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ADVANCED WEAPON SYSTEMS STUDY

PART III - ADVANCED MILITARY SATELLITES

VOLUME I

SUMMARY

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INTRODUCTION

This volume, the first of four, is a summary of the main results of an investigation of the feasibility of various applications of high-altitude satellites, especially satellites in 24-hour orbits. The applications discussed briefly in this volume and analyzed in detail in the other volumes include:

(1) Communication relay, especially by 24-hour satellites,

[redacted]

The feasibility of other applications, such as bombing, direct electronic or photographic reconnaissance, navigation, and guidance, was not investigated in this study because other techniques for accomplishing the same objectives appeared to be more effective or much simpler and cheaper.

Recent work by BMD, the Army Signal Corps, and Space Technology Laboratories, Inc. in the planning of the ARPA program for the development of communication satellites was greatly facilitated by the material in Volumes II, III, and IV, published in September 1958.

Since the publication of these volumes, the situation concerning the availability of prospective propulsion stages has changed considerably. Also, additional studies and analyses have been made in connection with the ARPA program. As a result, certain detailed portions of this report are no longer applicable, and therefore this summary volume is confined for the most part to the more general ideas presented in Volumes II, III, and IV. Certain corrections developed during subsequent studies are given in footnotes.

CHAPTER 1

COMMUNICATION SATELLITES

1.1 LONG-RANGE COMMUNICATION

Present-day systems and techniques for military communications do not adequately meet military needs for reliability and traffic handling capacity. The available wire lines have a low capacity and, in any case, they can be easily put out of action. Low-frequency radio transmission suffers from the strong atmospheric noise, and the antennas are notoriously inefficient. Medium frequencies can be used for long-range communication only at night. High-frequency signals are subject to severe disturbances by various solar emissions, such as flares; and additional high-frequency absorption, sometimes lasting for several days, occurs in the polar regions. Radio waves in the lower VHF range can be used for communication, but only at ranges limited to about 1400 miles. Longer ranges can be attained by series of short-range links, such as the currently used microwave relay links; but then the multiplicity of equipment decreases reliability, complicates maintenance, and raises costs.

Placing active, real-time, radio repeater stations in high-altitude earth satellite orbits will serve to avoid many of these difficulties and consequently will make significant improvements in the reliability and traffic capacity of military long-range communication circuits. As shown below, 24-hour satellites,* moving in circular orbits, are particularly well suited for this purpose--especially 24-hour equatorial satellites (which rotate with the earth and hence retain their positions above preassigned points on the equator) and 24-hour polar satellites (which move in planes containing the earth's axis). The altitude of a 24-hour satellite above the earth's surface is about 19.4×10^3 nautical miles.

* The term "24-hour satellite" is a short name for a satellite having the period of one sidereal day, which is the period of the earth's rotation relative to the fixed stars and is equal to 23 hours, 56 minutes, and 4.09 seconds.

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To determine the best positions of 24-hour communication satellites, we consider Figure 1-1a, in which the shading shows the areas where almost all of the communication signals of importance to the U.S. military forces originate and are received. It is apparent that high-density communication service must be provided (1) between North America-Greenland and Europe, (2) between North America and Asia, and (3) between the various points in the ZI. By contrast, the communication service required between the remaining areas can be a low-density service.

1.2 HIGH-DENSITY SERVICE

Figures 1-1b and 1-1c show the earth as it would be seen from 24-hour equatorial satellites stationed above points on the equator at the longitudes 30°W and 170°W. It is apparent that such a pair of satellites would provide for the requirements (1) and (2) listed above; moreover, as shown by the rectangles in these figures, the satellite antennas could have relatively narrow beams, with a consequent decrease in the required power. A 24-hour equatorial satellite placed anywhere between the longitudes 45°W and 130°W would "cover" all of ZI. Consequently, three equatorial 24-hour satellites could meet all the high-density traffic requirements.

The main advantages of equatorial 24-hour communication satellites are: (a) the desired coverage for high-density traffic can be obtained with only three such satellites, whereas the use of other orbits cannot reduce this number and will require a substantially greater number if the altitude is decreased appreciably; and (b) the "stationary" character of an equatorial 24-hour satellite reduces--if it does not completely eliminate--the problem of antenna tracking, both on the ground and in the satellite; this permits the use of larger antennas and yields a greater communication capability with the same transmitted power. (If proper directional antennas are used, the power required for a satellite depends on the ground area to be covered, but not on the altitude of the satellite.) As to propulsion, it is shown in Volume II that even a Thor booster with some modification can put a useful payload into a 24-hour orbit.

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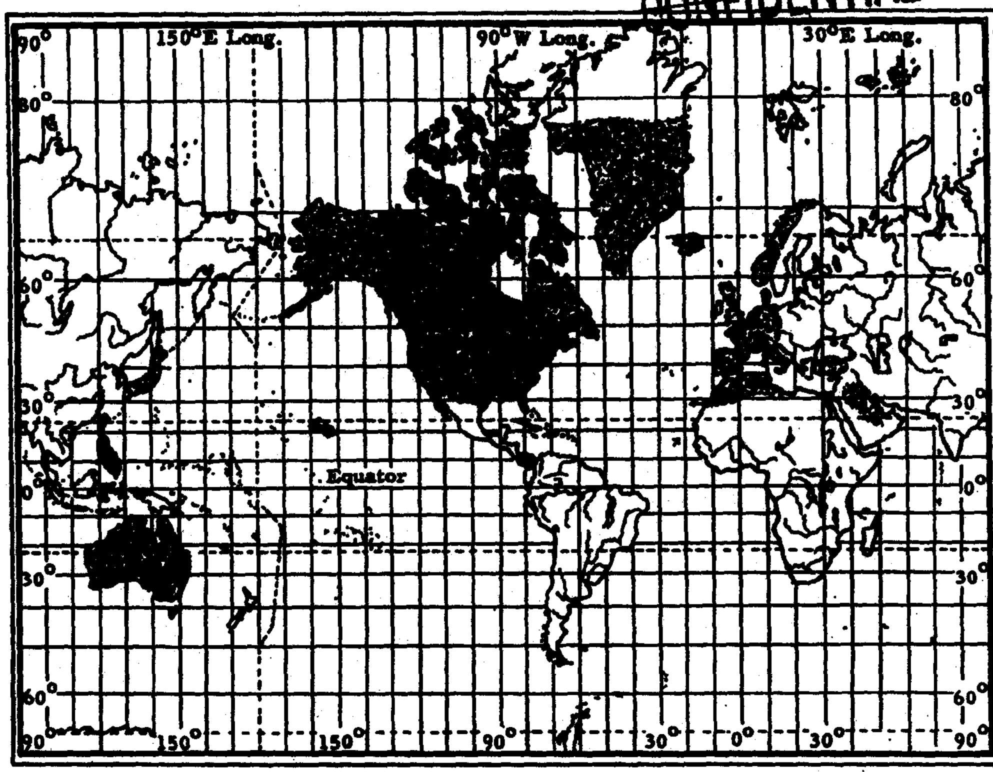

Figure 1-1a. Areas (shaded) Originating Great Majority of Communications of Military Interest.

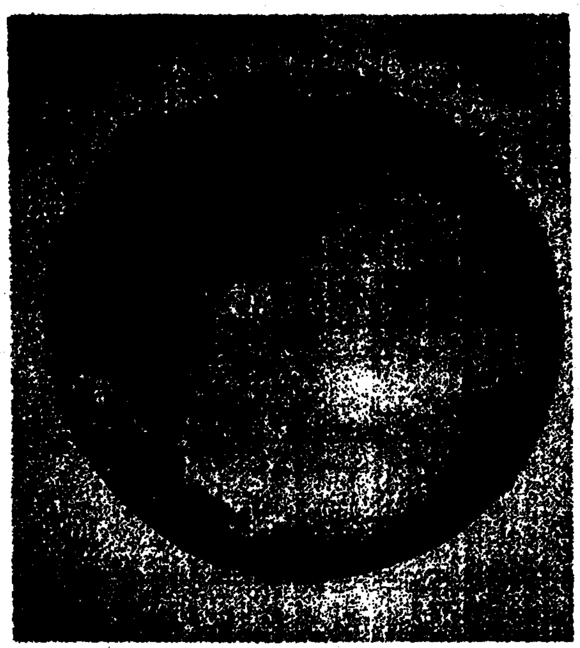

Figure 1-1b. Longitude 170°W.

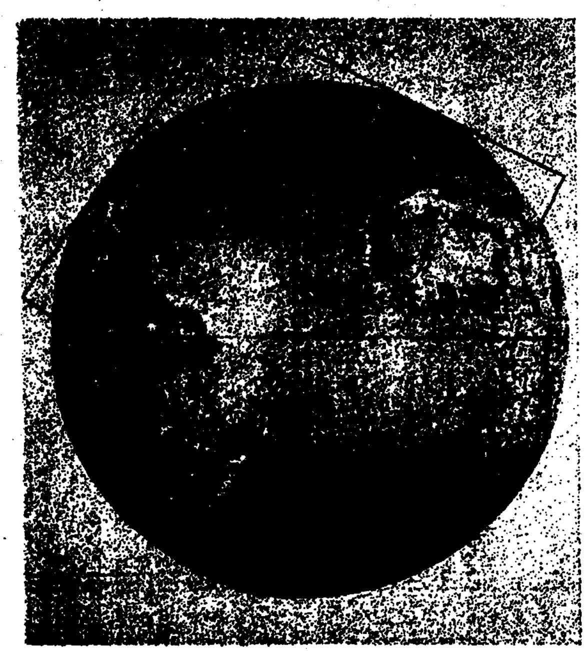

Figure 1-1c. Longitude 30°W.

Figure 1-1b and 1-1c. Earth as Seen from Satellite in 24-Hour Orbit (coverage areas outlined).

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1.3 LOW-DENSITY SERVICE

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The communication equipment of satellites intended for low-density traffic will be different from that required for high-density traffic. Thus, a much smaller bandwidth will be adequate, but the power transmitted by the satellite will be radiated to a substantially larger area. Also, communication with mobile units, such as aircraft, will have to be established, and, therefore, the dimensions (and gains) of the "ground" antennas will be considerably smaller than those which can be used for the high-density service.

The north polar regions, which are of the utmost importance for the SAC communication problem, cannot be "covered" by equatorial satellites. This fact plays a critical part in the choice of orbits for satellites serving the low-density areas.

The orbits recommended in this report for polar coverage are 24-hour polar orbits. The earth track of a satellite moving in such an orbit is a figure-eight, extending from pole to pole. The northern half of such a track is shown in Figure 1-2, for the case when the satellite crosses the equatorial plane at the longitude of 30°W (the satellite crosses the equatorial plane from South to North and from North to South at the same meridian, because during the 12 hours which it spends in either the northern or the southern hemisphere the earth turns through 180°). An important feature of a 24-hour satellite is that it keeps passing every 24 hours over the same points on the earth, which reduces the tracking problem considerably. If three such satellites, crossing the equator at 60°W and spaced 8 hours apart, should be put up, then at least one of them will be above the 30°N latitude at all times, and the three of them will provide continuous line-of-sight communication between the entire United States and certain large areas shown in Figure 1-2 and identified in the caption of that figure.

The polar orbits described above cover very little of Asia and Africa, and miss large portions of the Pacific Ocean. These areas can be covered by two equatorial 24-hour low-density satellites, one at 170°W longitude, over the Pacific, the other at 65°E longitude, over the Indian Ocean. However, the Indian Ocean satellite could not communicate with the United States directly, but only through a relay station in Japan or Europe.

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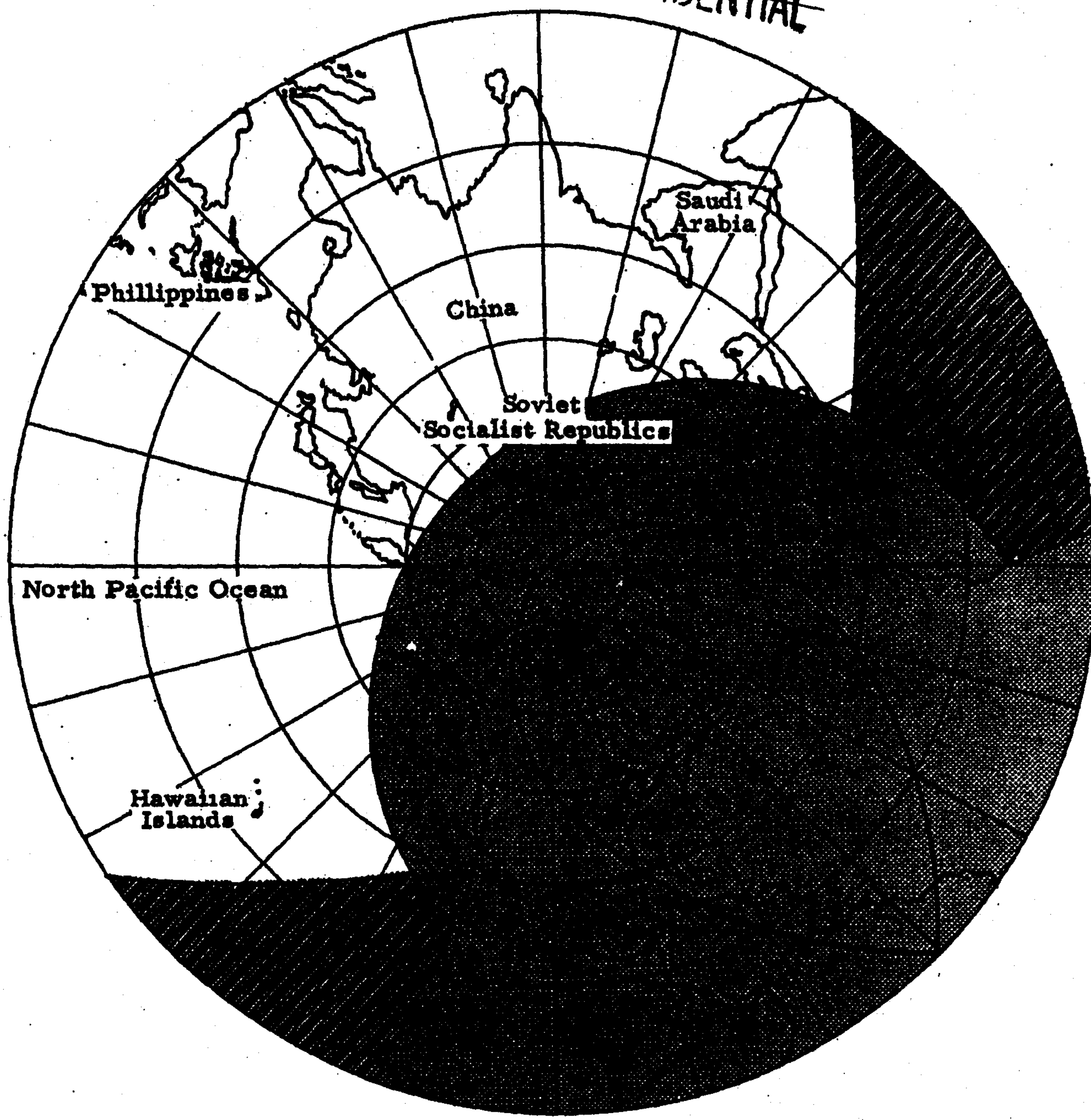

Area within continuous line-of-sight of recommended polar satellites when they are also within line-of-sight of central U.S.A.

Additional area within continuous line-of-sight of recommended polar satellites.

Figure 1-2. Continuous Coverage Areas for Low-Density Polar Relay Satellites.

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1.4 NUMBER OF SATELLITES

Both the high-density and the low-density traffic areas of the earth can be covered by the following eight 24-hour satellites.

<u>High-Density Traffic</u>

ZI to Europe	Equatorial, Longitude = 30°W
ZI to Asia	Equatorial, Longitude = 170°W
ZI Internal	Equatorial, Longitude = 45°W to 130°W

<u>Low-Density Traffic to Aircraft, etc.</u>

Polar, North America, North Atlantic, Europe, North Africa, and portions of North and South Pacific Ocean, South Atlantic Ocean, and South America	3 Polar Satellites crossing Equator at 60°W
Pacific Area	1 Equatorial, Longitude ≅ 170°W
Asia, Africa and Indian Ocean	1 Equatorial, Longitude = 65°E

Note that at 170°W both a high-density and a low-density satellite are required. One double-purpose satellite could be used instead, provided that an adequate payload could be put into a 24-hour orbit.

1.5 BANDWIDTH REQUIREMENTS

To establish and maintain a real-time satellite relay service is a complicated undertaking; therefore, it is important to use a conservative design and, if feasible from the standpoint of weight, to set up an extremely high-quality service (one with a high signal-to-noise ratio). A high signal-to-noise ratio will also reduce the susceptibility to enemy jamming, although jamproof operation will require, in addition, some technique for spreading the signal over the total available bandwidth to prevent the enemy from effective spot-jamming. Therefore, in this study the peak signal-to-noise ratio was set at 30 db, except for using a 20-db ratio in cases where the power (and satellite weight) required to provide a 30-db ratio for acceptable bandwidths would be excessive. The 30-db figure permits transmitting high-quality video and voice information, while providing a negligible error rate for normal digital signals, such as teletype. For the low-density relays, where the 20-db signal-to-noise ratio must be accepted, the limited bandwidth will be inadequate for transmitting video information, but the relays will still be comparable to wire-lines for transmitting digital signals.

For an excellent high-density service, a bandwidth of 50 megacycles (25 megacycles each way) between the terminals of an intercontinental relay is desirable. This bandwidth would provide, for example, for 5 video channels and 625 voice channels in each direction for a two-way system. Thirteen or more teletype or data-link channels could be substituted for a single voice channel.

For a minimal low-density traffic service the video is unnecessary, and there is no need for more than 5 to 10 per cent of the capability for voice, teletype, or data-link signals required for the high-density traffic. Therefore, the bandwdith for a low-density system needs to be only 100 kilocycles, which will provide either 5 high-speed data-link channels, or 25 voice channels, or over 300 teletype channels, or some combination of these channels.

The choice between traveling-wave tubes and klystrons as amplifiers for the low-density service will be determined by the availability of tubes and the desire to minimize the total power required for the system.

The bandwidths suggested above are merely design objectives; in fact, it may prove desirable to reduce the system signal-to-noise ratio to obtain increased bandwidth, or to increase reliability by providing redundant equipment.

1.6 POWER REQUIREMENTS

The maximum gain achievable with a satellite antenna is determined by the area to which it must transmit; and since effective satellite antennas may be limited to about the size of the satellite itself, it follows that the highest possible operating frequency should be used to minimize the antenna packaging problems. However, because amplifier tubes are not available at frequencies above 12 kmc, the frequency of 10 kmc was selected in this study for determining the power requirements.

Polyrod antenna arrays are the most suitable type of antenna for the satellite, where limitations upon size prevent the use of the larger, but more efficiently fed, parabolic dish antenna. The total weight of the largest array considered (16 rods, $\lambda/2$ in diameter, and 10 λ in length) should not weigh more than 10 to 20 pounds at S-band or 1 pound or so at X-band.

The "size" (equivalent area) is the only parameter of the ground antennas affecting the power requirements for satellite-to-ground transmission; and, therefore, by the reciprocity theorem, only the size of these ground antennas affects the power requirements for ground-to-satellite transmission.

1.6.1 <u>ZI Relay</u>

The West-to-East angle subtended by the United States at a 24-hour satellite placed at about 95°W longitude will be about 4 degrees; the North-to-South angle will be about 10 degrees. At 10 kmc this coverage would require an antenna array of only about 5 inches by 10 inches by 1 foot or so in depth, and would imply a gain of about 30 db with about a 50 per cent radiation efficiency.* The ground antenna would be a 60-foot dish with a beamwidth of 0.1 degree, which is compatible with the tracking accuracies and would provide a gain of about 63 db at 10 kmc.** The radiation efficiency of this antenna may be expected to be over 90 per cent.

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* A review of current antenna design practice indicates that, if the assumed beamwidths are obtained, the gains may fall short of the assumed gains by as much as 1.7 db. This would imply a decrease in bandwidth capability of one-third, an increase in satellite transmitted power of 50 per cent, or an acceptance of a decrease of 1.7 db in signal-to-noise ratio.

** The mechanical tolerances on a dish of this size may prove severe at 10 kmc, and if the tolerances cannot be met, the assumed gain will not be attained. However, the conclusions regarding system performance are not changed if the frequency is lowered, the decreased gain at the lower frequencies being exactly made up by the decreased path loss of the propagation path. The major system change required by a reduction in frequency would be an increase in the size of the satellite antenna required to produce the desired beamwidths; recent studies on antenna structures suitable for satellite vehicles indicate no severe problems in the production of lightweight foldable antenna structures having the required characteristics.

With a 30-db signal-to-noise ratio, the ZI communication relay service would require 5 watts per megacycle, or a total of 250 watts for a 50-megacycle bandwidth. Since there is little limitation on power for the ground equipment, it is advantageous to transmit at higher power levels from the ground, so as to decrease the amplification required in the satellite.

1.6.2 Intercontinental Relays

For a relay between eastern North America and Europe, antennas having beamwidths of about 2 degrees by 9 degrees are required. An antenna of about 6 inches by 2 feet by 1 foot or so would provide a gain of about 33 db. If the ground antennas are 60-foot dishes, and if the signal-to-noise ratio is to be 30 db, then a transmitter power of about 250 watts is required for two-way transmission of signals having a 50-megacycle bandwidth.

A relay between Asia and western North America requires similar antennas, with a third antenna, if desired, to serve Hawaii. A beamwidth of 1 degree will cover Hawaii; it can be provided by an antenna of about 4 feet by 4 feet by 1 foot without complicating unduly the attitude stabilization problem of the satellite. With a 30-db signal-to-noise ratio, the necessary transmitter power is about 10 watts. Hence, the total power required for this relay is about 260 watts, for a bandwidth of 50 megacycles.

1.6.3 Low-Density Traffic

For the low-density polar service, shown in Figure 1-2, the satellite antennas must have 16-degree beamwidths, since the total line-of-sight region subtends an angle of about 16 degrees at a polar 24-hour satellite. The necessary parabolic dish has a diameter of only about 5 inches and a high radiation efficiency. Since the attitude of the satellite will be stabilized, fixed antennas may be used to obtain the desired coverage. However, the ground antennas must track the relay satellite. In addition, "ground" stations may include aircraft, and such mobile antennas will be small. A 15 by 15 by 13 inch aircraft antenna will have a beamwidth of about 3 degrees at 10 kmc, which is wide enough to permit tracking from aircraft and which will provide an antenna gain of about 33 db.

Under these conditions, 2.2 kilowatts are required to provide a 30-db signal-to-noise ratio in the 100-kilocycle bandwidth assumed for the low-density service. But this power may be reduced to 220 watts if the signal-to-noise ratio is reduced to 20 db. If more power is available, a lower signal-to-noise ratio could also be used to increase the bandwidth.

The power required for a low-density 24-hour equatorial satellite placed over the Pacific or over the Indian Ocean proves to be essentially the same as that computed above for a relay moving in a polar orbit.

1.7 "INTERIM" COMMUNICATION SYSTEMS

The satellite relay system described above is an initial global system, in the sense that, as the achievable payloads increase, it would be replaced by a higher-capability system. However, the time required for developing even the initial system is around four to five years, and hence it is useful to investigate some interim systems, which would have a lesser capability but could be developed in half that time by making full use of the presently available devices and techniques.

Two general classes of "interim" systems have been investigated: (1) systems providing relay service from point to point or, at most, from a restricted area to a restricted area, and having a bandwidth of a few megacycles; and (2) systems for relay service between any two points within a large area with a bandwidth of a few kilocycles.

Three point-to-point systems have been investigated, which provide a reasonable range of capability and from which the requirements for this class of systems as a whole may be estimated. They include: (a) point-to-point simplex (one way at a time, but reversible) transmission of FM multiplexed signals with a nominal information bandwidth of 5 megacycles; (b) point-to-point duplex (simultaneous two-way transmission) of FM multiplexed signals of the same nominal bandwidth; and (c) simplex transmission from a restricted area to a restricted area, with a nominal total bandwidth of 2.5 megacycles.

One relay system for service between points within a large area was studied. Such a system should permit communication between mobile stations, including aircraft. Therefore, voice transmission is desirable, although a high-speed teletype capability might be acceptable. The system that was investigated uses the 200 to 400 megacycle UHF communication band, with a bandwidth of 3 kilocycles. Dipole antennas are used to eliminate the need for antenna tracking, with the result that the UHF equipment now available for aircraft communications may be used in its entirety, and new development is necessary only for the satellite-borne equipment. The UHF voice relay would be similar in its main features to the ZI relay for the initial systems. However, the UHF communication band permits the use of more conventional amplifier tubes and design techniques. The estimated weight and power requirements for the UHF voice system are about 20 pounds and about 258 watts.

Some parameters of the initial system are compared with those of an interim system in Table 1-1.

Table 1-1. Communication Parameters of Relay Systems.

Low-Density Traffic

	Interim, UHF Voice	Initial
Continuous Coverage (3 polar, 1 Pacific)	To aircraft over Polar Regions, Northern Oceans, Europe, North America	
Bandwidth	3 kc	100 kc
Signal-to-Noise Ratio	15 db	20 db
Satellite Antenna	dipole	5-in. dish
Required Transmitter Power	135 watts	220 watts
Total Satellite Communication Power	258 watts	1725* watts
Satellite Communication Weight	20 lb	73* lb
Ground (or aircraft) Antenna	dipole	15-in. sq array

High-Density Traffic

	Interim	Initial
Typical Coverage of One Antenna	Britain	Greenland to near East Russia to Spain
Bandwidth	5 mc	50 mc
Signal-to-Noise Ratio	20 db	30 db
Satellite Antenna	4 ft sq array	6-in. by 2-ft array
Required Satellite Transmitter Power	5 watts	250 watts
*Total Satellite Communication Power	210 watts	1760 watts
*Satellite Communication Weight	86 lb	100 lb
Ground Antenna	30-ft dish	60-ft dish

* Using periodically focused tubes.

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CHAPTER 3

ASCENT TO 24-HOUR ORBITS

3.1 ASCENT TRAJECTORIES

Equatorial 24-hour orbits. The trajectory recommended for ascent from a hypothetical equatorial launch site is described in Figure 3-1. The first portion of powered flight begins on the launch stand and ends when the vehicle is moving horizontally with the speed of about 25,000 feet per second, at the altitude of about 140 nautical miles. This is followed by a coast in the 140-nautical mile circular orbit. When the vehicle is at the proper longitude for transfer to the desired position in the 24-hour orbit, a "perigee burning" adds about 8000 feet per second to its speed, and sends the vehicle into the transfer ellipse (more precisely, into the minimum-energy transfer ellipse, called the Hohmann ellipse). The missile coasts in this ellipse through 180°, when the "apogee burning" adds about 5300 feet per second to its speed, and injects it into the circular 24-hour orbit. The longitude in which the satellite will hover above the equator is determined by the length of the coast in the low-altitude orbit. (The terms "perigee burning" and "apogee burning" are used because they refer to the burning at the perigee and apogee of the transfer ellipse.)

The sequence of events recommended for launching an equatorial 24-hour satellite from Cape Canaveral (latitude 28.5°N), illustrated in Figure 3-2, runs as follows: The vehicle is launched eastward into a circular 140-nautical mile orbit (inclined 28.5° to the equator) where it coasts until it crosses the equatorial plane at a point determined by the desired ultimate position of the satellite relative to the earth. At this crossing point the perigee burning sends it into a transfer ellipse, whose plane is inclined 28.5° to the equator. When the vehicle reaches the equatorial plane (after coasting 180° in the elliptic orbit), the apogee burning provides a thrust of the correct magnitude and direction to inject the missile into its final circular 24-hour equatorial orbit. Comparison of Figures 3-1 and 3-2 shows that in the case of launch from Cape Canaveral, the extra speed to be added at apogee for injection into the equatorial plane is 6013 - 4826 = 1187 feet per second. Since Cape Canaveral turns with the earth

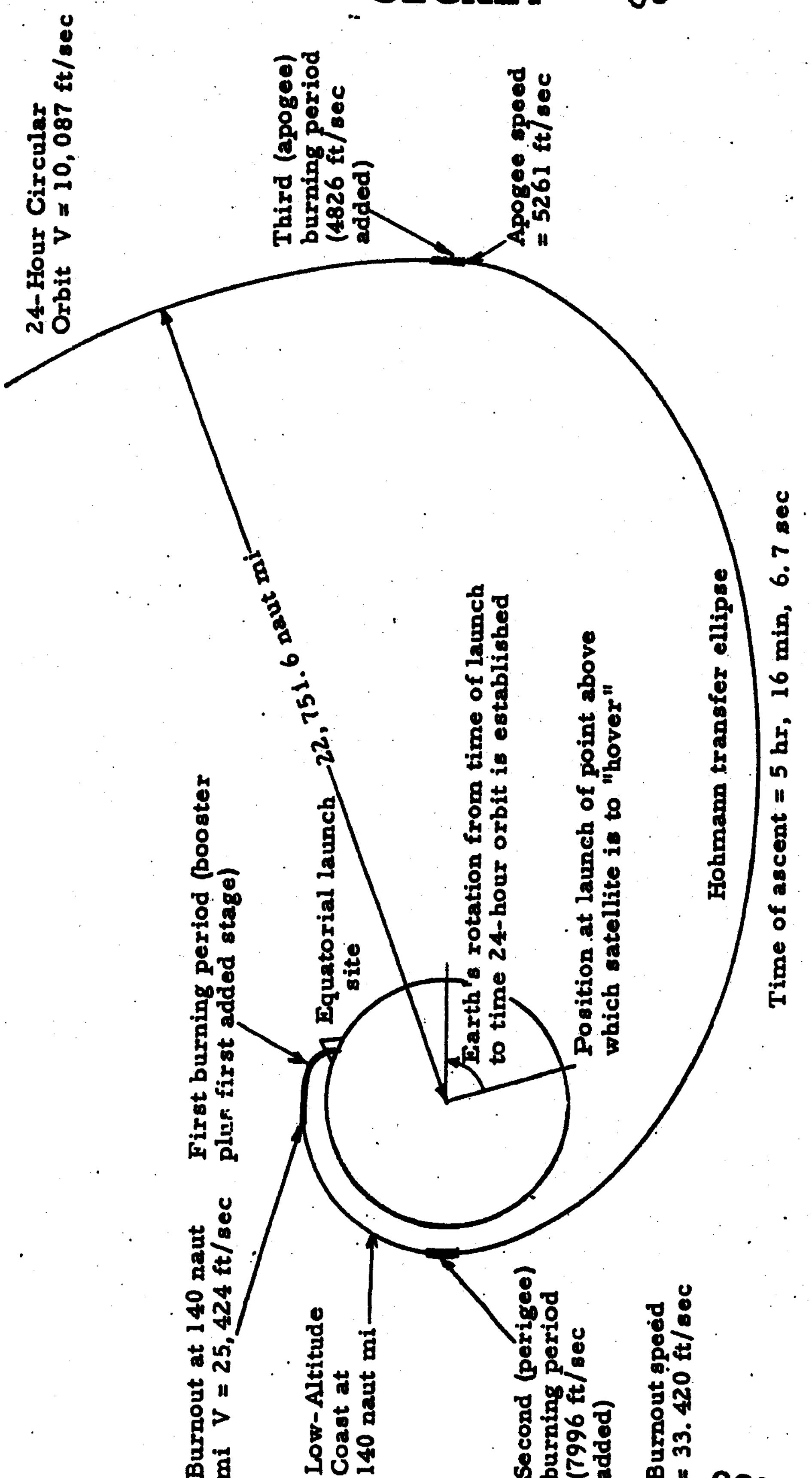

Figure 3-1. Total Trajectory for Satellite Launched on Equator.

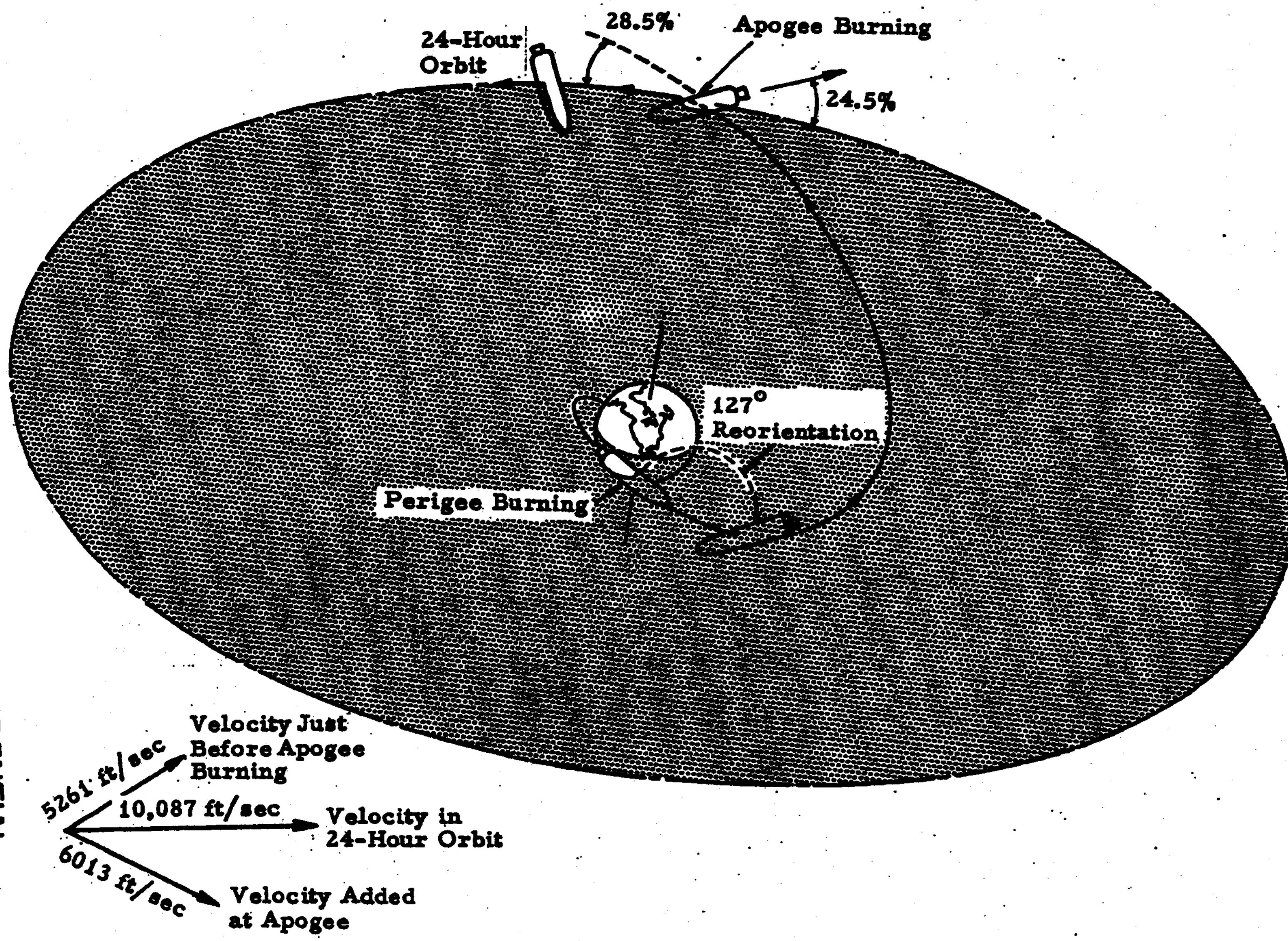

Figure 3-2. Flight Path and Attitude of Vehicle During Ascent Coasting, and Apogee Burning Conditions (Perigee Burning on Southward Crossing of the Equator).

slower than a point on the equator, the total penalty is 1187 + 184 = 1371 feet per second.

If the perigee burning is scheduled for the time of the first crossing of the equator, the satellite will take up a station on the longitude of Singapore. Other final longitudes can be attained by longer coasts in the low-altitude orbit or by departing from the strictly eastward launch.

While the vehicle ascends in the transfer ellipse, it moves relative to the rotating earth as shown in Figure 3-3.

Polar 24-hour orbits can be established by firing to the south (that is, somewhat west of south relative to the ground) from the Vandenberg Air Force Base. The method of ascent is similar to that in the equatorial case. However, in a southward launch one gains no advantage from the earth's rotation.

3.2 ASCENT GUIDANCE

Placing a satellite into a circular 24-hour orbit involves "ascent guidance" of the satellite into an approximately correct orbit and then making the necessary "orbital corrections" with the help of observations taken on the ground.

The apogee of the transfer ellipse will always be within a line-of-sight of the United States or a friendly land area, because its position is determined by the communication requirements. On the other hand, the perigee of this ellipse does not always have this property; for example, if the satellite is intended for communication between the United States and Asia, the perigee lies over the Indian Ocean. Therefore radio guidance can always be used at apogee, but not always at perigee.

By contrast, certain available all-inertial guidance systems are satisfactory for the booster-powered flight and at perigee; but the present inertial systems are not adequate for the apogee burning without external corrections, because of the cumulative gyro drift for the long flight period. Therefore both radio and inertial guidance must be provided for 24-hour communication satellites.

Guidance during the booster-powered flight is done with a gyro-stabilized inertial platform, mounting three accelerometers. This platform works together

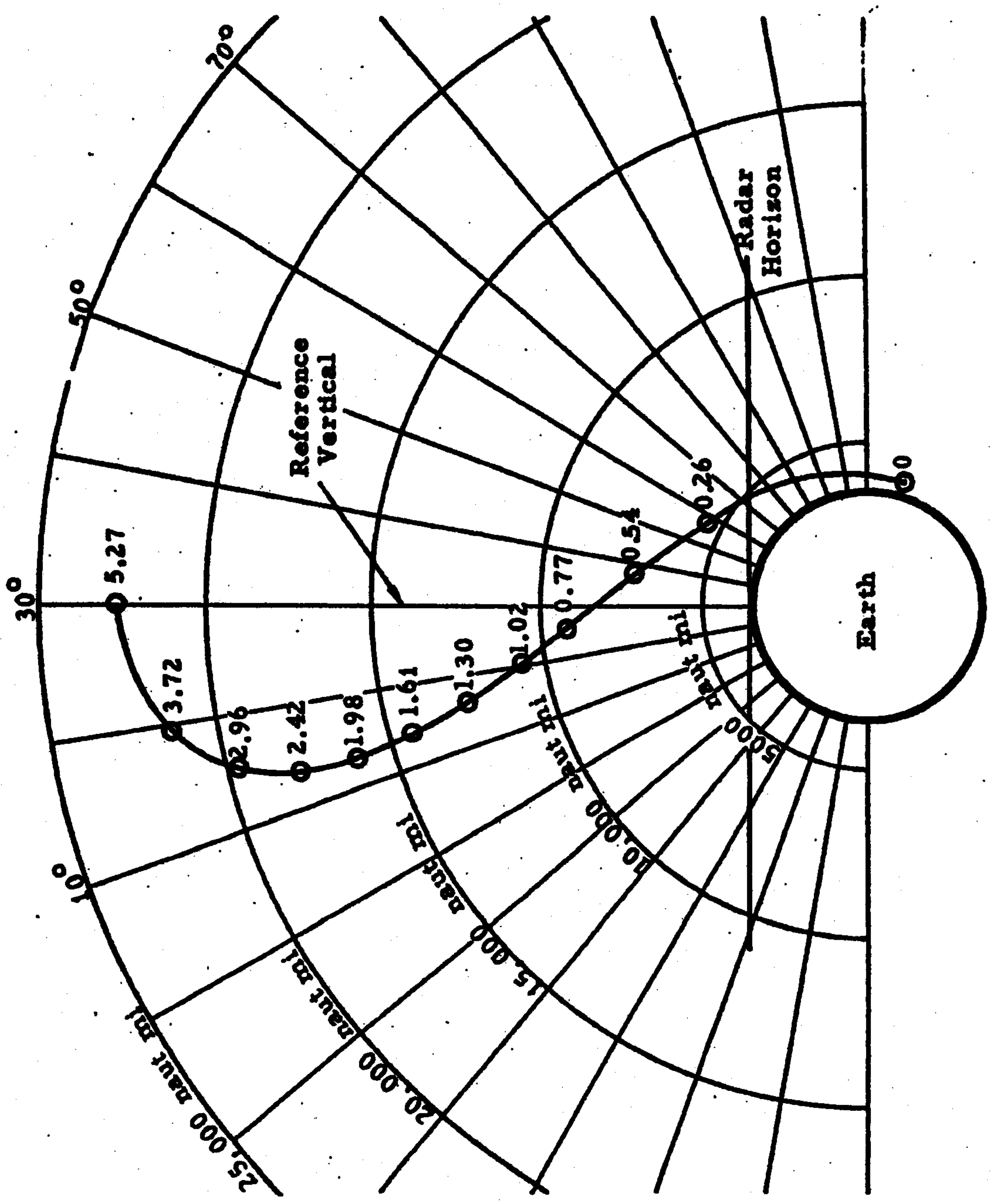

Figure 3-3. Position of Vehicle in the Equatorial Plane During Ascent Relative to the Rotating Earth.

with a "primary" guidance digital computer. During the first-stage booster burning period, the vehicle follows a programmed pitch attitude and is stabilized in roll and yaw using the platform as a reference. The primary guidance computer calculates position and velocity throughout this period. Attitude control and steering instructions for subsequent stages in the climb to the low-altitude orbit are provided by the primary computer, which can also issue engine start and cutoff signals.

The primary computer is relatively heavy, because the computations to be done before perigee burning are complex. However, the computations required during and after the perigee burning are simple enough for a rudimentary digital computer, here called the "steering and shutoff computer." Therefore, the primary guidance computer is housed in the first added stage and is jettisoned before perigee burning. The steering and shutoff computer is housed in the final stage.

Low-Altitude Coasting. The inertial guidance system can measure the velocity and position of the vehicle at burnout more accurately than it can control these quantities. The resultant control errors in altitude and speed at the point of injection into the low-altitude orbit affect the low-altitude coast period and, together with the downrange error at injection into the low-altitude orbit, cause an error in the downrange position of the satellite at the nominal time of perigee burning. This can be compensated by adjusting the ignition time for perigee burning. Velocity and altitude errors at the actual time of perigee burning can be compensated by adjusting the vector velocity increment to be added at perigee. The perigee correction biases are computed by the primary guidance computer, and are transferred to and stored in the smaller steering and shutoff computer before the first added stage is separated from the vehicle.

After the separation of the first added stage, the vehicle is reoriented to the attitude required for perigee burning, and is held in this attitude during the coast in the circular low-altitude orbit. A simple analog computer commands the attitude control jets by converting platform gimbal angles into on-off signals for the jets.

Perigee burning adds about 8000 feet per second to the speed, the actual amount depending on the perigee impulse correction stored in the steering and shutoff computer, which provides its commands on the basis of the accelerometer readings. Steering commands go to the perigee-apogee autopilot.

During the final two or three hours of the ascent coasting in the transfer ellipse the vehicle must be tracked from the ground, to obtain its position and velocity. The tracking data are fed into a ground digital computer, and apogee burning corrections are sent to the vehicle as modulations on the tracking waves. Attitude stabilization during ascent coasting is provided by the platform. During this coast period the vehicle must be reoriented so as to permit the motor used for perigee burning to inject it into the 24-hour orbit.

Apogee burning is commanded from the ground when the vehicle reaches the correct angle relative to the earth (the correct latitude for a polar satellite and the correct longitude for an equatorial satellite). The accumulated measurement and control errors will prevent it from occurring precisely at apogee, or at the precisely correct altitude. The procedure that simplifies the problem of orbital correction is to strive for attaining a horizontal burnout velocity that would lead to a satellite period as close to one sidereal day as possible.

After apogee burning the vehicle is reoriented by 90 degrees so as to point its nose-mounted antennas toward the ground stations. This is done with gas jets, using data from the inertial platform. After this rough orientation, a more precise attitude is achieved and the unwanted angular rates are eliminated by the attitude control system, which must maintain the correct orientation of the satellite throughout its life.

3.3 ORBITAL CORRECTIONS

If the period of an equatorial satellite is different from one sideral day, the satellite will keep drifting "off station" to the east or west. To permit tracking by moving only the feed of the ground antenna it is probably desirable to limit the total drift during the operational life of the satellite to about 2 degrees; this corresponds to a total drift of about 800 nautical miles along the orbit, and to a total shift of the boundaries of the "covered region" amounting to about 120 nautical miles on the equator. On the assumption of an operational life of 9 months, this limitation permits an average drift rate of about 0.2 foot per second. If the orbit is slightly eccentric but the difference between the altitudes of the satellite at perigee and apogee is limited to 200 nautical miles, the satellite will undergo a daily east-west oscillation limited to only one-half a degree. Similarly, the error in the inclination of the orbital plane is likely to be quite small compared to 2 degrees. In any case, such discrepancies can be reduced within one or two days, as illustrated below for the case when the eccentricity of the orbit is to be reduced.

Consider a satellite injected into a precisely 24-hour equatorial orbit, but suppose that this orbit is slightly elliptic and the actual altitude at apogee burning is greater than the nominal altitude, as indicated in Figure 3-4. Since the angular eastward motion of the satellite about the earth is slowest when it is at apogee, and since its average angular rate is the same as that of the earth, the satellite will first recede to the west of its intended longitude. As the altitude decreases, the apparent angular regression slows up; and 6 hours after apogee, when the satellite crosses the altitude of the nominal circular orbit, the westward regression changes to an eastward progression, whose rate reaches a maximum when the satellite is at perigee. The path of the satellite relative to an observer on the earth is, to a first approximation, an ellipse, as shown in Figure 3-5. Detailed first-order analysis shows that the major axis of this ellipse is horizontal and is twice as long as the minor axis. In Figure 3-5 the point C lies on the desired meridian at the altitude of a circular orbit having the period of one sidereal day. The point A is the position of the satellite at the end of apogee burning. Six hours later the satellite is at B. At this point an outward radial impulse is applied which halves the inward radial velocity component. As a result, the apparent motion of the satellite during the next 12 hours is along the arc BC rather than along the arc BDE. When the satellite arrives at C, a second impulse is applied, equal and opposite to that applied at point B. This impulse stops the relative motion.

An error analysis indicates that the amplitude of the north-south oscillation caused by having the orbital plane inclined to the equatorial plane may not be

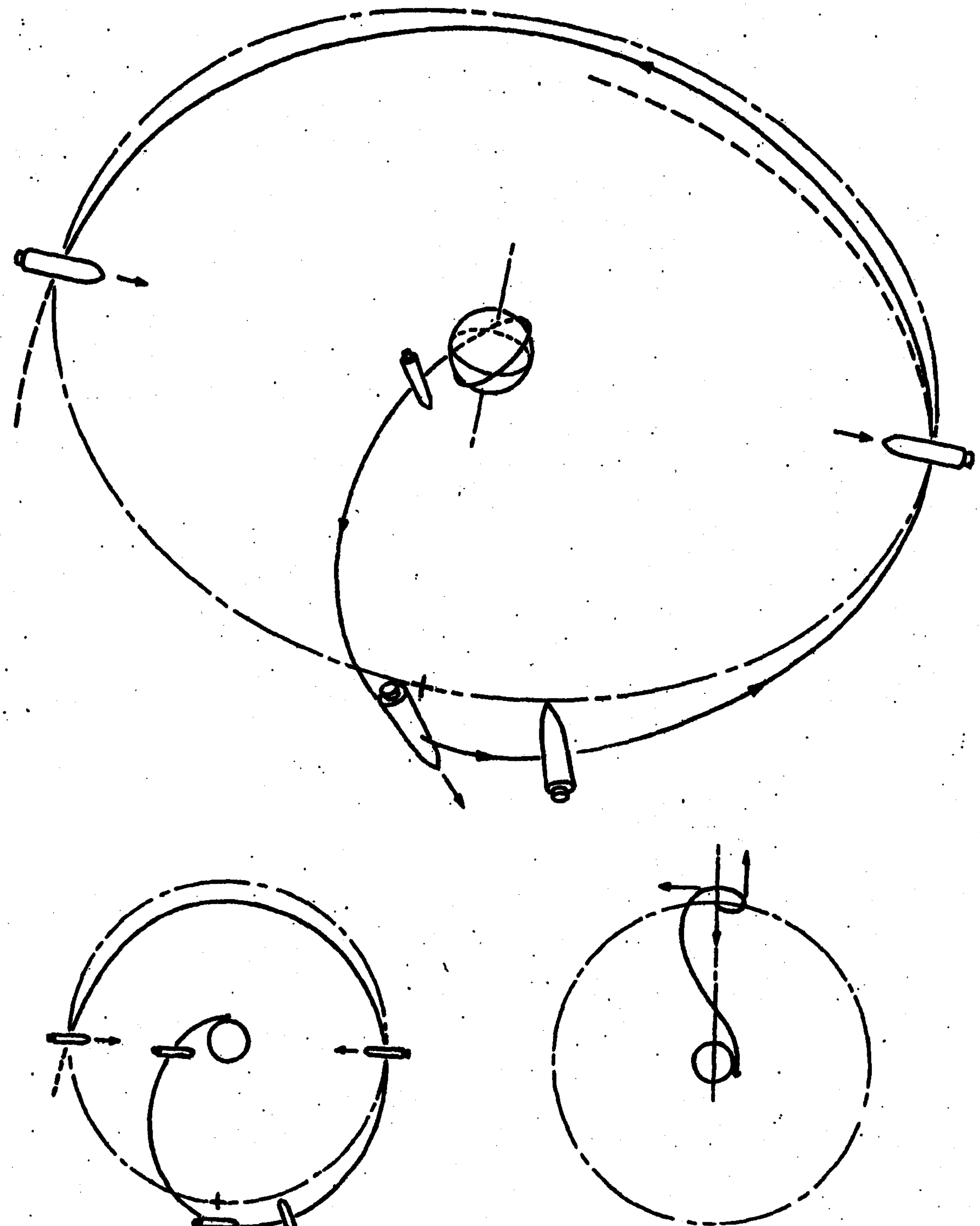

Figure 3-4. Method for Correcting Eccentricity of Final 24-Hour Orbit.

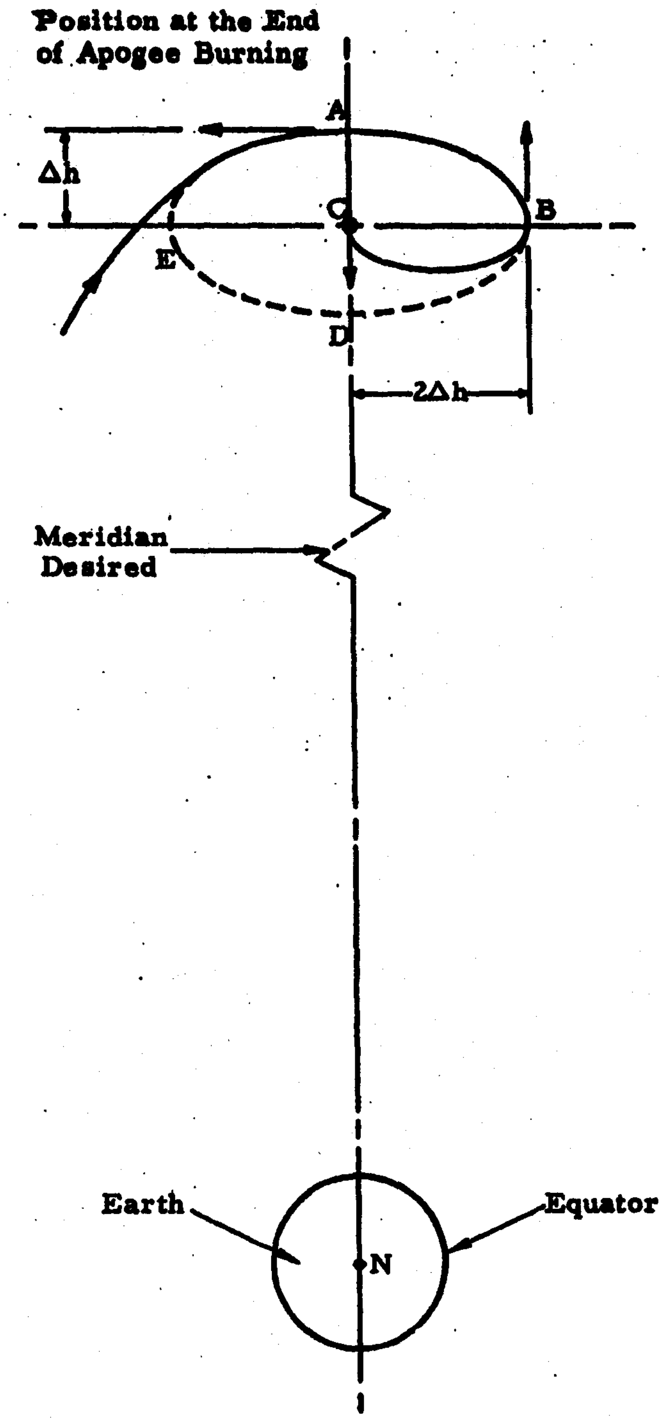

Figure 3-5. Apparent Path of Satellite as Viewed from the Rotating Earth.

large enough to warrant correction. However, if it is necessary to reduce this oscillation, this can easily be done by applying an impulse to the satellite in the north-south direction as it crosses the equatorial plane.

The discussion given above relates to equatorial orbits. The polar orbital correction problem is somewhat different, because in that case it is not necessary to control as closely either the eccentricity or the orbital inclination.

Orbital correcting impulses can be supplied by gas jets controlled by the steering and shutoff computer, which receives commands through a decoder from the ground tracker and computer (such a data link and decoder, associated with the transponder beacon, is now used on the Atlas missile). Since the jets cannot be calibrated precisely, a vernier acceleration sensing system must be used to control the velocity impulses by a feedback loop through the steering and shutoff computer. The sensitive accelerometers used during the correction period may remain caged until the apogee burning is finished.

Estimates of certain weights and power requirements are given in Table 3-1 (the actual power consumption is quite uneven, because of the intermittent use of the components). The weight of airborne guidance in the last stage, required to localize the satellite above any point on the equator accurately enough for any of the applications discussed in Chapter 2, proves to be about 370 pounds.

Analysis of the accuracy of the system shows that the suggested combination of inertial and radio measurements at appropriate points on the ascent trajectory would stabilize for 5 years or more both the position and the altitude of a 24-hour satellite. The vernier velocity impulses required to stabilize the orbit probably amount to about 100 feet per second total.

The guidance system recommended for polar satellites differs slightly from that for equatorial satellites, because only broad-beam coverage is required; that is, after the initial velocity corrections are made, a polar satellite need not be stabilized against rotation about the axis directed from the satellite toward the center of the earth.

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Table 3-1. Power Breakdown for Each Phase of Orbit-Establishing Sequence.

Airborne Component	Preorbital Trajectory		Low-Altitude Coasting (10 hours)	Perigee Burning	Ascent Coasting (5.3 hours)	Apogee Burning	Vernier Correction (24 to 48 hours)
	Weight (pounds)	Power (watts)	Power (watts)	Power (watts)	Power (watts)	Power (watts)	Power (watts)
1. Primary guidance computer (digital)	90	250					
2. Booster autopilot[a]	15	35					
	110	285					
3. Gyro-stabilized inertial platform and electronics	235	400	400	400	400	400	
4. Steering and shutoff computer (digital)	20		40	40	40	40	40
5. Apogee and perigee autopilot[a]	10			35		35	
6. Tracking beacon with data link and decoder[b]	15				5[c]		5[c]
7. Vernier acceleration-sensing system	20						10
8. Clock	1						
9. Cabling	25						
Total	326	400	440	475	445	475	55

[a] Does not include actuators

[b] 50 watts

[c] Tracking for 2 seconds every 2 minutes

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CHAPTER 4

ATTITUDE CONTROL, ELECTRIC POWER, AND VEHICLE DESIGN

4.1 ATTITUDE CONTROL

The attitude of the vehicle before reaching the 24-hour orbit may be controlled by the inertial platform and a "coarse" gas jet system, which can orient the vehicle properly during low-altitude and ascent coasting, and point its nose toward the earth after apogee burning. This system can also add the small velocity increments needed for orbital correction; for this purpose the jet pairs would be oriented for thrusting rather than for torquing. A simple jet system is shown in Figure 4-1. Finer attitude control during low-altitude and ascent coasting is achieved by using the inertial platform and a "fine" gas jet system.

After injection into a circular 24-hour orbit the satellite should move with its nose constantly pointing at the center of the earth. The term <u>roll</u> then means rotation about the velocity vector, <u>yaw</u> means rotation about the vertical, and <u>pitch</u> means rotation about the axis perpendicular to the plane of the orbit. A properly stabilized 24-hour communication satellite will turn in the pitch plane through 360° each sidereal day.

When in orbit, the satellite is stabilized against roll and yaw by a 10- to 12-pound flywheel, whose axis is perpendicular to the plane of the orbit, as in Figure 4-2. The angular speed of this wheel is kept constant by an electronic clock. A small variable-speed wheel, whose axis is parallel to the axis of the large flywheel, is used to "absorb" small disturbances in the pitch angular momentum caused, say, by meteor impacts. If an accumulation of disturbances should cause the small wheel to exceed its design speed, it can be stopped (while the pitch gas jets are used to hold attitude) and left free to absorb future disturbances in pitch. If the available information on meteor frequency as a function of a size is correct, then it is likely that the small pitch wheel alone will be able to compensate for meteor impact and the fine gas jets will have to be used only during the orbital correction period (24 to 48 hours), when relatively large accidental torques may result from an unbalance of the coarse jets.

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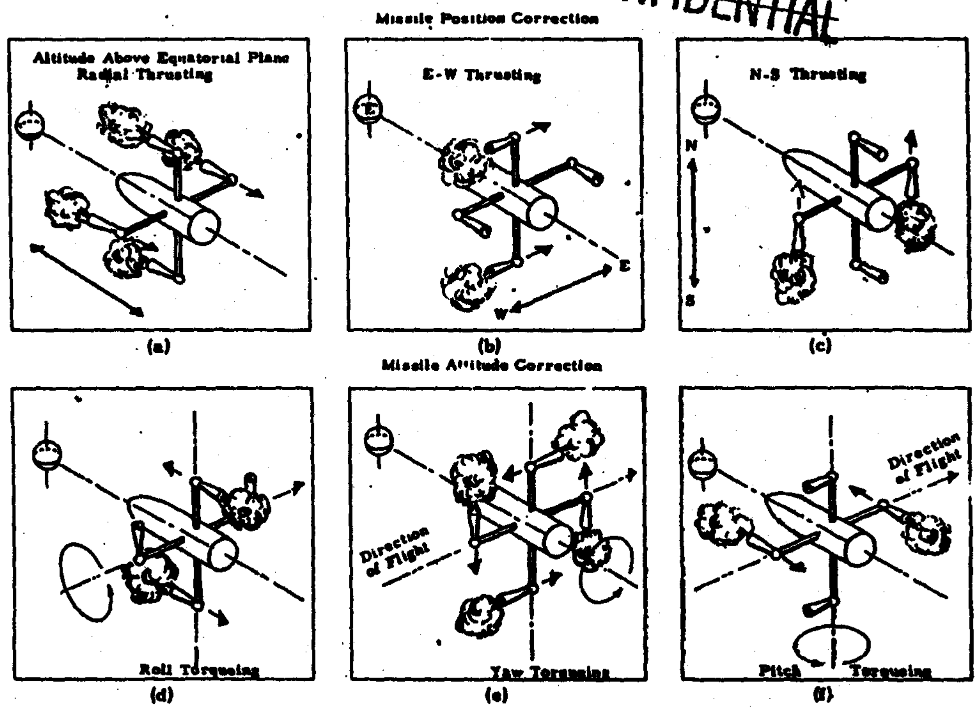
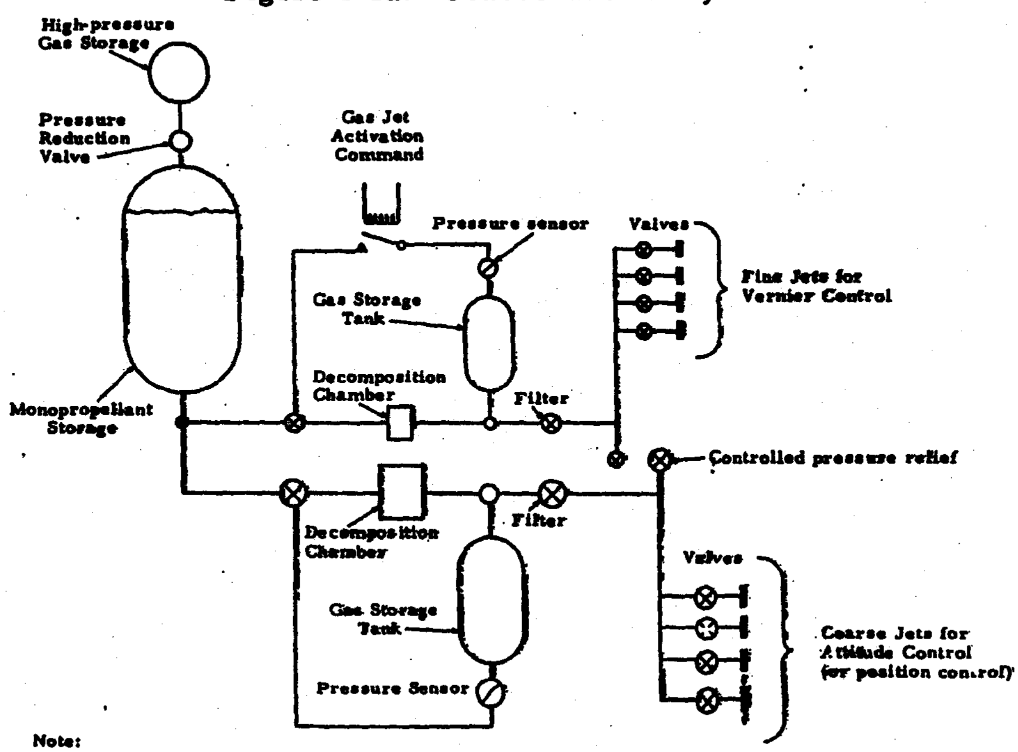


Figure 4-1a. Coarse Gas Jet System.



Note:
1. All valves have electric control
2. Jets rotate about one axis

Figure 4-1b. Gas Jet System for Attitude Control and Velocity Correction in 24-Hour Orbit.

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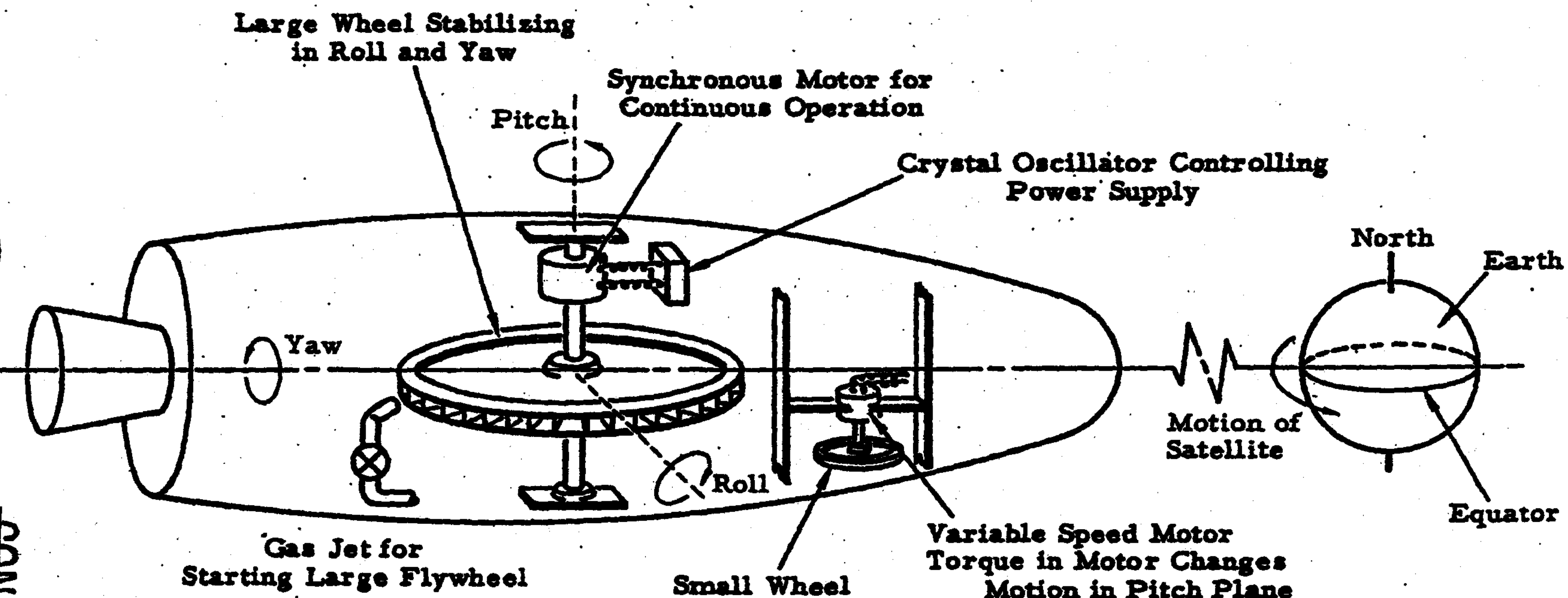

Figure 4-2. Schematic of Stabilizing Flywheel (Axis Perpendicular to Plane of Orbit).

The primary purpose of an attitude control for a communications satellite is to keep the communications antennas pointed at the ground stations. For the "stationary" equatorial satellite two small monopulse receiving antennas with a relatively broad receiving pattern can be used as finders for the relatively narrow-beam communications antennas. Error signals derived from the monopulse receiving system can be used to control the fine gas jet system and the pitch wheel. For a polar satellite it is only necessary to keep one 16-degree antenna pointed at the earth, and there is no need for yaw control. Pitch and roll measurements for a polar satellite can be made by an infrared horizon scanner and can be used to control the fine gas jet system and the small pitch wheel. Table 4-1 summarizes certain weights and powers required for attitude control if a Thor or a Titan should be used as a booster.

4.2 ELECTRIC POWER

Estimates of the power required for satellite-borne communication equipment were given in Table 1-1 on page I-13 for both low-density and high-density service and for both the <u>interim</u> system (which can be built in 2 to 3 years) and the <u>initial</u> system (which can be built in 4 to 5 years). Power estimates for other equipment are given in Tables 4-2 and 4-3, for the case of high-density service. Table 4-2 relates to the interim system and a Thor booster, Table 4-3 to the initial system and a Titan booster. The use of periodically focused traveling-wave tubes is presupposed in both tables, because these tubes can save considerable power.

Photovoltaic cells for converting solar energy directly into electrical energy appear to be able to supply the necessary power. Other devices involve large weight penalties or require much additional development--as do light-weight nuclear power systems, which should, however, be considered for more advanced communication satellites.

Electric power must be available in the satellite as ac, low-voltage dc, and high-voltage dc. If the efficiency of the inverter delivering the ac is 80 per cent and the combined efficiency of the transformer and rectifier delivering the high-voltage dc is 90 per cent, the low-voltage power necessary

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Table 4-1. Data on Attitude Control System if Thor or Titan is Used as Booster.

Component	Weight (pounds)	Power (watts)
Coarse Gas Jets (plumbing and main tank)	15	
Gas for Velocity Correction (Thor)	45**	
(Titan)	90**	
Gas for Reorientation	6**	20*
Fine Gas Jets (plumbing)	5	
Gas for Fine Reorientation	0.6**	
Gas for Continuous Attitude Correction in 24-hour Orbit for Lifetime of One Year	5**	
Monopulse System	20	75*
Attitude Stabilization Computer (analog)	15	20
Large Flywheel	12	20
Small Flywheel	2	10
TOTALS (Thor)	125.6	145
(Titan)	170.6	

* Used very intermittently

** The weight of this gas is based upon a specific impulse (62 sec/lb) of catalytically decomposed monopropellant. If a higher specific impulse is found to be feasible, the weight of the gas will be proportionately smaller.

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Table 4-2. Electric Power Requirements for Interim System (Thor Booster, Periodically Focused Tubes).

	Power Requirements (watts)			
	Low-Altitude Coast	Ascent Coast (5.3 hours)	Apogee Burning	24-Hour Orbit
Guidance	360	384	635	3*
Attitude Stabilization, etc.	20	20	20	30
Coolant Pump	5	5	5	5
Communication				210*
Solar Controls				10
Total	385	409	660	258

* 180 watts during first 24 hours, when communication equipment is not on.

Table 4-3. Electric Power Requirements for Initial System (Titan Booster, Periodically Focused Tubes).

	Power Requirements (watts)			
	Low-Altitude Coast	Ascent Coast (5.3 hours)	Apogee Burning	24-Hour Orbit
Guidance	360	384	635	3*
Attitude Stabilization, etc.	20	20	20	30
Coolant Pump	(15)	(15)	(15)	(15)
Communications				1760
Solar Controls				(20)*
Total	395	419	670	1828

* If solar cells are used.

to provide the 258 watts listed in Table 4-2 proves to be about 350 watts. During the eclipses of the satellite this power must be provided by storage cells, which must be charged by the solar cells while they are illuminated. This circumstance raises the demand on the solar cells to 378 watts for normal incidence of sunlight, a figure based on the assumption of a charge/discharge efficiency of 70 per cent and on a study of the frequencies and durations of satellite eclipses at various seasons of the year.

The area of the cell-covered surface required to provide the necessary power depends on the conversion efficiency of photovoltaic cells (which is a function of the temperature), on the effect of a covering for protection against meteoric pitting, and on the fact that the available surface cannot be fully utilized because of various clamps and connectors. Detailed computations show that, in the case of silicon cells, the area of about 54 square feet is required to provide 378 watts at normal incidence. This figure is listed in Table 4-4, together with other relevant data for the interim system. Table 4-5 gives similar data for a satellite of the initial system.

The orientations of an equatorial 24-hour satellite at different times of day and of the solar cells at different times of year are pictured in Figure 4-3.

4.3 VEHICLE DESIGN

This study included a detailed examination of the various propulsion and vehicle design problems associated with the launching of 24-hour satellites with a Thor, a Titan or an Atlas booster. Various added-stage combinations were considered, to determine the payloads as functions of staging, specific impulse, and structure. The resulting recommendations are given in detail in Volume IV of this report. The present summary is confined to some of the results obtained for the case of an Atlas or Titan booster.

Figure 4-4 is a sketch of a 24-hour satellite vehicle boosted by an Atlas or an uprated Titan (400,000-pound thrust). The two added stages have a gross weight of 20,000 pounds at take-off. Figure 4-5 is a preliminary sketch of the added stages that could be used with these boosters. Two alternatives for the 12,000-pound thrust engine are shown, a pressure-fed engine and a turbopump-fed engine. Table 4-6 presents a structural weight breakdown for the two

Table 4-4. Solar Power System, <u>Interim</u> System.

Item		Weight (lb)
Storage batteries for eclipses		
345 watts for 1.3 hours		32*
Solar cells		
Supply 378 watts with normal light incidence (54 sq ft)		27
Axle, gear train, motor		10
Servocontrols for surface orientation		10
Inverter, transformer, rectifier		10
Payload components in 24-hour orbit		89
Chemical batteries		
For low-altitude coast (jettisoned before perigee burning)		
475 watts for 10 hours (maximum)	71	14 (equivalent payload)
For transfer ellipse (jettisoned before apogee burning)		
508 watts for 5.3 hours	40	17 (equivalent payload)
Total equivalent payload for entire electric power system		120

Table 4-5. Solar Power System, <u>Initial</u> System.

Item		Weight (lb)
Storage batteries for eclipses		
2498 watts for 1.3 hours		232*
Solar cells		
Supply 2737 watts with normal light incidence (391 sq ft)		195
Axle, gear train, motor		72
Servocontrols for surface orientation		10
Inverter, transformer, rectifier		30
Payload components in 24-hour orbit		539
Chemical batteries		
For low-altitude coast (jettisoned before perigee burning)		
475 watts for 10 hours (maximum)	71	14 (equivalent payload)
For transfer ellipse (jettisoned before apogee burning)		
508 watts for 5.3 hours	40	17 (equivalent payload)
Total equivalent payload for entire electric power system		570

* Based on 70 per cent discharge/charge efficiency and a weight of 50 pounds per kilowatt hour of input electric power.

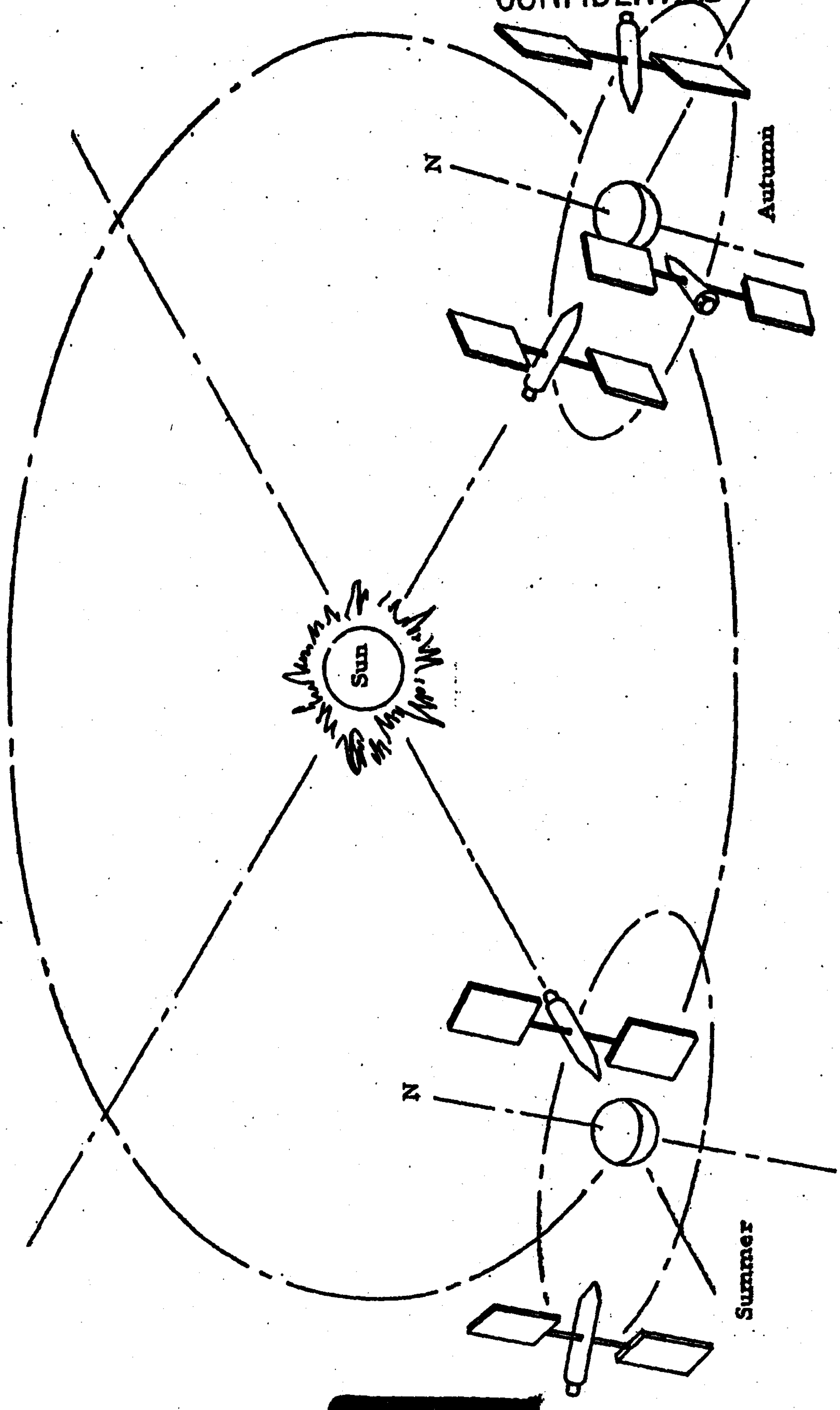

Figure 4-3. Orientation of Solar Power Surfaces for Equatorial Orbit (Two Surfaces, One Axis of Rotation).

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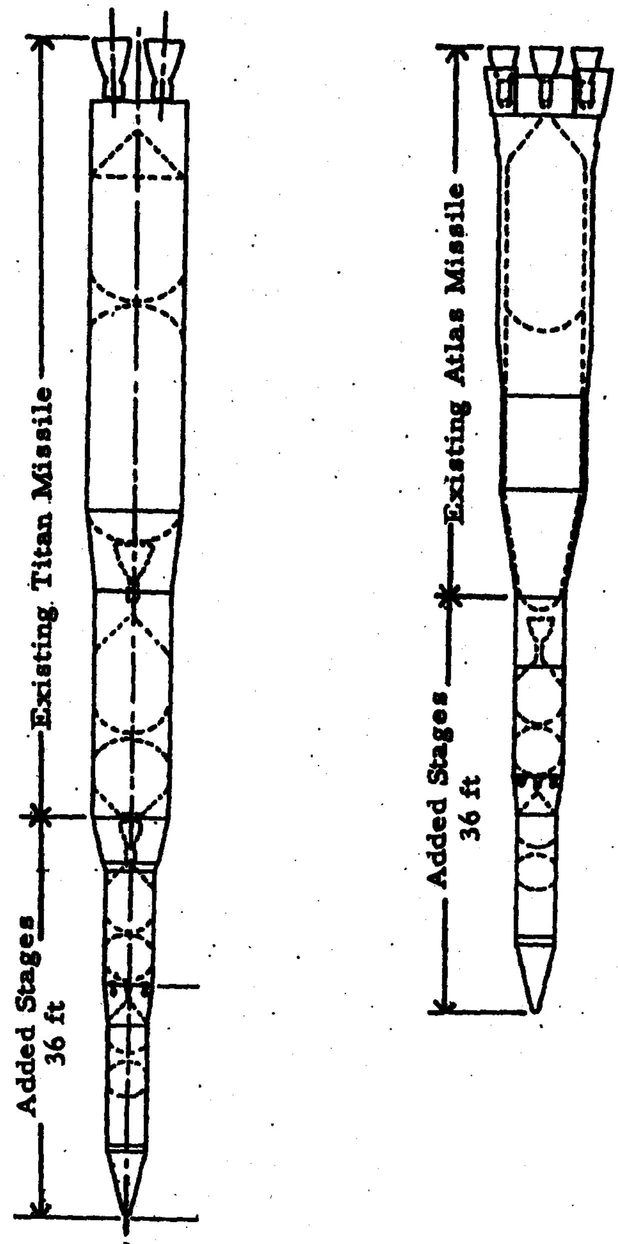

Figure 4-4. 24-hr Satellite Vehicles with Atlas and Titan Boosters

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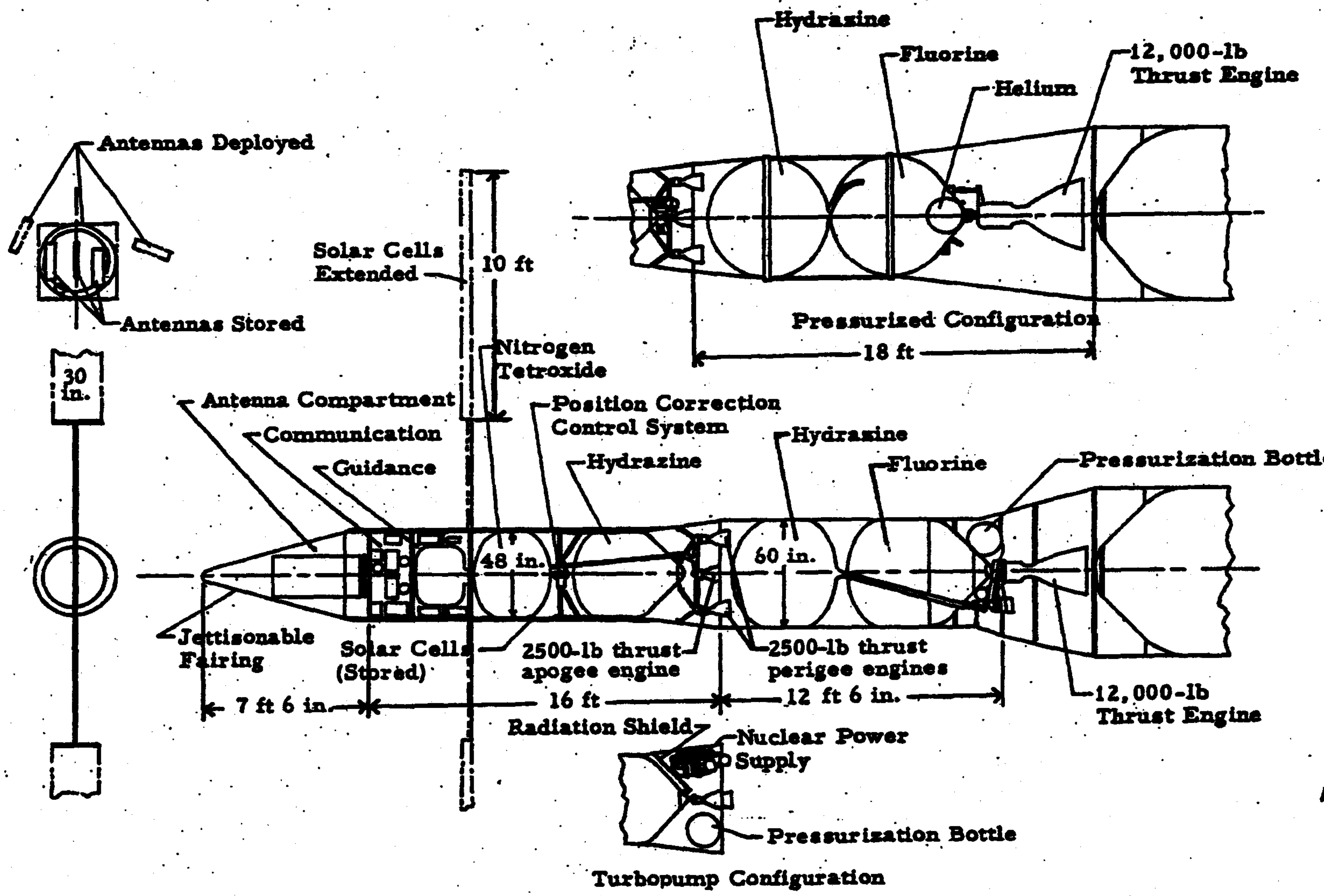

Figure 4-5. Design of the First Added Stage and Final Stage of the 24-Hour Satellite Vehicle with Titan Booster.

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Table 4-6. Structural Weight of Titan-Boosted First Added Stage (Using 12,000-lb Thrust F_2 - N_2H_4 Nomad Engine).

	Pressure Feed (lb)	Turbopump Feed (lb)
Propellant Tanks	208	133
Intertank Structure and Thrust Frames	214	178
Pressurization System	110	90
Start Rocket Casings (2)	20	20
Forward Interstage Structure	46	46
Motor (wet) with Flexure and Actuators	129	125
Turbopumps and Plumbing (wet)	--	39
Plumbing	19	--
Residual Propellants (1 per cent)	115*	115*
TOTAL	861*	746*

* Add 10 pounds for polar orbit.

alternate designs of the first added stage. With slight modifications, this stage can be mounted on either the Atlas or the Titan.

The final stage has storable propellants, nitrogen tetroxide and hydrazine; they are pressurized, and thus boiloff during long coast periods is avoided. Since weight is not extremely critical in this design, three engines are used, one 2500-pound thrust engine for perigee burning and two 1250-pound thrust engines for apogee burning. The use of different engines at apogee and perigee burning eliminates the reignition problems.

Table 4-7 is a weight summary of the Titan-boosted 24-hour satellite vehicle, and Table 4-8 describes a possible allocation of the payload suitable for either the interim or the initial systems when the Atlas or Titan is used as boosters. Although the weights allocated to the components are rather generous, a considerable amount of weight remains unassigned. Guidance weight has been increased by 100 pounds, to make the specifications easier to meet. However, it is likely that the weights of various components will in general decrease in time, and consequently the figures given in Table 4-8 are very conservative.

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Table 4-7. Weight Summary for Titan-Boosted Vehicle.

	Equatorial Orbit (Cape Canaveral) (pounds)		Polar Orbit (Vandenberg AFB) (pounds)	
First Added Stage				
Initial Weight		20,000		20,000
Propellant Consumed	11,514		12,505	
Burnout Weight		8,486		7,495
Structure and Residuals Jettisoned	861		871	
Final Weight		7,625		6,624
Final Stage				
Batteries Jettisoned	71		12	
Initial Weight at Perigee		7,554		6,612
Propellant Consumed	4,105		3,593	
Burnout Weight at Perigee		3,449		3,019
Batteries Jettisoned	40		40	
Initial Weight at Apogee		3,409		2,979
Propellant Consumed	1,519		1,123	
Burnout Weight at Apogee		1,890		1,856
Structure and Residuals	586		577	
Gross Payload		1,304		1,279

Table 4-8. Payload Weight Summary (Titan Booster, Equatorial Orbit).

	Interim (pounds)		Initial (pounds)	
Payload Capability		1304		1304
Communications	176		220	
Guidance	470		470	
Attitude Stabilization	70		70	
Vernier Jet System	70		70	
Power	135		201	
Cooling	18		51	
Total	939	939	1082	1082
Unassigned		365		222

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