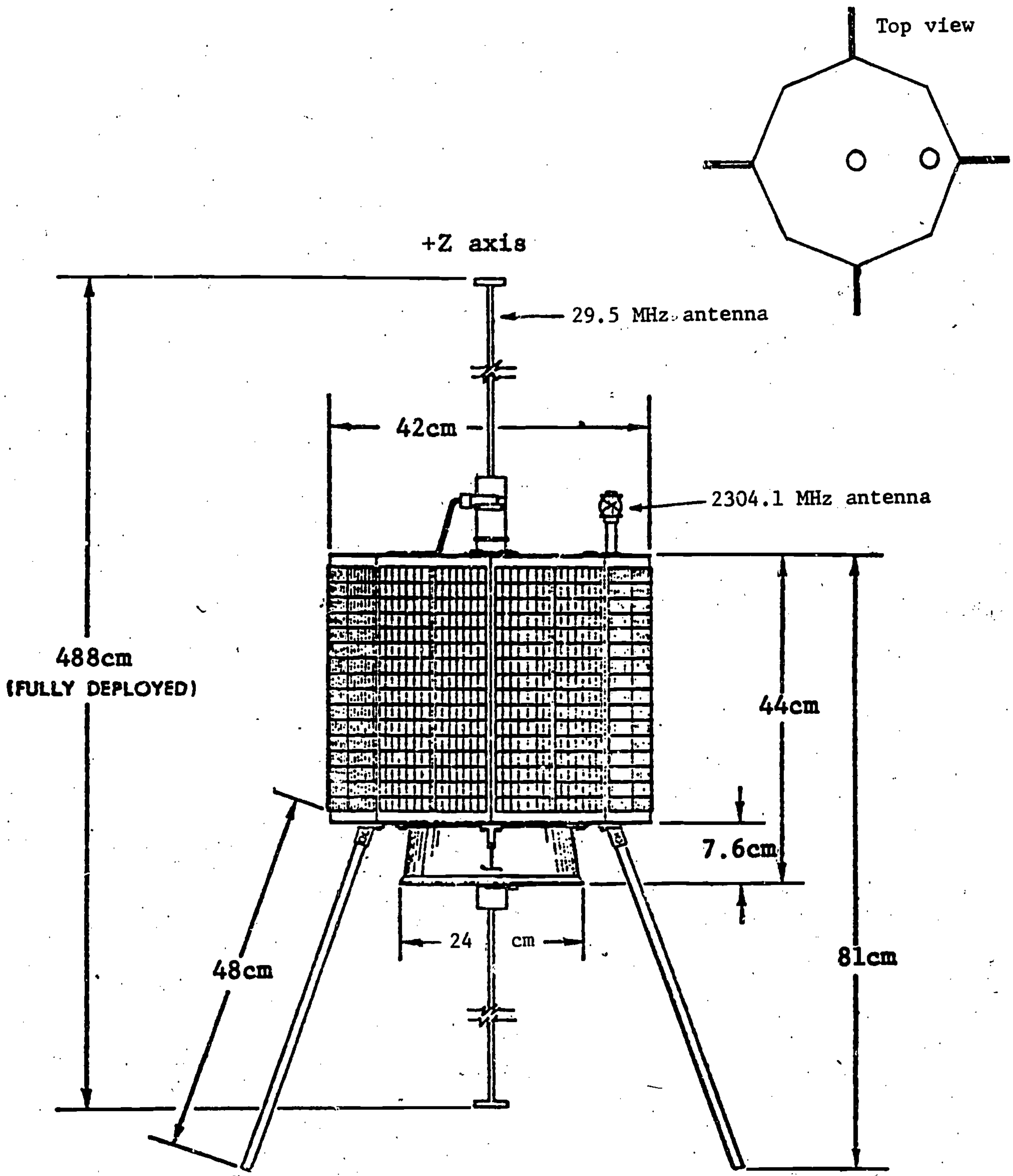


Figure 1. AMSAT-OSCAR 7.

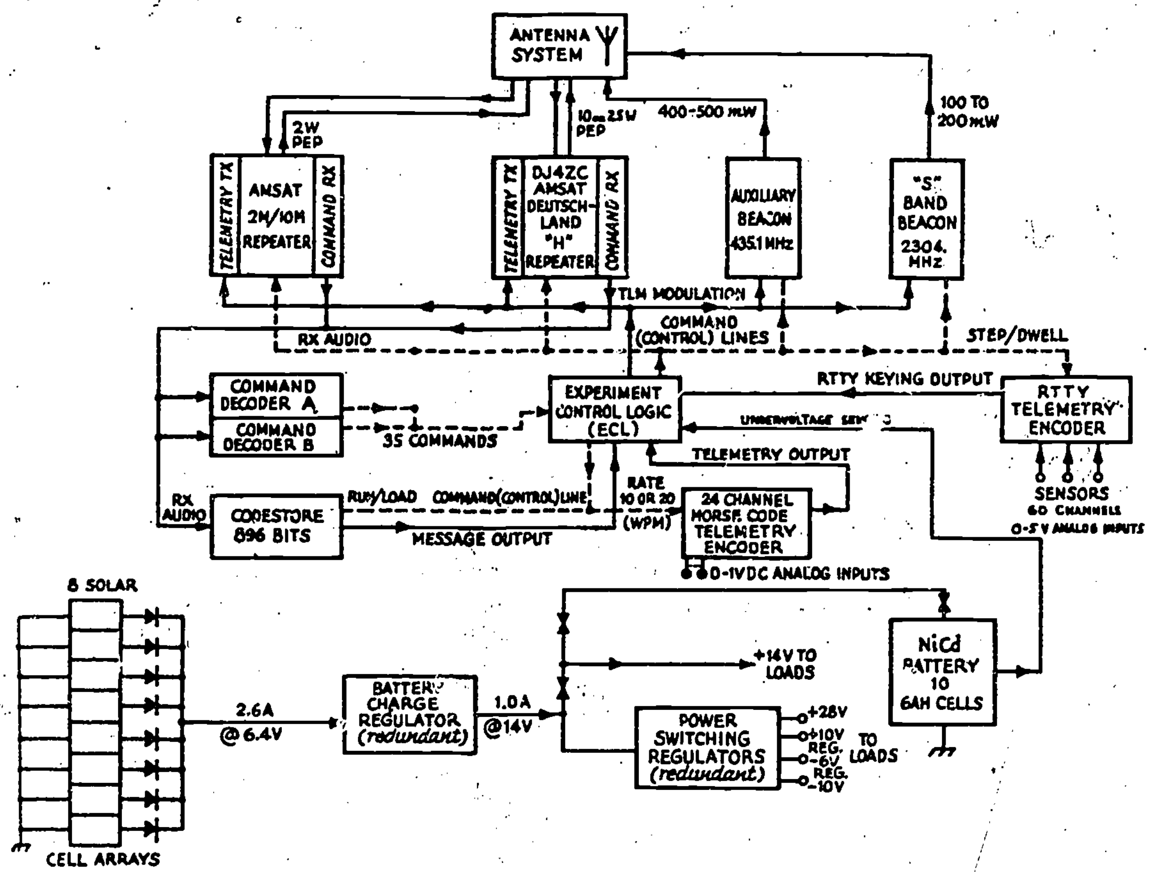


Figure 2. AMSAT-OSCAR 7 block diagram.

SUBSYSTEM DESCRIPTION

3.1 Beacons

29.502 MHz. (200 milliwatts) Used in conjunction with Mode A. Usually Morse code telemetry but may be Codestore. Excellent for educational applications.

145.972 MHz. (200 milliwatts) Used in conjunction with Modes B and C. Usually radioteletype telemetry but may be Morse code telemetry or Codestore. Excellent for educational applications.

435.100 MHz. Due to an intermittent problem this beacon is only turned on for special tests. Power output switches between 400 milliwatts and 10 milliwatts. Check Morse code telemetry channel 6B on 29.502 MHz beacon when spacecraft is in Mode A to determine if 435 MHz beacon is operating and power level. A well equipped ground station can hear the beacon at the 10 milliwatt level. This beacon can not be used in conjunction with Modes B or C.

2304.1 MHz. (40 milliwatts) AMSAT does not have permission to activate this beacon. It may be possible to obtain a special temporary authorization for its use from the Federal Communications Commission for educational programs.

3.2 Telemetry

Formats available: Morse code, radioteletype, Codestore

Morse code telemetry

frame: A frame contains 24 channels, 6 lines by 4 columns. Parameters are sent in a fixed serial format.

channel: A channel consists of a three digit number. The first digit is a line identifier, the last two digits encode the data. See Table 1 for decoding information.

speed: The telemetry is usually sent at 20 words per minute and a complete frame requires about 75 seconds. The speed can be reduced to 10 wpm by ground command for special demonstrations.

sample data: Table 2 contains sample data for an entire pass.

Radioteletype telemetry

frame: This system is normally operated in a serial mode where the 80 channels are sent sequencially. The first 60 channels (00-59) contain analog information and the remaining 20 channels contain repeated command/status information. This system can be operated in a dwell mode in which a specific channel is sampled continuously.

analog channels: Each analog channel contains five digits. The first two digits identify the channel (00-59) while the last three digits encode the sensor data. See Table 3.

command/status channels: Channels 60-79 (inclusive) describe spacecraft status -- even numbered channels (60-78) contain information on satellite clock time and odd number channels (61-79) contain information on operating modes which can be commanded from the ground. See Tables 4a and 4b. A clock time or state/mode word consists of five octal integers. Clock time words are decoded as follows. Ignore the first (left-hand) digit. Transform the remaining 4 digit octal

AMSAT - OSCAR - 7
CW TELEMETRY DECODING

CHANNEL	MEASURED PARAMETER	MEASUREMENT RANGE	CALIBRATION EQUATION
1A	Total Solar Array Cur.	0 to 3000 ma	$I_T = 29.5\ N$ (ma)
1B	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1970 - 20\ N$ (ma)
1C	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1970 - 20\ N$ (ma)
1D	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1970 - 20\ N$ (ma)
2A	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1970 - 20\ N$ (ma)
2B	RF Pwr. Out 70/2	0 to 8 watts	$P_{70/2} = 8(1 - 0.01\ N)^2$ (watts)
2C	24-Hr. Clock Time	0 to 1440 min.	$t = 15.16\ N$ (min) or $0.253\ N$ (hrs)
2D	Batt Chg/Dischg Cur.	-2000 to +2000 ma	$I_B = 40\ (N - 50)$ (ma)
3A	Battery Voltage	6.4 to 16.4 V.	$V_B = 0.1\ N + 6.4$ (volts)
3B	Half-Batt Voltage	0 to 10 V.	$V_{\frac{1}{2}B} = 0.10\ N$ (volts)
3C	Bat. Chg. Reg. #1	0 to 15 V.	$V_{cr1} = 0.15\ N$ (volts)
3D	Battery Temperature	-30 to +50°C	$T_{Bat} = 95.8 - 1.48\ N$ (°C)
4A	Baseplate Temp.	-30 to +50°C	$T_{bp} = 95.8 - 1.48\ N$ (°C)
4B	PA Temp. - 2/10 Rptr	-30 to +50°C	$T_{10} = 95.8 - 1.48\ N$ (°C)
4C	+X Facet Temp.	-30 to +50°C	$T_{+x} = 95.8 - 1.48\ N$ (°C)
4D	+Z Facet Temp.	-30 to +50°C	$T_{+z} = 95.8 - 1.48\ N$ (°C)
5A	PA Temp. - 70/2 Rptr	-30 to +50°C	$T_2 = 95.8 - 1.48\ N$ (°C)
5B	PA Emit. Cur. 2/10	0 to 1167 ma	$I_{10} = 11.67\ N$ (ma)
5C	Modul. Temp. 70/2	-30 to +50°C	$T_m = 95.8 - 1.48\ N$ (°C)
5D	Instr. Sw. Reg. Input Cur.	0 to 93 ma	$I_{isr} = 11 + 0.82\ N$ (ma)
6A	RF Pwr Out - 2/10	0 to 10,000 mw	$P_{2/10} = \frac{N^2}{1.56}$ (mw)
6B	RF Pwr Out - 435	0 to 1,000 mw	$P_{435} = 0.1(N^2) + 35$ (mw)
6C	RF Pwr Out - 2304	0 to 100 mw	$P_{2304} = 0.041\ N^2$ (mw)
6D	Midrange Telemetry Calibration	0.500 V.	$V_{cal} = 0.01\ N$ (0.50 ± 0.01)

Table 1. AMSAT-OSCAR 7 Morse code telemetry

[illegible]	B	1C	1D	2A	2B	2C	2D	3A	3B	3C	3D
	36	57	80	56	00	57	52	80	71	34	52
		64	75	41	00	57	51	76	69	33	53
0		65	71	36	00	61	51	79	70	31	51
00	[illegible]	76	68	49	00	59	51	75	66	31	52
00	38	73	82	72	00	59	49	72	67	30	52
00	2	75	71	82	01	59	50	72	66	31	54
00	[illegible]	70	75	92	00	59	52	73	67	30	52
00	52	87	54	90	00	59	50	74	67	30	52
00	73	79	54	94	00	59	49	71	66	31	52
00	84	69	37	92	00	59	52	70	68	33	52
00	87	65	42	93	00	59	55	78	68	34	52
00	89	63	65	90	00	59	54	74	70	34	52

4A	4B	4C	4D	5A	5B	5C	5D	6A	6B	6C	6D
53	33	57	56	54	11	55	49	38	07	01	49
52	32	56	56	53	13	54	51	40	05	00	50
52	32	57	56	51	13	54	50	37	05	00	51
52	32	56	57	52	14	54	50	43	05	01	50
53	32	55	50	53	12	54	50	43	10	00	49
54	32	54	54	56	11	54	49	38	12	00	49
51	34	55	54	55	12	54	51	38	12	00	50
52	33	52	53	50	14	54	50	38	11	01	51
49	34	53	54	54	11	54	50	38	12	01	49
49	32	54	54	52	12	55	50	40	12	00	51
52	32	55	54	53	12	54	52	40	11	00	51
55	32	54	55	51	12	54	50	40	12	00	50

Table 2. Sample AMSAT-OSCAR 7 Morse code telemetry data (orbit 2041; April 27, 1975). Record contains 12 frames. The 00 readings in channel 1A are due to an RF interference problem. Data provided by John Fox (WØLER).

number into decimal form. The resulting number tells how many time intervals have elapsed since the clock was last reset. The clock is wired to increment every 96 minutes and reset to zero every 273 days.

State/mode words are decoded as follows. Ignore the first digit. The middle three digits identify the command as per Table 4. The fourth digit indicates the operating mode. The fifth digit indicates if the satellite is receiving a command signal. Commands are described in Table 4b. At times, the second and third digits of the state/mode words are triggered by noise. If a state/mode word does not correspond to a command (Table 4b) or is obviously erroneous it is likely that this has occurred. The problem is most serious when a command is not being received (7 in 5th digit).

speed: Radioteletype telemetry is transmitted at 45.5 Bauds

comments: Radioteletype telemetry is usually transmitted using a mark only format in order to conserve spacecraft power.

sample data: Table 5 contains a complete RTTY telemetry frame. Three examples of decoding the sample data follow.

(1) The number 32198 is contained in RTTY channel 32. This is an analog channel measuring (see Table 3) the power output of the 70 cm / 2 m transponder which is 5.15 watts.

(2) The number 03544 is contained in RTTY channel 64. Channel 64 is measuring clock time information. 3544 (octal) equals 1892 (decimal) which corresponds to 126 days, 3 hours, 12 minutes (plus up to 96 additional minutes) since last reset.

(3) The number 06167 is contained in RTTY channel 73. This is state/mode information. The 616 corresponds to command 16 and the trailing six refers to Mode B. So, the satellite was programmed for Mode B RTTY and a command signal was not being received when channel 73 was transmitted.

Codestore: The Codestore module uses an 896 bit COS-MOS shift register memory which can be loaded by ground stations with a message in any digital code. The Morse code is usually employed. The playback speed, using Morse code is about 13 words per minute.

3.3 Command System

The satellite contains a command system decoder which recognizes 35 commands of which six are redundant. See Table 4b.

3.4 Transponders

Transponder I: Mode A (2m/10m)

type: linear, non-inverting

uplink passband: 145.850 - 145.950 MHz

downlink passband: 29.400 - 29.500 MHz

translation equation: downlink freq. (MHz) = uplink freq. (MHz) - 116.450 MHz ± Doppler

output power: about 1.3 watts PEP (See Morse code telemetry channel 6A)

Table 3. AMSAT-OSCAR 7 RTTY telemetry (channels 00-59)

AMSAT - OSCAR - 7

TELETYPE TELEMETRY DECODING

XXYYY (typical channel)

channel number N

CHANNEL	MEASURED PARAMETER	MEASUREMENT RANGE	CALIBRATION EQUATION (25° C)
00	PA Temp. - 70/2 Rptr	-30 to +50°C	$T_2 = 95.79 + \frac{1}{0.001271 - 6.756/N}$
01	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1995 - 2.191\ N$ (ma)
02	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1968 - 2.175\ N$ (ma)
03	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1953 - 2.150\ N$ (ma)
04	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1954 - 2.150\ N$ (ma)
05	+Z Axis Orientation	0 to 90°	θ_z = arccos (N/1000) (degrees from Z axis)
06	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1995 - 2.191\ N$ (ma)
07	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1968 - 2.175\ N$ (ma)
08	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1953 - 2.150\ N$ (ma)
09	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1954 - 2.150\ N$ (ma)
10	-Z Axis Orientation	0 to 90°	θ_{-z} = arccos (N/1000) (degrees from Z axis)
11	Battery Voltage	6.4 to 16.4 V.	$V_B = 0.01\ N + 6.40$ (volts)
12	Half-battery voltage	0 to 10 V.	$V_{hB} = 0.01045\ N$ (volts)
13	28V. Regulator	0 to 34 V.	$V_{28} = 0.034\ N$ (volts)
14	10V. Regulator	0 to 15 V.	$V_{10} = 0.01558\ N$ (volts)
15	9V. Regulator	0 to 10 V.	$V_9 = 0.01\ N$ (volts)
16	Bat. Charge Reg. #1	0 to 15 V.	$V_{cr1} = 0.01547\ N$ (volts)
17	Bat. Charge Reg. #2	0 to 15 V.	$V_{cr2} = 0.01535\ N$ (volts)
18	Zero V. Telemetry Calibration	0 V.	$V_o = \frac{N}{100}$ (volts); N=000 ± 1 count
19	2304 Beacon Pwr Out	0 to 100 mw	$P_{2304} = 3.0(N/100)^2$ (mw)
20	Bat. Chg-Dischg Cur.	-1500 to +1500 ma	$I_B = 3.17\ N - 1474$ (ma)
21	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1995 - 2.191\ N$ (ma)
22	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1968 - 2.175\ N$ (ma)
23	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1953 - 2.150\ N$ (ma)
24	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1954 - 2.150\ N$ (ma)
25	+Z Axis Orientation	0 to 90°	θ_{+z} = arccos (N/1000) (degrees from Z axis)
26	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1995 - 2.191\ N$ (ma)
27	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1968 - 2.175\ N$ (ma)
28	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1953 - 2.150\ N$ (ma)
29	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1954 - 2.150\ N$ (ma)
30	-Z Axis Orientation	0 to 90°	θ_{-z} = arccos (N/1000) (degrees from Z axis)
31	2/10 Rptr Pwr Out	0 to 10,000 mw	$P_{2/10} = 0.641(N/10)^2$ (mw)
32	70/2 Rptr Pwr Out	0 to 8 watts	$P_{70/2} = 8\ (1 - 0.001\ N)^2$ (watts)
33	435 Beacon Pwr Out	0 to 1000 mw	$P_{435} = 0.001\ N^2 + 35$ (mw)
34	Total Solar Panel Cur.	0 to 3000 ma	$I_T = 3.115\ (N - 7)$ (ma)
35	Battery Temperature	-30 to +50°C	T_{Bat} = (same as channel 00)
36	Baseplate Temperature	-30 to +50°C	T_{bp} = (same as channel 00)
37	+X Facet Temp.	-30 to +50°C	T_{+x} = (same as channel 00)
38	+Z Facet Temp.	-30 to +50°C	T_{+z} = (same as channel 00)
39	2304 Beacon Temp.	-30 to +50 °C	T_{2304} = (same as channel 00)
40	Midrange Telemetry Calibration	0.500 ±0.005 V.	$V = 0.001\ N$ (volts) ($N = 500 \pm 5$)
41	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1995 - 2.191\ N$ (ma)
42	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1968 - 2.175\ N$ (ma)
43	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1953 - 2.150\ N$ (ma)
44	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1954 - 2.150\ N$ (ma)
45	+Z Axis Orientation	0 to 90°	θ_{+z} = arccos (N/1000) (degrees from Z axis)
46	+X Solar Panel Cur.	0 to 2000 ma	$I_{+x} = 1995 - 2.191\ N$ (ma)
47	+Y Solar Panel Cur.	0 to 2000 ma	$I_{+y} = 1968 - 2.175\ N$ (ma)
48	-X Solar Panel Cur.	0 to 2000 ma	$I_{-x} = 1953 - 2.150\ N$ (ma)
49	-Y Solar Panel Cur.	0 to 2000 ma	$I_{-y} = 1954 - 2.150\ N$ (ma)
50	-Z Axis Orientation	0 to 90°	θ_{-z} = arccos (N/1000) (degrees from Z axis)
51	Battery Voltage	6.4 to 16.4 V.	$V_B = 0.01\ N + 6.45$ (volts)
52	Half-battery Voltage	0 to 10 V.	$V_{hB} = 0.01052\ N$ (volts)
53	70/2 Rptr AGC Level	0 to 29 db	$AGC_{70/2} = 20 \log_{10} (948.9 - N) - 57.92$ (db)
54	70/2 Rptr TX Osc. Test	75.0 to 95.0%	TX Osc = 0.1 N (88.0% typical)
55	70/2 Rptr RX Osc. Test	10.0 to 40.0%	RX Osc = 0.1 N (24.0 % typical)
56	70/2 Rptr Mod. Out	0 to 1 V.	MOD = 0.00128 N (volts)
57	70/2 Rptr Envelope Level	0 to 1 V.	ENV = 0.00128 N (volts)
58	2/10 Rptr AGC Level	0 to 27 db	$AGC_{2/10} = 10 \log_{10} (1605 - 2.48\ N)$ (db)
59	70/2 Rptr Conv Osc Test	0 to 10 V.	CONV = 0.01 N (volts)

A07

STATUS WORD 1	2	3	4	5	COMMAND # (see Table 4b)
	0	3	odd		33
	0	5	odd		34
	0	6	odd		35
	0	7	even		32
	1	1	odd		12
	1	2	odd		14
	1	3	even		10
	1	4	odd		15
	1	5	even		11
	1	6	even		13
	2	1	odd		22
	2	2	odd		24
	2	3	even		20
	2	4	odd		25
	2	5	even		21
	2	6	even		23
	3	0	odd		05
	3	1	even		02
	3	2	even		03
	3	4	even		04
	4	1	odd		28
	4	2	odd		30
	4	3	even		26
	4	4	odd		31
	4	5	even		27
	4	6	even		29
	5	0	odd		09
	5	1	even		06
	5	2	even		07
	5	4	even		08
	6	0	odd		19
	6	1	even		16
	6	2	even		17
	6	4	even		18
	7	0	even		01

Fourth Status bit	Mode
4 or 5	Mode A
6 or 7	Mode B
2 or 3	Mode C
0 or 1	Mode D

Fifth Status bit	
7	Command receiver does not detect command signal
0	Command receiver is detecting command signal

Table 4a. State /Mode words in teletype telemetry channels 61, 63, 65, 67, 69, 71, 73, 75, 77 and 79. (AMSAT-OSCAR 7).

CMND #

1 - MODE A SELECT - 2/10 RPTR ON AND/OR 435 BEACON ON, 70/2 RPTR OFF.
2 - MODE B SELECT - 70/2 RPTR ON AT FULL-POWER, 2/10 RPTR OFF, 435 BEACON OFF.
3 - 435 BEACON ON (DURING MODE A OR MODE D ONLY).
4 - 435 BEACON OFF.
5 - CODESTORE - RUN MODE.
6 - CODESTORE - LOAD MODE.
7 - MORSE CODE TELEMETRY - 20 WPM.
8 - MORSE CODE TELEMETRY - 10 WPM.
9 - 2/10 RPTR - FULL SENSITIVITY.
10 - 2/10 RPTR - REDUCED SENSITIVITY (-14 DB).
11 - 435 BEACON - CODESTORE KEYING.
12 - 435 BEACON - MORSE CODE TELEMETRY KEYING.
13 - 435 BEACON - TELETYPE TELEMETRY KEYING.
14 - 29.50 OR 145.98 BEACON - MORSE CODE TELEMETRY KEYING.
15 - 29.50 OR 145.98 BEACON - CODESTORE KEYING.
16 - 29.50 OR 145.98 BEACON - TELETYPE TELEMETRY KEYING.
17 - RESET 24-HOUR SATELLITE CLOCK.
18 - MODE C SELECT - 70/2 RPTR ON AT QUARTER-POWER, 2/10 RPTR OFF, 435 BEACON OFF.
19 - MODE D SELECT - RECHARGE MODE; 70/2 RPTR OFF, 2/10 RPTR OFF. NOTE - 435 BEACON CAN BE OPERATED IN THIS MODE.
20 - CHARGE REGULATOR NO. 1 SELECT.
21 - CHARGE REGULATOR NO. 2 SELECT.
22 - TELETYPE TELEMETRY - DWELL MODE.
23 - TELETYPE TELEMETRY - RUN MODE.
24 - 2304 BEACON ON (FOR 14 MINUTES ONLY). NOTE - THIS BEACON IS OPERABLE IN ALL MODES.
25 - 2304 BEACON OFF.
26 - 2304 BEACON - INTERNAL KEYING.
27 - 2304 BEACON - MORSE CODE TELEMETRY KEYING.
28 - TELETYPE TELEMETRY - FSK MODE.
29 - TELETYPE TELEMETRY - AFSK MODE.
30 - MODE A SELECT (REDUNDANT TO #1).
31 - MODE B SELECT (REDUNDANT TO #2).
32 - 435 BEACON ON (REDUNDANT TO #3).
33 - 435 BEACON OFF (REDUNDANT TO #4).
34 - MODE C SELECT (REDUNDANT TO #18).
35 - MODE D SELECT (REDUNDANT TO #19).

Table 4b. AMSAT-OSCAR 7 command states.

00461	01864	02357	03743	04884	05252	06880	07366	08734	09898
10000	11703	12642	13018	14622	15000	16274	17423	18016	19000
20489	21904	22398	23680	24891	25043	26893	27408	28665	29896
30000	31000	32198	33000	34000	35447	36481	37472	38483	39514
40498	41910	42475	43566	44890	45020	46915	47503	48543	49902
50000	51702	52639	53610	54878	55254	56423	57406	58000	59855
03544	06167	03544	06167	03544	06167	03544	06167	03544	06167
03544	06167	03544	06167	03544	06167	03544	06167	03544	06167

Table 5. Sample AMSAT-OSCAR 7 RTTY telemetry frame. (Courtesy of John Fox)

uplink EIRP: a maximum of 100 watts is recommended

comments: The same basic transponder design has been used on ASAT-OSCARs 6, 7, and 8. A block diagram is contained with the AMSAT-OSCAR 8 description. Delay time about 10^{-6} seconds.

Transponder II: Mode B, Mode C (70cm/2m)

type: linear, inverting

uplink passband: 432.125 - 432.175 MHz

downlink passband: 145.975 - 145.925 MHz

translation equation: downlink freq. (MHz) = 578.100 MHz - uplink freq. (MHz) ± Doppler

output power: about 8 watts PEP Mode B (agc based on peak power); about 2.5 watts PEP Mode C (agc based on average power). Monitored by Morse code telemetry channel 2B.

uplink EIRP: a maximum of 80 watts is recommended

delay time: less than 10 microseconds (estimate)

comments: The transponder, designed by Dr. Karl Meinzer, employes the envelope elimination and restoration technique described in reference 3 of Chapter 3.

3.5 Attitude Stabilization

Primary control: Four Alnico 5 bar magnets, each approximately 15 cm long with a square cross-section of about .6 cm are symmetrically placed with respect to the Z-axis of the spacecraft and parallel to it. The resultant far field is similar to that produced by a single 30,000 pole-cm magnet. As the satellite moves along its orbit the Z-axis of the spacecraft constantly changes its direction in inertial space in order to remain aligned parallel to the local direction of the earths magnetic field. The +Z-axis (top) of the satellite points in the direction of the earths north magnetic pole.

Secondary control: Reflective and non-reflective optical coatings on the elements of the canted turnstile antenna cause the satellite to spin about the Z-axis due to solar radiation pressure (radiometer effect).

Damping: The spacecraft contains eight Allegheny Ludlum type 4750 permalloy hysterisis damping rods. The rods are about 30 cm long, .32 cm in diameter, and bent in the center at a 135° angle. Four are mounted just under the faces of the spacecraft in a plane paralled to the top (perpendicular to the Z-axis) and about 7 cm from it. The remaining four are similarly mounted in a plane parallel to, and about 7 cm from, the bottom plate.

3.6 Antennas (See Figure 1)

29.5 MHz: The 29.5 MHz antenna is a half wavelength dipole (about 4.9 m) mounted along the Z-axis.

146 MHz and 432 MHz : A single antenna, known as a canted turnstile, is used to receive on 146 MHz for Mode A, and to simultaneously receive on 432 MHz and transmit on 146 MHz for Mode B. The antenna consists of two "inverted V" shaped dipoles mounted at right angles to each other on the base (-Z face) of the spacecraft. Each dipole consists of two 48 cm spokes (about 1/2 wavelength at 146 MHz, 3/2 wavelength at 432 MHz). The canted turnstile produces an elliptically polarized (circularly polarized along the -Z-axis) radiation field over a very large solid angle. There is some gain along the -Z axis and some shadowing along the +Z-axis.

2304 MHz: A quadrifilar helix mounted on the top plate is used in conjunction with the 2304.1 MHz beacon. It has the unusual property of producing a nearly circularly polarized radiation field everywhere in the hemisphere defined by the top plate.

notes: Signal polarizations of the spacecraft antennas are summarized in Table 6.

3.7 Energy-Supply and Power Conditioning

The main components of the AMSAT-OSCAR 7 energy-supply and power conditioning subsystem were shown in Figure 2. The basic solar cell module consists of 112 individual cells (7 parallel cells form a unit, 16 units in series). The modules are interconnected as shown in Figure 3. Each of the eight spacecraft facets (faces) contains two modules. Four modules on two adjacent facets are connected together to form a quadrant. Provisions are made for monitoring the current output of each of the four quadrants (+X, -X, +Y, -Y) and the total output via telemetry. For additional information on the solar cell arrangement see Chapter 6, Project SP 7.

AMSAT-OSCAR 7 ANTENNA POLARIZATIONS

System	Spacecraft Polarization
2m/10m transponder uplink (146 MHz)	left-hand circular (*1)
2m/10m transponder downlink and 29 MHz beacon	linear
70cm/2m transponder uplink (432 MHz)	right-hand circular (*1)
70cm/2m transponder downlink and 146 MHz beacon	right-hand circular (*1)
435.1 MHz beacon	left-hand circular (*1)
2304.1 MHz beacon	right-hand circular (*2)

Note: (*1) Polarization sense referenced to +Z-axis of spacecraft. Ground stations off the +Z-axis will observe elliptical polarization. Stations north of the magnetic equator (see Figure 3.6) will generally find that the circular component is as indicated in the table. Stations in the southern hemisphere will generally find the circular component opposite to that shown.

(*2) The polarization remains nearly circular in the entire hemisphere defined by the +Z face of the satellite.

Table 6. AMSAT-OSCAR 7 antenna polarizations.

Configuration of quadrant. Legend: basic module shown 112 cells.

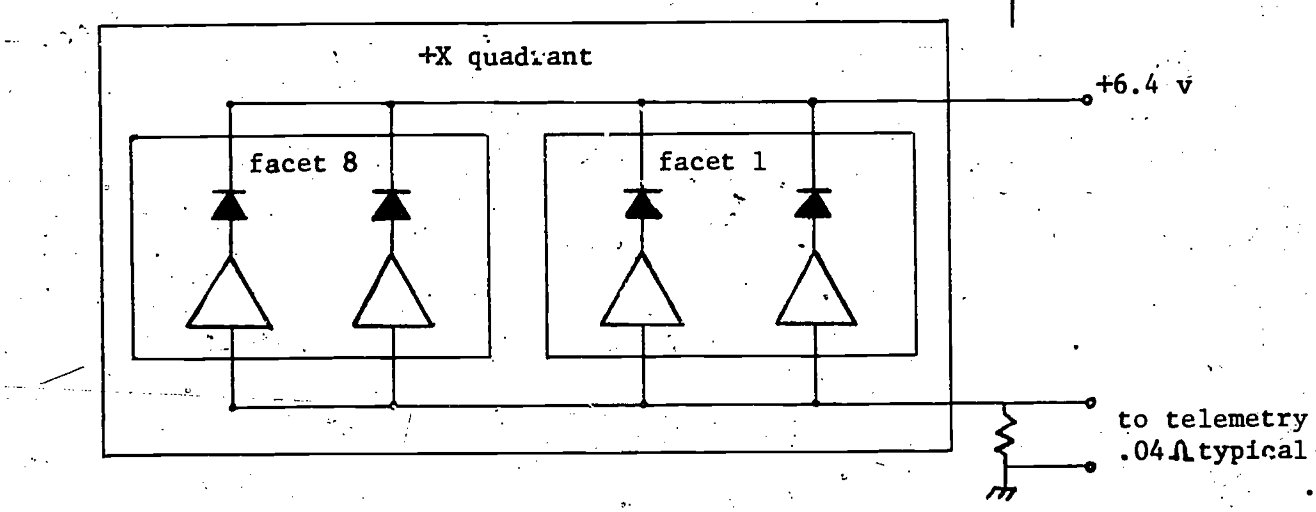


Figure 3a. AMSAT-OSCAR 7 solar cell configuration.

Solar Cell Characteristics

type: n on p silicon
size: 1 cm x 2 cm
total number: 1792
total surface area: 3,580 cm^2
protective cover: .015 cm glass cover slide
maximum power output per cell: 23 mW (58 mA x .405 v)
efficiency: 9% (before launch)
peak array output: 15 W (optimal sun orientation)

Solar Cell Configuration

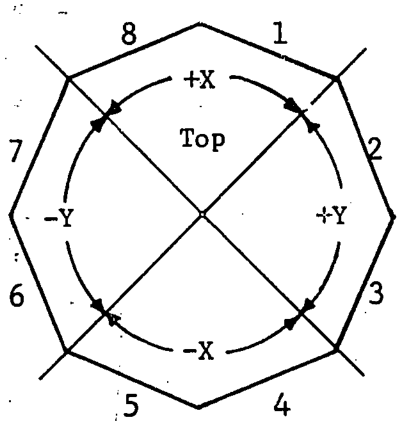
basic module: 7 parallel cells form a unit, 16 units in series
total number of modules: 16
location: Each spacecraft facet contains two modules in parallel; two adjacent facets paralleled to form quadrant; spacecraft contains four quadrants: +X, -X, +Y, -Y.

Storage Battery

type of cell: Nickel-Cadmium
voltage/cell. 1.35 v
capacity/cell: 6 Ampere-hours (Ah)
configuration: 10 cells in series
battery (100% charged): 13.5 v, 6 Ah
battery (50% charged): 12.1 v, 3 Ah

Switching Regulators

battery charge regulator: Converts 6.4 v solar array bus to 12 v (unregulated) main spacecraft power bus. Prevents battery from overcharging when it reaches 14 v. Fully redundant and autoswitching if regulator senses open or short.
instrumentation switching regulator: Provides well regulated + 10 v, -6 v and -10 v and precision reference of 0.5 v for all spacecraft logic functions and command receiver. Fully redundant except +.5v reference.
transponder regulator: Converts 12 v unregulated spacecraft bus to 24-28 v for use by the 2m/10m transponder power amplifier and driver stages. Fully redundant.



Cross-section parallel to top of AMSAT-OSCAR 7 showing relation between 8 facets (faces) and 4 quadrants.

Figure 3b. AMSAT-OSCAR 7 solar cell configuration.

SPACECRAFT NAME: AMSAT-OSCAR 8

GENERAL

1.1 Series: AMSAT Phase II (low-altitude, long-lifetime)

1.2 Launch
Date: March 5, 1978
Vehicle: Two-stage Delta 2910
Agency: U.S. National Aeronautics and Space Administration
Site: NASA Western Test Range; Lompoc, California (Vandenberg Air Force Base)

1.3 Orbital Parameters
General designation: low-altitude, near-circular, sun-synchronous
Period: 103.23 minutes/orbit
Apogee altitude: 912 km
Perigee altitude: 908 km
Eccentricity: .0009 (near-circular)
Inclination: 98.99°
Longitude increment: 25.72° west/orbit
Maximum access distance: 3,250 km

1.4 Ground Track Data
Data for drawing ground tracks is contained in Chapter II, Table 2.4.

1.5 Operating Schedule (tentative)
Monday-Friday (UTC): Mode A (2m/10m transponder and 29.402 MHz telemetry beacon)
Saturday-Sunday (UTC): Mode J (2m/70cm transponder and 435.095 MHz beacon)
Notes: On Wednesdays (UTC) the transponders are reserved for special experiments coordinated in advance with ARRL. On Mondays (UTC) transponder users are requested to limit themselves to 10 watts EIRP. Users should check QST or with ARRL (see Appendix A) to obtain updated schedule.

1.6 Construction Credits
Project management: AMSAT U.S.
Spacecraft subsystems: Designed and built by groups in Canada, Japan, United States, West Germany.

1.7 Primary Reference
P. Klein and J. Kasser, "The AMSAT-OSCAR D Spacecraft", AMSAT Newsletter, Vol. IX, no. 4, Dec. 1977, pp. 4-10.

SPACECRAFT DESCRIPTION

2.1 Physical Structure
Shape: Rectangular solid as shown in Figure 1, approximately 33 cm (height) by 38 cm (width) by 38 cm (depth).
Mass: 25.8 kg

2.2 System Description
Block diagram: See Figure 2.

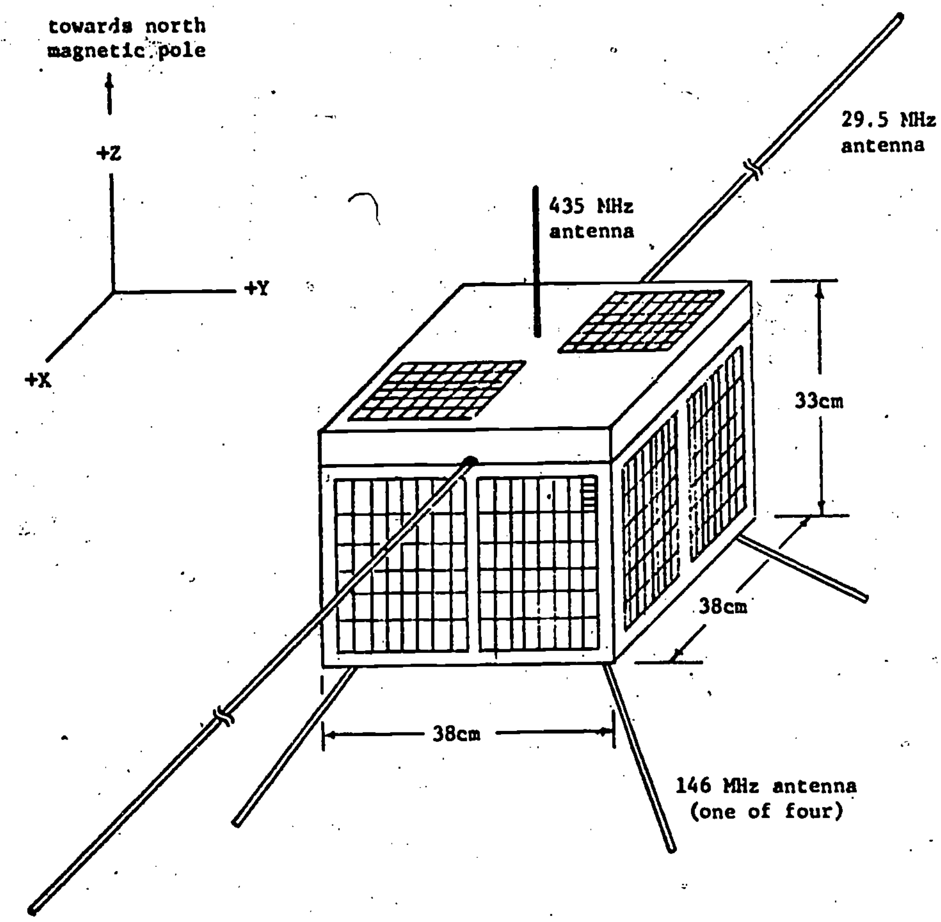


Figure 1. AMSAT-OSCAR 8.

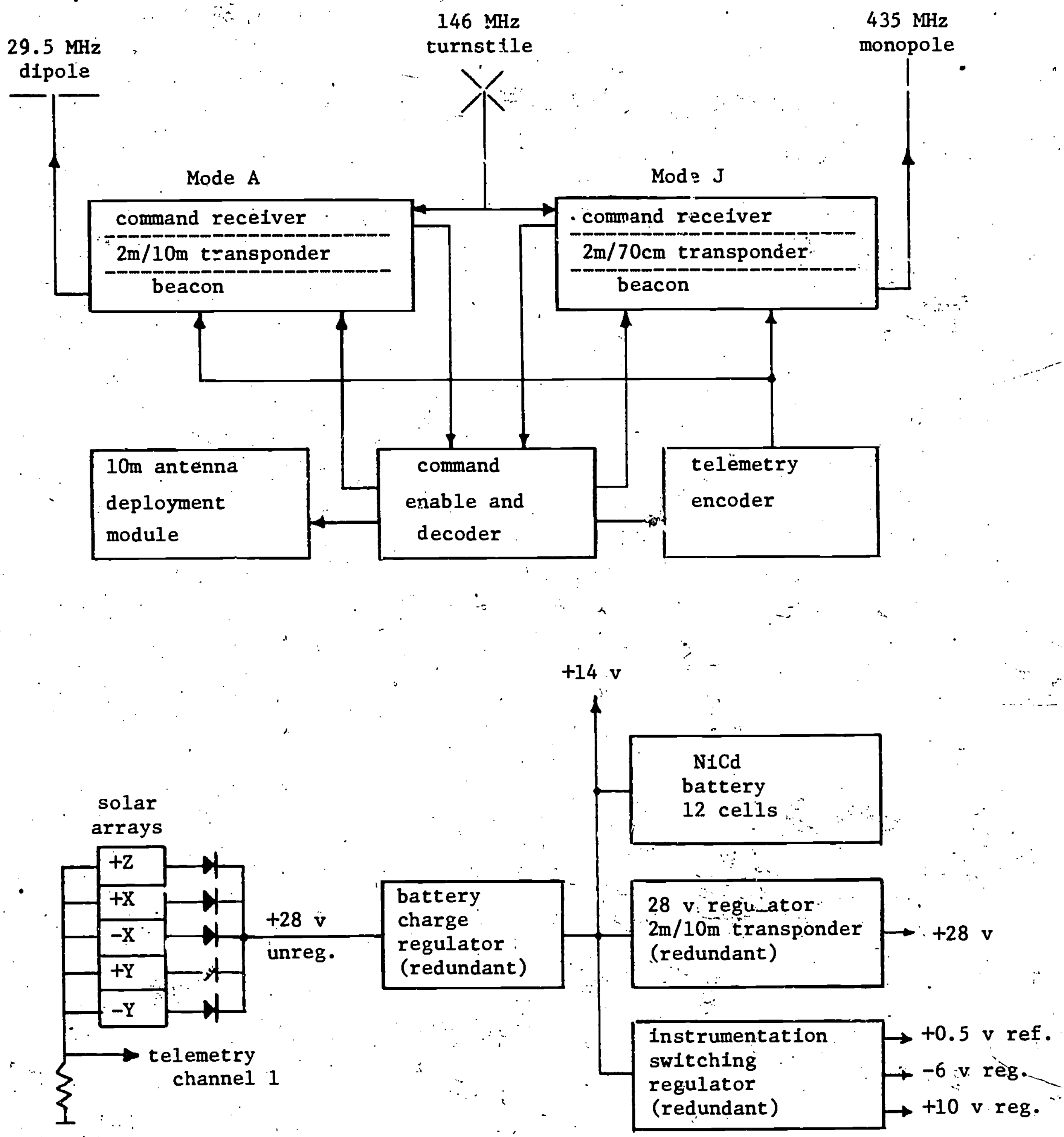


Figure 2. Block diagram of AMSAT-OSCAR 8

SUBSYSTEM DESCRIPTION

3.1 Beacons

29.402 MHz: (110 milliwatts) Operates in conjunction with Mode A.
435.095 MHz: (100 milliwatts) Operates in conjunction with Mode J.

3.2 Telemetry

Formats available: Morse code, special features

Morse code telemetry

frame: A frame contains six channels (six lines by one column). Parameters are sent in a fixed serial format.

channel: A channel consists of a three digit number. The first digit is a line identifier. Because of the single column format, the first digit uniquely identifies the parameter being measured. The last two digits in a channel encode the data as per Table 1.

speed: The telemetry is sent at 20 words per minute. A complete frame requires about 20 seconds.

sample data: Data from a nearby pass is shown in Table 2.

Special features

command enable: When the command system has been enabled and is ready to accept a command, the Morse code telemetry is interrupted and an unmodulated carrier is transmitted on the beacon frequency.

10 m antenna status: When the 10 m antenna deployment command is received at the satellite the beacon transmits a series of pulses. The pulse rate is a function of tip-to-tip antenna length. See 3.6 -- 29.5 MHz antenna.

Channel 1: Total Solar Array Current;	I = 7.15(101-N) ma. (*1)
Channel 2: Battery Charge-Discharge Current;	I = 57(N-50) ma. (*2)
Channel 3: Battery Voltage;	V = (0.1N + 8.25) volts
Channel 4: Baseplate Temperature;	T = (95.8 - 1.48N) °C
Channel 5: Battery Temperature;	T = (95.8 - 1.48N) °C
Channel 6: 435 MHz Transmitter Power Output;	P = 23 N milliwatts (*3)

Table 1. AMSAT-OSCAR 8 Morse code telemetry decoding information.

(*1) Whenever N is less than 10 assume that an overrange condition has occurred. For example, as the satellite enters the earth's shadow a reading of 101 is transmitted. This refers to channel 1, N = 01. Since N is less than 10 we assume that overranging has occurred and the actual N is 101 which corresponds to zero current.

(*2) There is a two-second integration time associated with the current telemetered on this channel.

(*3) There is a 2.5 second integration time associated with the power telemetered on this channel.

	1N	2N	3N	4N	5N	6N
(*1)	01	47	82	50	48	01
	01	47	82	51	48	01
	01	46	82	--	--	--
	--	--	--	--	--	--
	--	--	--	--	--	--
	01	46	--	--	--	--
	01	46	81	51	47	01
	01	46	81	51	47	01
(*2)	mm continuous tone mmm					
(*3)	01	41	81	51	47	12
	01	42	81	--	46	18
	01	41	81	51	--	15
	01	41	81	52	47	17
	01	--	82	51	46	17
	01	41	79	51	47	18
	01	44	79	51	47	18
	01	41	79	51	47	11
	01	41	79	51	48	19
	01	41	79	51	46	22
	01	41	79	51	46	17
	01	41	79	51	46	17
	01	41	79	51	46	23
	01	41	78	51	46	23
	01	41	78	51	46	19
	01	41	79	51	46	23
	01	41	78	52	46	21
	01	41	78	52	46	25
	01	41	78	52	46	24
	01	41	78	52	46	26
	01	41	78	52	46	23
	01	43	78	52	47	22
(*4)	98	43	79	52	48	22
	61	48	81	52	47	26
	51	48	81	52	47	17
	48	47	81	52	48	23
	51	48	81	52	48	23
	66	51	81	52	48	26
	53	49	81	52	47	19
	46	49	80	52	47	20
	58	49	80	52	47	18
	--	49	80	52	47	21
	64	48	80	52	48	22
	63	49	80	52	48	07
	63	49	80	52	48	13
	52	49	80	52	47	16
(*5)	51	49	80	52	48	--

	1N	2N	3N	4N	5N	6N
(*6)	01	42	76	50	47	17
	01	41	77	50	47	17
	01	41	77	50	47	17
	01	41	76	50	47	--
	01	41	76	50	47	21
	01	40	76	50	46	21
	01	41	76	51	47	17
	01	41	76	51	47	14
	01	41	76	51	47	20
	01	41	76	51	46	mm
(*2)	mm continuous tone mmm					
(*7)	01	46	77	51	47	01
	01	46	77	51	47	01
	01	45	77	51	47	01

(*1) Acquisition of orbit #61 at 02:12:28 UTC, 10 Mar. 1978 (ascending node 02:09:20 UTC, 69.9°W).

(*2) Command station accessing satellite.

(*3) Mode J turned on (see channel 6), mode A remains on (telemetry being copied on 29.402 MHz).

(*4) Satellite crossing terminator into daylight (see channels 1 and 2).

(*5) Loss of orbit #61 at 02:28:25 UTC.

(*6) Acquisition of orbit #62 at 03:59:25 UTC, 10 Mar. 1978 (ascending node 03:52:32 UTC, 95.7°W).

(*7) Mode J turned off, mode A remains on.

Table 2. AMSAT-OSCAR 8 telemetry copied on 29.402 MHz beacon 10 March 1978. Courtesy of Richard Zwirko.

3.3 Command System

The command system recognizes five commands as per Table 3.

Command	Spacecraft Status
Mode A Select	2m/10m transponder and 29.402 MHz beacon ON
Mode J Select	2m/70cm transponder and 435.095 MHz beacon ON
Mode D Select	Recharge mode. Both transponders and beacons OFF
10 m Antenna Deployment	Activates 10 m antenna deployment mechanism and switches telemetry to pulse format encoding tip-to-tip length of antenna.
10 m Antenna Reset	Stops deployment of 10 m antenna (deployment can not be reversed). Switches telemetry back to Morse code.

Table 3. AMSAT-OSCAR 8 Commands.

3.4 Transponders

Transponder I: Mode A (2m/10m)

type: linear, non-inverting
uplink passband: 145.850 - 145.950 MHz
downlink passband: 29.400 - 29.500 MHz
translation equation: downlink freq. (MHz) = uplink freq. (MHz) - 116.458 MHz ± Doppler
output power: 1 to 2 watts PEP
uplink EIRP: (tentative) a maximum of 80 watts is recommended
delay time: about 10 microseconds
comments: The same basic Mode A transponder has been used on AMSAT-OSCARs 6, 7, 8. A block diagram is shown in Figure 3.

Transponder II: Mode J (2m/70cm)

type: linear, inverting
uplink passband: 145.900 - 146.000 MHz
downlink passband: 435.100 - 435.200 MHz
translation equation: downlink freq. (MHz) = 581.100 - uplink freq. (MHz) ± Doppler
output power: 1 to 2 watts PEP. Telemetry channel six measures the output power using a 2.5 second integration time.
uplink EIRP: (tentative) a maximum of 8 watts is recommended. Under certain conditions of spacecraft temperature and battery voltage transponder sensitivity may decrease and 80 watts may be needed.
delay time: less than 10 microseconds (estimated)
comments: This transponder was constructed by the Japan AMSAT Association of Tokyo to test the effectiveness of this link for low-altitude spacecraft.

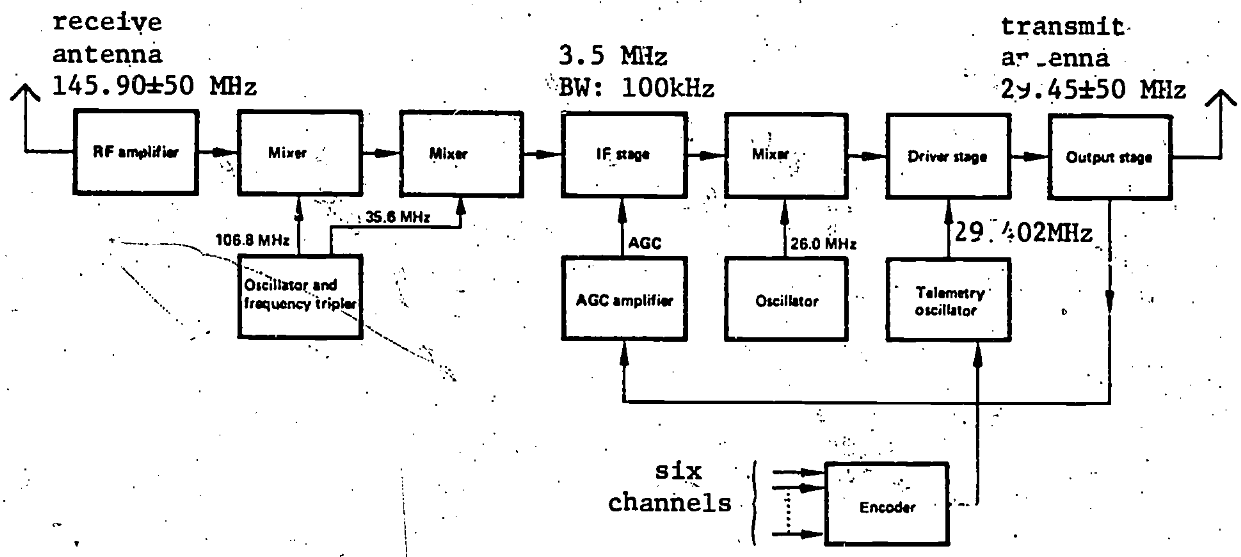


Figure 3. Block diagram of AMSAT-OSCAR 8 transponder.

3.5 Attitude Stabilization

Primary control: Four Alnico 5 bar magnets, each approximately 15 cm long and with a square cross-section of about .6 cm by .6 cm are mounted parallel to the Z-axis of the spacecraft. The resultant far field is similar to that produced by a single 30,000 pole-cm magnet. As the satellite moves along its orbit the Z-axis of the spacecraft constantly changes its direction in inertial space in order to remain aligned parallel to the local direction of the earths magnetic field. The +Z-axis (top) of the satellite points in the direction of the earth's north magnetic pole.

Damping: Allegheny Ludlum type 4750 permalloy hysterisis damping rods (.32 cm diameter) are mounted behind, and parallel to, the +X, -X, +Y, and -Y solar panels (perpendicular to the Z-axis) to damp out rotational motion about the Z-axis.

3.6 Antennas (See Figure 1)

29.5 MHz: The 29.5 MHz transmitting antenna is a half wavelength dipole (about 4.9 m) mounted perpendicular to the Z-axis. It is composed of tubular extendable members which are deployed by small motors activated by ground command. To prevent damage this antenna will not be deployed until the satellite spin rate is less than 2 rpm. This should occur within one week of launch. The deployment process takes about 15 seconds, it's non-reversible, and it can be monitored by ground stations listening to the telemetry signals. When the satellite receives the 10 m Antenna Deployment command the telemetry system sends a series of pulses. The pulse rate is a function of tip-to-tip antenna length. In the fully retracted position (launch state) the rate is about 15 pulses/sec. When the antenna is fully deployed the rate is 1.8 pulses/sec.

146 MHz: The 146 MHz receiving antenna for both transponders is a canted turnstile. It consists of two "inverted V" shaped dipoles mounted at right angles on the base (-Z face) of the spacecraft. Each dipole consists of two 48 cm spokes (1/4 wavelength) constructed from a material similar to 1 cm wide carpenter's rule. The turnstile is fed by a hybrid ring and matching network. It produces an elliptically polarized radiation field (circularly polarized along -Z-axis) over a large solid angle. The gain approaches 5 dB along the -Z axis; there's some shadowing along the +Z-axis.

435 MHz: The 435 MHz transmit antenna is a 1/4 wavelength monopole mounted on the top (+Z face) of the spacecraft.

Notes: Signal polarizations of the spacecraft antennas are summarized in Table 4.

AMSAT-OSCAR 8 ANTENNA POLARIZATIONS

System	Spacecraft Polarization
2m/10m transponder uplink (146 MHz)	left-hand circular (*1)
2m/10m transponder downlink and 29 MHz beacon	linear
2m/70cm transponder uplink (146 MHz)	right-hand circular (*1)
2m/70cm transponder downlink and 435 MHz beacon	linear

Note: (*1) Polarization sense referenced to +Z-axis of spacecraft. Ground stations off the +Z-axis will observe elliptical polarization. Stations north of the magnetic equator (see Figure 3.6) will generally find that the circular component is as indicated in the table. Stations in the southern hemisphere will generally find the circular component reversed.

Table 4. AMSAT-OSCAR 8 antenna polarizations

3.7 Energy-Supply and Power Conditioning

The main components of the AMSAT-OSCAR 8 energy-supply and power conditioning subsystem are shown in Figure 2.

<u>Solar Cell Characteristics</u>

type: n on p silicon
size: 1 cm x 2 cm
total number: 1,920
total surface area: 4,005 cm^2
protective cover: .015 cm glass cover slide
maximum power output per cell:
efficiency: 8% (before launch)
peak array output: 15 w (optimal sun orientation)

<u>Solar Cell Configuration</u>

basic module: 80 cells in series
total number of modules: 24
location: +X, -X, +Y, -Y facets have 5 modules each; +Z facet has 4 modules.

<u>Storage Battery</u>

type of cell: Nickel-Cadmium
voltage/cell: 1.45 v (fully charged)
capacity/cell: 6 Ampere-hours (Ah)
configuration: 12 cells in series
battery (100% charged): 17.4 v, 6 Ah
battery (50% charged): 14.5 v, 3 Ah

<u>Switching regulators</u>

battery charge regulator: Converts 28-30 volt solar array bus to 14-16 volt main spacecraft power bus. Tapers charge rate to prevent overcharging at a battery voltage of 17.4 volt. Fully redundant and autoswitching if regulator senses open or short.
instrumentation switching regulator: Provides well regulated +10 v, -6 v, and precision reference of +0.5 v for all spacecraft systems. Fully redundant.
transponder regulator: Converts 14-16 volt unregulated spacecraft bus to 24-28 volts for use by the 2m/10m transponder power amplifier and driver. Fully redundant.

SPACECRAFT NAME: AMSAT Phase III-A

GENERAL

1.1 Series: AMSAT Phase III (high-altitude, long-lifetime)

1.2 Launch
Date: late 1979 or early 1980
Vehicle: Ariane
Agency: European Space Agency (ESA)
Site: Kuru, French Guiana

1.3 Orbital Parameters (projected)
General designation: elliptical, synchronous-transfer
Period: 660 minutes
Apogee altitude: 35,800 km
Perigee altitude: 1,500 km
Eccentricity: .685
Inclination: 57°
Longitude increment: 165°west/orbit
Maximum access distance: 9,040 km
Argument of perigee: biased about two years before southern most point (270°)
Note: The Phase III-A spacecraft will use its own propulsion system to attain the above orbit after being placed in a temporary orbit by the launch vehicle. The projected temporary orbit is:
apogee altitude: 35,000 km
perigee altitude: 200 km
inclination: 17°

1.4 Ground Track Data
Ground track data will be disseminated shortly before launch.

1.5 Operating Schedule
Transponders will operate 24 hours per day. Details available shortly before launch.

1.6 Construction Credits
Project management: AMSAT U.S.
Spacecraft subsystems: Designed and built by groups in Canada, United States, West Germany.

1.7 Primary References
1) J. A. King, "Phase III: Toward the Ultimate Amateur Satellite",
Part 1, QST, Vol. LXI, no. 6, June 1977, pp. 11-14;
Part 2, QST, Vol. LXI, no. 7, July 1977, pp. 52-55;
Part 3, QST, Vol. LXI, no. 8, Aug. 1977, pp. 11-13.
2) M. Davidoff, "The Future of the Amateur Satellite Service",
Ham Radio, Vol. 10, no. 8, Aug. 1977, pp. 32-39.

SPACECRAFT DESCRIPTION

2.1 Physical Structure
Shape: Tri-star as shown in Figure 1.
Mass: Approximately 68 kg at launch (about 65% of mass is fuel)

2.2 System Description
Block diagram: See Figure 2.

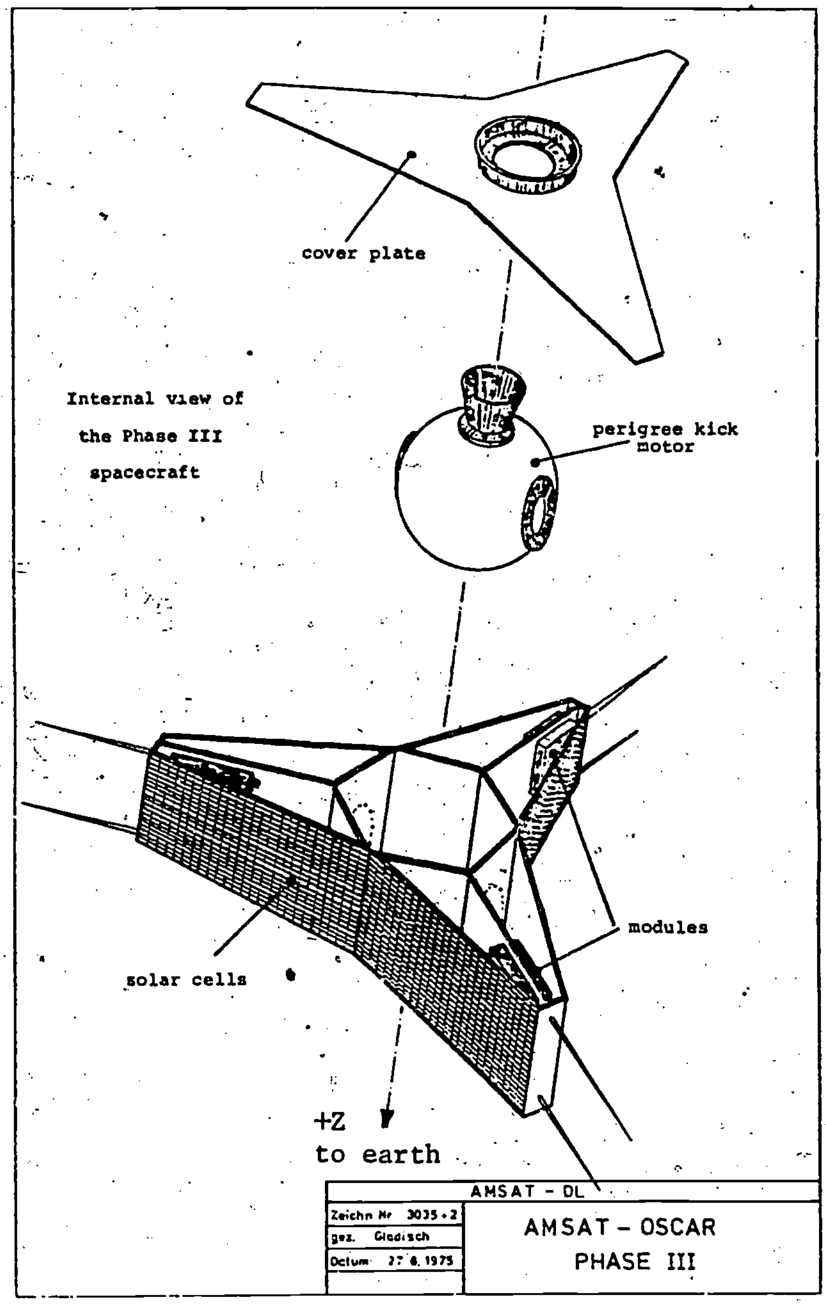


Figure 1. AMSAT Phase III-A

PHASE - III S/C FUNCTIONAL BLOCK DIAGRAM

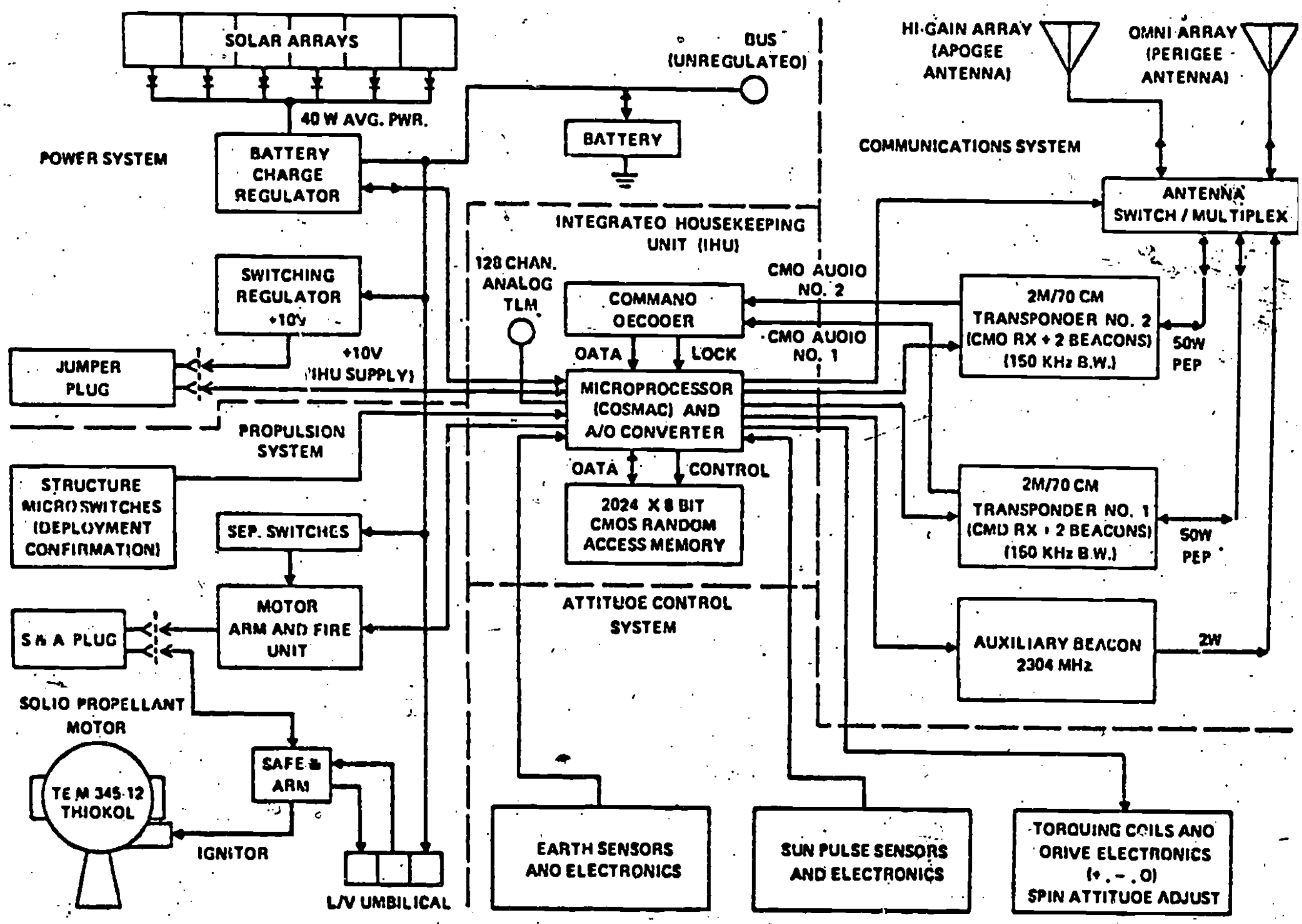


Figure 2 AMSAT Phase III-A satellite functional block diagram.

SUBSYSTEM DESCRIPTION

3.1 Beacons (tentative)

145.845 MHz: Mode B engineering beacon (PSK modulated)
145.995 MHz: Mode B educational beacon (FSK modulated)

435.145 MHz: Mode J educational beacon (FSK modulated)
435.300 MHz: Mode J engineering beacon (PSK modulated)

Notes: The educational beacons will probably carry Morse code telemetry and Codestore messages. The engineering beacons will be used for high data rate -- 400 bits-per-second (PSK/PCM) -- telemetry. The Mode J beacons might not be included on the first Phase III satellite. An S-band (2304.1 MHz) beacon may be included on Phase III-A. It will have a power output of 2 watts and be operated on an as-needed basis.

3.2 Telemetry

Formats available: Morse code, radioteletype, Codestore, others. The telemetry format will be controlled by software residing in the spacecraft computer. The software can be altered via the command links or stored program.

Comments: Plans are to program the satellite computer so that the educational beacons will use Morse code telemetry and Codestore. For example, Codestore might be used the first five minutes of each hour and Morse code telemetry the rest of the time. Engineering beacons will be used for high speed -- probably 400 bits-per-second -- data acquisition by ground stations equipped with microprocessors. Decoding information will be made available shortly before launch.

3.3 Command System

See block diagram of spacecraft (Figure 2). Uplink will load new program directly into memory via microprocessor interrupt feature.

3.4 Transponders

Transponder I: Mode B (70cm/2m)

type: linear, inverting
uplink passband: 435.150 - 435.290 MHz
downlink passband: 145.850 - 145.990 MHz
translation equation: downlink freq. (MHz) = 581.140 - uplink freq. (MHz) ± Doppler
output power: about 50 watts PEP
uplink EIRP: about 500 watts (tentative)
comments: The transponder, designed by Dr. Karl Meinzer, employs the envelope elimination and restoration technique described in Reference 3 of Chapter III. An article describing the design of this transponder is being prepared for publication.

Transponder II: Mode J (2m/70cm)
- type: linear, inverting
- uplink passband: 145.850 - 145.990 MHz
- downlink passband: 435.150 - 435.290 MHz
- translation equation: downlink freq. (MHz) = 581.140 - uplink freq. (MHz) ± Doppler
- output power: about 50 watts PEP
- uplink EIRP: 500 watts (tentative)
- comments: Due to time constraints and/or results with AMSAT-OSCAR 8 this transponder may be replaced with a second Mode B unit aboard Phase III-A. Also see comments under Transponder I.

3.5 Attitude Stabilization

General: The satellite will be spun about its Z-axis at 90-120 rpm. This will serve to "fix" the direction of the Z-axis in inertial space. Initial spin-up will be accomplished by three solid propellant spin rockets, one at the end of each arm. Torquing coils will provide a back up to the spin rockets and also permit changes in the direction of the spin axis and spin rate throughout the satellites lifetime. Pulsing these coils with current at the proper rate and time (near perigee) can produce changes in spin rate and direction of spin axis through the interaction between the magnetic fields of the earth and the coils. A ground station will load the satellite computer with a pulsing program when needed. Generally, the spin axis will be adjusted to point toward the geocenter at apogee. However, this may have to be modified at times if the resultant sun orientation with respect to the spacecraft solar cells is very poor.

Sensors: Details of the sun and earth sensors are not yet available.

3.6 Antennas (See Figure 1)

146 MHz: The 146 MHz antenna uses the six 48 cm spokes extending from the arms of the spacecraft. It is composed of two sets of three elements, each set lying in a plane perpendicular to the Z-axis. Elements in a set are fed using phase delays of 0°, 120°, and 240°. Gain: about 9 dB_i along axis.

435 MHz: The 435 MHz antenna will consist of either (1) a quadrifilar helix mounted on the +Z face providing about 7 dB_i gain, (2) the 146 MHz antenna reused in a third harmonic mode, also providing about 7 dB_i gain or (3) a second set of 3 elements mounted on the +Z face fed by 120° phase delay lines, gain about 9.5 dB_i.

2304.1 MHz: Not yet determined

3.7 Energy-Supply and Power Conditioning

The main components of the Phase III-A energy-supply and power conditioning subsystem are shown in Figure 2.

Solar Cell Characteristics (all values tentative)

type: n on p silicon
size: 2 cm x 2 cm
total number: 2,700
total surface area: 1.1m^2
protective cover: .03 cm glass cover slide (to minimize radiation degradation cover slides are thicker than on Phase II satellites)
maximum power output per cell:
efficiency: greater than 10% (before launch)
peak output: about 45 watts (optimal sun orientation)

Solar Cell Configuration (all values tentative)

basic module: 3 cells in parallel by 68 in series = 204 cells/module
total number of modules: 12 (2 per spacecraft facet)
location: The six facets of the spacecraft will contain identical solar panels each consisting of about .2 m^2 of cells.

Storage Battery (all values tentative)

type of cell: Nickel-Cadmium
voltage/cell: 1.45 v
capacity/cell: 6 Ampere-hours (Ah)
configuration: 12 cells in series (subject to change)
battery (100% charged): 17.4 v, 6 Ah
battery (50% charged): 14.5 v, 3 Ah

Switching Regulators

See Figure 2.

3.8 Propulsion System

A solid-propellant motor (apogee-kick-motor) containing approximately 35 kg of a mixture of powdered aluminum and organic chemicals in a spherical shell with a single exit nozzle will produce a velocity change of about 1,600 m/s during its single 20-second burn.

3.9 Integrated Housekeeping Unit (IHU)

As shown in Figure 2, the IHU will consist of a CMOS microprocessor (RCA COSMAC), at least 2048 bytes of random access memory (RAM), a command decoder and an analog-to-digital converter. The IHU is responsible for controlling virtually every function on board the spacecraft. It will execute all telemetry and command requirements, monitor the condition of the power and communications systems, and take corrective actions as necessary. It will establish clocks needed for various spacecraft timing functions, and it will interact with the attitude sensors. In addition, the IHU will make the final decision on whether all on board systems are "go" for the kick-motor firing. If confirmed, the computer will send the command to fire, not a ground control station.

SPACECRAFT NAME: "RS-1"

GENERAL

Note: All information below is based on prelaunch specifications and must be regarded as tentative.

1.1 Series: U.S.S.R. Amateur System "RS" (low-altitude, long-lifetime)

1.2 Launch

Date: Expected mid 1978
Site: Pletsetsk, U.S.S.R.

1.3 Orbital Parameters

General designation: low-altitude, near-circular
Period: 102 - 104 minutes
Altitude: 860 - 950 km
Inclination: 82°
Maximum access distance: 3,140 km at 860 km altitude

1.4 Ground Track Data

Data for plotting the ground track for 102 minute period, 860 km altitude, orbit is contained in Chapter II, Table 2.4.

1.5 Operating Schedule

Information not available at presstime.

1.6 Construction Credits

Designed and built by members of the Central Radio Club of Moscow and the Radio Club of the Technical University of Moscow.

1.7 Primary References

(1) Special Section No. SPA-AA/159/1273 annexed to International Frequency Registration Board Circular No. 1273 dated 12 July 1977 submitted by USSR Ministry of Posts and Telecommunications
(2) V. Dobrozhanskiy, "Radioamateur Satellites; The Repeater: How is it used?", Radio, No. 9 (Sept.), 1977, pp. 23-25.
(3) Also see July, Oct., and Nov. 1977 issues of Radio for additional information.

SPACECRAFT DESCRIPTION

2.1 Physical Structure

Information not available at presstime

2.2 System Description

Information not available at presstime

SUBSYSTEM DESCRIPTION

3.1 Beacons

29.398 MHz: (100 milliwatts ?)
Note: Additional beacons may be present.

3.2 Telemetry

Morse code telemetry system expected

3.3 Command System

A command system is included in the spacecraft. The announced operating frequency is in the vicinity of 145.850 MHz.

3.4 Transponders

Transponder I: Mode A (2m/10m)

type: linear, non-inverting
uplink passband: 145.800 - 145.840 MHz
downlink passband: 29.350 - 29.390 MHz
translation equation: downlink freq. (MHz) = uplink freq. (MHz) - 116.450 ± Doppler
output power: 1.5 watts PEP
uplink EIRP: A maximum of 100 watts is recommended. Downlink signals should never be stronger than the beacon.

Transponder II:

Plans are to include two transponders on RS-1. They will probably be either identical or nearly so (differing slightly in uplink and downlink frequencies).

3.5 Attitude Stabilization

Primary control: probably permanent magnet as on recent AMSAT satellites

3.6 Antennas

29.5 MHz: The 29.5 MHz antenna is described as "Half-wave [...], G = 1, circular characteristic". It appears that an attempt will be made to use a circularly polarized turnstile but mechanical problems may necessitate a simple half-wavelength dipole.

146 MHz: The 146 MHz antenna is described as "Quarter wave [...], G = 1, circular characteristics". This probably refers to a turnstile.

3.7 Energy-Supply and Power Conditioning

Although no information has been published the fact that this is a long lifetime spacecraft suggests that solar cells and Nickel-Cadmium batteries will be used.

CHAPTER V

GROUND STATION EQUIPMENT

The objectives of this chapter include:

1. Describing the radio equipment needed to receive signals from the AMSAT-OSCAR satellites,
2. Discussing antenna characteristics and how these characteristics may be matched to ground station requirements when choosing an antenna system,
3. Presenting construction information for simple ground station antennas,
4. Describing some accessory equipment useful in setting up and operating a ground station,
5. Discussing licensing requirements and transmitting equipment for uplinks.

We begin this chapter by discussing the requirements for a basic ground station designed to receive radio signals from the AMSAT-OSCAR satellites. After the basic ground station is presented we turn first to a detailed discussion of receivers and then to antennas, focussing on topics important to satellite users. The emphasis is always on understanding the function of and requirements for each piece of equipment so that readers can put together a station that meets their needs in terms of equipment already available, proficiency in electronics, and time and funds which can be devoted to the project. This chapter does not include detailed construction information for electronics equipment. However, references are provided to a number of excellent construction articles for readers interested in this area and construction information is included for some simple antennas. Please note that a list of addresses for all manufacturers and publishers mentioned in this chapter is contained in Appendix A.

5.1 BASIC 29 MHz GROUND STATION

The basic 29 MHz ground station, as shown in Figure 5.1, consists of an HF (high frequency) communications receiver and an antenna. We look first at the receiver. The HF receiver can be either a GC (general coverage) or amateur-bands-only model, as long as it covers 29.000 - 29.500 MHz, the frequency region where the signals from the AMSAT-OSCAR and Russian RS satellites can be found. It must be a communications receiver, i.e., capable of detecting Morse code and SSB (single sideband) voice signals. Most receivers produced in the last 20 years for the amateur radio operator market which cost $250. or more when new will work well if in decent condition. A new top-of-the-line receiver produced by one of the manufacturers listed in section 5.2 will provide excellent results for about $600. Used receivers of the type needed are often available for well under $100. When considering a receiver keep in mind that an amateur-bands-only model will generally outperform a GC receiver of similar cost.

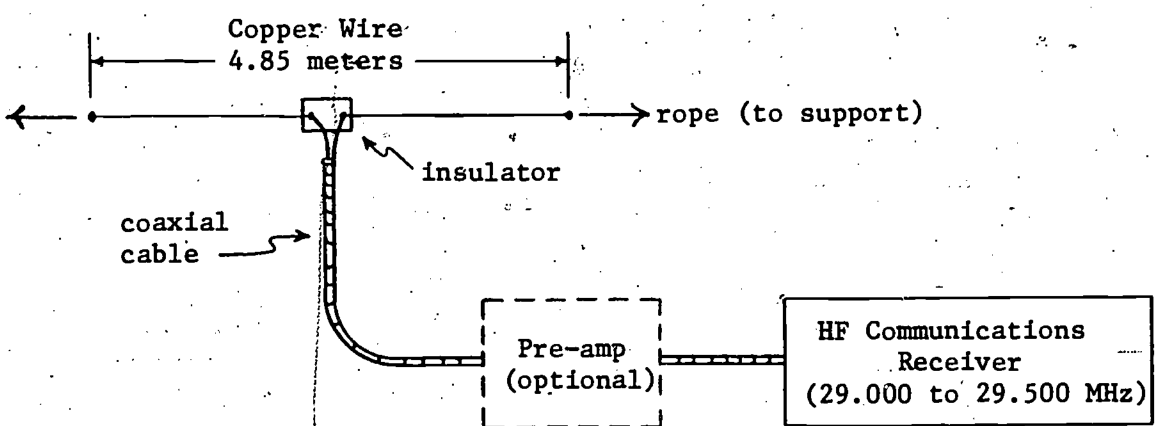


Figure 5.1. Basic satellite ground station for 29.000 to 29.500 MHz. The following coaxial cables can be used (in order of preference): RG-8/U, RG-11/U, RG-58/U, RG-59/U.

A convenient antenna for the basic ground station is the half-wave dipole pictured in Figure 5.1. It can either be mounted horizontally as shown or hung vertically from one end. The best performance will be obtained if it is mounted high and clear of surrounding objects.

Most educators working with the AMSAT satellites begin with a station like the one just described. Once some experience is gained additional features can be added. For example, if the signals received from the satellite are very weak a pre-amplifier inserted between the antenna and receiver often significantly improves results at a cost of about $20. for a new unit (see section 5.2). This improvement is often noted even with new expensive receivers since these models weren't optimized for the low background noise encountered when receiving signals from space. We now turn to a detailed discussion of communications receivers and antennas for satellite ground stations.

5.2 RECEIVERS

Performance requirements for a receiver used for satellite reception are outlined below. We begin by considering the 29 MHz downlinks and then turn to the higher frequency links.

<u>29 MHz Receivers.</u> A receiver can be characterized on a number of performance parameters including:

<u>(1) Tuning Range.</u> The receiver must tune the frequencies between 29.000 and 29.500 MHz where the downlink radio signals from the satellites are found. The specific frequencies are listed in Chapter IV. As we'll see shortly, a range of 28.000 to 30.000 MHz is desirable so that equipment for receiving the 146 MHz and 435 MHz satellite downlinks can be added at a later date.

<u>(2) Detector.</u> The receiver must contain a detector which is capable of demodulating the continuous wave (cw) Morse code and single sideband suppressed carrier (SSB) signals which are found on the downlinks. This requirement will be met if the receiver contains either a beat frequency oscillator (bfo) or product detector.

<u>(3) Stability.</u> The receiver must be stable -- the frequency it is tuned to should not drift more than 1 kHz over a period of 30 minutes (after an initial 30 minute warmup period). In addition, any frequency changes due to line voltage fluxuations or mechanical vibrations (slamming doors, etc.) should not exceed 200 Hz.

<u>(4) Tuning Mechanism.</u> The receiver should have a good quality tuning mechanism. Frequency shift when the tuning knob is released (backlash) should not exceed 200 Hz. Frequency readout should be better than 5 kHz (preferably better than 1 kHz) so that the beacons can be easily located.

<u>(5) Sensitivity.</u> The receiver should be extremely sensitive -- a signal of 0.1 microvolt at 29.5 MHz should be perfectly discernable.

A receiver meeting all of the above requirements will provide excellent performance when used to receive satellite signals. A receiver which doesn't meet all of the requirements can sometimes be used if it is modified or used in conjunction with outboard equipment. For example, a receiver with poor frequency readout and long term stability can still be used for accurate frequency measurements if an external oscillator, tuned to the incoming signal, and a frequency counter are available. In general, any amateur or communications type receiver covering the high frequency (HF) portion of the radio spectrum (3 - 30 MHz) purchased in the last 20 years which sold for $250. or more when new should meet all of the requirements except for (5) (sensitivity). Most receivers, even the most expensive models, do not meet this requirement because they've been designed for communications via the ionosphere. Space communications can effectively utilize higher sensitivity. This problem is easily remedied by adding a pre-amplifier between the antenna and receiver. Suitable pre-amps can be purchased for under $20. (see Table 5.2) or constructed using a few dollars worth of parts. For construction information request the AMSAT Newsletter Reprint Booklet from the ARRL.

If you purchase a new receiver plan to spend between $300 and $600. Some suitable ground station receivers which are currently available are listed in Table 5.1 -- the list is representative, not inclusive. Keep in mind that the comments in Table 5.1 only reflect one person's subjective views and that every receiver mentioned performs adequately at 29.5 MHz when used with a pre-amp.

If you are comfortable around an electronics shop and have the proper facilities, an older used receiver, either purchased for considerably under $100. or rescued from a storeroom shelf, can provide good service. A proficient electronics constructor has a number of additional options for a ground-station receiver. Possibilities include converting a CB radio (the 23 channel SSB models with offset tuning are easy to convert and often available very inexpensively) or using a modified broadcast band radio (car radios are generally good for this purpose) and a "home brew" converter from 29.5 MHz to the broadcast band. For details the reader should consult recent issues of the radio amateur magazines.

Since all the downlink radio signals from the AMSAT-OSCAR satellites are in amateur designated bands, you may find that one of your students holds an amateur license and owns a suitable receiver. The student might be willing to help with a demonstration or be interested in doing laboratory projects using his/her own equipment.

VHF and UHF Receivers. The very high frequency (VHF) portion of the radio spectrum runs from 30 to 300 MHz, the ultra high frequency (UHF) portion from 300 MHz to 3 GHz. Users of AMSAT-OSCAR satellites will mainly be interested in that part of the VHF spectrum between 145.500 and 146.000 MHz and the portion of the UHF spectrum between 435.000 and 435.500 MHz. Specific beacon and transponder frequencies are listed in Chapter IV. Techniques and equipment for the two ranges are similar. Although one can purchase a separate VHF or UHF receiver most ground stations receive these frequencies by adding a converter and appropriate antenna ahead of their HF receiver. A block diagram of a sophisticated ground station with VHF and UHF capabilities and a number of other features is shown in Figure 5.2.

A converter "converts" (shifts) a slice of the radio spectrum to a different region of the spectrum. For example, when the appropriate VHF converter is used, tuning the HF receiver between 28.000 and 30.000 MHz will enable one to tune between 144.000 and 146.000 MHz. This specific frequency combination is often employed. The cost of a new VHF or UHF converter is from $30. to $90. If you purchase a used unit make sure it is a recent model since these devices have seen a great deal

Receiver	Comments
Drake R4C ($600.)	Excellent for satellite groundstation.
Drake SPR-4 ($550.)	Design optimized for laboratory and general purpose uses rather than SSB and CW communications.
Drake SSR-1 ($300.)	General coverage receiver. Does not perform as well as higher priced units but adequate for simple experiments if pre-amp is used.
Heath SB303 ($330.)	Excellent receiver but only available in kit form.
Heath HR-1680 ($200.)	This receiver requires a simple modification to cover 29.000 to 29.500 MHz -- details available from Heath. Performs well when used with a pre-amp. Only available in kit form.
Kenwood R599D ($460.)	Excellent receiver. An R599A has logged over 2,000 miles in the trunk of this author's car for demonstrations without a single malfunction. The dial readout is awkward on OSCAR downlink frequencies. A built in 146 MHz converter is available but it requires a pre-amp for adequate sensitivity.
Kenwood R300 ($240.)	General coverage receiver. Does not perform as well as higher priced units but adequate for simple experiments if pre-amp is used.
Yaesu FR101 ($500.)	Excellent receiver.
Yaesu FRG-7GC ($300.)	General coverage receiver. Does not perform as well as higher priced units but adequate for simple experiments if pre-amp is used.

Table 5.1. Some currently available HF receivers suitable for use with OSCAR satellites. List does not include transceivers, nor does it include receivers costing more than $600. 1978 list prices shown in ( ). Please note, the absence of critical comments usually implies that this author has not had the opportunity of testing the unit described.

of improvement in recent years. A number of converter and pre-amp manufacturers are listed in Table 5.2 (addresses are contained in Appendix A). The higher priced units are generally more sensitive and less prone to interference from radio transmitters in other parts of the radio spectrum. If an older model or low cost converter is available, a pre-amp will often improve reception considerably. A noise figure below 2 dB is desirable for the first stage following the antenna (the lower the noise figure the better the performance). At VHF and UHF it sometimes helps to mount a pre-amp directly at the antenna, even when a good converter is used. However, this usually entails placing the pre-amp where it is exposed to the weather which may cause reliability problems. Mounting the pre-amp at the antenna is usually not necessary unless one uses more than 30 meters of coaxial cable (see section 5.4) between the antenna and converter (or pre-amp) or desires optimal performance.

Manufacturer	146 MHz converter	435 MHz converter	146 MHz pre-amp	29.5 MHz pre-amp
Data Signal	---	---	$14/$28	$14/$28
Janel	$80	$85	$16/$20	$20
Kenwood	$29 (mounts in R599 receiver)	---	---	---
Spectrum International	$54	$54	$30	?
Vanguard	$55	$60	$30/$37	$30
Yaesu	$39 (mounts in FR101 receiver)	---	---	---
Microwave Modules	?	?	?	?

Table 5.2. Converter and pre-amp manufacturers. Approximate 1978 prices shown. Addresses of manufacturers are in Appendix A.

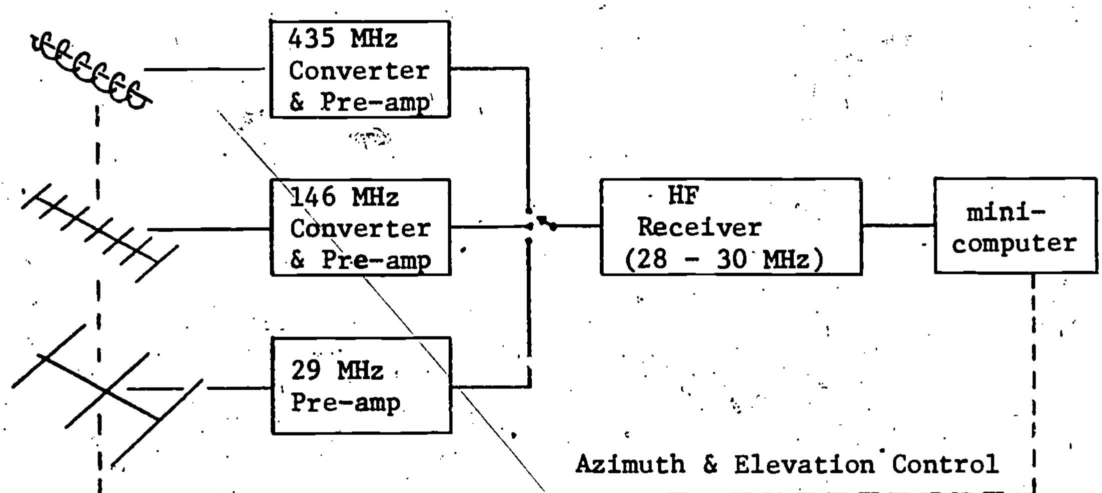


Figure 5.2. Advanced ground station for working with OSCAR satellites.

5.3 ANTENNA BASICS

The ground station receiving antenna has an extremely important effect on overall station performance. Choosing an antenna consists of matching (under various constraints) antenna properties to one's receiving requirements -- there is no single "best" antenna nor is an antenna with higher gain necessarily better than one with lower gain. It is therefore important to have a general understanding of antenna properties and how they are specified. Our objective when choosing an antenna is to provide an adequate signal to noise (S/N) ratio at the output of the receiver while minimizing the complexity of the system. What constitutes an adequate S/N ratio depends on the experiments to be performed but, in general, our requirements are modest. We do <u>not</u> need 99.9% reliability nor a dynamic range suitable for symphonic music. Once again our approach is practical, qualitative and intuitive. For a more comprehensive treatment see the <u>ARRL Antenna Book</u> and <u>Antennas</u> by Kraus [1]. We begin this section by discussing antenna properties and then turn to some practical downlink antennas. The antenna properties we will be looking at include:

1. Directional properties (patterns, directivity, gain, efficiency),
2. Transmitting vs. Receiving properties,
3. Polarization.

<u>Antenna Directional Properties</u>. One very important characteristic of a receiving antenna is its relative effectiveness at extracting power from radio signals arriving from different directions. An antenna that extracts power equally well from signals arriving from any direction is called an isotropic antenna. No practical antenna has this property but we introduce the concept because the isotropic antenna is a useful "measuring stick" for comparing other antennas to. An antenna that selectively extracts power from a certain preferred direction is called a beam. There are many types of beam antennas.

Problem 5.1.

Assume that we have a choice of two antennas to use for receiving satellite downlink signals -- the "imaginary" isotropic and a beam with a small solid angle for accepting radiation. Compare the expected performance of these two antennas.

Answer: We will assume that we have a very sensitive receiver and that any noise at the output of the receiver is due to noise coming in through the antenna. Such noise, due to cosmic and terrestrial sources, is present in all parts of the radio spectrum. Consider the beam antenna first. If it were pointed at the satellite it would "capture" a certain amount of power from the desired incoming signal. In addition it would capture some background noise power. But, it would only capture this noise from the limited solid angle over which it accepts power. Now consider the isotropic antenna. It could conceivably capture the same amount of power from the desired signal (if the overall dimensions and efficiencies of both antennas are similar) but, it would capture noise power arriving from all directions. The result is that the signal power to noise power (S/N) ratio presented to the receiver by the beam antenna would be much higher than for the isotropic antenna. Note that the beam antenna only is superior when it is pointed in the proper direction.

Because it is difficult to draw a three-dimensional picture representing the directional properties of an antenna, the usual approach is to draw two cross-sections of the three-dimensional pattern, one in the horizontal plane (horizontal pattern) as shown in Figure 5.3 and one in the plane perpendicular to the horizontal plane (vertical pattern). The vertical cross-section is taken in the direction of maximum horizontal plane gain. The terms horizontal and vertical refer to an antenna mounted in its usual configuration for terrestrial communications. Actually, patterns must always include information on the antenna orientation so as not to be ambiguous. An antenna whose horizontal pattern is a circle is called an omni-directional antenna.

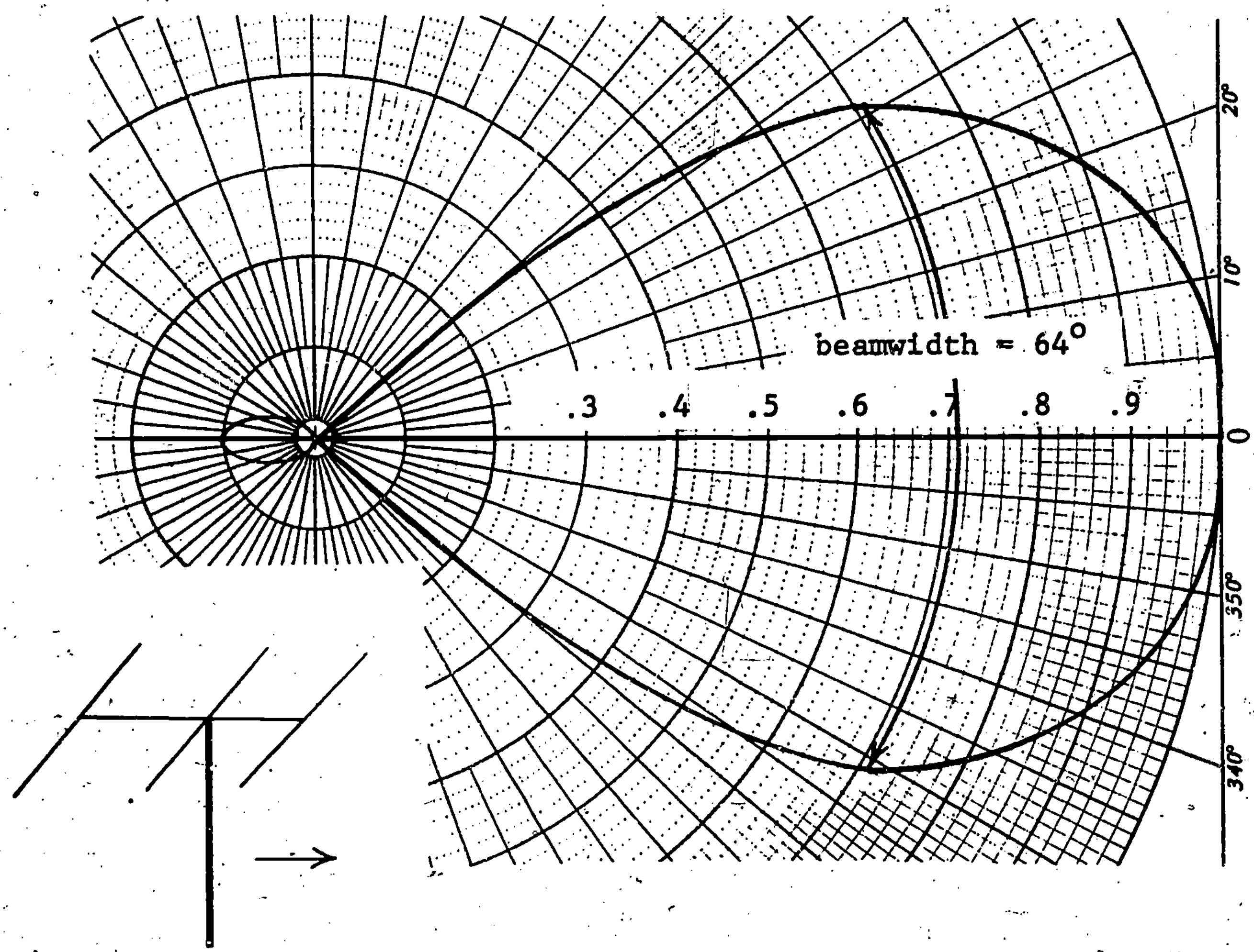


Figure 5.3. Horizontal field intensity pattern for 3-el yagi beam with .1 wavelength director spacing and .15 wavelength reflector spacing. Element length adjusted for maximum gain with antenna mounted 1/2 wavelength above ground. Beamwidth measured at half power (3 dB) points (field intensity = .707 maximum).

Problem 5.2

Specify the horizontal and vertical patterns for an isotropic antenna.

Answer: Every cross-section taken from a three-dimensional representation of the isotropics performance will be a circle.

Beam antennas do not give something for nothing. They just confine their response to certain directions at the expense of others. The directivity of a receiving antenna is a measure of its relative ability to concentrate its response to a specific direction compared to a standard antenna. A qualitative definition of directivity will be given in the next section. The horizontal beamwidth of an antenna is specified by measuring the angle in the horizontal pattern between the points where the field intensity has dropped by 3 dB from its maximum value. The 3 dB (half power) points are shown in Figure 5.3. Vertical beamwidth is defined in a similar manner by referring to the vertical pattern. If a pattern has more than one lobe, we confine our attention to the main lobe. In general, directivity and beamwidth are closely related -- the higher the directivity, the narrower the beamwidth.

Transmitting vs. Receiving Properties. Most applied discussions of antennas emphasize the transmission properties while we have been emphasizing the receiving properties. To relate receiving and transmitting properties of an antenna, consider Figure 5.4 which shows a radio link involving two stations -- A and B. We begin by assuming that A transmits and B receives. Station A has a choice of two antennas: one is a beam, the other an isotropic. Station B has a single antenna, the type is not

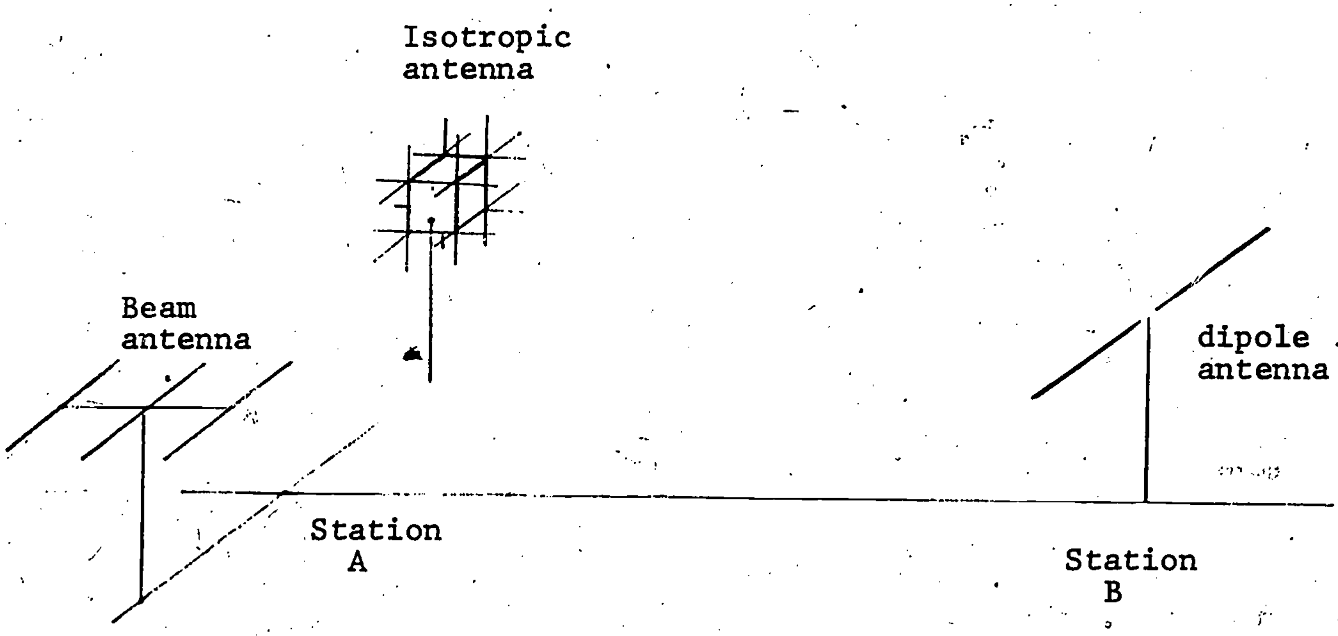


Figure 5.4. Imaginary test setup used (1) to illustrate reciprocity between transmitting and receiving properties of an antenna and (2) to define "gain" and "EIRP".

important. When A uses the isotropic to transmit, the radiated signal spreads out equally in all directions and only a small amount of power reaches B. If A switches to the beam, keeping the transmitted power constant, a much larger amount of power is directed toward B. B will actually extract more signal power from the antenna when A is using the beam. In both cases B will receive the same amount of noise power so the signal to noise (S/N) power ratio seen by B will be considerably higher when A is using the beam.

Now let's consider the reciprocal problem with B using the transmitter and A at the receiving end of the link. The amount of signal power reaching the vicinity of A is fixed. The amount of power an antenna can extract from its environment is roughly dependent on its physical size. If the isotropic and beam antennas are about the same size, each will extract the same amount of power from the desired signal. However, as discussed in Problem 5.1, the beam will only accept noise power from a limited direction so it will provide an improved S/N power ratio. For well-designed antennas, the improvement in the S/N power ratio for the link will be the same whether the antenna changes described in the example are made at the receiving end or at the transmitting end of the link (reciprocity principle).

When an antenna is 100% efficient, the terms "gain" and "directivity" are synonomous. To simplify our discussion, we'll assume 100% efficiency. This is a reasonable assumption for well-designed antennas with elements larger than about 1/4 wavelength. Let's return to the case where A is transmitting with a beam whose characteristics we wish to describe. We can ask the question "What transmitter power (P_i) does A need in conjunction with the isotropic antenna to produce the same signal power at B as produced by a transmitter power (P) and the beam?" The answer is used to quantify the concept of "gain". The gain (G) of a beam antenna with respect to an isotropic reference antenna is given by

$$G = \frac{P_i}{P} \quad (5.1)$$

The quantity P_i, known as the EIRP (effective isotropic radiated power), is a good measure of a transmitting stations capabilities. For example, 100 watts fed to an antenna with a gain of 6 yields the same EIRP (600 watts) as 200 watts fed to an antenna with a gain of 3. Either setup, used at A, will produce the same signal level and S/N ratio at receiving station B. It's oftentimes convenient to express gain as a decibel (dB) ratio:

$$G(\mathrm{dB}) = 10 \log \frac{P_i}{P} \quad (5.2)$$

Although we have specified antenna gain in terms of a transmitting antenna, the reciprocity relations make it a meaningful quantity in terms of receiving antennas.

Problem 5.3.

Phrase a question similar to the one just discussed which will enable us to define gain for a receiving antenna.

Answer: Let station A be on the receiving end of the link. When A is receiving on the isotropic antenna and station B is transmitting using P_i watts a specific S/N power ratio results. What power (P) will station B have to use to produce the same S/N power ratio at A when station A switches to receiving on the beam antenna? Eqs. 5.1 and 5.2 yield the same numeric value for receiving gain as for transmitting gain.

The gain of an antenna is usually specified with respect to an isotropic antenna. The half-wave dipole is also frequently used as a reference antenna. Since the half-wave dipole has a gain of 2.14 dB over an isotropic antenna, the gain of a specific antenna will be 2.14 dB higher when referenced to an isotropic as stated in Eq. 5.3. Gains are sometimes expressed in dB_i or dB_d where the subscript specifies the reference antenna as isotropic or dipole. Obviously, statements about

(5.3) Gain over dipole = Gain over isotropic – 2.14 dB

gain (like those seen in some advertisements) must include the identity of the reference antenna to have any meaning. Up to this point we have been discussing the directional properties of antennas. We now turn to another property -- polarization.

Antenna Polarization. We begin this section by taking a look at some properties of radio waves. One can conceptualize an incoming radio wave at a point in space as consisting of orthogonal electric field (E) and magnetic field (H) vectors which vary with time. The antennas we are interested in respond to the electric field vector so we confine our attention to it. If we picture the tail of the E vector as being tied to a particular point in space the tip will, in the most general case, trace out an ellipse during each cycle of the incoming wave. Such a wave is said to be elliptically polarized. If the magnitude of the E vector remains constant as it rotates, the ellipse degenerates to a circle and we have circular polarization. If the minor axis of the ellipse decreases to zero we have linear polarization. Circular polarization (CP) and linear polarization (LP) are just two special cases of elliptical polarization and, an elliptically polarized wave can be treated as if it consisted of a circularly polarized component and a linearly polarized component along the major axis -- both components having the same frequency and phase.

Like radio waves, antennas can be assigned a polarization -- the polarization of the waves that they transmit or respond to (in direction of maximum gain). In general, an antenna that transmits a linearly polarized wave will respond most effectively to a linearly polarized wave having the same orientation; an antenna that transmits a circularly polarized wave will respond most effectively to a received signal that is circularly polarized and of the same sense -- right hand circular polarization (RHCP) or left hand circular polarization (LHCP). To determine the polarization sense of the antenna, imagine that you are standing behind it watching a wave being transmitted. If the E vector at a specific point rotates clockwise, we have RHCP. To extract the greatest amount of power from an incoming wave, the polarization response of the antenna should be matched to the incoming wave. Polarization mismatch can easily make the difference between strong solid signals and no signals whatsoever!

Let's look into the polarization matching problem further by examining a communications link involving two stations -- station T, the transmitting station, and station R, the receiving station. Each station has a choice of antennas which provide RHCP, LHCP, or LP. The orientation of the LP antenna can be varied by rotating it about the line joining T and R. All antennas are assumed to have the same gain and each is aimed at the other station. Possible link combinations can be characterized by the polarization at T, the polarization at R, and the relative orientation or sense of the antennas used at T and R. For example, (LP, CP, random) can mean either that T has a LP antenna and R a CP antenna or vice-versa and that the orientation of the LP antenna is random. The ambiguity is intentional -- since the reciprocity relation previously discussed states that system performance will be the same in both cases, there is no need to distinguish between them. Only five distinct combinations need be considered:

Type 1 link	(LP, LP, matched)
Type 2 link	(LP, LP, random)
Type 3 link	(LP, CP, random)
Type 4 link	(CP, CP, same sense)
Type 5 link	(CP, CP, opposite sense)

Arbitrarily choosing the Type 1 link as a reference, we examine the other four combinations.

1. (LP, LP, matched). The received signal level is constant. This link is our reference.
2. (LP, LP, random). The received signal strength on this link varies monotonically from a maximum equal to the reference level when the two antennas are aligned parallel to a minimum about 30 dB below the reference level when the two antennas are perpendicular. (30 dB is a realistic number, a simple theoretical model predicts infinite attenuation for the perpendicular situation).
3. (LP, CP, random) The received signal strength on this link will be constant for all orientations of the linear antenna and 3 dB down from the reference level.
4. (CP, CP, same sense). The received signal strength on this link will equal the reference level.
5. (CP, CP, opposite sense). The received signal strength on this link will be constant and about 30 dB down from the reference level.

Having looked at the performance of the five basic links, we return to the main problem -- choosing a ground station antenna to operate in conjunction with a specific type of satellite antenna. If the satellite antenna is linearly polarized, our choice of ground station antenna is equivalent to choosing a Type 1, 2, or 3 link. Of the three, the Type 1 link is preferable since it provides the strongest signals. However, from a practical viewpoint, it is almost impossible to implement since the orientation of the incoming wave is constantly changing. In reality our choice is therefore limited to a Type 2 or Type 3 link. Of these the Type 3 link is preferable from a performance viewpoint. It is true that a small percentage of the time the Type 2 link will provide up to 3 dB stronger signals but half of the time it will produce weaker signals that will often completely fade out.

We can execute a similar analysis for a satellite antenna which is circularly polarized. The choice of ground station antenna here is equivalent to choosing a Type 3, 4, or 5 link. A Type 4 link is preferable. However, the Type 3 link results in signals that are only 3 dB weaker with none of the severe fading problems associated with Type 2 links. A Type 3 link is often chosen because linearly polarized antennas are simpler from a mechanical viewpoint and the 3 dB reduction in received signal strength can be tolerated.

This analysis is overly simplified since the relative antenna orientations involve other degrees of freedom. However, since a CP ground station antenna produces the best results in both cases (LP or CP at the satellite), a CP ground station antenna of the proper sense will also provide the best results in the more general case of receiving elliptically polarized signals from the satellite.

In summary, a circularly polarized ground station antenna will produce the best performance for either circularly or linearly polarized satellite antennas. When the satellite antenna is circularly polarized, the ground station can use linear polarization with only a slight reduction in signal strength. When the satellite antenna is linearly polarized the use of linear polarization by the ground station will result in periodic severe fading due to polarization mismatch but strong signals will be received a small percentage of the time. Many groundstations elect to use linear polarization and accept the tradeoff of performance for mechanical simplicity.

5.4 PRACTICAL GROUND-STATION ANTENNAS

In this section we discuss some of the practical considerations related to choosing: (1) antenna location, (2) between beams and omnidirectional antennas, and (3) between linearly and circularly polarized antennas. It is strongly recommended that the simplest possible antenna system be used initially. For low-altitude satellites, such as AMSAT-OSCAR 7 and 8 and the Russian RS-1, this means fixed omni-directional or broadly directional antennas. The high-altitude AMSAT Phase III series will require moderate gain beam antennas at the ground station. By starting with simple antennas one can evaluate the performance improvements obtained by various changes in the system and weigh the enhanced performance against the added system complexity. Later in this section we describe a number of specific antennas for ground station use.

(1) Antenna Location. Generally, the best location for an antenna is as high as possible and as far from surrounding objects as possible. However, keeping in mind our design objective of producing an adequate S/N ratio at the receiver output while minimizing system complexity, it is often best to compromise on a convenient location. Losses in the feedline can be a serious problem, especially at VHF and UHF frequencies (see section 5.5 -- coaxial feedlines). It is possible to reach a point where improvements due to raising the antenna are offset by feedline losses unless one uses a pre-amp mounted at the antenna. Signals of the 146 MHz AMSAT-OSCAR 7 downlink are so strong that this author has often obtained good results during demonstrations (even in steel and concrete buildings) with a simple ground plane antenna held in one hand while tuning the receiver with the other. However, don't count on being this lucky -- get your antenna on the roof and above nearby objects insofar as possible.

(2) Beams vs. Omni-directional antennas. Although beams can produce better S/N ratios at the ground station, using them entails constructing an antenna system with at least one or two rotators and tracking the satellite during each pass. Beginners should start with either omni-directional antennas or those having a very broad pattern so that rotators are not needed. Note: moderate gain antennas will be needed with Phase III satellites except near perigee.

(3) Polarization. Although we have discussed the advantages of using circular polarization at the ground station, once again it is much simpler to start off using linearly polarized antennas which are simpler to construct. Once some experience has been acquired, one can experiment with circularly polarized antennas if improved performance is required.

Half-wave dipole. (Only suitable for low-altitude (Phase II) satellites). Perhaps the simplest antenna for ground station use is the horizontally mounted half-wave dipole mentioned in conjunction with the simple ground station of section 5.1 and shown in Figure 5.1. The half-wave dipole (4.85 m at 29.5 MHz, 96.5 cm at 146 MHz, 32.3 cm at 435 MHz), fed in the center with coaxial cable, is extremely simple to construct. The horizontal pattern of a half-wave dipole is shown in Figure 5.5. A

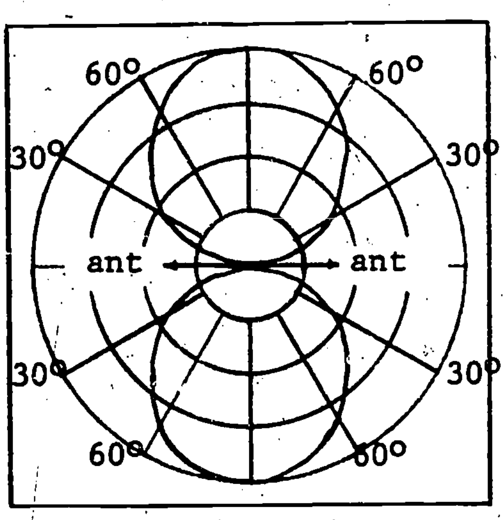


Figure 5.5.

Horizontal pattern of half-wave dipole. The radial scale represents normalized field intensity. Beamwidth = 78°.

horizontally mounted half-wave dipole can be classified as a broadly directional linearly polarized antenna. Dipoles are used at a great many ground stations for reception at 29.5 MHz. If one is mainly interested in the mid-morning (local time) passes of AMSAT-OSCAR 7 and 8 which generally go from northeast to southwest, a single fixed dipole oriented northwest to southeast will provide an appropriate horizontal pattern. It will also work well for the Russian RS-1 satellite. Many stations use two horizontal dipoles mounted at right angles, each with its own feedline. They can then switch from one dipole to the other, choosing the antenna that provides the best signal.

A simple variation of the dipole is the inverted V antenna shown in Figure 5.6. From a construction viewpoint, it is just a dipole with

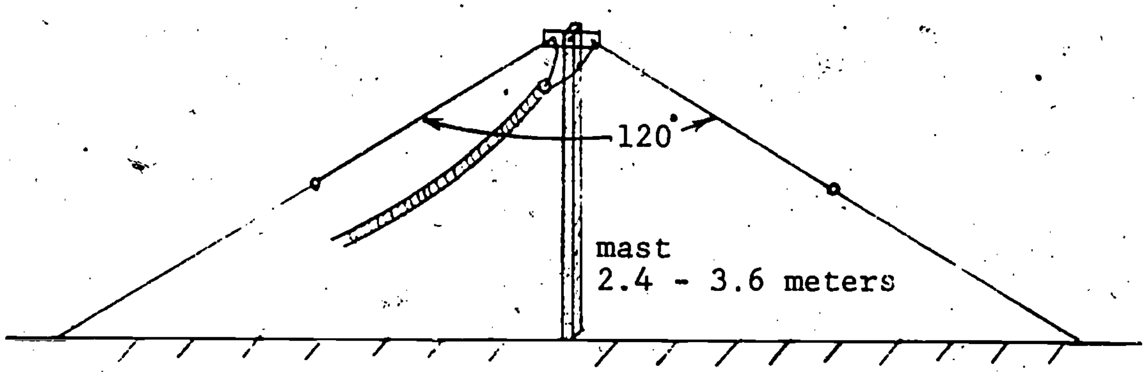


Figure 5.6. 29.5 MHz inverted V antenna. See Figure 5.1 for dimensions. The following coaxial cables may be used (in order of preference): RG-8/U, RG-11/U, RG-58/U, RG-59/U.

drooping ends. The horizontal pattern of an inverted V is shown in Figure 5.7. Because the pattern is relatively omni-directional in the horizontal plane, a single inverted V can provide reasonable coverage in all directions.

So far we have been considering horizontal plane patterns. We now consider the vertical plane pattern of the half-wave dipole. Vertical patterns are strongly influenced by the presence of the ground. Three possible vertical patterns for a half-wave dipole above a perfect ground are shown in Figure 5.8. The effective electrical ground does not generally correspond with the actual surface level so one cannot just use a tape measure to figure out which pattern is appropriate. A pattern like that shown in Figure 5.8c is very undesirable since each time the satellite passes through one of the nulls, the downlink signals will fade. In reality the nulls do not appear as sharp as shown because (1) the ground is not a perfect conductor and (2) satellite signals often arrive at the receiving antenna after being reflected off nearby objects.

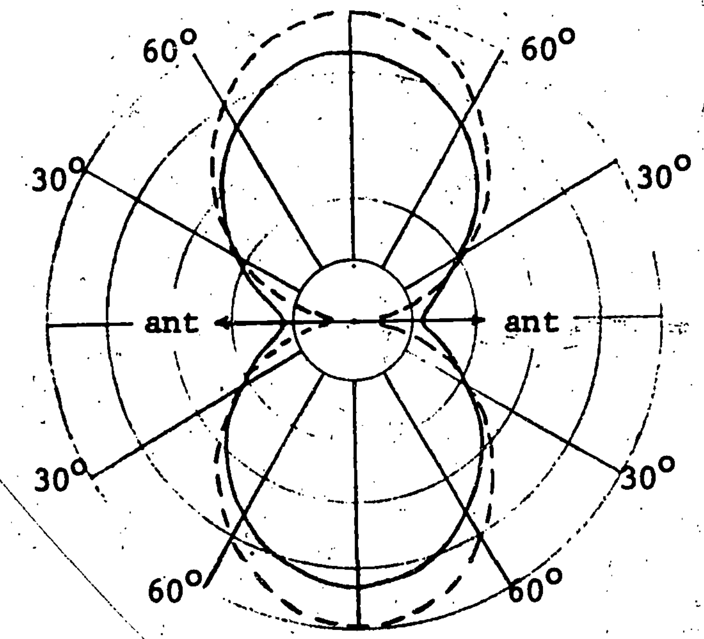


Figure 5.7.

Horizontal pattern of inverted V with 120° apex angle. The radial scale represents relative field intensity. A half-wave dipole (dashed line) is shown for comparison. See reference [2].

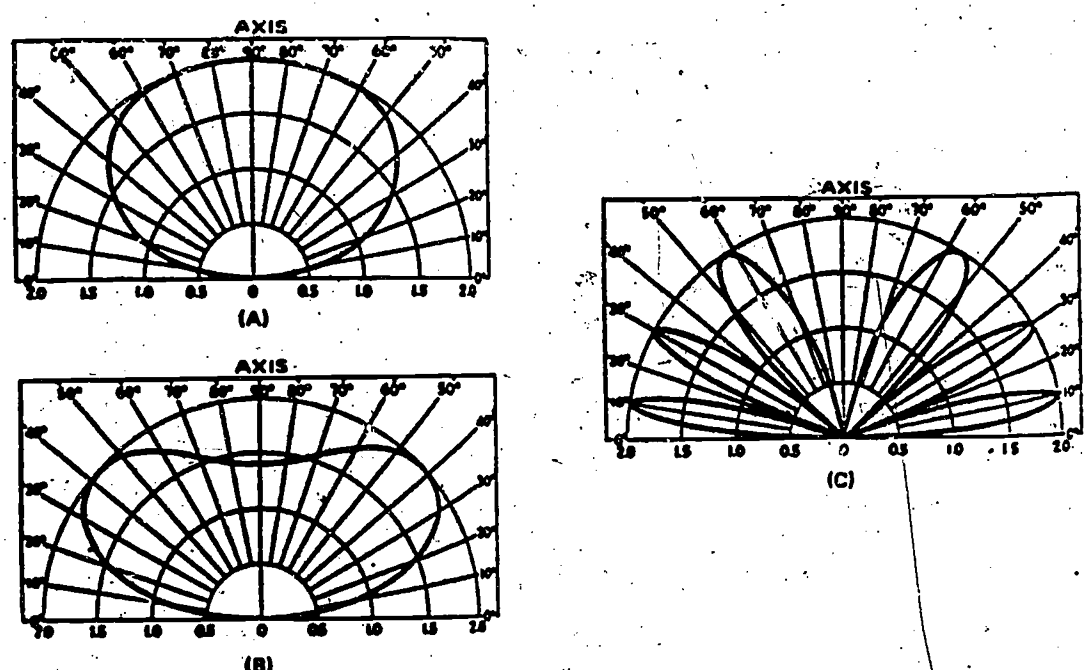


Figure 5.8. Vertical plane (elevation) patterns for half-wave dipole above perfect ground. Pattern at right angles to dipole. Pattern A is for height of 1/4 wavelength, B for 3/8 wavelength, C for 1.5 wavelength.

Many dipole users just orient the antenna with regard to the horizontal pattern and mount it as high and as clear of surrounding objects as possible. Although this does not usually result in optimum system performance, the results are usually adequate. Some users have tried to obtain the desired vertical pattern (Figure 5.8a or 5.8b) by simulating a ground with a grid of wires placed beneath the dipole as shown in Figure 5.9. Subjective

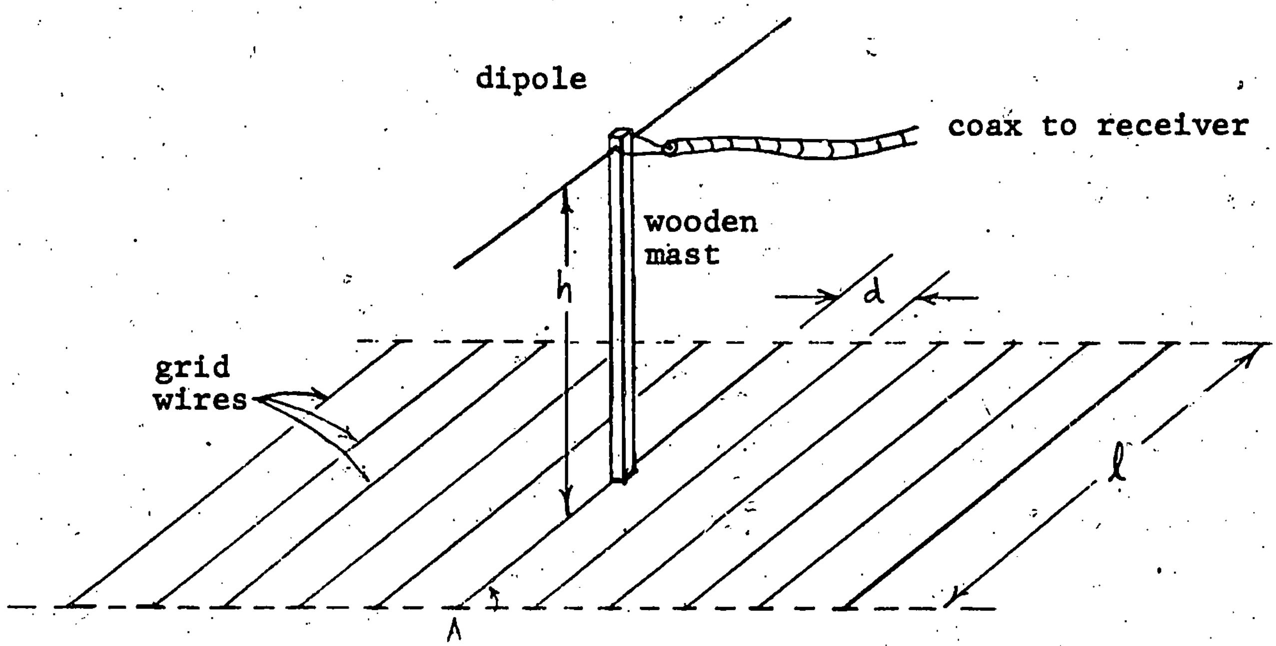
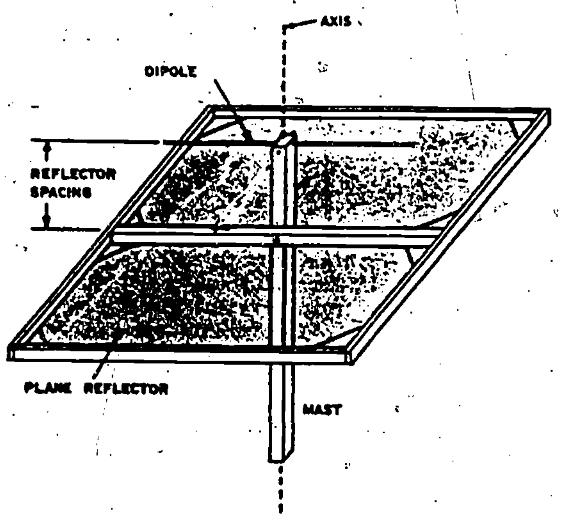


Figure 5.9. Dipole mounted above reflecting screen. Best results are obtained when h = 3/8 wavelength, d is less than .1 wavelength, and ℓ = .6 wavelength. Note that it is not necessary to physically connect the grid wires to the dipole or feedline.

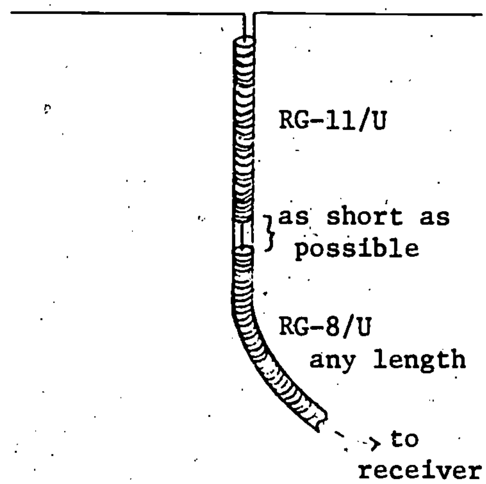
reports suggest that even a single wire (the one labeled A) so placed may improve reception. At 146 MHz the ground can be simulated by a reflecting screen so that the vertical pattern of Figure 5.8b can be obtained with the antenna mounted in a desirable high location. See Figure 5.10 for construction details. The feedline matching system shown in Figure 5.10b may improve reception when using this antenna. Some users have experimented with the inverted V antenna above a simulated ground. Although hard data is not available subjective reports are favorable.

The basic half-wave dipole can also be mounted vertically. When placed this way, it will produce an omni-directional pattern in the horizontal plane. However, performance at high vertical angles may be poor. When used in this configuration the feedline should be routed at right angles to the antenna for at least a half wavelength (five meters at 29.5 MHz).



(a)

The total length of the dipole should be 1/2 wavelength (97.2 cm). It can be made from copper or brass tubing. Hardware cloth on a wooden frame is used for the screen which should be at least .6 wavelengths (122 cm) on a side. A reflector spacing of 3/8 wavelength (72.4 cm) should be used. Feed with RG-11/U or the impedance matching transformer shown in (b).

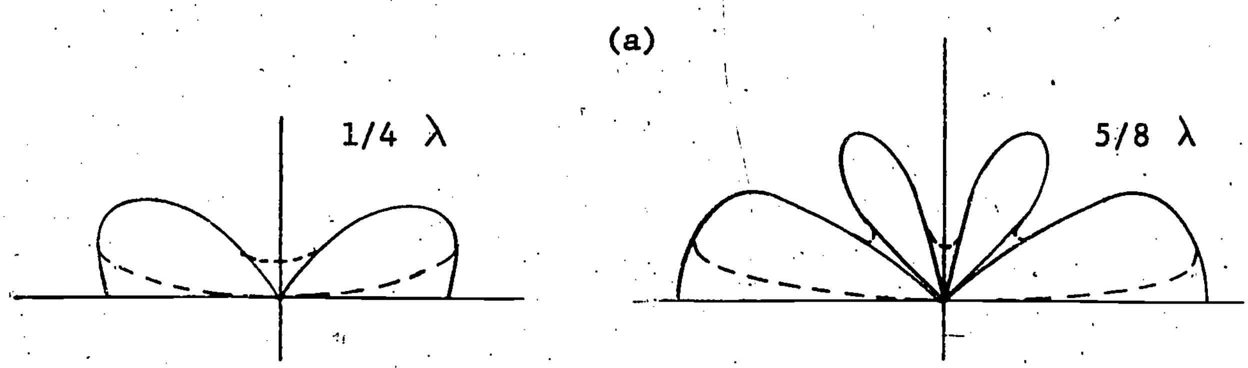


(b)

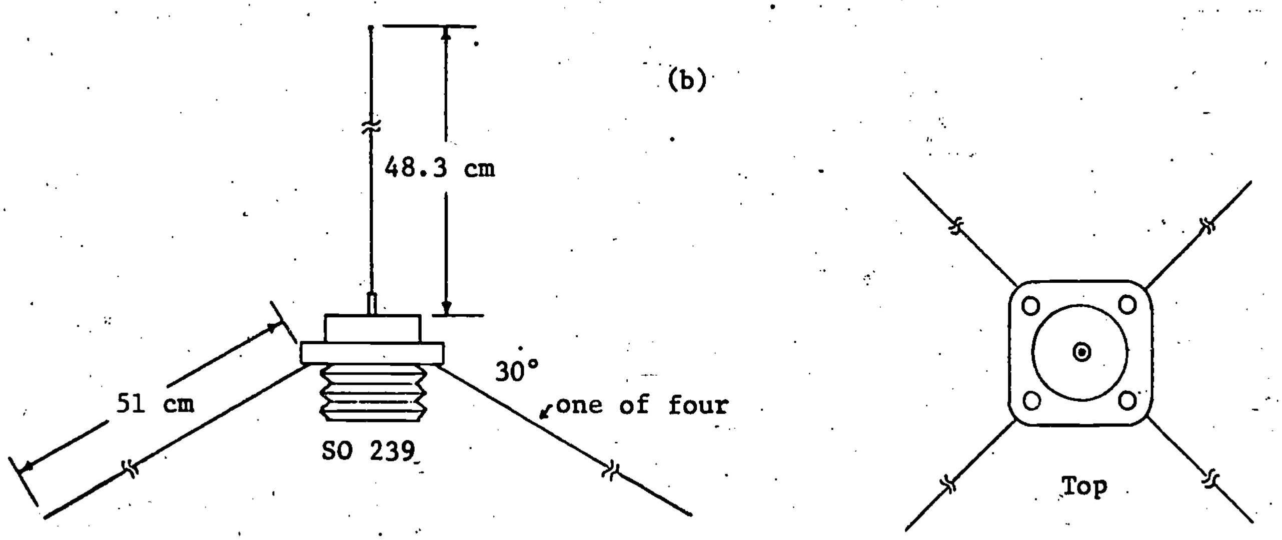
Alternative feed system. The RG-11/U should be an electrical 1/4 wavelength at 146 MHz (31.9 cm for solid dielectric cable; 36.2 cm for foamed dielectric cable).

Figure 5.10. Dipole above reflecting screen (a) and feed system (b) with dimensions for 146 MHz.

<u>Ground plane antenna</u>. (Only suitable for low-altitude (Phase II) satellites). Another simple antenna is the ground plane (GP) which consists of a 1/4 or 5/8 wavelength vertical element and three or four horizontal, or drooping, spokes about 5% longer than the vertical element. One is shown in Figure 5.11b. Metal sheet or screening is sometimes used instead of the horizontal spokes. Dimensions for the vertical element (1/4 wavelength, 5/8 wavelength) are -- 2.4 m, 6.1 m at 29.5 MHz; 48.3 cm, 121 cm at 146 MHz; 16.2 cm, 40.4 cm at 435 MHz. The GP has an omni-directional horizontal plane pattern. The radiation is linearly polarized. Vertical plane patterns for 1/4 wavelength and 5/8 wavelength GP antennas are shown in Figure 5.11a. A number of stations have used GP antennas at 29.5 MHz and 146 MHz with excellent results. Although the vertical plane pattern suggests that GP performance should be poor when the satellite is overhead, this is frequently not the case. The overhead null is often not observed



(a) Vertical field patterns showing relative intensity for 1/4 wavelength and 5/8 wavelength groundplane antennas. Solid lines show pattern over ideal earth (conductivity = 100%). Dashed lines represent more realistic pattern. Ground losses greatly reduce intensity at low elevation angles. Anisotropic ground conductivity tends to reduce depth of nulls.



(b) Construction details for 1/4 wavelength groundplane antenna using SO 239 female UHF coaxial connector.

Figure 5.11. The groundplane (GP) antenna.

because satellite signals reflected off nearby objects arrive at the GP at low angles when the satellite is overhead. If a GP is mounted high and clear of nearby objects, the overhead null may be observed.

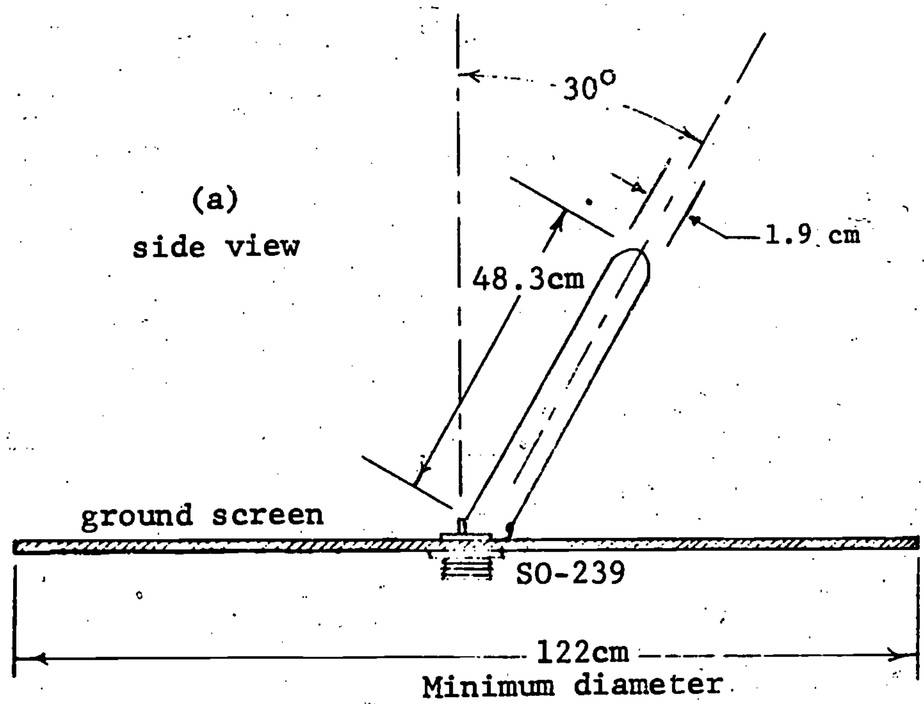
GP antennas designed for 27 MHz citizens band (CB) operation are widely available and inexpensive. They can be modified for use on 29.5 MHz by shortening the vertical element by 9%. CB salespersons will inevitably try to sell the "bigger and better" 5/8 wavelength model but, at 29.5 MHz, better results are usually obtained with the 1/4 wavelength model.

GP antennas designed for 146 MHz amateur operation are available commercially at moderate cost. I've had equally good results using 1/4 and 5/8 wavelength models. A 1/4 wavelength ground plane can be constructed at extremely low cost using an SO 239 UHF female coax connector as shown in Figure 5.11b. You'll also need about 2.6 meters of #12 copper wire -- the type used for house wiring is fine. Solder the vertical element and the four spokes to the coax connector as shown in the figure. To mount the antenna, connect a feedline and tape the feedline to a mast.

Construction details for a variation of the GP using a tilted vertical element are shown in Figure 5.12a [3,4]. When the vertical element of the 1/4 wavelength GP is tilted at 30° to the vertical, the resulting vertical plane pattern is as shown in Figure 5.12b. Notice that the overhead null has been eliminated. The horizontal pattern remains nearly omni-directional. Stations using this antenna at 146 MHz report good results. It should also work at 29.5 MHz (when the dimensions are scaled) but I'm not aware of anyone who has tested it at this frequency.

Beam Antennas. The majority of the experiments described in Chapter VI can be done with simple fixed omni-directional or broad beamwidth antennas of the types just described when working with low-altitude AMSAT and Soviet RS satellites. Beam antennas will be needed with Phase III satellites during most of their orbit and they may be employed when higher S/N ratios are desired with low-altitude spacecraft. Beams are usually used in conjunction with azimuth and elevation (or polar mounted) rotators so that they can be pointed at the moving target -- the satellite. Station operation using beams becomes more complex because one must "ride" the rotator controls during a satellite pass. With Phase III this will much less of a problem since the azimuth and elevation change slowly except near perigee.

There are various types of beam antennas. Three of the most common are shown in Figure 5.13. The yagi and quad produce linearly polarized radiation while the helix produces circularly polarized radiation. Dimensions for beam antennas depend on the operating frequency. A three element yagi (3-el yagi) for 29.5 MHz will be about five times the size of a 146 MHz 3-el yagi but both will have the same gain and patterns. The horizontal pattern for a 3-el yagi was shown in Figure 5.3. As the boom length and number of elements of a well designed yagi increase the gain usually increases and the beamwidth decreases. Yagis and quads for 29.5, 146, and 435 Mhz are available from a number of commercial sources (see



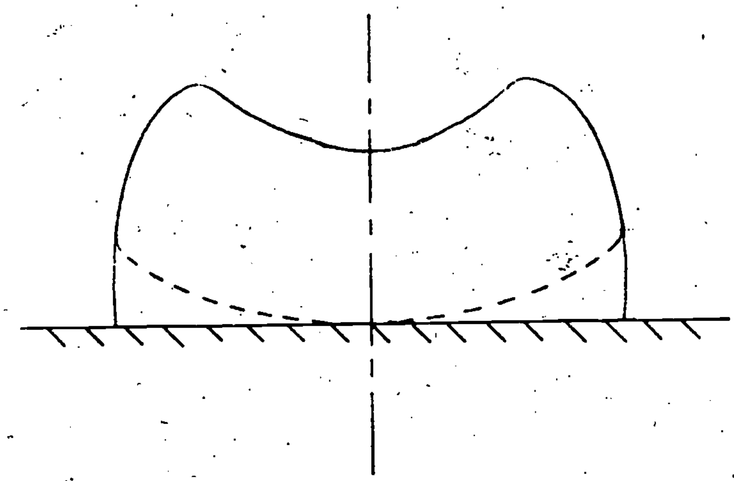


Figure 5.12. (a) Ground plane antenna with tilted vertical element. The vertical element can be heavy copper wire. One end is soldered to the center of the SO-239 coax connector, the other end is soldered to a ground lug at an SO-239 mounting screw. Use RG-8/U or RG-11/U feedline. Groundplane may be square or circular.

(b) Vertical field relative intensity pattern for antenna shown in (a).

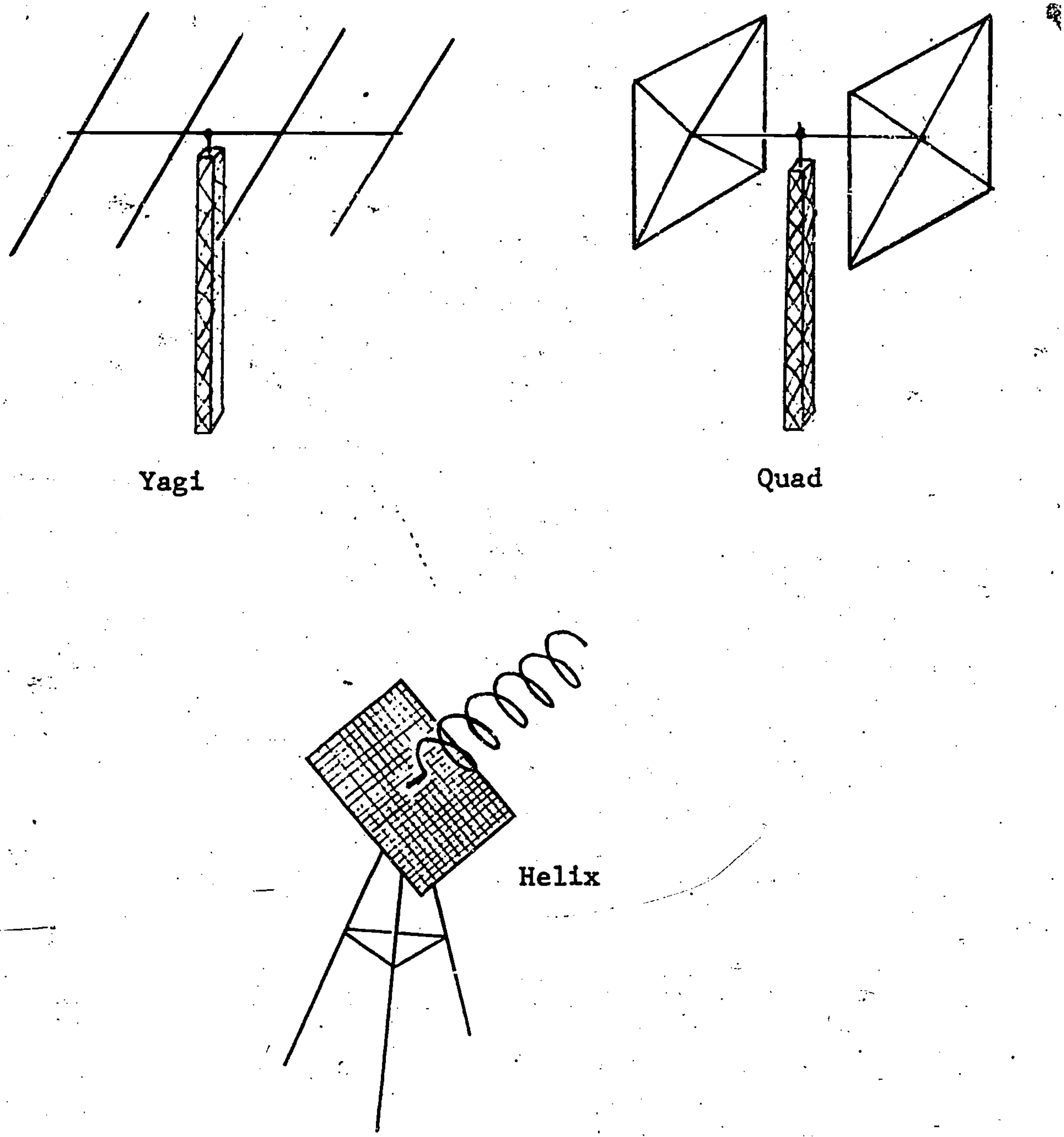


Figure 5.13. Three types of beam antennas: Yagi, Quad, Helix

advertisements in a recent issue of QST or Ham Radio or contact the manufacturers in Table 5.3 for literature).

Antenna Manufacturers
KLM
Cush-Craft
Hy-Gain
Spectrum International
Heath (146 MHz GP)
Radio Shack (CB GP for 29 MHz)

Table 5.3. Manufacturers producing antennas suitable for OSCAR ground station use. Addresses in Appendix A.

Home built yagis and quads for 29.5 MHz usually perform well. See ARRL Antenna Book for plans. However, the dimensions of a yagi antenna are critical and obtaining the desired performance at 146 MHz or 435 MHz usually requires good test equipment. Unless one has had experience in the construction of VHF or UHF antennas, it is best not to attempt to build a yagi. In contrast, the dimensions of the helix antenna are not critical, making construction of one for 146 MHz or 435 MHz practical [5,6]. Another easily duplicated antenna for 146 MHz or 435 MHz is the Quagi which is described in reference [7].

When working with low-altitude satellites it's best to stick to relatively low gain beams -- 3-el to 6-el yagis or 2 to 6 turn helixes -- so tracking will not be overly difficult. When working with Phase III satellites (except near perigee) beams having gains of 10 to 14 dB_i should be used.

Antennas: General. It's often convenient to use more than one receiving antenna at a specific downlink frequency, switching to whichever antenna performs best at a given time. Antennas with complementary characteristics are generally chosen. The antennas can be complementary with respect to (1) azimuth response (for example: two perpendicular horizontal dipoles), elevation response (examples to follow), or (3) polarization response (for example: a horizontal dipole and a GP). An example of elevation angle complementarity would be to use a 1/4 wavelength GP when the satellite elevation angle is below 40° and an inverted V mounted close to the ground for high elevation angles (common combination at 29.5 MHz). Another common pair at 29.5 MHz consists of a yagi mounted horizontally with only azimuth control (used at low elevation angles) and an inverted V mounted close to the ground (used at high elevation angles). At 146 MHz many stations find that a small yagi (5-el) mounted on an azimuth rotator but at a fixed elevation angle of 25° works well for elevation angles up to 60°. At higher elevation angles a fixed antenna with good overhead performance, such as the tilted GP shown in Figure 5.12 is switched to.

We've discussed the advantages which result from using circularly polarized antennas at the ground station. However, you've probably noticed that all of the antennas discussed so far, except for the helix, produce linearly polarized radiation. Fortunately, linear antennas can be combined to produce circular polarization. This is frequently done using dipoles and yagis. Details can be found in the references at the end of this chapter [8,9,10]. If one used two linear antennas for a given link (mounted at right angles with respect to polarization while aimed in the same direction) with separate feedlines it is possible to construct a switching system at the receiver which allows the selection of either of two orthogonal linear polarizations or circular polarization of either sense. Operators at many ground stations have found the performance of linearly polarized antennas to be satisfactory. As a result, they've not gone to the trouble of changing to circular polarization.

5.5 STATION ASSEMBLY: GENERAL

Coaxial Feedline. Coaxial feedline is almost always used to connect the antenna and receiver. One should always try to choose the cable that results in minimum attenuation of the desired signal. The most commonly encountered cables are listed in Table 5.4. To minimize antenna system

Cable	Characteristic Impedance	Loss (dB per 30 m) 29.5 MHz	146 MHz	435 MHz	Diameter
RG-8/U	52 ohms	1.0	2.3	4.2	1.0 cm
RG-58/U	53.5 ohms	1.8	4.2	7.8	.50 cm
RG-11/U	75 ohms	1.2	2.8	5.2	1.0 cm
RG-59/U	73 ohms	2.3	6.0	14.	.61 cm

Table 5.4. Characteristics of common coaxial cables. Losses are for solid dielectric cable. For foamed dielectric cable losses decrease by about 10%. Velocity factor for solid dielectric cable = .66 (approximate). Velocity factor for foamed dielectric cable = .80 (approximate).

losses, the characteristic impedance of the feedline should be matched to the impedance of the antenna and receiver. This is especially true at 146 MHz and 435 MHz. At 29.5 MHz small mismatch losses may not be significant. Most receivers are designed for 50 ohm input impedance, therefore we'll generally stick with 50 ohm systems. When working with antennas which do not have a 50 ohm impedance, we sometimes use short sections of coaxial cable as an impedance transformer between the antenna and the main feedline as in Figure 5.10b.

All feedlines produce some attenuation of the desired signal, even when perfectly matched. These losses vary directly as the line length and increase with frequency. Each of the cables listed in Table 5.4 can be purchased with a foamed dielectric instead of a solid dielectric. The foamed dielectric cable has about 10 percent less loss but, in most systems, the difference will not be observable. RG-58/U is cheaper, thinner and lighter in weight than RG-8/U (both are 50 ohm). However, the attenuation data show that, even at 29.5 MHz, RG-58/U should only be used for runs of less than about 30 meters. As a rough guideline, a station desiring good but not necessarily optimum performance should limit RG-58/U runs to less than 10 meters at 146 MHz, and to less than 3 meters at 435 MHz.

Two types of coaxial connectors are commonly used -- the UHF series and the BNC series. The UHF series can be used with RG-8/U, RG-58/U, RG-11/U and RG-59/U cable. The BNC series can only be used with RG-58/U and RG-59/U cable. Each series contains various male and female connectors and adaptors. The more exotic types of coaxial connectors offer very little improvement at 29.5 MHz, a marginal amount at 146 MHz, and a noticeable increase in performance at 435 MHz. The type N series of connectors should be considered at 435 MHz if optimal results are required.

Ready-made cables of various lengths with UHF connectors are available from distributors servicing the citizens band radio market. It's often cheaper to buy these cables than to purchase the components separately.

<u>Accessories</u>: <u>Basic</u>. Three accessories are discussed in this section: digital frequency readouts, frequency calibrators, and weak signal sources.

Many receivers provide access to the variable frequency oscillator (vfo) used for tuning. If a receiver has a vfo output, a standard frequency counter having an appropriate upper frequency limit can be used to monitor the vfo frequency. Note that the counter reads the vfo frequency, not the frequency of the incoming signal. To calculate the frequency to which the receiver is tuned to, a constant must be added to, or subtracted from, the counter reading. The constant, which will usually be different for each band, may be obtained either by (1) referring to the mixing scheme shown in the receiver instruction manual or by (2) tuning the receiver to a known frequency (using a frequency standard) and subtracting to obtain the difference frequency. For many experiments one is primarily interested in frequency changes and not in true frequency. Frequency changes can be obtained from a counter connected to the vfo without ever calculating the received frequency. Accurate measurements of the true frequency of an incoming signal involve a number of considerations not discussed here which make other techniques more appropriate -- see Project SP 4 in Chapter VI. Some modern receivers provide outputs from all frequency determining oscillators so that they can be used with companion direct frequency readout counters available as accessories. These special counters combine the outputs of all frequency determining oscillators in the proper manner to arrive at the received frequency. Old receivers which do not provide access to the vfo can usually be modified by constructing a source or cathode follower and connecting it to the vfo (see <u>The Radio Amateur's Handbook</u> for details).

A crystal controlled calibration oscillator which provides markers every 100 or 25 kHz is a very useful and inexpensive (about $25) station accessory. Calibration oscillators greatly improve the accuracy of analog (dial) receiver frequency readout. Many of the HF receivers mentioned in section 5.2 come with a built-in calibrator. Calibrators can also be purchased separately. See the advertisements in recent issues of QST and Ham Radio.

A crystal controlled oscillator for spot frequencies close to the 29.5 MHz and 146 MHz beacons aboard the OSCAR satellites is useful for testing the receiving system and optimizing the various component parts. Built-in calibration oscillators often do not serve this purpose because their harmonics at 146 MHz cannot be heard and/or their output is injected directly into the receiver first stage bypassing the antenna, feedline, pre-amp and converter. The 18th harmonic of a crystal cut for 8.108 MHz will provide a useful marker near 145.944 MHz and the 4th harmonic of a crystal cut for 7.369 MHz will provide a useful marker near 29.476 MHz. Inexpensive low tolerance crystals are usually fine since the exact frequencies are not very important. A kit with all parts for an oscillator covering the proper frequency range is available from the International Crystal Manufacturing Company for under $5. Crystals are available from the same source for about $4 -- model EX crystals and oscillator kits.

Accessories: Advanced. In this section we look at some accessories useful to the advanced experimenter. We begin by discussing the equipment needed to decode the radio teletype (RTTY) telemetry from AMSAT-OSCAR 7.

An RTTY demodulator and a teleprinter or video display unit must be added to the regular station setup in order to decode RTTY telemetry (see Figure 5.14). The RTTY demodulator is an electronic device that

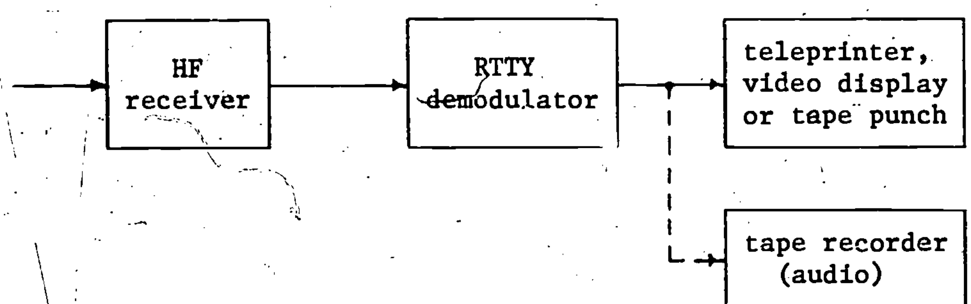


Figure 5.4. Equipment connected to audio output of receiver for decoding of RTTY telemetry.

converts the audio signal from the receiver output into a format useable by a standard teleprinter or video display unit. Advocates of the teleprinter stress the fact that it provides hard copy while the video display unit does not. However, a regular audio tape recorder will function as a memory for the video display unit. The video display unit is quiet and usually very reliable while the teleprinter, with its myriad of moving mechanical parts, makes considerable noise and needs frequent maintenance. Teleprinters are sometimes available at very low cost which could be an overriding factor. Anyone contemplating setting up to receive RTTY telemetry should first read Specialized Communications Techniques for the Radio Amateur, Chapter 5, for general background on RTTY techniques. Most RTTY operation is done using two closely spaced frequencies, f_1 and f_2, referred to as the mark and space. The transmitter operates continuously during a transmission period switching back and forth between mark and space. The OSCAR 7 system differs from normal RTTY in that only the mark is transmitted. This is done to conserve satellite battery power. Since the space information is actually redundant, the techniques works. However, redundancy does improve the accuracy of the received copy. Under most conditions the savings in battery power are considered worth the performance degradation. Minor changes must be made in most RTTY demodulators in order to work with the OSCAR 7 system. Details are contained in reference [11]. Some stations have directly interfaced the receiver audio output with a minicomputer system. The minicomputer can then be programmed to decode the RTTY. When new satellites use other digital codes the user need only reprogram the minicomputer. RTTY demodulators and video display units are available from HAL Communications Corp. (see Appendix A).

Radio Frequency Interference (RFI). One of the most serious and widespread problems facing satellite users is electromagnetic pollution -- the unintentional radio-frequency noise generated by many electrical devices. Much of this noise cannot be filtered out by any fancy noise limiter or blanker on the receiver. One must either eliminate the noise at the source or somehow operate around it. Both approaches will briefly be discussed here. Common noise sources are fluorescent lights or starters which are not operating properly, thermostats associated with furnaces, stoves, and elevators, and brush type motors. When setting up a station in or near a room with fluorescent lights, always check for RFI by switching the lights on and off while monitoring the receiver. If noise is a problem it can sometimes be cured by replacing faulty lamps or starters. Noise sources can often be located by a trial-and-error process: turn the power to suspected devices on and off while monitoring the receiver. Many manufacturers provide free kits designed to reduce RFI caused by their appliances upon request [12]. If RFI is being caused by an appliance not under one's control and requests to the owner to fix the offending unit do not produce results, a formal complaint can be filed with the Federal Communications Commission but it often takes a very long time to get responsive action.

If the noise source cannot be eliminated (or located), there are still a number of strategies for getting around the problem. Perhaps the best advice is -- be flexible, try varying operating times, frequencies, station location, and the antenna system. You may find that certain noise sources are turned off during lunch hour. Or, a noise source that makes reception at 29.5 MHz impossible may not even be discernable at 146 MHz. Sometimes moving the station to an adjacent room, or switching to an ac outlet on a different line may greatly reduce the problem. Moving the location of the antenna system, switching to a beam antenna from an omni-directional array, or running a ground lead from the receiver to a cold water pipe may also produce significant improvements.

Construction Information. Directions for constructing specific electronic devices have generally been omitted from this text. Readers who are experienced in electronic construction can refer to the books and construction articles cited. Those who are not experienced at electronics construction will find it much less frustrating to use commercially available equipment. The ARRL series of books are an excellent source for practical information on the construction of communications equipment. The following books in the series are especially valuable to those building and operating satellite ground stations:

The Radio Amateur's Handbook,
The ARRL Antenna Book,
The Radio Amateur's VHF Manual,
Specialized Communications Techniques for the Radio Amateur,
Understanding Amateur Radio.

Seeking Assistance. You may find that some of your students or other teachers have amateur radio licenses. Most licensed amateurs have had practical experience with the HF and VHF communications techniques and equipment we have been discussing. They can be an excellent resource for obtaining answers to specific questions and, oftentimes, will enjoy helping to set up a ground station. If there is an amateur radio club in the area, phone one of the officers and ask for the names of some members who are active OSCAR users. Most amateurs enjoy showing off their stations and a call expressing interest in OSCAR communications will likely result in an invitation for a demonstration during an OSCAR pass. Over 5,000 American amateurs have participated in two-way communications via the OSCAR satellites and many more have listened to the downlinks so it is usually possible to find someone experience in OSCAR communications locally. The OSCAR Educational Programs Office at the ARRL maintains a list of radio amateurs who have volunteered to assist in educational activities. Contact the office to find out if there is a volunteer in your area.

Portable Ground Station. A special portable ground station designed for educational demonstrations has been assembled with funds provided by the National Science Foundation. It is being used for an ongoing program of demonstrations at science education conferences. The portable ground station has provisions for receiving downlink signals at 29.5 MHz, 146 MHz and 435 MHz. It is also capable of tranmitting to the satellite (voice or Morse code) at 435 MHz. The total weight of the ground station, including all accessories, is under 20 kg and it operates off 115 v ac or 12 v dc. A list of station components follows:

Kenwood R599A receiver
Kenwood 146 MHz converter
Janel 146 MHz pre-amp
KLM ECHO 70 SSB Transceiver (for 432 and 435 MHz)
Heathkit HWA 202-1 power supply (provides 12 v dc for ECHO 70)
29.5 MHz antenna: wire dipole
146 MHz antenna: 1/4 wavelength groundplane
435 MHz antenna: 7-el KLM yagi mounted on camera tripod
Miscellaneous: coaxial cables, speaker, compass, orbit calculator

5.6 TRANSMITTING CONSIDERATIONS

All radio transmissions are regulated by national governments in compliance with international treaties. In the United States, the rules and regulations governing the transmission of radio signals are implemented by the Federal Communications Commission (FCC), in Canada by the Department of Communications (DOC). In order to transmit uplink signals for relay by, or control of, an OSCAR satellite, one must be licensed by his/her government. The uplink frequencies are assigned to the amateur radio service so the appropriate license is usually an amateur one.

In the United States the FCC rules and regulations state that the ground station transmitter must be under direct control of a licensee having an appropriate grade of license (technician class for working with AMSAT-OSCAR transponders). The key word is "control". If an instructor with a technician or higher grade license is present in a laboratory to see that transmitting equipment is being operated in compliance with FCC regulations, an unlicensed student may operate a transmitter. This makes it possible for anyone, regardless of whether or not they have a license, to perform sophisticated experiments which require transmitting: precise ranging and tracking, observations of compound Doppler shifts, testing of VHF and UHF transmitters, etc.

This text emphasizes educational applications that only require receiving capabilities at the ground station. This is done because a receive only station (1) is simpler to assemble than a station including transmitting capabilities, (2) requires no license, and (3) will permit the user to perform a great many experiments. However, some educators may decide to include transmitting facilities at their ground station. It is strongly suggested that a receive only station be assembled and experience gained in its use before a decision is made on whether or not to add transmitting capabilities. The remainder of this section will briefly introduce practical considerations related to the assembly of uplink facilities. It is assumed that the reader has the technical background needed for a Technician Class amateur license or a Second Class commercial license.

Anyone planning to use the transponders aboard the AMSAT-OSCAR series of satellites should be aware that all users share the available satellite power. Cooperation among users is therefore essential. Stations employing too high an effective radiated power (EIRP) will use more than their share of power and may even activate the automatic gain circuitry aboard the satellite making it impossible for low power stations to use the transponder. For general communications, CW and SSB are recommended. SSB has a high peak to average power characteristic and the transmitting duty cycle for CW is usually considerably less than 50 percent so both these modes use the available satellite power effectively and efficiently. Users are discouraged from using FM, SSTV (slow scan TV), AM or SSB with speech processing for general communications because the high transmitting duty cycle of these modes uses an excessive amount of the available satellite power. However, these modes may be used for special experimental purposes.

Suggested EIRP levels are discussed in conjunction with each satellite. These levels will provide moderately reliable communications with little chance of overloading the satellite except when it is almost directly overhead. Higher EIRP levels will increase communications reliability and use of such levels is legal as long as downlink signals are constantly monitored to insure that the transponder is not being overloaded and that one is not monopolizing the available power. The FCC has announced that stations intentionally overloading the satellite will be considered in violation of Sections 97.67(b) and 97.125 of the Rules and Regulations. Such violation could result in license suspension and fine. Stations using EIRP levels higher than those recommended must therefore provide a means for decreasing their signal level at the satellite. Two common methods for accomplishing this are: (1) use of a linear amplifier which may be switched in or out, (2) deliberate misaiming of a beam antenna. To minimize interference between SSB, Morse code, and other modes the bandplan shown in Figure 5.15 is suggested for all AMSAT transponder downlinks. We now look at the transmitting equipment employed at many OSCAR ground terminals.

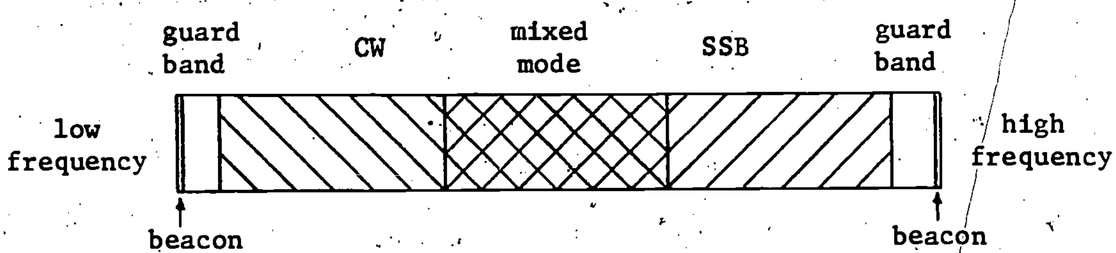


Figure 5.15. Band plan adopted for all AMSAT transponder downlinks. The guard bands are each 5% of the total bandwidth. The CW, mixed mode, and SSB segments are each 30%.

<u>146 MHz Uplink</u>. Most AMSAT-OSCAR and Russian RS satellites have transponders with input frequencies in the vicinity of 146 MHz and output frequencies near 29.5 MHz. Specific frequencies are listed in Chapter IV. A ground station EIRP of about 100 watts is generally required to access the 146 MHz transponders currently in orbit. Specific values for each satellite are given in Chapter IV. Phase III satellites may also have a transponder with an input at 146 MHz but they'll require an EIRP of about 500 watts (50 watts plus 10 dB_i antenna gain).

Problem 5.4.

Assume that a 146 MHz CW transmitter having an input of 100 watts and a 3 element yagi are available for a ground station and that 30 meters of RG-8/U will be used as feedline. What will the effective radiated power of this system be?

Answer: Lacking more information, it is reasonable to assume that the 146 MHz transmitter final amplifier will have an efficiency of about 50%. The loss in the coaxial cable will be about 2.4 dB (see section 5.4). A well designed 3-el yagi has a gain of about 8 dB_i. The EIRP is therefore roughly 200 watts.

The use of a high rf power level and a low gain broad pattern antenna to achieve the recommended EIRP is very desirable for transmitting to low-altitude satellites since it minimizes the need for antenna aiming. A transmitter power of at least 30 watts should be used at the ground station. Approaches used to produce the desired rf power include the following:

1. Purchase of Amateur Service 146 MHz CW and SSB Transmitter. Amateur equipment designed for SSB and/or CW operation on 146 MHz is available. Transceivers for 146 MHz have recently been placed on the market by KLM, Yaesu, Kenwood and ICOM. Many of these units only produce about 10 watts output but accessory amplifiers putting out up to 140 watts are available. When buying a SSB transmitter, make sure that it can operate upper and lower sideband. A number of other manufacturers have marketed items in the past which have been discontinued. These items can often be obtained on the used equipment market at very reasonable prices. One widely used CW transmitter in this category is the AMECO TX-62. If an HF SSB/CW transmitter is available consider purchasing a transverter (a transmitting converter). Transverters are available with inputs at 14, 21, 28, or 50 MHz and outputs at 146 or 435 MHz. Be sure to check the advertisements in recent issue of QST and Ham Radio since new pieces of equipment are likely to be introduced in the near future.

2. Conversion of Commercial VHF FM Transmitter Strip. Transmitter strips from commercial VHF FM equipment designed for 130-160 MHz can usually be converted to 146 MHz CW service. Conversion involves constructing a power supply, purchasing a crystal, retuning resonant circuits to the desired frequency and adding provisions for keying -- grid-block keying of the driver and final is usually employed. Transmitter strips rated at 30 or 60 watts output in commercial service can safely provide 45 or 90 watts CW. The strips themselves can be obtained inexpensively from a number of sources -- see advertisements in a recent issue of Ham Radio.

3. Modifying Amateur 146 MHz FM Transceiver. Most amateur transceivers designed for FM operation between 146 and 148 MHz can be modified for CW operation on the OSCAR uplink frequencies. Modification may be as simple as plugging in an appropriate crystal, disconnecting the mike or setting the deviation to zero, and keying the push-to-talk switch. It is usually desirable to modify the push-to-talk circuitry so that the unit can be left in transmit and only the driver and final keyed.

4. Construction of Transmitter or Transverter. A transmitter or transverter may be constructed. Suitable plans are contained in the ARRL VHF manual. Collecting components, building, and debugging usually involve a great deal of time and the total expense, when starting from scratch, is comparable to other approaches. This approach is only recommended when the educational aspects of the actual construction are of primary interest.

<u>432/435 MHz Uplink</u>. A 432 MHz input is included on AMSAT-OSCAR 7 and 435 MHz inputs are planned for Phase III. AMSAT recommends that a maximum EIRP of 80 watts be used to access the mode B transponder aboard OSCAR 7. Phase III will probably require an EIRP of about 500 watts (50 watts plus 10 dB_i antenna). Transmitting equipment for 432 or 435 MHz is more difficult to obtain than for 146 MHz. A great many users have had good success with 10 to 20 watts of rf and a moderate gain antenna (8-10 dB_i) when working with AMSAT-OSCAR 7. These figures do not contradict the 80 watt maximum EIRP since feedline losses are not included. At this frequency a well designed 6-el yagi can give about 10 dB_i gain with dimensions of only 60 cm for the boom and 35 cm for the largest element. A large percentage of mode B CW operation is done using converted commercial UHF FM transmitting strips designed for either 420 MHz or 460 MHz. Strips rated at 15 watts commercial FM service safely provide more than 20 watts CW when used with an appropriate power supply. Comments regarding the conversion of VHF strips to 146 MHz also apply to UHF strips. A few manufacturers are producing 430 MHz transmitting equipment suitable for OSCAR use. Notable are: <u>KLM</u> (ECHO-70 10 watt CW/SSB transceiver, antennas, power amplifiers), <u>Amateur Radio Components Service</u> (transverter, high power amplifiers), <u>VHF Engineering</u> (CW exciters, amplifiers), <u>Spectrum International</u> (varactor triplers), <u>Texas RF Distributors</u> (transverters by Microwave Modules), <u>Hamtronics</u> (CW exciters, amplifiers). Addresses for these manufacturers are contained in Appendix A. It is suggested that recent issues of <u>QST</u> and <u>Ham Radio</u> be consulted for up-to-date equipment availability. At 435 MHz every effort should be made to use the shortest length of feedline and the highest quality coaxial cable for feeding the antenna. Additional information on transmitting equipment is contained in <u>The Radio Amateur's VHF Manual</u> and in reference [13].

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Chapter V

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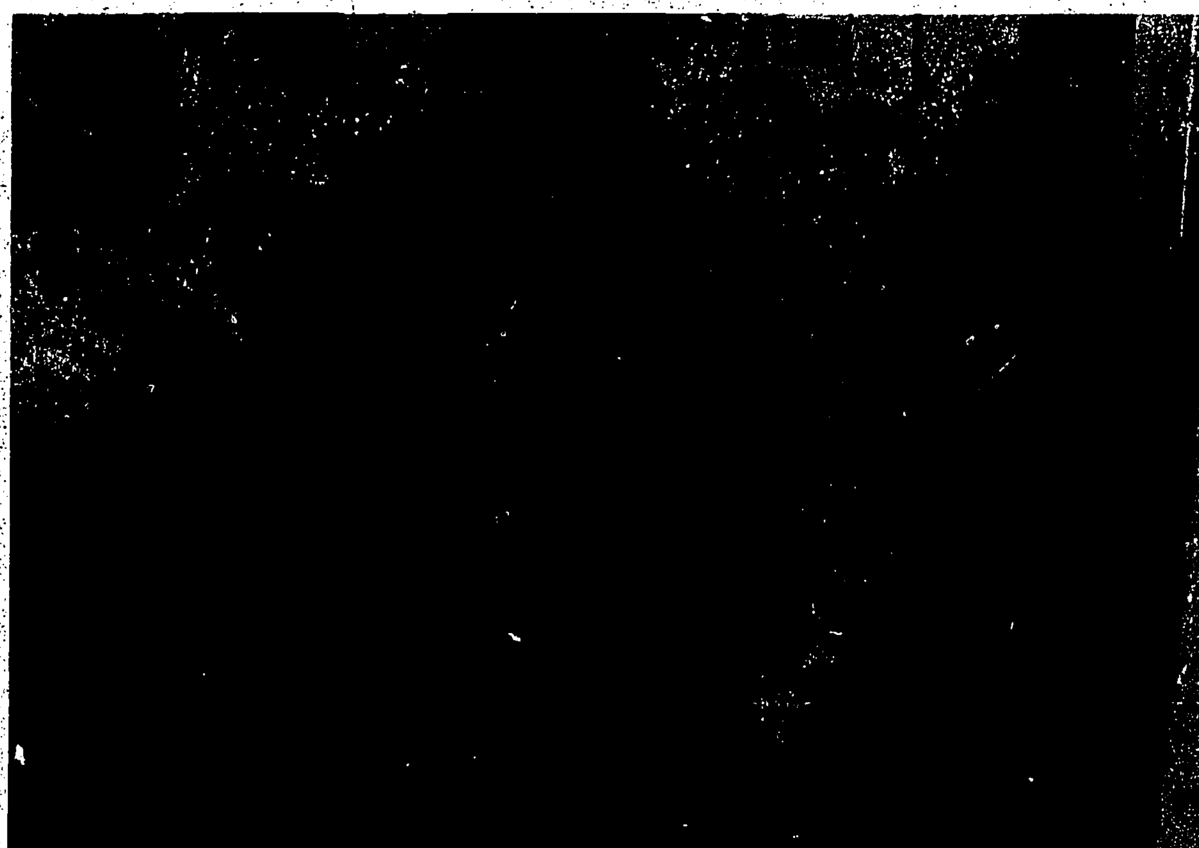


Students in the Electronics Engineering Technology B. S. program at Trenton State College (N. J.) are shown here operating a satellite ground station. They also have designed and constructed prototype flight model subsystems for AMSAT spacecraft.

CHAPTER VI

EDUCATIONAL EXPERIMENTS AND ACTIVITIES

The objective of this chapter is:

to describe several satellite experiments and activities suitable for regularly scheduled student laboratories, student project laboratories, demonstrations, and original research projects.

This chapter contains four satellite experiments (SEs), fifteen satellite projects (SPs) and nineteen satellite miscellaneous activities and topics (SMs). The SEs are in the form of sample student handouts which were used as part of regularly scheduled laboratories. The SPs consist of detailed outlines which can easily be turned into formal experiments. The SMs briefly introduce ideas which can be developed into projects or experiments by interested readers. The SEs and SPs are grouped by general topic into seven sections:

1. Developing Tracking Data,
2. Derivation of Tracking Equations,
3. Doppler Effect,
4. Telemetry Reduction,
5. Propagation,
6. Satellite System Design,
7. Satellite Ranging.

SMs are presented in section 8. All references [n] are at the end of the chapter.

6.1 DEVELOPING TRACKING DATA

This section includes experiments which involve determining one or more of the orbital parameters of a satellite. Tracking experiments can be employed to:

1. directly measure the period of a satellite;
2. introduce satellite terminology;
3. illustrate Kepler's laws;
4. predict when future satellite passes will be within range;
5. determine all orbital parameters of a given satellite.

Tracking experiments are among the most widely applicable and interesting activities involving the OSCAR satellites.

We begin this chapter by presenting a simple experiment, using AMSAT-OSCAR 7, performed by students in the electronics technology program at Catonsville Community College. With minor modifications the experiment is suitable for introductory physics laboratories for engineering students. Typical results are included. The experiment requires a single laboratory session of at least three hours. Although the total amount of laboratory time for this experiment is greater than usual, a good deal of the time is spent waiting for signals to appear and this waiting time can be profitably used.

Tracking experiments involving Doppler shift measurements (section 6.3) and Ranging (section 6.7) will be presented later in this chapter.

EXPERIMENT SE 1
Electronics Technology Department
Catonsville Community College

SATELLITE TRACKING

OBJECTIVES

The objectives of this experiment include:
1. learning basic satellite-orbit terminology;
2. determining the period of AMSAT-OSCAR 7 (circular orbit);
3. predicting times during which the satellite will be in range;
4. determining the altitude of the satellite.

EQUIPMENT

1. Basic satellite groundstation consisting of:
   HF Receiver
   Converter for 146 MHz (if needed)
   Groundplane antenna (29 MHz or 146 MHz as needed)
   Clock
   Weak signal source
2. Textbook: Using Satellites in the Classroom: A Guide for Science Educators, M. Davidoff.

NOTE

In order to assure that signals will be received, this experiment should only be scheduled during either morning laboratories (between 6 a.m. and noon local standard time) or evening laboratories (between 5 p.m. and 11 p.m. local standard time).

PROCEDURE

1. Set the clock using time signals from the National Bureau of Standards (WWV) at 5.000, 10.000, or 15.000 MHz. The Canadian National Observatory (CHU) at 7.335 MHz may also be used.

2. Set the receiving system to the correct frequency. The satellite beacon transmits on 29.502 MHz when it is in Mode A and on 145.972 MHz when it is in Mode B. Operating schedule information is contained in Chapter IV. Adjust the receiver for maximum sensitivity (rf gain full clockwise, AVC off) and widest selectivity. If available, use the weak signal source to check the receiving system. The beacon frequencies specified are approximate. At the beginning of a pass the frequency will be slightly higher (about .5 kHz at 29 MHz, about 3 kHz at 146 MHz) while at the end of the pass they will be slightly lower. The reason for this phenomena, known as Doppler shift, will be discussed in the next experiment.

3. Once the equipment is set up and checked you can sit back and relax. When satellite signals are heard fill in Table I. If you're not sure of

what to listen for ask the instructor for the cassette tape of AMSAT-OSCAR 7 signals. If nothing is heard in two hours check with the instructor -- something is wrong! Data from two consecutive orbits, called A and B, will be needed.

TABLE 1	
Date	28 MAY 1976
Frequency	29.500 MHZ
Orbit A: AOS time	08:08 EDT
Orbit A: LOS time	08:22 EDT
Orbit B: AOS time	09:59 EDT
Orbit B: LOS time	10:19 EDT

AOS and LOS stand for acquisition of signal and loss of signal. Another abbreviation we'll be using is TCA which stands for time of closest approach.

4. Using Table 1 estimate TCA for each orbit using the following approximation.

$$TCA = \frac{AOS + LOS}{2} = \text{time midway between AOS and LOS}$$

Orbit A:	TCA =	08:15 EDT
Orbit B:	TCA =	10:19 EDT

The elapsed time between two consecutive TCA's will serve as an estimate of the period (T).

Period =	114 MINUTES

5. Using the estimated period calculate the altitude (h) of the satellite using the equation

$$T^2 = \left(\frac{4\pi^2}{GM}\right) r^3 \; ; \; r = (.3315) \times 10^6 \times T^{2/3}$$

where T is measured in minutes, r is measured in meters and $r = R_e + h$ (R_e is radius of earth $= 6.37 \times 10^6$m).

h =	1.42×10^6M

$$r = (.3315) \times 10^6 \times (114)^{2/3} = 7.79 \times 10^6 M$$

$$h = (7.79 - 6.37) \times 10^6 M = 1.42 \times 10^6 M$$

6. Using your estimate of the period and your TCA data predict when the next pass will occur. If this pass occurs before noon local standard time (morning laboratory) or 11 p.m. local standard time (evening laboratory) listen for it.

Orbit C: predicted TCA 12:03 EDT

Orbit C: AOS 11:53 EDT

Orbit C: LOS 12:10 EDT

Orbit C: TCA estimated from AOS and LOS 12:01½

New estimate for period using data from Orbits A and C:

T = 113.3 MINUTES

Use TCA data from orbits A and C to compute a new estimate for the period. Show work below.

$$\frac{\text{TCA (ORBIT C)} - \text{TCA (ORBIT A)}}{\text{2 ORBITS}} = \frac{226\tfrac{1}{2}}{2} = 113.3 \text{ MINUTES}$$

7. Using your most recent estimate of the period and the TCA data which you have accumulated, predict whether a pass will occur tomorrow having a TCA between that of orbit A and B. If possible listen, or arrange for another pair of students to listen, for the satellite starting at least a half period before the predicted time. Show all computations and discuss results.

TCA predicted: date 29 MAY time 08:48 frequency 145.973 MHZ

STARTING WITH ORBIT A

113.3 MIN. X 13 ORBITS = 1473 MIN. = 1 DAY + 33 MIN.

PREDICTED TCA = ORBIT A TCA PLUS 33 MIN.

AOS 8:59
LOS 9:18
TCA 9:08½

DATA WHICH WOULD BE OBTAINED IF MEASUREMENTS WERE MADE

New period estimate 114.9 MIN. (1493.5 MIN./13 ORBITS)

Comments on SE 1

The experiment just described works well with AMSAT-OSCAR 7. Period estimates using data from one or two complete orbits are generally within one to two percent of the actual value, lying on the low side. This accuracy is sufficient to make predictions for the following day which, in turn, will generally yield period estimates having an accuracy of better than 0.1 percent. To simplify the experiment a number of assumptions were made: (1) the orbit is circular, (2) the period is approximately equal to the time between successive TCAs and (3) TCA occurs midway between AOS and LOS. These approximations work well with OSCAR 7 but they may not work with other satellites. This experiment should work well with AMSAT-OSCAR 8 and with Soviet RS-1 and ground stations with only 29.5 MHz receive capabilities should adapt the experiment to one of these spacecraft. OSCAR 8 should be available every weekday morning on 29.5 MHz and RS-1 may provide some afternoon passes so it may be possible to relax some of the time restrictions and careful attention to schedules discussed in the experiment. See Chapter IV for beacon frequencies and operating schedules.

A number of additions or variations to this experiment suggest themselves. For example, a directive antenna could be used to obtain additional information which could enable one to roughly gauge the orbit inclination. To confirm the circular nature of the orbit consider assumption (3). For elliptical orbits assumption (3) is poor. As a result, period measurements would vary greatly. The small variation in period measurements therefore suggests a nearly circular orbit.

6.2 DERIVATION OF TRACKING EQUATIONS

A number of the basic tracking equations presented in Chapter I are derived in this section:

SP 1. Eqs. 1.11 and 1.12 -- Satellite elevation angle and slant range;

SP 2. Eq. 1.15 -- Satellite position in orbital plane as a function of time (Kepler's equation);

SP 3. Eqs 1.6 and 1.7 -- Satellite ground track.

These derivations have been found useful as illustrations of practical applications in calculus and in programming and numerical analysis courses.

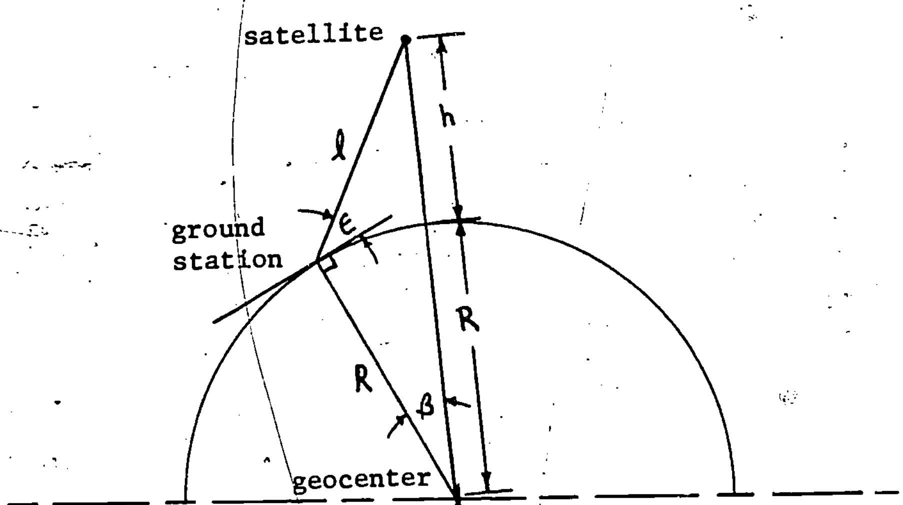
PROJECT SP 1

ELEVATION ANGLE AND SLANT RANGE

Objective: To illustrate how the equations for satellite elevation angle and slant range are derived. The derivation only involves two-dimensional trigonometry, the law of sines and the law of cosines so it is suitable for pre-college instruction.

Derivation. The instantaneous elevation angle (ϵ) of a satellite can be obtained if (1) the instantaneous height (h) of the satellite above the surface of the earth and (2) the surface distance (s) between the sub-satellite point and one's ground station are known. For the calculation we assume that s and h are known. For circular orbits h is, of course, constant. A technique for obtaining h for elliptical orbits will be covered in Project SP 2 -- Satellite motion in the orbital plane. Determining the surface distance between two points on the surface of the earth, Eq. 1.9, is a standard navigation problem and will not be covered here. In the course of determining the elevation angle the slant range (line-of-sight distance between satellite and ground station) will also be found.

Deriving the elevation angle of a satellite appears to be a simple problem in plane trigonometry but the apparent simplicity is deceptive. Try assigning the problem to your students -- provide Figure 1 to standardize notation and specify that you want them to obtain ϵ as a function of s and h -- and see how many can solve it.



The solution is obtained as follows:

(1) $(R+h)^2 = \ell^2 + R^2 - 2R\ell\cos(\epsilon+90°)$ (law of cosines)

(2) $\dfrac{R+h}{\sin(\epsilon+90°)} = \dfrac{\ell}{\sin\beta}$; $\ell = (\sin\beta)(R+h)/\cos\epsilon$ (law of sines)

(3) $\ell^2 = R^2 + (R+h)^2 - 2R(R+h)\cos\beta$ (law of cosines)

Substitute (2) and (3) in (1) and switch to radian measure with $s = R\beta$.

(4) $\tan\epsilon = \dfrac{(R+h)\cos(s/R) - R}{(R+h)\sin(s/R)}$

Step (3) yields the slant range; step (4) yields the elevation angle.

PROJECT SP 2

MOTION IN THE ORBITAL PLANE

Objective: To illustrate how the equation governing satellite motion in the orbital plane (Eq. 1.15) as a function of time (Kepler's equation) is derived. This example is useful in the latter part of an introductory Calculus course or one in numerical analysis.

Derivation. Refer to text Figure 1.1 (Chapter I). Assume that the satellite is moving counterclockwise and that our time measurements start from zero at perigee. From Kepler's III Law (Equal areas (ΔA) swept out in equal time intervals (Δt)) we obtain

(1) $\Delta A = k\Delta t$

The proportionality constant, k, is found by considering one complete orbit:

$\Delta A = \pi ab$ = area of ellipse

$\Delta t = T$ = period of satellite

(2) $k = \pi ab/T$

The area element in polar coordinates is given by $dA = (1/2)[r(\theta)]^2 d\theta$. For an ellipse $r(\theta) = a(1-e^2)/[1+e\cos(\theta)]$. Integrating we obtain

(3) $$\int_0^\theta dA = (1/2)\int_0^\theta r^2(\theta)d\theta = \left(a^2(1-e)^2/2\right)\int_0^\theta d\theta[1+e\cos(\theta)]^2$$

which can be evaluated using most tables -- for example Pierces no.'s 317 and 319.

(4) $$\int_0^\theta dA = \frac{a^2(1-e)^2}{2(1-e^2)}\left\{\frac{-e\sin(\theta)}{1+e\cos(\theta)} + \frac{2}{(1-e^2)^{.5}}\arctan\left[\frac{(1-e^2)^{.5}\tan(\theta/2)}{1+e}\right]\right\}$$

Introducing the abbreviation

(5) $$E(\theta) = \arcsin\left[\frac{(1-e^2)^{.5}\sin(\theta)}{1+e\cos(\theta)}\right] = 2\arctan\left[\left(\frac{1-e}{1+e}\right)^{.5}\tan(\theta/2)\right]$$

known as the eccentric anomaly and using (1) and (2)

(6) $$\int_0^\theta dA = \pi abt/T = \pi a^2(1-e^2)^{.5}/T$$

Equating (4) and (6) and solving for t we obtain

(7) $$t = (T/2\pi)[E(\theta) - e\sin E(\theta)].$$

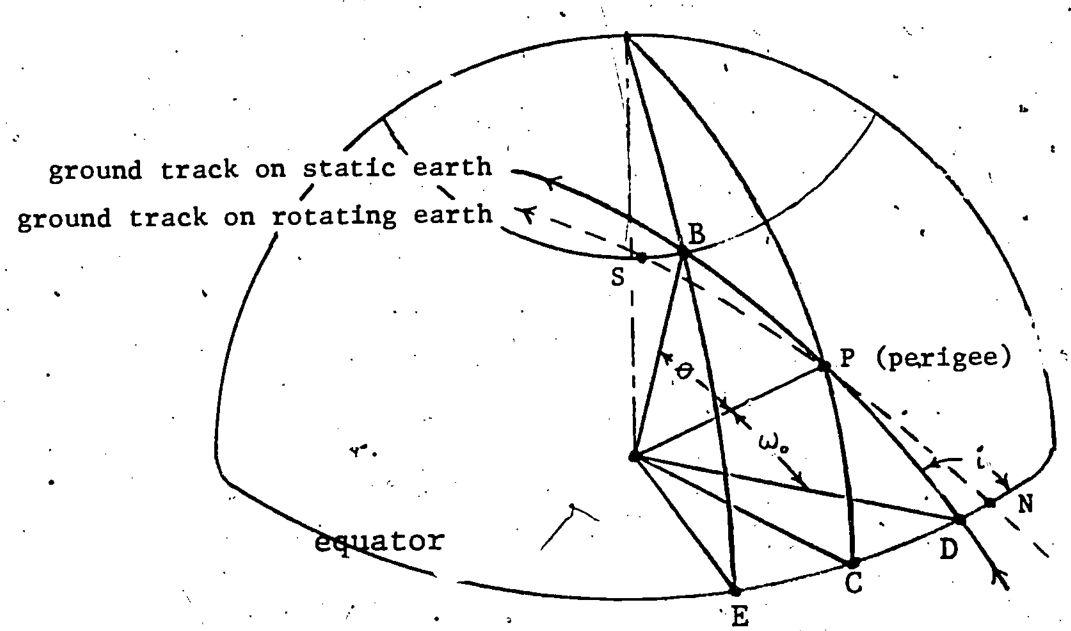
Problem: Verify that the two expressions given for $E(\theta)$ in (5) are equivalent.

PROJECT SP 3

GROUND TRACK

Objective: To illustrate how the equations (Chapter I, Eqs. 1.6 and 1.7) describing the satellite ground track (imaginary curve on surface of earth generated by the point directly below satellite) are derived. This project is an interesting application of spherical trigonmetry.

Derivation. Figures 1 and 2 show the geometry of the ground track problem that we begin with and explain the notation used



	B	C	D	E	N	P	S
Latitude	$\phi(t)$	$0°$	$0°$	$0°$	$0°$	ϕ_p	$\phi(t)$
Longitude	$\lambda^*(t)$	λ_p	$\lambda^*_\circ$	$\lambda^*(t)$	$\lambda_\circ$	λ_p	$\lambda(t)$

Figure 1. Ground Track (* indicates coordinate on static earth)

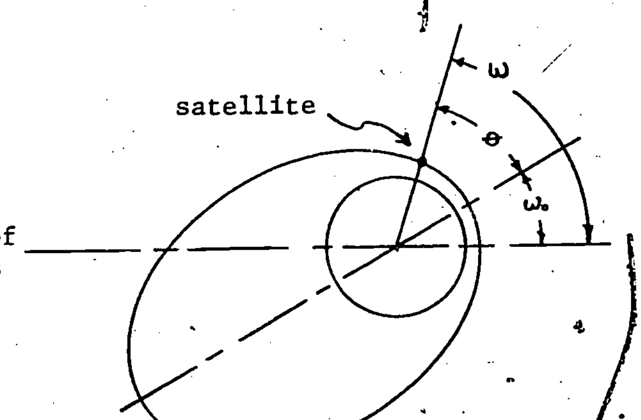


Figure 2. Orbital Plane

$\omega_\circ$ = argument of perigee

θ : measured from perigee

We assume that the latitude and longitude at perigee ($\emptyset_p$, λ_p), the orbital inclination i, and the direction of motion (north or south) of the satellite at perigee are known. Time (t) is measured from perigee, positive following perigee and negative before it. To derive an expression for the latitude $\emptyset(t)$ and the longitude $\lambda(t)$ of the satellite at any time we perform the following steps in order.

1. Compute the time (t_p) and the angle in the orbital plane (ω_0) between the previous ascending node and perigee. ω_0 is called the argument of perigee.

2. Compute the longitude at the previous ascending node using a static (non-rotating) earth model, λ_0^*.

3. Take the rotation of the earth into account to obtain the true longitude at ascending node, λ_0.

4. Compute the latitude of the satellite at any time using a static earth model, $\emptyset(t)$. The rotation of the earth does not effect latitude.

5. Compute the longitude of satellite at any time using a static earth model, $\lambda^*(t)$.

6. Take the rotation of the earth into account to obtain the true longitude, $\lambda(t)$.

7. Generalize (5) and (6) to any octant of earth and any orbital inclination.

The crux of the problem involves applying basic spherical trigonometry formulas to right triangles formed by great circles on the surface of the earth. All longitude lines are great circles. The equator is the only latitude line that qualifies as a great circle. The satellite ground track on a static earth is a great circle but the true ground track is not.

We begin with right spherical triangle PCD (Figure 1). Note that North latitudes and East longitudes are considered positive, South latitudes and West longitudes are considered negative.

$$\sin\emptyset_p = \sin(i)\sin(\omega_0)$$

(1) $$\omega_0 = \arcsin[\sin\emptyset_p \,/\, \sin(i)]$$

$$\cos(\omega_0) = \cos\emptyset_p\cos(\lambda_0^* - \lambda_p)$$

(2) $$\lambda_0^* = \lambda_p + \arccos[\cos(\omega_0) \,/\, \cos\emptyset_p]$$

(3) $$\lambda_0 = \lambda_p + \arccos[\cos(\omega_0) \,/\, \cos\emptyset_p] + |t_p|/4$$

The term t_p in (3) is obtained from ω_0 and Kepler's equation (see Project SP 2). We now look at right spherical triangle BED.

(4) $$\emptyset(t) = \arcsin[\sin(i)\sin(\theta(t)+\omega_0)]$$

(5) $$\lambda^*(t) = \lambda_0^* - \arccos[\cos(\theta(t)+\omega_0) \,/\, \cos\emptyset(t)]$$

(6) $$\lambda(t) = \lambda_0 - \arccos[\cos(\theta(t)+\omega_0) \,/\, \cos\emptyset(t)] - t/4 - |t_p|/4$$

Generalizing (1), (3), (4), and (6) to all octants of the earth and all orbital inclination angles we obtain:

(1a) $$\omega_o = n_1\pi + (-1)^{n_1}\arcsin[\sin\phi_p / \sin(i)]$$

where $n_1 = \begin{cases} 0 \text{ (satellite headed North (or West) at perigee)} \\ 1 \text{ (satellite headed South (or East) at perigee)} \end{cases}$

(3a) $$\lambda_o = \lambda_p + (-1)^{n_2+n_3}\arccos[\cos(\omega_o) / \cos\phi_p] + |t_p|/4$$

$n_2 = \begin{cases} 0 \ (90° \leq i \leq 180°) \\ 1 \ (0° \leq i < 90°) \end{cases}$; $n_3 = \begin{cases} 0 \text{ (perigee in Northern hemisphere)} \\ 1 \text{ (perigee in Southern hemisphere)} \end{cases}$

(4a) $$\phi(t) = \arcsin[\sin(i)\sin(\theta(t)+\omega_o)]$$

(6a) $$\lambda(t) = \lambda_o - (-1)^{n_2+n_4}\arccos[\cos(\theta(t)+\omega_o) / \cos\phi(t)] - t/4 - |t_p|/4$$

$n_4 = \begin{cases} 0 \text{ (satellite in Northern hemisphere, } \phi(t) \geq 0) \\ 1 \text{ (satellite in Southern hemisphere, } \phi(t) < 0) \end{cases}$

The sign conventions for time, latitude, and longitude are summarized below:

North Latitudes (+)
South Latitudes (-)

East Longitudes (+)
West Longitudes (-)

Time after perigee (+)
Time before perigee (-)

6.3 DOPPLER EFFECT

Given a monochromatic source of electromagnetic or acoustical energy, two observers, one moving with the source and one in motion with respect to the source will measure different source frequencies. This is known as the Doppler effect (Johann Doppler 1803-1853). The acoustic Doppler problem is nonsymmetric -- different expressions for the frequency shift are obtained when considering (1) the case where the source is at rest with respect to (wrt) the transport medium while the listener is in motion wrt the transport medium and (2) the case where the source is in motion wrt the transport medium while the listener is at rest wrt the transport medium [1]. For electromagnetic waves there is no preferred reference frame and we obtain a single expression for the frequency shift which differs from both the acoustic expressions. However, to first order in the ratio of the source, observer, or relative velocity to the propagation velocity of the acoustic or electromagnetic waves the expressions will be identical.

The opening experiment illustrates the electromagnetic Doppler effect using beacon signals from the AMSAT-OSCAR 7 satellite. It is designed for introductory physics students. With a change of emphasis it is suitable for astronomy or electronics technology courses. Throughout the experiment we assume that the relative velocity between source and listener is always much less than the speed of light (c) so we can neglect all terms above first order in v/c. This experiment should work well with AMSAT-OSCAR 8 and with the soviet RS satellites. The following experiment can be performed in a single laboratory session. Setting up equipment and taking data requires about one hour for a student familiar with radio equipment. For a more detailed discussion of the Doppler effect and satellites see: Stahl, "Doppler Equations for Satellite Measurement," Proceedings of the IRE, May 1958, p.915.

In addition to experiment SE 2 this section contains three projects. Project SP 4 treats accurate frequency measurement. Project SP 5 introduces the calculation of theoretical Doppler curves for satellites and outlines how such curves can be used in an iterative method for determining orbital parameters. Project SP 6 discusses an interesting anomaly observed in the Doppler curves from satellite beacons which was discovered by ground stations using AMSAT-OSCAR 6.

EXPERIMENT SE 2

DOPPLER EFFECT

OBJECTIVES

The objectives of this experiment include:

1. learning how to predict when the AMSAT-OSCAR 7 satellite will be within range;
2. compiling a Doppler curve for a single pass;
3. accurately determining the time of closest approach;
4. accurately determining the distance between ground station and satellite at closest approach.

EQUIPMENT

1. Basic satellite groundstation consisting of:
   - HF receiver
   - Converter for 146 MHz (if needed)
   - Groundplane antenna (29 MHz or 146 MHz as required)
   - Clock, Weak signal source
2. Polar tracking calculator, instructions, and orbit calendar
3. Textbook: Using Satellites in the Classroom: A Guide for Science Educators, M. Davidoff.

INTRODUCTION

This experiment involves measuring the frequency of downlink signals from one of the beacons aboard the OSCAR 7 satellite. Measurements need only be made during a single pass having a duration of about 20 minutes. For best results choose a pass that is nearly overhead. Appropriate passes generally cross the equator heading north* within plus/minus one hour of: morning laboratories -- 7:30 a.m. local standard time (8:30 a.m. local daylight time); evening laboratories -- 9:00 p.m. local standard time (10:00 p.m. local daylight time). If possible choose a mode B (146 MHz) orbit.

PROCEDURE

1. The orbit calculator and terminology.

Look up the following terms in the text Glossary: AOS, LOS, TCA, Slant range. Familiarize yourself with the operation of the polar projection tracking calculator. Choose an appropriate pass and begin filling in Table I.

*The times are for south to north equatorial crossings as listed in orbit calendar. The satellite will actually be headed south when it crosses the United States during morning (local time) passes.

Table I		
Satellite: AMSAT-OSCAR 7		
Orbit reference no.:		
date:		
downlink beacon frequency:		
	Predicted	Observed
AOS		
TCA		
LOS		

2. Setting the clock.

Set the clock using time signals from the National Bureau of Standards (WWV) at 5.000, 10.000, or 15.000 MHz. The Canadian National Observatory (CHU) at 7.335 MHz may also be used.

3. The Receiving System.

We must first determine the frequency of the satellite beacon which we'll be listening for. Referring to the orbital calendar, the letters A and B following the orbit number specify which beacon will be operating. The mode A beacon operates on approximately 29.502 MHz. The mode B beacon operates on approximately 145.972 MHz. The mode B beacon is preferred for this experiment. To receive the mode A beacon connect the 29 MHz antenna directly to the HF receiver and tune the receiver to 29.502 MHz. To receive the mode B beacon connect the 146 MHz antennna to the VHF converter and connect the converter output to the HF receiver. With the converter connected the HF receiver tuning range 29.500 - 30.000 MHz will cover 145.500 to 146.000 MHz.

The weak signal source should be used to check receiver operation and dial alignment. Set the weak signal source to a frequency close to the beacon and tune the receiver until the signal is heard. Now set the receiver tuning dial to the exact frequency specified on the weak signal source and adjust the receiver RIT control for "zero beat" -- the pitch of the signal goes to zero. The dial frequency is now calibrated. All frequency measurements during this experiment will be made by adjusting the main tuning dial for zero beat and reading the dial setting to the closest 0.2 kHz.

4. Collecting data.

You're now ready to collect data. Prepare a data sheet with two columns; the first for time, the second for frequency. The time column should begin about 3 minutes before predicted AOS and end about 3 minutes after predicted LOS. Use one minute intervals near AOS and LOS and half minute intervals near the predicted TCA. The frequency column will be filled in during the pass. Practice your procedures with your laboratory partner before the pass -- things get pretty hectic once the satellite is within range.

5. Doppler graph.

After the pass prepare a Doppler graph -- frequency (ordinate) vs. time (abscissa).

6. Interpreting: Doppler graph.

Mark the point on the graph where df/dt is a maximum. It will be shown later that this is the point of closest approach. Note the observed TCA, the frequency at TCA and the slope of the Doppler curve at TCA.

TCA (observed): (fill in Table I)

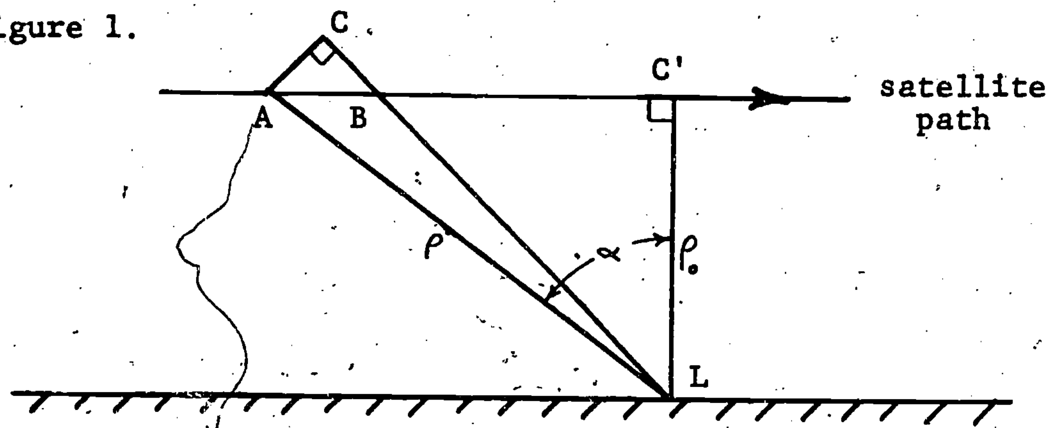
f_o:

df/dt at TCA:

7. Doppler shift model.

To compute the slant range at TCA we need a quantitative Doppler equation. Many texts derive the Doppler equations for the case where the relative velocity between source and listener is along the line joining them. Since the satellite does not pass through our ground station we need a more sophisticated model. The model we'll be using is shown in Figure I. As a first approximation we neglect (1) curvature of the earth, (2) curvature of the satellite orbit, (3) rotation of the earth, (4) relativistic effects.

Figure 1.



t = time from TCA (negative before TCA, positive after TCA)

V = satellite velocity

c = speed of light

f_o = transmitted frequency

f = received frequency as measured by listener at L

T_o = time between two successive crests as measured by observer on satellite (period)

T = time between two successive crests as measured by listener at L

ρ = slant range

ρ_o = slant range at closest approach

α : positive before TCA, negative after TCA

Points A and B in Figure 1 are the positions of the satellite as two successive crests are transmitted. Since the distance BL is less than AL the time it takes for crest B to arrive at L will be less than the time it takes for the crest A to arrive at L. The result is that the time between crests A and B as measured by a listener at L will be shorter than that measured by an observer on the satellite (before TCA). Our definition of average frequency during a cycle will be the reciprocal of the time interval between two successive crests. We now quantify the above description.

The geometry of Figure I shows angle α and angle BAC to be equal.

$\overline{AB}$ is the distance traveled by the satellite during one cycle

$$\overline{AB} = VT_o$$

Referring to right triangle ACB we see that

$$\sin\alpha = s/\overline{AB} = s/VT_o \qquad \text{(where } s = \overline{CB}\text{)}$$

Therefore

$$s = VT_o \sin\alpha$$

The time between two successive crests is approximately given by

$$T \cong T_o(1 - \frac{V}{c}\sin\alpha)$$

This last equation is an approximation because it assumes distances AL and CL to be equal. They differ slightly, but by terms of second and higher order which we've been omitting. Inverting we obtain the expression

$$f \cong f_o(1 - \frac{V}{c}\sin\alpha)^{-1}$$

Expanding this expression in a power series using the binomial theorem and again dropping terms in V/c of second and higher order we obtain

(1) $$f \cong f_o(1 + \frac{V}{c}\sin\alpha)$$

To show that df/dt is a maximum at TCA we must look at the second derivative of f with respect to t. This is facilitated by using the chain rule and the expression $\tan\alpha = -Vt/\rho_o$ to obtain $d\alpha/dt$,

(2) $$\frac{df}{dt} = \frac{f_oV}{c}\frac{d}{dt}(\sin\alpha) = -\frac{f_oV^2}{c\rho_o}\cos^3\alpha$$

(3) $$\frac{d^2f}{dt^2} = \frac{f_oV}{c}\left\langle \cos\alpha\frac{d^2\alpha}{dt^2} - \sin\alpha\left(\frac{d\alpha}{dt}\right)^2\right\rangle$$

The only place within range where the right-hand side of Eq. 3 will be zero is when $\alpha = 0$. Verify all the above equations in your laboratory report. At TCA we therefore have

$$(df/dt)_{TCA} = (df/dt)_{max}$$ and, from the geometry, $\cos\alpha = 1$

Evaluating Eq. 2 at TCA we obtain

(4) $$\rho_o = -\frac{f_oV^2}{c(df/dt)_{max}}$$

If we assume that the orbit is circular and that the period is known

(114.9 minutes) we can obtain V from Eq. 5

$$(5) \quad V = (2\pi GM/T)^{1/3}$$

Verify this equation (see text section 1.3) and compute V.

Finally, compute ρ_o (Eq. 4) using f_o and $(df/dt)_{max}$ as determined from your graph (step 6) and V as computed above.

8. Comments.

We've seen how a Doppler curve can enable us to accurately determine TCA, slant range at closest approach, and the actual transmission frequency. A single Doppler curve actually provides us with a unique signature for a satellite orbit. Using a more sophisticated Doppler model one can actually determine the 6 parameters needed to characterize an elliptical orbit or the 4 parameters needed to characterize a circular orbit. The task of determining the orbital parameters (elements) is usually accomplished using a combination of Doppler and ranging measurements (see section 5.7). A closely related problem is that of using Doppler data to determine the latitude and longitude of one's ground station with a satellite whose orbital elements are very accurately known. This technique is used with navigation satellites such as the Transit series.

9. Question.

In step 7 we made a number of simplifying assumptions. Indicate how the problem could be solved without assumptions (1), (2) and (3) (do not solve). Is there any necessity of eliminating assumption (4)?

SAMPLE RESULTS FOR SE 2

Satellite: AMSAT-OSCAR 7
Orbit: 7603 BX
Date: 14 July 1976
Ascending Node: 1346:39 UTC; 256.5°W
Ground Station Location: Baltimore, MD.; *Latitude*: 39.35°N, *Longitude*: -76.6°W
Notes: Descending orbit passing nearly overhead

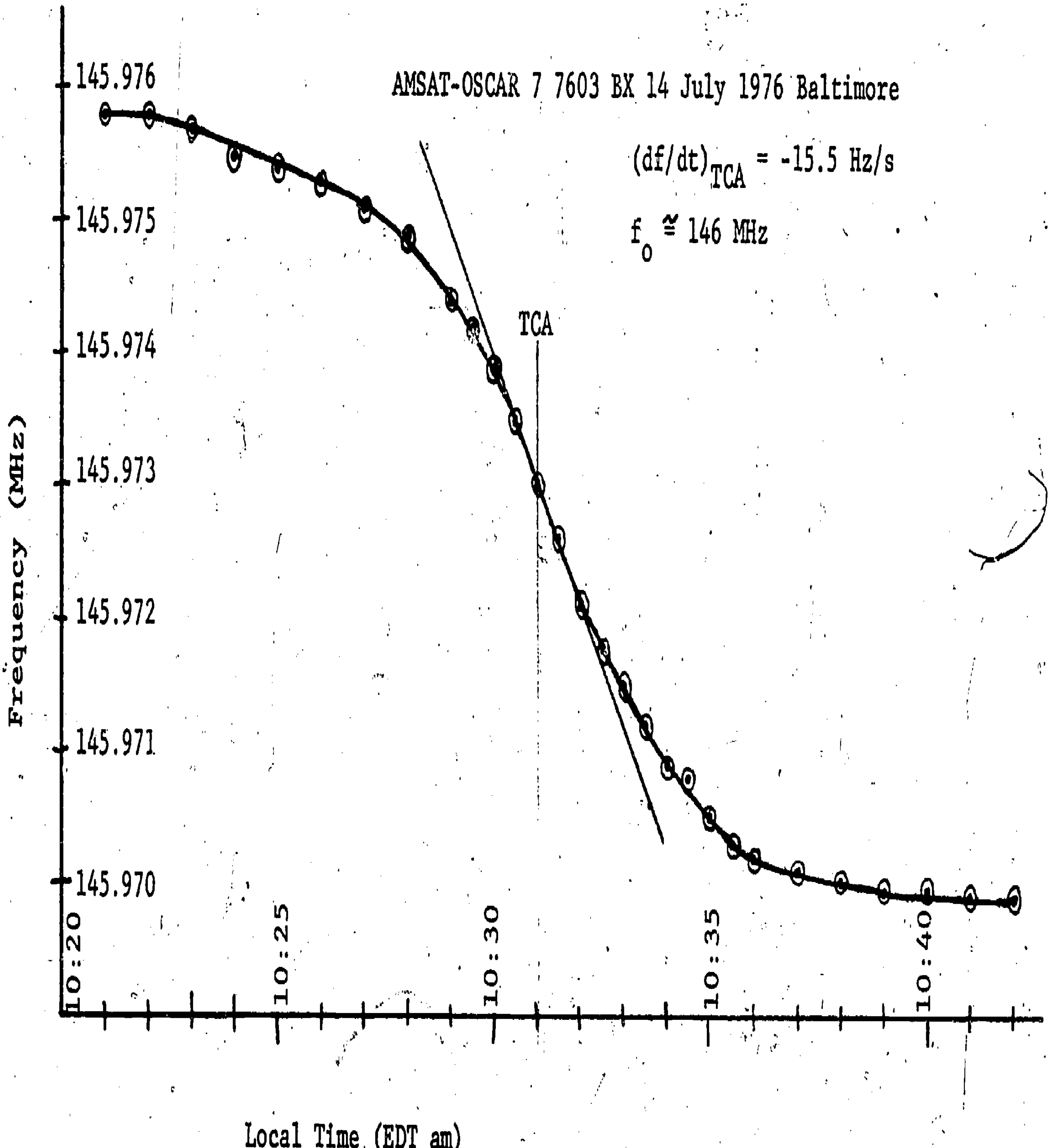
Doppler Curve: See next page.

Slant Range at TCA

$V = 7.11$ km/s

from graph: $(df/dt)_{TCA} = -15.5$ Hz/s, and $f_o = 145.97$ MHz

$$\text{slant range} = -\frac{f_o V^2}{c(df/dt)_{TCA}} = 1.59 \times 10^6 \text{ m} = 988 \text{ miles}$$


AMSAT-OSCAR 7 7603 BX 14 July 1976 Baltimore
$(df/dt)_{TCA}$ = -15.5 Hz/s
$f_0 \cong$ 146 MHz
TCA
145.976
145.975
145.974
145.973
145.972
145.971
145.970
Frequency (MHz)
10:20
10:25
10:30
10:35
10:40
Local Time (EDT am)

PROJECT SP 4

ACCURATE FREQUENCY MEASUREMENT

Objective. The objective of this project is to illustrate methods for accurate frequency measurement using satellite downlink signals as a signal source. The project, using a variation of the Doppler Shift Experiment (SE 2), was designed for electronics technology students at Catonsville Community College. It essentially omits the mathematics contained in step 7 of SE 2 and emphasizes accurate frequency measurement techniques instead. Discussion questions included in this experiment are reproduced below.

DISCUSSION QUESTIONS

3.1 Most receivers used for voice communications are designed so that the audio response falls off below 400 Hz. The Heath receiver low frequency audio response has been enhanced for this experiment by increasing the size of the interstage coupling capacitors. Why was this modification necessary?

3.2 List possible methods of obtaining time signals without synchronizing with WWV or CHU.

3.3 In 2.1 why do we specify synchronizing the receiver with the "highest WWV frequency" that can be received?

3.4 The satellite beacon transmitter is crystal controlled. In our Doppler shift measurements we assumed that the beacon frequency was constant. Is this a reasonable assumption? Explain.

3.5 What effect would Doppler shift have if a transmitter in the satellite were transmitting a SSB audio signal?

3.6 Noting that the satellite carries a linear transponder, how would the uplink and downlink Doppler shifts combine? Give qualitative answer.

3.7 What are the advantages and disadvantages of using a power sharing limited bandwidth transponder, like the one carried by Oscar 7, to relay FM signals like those used in the land mobile services?

3.8 Given a $10,000 budget, discuss how you would design an optimal ground station to be used for making accurate frequency measurements on an incoming signal that is being shifted in frequency by Doppler effects. Be sure to consider phase locked loops and premixing of receiver oscillators before injection into counter.

3.9 Consider the land mobile services again. Contrast the advantages and disadvantages of a repeater system using a single stationary (synchronous) satellite (altitude = 22,300 miles) to one using a number of non-synchronous (altitude = 900 miles) satellites.

PROJECT SP 5

THEORETICAL DOPPLER CURVES

Objective: To compute a set of theoretical Doppler curves for the radio beacon aboard AMSAT-OSCAR 7. A student capable of independent work might undertake this activity as a term project. The project requires a working knowledge of basic physics, analytic geometry and calculus, and computer programming and it assumes a familiarity with parts of this text. Specifically, the student should

1. perform experiments SE 1 and SE 2,
2. work through projects SP 1, SP 2, and SP 3,
3. read Chapter I and Chapter II sections 1 and 3,

before or during this project.

Procedure. Use the tracking calculator to identify the equatorial crossing that will result in a northbound overhead pass. Doppler graphs will be prepared for this ascending node and ascending nodes in ± 15° increments in longitude. A total of seven graphs will be prepared. For example, a station in Washington, D.C. would prepare graphs for ascending nodes at 19°, 34°, 49°, 64° (overhead pass), 79°, 94°, 109°. The following steps describe how each graph is prepared.

1. Note the latitude and longitude of your ground station. Start with the westernmost ascending node.

2. Compute the slant range at 1 minute intervals. This can be done by using text Eqs. 1.6, 1.7, and 1.8 to determine the latitude and longitude of the subsatellite point. Next use Eqs. 1.12 and 1.9 to determine slant range.

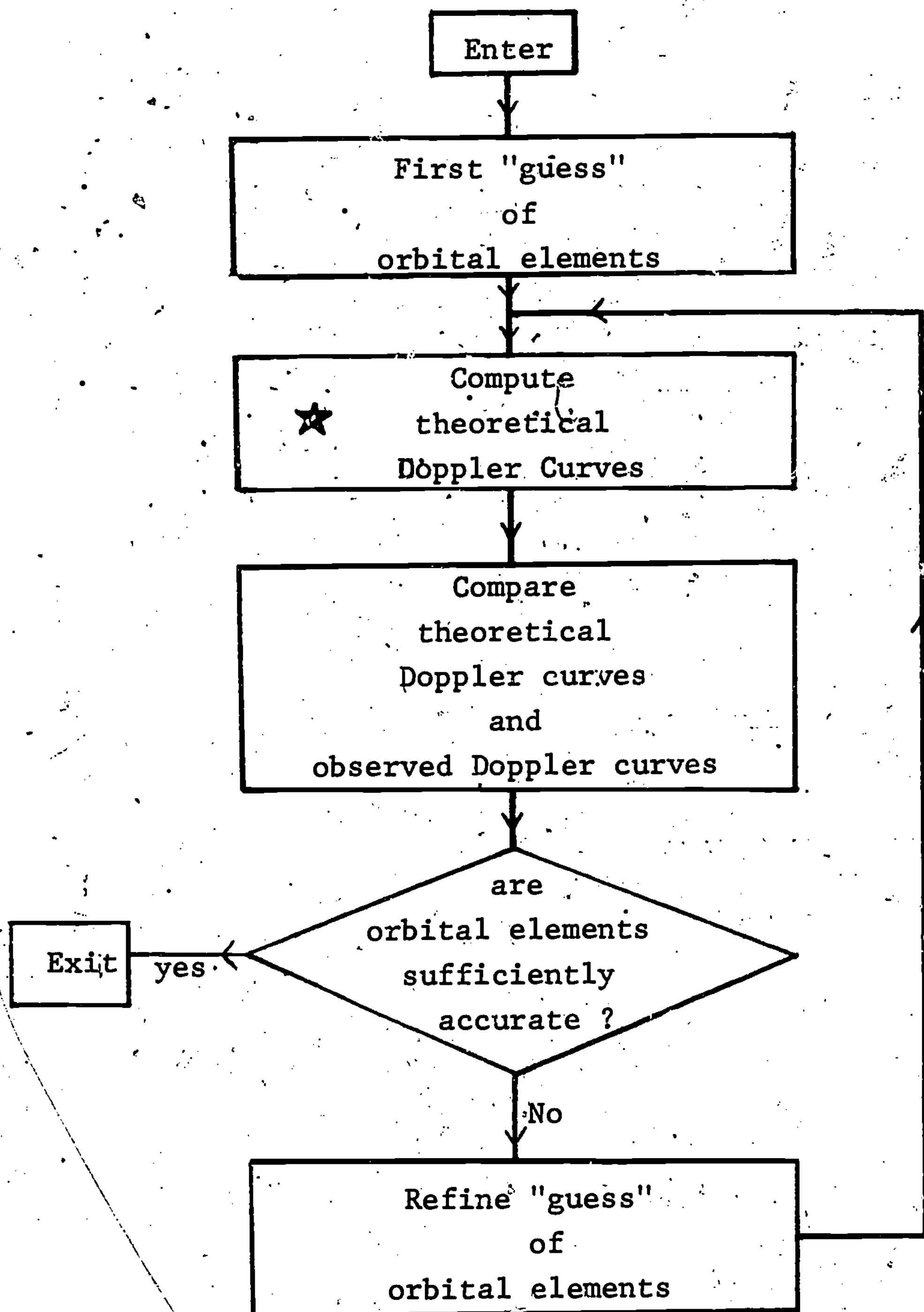
3. Compute an average period for each one minute interval assuming that the actual satellite beacon frequency is 146.000 MHz. This step closely parallels the computation performed in experiment SE 2, Procedure step 7. However, instead of averaging over 1 cycle we average over the number of cycles that occur in one minute. Neglect all terms of second or higher order.

4. Convert the average period calculations to frequencies.

5. Plot these frequencies as a function of time.

6. Return to step 2 and redo the calculations for the ascending node which occurs 15° further east.

Comments. Note that the Doppler curves you have prepared are asymmetrical. Had we started with an elliptical orbit these asymmetries would be even greater. It is precisely these asymmetries which enable us to solve the "inverse" of the above problem -- determining the orbital elements when Doppler curves are available. A flow chart for the orbit determination problem follows. This project (SP 5) constitutes a crucial part of the orbit determination problem. The computer program developed for SP 5 can be incorporated in the orbit determination problem at the step (*) and the curves produced can be used to help one devise a curve fitting technique and a method for refining the guess of the orbital elements so as to converge to the actual values.



Flow chart: Determining orbital elements of satellite

PROJECT SP 6

ANOMOLOUS DOPPLER

<u>Objective</u>: To introduce the reader to the anomolous Doppler effect and to illustrate how careful measurements similar to those described in experiment SE 2, using relatively simple equipment, can lead to meaningful scientific discovery.

In 1972 an experimenter collecting Doppler data on the 435 MHz beacon aboard AMSAT-OSCAR 6 noticed a strange effect on a northbound pass [2, 3]. For the first few minutes after AOS the frequency of the observed signal increased instead of decreasing as expected. After thoroughly checking a number of factors which could have accounted for the observations (i.e., drift in ground station frequency measuring equipment, change in satellite temperature affecting beacon oscillator frequency, etc.) it was decided that an interesting physical anomaly was being observed. The effect was later observed on navigational satellites operating in the vicinity of 400 MHz. An exhaustive experimental investigation was undertaken to delineate the geographical and temporal (time of day, season of year, etc.) extent of the anomaly and to determine the frequency range over which it occurred. In addition, an attempt was made to correlate the effect with physical changes in the ionosphere which were suspected of being related.

Although a great deal of data was collected no firm conclusions have ever been reached. To this authors knowledge the anomolous Doppler effect has not been observed on satellites operating at either 137 MHz or 146 MHz but experimenters should be alert to the possibility of this occurring since the effect is a small one and may be overlooked. Changes of up to 700 Hz have been observed at 435 MHz so any effect observed at 146 MHz will probably be less than 250 Hz. If your equipment is capable of this resolution and you observe the effect at 146 MHz please notify AMSAT with pertinent information.

6.4 TELEMETRY REDUCTION

This section contains two experiments and three projects. The experiments, SE 3 and SE 4, involve decoding the telemetry information being sent from AMSAT-OSCAR 7. The telemetry information is used to illustrate:

1. how energy balance concepts determine the equilibrium temperature of a satellite,
2. how the orientation of a satellite may be determined from solar array current measurements.

SE 3 and SE 4 are suitable for physics and engineering student laboratories. The format was designed so that all data could be collected in a single two hour laboratory session. These experiments are exceptionally good takeoff points for term projects, independent study laboratories, open ended laboratory projects, senior theses, etc. Experiments SE 3 and SE 4 can also be performed using AMSAT-OSCAR 6. Telemetry decoding information is contained in Chapter III and Chapter IV.

This section also contains three projects, SP 7, SP 8, and SP 9, which discuss how AMSAT-OSCAR 7 can be used to:

1. relate energy balance concepts to power available for satellite instrumentation,
2. determine the solar constant,
3. measure the earth's albedo.

Satellite telemetry is sent in a number of formats (see Chapter III and Chapter IV). Experiments SE 3 and SE 4 focus on Morse code telemetry because (1) no special equipment is needed to decode telemetry in this format and (2) it can be received with the simplest ground station set up for 29.5 MHz operation. The experiments can easily be generalized to employ other telemetry formats if desired.

The AMSAT-OSCAR 7 Morse code telemetry system is discussed in Chapter III and Chapter IV. Briefly, satellite physical parameters of interest are measured, converted to digital form, and transmitted down to earth in the form of Morse code integers. Because of the redundancy of the Morse code digits the telemetry can be easily decoded using the Morse code numbers chart included with the experiment sheet. A multi-speed tape recorder (record at high speed, play back at low speed) may be of help.

The experiments in this section all assume a familiarity with a tracking technique and with basic satellite terminology such as would be acquired by performing experiment SE 2. We begin with SE 3 -- ENERGY FLOW IN SPACE.

EXPERIMENT SE 3

ENERGY FLOW IN SPACE

OBJECTIVES

The objectives of this experiment include:

1. learning how to decode AMSAT-OSCAR 7 telemetry,
2. studying how energy balance concepts are related to satellite equilibrium temperature.

EQUIPMENT

1. Basic satellite groundstation consisting of:
   HF receiver for 29.5 MHz
   Groundplane antenna for 29.5 MHz
2. Polar tracking calculator, instructions, and orbit calendar
3. Three speed audio tape recorder
4. Reference text: Using Satellites in the Classroom
5. Weak signal source
6. Clock

INTRODUCTION

This experiment involves three parts.

1. Raw data collection. A tape recording of satellite telemetry signals will be acquired during a single pass.
2. Data reduction. The raw data will be translated into actual parameter values for the systems of interest.
3. Data interpretation. Measured parameter values will be related to theoretical values obtained from energy flow concepts.

PROCEDURE

1. Raw data collection.

Using the orbital calculator and orbit calendar, choose an orbit that will be within range for at least 15 minutes during a pass when the satellite is in mode A. Tune the receiver to time signals from the National Bureau of Standards (WWV) at a convenient frequency (5.000, 10.000, or 15.000 MHz). Set the clock. Use a jumper to directly connect the receiver and tape recorder -- do not use a microphone to record from the receiver speaker. Set the recorder to 7-1/2 ips and adjust the receiver and recorder volume controls to the proper levels. If possible, monitor the recorder output. If the recorder does not have provisions for monitoring during record then the receiver output should be monitored. Check for proper operation of the system by playing back the WWV recording. Now set the receiver as you would for Doppler measurements on the 29.5 MHz beacon. Check the receiving system once again, this time using the weak signal source near 29.5 MHz.

Start the recorder a few minutes before AOS. Adjust the main tuning dial ± a few kHz until the beacon is acquired. During the pass the tuning will have to be periodically adjusted to compensate for Doppler shift of the beacon frequency.

2. Data reduction.

Information on data reduction is contained in Section 3.2 of the text -- see especially p.3-13 ff. Channels of special interest for this experiment are 3D, 4A, 4B, 4C, 4D, 5A, 6D. Play back your recording at low speed and decode the digits using the chart below. Fill all numbers (N) in Table 2.

1	· − − − −	6	− · · · ·
2	· · − − −	7	− − · · ·
3	· · · − −	8	− − − · ·
4	· · · · −	9	− − − − ·
5	· · · · ·	0	− − − − −

Table 1.

Complete filling in Table 2 using the relation between temperature (T) measured in degrees Centegrade (°C) and the contents of the channel (N).

$$T(^\circ C) = 95.8 - (1.48)N$$

This equation comes from Table 1 on page 4-8.

3. Data Interpretation

In this section we compute the power input to (P_{in}) and power outflow from (P_{out}) the satellite and see how conservation of energy determines the equilibrium temperature of the spacecraft.

In the vacuum of space we need only consider radiant forms of energy. The sun is the sole source of energy for the satellite:

(1) $$P_{in} = P_o \langle A \rangle \alpha \beta$$

where P_o (solar constant) = incident energy per unit time on a surface of unit area (perpendicular to direction of radiation) at 1.49 x 10^{11} m (earth-sun distance) from the sun. P_o = 1,380 watts/m^2

$\langle A \rangle$ = effective capture area of satellite for solar radiation.

α = eclipse factor (fraction of time satellite exposed to sun).

β = absorptivity (fraction of incoming radiation absorbed by satellite).

CHANNEL	PARAMETER	FRAME NUMBER Contents / Parameter Value											
		/	1	2	3	4	5	6	7	8	9	10	11
3D	Battery temperature	N °C											
4A	Baseplate temperature	N °C											
4B	Power Amp (29.5 MHz) temperature	N C											
4C	+X facet temperature	N °C											
4D	+Z facet temperature	N °C											
5A	Power Amp (146 MHz) temperature	N °C											
6D	Calibration N *	N											

*If telemetry system is operating correctly this channel will register 50±1.

Table 2.
AMSAT—OSCAR 7 Telemetry

Power outflow from the satellite consists of blackbody radiation at temperature T (P_{BB}) and radio emissions (P_{EM}). Since P_{BB} is very much greater than P_{EM} we temporarily ignore the latter (we verify this later).

(2) $$P_{out} = A\sigma cT^4$$

where A = surface area of satellite

σ = Stefan-Boltzmann constant = $\dfrac{5.67 \times 10^{-8} \text{ joules}}{K^4 m^2 s}$

c = average emissivity factor for satellite surface

T = temperature (K)

At equalibrium $P_{in} = P_{out}$.

(3) $$P_o \langle A \rangle \alpha \beta = A\sigma cT^4$$

(4) $$T^4 = \frac{P_o \alpha \beta}{\sigma c} \frac{\langle A \rangle}{A}$$

Reasonable values for the various constants are $\alpha = .8$, $\beta = .8$, $c = .5$. Calculating A for AMSAT-OSCAR 7 (see text page 4-5, Figure 1), A = 7770 cm^2. It only remains for us to calculate $\langle A \rangle$ in order to solve Eq. 4 for T.

The calculation of $\langle A \rangle$. To calculate $\langle A \rangle$ we must average over various orientations of the satellite. The technique used depends on a number of factors including satellite geometry, stabilization system, etc. The approach we use is appropriate for the geometry of AMSAT-OSCAR 7 when one assumes that all spacecraft orientations are equally probable.

Let the spacecraft Z axis be perpendicular to the sun-spacecraft line and assume a constant rotational speed about this axis. See Figure 1. We compute the average effective surface area for one of the 8 satellite faces (true area = A_o) as the satellite undergoes one complete rotation.

$$A^* = \frac{1}{2\pi} \int_{-\pi/2}^{+\pi/2} A_o \cos(\theta)\, d\theta = A_o/\pi$$

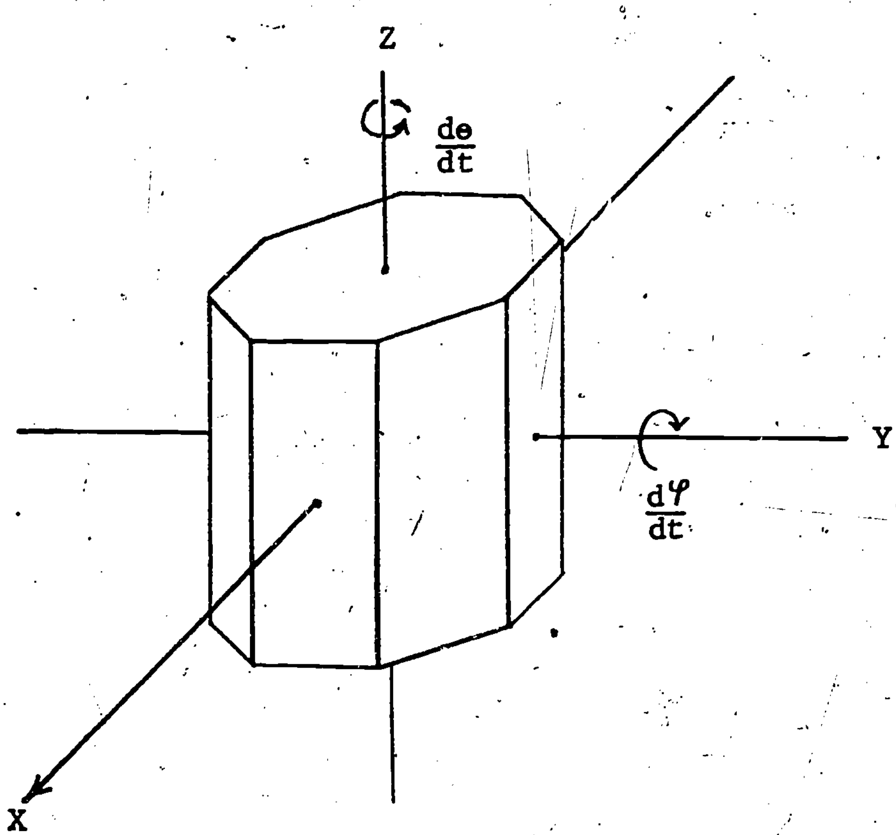
where the limits of integration take into account the fact that A_o does not contribute to A* when it faces away from the sun. The effective area for all 8 faces in this particular orientation is therefore

$$8A_o/\pi = 1,485 \text{ cm}^2.$$

Adding an additional 180 cm^2 for the contribution from the mounting collar (this value is discussed later) we obtain

$$\langle A_1 \rangle = 1,665 \text{ cm}^2$$

Figure 1.



Now consider rotation of the satellite about the y axis. Let A_2 be the area of the top (or bottom) plate. From text Figure 1 on page 4-5 $A_2 = 1{,}270\ cm^2$. A_2 and $\langle A_1 \rangle$ will both contribute to the total effective area $\langle A \rangle$.

$$\langle A \rangle = 2\,\frac{1}{2\pi}\int_{-\pi/2}^{+\pi/2} \langle A_1 \rangle \cos\varphi\, d\varphi + 2\,\frac{1}{2\pi}\int_{-\pi/2}^{+\pi/2} A_2 \cos\varphi\, d\varphi$$

$$\langle A \rangle = (2/\pi)(\langle A_1 \rangle + A_2) = 1{,}870\ cm^2$$

Finally, inserting the value for $\langle A \rangle$ in Eq. 4 we obtain

$T = 294\ K = 21\ °C.$

The term $\langle A \rangle / A$ which appears in Eq. 4 is called the aspect ratio.

DISCUSSION

1. From Table 2 we see that the satellite is not at a single temperature. A good average T for the spacecraft is the baseplate temperature (channel 4A) because it has a large heat capacity and is in good thermal contact with the rest of the spacecraft. Why is the channel 4B temperature much higher than the others? Suggest some possible causes for the variation in the other temperature readings.

Channel 4B is the temperature of the final amplifier of the mode A transmitter which is operating. Since no amplifier is 100% efficient, it also produces heat.

We can picture the satellite as consisting of a number of masses of different heat capacities connected by thermal links of various impedances. Thermal input from the sun to each mass is different. Under these conditions we expect different temperatures. The temperature differences will be small when the link impedances are low.

2. Use Eq. 2 to compute the power being radiated by the satellite in the form of blackbody radiation. Compare this to the average radiated radio power which is about 2 watts.

$$P_{BB} = A\sigma\varepsilon T^4 = .777 \text{ m}^2 \frac{5.67 \times 10^{-8} \text{ joules}}{K^4 m^2 s} (.5)(294)^4 K^4$$

$$P_{BB} = (.777)(5.67)(2.94)^4(.5) \text{ joules/s} = 165 \text{ watts}$$

P_{EM} *is less than 2% of* P_{BB}.

3. What effect would an error of ± 10% in our calculation of P_{in} or P_{out} have in our results for T?

from Eq. 4: $T^4 = k$. *A ±10% change in k produces* $\tilde{T}$ *where*

$$\tilde{T}^4 = 1.1\ k \quad \text{or} \quad .9\ k$$

$$\tilde{T} = T(1.1)^{1/4} \text{ or } T(.9)^{1/4} \quad \text{where } T = 294 \text{ K}$$

$$\tilde{T} = 301 \text{ K} \quad \text{or} \quad 286 \text{ K}$$

4. The value for the solar constant, P_o = 1,380 watts/m^2, used in this experiment was obtained from satellite measurements. It varies about ±2% depending on solar activity. Using this value of P_o compute the total power being radiated by the sun.

r = *mean sun-earth distance* = 1.49×10^{11} m

Power output of sun = $4\pi r^2 P_o = 3.85 \times 10^{26}$ *watts*

5. Assuming that the sun behaves like a spherical blackbody of radius R = 7.03 x 10^8 m, what temperature would it be at to produce the power computed in 4 ?

Power output of sun = (power output per unit area) (surface area)

$$= (\sigma T^4)(4\pi R^2) = 3.85 \times 10^{26} \text{ watts}$$

$T = 5,750.$ K

6. We used energy conservation principles to compute T. We could have used these principles and the measured value of T to compute β/c (absorptivity/emissivity). β and c are not actually constants. They depend on the surface material and on the relevant radiation wavelengths. The values we assigned to them are averages. How can we use the satellite telemetry to separate c and β? Consider the fact that c and β are different for different surfaces and that we have temperature measurements for various surfaces. Would data on baseplate temperature as a function of time as the satellite enters eclipse be of use? Would solar panel currents which tell which panels are facing the sun be of use?

7. Our eclipse factor α was chosen as .8. Actually, it varies over the course of a year. P_o also varies in the course of the year since the earth is closer to the sun in the winter (northern hemisphere). Given the maximum average temperature (286 K on Jan. 30) and minimum average temperature (276 K on June 30) of the satellite during a particular year and ignoring the change in P_o compute α_{max} and α_{min}. Assume that $\alpha = .80$ when T = 281 K.

Using Eq. 4: $T^4 = k\alpha$; $(281)^4 = k(.80)$; $k = 77.9 \times 10^8$

$T^4 = 77.9 \times 10^8 \alpha$

$\alpha_{min} = .74$

$\alpha_{max} = .86$

8. Compute the maximum and minimum cross-sectional areas of the OSCAR 7 satellite for a plane containing the z axis (ignore the mounting collar).

maximum cross section = $42.4 \times 36 = 1,526$ cm^2

minimum cross section = $1,410$ cm^2

$$\left.\begin{array}{l} \text{arithmetic mean} = 1,468 \text{ cm}^2 \\ 8\langle A\rangle = 1,485 \text{ cm}^2 \end{array}\right\} \text{differ by about } 1\%$$

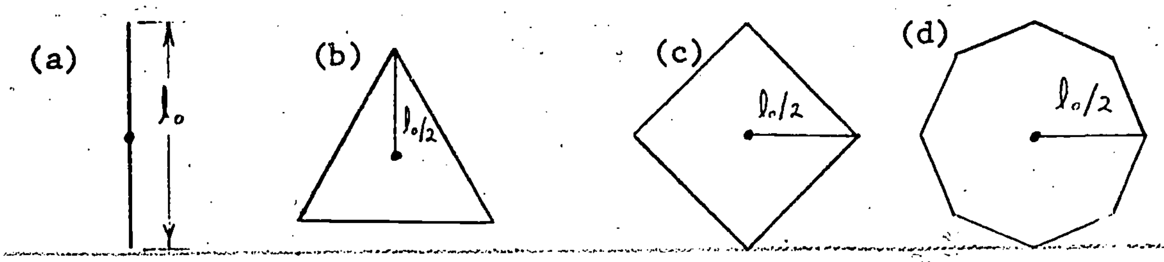
How does the arithmetic mean of these two values compare with the effective area of the 8 faces averaged with respect to rotation about the z axis which was previously calculated? We allowed 180 cm^2 for the effective area of the mounting collar $\langle A_c\rangle$. This value is simply the cross-sectional area. If this estimate of collar effective area is off by ±10% what would the percentage error in $\langle A_1\rangle$ be?

$$\langle A_1\rangle = 8\langle A\rangle + \langle A_c\rangle$$

A 10% error in $\langle A_c\rangle = 18$ cm^2

percentage error in $\langle A_1\rangle \cong 18/1,800 \cong 1\%$

9. Find the effective areas of the following figures. All are rotating at constant velocity about the z axis (perpendicular to plane of paper) and have a height of 1 cm.



Write a general expression for the effective area of an n sided object having the symmetries of the objects shown with a maximum radius of $\ell_0/2$. Does the maximum effective area converge as $n \longrightarrow \infty$? If so, to what does it converge?

area of one side $= \ell_0 \sin(\pi/n)$

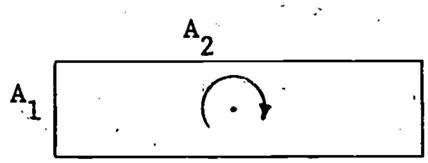
$$\text{effective area of one side} = \frac{1}{2\pi}\int_{-\pi/2}^{+\pi/2} \ell_0 \sin(\pi/n)\cos\theta\, d\theta = \frac{\ell_0}{\pi}\sin(\pi/n)$$

$$\text{total effective area} = \langle A\rangle = \frac{n\ell_0}{\pi}\sin(\pi/n)$$

total effective area as $n \longrightarrow \infty$

$$\lim_{n\to\infty}\langle A\rangle = \lim_{n\to\infty}\left[\frac{n\ell_0}{\pi}\left(\frac{\pi}{n} - \frac{1}{3!}\left(\frac{\pi}{n}\right)^3 + \frac{1}{5!}\left(\frac{\pi}{n}\right)^5 - \cdots + \cdots\right)\right] = \ell_0$$

10. Compute the effective area for the following figure. Show that as $A_1 \longrightarrow 0$ we obtain case (a) of 9.



$$\omega = \frac{d\theta}{dt} = \text{constant}$$

$$\langle A\rangle = 2\frac{1}{\pi}\int_0^{\pi/2} A_1\cos\theta\, d\theta + 2\frac{1}{\pi}\int_0^{\pi/2} A_2\cos\theta\, d\theta = \frac{2}{\pi}A_1 + \frac{2}{\pi}A_2 = \frac{2}{\pi}(A_1 + A_2)$$

$$\lim_{A_1\to 0}\langle A\rangle = \frac{2}{\pi}A_2$$ *This checks with case (a) of 9. when both sides (n=2) are considered.*

11. Compute the equilibrium temperature of another satellite -- the earth. Discuss your assumptions and answer.

Using a simple model involving only radiation and assuming that $\beta = c$ *we obtain* $T = 6°C$.

EXPERIMENT SE 4

SATELLITE ORIENTATION

OBJECTIVES

The objectives of this experiment include:

1. learning how to decode AMSAT-OSCAR 7 telemetry,
2. determining satellite orientation from telemetry data.

EQUIPMENT

1. Basic satellite groundstation consisting of:
   HF receiver for 29.5 MHz
   Groundplane antenna for 29.5 MHz
2. Polar tracking calculator, instructions, and orbit calendar
3. Three speed audio tape recorder
4. Reference text: Using Satellites in the Classroom
5. Weak signal source
6. Clock

INTRODUCTION

This experiment involves three parts:

1. Raw data collection. A tape recording of satellite telemetry signals will be acquired during a single pass.
2. Data reduction. The raw data will be translated into actual parameter values for the systems of interest.
3. Data interpretation. Measured parameter values will be used to compute satellite orientation as a function of time.

PROCEDURE

1. Raw data collection.

Using the orbital calculator and orbit calendar, choose an orbit that will be within range for at least 15 minutes during a pass when the satellite is in Mode A. Tune the receiver to time signals from the National Bureau of Standards (WWV) at a convenient frequency (5.000, 10.000, or 15.000 MHz). Set the clock. Use a jumper to directly connect the receiver and tape recorder -- do not use a microphone to record from the receiver speaker. Set the recorder to 7-1/2 ips and adjust the receiver and recorder volume controls to the proper levels. If possible, monitor the recorder output. If the recorder does not have provisions for monitoring during record then the receiver output should be monitored. Check for proper operation of the system by playing back the WWV recording. Now set the receiver as you would for Doppler measurements on the 29.5 MHz beacon. Check the receiving system once again, this time using the weak signal source near 29.5 MHz.

Start the recorder a few minutes before AOS. Adjust the main tuning dial ± a few kHz until the beacon is acquired. During the pass the tuning will have to be periodically adjusted to compensate for Doppler shift of the beacon frequency.

2. Data reduction.

Information on data reduction is contained in Section 3.2 of the text -- see especially p.3-13 ff. Channels of special interest are 1A, 1B, 1C, 1D, 2A, 6D. Play back your recording at low speed and decode the digits using the chart below. Fill all numbers (N) in Table 2.

1	· — — — —	6	— · · · ·
2	· · — — —	7	— — · · ·
3	· · · — —	8	— — — · ·
4	· · · · —	9	— — — — ·
5	· · · · ·	0	— — — — —

Table 1.

Complete filling in Table 2 using the following formulas:

$$I \text{ (ma)} = 1{,}970 - 20(N) \quad \text{(Channels 1B, 1C, 1D, 2A)}$$

$$I \text{ (ma)} = (29.5)N \quad \text{(Channel 1A)}$$

These formulas were obtained from Table 1 on page 4-8. Note that channel 1A has an intermittent problem which may cause erroneous readings. If channel 1A contains all zeros or takes large erratic jumps it should be discounted.

3. Data Interpretation.

In this section we compute the orientation of the satellite with respect to the sun. This will be done for each telemetry frame so that we can follow the satellite orientation as a function of time while the pass is within range. Satellite orientation will be specified by relating a fixed cartesian reference frame in the satellite to a line joining the satellite and sun.

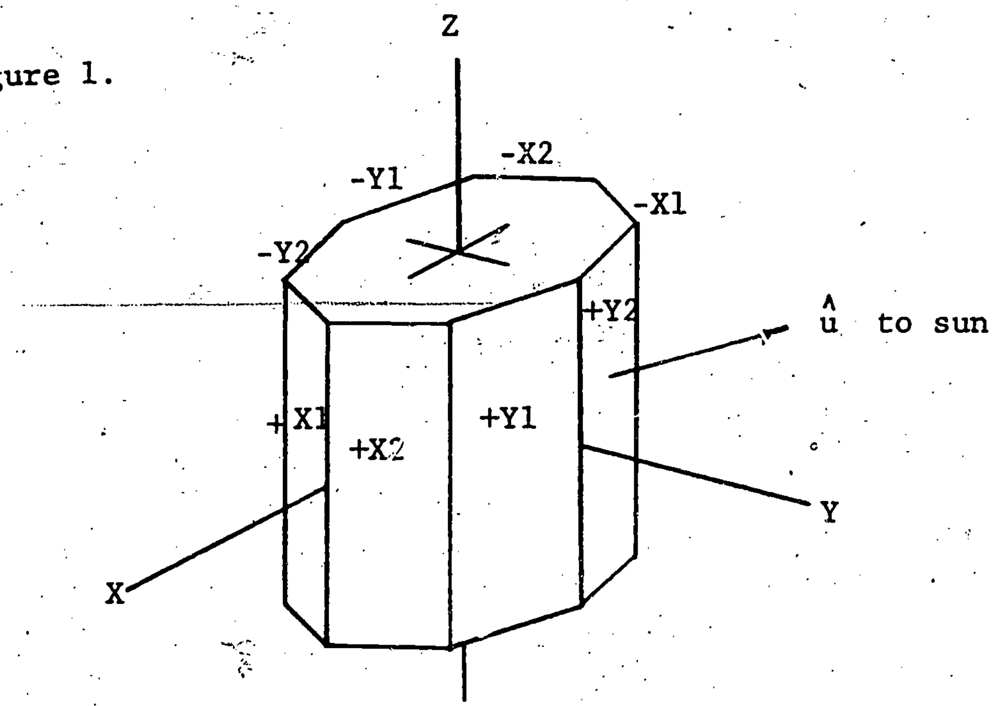
CHANNEL	PARAMETER	FRAME NUMBER Contents / Parameter Value												
			1	2	3	4	5	6	7	8	9	10	11	12
1A	Total Solar Panel current	N I(ma)												
1B	+X facet current	N I(ma)												
1C	+Y facet current	N I(ma)												
1D	-X facet current	N I(ma)												
2A	-Y facet current	N I(ma)												
6D	Calibration N *	N												

*If telemetry system is operating correctly this channel will register 50 ± 1

Table 2.
AMSAT-OSCAR 7 Telemetry

Geometry. The cartesian reference frame attached to the satellite is shown in Figure 1.

Figure 1.



The sun vector (a unit vector pointing from the satellite to the sun) is given by

$$\hat{u} = \cos\alpha\,\hat{i} + \cos\beta\,\hat{j} + \cos\gamma\,\hat{k}$$

where α, β, γ are the angles between the satellite +X, +Y, +Z axes and the sun vector respectively. Normal unit vectors to the +X1 and +X2 faces are given by

$$\hat{x}_{+1} = \cos(22.5°)\,\hat{i} - \sin(22.5°)\,\hat{j}$$

(1)

$$\hat{x}_{+2} = \cos(22.5°)\,\hat{i} + \sin(22.5°)\,\hat{j}$$

Similar notation is used for the other 6 facets.

Solar Panel Currents. The output current (I) of a solar panel is given by:

(2) $$I = I_{max} \cos\varphi = I_{max}\,\hat{u}\cdot\hat{n}$$

where φ is the angle between the normal to the panel and the sun vector. This equation only holds for $-90° \le \varphi \le +90°$. We restrict our analysis to the +X quadrant. However, the results are immediately generalizable to the other three quadrants. The AMSAT-OSCAR 7 +X quadrant current is the sum of the +X1 facet current and the +X2 facet current:

$$I_{+X} = I_{+X1} + I_{+X2}$$

Telemetry only gives us the quadrant current. We usually do not have access to each of the facet currents.

Using Eq. 2 for the current produced by each facet and Eq. 1 for the facet normal we obtain

$$(3) \qquad I_{+X} = I^{\circ}_{+X1}\,(\hat{x}_{+1}\cdot\hat{u}) + I^{\circ}_{+X2}\,(\hat{x}_{+2}\cdot\hat{u})$$

where the superscript ° refers to a maximum value. Note that this equation is only true when $\hat{x}_{+1}\cdot\hat{u} \geq 0$ <u>and</u> $\hat{x}_{+2}\cdot\hat{u} \geq 0$. This condition just means that both facets of the quadrant are illuminated. If we assume that $I^{\circ}_{+X1} = I^{\circ}_{+X2}$ then Eq. 3 simplifies to

$$(4) \qquad I_{+X} = [2I^{\circ}_{+X1}\cos(22.5°)]\cos\alpha$$

similarly

$$(5) \qquad I_{+Y} = [2I^{\circ}_{+Y1}\cos(22.5°)]\cos\beta$$

Just before launch the measurements contained in column 2 of Table 3 were made on AMSAT-OSCAR 7.

facet	Maximum current (measured *)	2I°cos(22.5°) (computed)
+X1	775.9 ma	1,445 ma
+X2	787.8 ma	
+Y1	780.9 ma	1,445 ma
+Y2	783.0 ma	
-X1	777.8 ma	1,452 ma
-X2	794.0 ma	
-Y1	818.3 ma	1,498 ma
-Y2	803.2 ma	

Table 3. *BOL (beginning of life) values. Maximum values measured at normal incidence using a radiation source producing 1,380 watts/m² at the solar cells. The source spectral output approximates the sun in the spectral region where the solar cells are active.

We initially use the data of Table 3 in Eq. 4. However, the output of the solar cells continuously degrades in space so some correction should be made. Three methods for correcting the data will be discussed later.

Example 1. The following data was obtained from a telemetry frame:

$$I_{+X} = 836 \text{ ma.}; \quad I_{+Y} = 1{,}015 \text{ ma.}; \quad I_{-X} = 0; \quad I_{-Y} = 0.$$

Using Eq. 4 we obtain $\alpha = 54.7°$. Using Eq. 5 we obtain $\beta = 45.4°$. γ is computed from the relationship obtained from $\hat{u}\cdot\hat{u}$.

$$(6) \qquad \hat{u}\cdot\hat{u} = 1 = \cos^2\alpha + \cos^2\beta + \cos^2\gamma$$

$\gamma = \pm 63.8°$. The ambiguity in the sign of γ cannot be removed using only Morse code telemetry.

Example 2. The following data was obtained from a telemetry frame:

$$I_{+X} = 1{,}065 \text{ ma.}; \quad I_{+Y} = 238 \text{ ma.}; \quad I_{-X} = 0; \quad I_{-Y} = 220 \text{ ma.}$$

Since I_{+X} is greater than 788 ma. (see Table 3) both +X facets must be illuminated. Eq. 4 can therefore be used to obtain : $\alpha = 42.5°$. Eqs. 4 and 5 are only applicable when both facets of a quadrant are illuminated and, in this case, only one facet of the +Y quadrant and one facet of the -Y quadrant can be illuminated. Referring to Figure 1, the I_{+Y} channel telemetry actually is the value of I_{+Y1}, and the I_{-Y} channel telemetry is actually the value of I_{-Y2}. Using Eq. 2 and the appropriate normal vectors one obtains:

$$I_{+Y1} = I^{\circ}_{+Y1} \left[\cos\alpha\sin(22.5°) + \cos\beta\cos(22.5°)\right] \quad \text{and}$$

$$I_{-Y2} = I^{\circ}_{-Y2} \left[\cos\alpha\sin(22.5°) - \cos\beta\cos(22.5°)\right].$$

Using the value of α previously obtained either of these equations can be employed to find β. It is perhaps best to obtain two independent values for β and use an arithmetic mean. The value obtained in this manner is $\beta = 89.1°$. Finally, solving for γ as in Example 1 we obtain $\gamma = \pm 47.5°$.

QUESTIONS

1. From the data in Table 2, plot α, β, and γ as a function of time. What conclusions can you come to about spin rates?

See graph following Questions.

2. The notation used seems cumbersome at first. Can you devise a simpler notation? Be sure your notation can be used to uniquely specify each facet, each quadrant, maximum and instantaneous current values, etc.

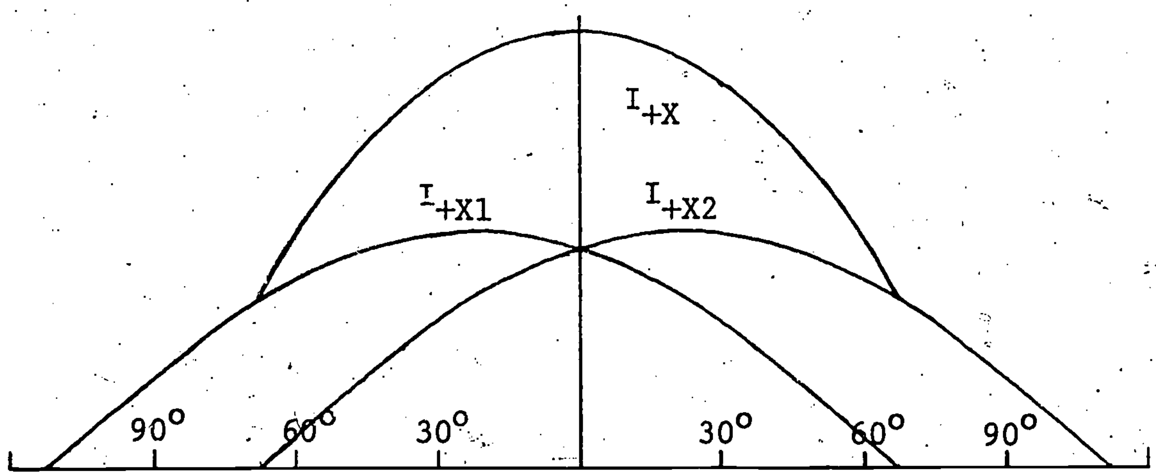
3. How can the values of Table 3 be corrected to allow for degradation of the solar cells?

   1. *Solar cell degradation results from a number of environmental factors. The rate depends on the orbit, solar activity, type of protective cover slips employed, etc. A very crude estimate for orbits below the Van Allen radiation belts (applies to A-0-7 and A-0-8) is that the cell efficiency will decrease 10% the first year and somewhat less each succeeding year.*

   2. *Assume that the maximum currents observed correspond to $21^{0}\cos(22.5^{0})$.*

   3. ~~*Assume that satellite Z axis is aligned parallel to earth's local magnetic field and compute γ. Use curve fitting techniques to solve Eq. 4 for α and Eq. 5 for β.*~~

4. Assume $\gamma = 90°$ (sun in XY plane); plot I_{+X1} for $-112.5° \leq \alpha \leq 67.5°$ and I_{+X2} for $-67.5° \leq \alpha \leq +112.5°$ on the same set of axes. On the same graph plot the expected I_{+X} telemetry channel current for $-112.5° \leq \alpha \leq +112.5°$. Note that Eq. 4 only holds for $-67.5° \leq \alpha \leq +67.5°$ in this case.

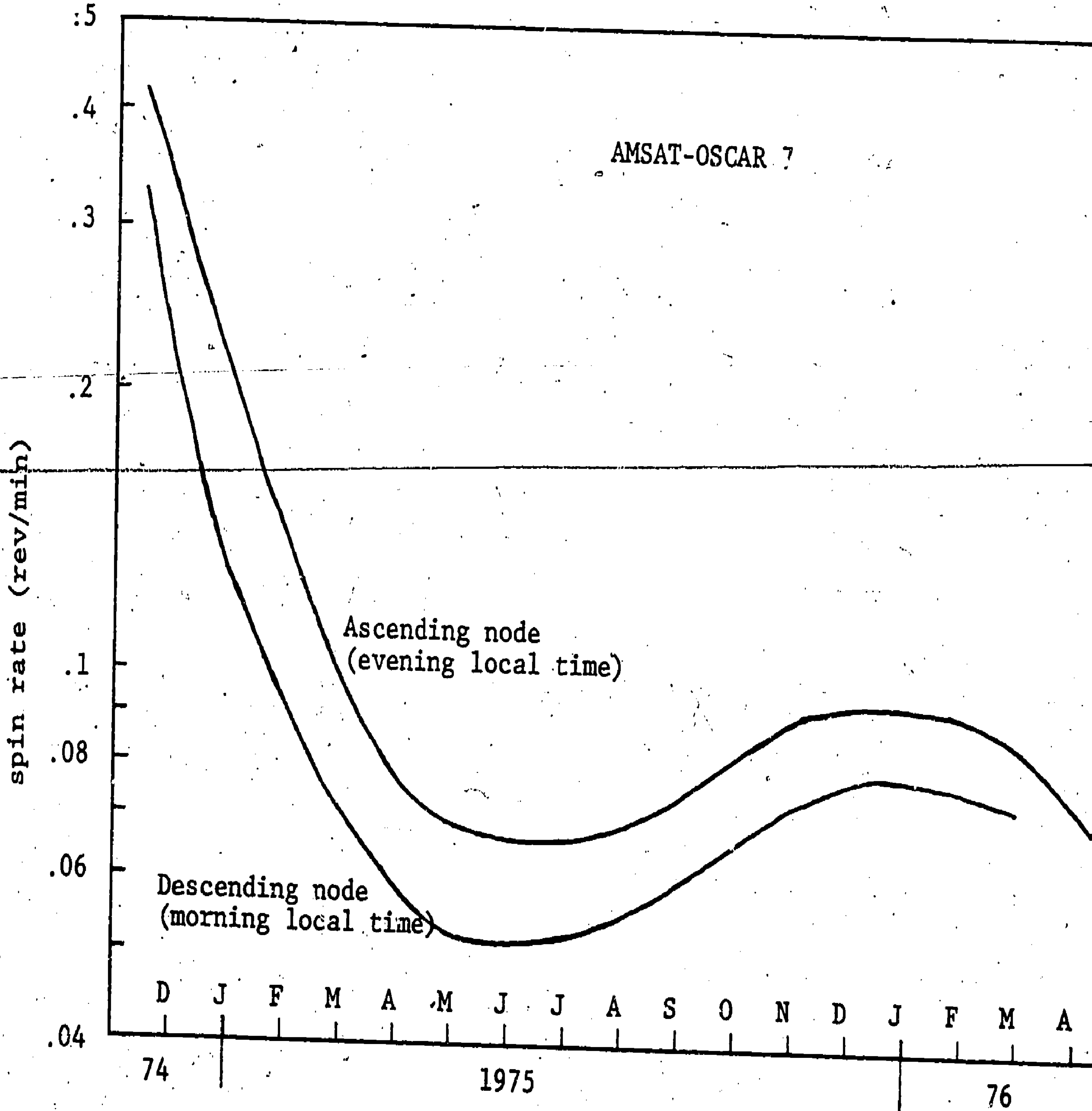


5. Write out the unit normal vector for each of the 8 facets (see Eq. 1).

6. In Example 2, solve the equations for I_{+Y1} and I_{-Y2} to obtain α and β. Compare your results to those obtained previously. Which method gives better results? Why?

7. The quadrants not facing the sun do not always read zero current. How do you account for this?

   Assuming that the electronics is working properly, they may be responding to reflected light from the earth. See Project SP 9.



Responses to question 1. can be used to extend the above graph which was provided by John Fox (WØLER).

PROJECT SP 7

SOLAR CELLS AND POWER BUDGET

<u>Objective</u>. The objective of this project is to calculate the average amount of power that AMSAT-OSCAR 7 can extract from incoming solar radiation. This calculation determines how much power we have available (power budget) for onboard satellite systems.

<u>Procedure</u>. The techniques used for the calculations in this project closely follow those in the procedure section of experiment SE 3. Our averaging technique is suitable for any satellite with an axis of symmetry where all possible orientations with respect to the sun are equally probable.

~~The power available from a planar panel of solar cells is given by~~

$$P = P_{max} \cos \varphi \quad \text{where } P_{max} = \eta P_o A_o$$

and P_o = solar constant,

A_o = surface area of solar cells,

η = solar cell conversion efficiency factor,

φ = angle between normal to surface and incoming radiation.

Solar cells used on satellites generally start out having an efficiency of about 11% (see Chapter III, p. 3-20). If normal precautions are taken to protect the cells from ultra-violet radiation by placing a thin sheet of glass (a cover slip) over them and if the satellite orbit does not encompass the Van Allen radiation belts, then the solar cell efficiency will generally degrade by about ten percent the first year and somewhat less in succeeding years. AMSAT-OSCAR 7 satisfies these requirements.

We now compute the average power output for one AMSAT-OSCAR 7 solar panel (each of the eight facets is treated as a single panel). Assume that the sun is in the satellite's xy plane and that all angles between the satellite x axis and the sun vector are equally probable.

$$P_1 = \eta P_o A_o \frac{1}{\pi} \int_0^{\pi/2} \cos(\theta) d\theta = \frac{1}{\pi} P_o A_o \eta$$

The contribution from all eight facets is therefore

$$P_8 = \frac{8}{\pi} P_o A_o \eta$$

Now, averaging over all possible sun vector angles with respect to satellite z axis (no solar cells on top or bottom plates) we obtain

$$P^* = \frac{8}{\pi} P_o A_o \frac{2}{\pi} \eta \int_0^{\pi/2} \cos\gamma \, d\gamma = \frac{16}{\pi^2} P_o A_o \eta = (2/\pi) P_8$$

Substituting appropriate values for AMSAT-OSCAR 7

$P_o = 1,380 \text{ watts/m}^2$

$A_c = (.358/8) \text{ m}^2$

$\eta = .08$

we obtain

$P^* = 8.1$ watts.

Introducing the eclipse factor, α, (about 0.8 when averaged over a year for AMSAT-OSCAR 7) which takes into account the fact that the earth sometimes shields the satellite from incoming radiation:

$P = \alpha P^* = 6.5$ watts.

In going from P_8 to P^* we assumed that all sun angles were equally probable. This resulted in a factor of $2/\pi$. Actually, an analysis of the magnetic attitude stabilization system would show that it favors desireable sun angles. As a result, the true averaging factor will be between 1 and $2/\pi$ raising P to about 8 watts for AMSAT-OSCAR 7.

Questions.

1. Compare the AMSAT-OSCAR 7 power budget to the average peak-transponder-power available.

2. How does the power budget affect the design of AMSAT-OSCAR 7?

3. How can our power budget calculations be empirically checked? Consider the information available by Morse code telemetry. For example, can information on (1) total solar panel current (when transponder is receiving heavy use and when transponder is not being used), and (2) battery voltage as a function of time during 24 hour mode B operation and during 24 hour mode A operation, aid us?

PROJECT SP 8

MEASURING THE SOLAR CONSTANT

Objective: To illustrate how AMSAT-OSCAR 7 can be used to measure the solar constant (solar power density at 1.49×10^{11} m from the sun incident on a plane surface perpendicular to the earth-sun line). The reader should be familiar with Experiment SE 4 as it forms the basis for this project.

Procedure. An important characteristic of solar cells (and panels) is that the output current at a specific incidence angle is directly proportional to the incident power density over a wide range of values: $I \propto P$. The proportionality constant may be determined by calibration. Table 3 in Experiment SE 4 contains the data for such a calibration. While a power density of 1,380 watts/m^2, corresponding to the actual solar constant, was used to compile Table 3, it is important to note that Experiment SE 4 could have been performed using some other calibration power density, for example 1,000 or 1,500 watts/m^2. Since the relationship $I \propto P$ is approximate, it is best to calibrate close to the actual solar constant value. Calibrations for actual space flights use a source modeling the sun as closely as possible. That's how Table 3 was produced.

For this project let us assume that the AMSAT-OSCAR 7 solar panels were calibrated before launch using a normal power density of 1,000 watts/m^2 and that the power source closely approximated the spectral energy distribution of the sun over the response range of the solar cells. Assume that, to 1% accuracy, all panels had the same output -- 573 ma. We now look at two methods of computing the solar constant from these assumptions. Both methods require the collection of a large data base of AMSAT-OSCAR 7 quadrant current measurements.

Method I. Assume that the largest quadrant current measurement in the data base was from the +X quadrant. Cos α (Eq. 4 in SE 4) must therefore be very close to 1 for this measurement so we obtain

$$I_{+X} \text{ (Maximum observed value) } \cong 2I^{\circ}_{+X1} \cos(22.5^{\circ})$$

solving for I°_{+X1}

$$I^{\circ}_{+X1} \cong \frac{I_{+X} \text{ (maximum observed value)}}{2\cos(22.5^{\circ})}$$

Using the direct proportionality of I and P and our calibration data we obtain

$$P_o \text{ (solar constant)} = \frac{I^{\circ}_{+X1}}{I_{+X1}\text{(calibration)}} \; P_{+X1}\text{(calibration)}$$

Method II. Method II illustrates statistical concepts and curve fitting techniques. Using our data base for the +X (or other) quadrant we can divide the range of observed currents into bins and count the number of observations in each bin. Using a probability model that assumes all orientations of the satellite equally likely we can compute the fractional number of measurements which we would expect in each bin -- the fractional number is equal to the solid angle subtended by the bin normalized by 4π sterradians. Using curve fitting techniques we can fit our measured distribution to the theoretical distribution to obtain I_{+X}(maximum). We then use this value to compute the solar constant as in Method I.

Questions.

1. Although the second method appears more sophisticated, it gives poorer results. Speculate as to why this might be so.

   The assumption that all angles are equally probable is not very good if data is only collected when the satellite is within range of a single ground station.

2. Why will these methods produce readings which are consistently a few percent low?

   Published data for the solar constant include non-thermal radiation which amounts to a few percent of the total. Our techniques assume a blackbody (thermal) energy distribution for the sun.

3. How would you design a calibration source to approximate solar characteristics?

PROJECT SP 9

DETERMINING THE EARTH'S ALBEDO

Objective: To illustrate how AMSAT-OSCAR 7 can be used to determine the earth's albedo (the fractional amount of incident solar energy reflected back into space) [4]. Before starting on this project one should be thoroughly familiar with the contents of experiments SE 3 and SE 4. The basic technique involves using data on currents from solar panels which are (1) facing a region of the earth illuminated by the sun and (2) shielded from direct solar radiation by the body of the satellite.

Procedure.

1. Obtain a good definition of planetary albedo and read about some of the techniques used to measure it.

2. Study the geometry involved in determining the albedo using a single planar solar panel as shown in Figure 1 where a number of vectors are introduced:

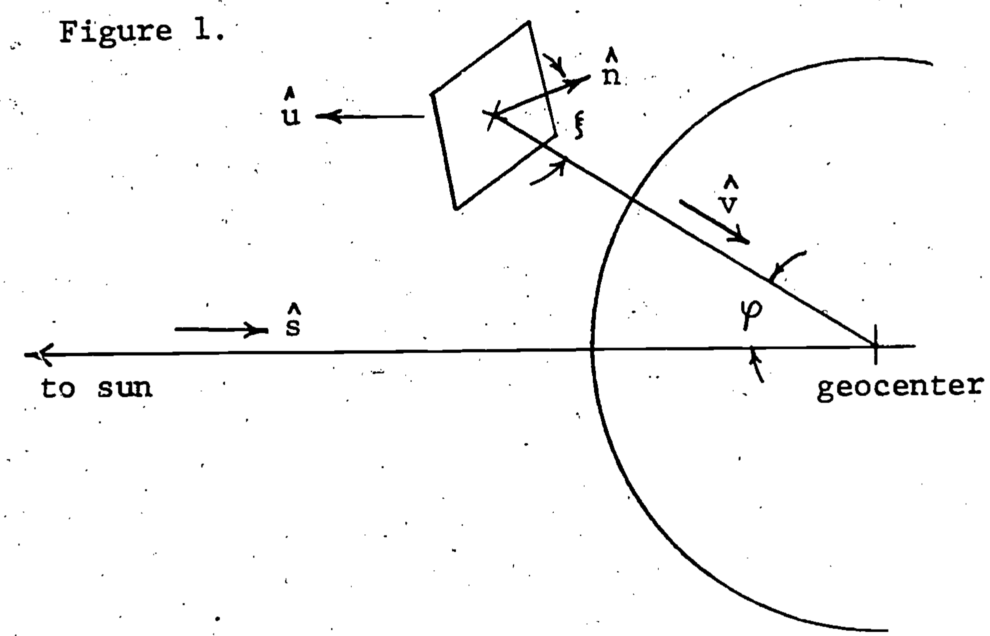
$\hat{u}$ (unit sun vector pointing from satellite to center of sun),

$\hat{v}$ (unit earth vector pointing from satellite to geocenter),

$\hat{n}$ (unit normal to solar panel),

$\hat{s}$ (unit sun-earth vector pointing from center of sun to geocenter).

Figure 1.



3. Using the approximations of geometrical optics, a solar panel illuminated by radiation reflected from the earth will produce a current directly proportional to

a) the cosine of the angle between the panel normal and the earth vector (ξ),

b) the cosine of the angle between the sun-earth vector and the earth vector (φ).

4. Use solar panel currents to determine spin rates in the manner of Experiment SE 4. The results will generally show the spacecraft to be undergoing between one and two complete revolutions per pass. The value should change by less than 5% in a week.

5. Once the spin rate is determined, the measurements needed to determine the albedo can be performed during a single pass. Choose a long pass where the portion of the earth "seen" by the satellite will be in sunlight during most of the time. Gather quadrant current data for the orbit. If the orbit is carefully chosen a plot of quadrant current versus time should show one or more large peaks which correspond to the quadrant being illuminated by the sun. The plot should also show a smaller current peak about midway between two direct sunlight peaks. The smaller peak is in response to reflected radiation from the earth.

6. Best results will be obtained when angles ξ and φ are close to zero. One can optimize the probability of this occurring by considering the following factors before choosing the pass during which the critical measurements will be made:

a) location of satellite pass,

b) time of year,

c) the direction of the earth's magnetic field at the satellite location (see Figure 3.6, page 3-17). Assume that the Z axis of the satellite is aligned parallel to the local terrestrial magnetic field vector.

6.5 PROPAGATION

This section contains four projects which focus on the propagation of radio signals. Experiments based on these projects are not limited to electrical engineering students studying propagation. By emphasizing various aspects of the projects experiments suitable for students in physics and geophysics (ionospheric composition, Faraday Rotation, Aurora, etc.) and Electronics Technology (antenna design, system design and implementation) can be devised. The specific effects discussed in this section are:

Faraday Rotation (SP 10),
Auroral Effects (SP 11),
Antipodal Reception (SP 12),
Extended Range Reception (SP 13).

PROJECT SP 10

FARADAY ROTATION

Objective: To show how AMSAT-OSCAR satellites can be used to study Faraday Rotation. The plane of a linearly polarized EM wave traveling through the ionosphere rotates about the direction of travel. The effect is called Faraday Rotation and it is due to the earth's magnetic field [5]. Ground stations using linearly polarized antennas can easily observe Faraday rotation on the 29.5 and 146 MHz AMSAT-OSCAR 7 beacons by its effect on downlink signal strength when using a linearly polarized antenna. Faraday rotation causes severe signal fading when the angle between the plane of polarization of the incoming wave and the plane of polarization of the receiving antenna is close to 90°. This experiment can also be performed using AMSAT-OSCAR 8 and the Soviet RS satellites.

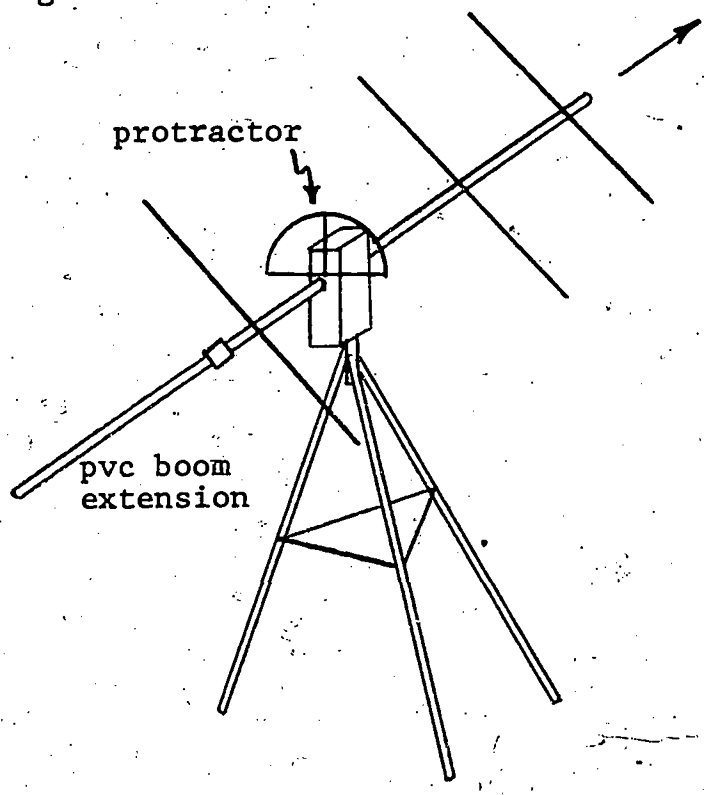
Procedure. Variations in downlink signal strength may be due to a number of factors including those listed in Table 1.

1. Satellite spin (changing orientation) and antenna pattern
2. Changing distance to the satellite (inverse power law)
3. Absorption in the ionisphere
4. Ground station antenna pattern
5. Faraday Rotation

Table 1. Factors affecting strength of received downlink signal.

Our problem is to separate Faraday rotation from the other factors listed. We describe one method of accomplishing this. Mount a linearly polarized beam antenna so as to permit rotation of the antenna about its axis (boom) as well as adjustment of azimuth and elevation. This can be done simply, as shown in Figure 1, by using a wooden surveying tripod at ground level. During a satellite pass the experimenter peaks received signal strength

Figure 1.



by (1) adjusting azimuth and elevation and (2) rotating the boom. A reference angle is read off a protractor mounted on the tripod and this angle is later plotted against time. Signal peaking is accomplished either by (1) using the receiver signal strength (S) meter in conjunction with a fast response time automatic volume control circuit (fast AVC) or by (2) ear with the AVC off.

Faraday rotation is frequency dependent -- the effect is more rapid at 29.5 MHz than at 146 MHz. However, a 3 to 6 element yagi for 146 MHz is much smaller and more convenient to work with than a 2 element full size yagi for 29.5 MHz so it is generally easier to perform this experiment at 146 MHz. The "circularly polarized" 146 MHz antenna aboard OSCAR 7 shouldn't adversely affect the experiment. Unless one is directly in line with the axis of the satellite antenna (an almost impossible situation for most ground stations) the 146 MHz beacon signal will have a strong linear component enabling us to measure Faraday Rotation. If a short yagi is available for 29.5 MHz the experiment should be simple to execute and should provide interesting results at this frequency. The construction of such an antenna is discussed in Section 6.8 -- SM 4. Faraday rotation periods of about 20 seconds have been observed at 29.5 MHz when the satellite is almost directly overhead. The period decreases as the satellite approaches the horizon. Different techniques must be employed to measure the rotation period when it decreases below about 10 seconds. One method is to record signal strength from two orthogonal linearly polarized antennas during a pass. A single receiver can be switched between the two antennas at a low audio rate.

In analyzing data from this experiment we see that factors 2 and 3 (Table 1) do not affect the plane of the received signals. Factor 4 will be relatively negligible when using a beam with a "clean" pattern. So the observed polarization variation is due to satellite orientation and Faraday Rotation. Results from Experiment SE 4 show the satellite's rotation period to be greater than 10 minutes. So, variations of polarization having a shorter period must be due to Faraday Rotation.

PROJECT SP 11

AURORAL EFFECTS

Objective: To (1) learn to identify radio signals affected by auroral activity and (2) study how auroral conditions affect radio propagation.

Procedure. Radio signals passing through zones of aurora activity acquire a characteristic distorted sound which has been described as: raspy, rough, hissy, fluttery, growling, etc. By listening to the AMSAT-OSCAR and Soviet RS satellite beacons as these spacecraft approach and depart from the polar regions one can learn to recognize auroral effects. Once one is able to recognize these effects a number of experiments are possible including the following:

1. The extent of the auroral zone, during a period when it is relatively constant, may be mapped [6];
2. The extent and severity of auroral effects can be compared at 29.5, 146 and 435 MHz.

PROJECT SP 12

ANTIPODAL RECEPTION

Objective: To identify instances of antipodal reception and to note the characteristics of the signal observed and the conditions under which antipodal reception occurs.

Procedure. Soon after Sputnik I was launched observers noticed that the 20 MHz signal from the satellite would often be heard for a short period of time when the satellite was located nearly antipodal to the observer. The phenomena was quickly dubbed the "antipodal effect" and a number of articles appeared in IEEE journals during the late 1950's discussing its causes. Antipodal effects were later observed on the 29.5 MHz beacon of OSCAR 5 [7].

The likelihood of antipodal reception is positively correlated with solar activity. During the sunspot maximum which is expected about 1980 it may again become common on the 29.5 MHz beacon signals. Although most occurences are felt to be the result of normal multihop propagation under the influence of a favorable MUF (Maximum useable frequency) signal strength is at times very high which suggests that a ducting mechanism may sometimes be responsible. It requires little effort to monitor the AMSAT-OSCAR 8 beacon frequency (29.402 MHz) when one's ground station is not being used for other purposes and chance observations of antipodal reception or other unusual propagation can prove very interesting.

Don't mistake AMSAT-OSCAR 8 for AMSAT-OSCAR 7 or the Soviet RS satellites. These satellites are easily identified by the beacon characteristics (frequency and telemetry format as described in Chapter IV).

PROJECT SP 13

EXTENDED RANGE RECEPTION

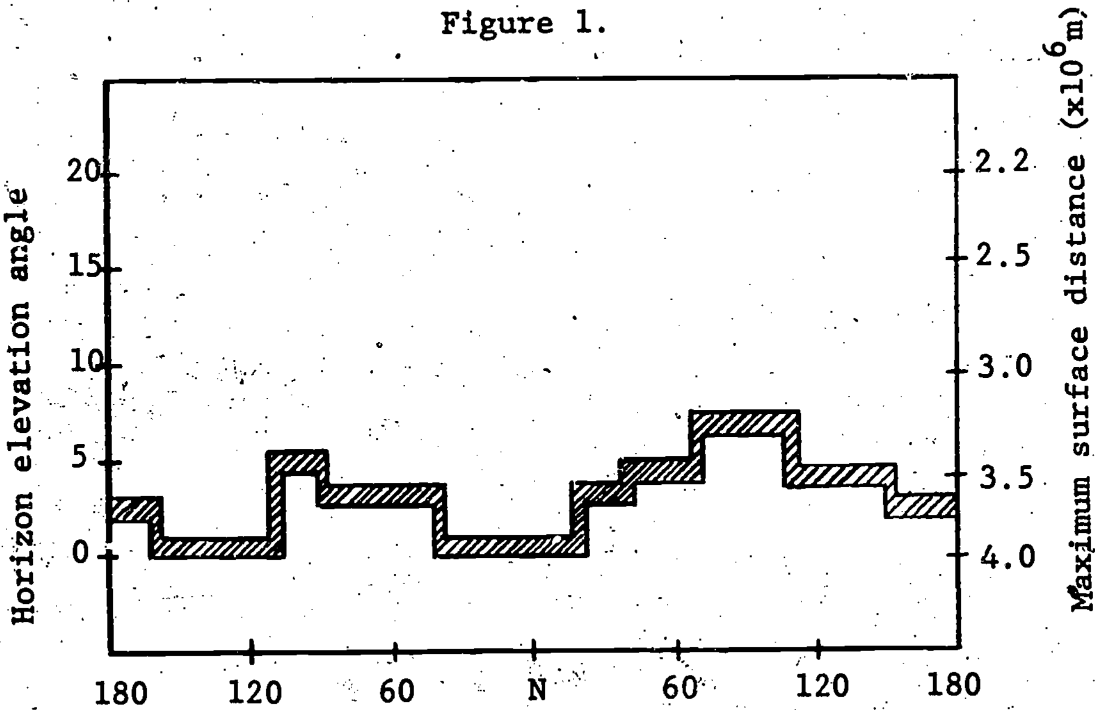
Objective: To identify instances of extended range reception and to determine the propagation mode responsible.

Procedure. If radio signals only propagated over line-of-sight paths it would be possible to predict AOS and LOS for AMSAT-OSCARs 7 and 8 to within a few seconds. One's radio horizon (minimum non-obstructed elevation angle vs. azimuth direction) at the receiving antenna could be established with a surveying transit. Using this data one could compute the maximum distance at which the satellite could be heard for each azimuth direction. However, observations would reveal AOS to frequently occur earlier and LOS to frequently occur later than this simple model predicts. These occurrences of extended range reception are due to various tropospheric and ionospheric mechanisms. Experimenters will find satellites an ideal tool for studying propagation.

In order to study extended range reception we must establish what "normal" reception means. This is essentially the same problem as determining one's radio horizon. The surveying transit approach mentioned earlier leads to problems. Does that big Oak qualify as an obstruction? Is that wood frame house an obstacle to radio waves? How about the house next door which is identical except for its aluminum siding? Because of these problems, it is best to establish one's radio horizon by processing data obtained from direct observations of AOS and LOS. One way of compiling a good data base is to require all students performing experiments SE 2, SE 3, and SE 4 to sign a log book which asks for orbit identification and accurate AOS and LOS times.

Processing the data base requires good judgment. The techniques will vary depending on the data available and one's objectives. One possible method is to draw a scatter plot using all observed data points on a set of axes suitable for a radio horizon graph. The horizontal (azimuth) axis is divided into bins, by eye, using the criteria -- the points in each bin appear relatively constant. The top 20% and botton 5% of values in each bin are temporarily discarded to allow for data collection errors and cases of extended propagation. Mean values are then used to construct a histogram as shown in Figure 1. Cross hatched horizontal areas reflect one standard deviation and vertical bands reflect an arbitrarily chosen 5° transition region.

Figure 1.



Now that the bounds for normal reception have been established, incidents of extended range reception can be identified. Determining the causes of extended range reception is an interesting problem in scientific detective work which can only be hinted at here. For example, if the number of incidents of extended range reception is greatest in a northern direction (in northern hemisphere) we might be observing auroral effects; if the incidents correllate positively with the presence of local stagnant air masses we might be observing a tropospheric effect; etc. For further information on propagation mechanisms which could produce extended range reception of satellite signals, see references [8, 9, 10, 11].

Propagation can also be studied by listening for the direct signals from stations transmitting to the satellite. Experiments along these lines led to the discovery of transequatorial (T-E) propagation at 146 MHz in early 1978 by stations in (1) Argentina, Puerto Rico, and Venezuela and in (2) Australia and Japan.

6.6 SATELLITE SYSTEM DESIGN

The tradeoffs involved in (1) the design of the OSCAR series of satellites and (2) the selection of their orbits provide a large number of interesting illustrations of system optimization involving real world constraints. Problems encountered include:

1. What orbits are most useful?

2. Starting from the initial orbits available, what orbits can we attain using a standard kick motor? This leads to the question of minimum energy required for orbit transfer.

3. For orbits and injection procedures under consideration, will the Van Allen radiation belts interfere with satellite electronics?

4. What are the satellite transmitter power requirements for a desired orbit? The answer depends on satellite antenna system which in turn depends on attitude stabilization system.

5. What are the solar cell and power budget requirements for a desired orbit and spacecraft shape?

6. What attitude sensors and stabilization schemes will be used? This directly affects possible antenna configuration and solar cell requirements.

7. Is hardware available? This leads to prototype hardware design and construction -- transponders, power storage and processing, computer control, sensors, etc.

As an example, we present a project which establishes criteria for the selection of an antenna for a satellite in an elliptical orbit. The problem was chosen because (a) it requires a minimum amount of prior specialized knowledge, (b) it is of interest to a varied audience (physics, mathematics and engineering students), and (c) it considers an aspect of satellite design which we haven't discussed elsewhere.

It should be noted that an original design project can lead to actual flight hardware or influence specifications for future satellites. For example, students at the University of Melbourne (Australia) built much of the hardware for AMSAT-OSCAR 5 and students at Trenton State College (N. J.) have built a prototype 435 MHz to 1,296 MHz MHz transponder for future OSCAR satellite use.

PROJECT SP 14

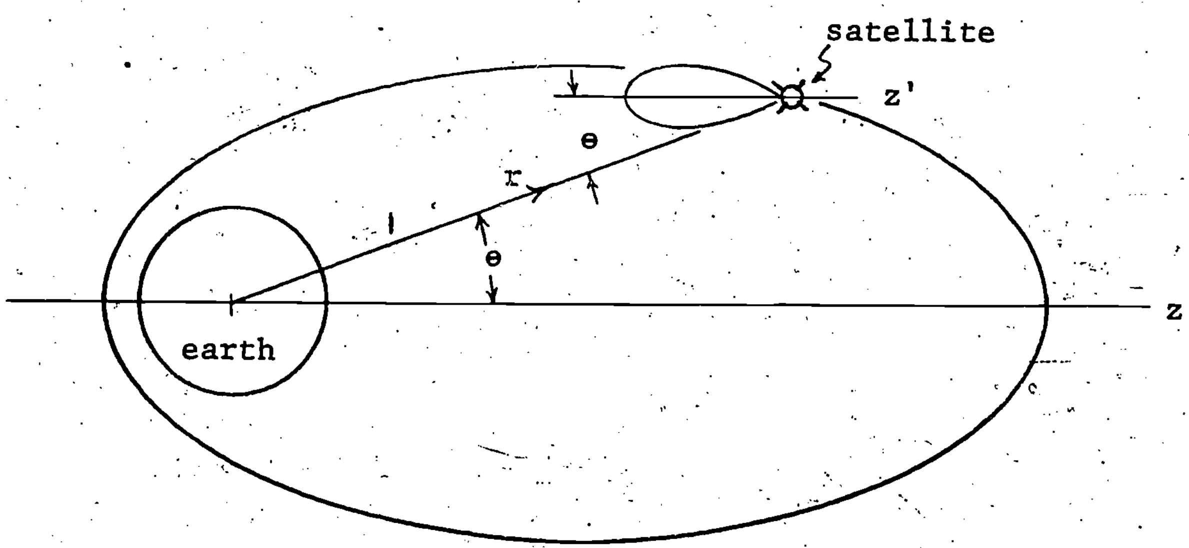
SELECTING A SATELLITE ANTENNA

Objective. This project illustrates how a study of the relationship between satellite antenna pattern and received signal strength at the ground can aid the satellite system designer.

Procedure. There are a number of factors that go into choosing the antenna(s) for a satellite. In this project the relationship between satellite antenna pattern and received signal strength at the ground is modeled [12].

Figure 1 shows the geometry of the orbital plane for a satellite in an elliptical orbit. Our objective is to compare the relative performance

Figure 1.



of various possible satellite antennas by computing received signal strength at the subsatellite point as a function of θ for each antenna.

Assumptions. We begin by making the following assumptions:

1. The satellite is spin stabilized with the spacecraft Z' axis lying in the orbital plane and pointing towards the geocenter at apogee. The Z' axis is parallel to the geocentered Z axis at all points on the orbit.

2. The satellite antenna pattern is symmetrical with respect to the Z' axis.

3. The satellite antenna pattern will have a shape given by $\cos^n(\theta)$ for $-90° \leq \theta \leq +90°$ and n given by one of (0, 1/2, 1, 2, 3, 4). There will be no radiation in the "back" (-Z') hemisphere.

4. Transmitter power aboard the satellite is constant.

Method. At any point on the orbit the radiation intensity at the surface of the earth, P*, is given by

(1) $P_n^*(\theta) \propto \frac{1}{(r-R_o)^2} U_n(\theta)$ where

(2) $r = \frac{a(1-e^2)}{1-e\cos(\theta)}$ and

R_o = radius of earth = 6.37×10^6 m
e = eccentricity of orbit
a = semimajor axis of orbit

(3) $U_n(\theta) = U_{max}\cos^n(\theta)$

Integrating the radiation intensity, $U_n(\theta)$, we obtain the satellite transmitter power, P_o, which is constant

(4) $$P_o = \int_{\phi=0}^{2\pi}\int_{\theta=0}^{\pi/2} U_n(\theta)\sin(\theta)\,d\theta\,d\phi = \frac{2\pi}{n+1}U_{max}$$

For an isotropic antenna $U_n(\theta)$ is a constant, U_i, so

(5) $P_o = 4\pi U_i$

Equating Eqs. 4 and 5 we obtain

(6) $U_{max} = 2(n+1)U_i$

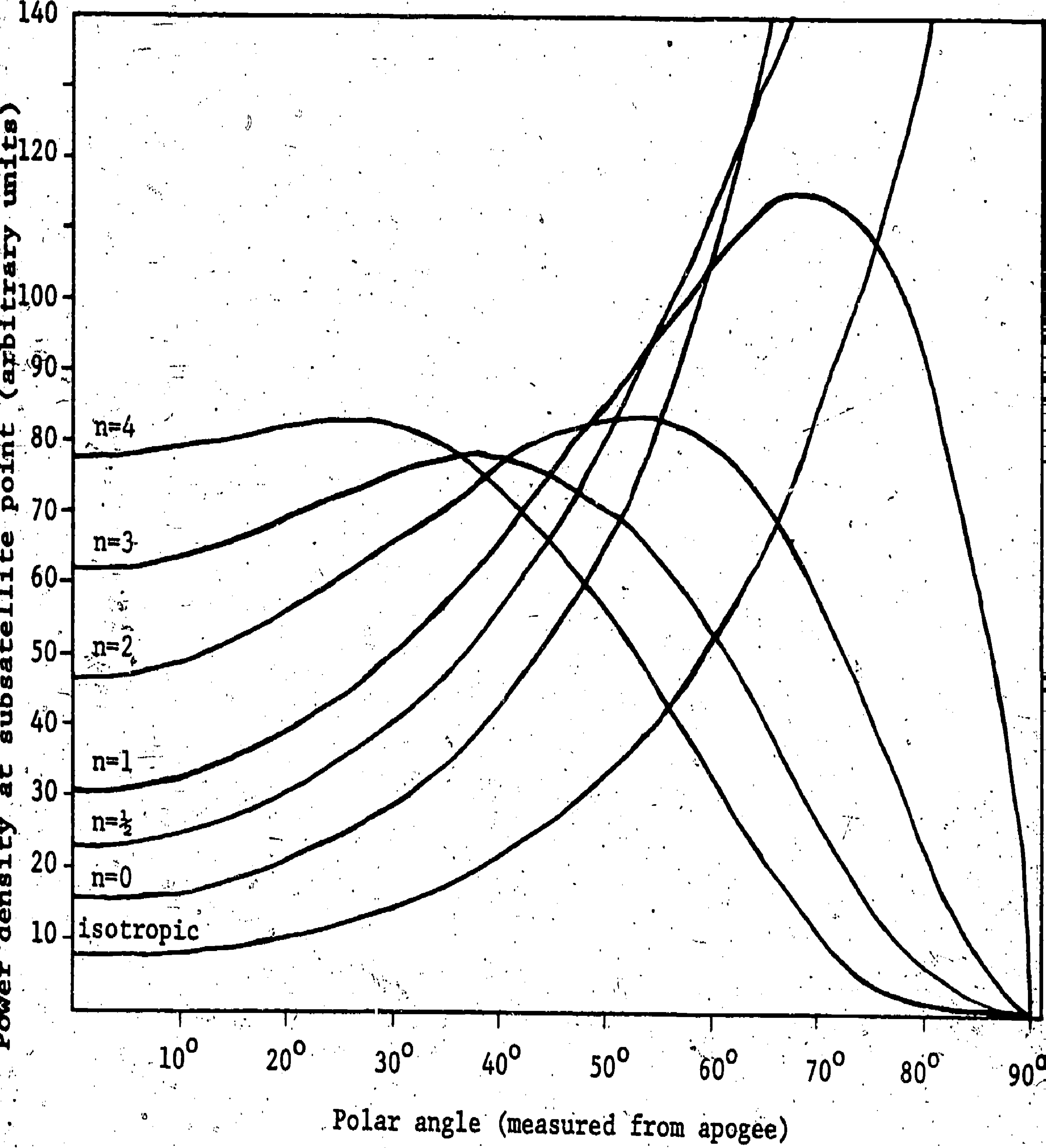
The quantity 2(n+1) is called the directivity of the antenna. For an antenna having an efficiency of 100% the terms directivity and gain are synonomous. Substituting Eq. 6 into Eq. 3 we obtain

(7) $U_n(\theta) = 2(n+1)U_i\cos^n(\theta)$

In summary, the desired solution is obtained by employing Eq. 2 and Eq. 7 to evaluate the righthand side of Eq. 1.

Question 1. We now look at a specific orbit being considered for a Phase III satellite -- eccentricity = .688, a = 3.93 R_o. Plot $P_n^*(\theta)$ vs. θ for n = (0, 1/2, 1, 2, 3, 4). Use arbitrary units for $P_n^*(\theta)$ and use 5° increments for θ.

See next page for graph.



Power density at subsatellite point (arbitrary units)
140
120
100
90
80
70
60
50
40
30
20
10
n=4
n=3
n=2
n=1
n=½
n=0
isotropic
10°
20°
30°
40°
50°
60°
70°
80°
90°
Polar angle (measured from apogee)

Question 2. Verify Eq. 4.

Isotropic: $U_i = P_o/(4\pi)$; $U_o(n) = U_{max}$

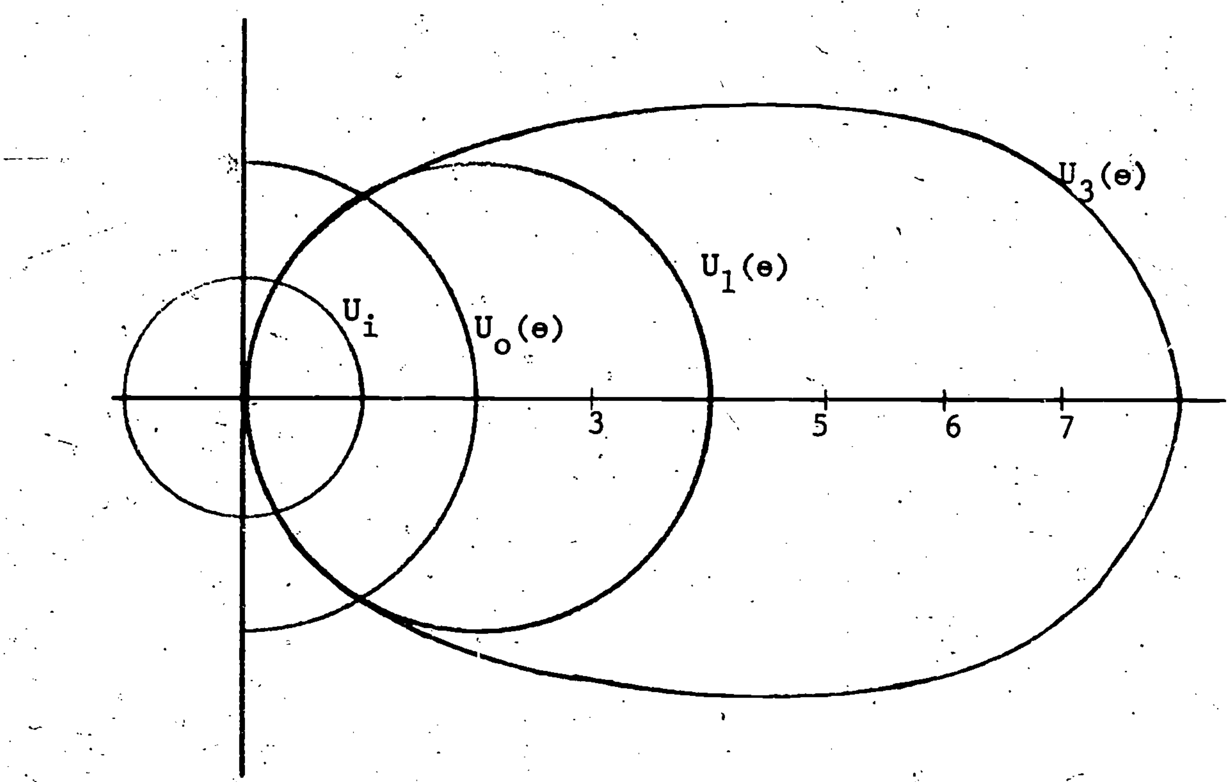
$$P_o = \int U_o(n) \cos^n(\theta) \sin(\theta)\, d\theta\, d\phi = U_o(n) \int_{\phi=0}^{2\pi} \int_{\theta=0}^{\pi/2} \cos^n(\theta) \sin(\theta)\, d\theta\, d\phi$$

Using integral tables (Pierce #277)

$$P_o = U_o(n) \int_0^{2\pi} \left[\frac{-\cos^{n+1}(\theta)}{n+1} \right]_0^{\pi/2} d\phi = U_o(n) \frac{1}{n+1} \int_0^{2\pi} d\phi = \frac{2\pi U_o(n)}{n+1}$$

$U_o(n) = \frac{n+1}{2\pi} P_o$ [QED] therefore $U_n(\theta) = \frac{n+1}{2\pi} P_o \cos^n(\theta)$

Question 3. Plot U_i and $U_n(\theta)$ using Eq. 7 for n = (0, 1, 3) on the same set of axes.



Question 4. Derive a general expression for the half power beamwidth of the antennas described by Eq. 7 and compute half power beamwidths for n = (0, 1/2, 1, 2, 3, 4).

$\theta = 2 \arccos (.5)^{1/n}$

n	Gain 2(n+1)	Gain dB_i	Half power beamwidth
isotropic	--	0	---
0	2	3.0	180
½	3	4.8	151
1	4	6.0	120
2	6	7.8	90
3	8	9.0	74.9
4	10	10.0	65.5

Question 5. Plans are to use an omni-directional antenna along the -Z axis of the first Phase III satellite. Spillover from this antenna will only illuminate a small part of the hemisphere centered about the +Z axis, perhaps up to 30°. Which of the antennas considered in Question 1 will give the best overall performance?

The n = 2 antenna produces a relatively constant signal at the ground for $-70° \leq \theta \leq +70°$. If satellite power is sufficient, this antenna will work well. One could argue that the n = 4 antenna will provide better results out to about 40° from apogee and that the satellite spends a very large percentage of each orbit near apogee. Also, the n = 4 antenna will still be providing stronger signals at 55° than the n = 2 antenna was providing at apogee. Since the choice is not very clear-cut, other factors such as available transmitter power and mechanical shape of the antennas will be an important consideration.

Question 6. Our computations for power density at the ground were for an observer at the subsatellite point. This greatly simplified our work but most ground stations will not be at this location. How useful is this model? Are there any simple changes which could be made to improve our model? How difficult would it be to solve this problem using an observer who always sees the satellite at a fixed elevation angle of 5°?

$P_n(\theta)$ (Eq. 1) involves two factors -- the first depends only on the distance r, the second only on the angle θ. The angular factor will be slightly high for some observers (not at subsatellite point) and slightly low for others. The distance factor will always be slightly high. Since a lower bound for $P_n(\theta)$ is most useful we could use the satellite radius instead of the altitude in the distance factor. It's not difficult to compute $P_n(\theta)$ for an observer who always sees the satellite at a 5° elevation angle (choose observer so $U_n(\theta)$ is minimum).

6.7 SATELLITE RANGING

In this section we discuss how "satellite ranging" can be used for a number of educational projects. Ranging data consists of information showing the distance between a satellite and ground station as a function of time.

PROJECT SP 15

SATELLITE RANGING

Objective: To introduce the reader to the various forms the ranging problem takes on and to discuss the experimental aspects of satellite ranging.

Procedure. The following discussion assumes that the reader is familiar with Chapter I of this text, Experiment SE 2 (Doppler Effect), and projects SP 1 (Slant Range) and SP 5 (Theoretical Doppler Curves).

Ranging problems involve three sets of parameters.

1. Range Curves. Range curves show slant range (ρ = distance between satellite and ground station) as a function of time;

2. Orbital Elements. Six orbital elements (parameters) are needed to describe a satellite in an elliptical orbit, four are needed for a circular orbit.

3. Ground Station Location. The ground station location is specified by latitude (ϕ_g) and longitude (λ_g).

A ranging problem involves finding one of the above sets of parameters when the other two are known. In practice the ranging problem generally occurs in one of the following two forms:

I. Navigation Problem. In the navigation problem we assume that range curves are available and orbital elements are known and that we wish to determine the two parameters which specify the location of the ground station.

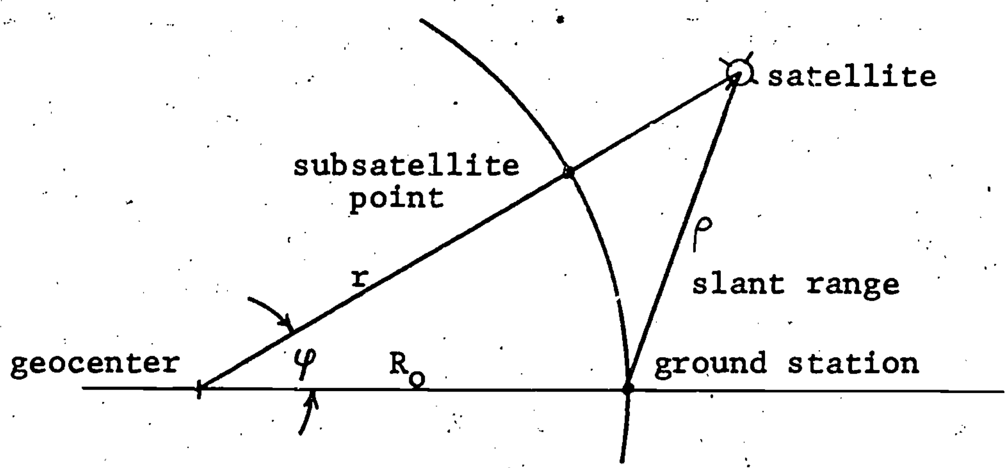
II. Tracking Problem. In the tracking problem we assume that range curves are available and the location of the ground station is known and that we wish to determine the six orbital elements.

We refer to the remaining form of the ranging problem as the

III. Theoretical Problem. In the theoretical problem we assume that the orbital elements and the location of the ground station are known and that we wish to calculate theoretical range curves. Each point on the range curve involves determining the value of one parameter.

The difficulty of each of the three forms of the ranging problem roughly parallels the number of variables to be determined. We turn first to III -- the theoretical problem -- because it is the simplest one.

Theoretical Problem. Problem III is the only form of the ranging problem that can be solved in closed form. Range curves can be computed, point by point, by following the steps outlined below.



1. Initialize θ (the angle that locates the satellite in the orbital plane).
2. Use text Eqs. 1.15, 1.20 and 1.21 to solve for the latitude and longitude of the subsatellite point.
3. Use text Eq. 1.14 to find the radial distance, r, between satellite and geocenter.
4. Use text Eq. 1.12,

   (1) $$\rho^2 = R_o^2 + r^2 + R_o r \cos\varphi$$

   to obtain the slant range.
5. Finally, $\cos\varphi$ is obtained from text Eq. 1.9

   (2) $$\cos\varphi = \sin\phi_s \sin\phi_g + \cos\phi_g \cos\phi_s \cos(\lambda_g - \lambda_s)$$

   where ϕ_s, λ_s are latitude and longitude of the subsatellite point and φ, ϕ_s and λ_s are implicitly time dependent. Steps 2-5 give us a single point on the range curve.
6. Increment θ by a convenient amount and return to step 2. Stop when the final value of θ is reached.

By following these steps one can plot theoretical range curves for any orbit of any satellite from any ground station. The accuracy of the curves is determined solely by the size of the θ increments chosen. We now turn to the navigation problem.

Navigation Problem. In the navigation problem two unknowns must be determined -- the latitude and longitude of the ground station. A simple graphical technique for solving the navigation problem involves selecting two points off a range curve and using Eq. 1 to obtain φ (essentially terrestrial distance) for each. These distances are used to draw circles on a globe about the corresponding subsatellite points. The two circles intersect at two points, one of which is the location of the ground station. The ambiguity is easily resolved by either (1) using a directional antenna to make a single crude azimuth measurement during the pass or by (2) drawing additional range circles along the subsatellite path -- the correct ϕ_g, λ_g will have a much smaller variance.

Another technique for solving the navigation problem suitable for satellites in circular orbits follows. Use the range curves to determine slant range and time at closest approach. Compute the location of the subsatellite point at this time and the heading (azimuth) of the satellite. The heading is most easily obtained by computing the position of the subsatellite one minute before and one minute after closest approach and then using standard terrestrial navigation methods to determine the bearing between these two points. The azimuth of the ground station with respect to the subsatellite point at closest approach is given by the satellite heading ± 90°. The terrestrial distance between subsatellite point and ground station at closest approach is easily calculated. The ground station location is therefore determined except for the sign ambiguity as in the previous technique.

Tracking Problem. The tracking problem involves determining six unknowns -- the six orbital elements. Six points on the range curve will yield (using Eqs. 1 and 2) six independent non-linear simultaneous equations. These equations can be solved using numerical techniques. For best results more than six points are taken from the range curve and curve fitting procedures are used. Although they're not necessary, range curves from ground stations at different locations will improve the accuracy with which orbital elements can be determined.

A similar approach to the tracking problem is to follow the procedure flow charted at the end of Project SP 5 (replace "Doppler curve" with "range curve" everyplace in the flow chart). The technique involves successive approximations -- one guesses the orbital elements and uses them to compute theoretical curves. The theoretical and actual curves are then compared and the results are used to produce a "better" guess of the elements. This loop is continued until the desired accuracy is attained. For additional information on these techniques see [13, 14].

Note: Experiments which require transmitting must comply with Federal laws cited in text section 5.6.

Experimental Considerations. Now that the three forms of the ranging problem have been discussed we turn to experimental aspects of obtaining range curves. The range to the satellite is obtained by transmitting a signal to the satellite and measuring the elapsed time (t) before it is returned by the transponder. As a first approximation the distance (ρ) to the satellite is given by

(3) $\rho = (1/2)ct$

where c is the speed of light. From Eq. 3 we see that the accuracy of our measurements depends on: (1) how closely the radio signal velocity is given by c and (2) how accurately the time interval is measured. We now examine each of these factors. The ionosphere tends to retard the velocity of signals propagating through it. The effect depends on the thickness of the ionosphere and on frequency. Errors as large as 500 km can occur at 29.5 MHz and low elevation angles. The problem is minimized by using the mode B or mode J transponders. Typical time intervals encountered will range from about 10 milliseconds for AMSAT-OSCAR 7 or 8 at closest approach to about .25 seconds for a Phase III AMSAT satellite near apogee. The transponder itself introduces a delay of typically 10 microseconds which can generally be ignored. The time delay in a selective communications receiver can amount to as much as 5 milliseconds and must not be neglected. The receiver delay time is usually directly measured and used to adjust the value of t. One method for determining receiver delay -- assuming mode B operation with a transmitter using a direct multiplier chain -- is to tune the 146 MHz receiver to 1/3 the transmitter frequency so that the low level stages can be monitored. A scope (preferably dual trace with delay line) is used to observe the transmitter output using a voltage probe (an AC probe from a VTVM is okay) and the receiver audio signal -- trigger on the transmitter signal. If the transmitter mixing scheme doesn't produce any output near 146 MHz, then time delay calibration can be accomplished by replacing the 146/29 MHz converter with a 432/29 MHz unit.

Actual ranging setups vary greatly in design. A triggered scope can be used to measure t with techniques similar to those employed for calibration. More accurate techniques have been described in [15] and [16].

At present there isn't any standard ranging system in use. Most systems are improvised using whatever equipment is available.

6.8 MISCELLANEOUS ACTIVITIES

In this section a number of activities involving, and topics concerning, satellites are briefly described. Each description is meant to introduce an idea -- comprehensive information is not provided. When available, references are given. All references, [N], are contained at the end of the chapter. This list of activities and topics can only begin to suggest the unending variety of experiments and projects involving satellites which can be used for science instruction.

SM 1. Construction of Satellite Hardware

Flyable hardware for future spacecraft is continually being solicited by AMSAT. Groups which have designed and constructed transponders, beacons, control circuitry, geophysical sensors, etc., are requested to submit the subsystems for testing and possible inclusion on satellites now in the planning stages. Once the Space Transportation System (Space Shuttle) is operational it is possible that an OSCAR satellite will be launched about every 18 months. This means that there will be a continual need for flight hardware. Past experience has shown that students get very involved in the design and construction of such hardware, especially when the odds are good that well-designed equipment will actually fly.

SM 2. Propagation Through the Ionosphere

In addition to the aspects of propagation discussed in section 6.5 there are other interesting topics worth investigating. These include: (1) unusual effects at AOS (acquisition of satellite) and LOS (loss of satellite) and (2) propagation velocity delay in the ionosphere. The AOS/LOS effects alluded to include: scintillations (large-amplitude, rapid variations) and signal splitting (signal appears to split into two or more components at slightly different frequencies). Some observers have speculated that splitting is due to signals arriving by different paths, each path having a slightly different Doppler shift. However, neither scintillations nor splitting are believed to be well understood. Propagation velocity delay was mentioned in conjunction with project SP 15 on ranging where it was regarded as an annoyance which introduced errors into our measurements. However, once the orbital elements of a satellite have been accurately determined, ranging can be used to obtain the signal propagation velocity which can, in turn, be used to infer information about the nature of the region of the ionosphere through which the signals are traveling.

SM 3. Satellite Monitoring

Hunting for downlink radio signals from unidentified satellites makes an interesting project. Once a satellite has been "found" an attempt may be made to determine the orbit and operating schedule and finally to identify the spacecraft. Sometimes this can be accomplished without ever understanding the content of the telemetry which is usually observed. Suggested frequency bands for listening to American and Russian non-amateur spacecraft are listed in the following paragraphs. Only downlinks below 500 MHz are listed because of equipment complexity and scarcity for higher ranges.

American satellites often use downlinks in the vicinity of

135.5 - 137 MHz (136 MHz band),

399.5 - 401 MHz (400 MHz band).

Commercial converters (for use with HF receiver) are available for the 136 MHz band making it easy to monitor. Converters for the 146/29 MHz range can usually be easily modified for the 136 MHz band. Adjustment involves retuning some resonant circuits and purchasing a new crystal for the injection oscillator chain.

Russian spacecraft often use downlinks in the vicinity of

15.008 MHz,

20.008 MHz,

121.50 and 121.75 MHz,

143.625 MHz.

The Russian frequencies are easy to monitor since the 15 and 20 MHz frequencies can be tuned on most HF receivers; the 121 MHz frequencies can be tuned on the inexpensive civilian aircraft monitor radios; the 143 MHz downlink can be tuned by stations set up to receive AMSAT-OSCAR 7 mode B. The 121 MHz links are often used for voice communications from manned spacecraft -- signal levels are very strong.

When monitoring, a 1/4 wavelength omnidirectional groundplane antenna works well for acquisition. When a possible satellite has been located, a directional antenna is often necessary to verify that the signals being received are from a non-terrestrial source. An axial mode helix is excellent for this purpose since a single helix can be designed to perform well over a wide range of frequencies including 120 - 150 MHz. Doppler shifts are also a clue as to the extra-terrestrial nature of the signals being received.

Monitoring projects can be used for a variety of educational purposes. At the simplest level they familiarize the student with communications equipment and satellite terminology. At a more advanced level they demonstrate how orbit determination utilizes all available resources. Information on operating frequencies of some American satellites can be obtained from [17].

SM 4. Antenna Design

The selection process for choosing the antenna(s) for one's ground-station serves as a good example of system optimization. Many of the tradeoffs were discussed in section 5.3 where simple antenna systems for the beginner were emphasized. Once one has experience with ground station operation, antenna improvements should be considered. Two areas of special interest are circularly polarized antennas and miniature antennas.

Circularly Polarized Antennas. Antennas which produce circularly polarized waves include:

a) axial mode helix [18],
b) TR-array (crossed dipoles above reflecting screen) [19],
c) crossed yagis [20],
d) miscellaneous [21],
e) quadrifilar helix [22].

The first three in the list produce circular polarization along a single axis. They were briefly discussed in Chapter V. In [21] Kraus discusses three types of antennas which produce omnidirectional circular polarization in a plane. Of special interest is the four in-phase 1/2 wavelength dipoles mounted around the circumference of an imaginary circle about 1/3 wavelength in diameter. If folded dipoles are used, a simple feed system could be improvised using four 1/2 wavelength sections of 300 ohm balanced line paralleled at the center and fed using a 1 to 1 balun with a slight mismatch. The quadrifilar helix (e) has the unusual property of producing nearly circular polarization in a hemisphere. It should be extremely useful for satellite ground stations.

Miniature antenna. It is possible to design highly directive antennas which are very small in terms of wavelength. These miniature antennas generally have a low efficiency which makes them unsuitable for transmitting. However, they can markedly improve reception when the received S/N ratio is being limited by noise arriving through the antenna. Miniature antennas are especially useful at 29 MHz where full size beams may be considered unwieldy. A 1/3 size two-element yagi for 29 MHz would have elements under 2 meters long and a boom of about 1.2 meters. General information on the construction of miniature beams is contained in references [23] and [24]. Two miniature yagis can be used for circular polarization as in [20].

SM 5. Antenna Measurements

Antenna characteristics (polarization, feed-point impedance, directivity, pattern and absolute gain) are very difficult to measure. As a result, realistic student laboratory experiments involving antennas are very scarce. Measurements of directivity, pattern and absolute gain (the characteristics treated here) are greatly simplified by using a signal source aboard a satellite.

Antenna measurements should ideally be made in outer space to guarantee that signals would travel by a single line-of-sight path. Although the ideal situation isn't possible we can closely approximate it by using a satellite beacon as a source while the antenna under test is used for reception at our ground station. Assume that the measurements are performed on a directive antenna when the satellite is at a high elevation angle. Any signals arriving at the antenna after being reflected off the ground will be way down on the directivity pattern. Multipath reception is therefore a much less serious problem when a satellite mounted beacon is used as a source than when a ground mounted source is employed.

Antenna directivity can be obtained by directly measuring the 3 dB beamwidth in the E plane (θ_E) and using the formula

$$D \approx \frac{40{,}000}{(\theta_E)^2} \quad [23].$$

This formula is only valid for directivity patterns which are nearly symmetric in the E and H planes. Most common antennas satisfy this requirement. Absolute gain measurements are made by comparing the antenna under test to a standard-gain antenna as suggested by the National Bureau of Standards.

Changes in signal level and direction of polarization may make it difficult to perform some of these measurements using Phase II (near earth) satellites. However, Phase III satellites near apogee should be almost ideal for the antenna measurements sketched above.

SM 6. Downlink Power Level

Standard techniques can be used to measure downlink power density levels from the satellite beacons. Since beacon output power is known we can use these measurements to:

1. evaluate various line-of-sight path loss models,
2. compute power loss due to propagation through the ionosphere,
3. deduce beacon power needed for other orbits.

SM 7. Satellite Beacons

The importance of the beacons for Doppler shift measurements, ground station antenna evaluation, and path loss computations has already been mentioned. In addition, the availability of a single well-characterized signal (frequency and power level) enables ground stations around the world to construct and accurately evaluate the performance of UHF and higher frequency equipment. The satellite beacon program produces a number of extremely important direct effects:

1. it encourages the development of equipment for higher frequencies,

2. it increases occupancy of higher frequency bands by serious experimenters which leads to better knowledge of propagation phenomena (for example, discovery of TE-propagation at 145 MHz in early 1978),

3. it increases the supply of trained personnel familiar with very high frequency techniques.

International regulations currently prohibit space beacons on many amateur bands (frequencies used for AMSAT-OSCAR satellites). As a result, beacons for many frequencies cannot be flown. A 2304 MHz beacon was placed on AMSAT-OSCAR 7 in the hope that a special temporary authorization for its use could be obtained. However the STA was not received. It would be of great utility to the world-wide educational community if telemetry beacons at 1,296 MHz, 2,304 MHz, and higher frequencies are permitted on future spacecraft.

SM 8. Orbital Transfer

Phase III AMSAT satellites will have an onboard propulsion system. For the first time, orbit changes will be possible. As a result, problems related to orbit change are now a reality for OSCAR designers. An orbital change may involve: change of period, change of eccentricity, change of orbital inclination, or a combination of the above. When one desires to make these orbital changes with the least expenditure of energy, it is often necessary to fire the propulsion system a number of times. The basic maneuver is referred to as a Hohmann transfer. A readable introduction to orbital transfer and rendezvous problems is contained in chapter 7 of reference [25].

AMSAT is primarily concerned with the following transfer problem. Using a kick motor capable of producing a velocity change of about 1,700 m/s (68 kg payload) which can only be fired once, what orbits are attainable from a given initial orbit? The solution of this problem leads to another problem. Which one of the attainable orbits is most desirable in terms of AMSAT's objectives? The tradeoffs here are very hard to quantify but it is generally agreed that an elliptical orbit with a period of 8 to 14 hours is probably best (see SM 9).

As mentioned in section 1.2 of the text, highly elliptical orbits are responsive to lunar and solar perturbations. These perturbations can cause large changes in the perigee altitude which may increase, decrease, or oscillate wildly. Orbital parameters will be carefully chosen to insure that the perigee does not decrease to the extent that the satellite will re-enter the atmosphere and burn up.

SM 9. Future Satellites: Orbit Selection

A number of constraints on orbit selection for future AMSAT spacecraft were mentioned in SM 8 including: initial orbits available, size of kick motor available, and effects due to solar and lunar perturbations. In this section we introduce two additional factors (1) operational features (from the point of view of those using the satellite) and (2) effects due to the earth's radiation belts.

Operational features. It is often thought that a synchronous orbit is most advantageous to the user. Let us briefly compare (1) a drifting synchronous satellite (satellite remains over the equator while longitude slowly changes) to (2) a satellite in a highly elliptical orbit like that being considered for Phase III (period = 11 hours, eccentricity = .69, inclination = 102°). Comparison criteria are necessarily subjective. For example, an analysis of a two-way communications link between New York and Frankfurt shows that communication would be possible about 17% of the time with a drifting synchronous satellite and about 70% of the time with the highly elliptical orbit when the spacecraft apogee is near the northernmost point. In addition, ground stations greater than 75° North or South of the equator will never be able to use the synchronous satellite. These selected comments only begin to show some of the different criteria which can be used to compare orbits. Synchronous orbits do have a number of well-known advantages so we have concentrated on the advantages of the elliptical orbit. Perhaps the reader may now deduce why the Russians have chosen highly elliptical orbits for their Molniya series of communications satellites. Both types of orbits have desirable features and, in the best of all possible worlds, both types of satellites would be available for educational use.

In order not to leave a false impression it should be stated that it requires a much greater amount of energy to achieve a synchronous orbit than the Phase III orbit used in the above example. AMSAT does not have the capability of putting a satellite into synchronous orbit using the "kick motor" approach being considered for the first two Phase III spacecraft.

Radiation belts. Soon after artificial satellites became a reality, scientists learned of the radiation belts surrounding the earth. Satellites passing through these radiation belts experience degradation of solar cell performance and an increased probability of catastrophic electronic system failure. Unless one is specifically interested in studying the radiation belts it is best to, in so far as possible, avoid them. The 1,400 km orbit of AMSAT-OSCAR 7 is safely inside the radiation zones. A synchronous orbit is safely outside. The elliptical orbits being considered for Phase III spacecraft pass through the belts twice each orbit.

Satellites studies have taught us a great deal about radiation damage so we can estimate the effects of such damage for different orbits. One important factor is in our favor. For the elliptical orbits being considered, the satellite is moving very rapidly near perigee as it traverses the radiation zones. Radiation damage is positively correlated with total exposure time. Calculations reveal that the total exposure should be within safe operating limits when planning for a six-year satellite lifetime. Projections of total radiation exposure depend upon the shape of the radiation zones. The zones cannot be adequately modeled using shells (radial symmetry). Reasonably accurate estimates of total exposure time require a three-dimensional model.

SM 10. Satellite Communications

The development of a satellite ground station with two-way communications capability can serve as a focal point for number of educational activities related to:

1. ground station design and evaluation,
2. a continuing program of ground station equipment development and evaluation,
3. system (satellite plus multiple ground station) evaluation.

The ground station will serve for ranging and other experiments, such as those discussed in SM 12, SM 13 and SM 14, requiring transmitting capabilities.

SM 11. Distance Between Two Satellites.

The distance between two satellites (for example, OSCARs 7 and 8) varies as a function of time. It is important to know when these satellites will be in close proximity because, when this occurs, it is possible to transmit uplink signals to OSCAR 7 (432 MHz) which are then relayed directly to OSCAR 8 (146 MHz) and returned to earth by OSCAR 8 (29.5 MHz). Computing the distance between OSCAR 6 and OSCAR 7 was assigned as a laboratory exercise in a course on computer programming with good results. This exercise was brought to my attention by C. J. Schmidt, Mathematics Department, Towson State College, Md.

SM 12. Satellite Communications Involving Satellite-Satellite Relay

The possibility of two-way communications between two groundstations using a satellite-satellite relay was mentioned in SM 11. Successful results of satellite-satellite relay experiments with OSCARs 6 and 7 are described in reference [26]. Satellite-satellite relays were observed with OSCARs 7 and 8 on the first day that OSCAR 8 was in orbit.

SM 13. Transponder Phase Distortion

In the Engineering Department of Trenton State College (N. J.) Dr. A. Katz is supervising students who are (1) making distortion measurements on the AMSAT-OSCAR satellite transponders and (2) analyzing how system nonlinearity is related to the generation of spurious signals and other aspects of transponder performance.

SM 14. Compound Doppler Shift

Experiment SE 2 outlined a Doppler shift model for a single link. The Doppler shift problem may be examined (experimentally and/or theoretically) for actual two-way communications involving an uplink and a downlink at different frequencies. A practical problem which might be considered follows. Assume two stations (A and B) are in contact via a satellite link and that A sets the uplink frequency so that downlinks from B and A coincide in frequency (as heard by A). Will B hear both downlinks on the same frequency? One might also try to analyze the Doppler shift effects occurring on a two-way communications link involving a satellite-satellite relay (see SM 12).

SM 15. Mobile Groundstation

It has been suggested that many land mobile radio services could benefit if repeaters were place aboard a network of low orbiting satellites instead of being mounted on towers. One approach to testing the feasibility of this idea is to build and evaluate the performance of a mobile ground station for use with AMSAT-OSCAR Phase II spacecraft. A sophisticated mobile station has been constructed by F. Merry of Albany, N. Y. The station has been used to send and receive EKG patterns from a moving vehicle to a fixed station at the National Institute of Health in Washington.

SM 16. Computer Access Via Satellite

AMSAT has received a special temporary authorization (STA) from the Federal Communications Commission which permits the use of ASCII code for satellite experiments. Ground stations can now use the satellite to interconnect microprocessors or to remotely access someone else's computer. It is conceivable that an on line time-sharing computer network utilizing AMSAT satellites may someday exist. Anyone contemplating ASCII experiments using OSCAR satellites should first contact AMSAT concerning the current status of the STA.

SM 17. Electro-Cardiograms (EKGs)

A number of techniques for encoding electro-cardiograms (EKGs) for transmission using a standard SSB transmitter have been tested. As mentioned in SM 15, EKGs have been successfully transmitted between a moving vehicle near Albany, N. Y. and a ground station at the National Institute of Health in Washington, D. C. via AMSAT Phase II satellites. For information on the encoding techniques and other aspects of these experiments see references [27] and [28].

SM 18. Emergency Locating Transmitters (ELTs)

Emergency Locating Transmitters (ELTs) operating on an international distress frequency of 121.5 MHz are carried by all aircraft in the U. S. and Canada and by aircraft of many other countries. The ELT is designed to automatically turn on at impact and it provides a signal for search aircraft to "home in" on. Recent experiments at the Communications Research Centre (CRC) of the Canadian Department of Communications using AMSAT-OSCARs 6 and 7 have shown that a satellite-aided search and rescue concept could reduce the costs and time associated with conventional methods for locating downed aircraft [29]. Similar experiments have been run by NASA at the Goddard Space Flight Center [30].

SM 19. Remote Geophysical Sensors

Low orbit satellites can be used to relay data from remote automated geophysical sensors. Groups are presently working on the design of automated data collection platforms using low power transmitters and omnidirectional antennas. Basic design decisions include the following. How is data recorded at the remote site? In what format is it transmitted? Will the transmitter be commanded by an internal clock or an external signal?

SM 20. Radio Interferometer

Radio interferometer principles can be illustrated by using two simple beam antennas at the ground station. Preliminary tests at 146 and 435 MHz with low altitude satellites and antenna spacings of three to five wavelengths show sharp, well defined nulls. Experimental suggestions are contained in reference [31].

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Chapter VI

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APPENDIX A

The addresses of equipment manufacturers, publishers and sources of information referred to in this text are listed below.

EQUIPMENT MANUFACTURERS

Amateur Radio Components Service (ARCOS), Box 546, East Greenbush, NY 12061

Cush Craft Corp., 621 Hayward St., Manchester, NH 03103

Data Signal, Inc., 2212 Palmyra Road, Albany, GA 31701

Drake (R. L. Drake Co.), 540 Richard St., Miamisburg, OH 45342

HAL Communications Corp., Box 365, Urbana, IL 61801

Hamtronics, Inc., 182-A Belmont Rd., Rochester, NY 14612

Heath Company, Benton Harbor, MI 49022

Hy-Gain Electronics Corp., 8601 NE Highway Six, Lincoln, NE 68507

International Crystal Mfg. Co., Inc., Box 32497, Oklahoma City, OK 73132

Janel Laboratories, 260 NW Polk Ave., Corvallis, OR 97330

Kenwood (Trio-Kenwood Communications, Inc.), 116 East Alondra, Gardena, CA 90248

KLM Electronics, 1600 Decker Ave., San Martin, CA 95046

Microwave Modules (see Texas RF Distributors)

Spectrum International, Box 1084, Concord, Mass. 01742

Texas RF Distributors, Inc., 4800 West 34th St., Suite D-12A, Houston, TX 77092

Vanguard Labs., 196-23 Jamaica Ave., Hollis, NY 11423

VHF Engineering, 320 Water St., Binghamton, NY 13902

Yaesu (Yaesu Musen USA, Inc.), 7625 E. Rosecrans Blvd., No. 29, Paramount, CA 90723

PUBLISHERS

ARRL (American Radio Relay League) Publications, 225 Main St., Newington, CT 06111
QST; The Radio Amateur's Handbook; The ARRL Antenna Book; OSCARLOCATOR; Getting to Know OSCAR; The Radio Amateur's VHF Manual; Specialized Communications Techniques for the Radio Amateur; Understanding Amateur Radio.

Communications Technology, Inc., Greenville NH 03048
Ham Radio Magazine; Satellabe

QST Magazine (See ARRL publications)

Ham Radio Magazine (See Communications Technology)

MISCELLANEOUS

AMSAT (Radio Amateur Satellite Corporation), Box 27, Washington, DC 20044

ARRL OSCAR Educational Programs Office (See publishers -- ARRL)

APPENDIX B

This appendix contains a short glossary of frequently encountered terms and abbreviations. Definitions focus on usage in the area of satellite communications.

altitude: The distance between a satellite and the point on the surface of the earth directly below it.

AMSAT: Radio Amateur Satellite Corporation. The construction of AMSAT-OSCARs 5, 6, 7 and 8 was coordinated by AMSAT.

AOS (acquisition of signal): The time at which a particular ground station begins to receive radio signals from a satellite.

apogee: The point on the orbit where a satellite is farthest from earth.

argument of perigee: The polar angle locating the perigee in the orbital plane -- measured counterclockwise from the line of nodes.

ARRL (American Radio Relay League): U.S. national organization of radio amateurs.

ascending node: Point on the satellite orbit (or ground track) where sub-satellite point crosses from southern hemisphere to northern hemisphere. Orbits are said to begin at the ascending node.

AU (Astronomical Unit): Mean sun-earth distance = 1.49×10^{11} m.

Codestore: A digital memory system aboard a satellite which can be loaded by ground stations for later rebroadcast in Morse or other codes.

descending node: Point on the satellite orbit (or ground track) where sub-satellite point crosses from northern hemisphere to southern hemisphere.

downlink: A radio link originating at a spacecraft and terminating at a ground station.

eccentricity: A parameter frequently used to describe the shape of an orbital ellipse.

equatorial plane: The plane containing the earth's equator.

ESA: European Space Agency.

ground station: A radio station on or near the surface of the earth used to receive signals from or transmit signals to a spacecraft.

inclination: The angle between the orbital plane of a satellite and the equatorial plane of the earth.

line of nodes: The line of intersection of a satellite's orbital plane and the earth's equatorial plane.

LOS (loss of signal): The time at which a particular ground station loses radio signals from a satellite.

NASA: National Aeronautics and Space Administration.

orbital elements: A set of parameters which completely describe an orbit. Six are needed for an elliptical orbit, four for a circular orbit.

orbital plane: The plane containing the satellite orbit.

OSCAR: Orbital Satellite Carrying Amateur Radio.

perigee: The point on the orbit where a satellite is closest to earth.

Phase I satellite: Label applied to early OSCAR satellites which were characterized by a short lifetime due to the fact that they did not use solar cells (OSCARs 1, 2, 3, and 5).

Phase II satellite: Label applied to OSCAR satellites characterized by long lifetimes and low altitudes (under 2,000 km). This group includes OSCARs 6, 7, and 8.

Phase III satellite: Label applied to post 1978 OSCAR satellites characterized by long lifetimes and high apogees (apogee height greater than 20,000 km).

reference node: The first ascending node of the UTC day for a given satellite.

slant range: Distance between satellite and a particular ground station (varies with time).

solar constant: Incident energy per unit time on a surface of unit area oriented perpendicular to direction of radiation at 1 AU from the sun.

sub-satellite point: Point on surface of earth directly below satellite.

TCA (time of closest approach): Time at which satellite is closest to a particular ground station during an orbit.

telemetry: Radio signals originating at a satellite which convey information on the performance or status of onboard subsystems. Also refers to the information itself.

transponder: A device which receives radio signals in a narrow slice of the spectrum, amplifies them, translates (shifts) their frequency, and retransmits them.

true anomally: The polar angle locating the satellite in the orbital plane -- measured counterclockwise from perigee.

uplink: A radio link originating at a ground station and terminating at a spacecraft.