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16. Abstract

Designs are synthesized for direct sound broadcast satellite systems for HF-, VHF-, L-, and Ku-bands. Methods are developed and used to predict satellite weight, volume, and RF performance for the various concepts considered. Cost and schedule risk assessments are performed to predict time and cost required to implement selected concepts. Technology assessments and tradeoffs are made to identify critical enabling technologies that require development to bring technical risk to acceptable levels for full scale development.

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FOREWORD

This report was prepared by Martin Marietta Denver Aerospace under Contract NAS3-24233. The contract was administered by the Lewis Research Center (LeRC) of the National Aeronautics and Space Administration (NASA). The study was performed from April 1984 to June 1985 and the NASA-LeRC project manager was Mr. Grady Stevens.

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GLOSSARY

A	Antenna Array Area
ACS	Attitude Control Subsystem
AM	Amplitude Modulation
APS	Auxiliary Propulsion Subsystem
ASSA	Advanced Space Systems Analysis
ATS-6	Applications Technology Satellite-6
A_p	Panel Area
A_{sa}	Solar Array Area
BOL	Beginning of Life
BS	Broadcast Satellite
BW	Beam Width
C	Average Unit Cost
C/N	Carrier-to-Noise Ratio
CDR	Critical Design Review
CER	Cost Estimating Relationship
CMG	Control Moment Gyro
CP	Circular Polarization
CPL	Capillary Pumped Loop
C_{ave}	Average Cost of Units
C_{u1}	First Unit Fabrication Cost
C_t	Total Fabrication Cost
D/A	Digital to Analog
A/D	Analog to Digital
D	Antenna Diameter
deg	Degree(s)
DOD	Department of Defense
DSB	Double Sideband
DVBS	Direct Voice Broadcast System
EOC	Edge of Coverage
EOL	End of Life
EPS	Electrical Power Subsystem
ESH	Equivalent Solar Hour
ETR	Eastern Test Range
FM	Frequency Modulation
F_{bv}	Battery Volume Factor
F_{bw}	Battery Weight Factor
F_c	Solar Cell Cover Glass Thickness Weight Factor
F_d	Battery Depth of Discharge
F_r	Radiation Degradation Factor
F_t	Temperature Adjustment Factor
GEO	Geostationary Earth Orbit
h	Hour(s)
HF	High Frequency

IF	Interface (Intermediate) Frequency
IR	Infrared
IUS	Inertial Upper Stage
I_{sp}	Specific Impulse
K	Solar Array Sizing Constant (1123 W/m^2)
K_d	Solar Array Density Coefficient
L/D	Length-to-Diameter (Ratio)
	Structural Dimension Ratio
L	Length of Radiating Dipole
LaRC	Langley Research Center
LEO	Low-Earth Orbit
LCC	Life-Cycle Cost
LSS	Large Space System
L_s	Spreading Loss
L_{ssl}	Spacecraft Stowed Launch Length
m	Slope of Learning Curve
	Meter(s)
MCC	Miniature Cassegrainian Concentrator
MeV	Megaelectron Volt(s)
MBG	Multiband Gap
MSFC	Marshall Space Flight Center
N	Number of Radiating Elements
NASA	National Aeronautics and Space Administration
NR	Nonrecurring
NRC	Nonrecurring Cost
N_c	Charger Efficiency
N_d	Diode Efficiency
N_e	Battery Efficiency
N_i	Solar Cell Efficiency
N_r	Distribution Efficiency
N_u	Total Number of Units Produced
N_{wr}	Wire Efficiency (Source to Bus)
N_{ws}	Wire Efficiency (Bus to Load)
O&S	Operation and Support
OPD	Orbital Plane Day
OTS	Off the Shelf
OTV	Orbital Transfer Vehicle
PDR	Preliminary Design Review
PFD	Power Flux Density
PPT	Pulsed Plasma Thruster
psi	Pound(s) per Square Inch
P_{es}	Eclipse Load Power
P_l	Load Power
P_{ls}	Sun Load Power
P_s	Power from Source
P_t	Transmitter Power

Q	Radiative Heat
R	Distance from Satellite to Receiver
RAMP	Risk Assessment and Management Program
RC	Recurring Cost
RCD	Rigid Body Control
RCS	Reaction Control System
RFC	Regenerative Fuel Cell
RTG	Radioisotope Thermoelectric Generator
s	Second(s)
S/C	Spacecraft
S/N	Signal-to-Noise (Ratio)
SAFE	Solar Array Flight Experiment
SAR	Synthetic Aperture Radar
SCIAP	Spacecraft Integrated Analysis Program
SDCM	System Design and Cost Model
SDR	System Design Review
SEMP	Systems Engineering Management Plan
SEP	Solar Electric Propulsion
Si	Silicon
SOA	State of the Art
SOD	Sidereal Orbital Day
SOW	Statement of Work
SRR	System Requirements Review
SSB	Single Sideband
SSPA	Solid State Power Amplifier
STS	Space Transportation System
TCOM	Tethered Communications Corporation
TOS/AMS	Transfer Orbit Stage/Apogee Maneuvering Stage
TT&C	Telemetry, Tracking, and Command
TWT	Traveling-Wave Tube
TWTA	Traveling-Wave Tube Amplifier
T_c	Temperature Collector
T_{cd}	Component Development Time
T_{cq}	Component Qualification Time
T_e	Time of Eclipse
T_{em}	Temperature Emitter
T_{sd}	Subsystem Development Time
T_{sq}	Subsystem Qualification Time
T_{sysq}	System Qualification Time
T_s	Time in Sun
UHF	Ultrahigh Frequency
USAF	United States Air Force
USIA	United States Information Agency
UTC	Universal Time Coordinated
VHF	Very High Frequency
VOA	Voice of America
Vol	Volume

V_b	Battery Volume
V_e	Equipment Bay Volume
V_s	Subsystem Hardware Volume
V_{sa}	Solar Array Volume
W	Watts
W_a	Antenna Weight
W_b	Battery Weight
W_{bc}	Battery Charger Weight
W_d	Power Distribution Weight
W_e	Equipment Bay Weight
W_s	Subsystem Weight
W_{sa}	Solar Array Weight
W_{sr}	Shunt Regulator Weight
W_{sw}	Power Switching Weight
W_t	Transmitter Weight
WTR	Western Test Range
α	absorptivity
ε	Emmissivity
σ	Stefan-Boltzmann Constant
λ	Wavelength

SATELLITE VOICE BROADCAST SYSTEM STUDY

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SUMMARY

The primary goal of this study was to develop technical, schedule, and cost data that can be used by the U.S. Information Agency to evaluate use of sound broadcast satellite systems to meet future international sound broadcast needs. Satellite systems launchable by the space shuttle were synthesized and analyzed for broadcast at four frequencies: 26 MHz (HF-band); 47 MHz (VHF-band); 1.5 GHz (L-band); and 12.2 GHz (Ku-band). Broadcast requirements for the study specified time of day, duration of broadcast, and ranges of ground signal strength. Results showed that satellite systems can meet Ku-band requirements. L-band systems were designed that can meet lower signal strength requirements. Neither VHFnor HF-band requirements can be met by realistic satellite systems. For these latter bands, the study results identified the maximum possible broadcast capabilities for each concept. Also, for HF-band systems, parametric relationships were derived to identify available signal strength and satellite mass vs satellite output power. Time and cost to implement each system were estimated, and risk assessments performed to identify 90 and 10% risk values of time and cost.

1.0 INTRODUCTION

The Satellite Voice Broadcast System Study was commissioned by NASA to investigate the feasibility of a Direct Voice Broadcast System (DVBS) in space. The study evaluated potential operating systems in four frequency bands: 26 MHz, 47 MHz, 1.5 GHz, and 12.2 GHz. Potential operational system concepts were defined to a depth sufficient to determine the relative technical characteristics, performance, and costs (development, construction, and operating), and to develop schedules of selected system concepts. In addition, an assessment of the impact of and need for advanced technology for these system concepts was performed.

1.1 BACKGROUND

The use of satellites to provide sound broadcasting was examined by NASA as early as 1967 (ref. 1 and 2). More recently, this service has received increasing attention for both national and international broadcasting interests. CCIR Report 955 (ref. 3) deals with the feasibility of sound broadcasting satellite systems operating in the range of 500 MHz to 2 GHz. The primary application in Report 955 is broadcasting to automotive or portable receivers having relatively low gain antennas; in this case rather large satellites are required due to large multipath fade margins.

An extension of this work by Chaplin, et al. considered only the rural broadcasting case and an improved receiver noise performance. Their analyses for this special case (ref. 4) indicates that national broadcasting at 1 GHz is feasible with rather conventional size spacecraft. Phillips and Knight (ref. 5) explored the same subject at 26 MHz. None of these studies considered the operational difficulties of worldwide sound broadcasting but confined themselves to restricted coverage, single satellite concepts.

The U.S. Information Agency (USIA)/Voice of America (VOA) is considering sound broadcasting by satellite as part of a program to renovate, modernize, and expand the existing worldwide USIA/VOA broadcasting network. With such comprehensive coverage, new difficulties are introduced to satellite broadcasting. Therefore, it is appropriate to examine worldwide conceptual and operational satellite sound broadcast systems to delineate these difficulties and to continue to examine the practicality of worldwide sound broadcasting by satellite. This will clarify the more subtle operational difficulties of satellite sound broadcasting and provide guidance to the more favorable broadcast bands and technologies to use.

1.2 PROGRAM OBJECTIVE

The objective of this study was to provide the data necessary to develop technical, schedule, and cost data to aid in evaluating alternatives for satisfying future international sound broadcasting needs of the U.S. Government.

Conventional terrestrial broadcasting techniques were excluded from this study. Satellite system concepts were synthesized and optimized for operation in each of four bands: 15.1-26.1 MHz, 47-68 MHz, 1.5 GHz, and 11.7-12.5 GHz. The technical and operating characteristics of the space segment were studied in sufficient detail to demonstrate technologically feasible and cost-effective launch, deployment, and operational capabilities; critical technologies were identified; project plans were prepared defining tasks and providing estimates of schedules and costs to construct and operate such systems. Project plans were separately addressed for the technical, schedule, and cost elements of development efforts required in each of the critical technology areas. Alternative approaches were developed that reduce risk and schedule associated with the development of these critical technologies. Systems costs (development, construction, implementation, and operation) and their associated funding profiles were delineated in sufficient detail to separately facilitate life-cycle and cost-effectiveness comparisons.

Also, the technical and operating characteristics of the telemetry, tracking, and control station and the associated feeder link were defined in sufficient detail to develop estimates of technical, schedule, and cost data for this segment. Global service coverage combined with centralized system control and program feed from the U.S. or its territories is a desirable system feature.

Program outputs are summarized in Table 1.

TABLE 1. - PROGRAM OUTPUTS

- For several sets of operating requirements, what are the most cost-effective satellite system concepts?
- What is the impact on selected systems concepts of variations in the operating requirements?
- What critical technology must be developed for the various sound DBS system options? What are the estimated development costs & schedule?
- What are the cost & schedule risks in developing the sound DBS system options?
- What is the least costly implementation approach to each of the sound DBS system options?

1.3 PROGRAM REQUIREMENTS

The VOA requirements include specification of zones to be covered, universal time coordinated (UTC) times and number of channels, frequency of operation, and power flux density (PFD). A variety of options were also studied to provide a broad data base to not only study system designs but also to provide insight into optional system requirements.

Figure 1 pictorially describes the 15 zones of interest. The broadcast requirements for the zones are presented in Figure 2. Times are presented in 15-minute increments (UTC times) for a 24-hour day. For Ku-band, L-band, and HF-band, all zones were to be covered. As a baseline for VHF-band, only Zones 9, 10, 12, and 14 were to be covered.

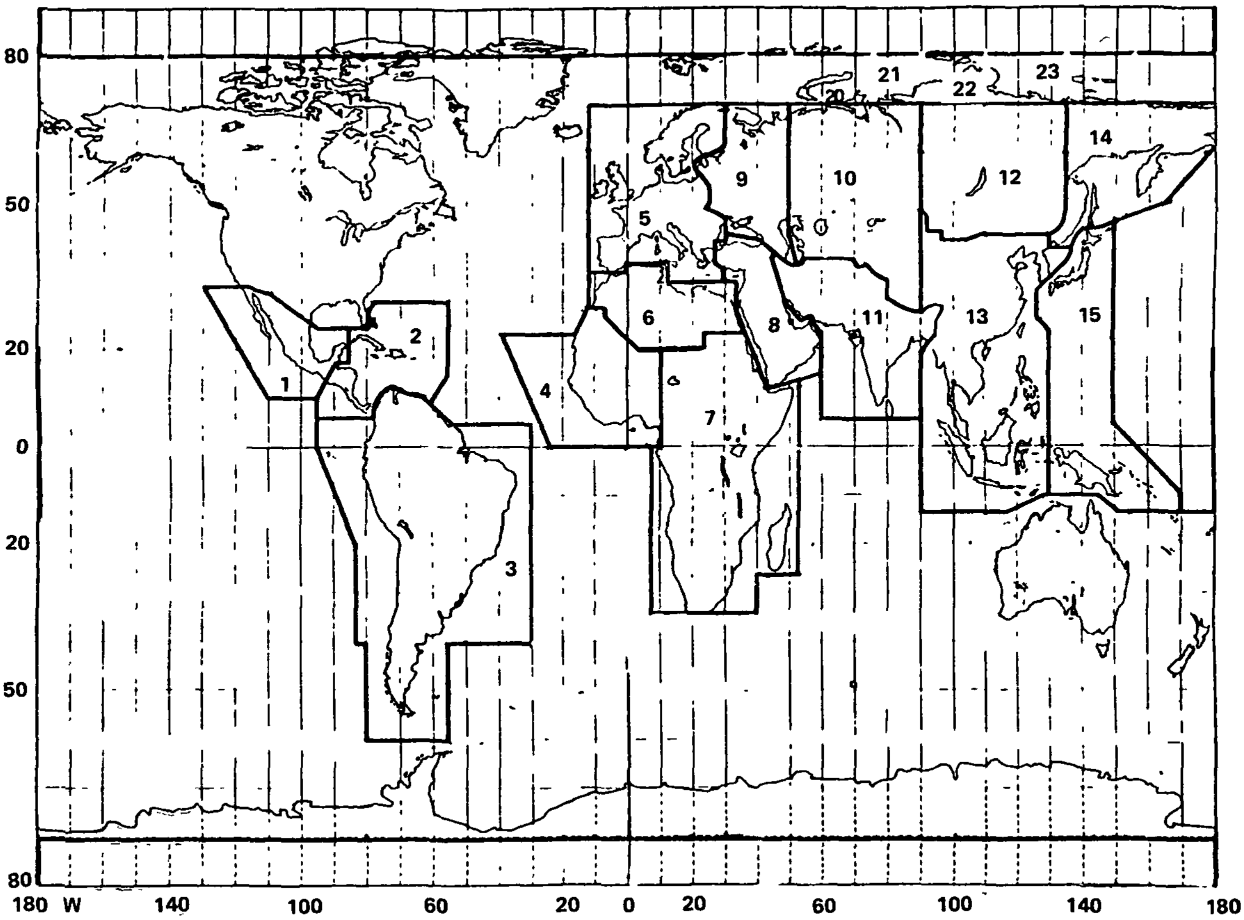

Figure 1. - VOA coverage zones.

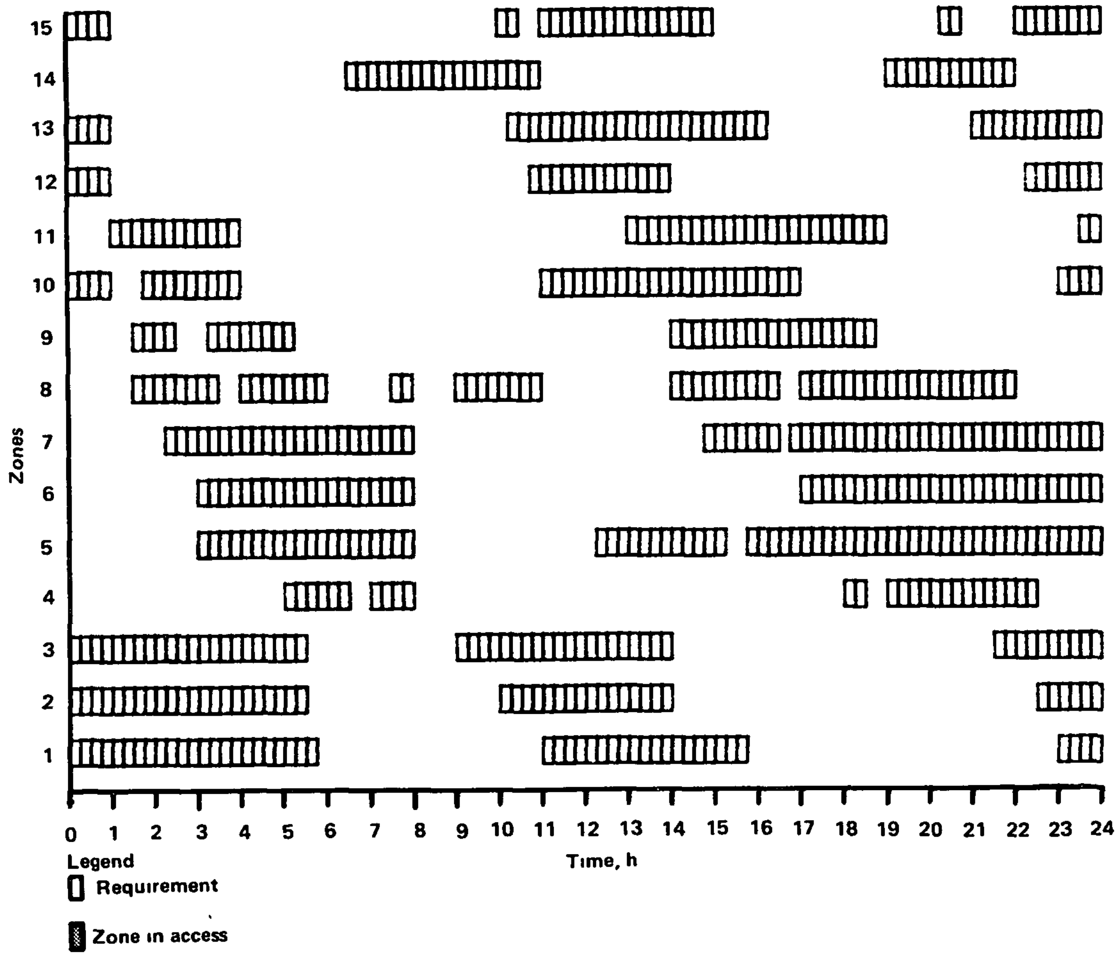

Figure 2. - VOA 24-hour broadcast requirements.

Table 2 presents the program requirements for Ku-band including frequency of operation, zones, maximum simultaneous channels and signal strength. No options were evaluated for the Ku-band system.

Table 3 presents the program requirements for L-band. Three signal levels were initially specified, however, due to high power requirements on the satellite for power levels P_1 and P_3, emphasis was placed on the P_2 level with a high and low power requirement.

TABLE 2. - PROGRAM REQUIREMENTS—KU-BAND: 11.7 GHZ

Zone	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
No. of channels	2	2	2	2	11	3	4	2	6	3	4	2	6	2	1
— Signal level. -128 dBW/m^2/4 kHz (maximum)															

TABLE 3. - PROGRAM REQUIREMENTS L-BAND· 1.5 GHZ ± 25 MHZ

Zone	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
No of channels	2	2	2	2	11	3	4	2	6	3	4	2	6	2	1

– Signal level P_1, P_2*, P_3

P_1 – (-91 2) Power flux density required to achieve an acceptable signal in a portable receiver or a receiver in an automobile Obtain this value as follows.

$P_1 = 107 + 20 \text{ LOG } f + M$

where $M = 12.5 + 0.17f - 0.17\phi + 1\ 65\ [6\ 4 - 1\ 19f - 0.05\phi]$

f = Frequency in GHz

ϕ = Elevation angle of satellite in degrees

P_2 – (-103 6) Power flux density sufficient to achieve 49 dB demodulated S/N ratio with a receiver inside a single family dwelling making use of an outside antenna.

P_3 – (-92.6) Power flux density sufficient to achieve 49 dB demodulated S/N ratio with a receiver & antenna inside a single family dwelling having an 11 dB wall attenuation.

*P_2 selected for satellite parametrics at two power levels (-103.6 dBW/m^2 & a less conservative -116 1 dBW/m^2) P_1 & P_3 power levels not achievable

Table 4 presents the program requirements for VHF-band. Only Zones 9, 10, 12, and 14 were specified for coverage. Three power levels were initially specified (250, 1000, and 5000 μV/m). The 1000 and 5000 μV/m signal levels were not achievable so program emphasis was placed on 250 μV/m with a 150 μV/m option and reduced channel options. A single orbiter was specified as the baseline but an option using a satellite in one orbiter and a large Centaur-type stage in a second orbiter was also considered.

TABLE 4. - PROGRAM REQUIREMENTS — VHF-BAND 47-68 MHZ

Zone	9	10	12	14
No of channels	6	3	2	2

– Signal level: 250, 1000*, 5000* μV/m FM
– Optional systems studied
  – Reduced channel requirements (selective reduction)
  – Reduced signal level. 150 μV/m
  – Satellite using full orbiter

*1000 & 5000 μV/m were not achieveable (150 & 250 μV/m were emphasized in program)

Table 5 presents the program requirements for HF-band. Three power levels were initially specified: 300, 500, and, 1000 μV/m. The 500 and 1000 μV/m signal levels were not achievable so program emphasis was placed on 200 μV/m with a 150 μV/m option. Reduced channel requirements and two reduced zone coverages were also to be evaluated. Single spacecraft in six different orbits were also evaluated at three signal levels for both double sideband (DSB) and single sideband (SSB). A full orbiter spacecraft was also investigated to provide greater capability on a single satellite.

TABLE 5. - PROGRAM REQUIREMENTS HF-BAND· 15.1-26.1 MHZ

Zone	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
No. of channels	2	2	2	2	11	3	4	2	6	3	4	2	6	2	1

– Signal levels: 300, 500*, & 1000* μV/m double sideband (DSB)
– Optional systems studied
  – Reduced channel requirements (six-channel max, one-channel max, selective reduction)
  – Reduced signal level. 150 μV/m
  – Small single spacecraft (DSB & single sideband [SSB]), 50, 150, 300 μV/m
  – Reduce coverage to 40° N.& 15° S. Lat. lat.
  – Reduce coverage to 40-70° N. & 15-60° S. lat.
  – Satellite using full orbiter

*500 & 1000 μV/m were not achievable (150 & 300 μV/m were emphasized in program)

2.0 SURVEY OF NONTERRESTRIAL BROADCAST TECHNIQUES

This survey identified and described existing and planned nonterrestrial broadcast techniques. Top-level analyses were performed on each technique to determine its feasibility for use as a sound broadcasting system.

In this introduction, it is useful to say a few words about why nonterrestrial transmission methods are superior to terrestrial methods. Radio signals in general propagate via a direct space wave out to a distance within the radio horizon, and via a surface wave considerably beyond this horizon. For terrestrially located transmitters, however, the horizon is only on the order of 65 km (40 miles) for tower or terrain elevations of up to 305 m (1000 ft). Although a reflected space wave can propagate over considerably greater distances via reflection or "skip" conditions, coverage is not continuous over the land, and varies considerably with time of day and sunspot activity. On the other hand the surface wave (also known as the ground wave) experiences a loss resulting from ground absorption in addition to its spreading loss of space wave propagation. This ground absorption loss increases with frequency, and therefore is not very useful for frequencies above 10 MHz. For example, over rich agricultural land with low hills, the absorption loss at 10 MHz and range of 100 km (62 miles) is about 70 dB. At 20 km (12 miles) the loss is about 90 dB (ref. 6).

If the antenna is elevated to heights available from balloons and powered heavier-than-air aircraft, or even more so to heights available from satellites, the radio horizon distance is considerably increased, and propagation over substantial distances via the space wave is possible.

The studies for nonterrestrial techniques have shown that while various nonorbital techniques can provide coverage, they suffer from some severe drawbacks. Most notably, they can only cover the edges of unfriendly territory, and many are required to cover an entire friendly zone. The number of individual signal sources raises the concern that there will be areas of interference where individual coverages overlap. Even in friendly territory, the need for logistics support for each platform can make the system nonviable.

Existing orbital communication systems operate at geostationary altitude. Equivalent coverage at HF and VHF would require very large antenna apertures. Also, the high-power requirements for these HF and VHF systems would be larger than any existing system. In the L-band, a DBSC satellite design could provide adequate power for the low-end power requirement and would provide beam sizes of the correct order of magnitude. In the Ku-band, several satellites designed for TV direct broadcast applications, (e.g, the Japan-Broadcasting satellite, the Hughes HS394, and the DBSC satellites, could be used; however, some modifications to the spacecraft antenna would be required. Also, an SBS-type satellite could provide adequate power levels and the proper size beams for Ku-band operation.

2.1 TECHNIQUES AND COVERAGE

To obtain some idea of the power that will be required for the VOA application, the VOA specified edge of coverage signal level was converted into an equivalent power required, per voice channel, for each band, for the largest and smallest areas to be covered. These areas were, respectively, South America (Zone 3) and the Eastern Europe region (Zone 9). This was done for various ground receive antenna, elevation angle of 20°. The approximate range of values for each band is summarized in Table 6.

The data contained in Table 6 assumes straight line propagation with no allowance for atmospheric or ionospheric losses. By the nature of the problem, these power requirements are, except for atmospheric or ionospheric considerations, independent of transmitter altitude, and thus apply to both nonorbital and orbital methods of coverage. It is obvious that with the exception of the power requirement for the Ku-band, and the lower range of L-band, no existing satellites can satisfy these power requirements. Discussion on what levels of power could be supplied by various techniques is in subsequent sections of this report.

TABLE 6. - VOA POWER REQUIREMENT RANGE (PER VOICE CHANNEL)

Band designation	Frequency range	Specified EOC signal level	Required power range
HF	15 0 - 26 0 MHz	300 μV/m	5 9 kW - 29 5 kW
VHF	47 0 - 68 0 MHz	250 μV/m	4.2 kW - 20 9 kW
L	1.5 GHz	P2	450.0 W - 2.2 kW
Ku	11.7 - 12 7 GHz*	-131 dBW/m^2	2.0 W - 10.0 W

*Maximum range per single transponder is 11.7 to 12.5 GHz.

2.1.1 Nonorbital Methods of Coverage

This section examines three nonorbital methods of providing nonterrestrial originated coverage. The three methods are:

1) Tethered lighter-than-air platforms,

2) Powered lighter-than-air platforms,

3) Powered heavier-than-air platforms.

2.1.1.1 Tethered Lighter-Than-Air Platforms

Low altitude (nonorbital) vehicles have an advantage over satellites in that they are retrievable, and can thus be repaired, or have their payloads changed. However, since they are susceptible to destruction in unfriendly territory, they are useful only in friendly territory. In addition, nonorbital vehicles also have a much lesser broadcasting range than satellites, so that many such vehicles must be used to cover one broadcast zone. The resulting segmentation of coverage results in a potential for undesirable interference in the regions where signal strengths from two or more sources of the same frequency are roughly equivalent. This could be a serious drawback to a highly segmented system, and should be studied further if such a system is to be considered.

The geometric line-of-sight coverage of low altitude platforms is, to a first order approximation, a function of the square root of altitude or elevation above the surrounding terrain. For radio frequencies in the approximate range of 100 MHz to 20 GHz, however, the radio propagation range is about 15% further than the geometric line of sight because of atmospheric refraction effects. These geometric line of sight and radio propagation distances (as a function of altitude) are shown in Figure 3.

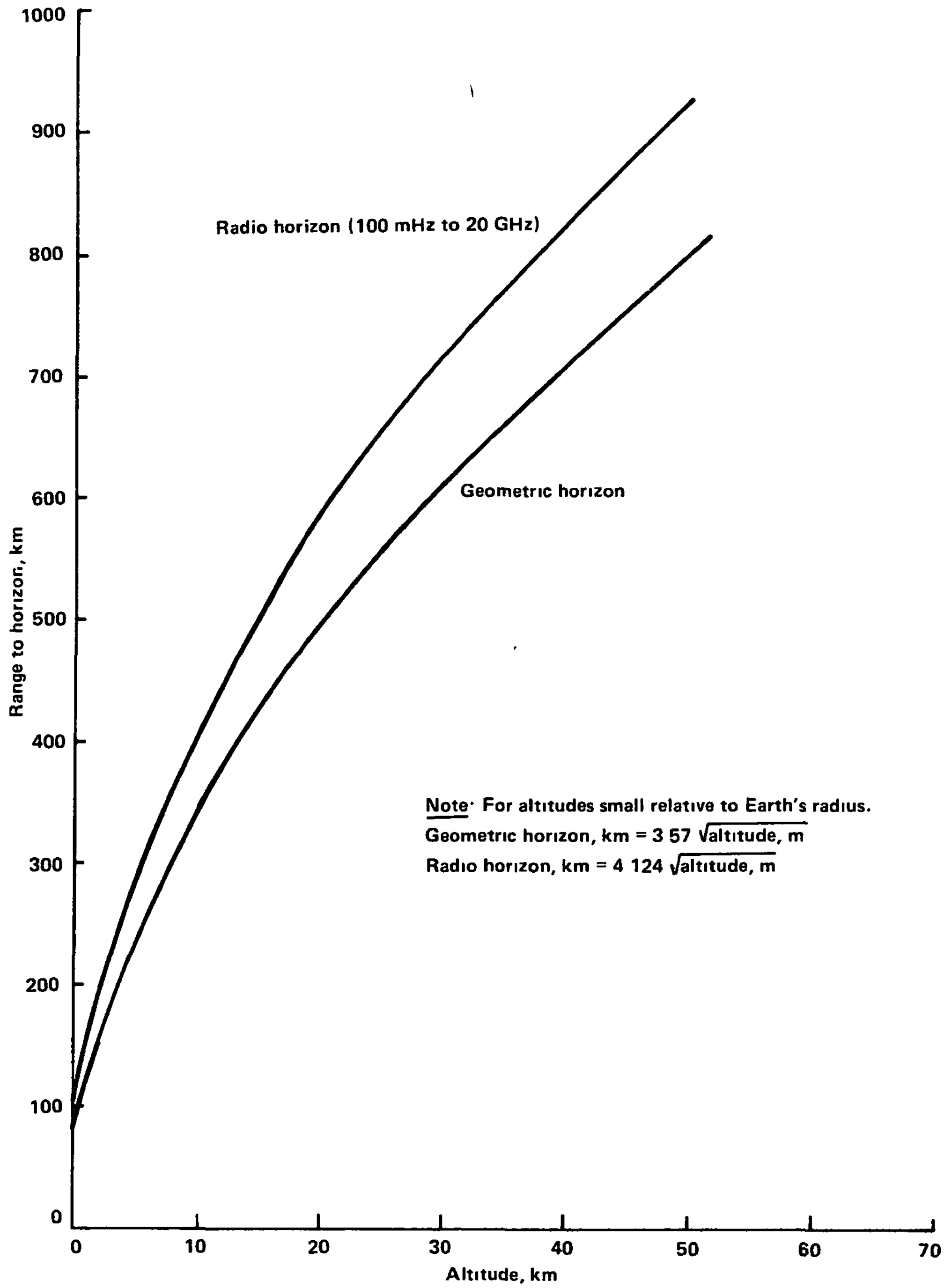

Figure 3. - Geometric and radio horizons versus altitude above terrain.

The tethered aerostat is a developed product, having already been used in broadcasting applications. Tethered aerostats can operate to altitudes of about 6000 m where they can carry payloads of 200 kg. At lower altitudes they can carry significantly heavier payloads. At 4500 m, line-of-sight distance is about 240 km (149 statute miles), as shown in Figure 3. To obtain reasonably full coverage of a region, tethered aerostats at this altitude would have to be placed in a grid with separations on the order of 500 km. For example, to cover a region such as Region 4 (Western Africa) which is roughly 4300 by 2700 km, an array of aerostats roughly 8x6 (on the order of 50 aerostats) would be required. (Tethered aerostats range in size from about 1400 m^3, 35 m long to about 17,000 m^3, 85 m long.)

The principal systems designer of tethered communications balloons is Tethered Communications Corporation, (TCOM) a subsidiary of Westinghouse Electric Company. Their product line generally falls into two categories:

1) A small, trailer-based transportable system capable of lifting 100 kg to an altitude of about 760 m (2500 ft) above sea level (ref. 8).

2) A permanent, installation-type system capable of lifting about 100 kg to 1830 m (6000 ft), or about 1000 kg to 1525 m (5000 ft) (ref. 8).

The payload for a tethered aerostat is suspended beneath the aerostat in a separate compartment. Systems built about ten years ago (refs. 9, 10, 11, 12) carried their own communications system power source, typically with the use of Sachs-Wankel rotary engines with a generating capacity of up to 5 kW. Under full-time use, fuel for the engines would typically last up to one week. More recently, power has been provided via the tether cable. This is accomplished by carrying high voltage, so as to minimize resistance losses, and stepping it down at the aerostat.

For a communication system, the problem of antenna orientation stabilization can be solved basically in one of two ways. The first way is to eliminate the problem by having an antenna pattern that is symmetrical about the vertical axis. In this case, rotations of the aerostat, with changing wind conditions, will not affect the coverage. The second method is to use an airborne mechanical system consisting of a two-axis gimbal, an azimuth drive, and a slip ring assembly package. The gimbal assembly acts as a pivot at the bottom of the aerostat hull from which the entire airborne payload is suspended in a pendulum fashion. Each axis is damped by a rotary viscous damper. The upper linkage on the gimbal assembly is mounted to the aerostat through a lightweight truss structure that distributes the airborne package weight and inertial loads throughout the balloon skin. The fixed shaft of the azimuth drive (with respect to the aerostat) is attached below the lower gimbal linkage. The azimuth drive is the mechanical portion of the azimuth heading servo loop. The drive system receives an electrical signal from the servo electronics and converts it into mechanical rotation of the payload package to maintain proper heading with respect to north, as the aerostat moves. The slip ring assembly incorporated into the airborne package allows unrestricted azimuth motion between the payload and the aerostat. The ring is located at the upper end of the azimuth drive where it is attached to the lower linkage of the gimbal. An azimuth positioning of $\pm$ 0.5° pointing accuracy, controllable in 0.1° increments, is achieved. The gimbal assembly isolates payload motion with respect to aerostat motion by a factor of 10 to 1.

A number of methods are available for antenna mounting on an aerostat. One method is to use the gyro-stabilized platform beneath the aerostat. An alternate method is to suspend the antenna inside the balloon, a method particularly applicable to long wire antennas such as might be required in HF- and VHF-bands. A third and as yet unproven idea is to create an antenna from a section of the balloon's surface allowing for the widest possible antenna dimensions.

Several means exist for transmitting the programming material to the aerostat:

1) It may be uplinked from a transmitter at or near the aerostat mooring site, via RF,

2) A microwave link may be used to carry the signal to the aerostat from some distance away. TCOM Corporation did this with an installation in Iran, using a microwave link 193 km (120 miles) long to its Mark VII balloon,

3) A fiber optic link inside the cable can be used. TCOM has used this method on smaller (700 m^3) STARS aerostats.

In summary, although tethered lighter-than-air balloons have advantages in that they are retrievable, are a developed product, require no fuel to remain aloft, and can be powered through the tether, they also have an obvious disadvantage in that they are extremely vulnerable to attack and must be organized in a grid pattern with the potential for radio interference in the intersecting regions.

2.1.1.2 Powered Lighter-Than-Air Platforms

Powered lighter-than-air vehicles (e.g, blimps, dirigibles, and airships) are possible, but not likely candidates for consideration as broadcasting platforms. They do however have a feature not realized with tethered aerostats, namely their ability to maneuver. However, this feature is probably not required for a VOA broadcasting system. On the negative side, conventional blimps do not have a tether, which, in aerostat systems, in addition to providing station keeping, also provides a means to transmit power to the broadcasting equipment. Without a tether, power must now be carried for both the broadcasting equipment and the station keeping engines of the aircraft. This severely limits the duration of continuous flights to the order of hours rather than the many days that are possible with tethered vehicles.

An alternate concept for the provision of power is to install a rectenna on the underside of the airship to beam energy to it from the ground. No working systems of this kind are now known.

The weight capacity of a dirigible is similar to that of a tethered aerostat, except that the means to generate power must now be counted as part of the payload.

For transmission of program material to powered lighter-than-air vehicles, the first two methods mentioned for tethered aerostats are applicable, leaving out the possibility of a fiber optic link.

In conclusion, powered lighter-than-air vehicles share many advantages with tethered balloons, (e.g, retrievability and buoyancy) and have an added advantage in that they are maneuverable. Unfortunately, they also have all the disadvantages found in tethered balloons (e.g vulnerability to attack and interference between broadcast platforms, as well as having to generate required power on board). Also important to consider is that although the technology required to build dirigibles is available, only a limited number of designs have been developed, designs that may not suit payload capacity and service ceiling requirements.

2.1.1.3 Powered Heavier-Than-Air Platforms

The usefulness of powered heavier-than-air vehicles, including fixed-wing aircraft and helicopters have been investigated. The results are not promising.

In this category, the helicopter has the advantage over fixed-wing aircraft in that it can operate without a landing strip, and can take off vertically and reach its desired position directly. This could be advantageous in remote mountainous coverage zones.

The largest helicopter on the market and also one of the most expensive is the Boeing UT234. Starting at approximately $17M for the basic aircraft, the UT234 (ref. 13) will also cost an estimated $3800 per flight hour to run. This vehicle sports a fairly impressive range of payload and service ceiling capabilities--from 7710 kg (17,000 lb) at 4570 m (15,000 ft) to 12250 kg (27,000 lb) at 2130 m (7,000 ft) as illustrated in Figure 4.

Another factor to keep in mind when considering helicopters is the close relationship between helicopter performance and air temperature. During periods of warmer weather, there is a notable decline in the service ceiling due to variation in air density. Figure 4 indicates that for a specified service ceiling, a 20° variation in air temperature can cause up to a 16% decrease in payload capacity.

Fixed-wing aircraft generally have much higher load and altitude capability than helicopters. A wide variety of midsized airplanes, from executive jets to propeller driven transport planes, are easily capable of doing the same job as the UT234 with power to spare. The greatest drawback with airplanes is not payload weight capacities, but limits on the size and shape of the broadcast antenna. The drag created by a 10-m dish, for example, would cripple all but the largest of these aircraft. This problem has

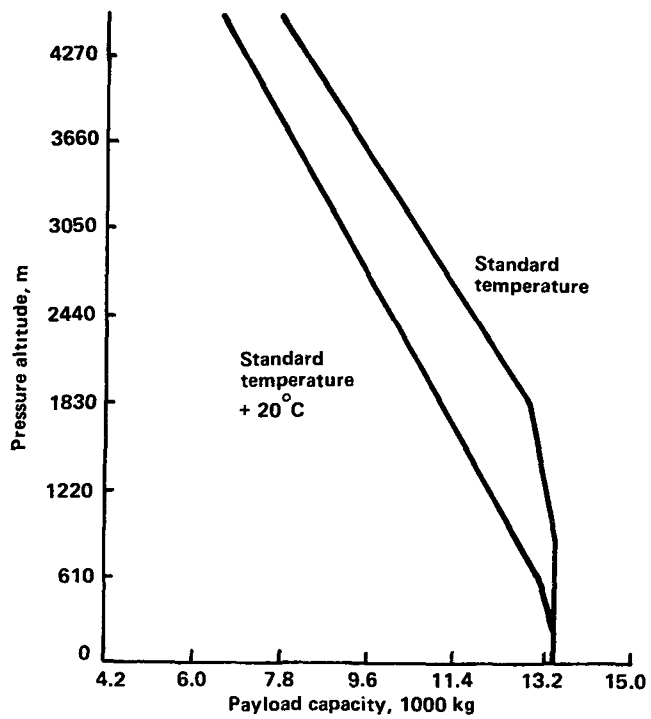

Figure 4. - UT234 hovering ceiling.

given rise to creative antenna designs that reconcile aerodynamics and broadcast efficiency. Examples of these are the disk-shaped antennas like those mounted on the Boeing 707 AWACS planes, and long trailing antennas developed by Lockheed.

The Lockheed C-130, a 4-engine turboprop, can be ordered with a 457-m (1500 ft) trailing antenna designed for broadcasting an AM signal. Used extensively in the past for broadcasting, this aircraft can carry up to 19500 kg (43,000 lb) at well above 3050 m (10,000 ft) for six or seven hours (specification provided courtesy of Wallace Robby, Lockheed Corp.). No existing helicopter can boast such impressive capabilities.

Helicopters and airplanes alike share one considerable limitation in that they can be flown only over friendly territory. Being easy targets for missile or other anti-aircraft attack, a very large percentage of the proposed coverage areas would be off limits to these vehicles. On the other hand, friendly territories can be serviced by radio towers (for years, the method preferred over airborne broadcast platforms).

Unlike lighter-than-air platforms, heavier-than-air platforms must continuously burn fuel to remain aloft. Fuel reserves set aside for this task would add considerably to a payload already made heavy by power generating fuel and equipment.

It is difficult to escape the conclusion that although heavier-than-air broadcast platforms are technically feasible, they do not represent an attractive alternate to the satellite or other conventional systems. Their two apparent assets, retrievability and maneuverability, seem far outshadowed by their numerous shortcomings:

1) Vulnerability to attack,

2) Must expend fuel to remain airborne,

3) Must generate all power onboard,

4) Limited weight and hovering capabilities,

5) Can remain airborne for only short periods of time,

6) Can use only specialized broadcast antennas.

2.1.2 Orbital Methods of Coverage

Various classes of satellite orbits are useful as broadcast platforms. Among these, equatorial satellite orbits (e.g, geostationary, geosynchronous, nonsynchronous, circular, or elliptical) form a special class in that they have no specific equator crossing, and hence no right ascension of their ascending node. This means that systems using circular equatorial orbits need not be concerned with precession of ascending node, as it does not exist. Elliptical equatorial orbits, however, would be concerned with the longitudinal location of the argument of perigee. All other satellite orbits are inclined, and must consider ascending node right ascension and its drift.

We will consider the various classes of satellite orbits and the characteristics that apply to VOA broadcasts, and give examples of existing or close-to-existing technology that are applicable.

2.1.2.1 Equatorial Orbit Characteristics

The simplest and most common equatorial orbit is the geostationary Earth orbit (GEO) with a period of one sidereal day, i.e, 23 hours, 56 minutes, 45 seconds. This orbit can provide uninterrupted coverage for a given region of the Earth up to $\pm$ 70° latitude. Almost all communication satellites are in geostationary orbit.

For lower altitude equatorial orbits, the orbital period is shorter, so that the satellite advances or creeps to the east. Thus, its coverage area also advances to the east. At submultiples of the day, the satellite will be over the same longitude.

Satellites below geostationary altitude have an applicable characteristic for VOA broadcasts in that they can cover different areas at different times of the day. Such orbits will be discussed later in this report.

2.1.2.2 Inclined Orbit Characteristics

The category of inclined orbits encompasses a number of different subcategories of orbits, for such orbits can be circular vs elliptical, geosynchronous vs nongeosynchronous, subsynchronous vs non subsynchronous, Sun synchronous vs non Sun synchronous, and of any angle of inclination (0 to 180°.

Satellites in circular orbits circle the Earth at a uniform rate sweeping the ground at the same uniform ground rate. Such orbits can be useful when a uniform rate of procession measured along the ground track is desired.

Elliptical orbits in the vicinity of the apogee have relatively low lineal and angular velocities, making them useful in applications where coverage is desired in one region for a longer period of time than that required for other regions, and that which is possible with a circular orbit of the same energy. To maintain the longer coverage time over the same geographical area on successive orbital revolutions, the line of apsides (the line from the Earth's center to the argument of perigee) must be restrained from rotating.

In a conventional sense, subsynchronous orbits map out the same ground track on the Earth every day. To do this, the orbital period must be an integral submultiple of a sidereal day corrected for orbital procession. The total time of orbits in one day is defined as the orbital plane day (OPD). Thus, for a satellite that loops twice around in a nominal 24-h period, and whose orbit precesses 0.9856° per day west, the OPD is 23 hours and 52 minutes, and the orbital period of the satellite is one half this value. In a more general sense, a subsynchronous orbit is any orbit whose ground track repeats with a predictable regularity. The orbit period of a satellite in such an orbit is the associated OPD multiplied by a rational number. The numerator of the rational number is the number of OPDs that occur before the ground track repeats, and the denominator of the rational number is the number of revolutions of the satellite in its orbit that occur before the ground track repeats. If, for example, the satellite period is OPD x 2/9 (4.5 revolutions per OPD), the ground track will repeat on every two OPDs, coincident with every nine orbital revolutions. On alternate days, the ground track will lie halfway between the components of the tracks made the previous day, and the two-day successive adjacent ascending nodes of right ascension of the orbit will be out of time phase by one-half an OPD. This more general type of subsynchronous orbit is mentioned here because such orbits may indeed be useful for VOA coverage.

A geostationary orbit is a special case of a subsynchronous orbit, in which the satellite period and the OPD are one sidereal day, the orbit is circular, and the inclination is zero.

It is important to note that although subsynchronous orbits repeat their ground tracks, they do not, in general, repeat at the same time on corresponding revolutions. Sun-synchronous orbits (but more precisely, Sun stationary orbits) are those that precess one revolution to the east per year. This maintains the plane of the orbit at a constant average angle relative to a line between the centers of the Earth and Sun, and the OPD when referred to a Sun-synchronous orbit is exactly 24 hours. The term average allows for the nonuniform motion of the Earth about the Sun, described by the equation of time. Sun-synchronous orbits, in themselves, do not require that their synchronous orbital period be a rational number multiple of the 24-h sidereal orbital day (SOD). They may be of any period which, in combination with the other orbital parameters, results in an eastward precession of the ascending node of one revolution per year. Orbits that are both subsynchronous and Sun synchronous may be useful for VOA applications.

Posigrade orbits are those whose satellite motion is to the east, the same as the direction of the rotation of the Earth. The inclination angle of such satellites is between 0 and 90°. Because of Earth's motion, satellites in such orbits require less energy for launch than do satellites in retrograde orbits whose satellite motion is to the west, with the Earth is motion hindering launch. Retrograde orbits have inclination angles between 90 and 180°. Orbits near 90° inclination are know as polar orbits. Their ground track extends to the polar regions. Since VOA does not require polar coverage, and further, since coverage to high latitudes can be provided by satellites of only moderate inclination, they appear to be inefficient orbits. They do, however, have a unique characteristic in that they can satisfy the conditions for circular, subsynchronous orbits. This is discussed further in the next section.

2.1.2.3 Some Special Orbits

1) An elliptical, subsynchronous orbit,

2) A class of circular, subsynchronous, Sun-synchronous orbits,

3) An elliptical, subsynchronous, Sun-synchronous orbit.

An elliptical, subsynchronous orbit. - An example of an elliptical, subsynchronous orbit is provided by the Molniya series of Russian satellites. These satellites are in highly elliptical orbits that precess to the west, and have a period of one-half of 23 hours, 56 minutes minus an allowance for precession (one half an OPD). The satellites are in an inclined orbit of 63.4°. The main characteristic of this angle and its complement of 116.6°, is that the line of apsides does not rotate. Thus, the irregular orbital pattern that results from the high eccentricity is held stationary in position, although not in time, so that the relatively long dwelltime at apogee is maintained over the same area of Earth. Figure 5 shows the ground track of a typical Molniya type trajectory.

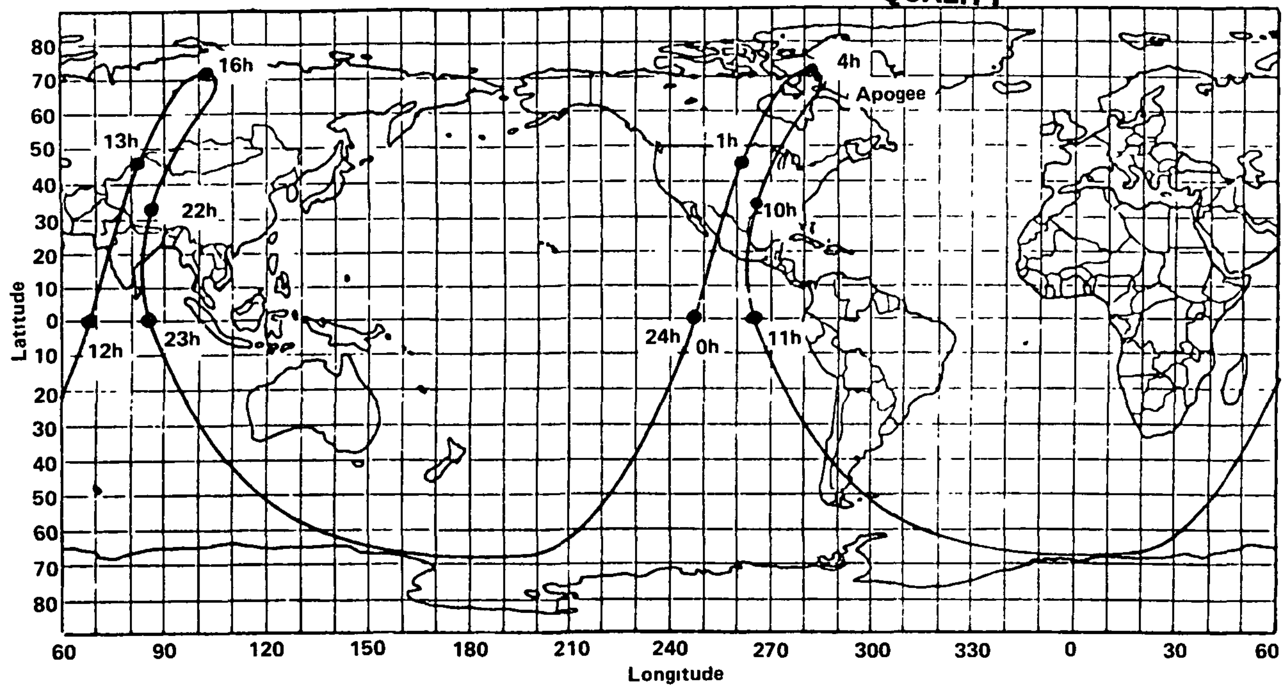

Figure 5. - Molniya typical ground trace—inclined orbit, 12-hour period.

Circular, subsynchronous, Sun-synchronous orbits. - An interesting class of orbits is the Sun synchronous, subsynchronous circular type. This class differs from mere subsynchronous orbits in that the ground track is repeated at the same clocktime, making them geosynchronous. The class differs from elliptical orbits in that there are no restraints on inclination angle, allowing rotation of the line of apsides. Without this restraint, there are a number of orbits of interest.

Table 7 lists characteristics of various satellites that have been launched.

TABLE 7. - EXAMPLES OF SATELLITES IN CIRCULAR, SUBSYNCHRONOUS, SUN-SYNCHRONOUS ORBITS

	Revolutions per solar day	Orbital altitudes	Inclination angle	Launch date	Mass
NOAA weather satellite	12.39	1510.0, km	101.9, deg	1976	340 0 kg
NOAA weather satellite	12 59	1450.0	101.4	1974	340.0
NIMBUS 6 weather technology	13 41	1106.0	99 8	1975	829 0
LANDSAT	13 97	905.0	98.8	1972	816.4
NOAA-7	14.13	848.0	98 9	1981	1405.0
DMSP-F3	14 24	811.0	98.6	1978	513.0
HCMM	14.83	623 0	97.6	1978	134.0
METIOR 1-29	14 91	595 0	97 9	1979	3800 0
SOLWIND P78-1	15 06	545 0	97.6	1979	1331.0
UOSTAT	15 08	539.5	97 8	1981	52 0

Elliptical, subsynchronous, Sun-synchronous orbit. - The next class of satellite trajectories contains a number of potential subclasses that are similar to the Molniya subsynchronized class, but, in addition, are in Sun-synchronous trajectories. To accomplish this, the inclination angle is 116.6°, and the other orbital parameters are defined such that the orbit drifts to the east one revolution per year, and the subsynchronous also becomes Sun synchronous. It turns out, however, that of all the potential subclasses of such orbits, only one is realizable.

2.1.3 Satellite State of the Art

This section looks at designs of some existing satellites, and some that are presently in a design phase, to determine to what extent such technology could reasonably satisfy VOA requirements. Before considering specific examples, it is well to note a general limitation in terms of satellite mass required per unit of primary power available. A survey of some satellites (other than experimental) shows a range of roughly 250 to 1400 grams per watt. The broadcast type satellites tend to cluster in the range of 350 to 475 grams per watt, while the fixed service satellites tend to cluster in the range of 600 to 1250 grams per watt.

2.1.3.1 The Applications Technology Satellite-6 (ATS-6)

The ATS-6 was launched in May 1974 to perform a number of experiments. One primary objective was to demonstrate the feasibility of deploying a 9.1 m (30 ft) parabolic reflector antenna. Its initial orbital mass in geostationary orbit was 1350 kg, and had 645 W of solar power for a relatively inefficient mass to primary power ratio of 2093 grams per watt. Several transmitters at frequencies ranging from 860 MHz to 315 GHz (including a 40 W transmitter at 1.55 GHz) with RF powers of up to 80 W were employed with the large parabolic reflector.

If the full resources of an ATS-6-type satellite were converted to VOA applications, the large parabolic reflector could provide the following beamwidths:

Frequency	Beamwidth
15.0 MHz	154.80°
26.0 MHz	88.80
47.0 MHz	49.10
68.0 MHz	33.90
1.5 GHz	4.0 (1/4 illumination of dish)
1.5 GHz	1.54 (full illumination of dish)
12.0 GHz	0.19

The beamwidths from 154.8° down through 33.9° clearly would not be efficient from geostationary altitudes, as the Earth extends only about 17° from this altitude.

If, in addition, the full primary resources were devoted to a single transponder voice channel, it is estimated that about a quarter of the 645 W available, or 160 W, could be available for transponder output. This would not be sufficient to produce the power needed in HF-, VHF-, and L-bands regardless of satellite altitude. In Ku-band, the available power would be sufficient but the beam size would be too small at geostationary altitude, and the antenna size would have to be reduced to meet coverage requirements. For HF- and VHF-bands, the antenna size from geostationary orbit provides whole Earth coverage. However, unless it is coupled with much higher power, the PFD on the ground would not meet the VOA requirements. Such higher power might then be able to power several channels simultaneously.

2.1.3.2 Japan Broadcasting Satellite

In April 1978, NASA launched for Japan a TV broadcast satellite (BS). This satellite has a mass of 678 kg, and a primary power of 1000 W (678 kg/kW). It has two 100 W transponders to handle two simultaneous TV broadcasts. A more recent BS-2 has increased the primary power to 1780 W, for a somewhat more efficient mass-to-power ratio. The antennas on the BS were 1 by 1.6 m, and produced a beam of approximately 2 by 1.4° at 12 GHz with 40.3 dB peak gain.

Satellites with the above parameters, as in the case of the ATS-6 technology, do not have sufficient power to be of use in HF- and VHF-bands. With a larger antenna, the stated power would be sufficient for two L-band voice broadcasts. However, the satellite would require significant redesign to be compatible with such an antenna. At Ku-band the antenna change could easily be accomplished and the resulting satellite could broadcast 12 channels or more.

2.1.3.3 Hughes DBS Satellites

Hughes Aircraft has introduced a high power DBS satellite (HS 394) capable of providing eight channels, each of 160 W output, with a total RF output in the 1200 to 1500 W range. Primary power is 3-4 kW, with about 1050 kW inorbit, for a mass-to-primary power ratio in the neighborhood of 300 kg/kW. Unlike conventional Hughes spin-stabilized satellites, the solar array is Sun stabilized, although the propulsion and attitude control section is spun. The antenna is sized to cover one half of the Continental U.S. at 12 GHz, resulting in a beam of about 3° in diameter, covering about three times the ground area of the Japanese BS.

As in the case of the Japan BS, power is insufficient for HF- and VHF-bands. This satellite is large enough to accommodate a significantly sized L-band-deployable antenna. This coupled with the large power source could provide several voice channels at L-band. In Ku-band, the 3° beam might be satisfactory from geostationary orbit for coverage of an area about half the size of Zone 8, (e.g, Saudi Arabia-Turkey). Therefore, a different antenna would be needed at Ku-band and this could be easily accommodated.

2.2 COST AND SCHEDULE

This section addresses the costs and schedules of the orbital and nonorbital systems described in Section 2.1. These systems have all been developed using existing technology, with all costs reported in fiscal 1984 dollars.

Costs were broken down into three sections: (1) nonrecurring (N/R) engineering cost; (2) production cost; and (3) operation and support (O&S) cost.

The N/R engineering cost includes direct engineering, overhead, general, and administrative costs, quality control, program management, and other technical services.

Production cost was derived assuming a minimal amount of time for startup and retooling of machinery. Since only the first unit was priced, no learning benefits were applied.

O&S cost was derived using a worst case operational life of seven years and an operation time of 12 hours per day. O&S for nonsatellites include fuel, repair labor, spares, technical publications, training, consumables, inventory control, and operations labor. O&S for satellites include launch and tracking costs. Insurance, and interest were not included in any costs. Table 8 shows the cost breakdown and development and production time per unit of five nonorbital and three orbital platforms.

For each coverage zone, the number of vehicles required to give full coverage is shown in Table 9. For nonorbital techniques, this number is independent of the operating band, assuming each vehicle would carry the appropriate antenna required. For orbital techniques, the number of vehicles is shown for a specific operating band.

TABLE 8. - ORBITAL AND NONORBITAL CONCEPTS— SINGLE UNIT COST AND SCHEDULE

Concept	N/R engr	Unit cost	O&S	Dev	Prod.
Goodyear B-type blimp	$ 1M	$10M	$ 5M	0.5, yr	1.0, yr
TCOM Aerostat	1	8	3	0 5	1 0
Boeing UT-234	2	15	45	1 0	1 5
Lockheed C-130	2	15	55	1 0	1 5
Boeing 707 (AWACS)	3	25	80	1.0	1.5
Fairchild ATS-6	10	65	135	2.0	1 5
DBSC	15	90	135	2 5	1 5
SBS	10	75	125	2 5	1 5

TABLE 9 - NUMBER OF VEHICLES REQUIRED PER ZONE

Zone	Blimp	Tethered balloon	UT-234	C-130	707	ATS-6 (L-band)	DBSC (L-band)	SBS (Ku-band)
1	219	73	72	36	25	8	0 6	Approx
2	213	71	70	35	24	7	0 6	six
3	719	240	236	119	83	25	2 0	total
4	375	125	123	61	43	13	1 0	
5	469	156	154	77	53	16	1 3	
6	188	63	61	31	21	7	0 5	
7	1000	333	328	164	114	35	3.0	
8	157	53	52	27	13	5	0 4	
9	163	54	53	27	18	6	0 5	
10	284	95	93	47	32	10	0 8	
11	344	115	113	56	39	12	1 0	
12	222	74	73	36	25	8	0 6	
13	844	281	276	138	96	29	2 5	
14	159	53	52	26	18	6	0 4	
15	406	135	133	67	46	14	1 0	
Total	5762	1921	1889	947	650	201x0 5 =100*	16	

*Assumes 50% reduction for a single satellite covering multiple zones

Table 10 represents the total life cycle cost and schedule for each of the specific concepts. The total cost includes nonrecurring engineering, unit, and operations and support costs. Total schedule includes development and production schedules. Schedules were based on the assumptions of production rates of 14 aircraft per month for the winged aircraft and 20 aircraft per month for the other two nonorbital aircraft. A production rate of two spacecraft per month was assumed for orbital platforms.

TABLE 10. - ORBITAL AND NONORBITAL CONCEPTS—TOTAL LCC AND SCHEDULE, $B

Concepts	Quantity	Total cost	Total schedule
Goodyear B-type blimp	5762	$ 86	25.5, yr
TCOM Aerostat	1921	21	9 5
Boeing UT-234	1889	113	13 5
Lockheed C-130	947	66	8.1
Boeing 707	650	68	6 4
Fairchild ATS-6	100	20	9 5
DBSC	16	4	6 8
SBS	6	1	6.3

3.0 SATELLITE SYSTEM MISSION ANALYSIS

Mission analysis was performed to characterize various satellite concepts, performance parameters, and hardware implementation options applicable to the DVBS satellite service requirements. The final results were a set of candidate satellite system concepts that satisfy the service requirements and are compatible with STS payload weight and volume restrictions and the requirements of an orbital transfer vehicle (OTV).

The analysis included an orbit and coverage analysis, a propagation analysis, a payload capability analysis, and a technology survey of subsystems for DVBS. The orbit and coverage analysis encompassed a wide range of orbits from low to high altitude for both elliptical and circular orbits. The propagation analysis reviewed data available on propagation parameters and used the parameter effects to yield the losses associated with transmission at each band. The payload analysis used projected STS payload capabilities and near-term OTV's to determine satellite limitations as to the total weight and volume that could be delivered to each orbit. The technology survey of applicable subsystems considered communication, power, ACS, stationkeeping and maneuvering, TT&C, thermal control, and the equipment bay (e.g, spacecraft body and subsystems).

3.1 ORBIT AND COVERAGE ANALYSIS

The full range of possible orbits was examined systematically. Both elliptical and circular orbits were considered. In recommending orbits for DVBS, the payload that can be placed into each orbit was considered along with the orbital constraints (e.g, eclipse time, Van Allen belt radiation, and orbit perturbations).

Additionally, orbits that provide repeatable ground tracks, repeatable time schedules, and long coverage time over a particular area were desirable to meet the VOA coverage requirements. These requirements could be met by a large number of satellites surrounding the globe. However, the number was reduced by choosing orbits with repeatable ground tracks over the required coverage area. Also, the number was reduced further by assuring either the same satellite or multiple satellites repeat the time schedule (e.g, the same schedule to the same area everyday). Coverage time was increased by using constellations of trailing satellites or by using an elliptical orbit with a high apogee occurring over the target area.

In order to provide a measure of the degree at which each recommended orbit met the VOA coverage requirements, a coverage analysis was performed using representative orbital positions. Each orbital position was assumed to have a capability to view the Earth to a 20° elevation angle for HF- and VHF-bands and a 11.5° elevation angle for L- and Ku-bands. Depending on the required number of voice channels and signal strength, the orbital position consisted of either a single satellite or a cluster of satellites.

3.1.1 Orbital Constraints

The orbital constraints that provided a basis to measure which orbits were possible candidates for DVBS included eclipse time, Van Allen belt radiation, and orbital perturbations.

Power requirements for the HF-, VHF-, and L-band satellites were such that the satellite operating during eclipse would need large battery packs or the satellite could not operate at all. In either case, minimizing eclipse time was desirable. In general, the eclipse time decreases as the satellite altitude increases. As it turned out, having to operate during eclipse was such a severe requirement for battery power that satellites recommended for HF-, VHF-, and L-band do not operate during eclipse. Figure 6 shows the percent of sunlight for a satellite vs the orbital altitude as orbit inclination is varied.

In addition to eclipse time, the charged particles in the Van Allen belts can cause serious deterioration of satellite solar panels and electronics. Providing the satellite operating in the Van Allen belts with enough end-of-life (EOL) solar power requires larger and heavier solar arrays than on a satellite outside of the Van Allen belts, and the weight of shielding needed for the satellite electronics increases. It was therefore desirable to minimize or avoid the Van Allen belt regions as much as possible.

Orbital perturbations include effects due to oblateness of the Earth, drag of atmosphere, solar and lunar gravity, solar radiation pressure, and electromagnetic drag. For the most part, all but the oblateness of the Earth are effects that the satellite can easily compensate for by using the stationkeeping system on the satellite.

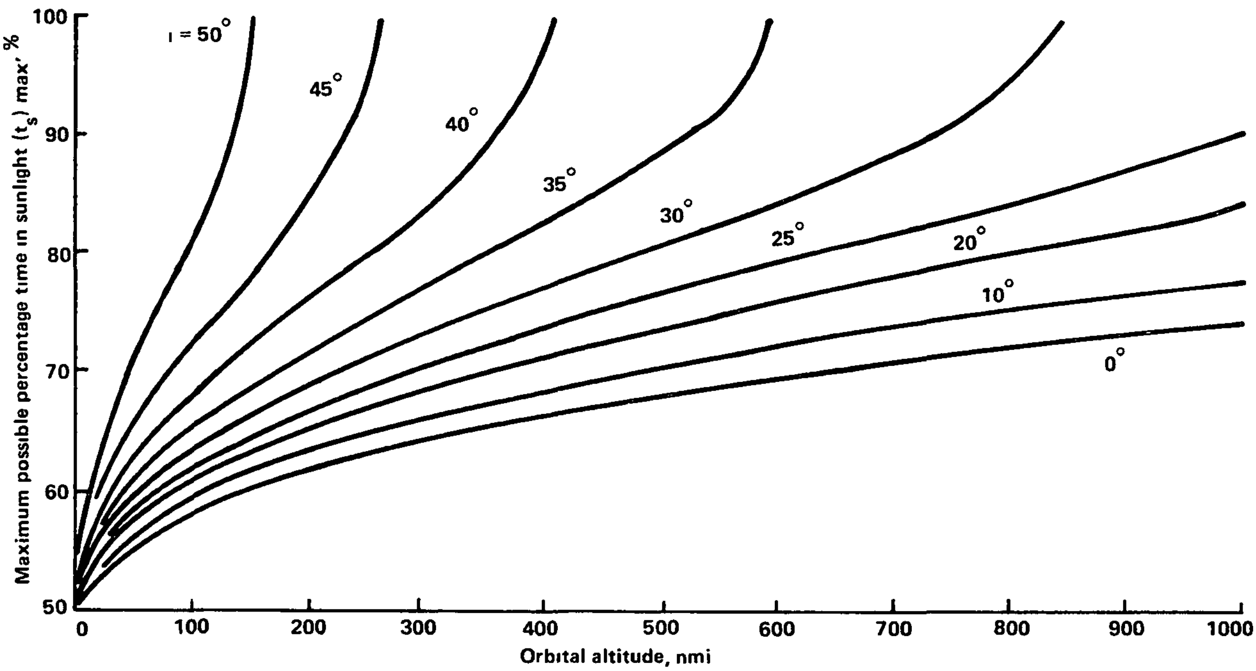

Figure 6. - Eclipse time versus orbital altitude.

The Earth's oblateness causes both periodic and steady (secular) changes in the orbital elements. The effects of Earth's oblateness were considered for each orbit and are included in the coverage analysis discussed later.

3.1.2 Elliptical Orbits

The orbital analysis conducted under this study looked at three types of elliptical orbits: Molniya, geosynchronous, and a Sun-synchronous subgeosynchronous orbit called a triply-synchronous orbit.

Elliptical orbits are used to maximize the dwelltime over a coverage zone. However, elliptical orbits usually require higher delta-v capabilities than circular orbits of the same orbital period. Consequently, there will be a reduction of the payload capability to the orbit.

Table 11 shows the four elliptical orbits studied under this contract. Two of the elliptical orbits are geosynchronous with different eccentricities and inclinations.

TABLE 11. - ELLIPTICAL ORBITS

Type	Period	Eccentricity	Inclination	Altitude		Payload capability
				Apogee	Perigee	
Triply-synch	3.000 h	0.3467	116 6 deg	7,843 km	521	3134 kg
Molniya	11 967	0.7720	63.4	39,375	1,000	9575
Geosynch	23 934	0.6000	60.0	61,085	10,488	8007
	23 934	0.3000	30 0	48,435	23,137	6544

The triply-synchronous elliptical orbit is retrograde and thus the orbital plane drifts to the east at the rate of +0.9856 deg/day. This equals the average rate of motion of the Earth around the Sun. Therefore, the orbital plane maintains a fixed orientation with respect to the Earth-Sun line. This allows the satellite to have nongimbaled solar arrays thereby reducing the weight and cost of the electrical power subsystem. Also, at the inclination of 116.57 deg, the orientation of the orbit in its plane does not change and thus the position of the perigee relative to the orbit remains fixed. Therefore, the orbit has a fixed time schedule and a fixed ground track. At the 3-hour period, the ground track repeats itself after eight revolutions in one day. Figure 7 shows the ground tracks of the triply-synchronous orbit.

The main drawback of this orbit is the large delta-v required to obtain high inclination. Therefore, satellites that could be placed in this orbit would be restricted to 3134 kg using a Centaur G as the orbital transfer vehicle.

The Molniya elliptical orbit (named after a class of Russian satellites) is characterized by a highly eccentric elliptical 12-hour orbit whose line of apsides does not precess. Satellites in this orbit would have long dwelltimes at apogee and short dwelltimes at perigee. For example, Figure 8 shows the ground track of a typical Molniya orbit. It can be seen that on each 12-hour revolution, 11 hours are spent in the northern hemisphere and only one hour in the southern hemisphere. The ground track is constant because the orbital period is adjusted to be exactly half a sidereal day corrected for the drift of right ascension. At an inclination of 63.4 deg, forces that produce the rotation of the line of apsides are balanced, therefore the location of the apogee remains constant.

The one item that is not fixed in this orbit is the time schedule. The satellite arrives above a given point on Earth about four minutes earlier each day.

The final elliptical orbit considered in this study was the elliptical geosynchronous orbit. The eccentricity used for the geosynchronous orbit designs were 0.6 and 0.3 for inclinations of 60 and 30°, respectively. This resulted in the ground trace becoming egg-shaped and provided an extensive period of time that a satellite would remain above a given coverage area. Figures 9 and 10 show the ground trace resulting from these two orbits. Both orbits allow the line of apsides to drift thereby the apogee will progress with time, however, it is possible to compensate for this effect through use of multiple satellites. The 60° inclination and 0.6 eccentricity orbit passes through the Van Allen radiation belt due to the low perigee altitude and therefore the 30° inclination and 0.3 eccentricity orbit is better suited for VOA applications.

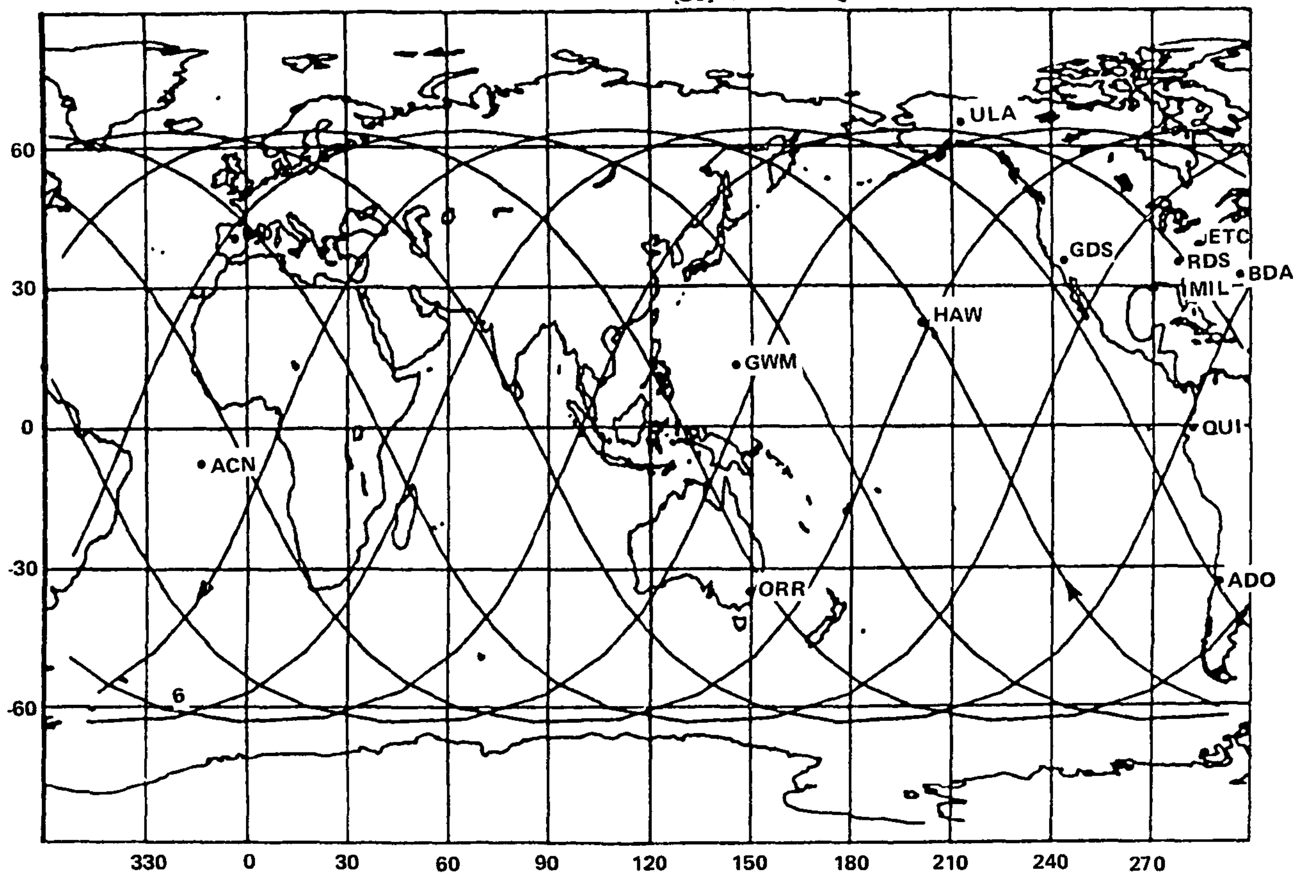

Figure 7. - Triply-synchronous elliptical orbit.

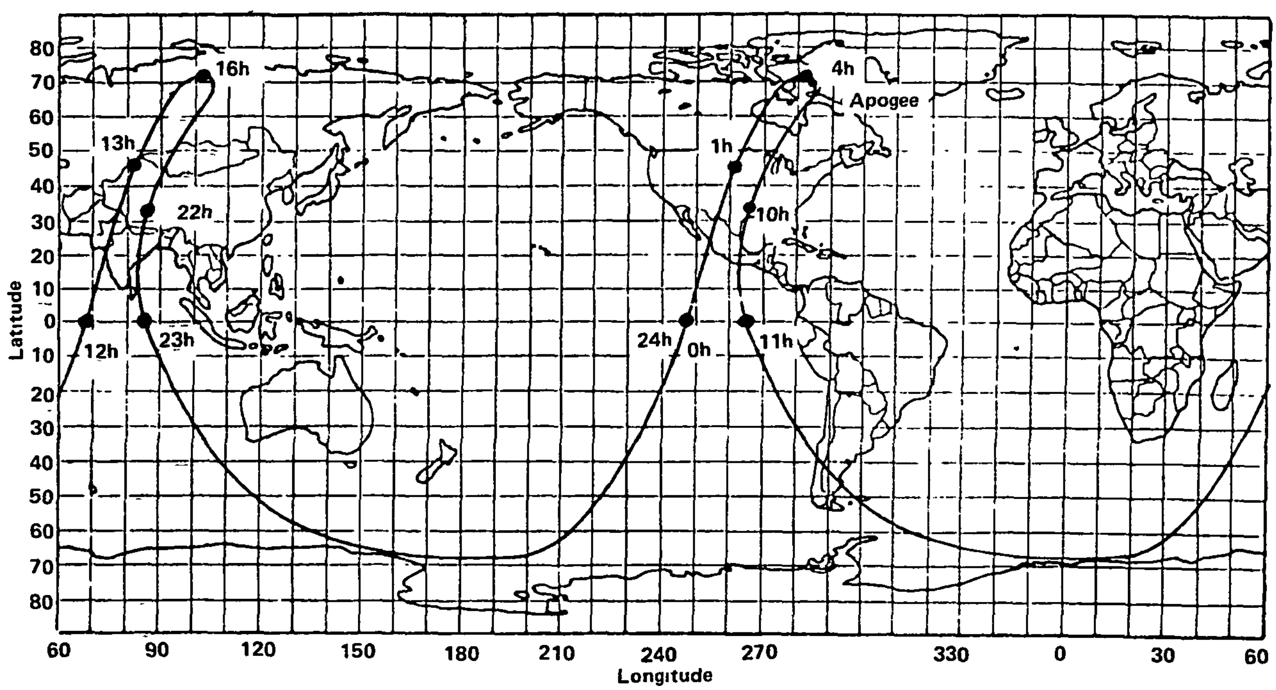

Figure 8. - Typical molniya elliptical orbit.

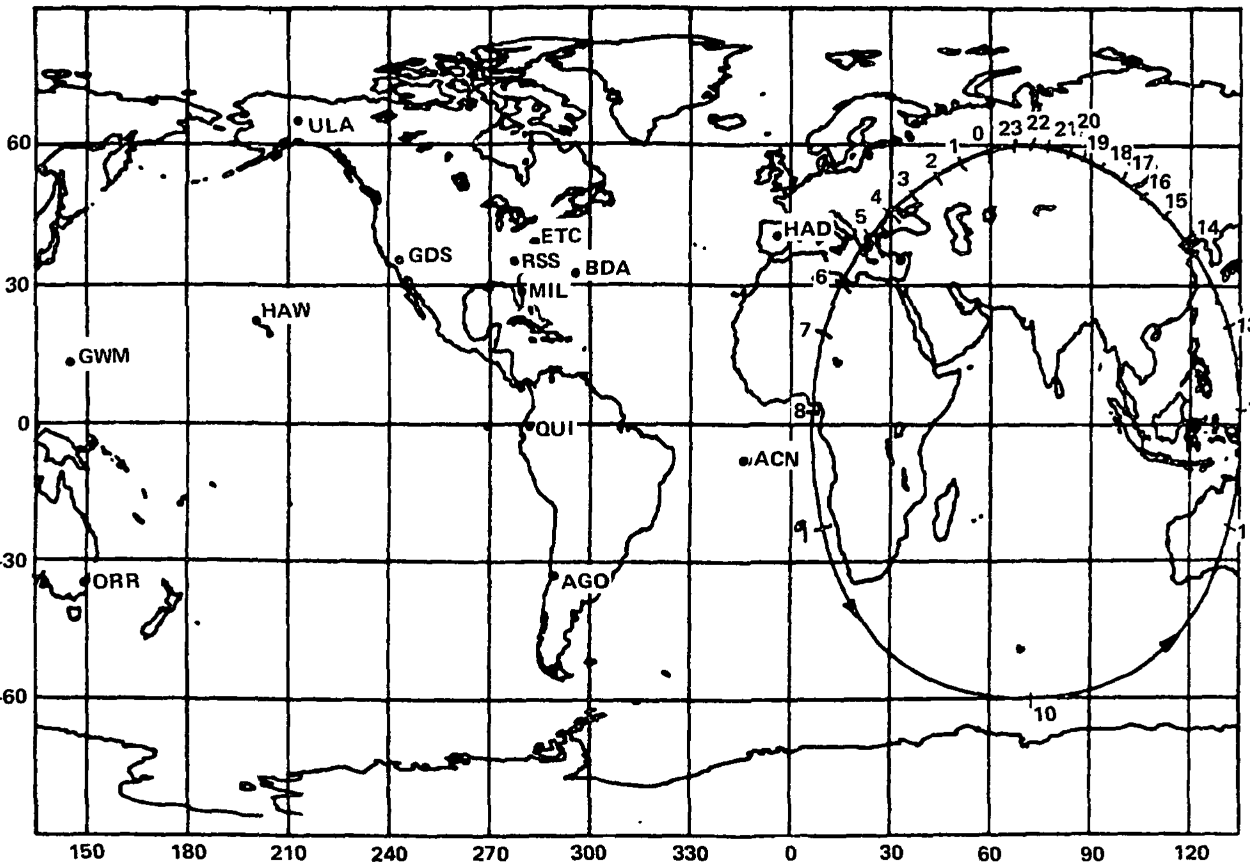

Figure 9. - Elliptical geosynchronous orbit.

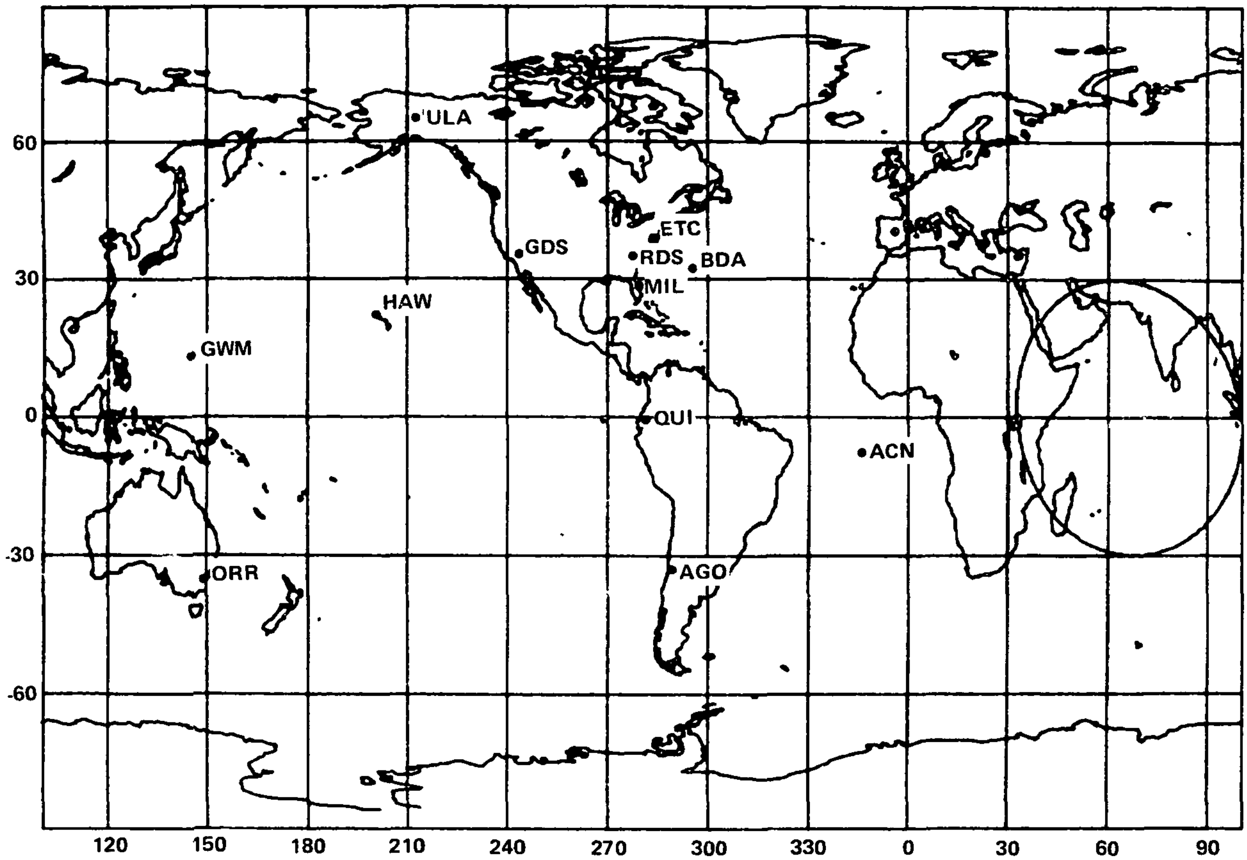

Figure 10. - Modified elliptical geosynchronous orbit.

3.1.3 Circular Orbits

The circular orbits studied under this contract consisted of three types: triply synchronous, geostationary, and subgeosynchronous (Table 12). For the subgeosynchronous orbits both 30 and 45° inclination orbits were examined to assess the impact on inclination vs coverage requirements.

TABLE 12. - CIRCULAR ORBITS

Type	Period	Inclination	Altitude	Payload capability
Triply synch	3.000 h	125.3 deg	4,182 km	2127 kg
Geostationary	23.934	0.0	35,786	4387
Subgeosynch	11.967	30.0 45 0	20,194	7236
	7.978	30.0 45 0	13,892	8910
	5.984	30 0 45.0	10,355	9993

The triply-synchronous circular orbit has characteristics similar to the triply-synchronous elliptical, i.e., a fixed ground track, a fixed time schedule and, a fixed orientation to the Sun. However, dwelltime over a particular spot is drastically reduced and therefore, would require more satellites to meet the same coverage requirements than the triply-synchronous elliptical orbit. Also, the inclination required is increased to 125.3° thereby requiring additional delta-v to achieve this orbit and restricting the payload to orbit from 3134 kg for the elliptical case to 2127 kg. Due to these reduced capabilities, the triply-synchronous circular orbit was not considered for full capability VOA applications.

The geostationary orbit is a standard for existing communication satellites. Satellites in geostationary orbits provide service to a wide coverage area. Dwelltime is constant since the satellite in geostationary orbit appears to be stationary to observers on Earth. However, the elevation angle from high altitude areas to a geostationary satellite will be low. Because of the high altitude, eclipse time is minimal.

The subgeosynchronous circular orbits studied ranged from approximately 12 to 6-hour periods. The inclinations of 30 and 45° were chosen since coverage requirements did not warrant higher inclinations and lower inclinations would produce low elevation angles when covering high latitude zones. All of the circular orbits studied had periods that were submultiples of one sidereal day. This assured the ground tracks of the orbits remain fixed. However, the time schedule was not fixed since the satellite will arrive over a given point 4 minutes earlier each day after completing one ground track cycle. Because of this time shift, multiple satellites in different orbital planes that generate the same ground track and follow one another were necessary to provide continuous service.

As the circular orbits were reduced in period and thus altitude, the following occurred: (1) the dwelltime of an individual satellite over a target shortened, (2) the eclipse time increased, and (3) the payload capability increased. As discussed later, the low altitude circular orbits, i.e. the 6- and 8-hour orbit, proved advantageous for use at the HF frequency due to this increased payload capability. Figures 11, 12, and 13 show the ground tracks for the 12-, 8-, and 6-hour orbits at 30° inclination respectively.

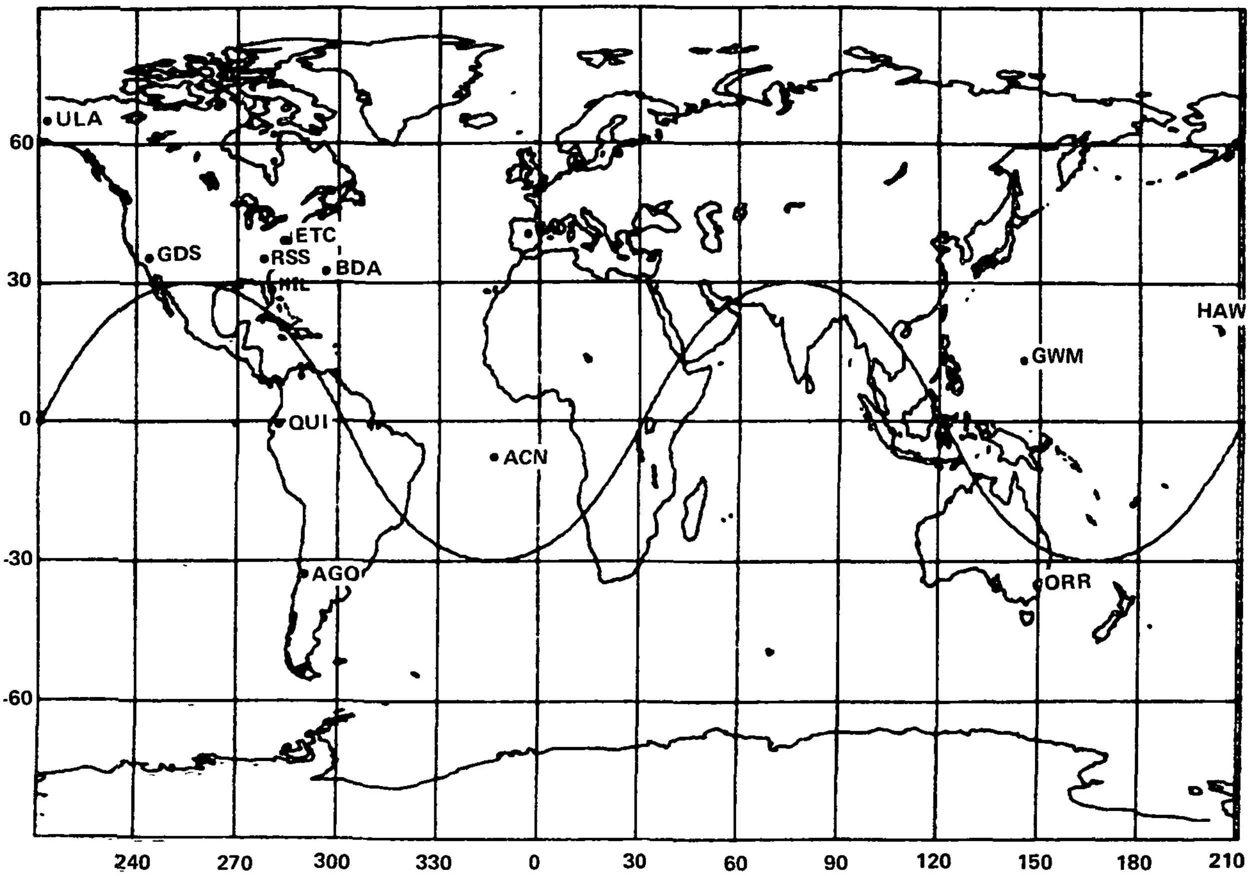

Figure 11 - Circular 12-hour orbit.

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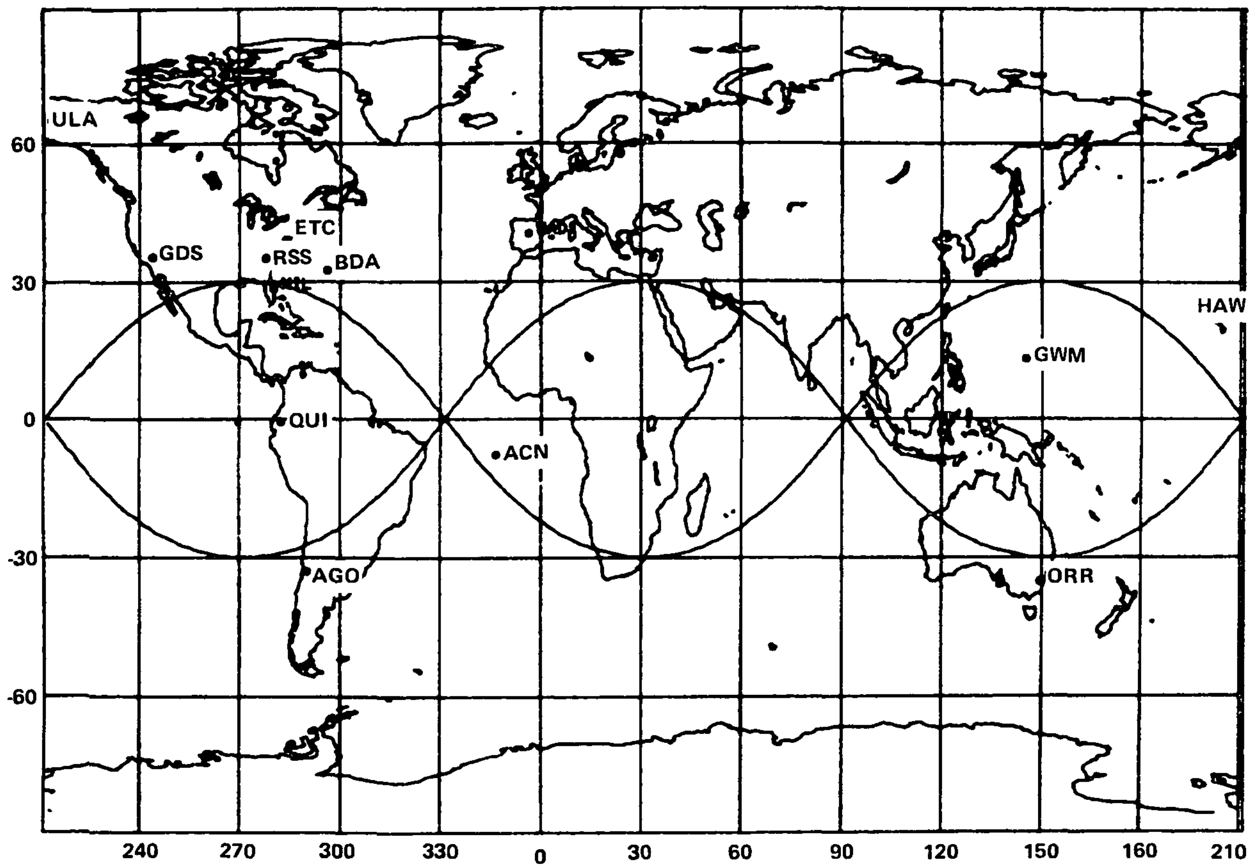

Figure 12. - Circular 8-hour orbit.

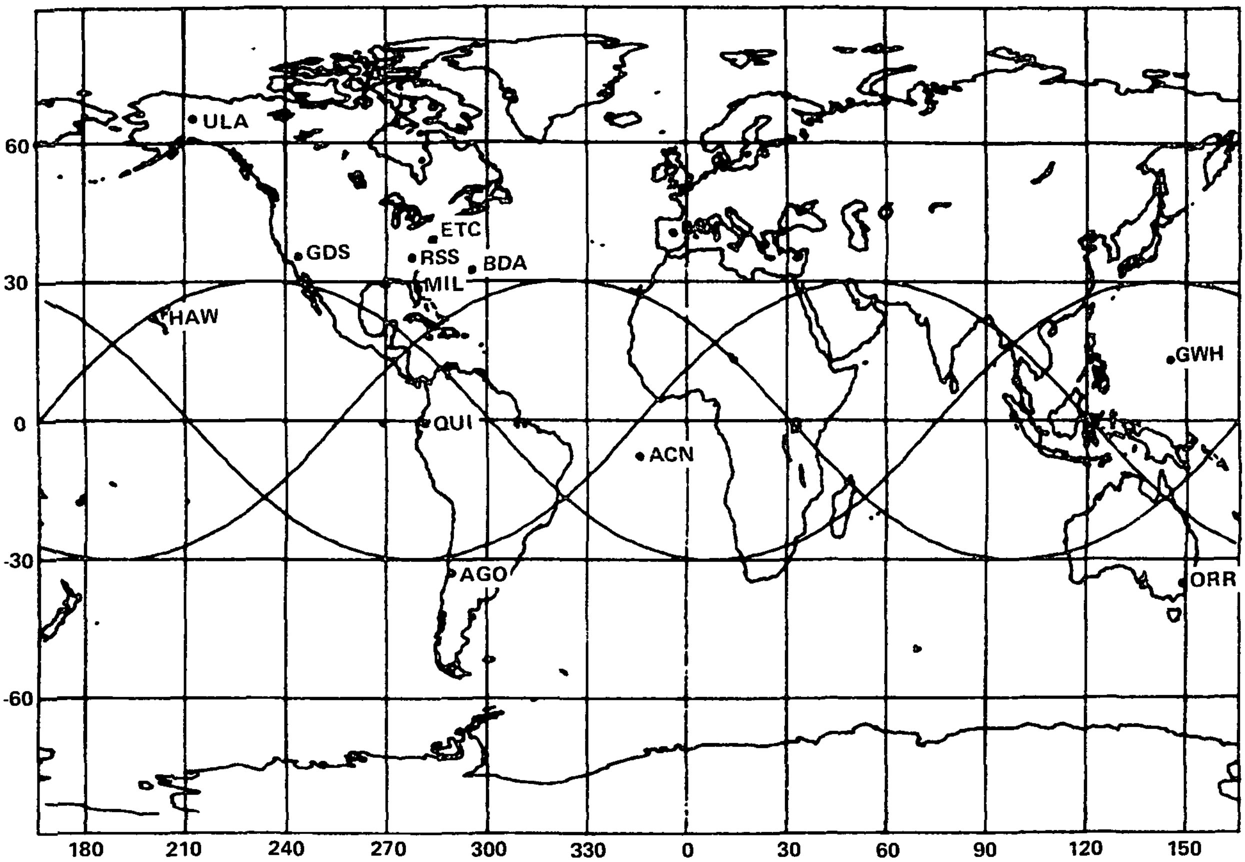

Figure 13. - Circular 6-hour orbit.

3.1.4 Coverage Analysis

After completing the orbital analysis, a coverage analysis was performed using the orbits best suited for VOA applications in order to provide some measure of the utility of proposed VOA satellite systems. The goal of the analysis was to measure the ability of alternative satellite systems to meet the VOA requirements. Figure 14 diagrams the method and logic flow of this analysis.

The VOA requirements for satellite broadcast to the zones of interest were described in matrix form in the statement of work (SOW). A graphical representation of the overall requirements (time vs zone) is shown in Figure 15. Note that this example is for the full HF broadcast requirements.

A time-step simulation of satellite access to the zones of interest was conducted using computer routines developed by the Martin Marietta Denver Aerospace Operations Analysis Department. Since the requirements were stated in 15-min increments throughout the day, the simulations were run at that interval as well. The simulations were conducted for 24 hours, since the requirements repeated daily and since the orbital positions chosen for analysis were reconfigured (that is, at their original positions relative to the Earth's surface) after 24 hours. (Exceptions to this 24-hour reconfiguration are discussed later in this section.)

The simulation routines look for access from the orbital position to geographic points on the Earth's surface within grazing angle limits. (Grazing angle and elevation angle are synonymous in this context, depending only on the point of reference; i.e., from the satellite or from the ground observer, respectively.) HF and VHF analyses assumed a minimum 20° elevation angle limit. For L- and Ku-band an 11.5° elevation angle was used. Figure 16 depicts access from two satellites. Each geographic zone of interest was simulated by multiple latitude/longitude ground points. Figure 17 shows the set of points that were used. Depending on the zone size, anywhere from three to six ground points were used to define each zone.

These access matrices were then compared, one orbital position at a time, to the requirements matrix. If a requirement to cover a zone existed and an orbital position had access to that zone, an assignment of orbital position to zone was made. Two possible transmit schemes were investigated. First, each orbital position was allowed to cover only one zone at a time, even if it had access to multiple zones. In the second scheme, each orbital position was considered to consist of a cluster of spacecraft all with the same ascending node, which has the capability to cover all zones in access at that time. These two schemes provided lower and upper bounds respectively on the performance capabilities of the systems examined.

Figure 18 shows an example of an orbital position's access overlaid on the requirements diagram from Figure 15. The scheme depicted here is that multiple zones can be covered at one time.

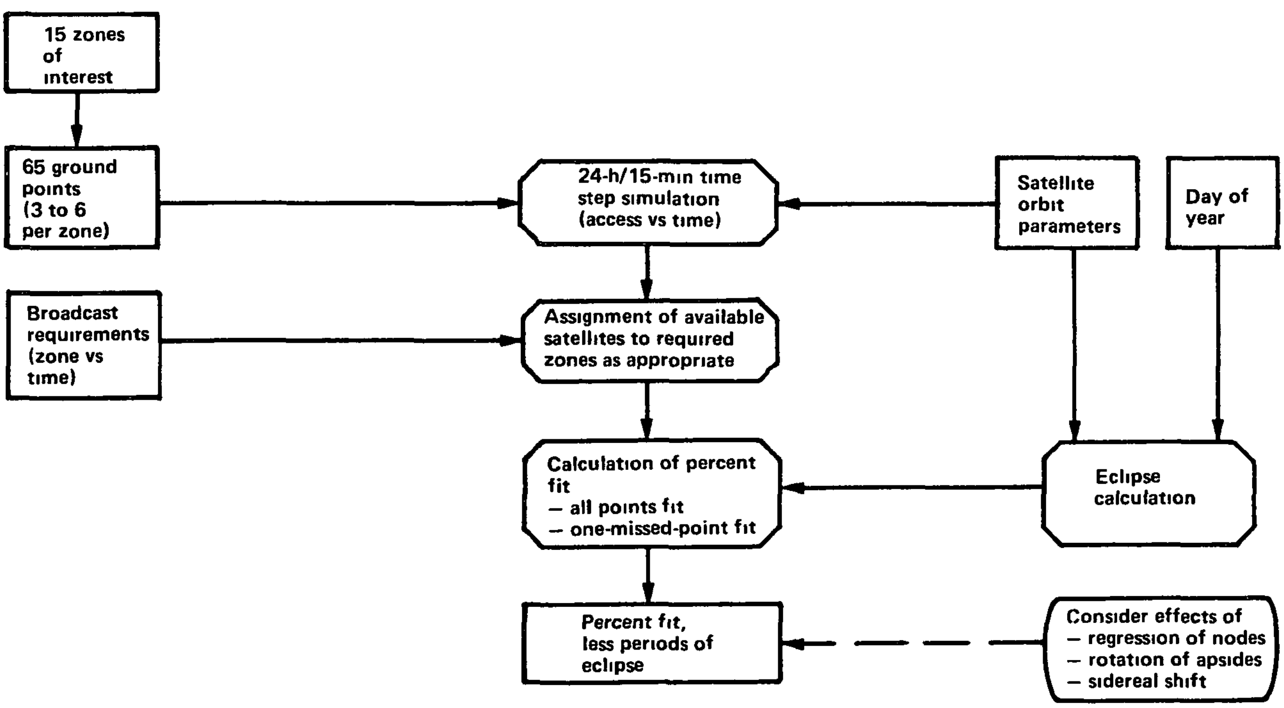

Figure 14. - Summary of utility analysis methodology.

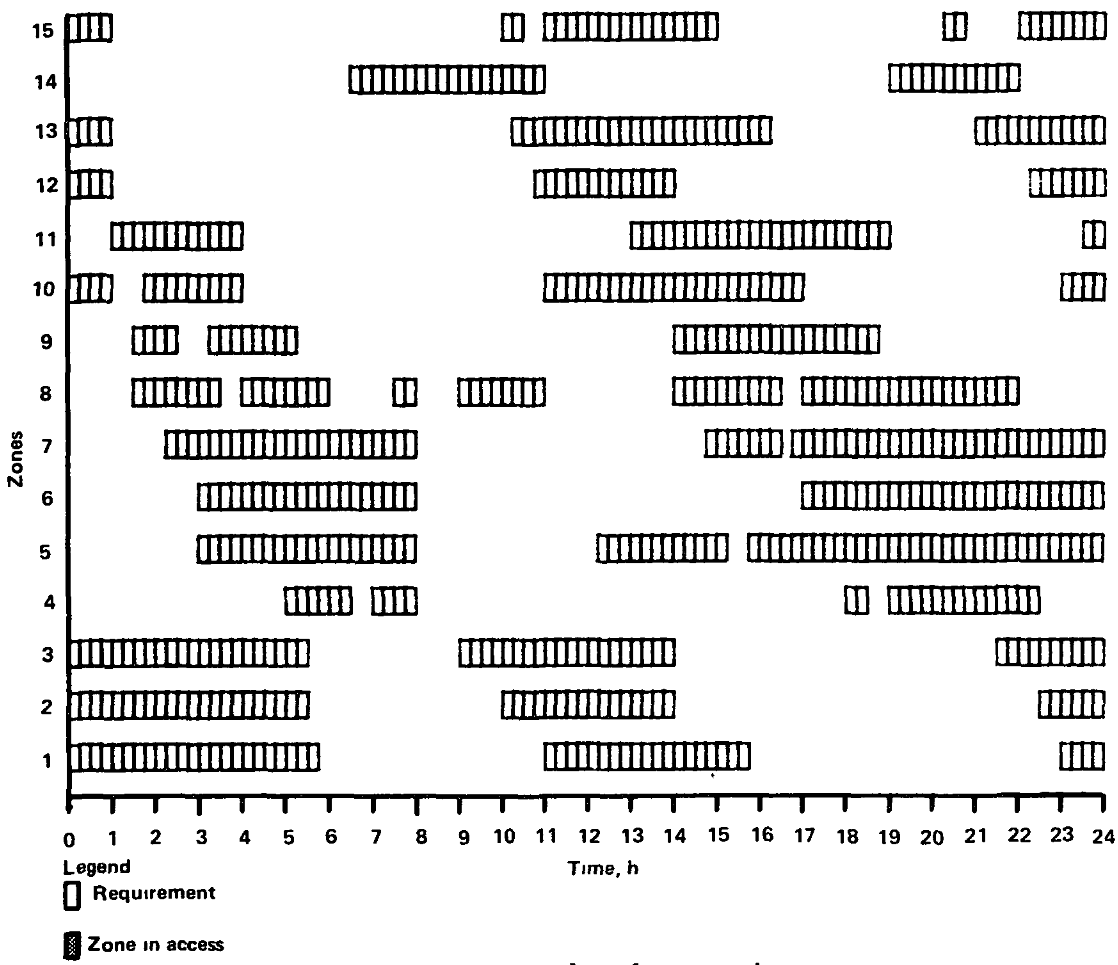

Figure 15. - VOA broadcast requirements.

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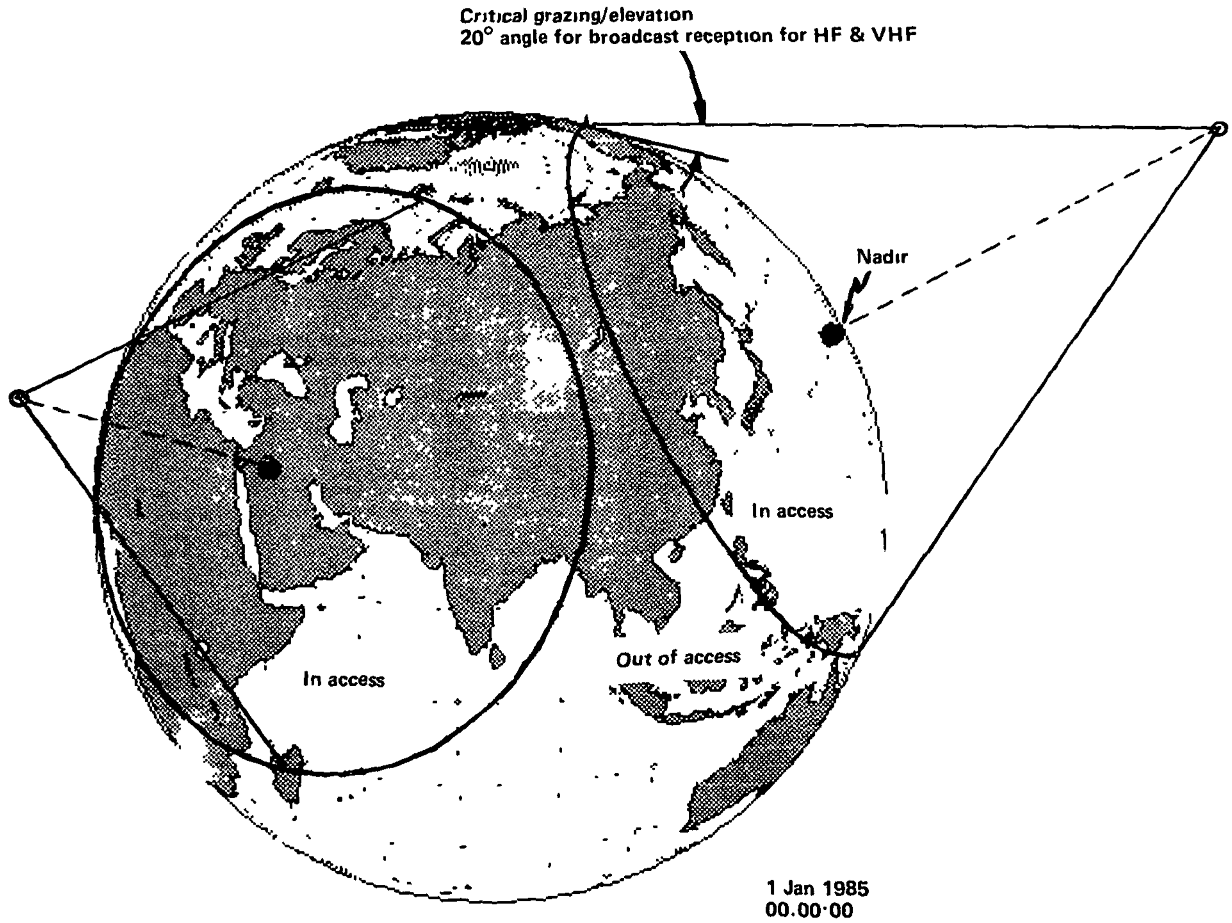


Figure 16. - Earth access from orbital position.

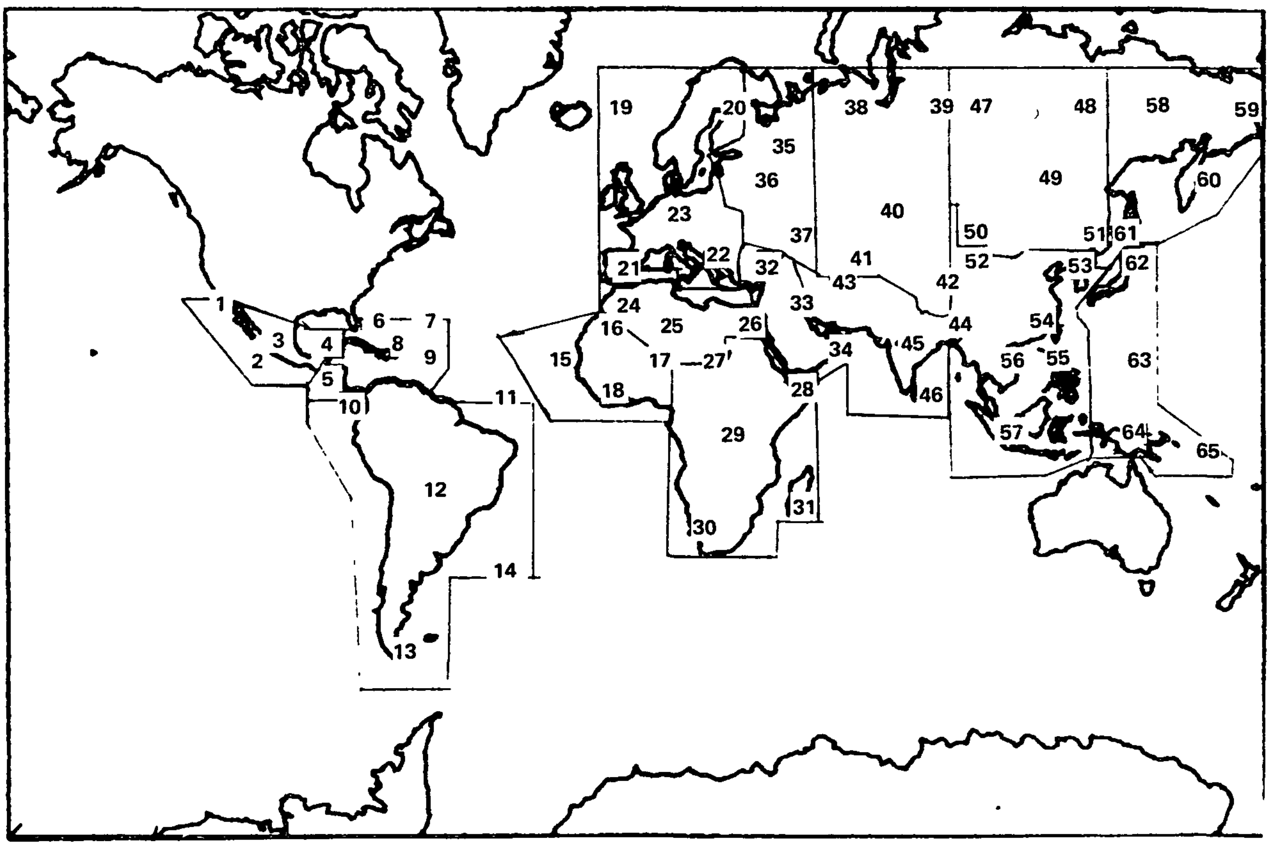


Figure 17. - Ground points used to define zones.

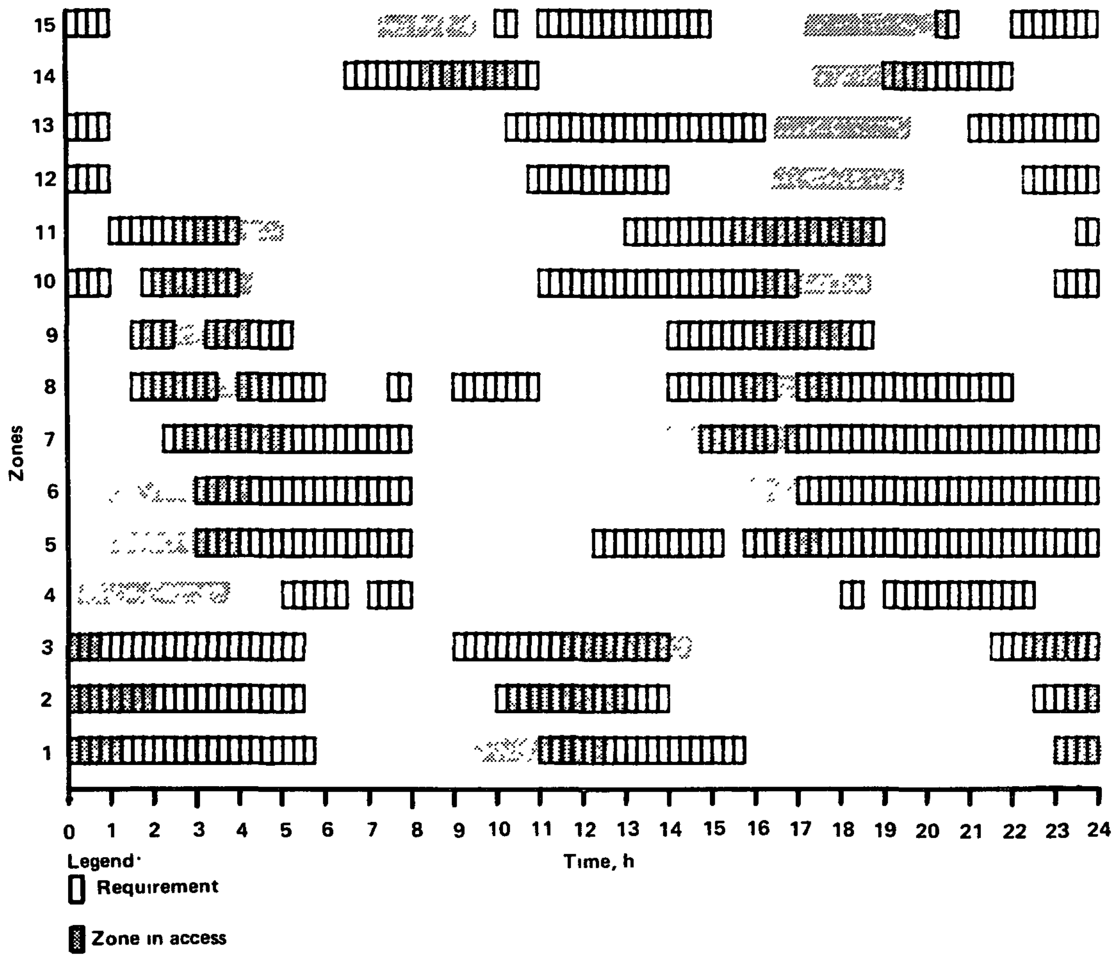


Figure 18. - One orbital position access on VOA requirements.

The procedure for assigning orbital positions to zones was repeated for each orbital position in the constellation. A resulting matrix was then formed to show the results of the simulation. Figure 19 is an example of an output matrix showing information about orbital position assignments over time and when, where, and why requirements were not met during the 24-hour period.

Once a broadcast assignment matrix was completed, system efficiency was calculated. For cases with multiple orbital positions, the ratio of coverage provided to the coverage required was calculated. That is, the denominator was the number of time period-to-zone combinations in the requirements matrix and the numerator was the number of time/zone assignments made. For single orbit position concepts, very few of the requirements could be met. Consequently, coverage efficiency would not be meaningful. However, the fraction of time that a single satellite would be transmitting is useful. This parameter can be obtained from charts like Figure 18 and has been provided in Table 13.

		Zones														
Time step	UTC	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
1	0.00	8	3	6	0	0	0	0	0	0	*	0	*	7	0	5
2	0.15	8	3	6	0	0	0	0	0	0	*	0	*	7	0	5
3	0.30	8	3	6	0	0	0	0	0	0	*	0	*	7	0	2
4	0 45	3	$	6	0	0	0	0	0	0	7	0	$	$	0	2
–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–	–
95	23.30	3	6	*	0	*	4	7	0	0	*	2	*	$	0	5
96	23 45	3	6	*	0	*	1	4	0	0	*	7	*	2	0	5
97	24.00	8	3	6	0	0	0	0	0	0	*	0	*	7	0	5.

8 orbital planes—limit of one zone access per satellite (8 satellites).

* – Access required, cannot see zone
$ – Access, but orbital position allocated to another zone.
0 – No access required

Other numbers are individual orbital position numbers.

Figure 19 - Results of tbe assignment process.

TABLE 13. - PERFORMANCE OF THE SATELLITE SYSTEM

Case	1	2	3	4	
1	0.45	0.59	0.67	0.84	*Coverage efficiency criteria
2	0.40	0.55	0 65	0 81	
3	0.62	0.85	0.83	0 97	1) Fraction of total requirements met
4	0.43	0.58	0 50	0.67	– One zone per satellite
5	0.63	0.71	0 73	0 80	– All of zone covered
6	0.52	0 68	0 91	1 00	2) Fraction of total requirements met
7	0.54	0 67	0 88	0.95	– One zone per satellite
8	0.52	0 68	0 91	1 00	– Most of zone covered
9	0.72	0.84	0 92	1 00	3) Fraction of total requirements met
10	0 33	0 45	0.38	0 49	– Multizones per satellite
11	0.58	0 65	0.72	0 76	– All of zone covered
12	0.69	0 91	0 84	1 00	4) Fraction of total requirements met
13	0 93	1 00	1.00	1.00	– Multizones per satellite
14**	0.50	0 56	0 74	0.81	– Most of zone covered
15**	0.50	0.58	0 76	0 82	
16**	0.72	0.76	0 96	1.00	**Hybrid constellations (geostationary/
17**	0.66	0 72	0 96	1.00	elliptical) Combined performance calculated
	Utilization factor for single satellites				
18				0.54	
19				0.69	
20				0.75	
21				0.67	
22				0.96	
23				0.96	

Note

L-band & Ku-band at three geostationary positions (20W, 15E, 110E) provided 100% coverage using 11.5° elevation

During the process of determining the coverage efficiency, it was found that many of the orbit positions covered all but a single ground point defining a zone, i.e., effectively covering 80% of the required zone. Since this was the case, two different efficiencies were calculated. First, all points of the zone were required to be in view for an access to be generated in the matrix. In the second calculation, one point in the zone could be out of view. Obviously, allowing one point in the zone to be out of access increases the measured performance. The operational significance of this depends on the impact of broadcasting to less than an entire zone. Figure 20 displays one satellite access to one zone and demonstrates the effect of this modification.

The analyses were completed by taking into account orbital effects, i.e, satellite eclipse time and Earth's oblateness.

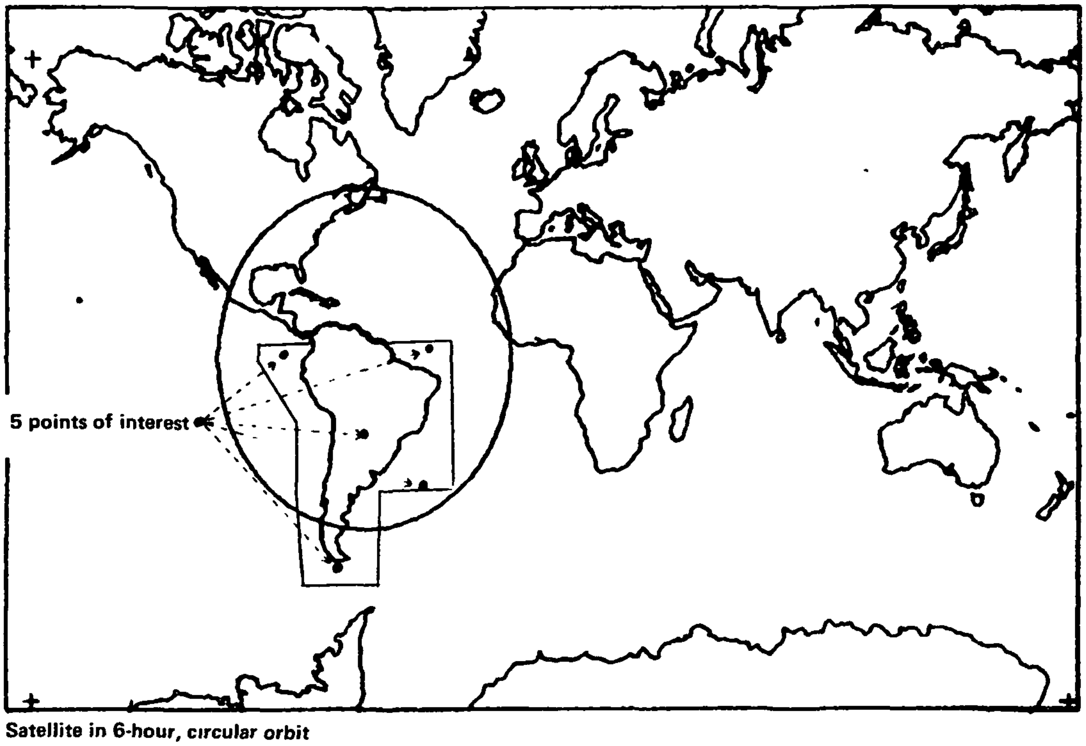


Figure 20. - Effect of one point in zone out of view.

3.1.4.1 Orbital Positions

Determining the optimum orbit to provide space-to-Earth communications is a nontrivial problem. While high orbit altitudes provide wide area access, lower altitudes are generally cheaper because of lower transmitter power and orbital insertion energy requirements. For this analysis, 23 orbital configurations were investigated under various zone coverage requirements. Although these configurations do not begin to exhaust the possible solutions, they do provide a range of capabilities for comparisons. Table 14 lists the orbital position configurations and zone requirements analyzed.

TABLE 14. - ORBIT POSITION CONFIGURATIONS AND REQUIREMENTS

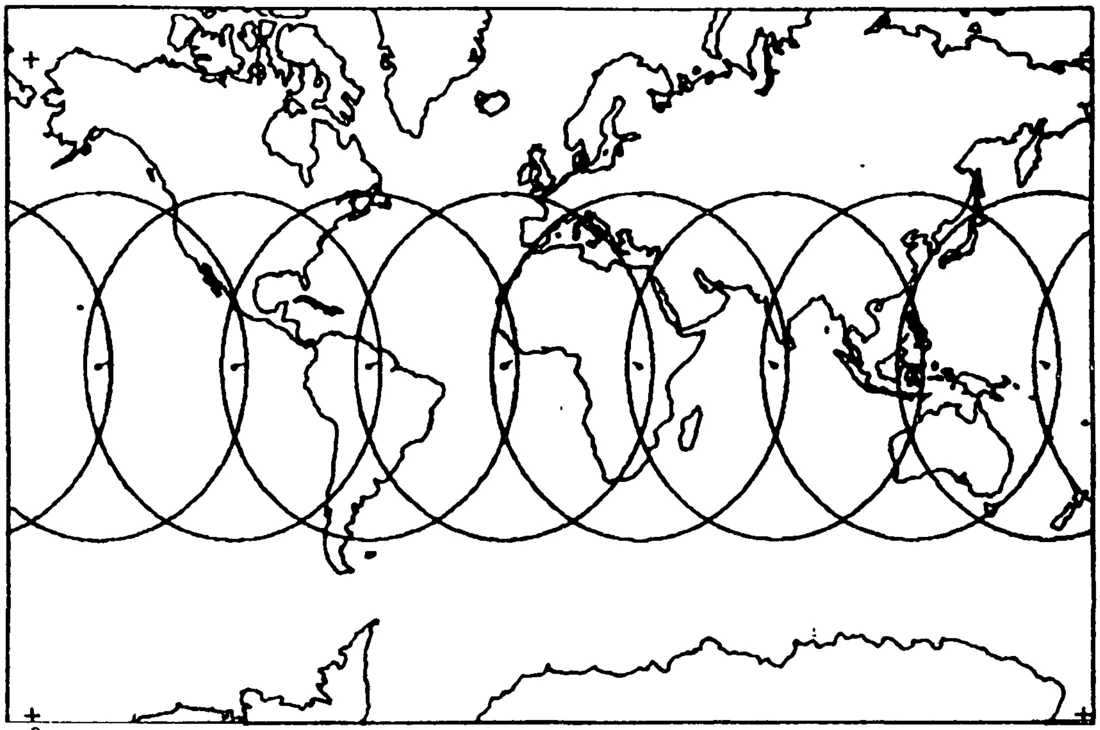
Case	Orbit parameters	Zones of access	Band
1	6-h circular, 30-deg, eight orbital positions	1-15	HF
2	6-h circular, 45-deg, eight orbital positions	1-15	HF
3	6-h circular, 30-deg, eight orbital positions	1-4, 6-8, 11, 13, 15	HF
4	6-h circular, 30-deg, eight orbital positions	3, 5, 9, 10, 12, 14	HF
5	6-h circular, 45-deg, eight orbital positions	3, 5, 9, 10, 12, 14	HF
6	8-h circular, 45-deg, eight orbital positions	1-15	HF
7	12-h circular, 30-deg, eight orbital positions	1-15	HF
8	12-h circular, 45-deg, eight orbital positions	1-15	HF
9	12-h circular, 45-deg, eight orbital positions	3, 5, 9, 10, 12, 14	HF
10	3-h elliptical, 117-deg, 0.347 eccen, eight orbital positions	1-15	HF
11	12-h circular, 30-deg, eight orbital positions	9, 10, 12, 14	VHF
12	12-h circular, 45-deg, three orbital positions	9, 10, 12, 14	VHF
13	24-h elliptical, 30-deg, 0.3 eccen, four orbital positio	9, 10, 12, 14	VHF
	24-h geostationary, three orbital positions	1-4, 6-8, 11, 13, 15	
14	24-h elliptical, 30-deg, 0.3 eccen, three orbital positio	5, 9, 10, 12, 14	L*
	24-h geostationary, three orbital positions	1-4, 6-8, 11, 13, 15	
15	24-h elliptical, 30-deg, 0.3 eccen, three orbital positio	5, 9, 10, 12, 14	L*
	24-h geostationary, three orbital positions	1-4, 6-8, 11, 13, 15	
16	24-h elliptical, 30-deg, 0.3 eccen, six orbital positions	5, 9, 10, 12, 14	L*
	24-h geostationary, three orbital positions	1-4, 6-8, 11, 13, 15	
17	24-h elliptical, 30-deg, 0.3 eccen, four orbitcal positio	5, 9, 10, 12, 14	Ku*
18	3-h elliptical, 117-deg, 0 347 eccen, one orbital positior	1-15	HF
19	6-h circular, 45-deg, one orbital position	1-15	HF
20	8-h circular, 45-deg, one orbital position	1-15	HF
21	12-h circular, 30-deg, one orbital positior	1-15	HF
22	24-h geostationary, one orbital position	1-15	HF
23	24-h elliptical, 30-deg, 0.3 eccen, one orbital positior	1-15	HF

*Subsequent to this analysis, L and Ku-band orbits were changed to geostationary with a 11 5° minimum elevation

Instantaneous access of eight orbital positions in circular orbit is shown in Figure 21 and represents Case 1. These eight orbital positions were configured in an eight-plane set in which all satellites pass the equator (and then reach maximum latitudes) at the same time. The circles shown around each nadir point represent a 20° grazing angle limit assumed in the analyses. At the 6-hour orbit period altitude of 10,350 km, Earth access is provided out to 5450 km from the satellite nadir points. As seen in Figure 21, all of the areas of interest are in access 1 1/2 hours after the satellites cross the equator. If the one-zone-per-orbit position transmit scheme is being considered however, only eight of the required zones may be covered at that time.

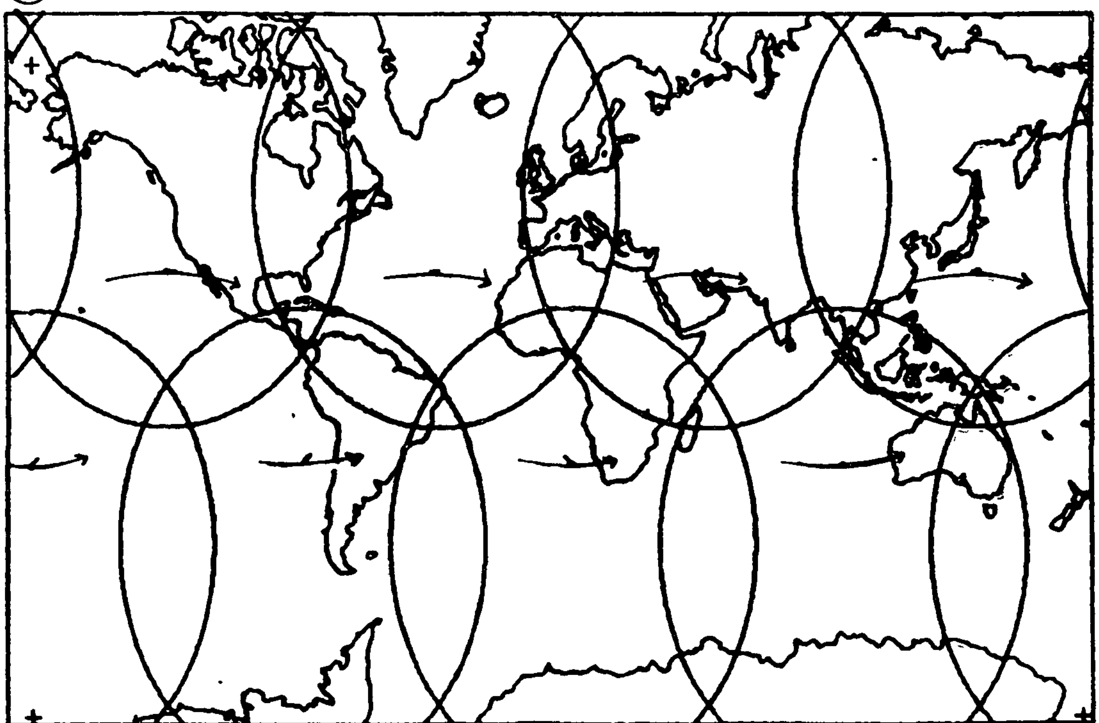
Figure 22 shows the satellite nadir traces for Case 1 and 2 constellations. The period and number of orbit positions chosen provides reconfigurations of the form of the constellation at 3-hour intervals. Every 24 hours each orbit position is at its original position relative to the Earth's surface.

24-hour elliptical orbits (Cases 13-17 and 23) provide a different type of access pattern. As shown in Figure 23, the combination of orbit position and Earth rotation results in a nadir point trace on the planet's surface in the shape of an oval. Note that the hourly interval marks in Figure 23 are closer together in the northern regions; this is due to the placement of apogee at the highest north latitude of the orbit. The resultant slower satellite velocity at the higher altitude maximizes the time viewing the higher latitudes.



20° elevation angle limit

(A) AT EQUATORIAL CROSSING



20° elevation angle limit

(B) 1½ HOURS AFTER EQUATORIAL CROSSING

Figure 21. - Eight satellite constellation access.

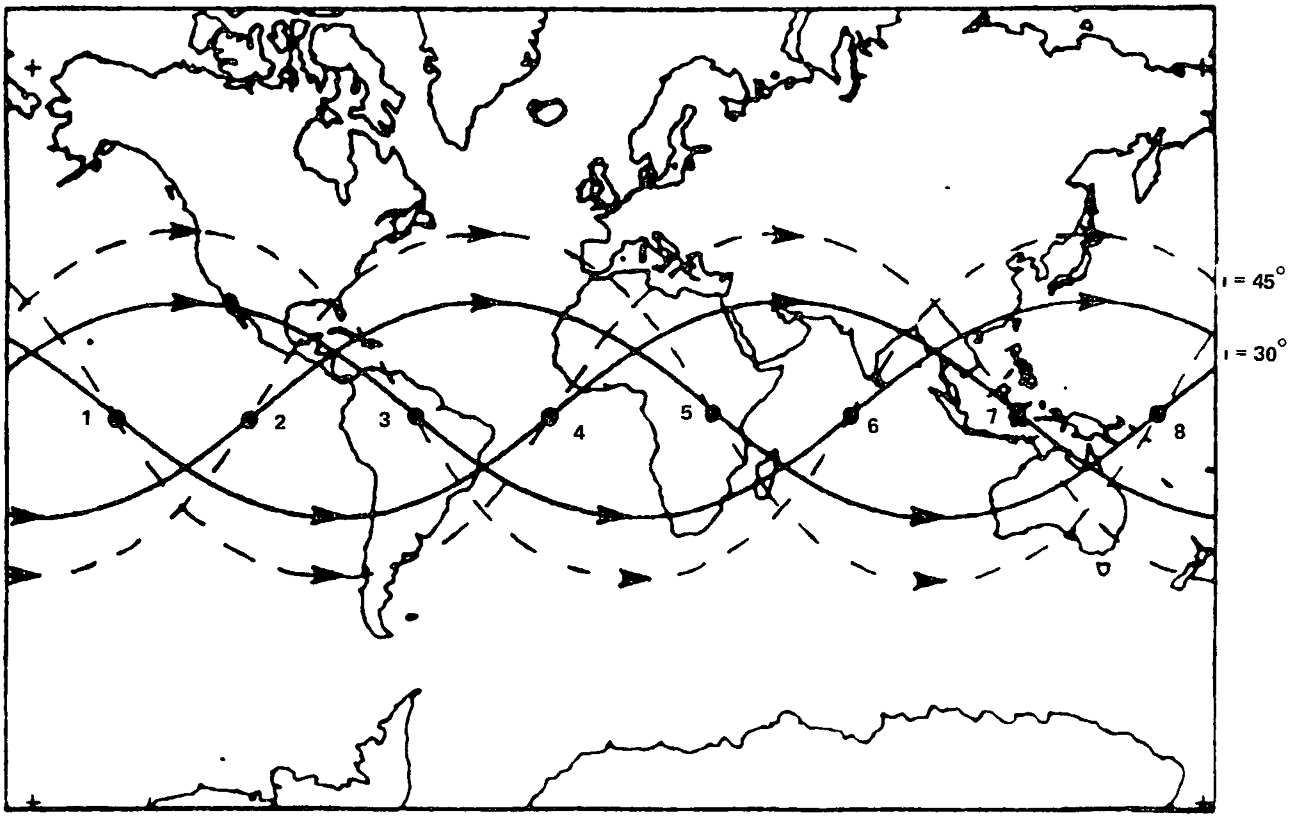


Figure 22. - Satellite nadir trace for 6-hour, circular orbits.

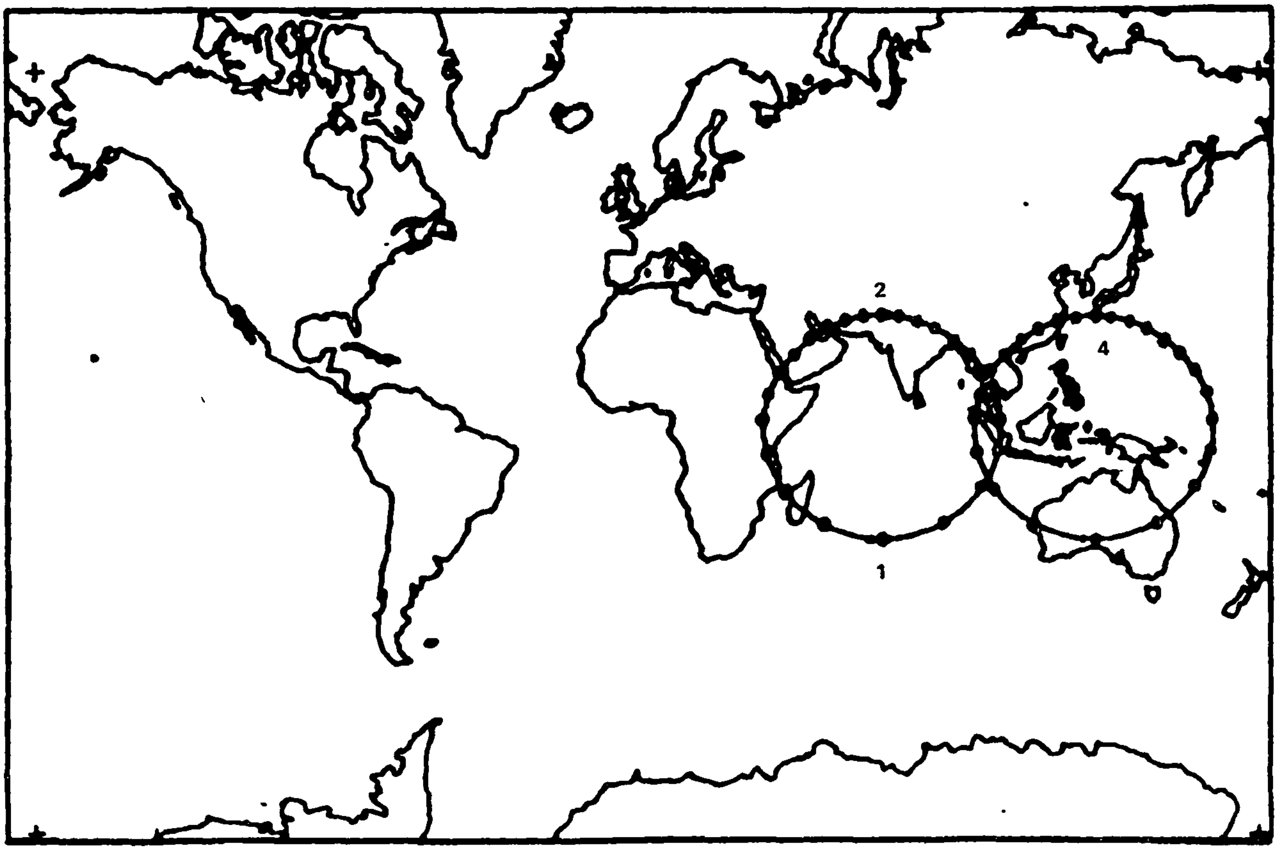


Figure 23. - Satellite nadir traces of elliptical 24-hour orbits.

The size of the access area for elliptical orbits varies throughout the satellites' passage from perigee to apogee, again because of the difference in altitude. Figure 24 shows the size of 20° grazing angle access areas at apogee and perigee for one of the satellites of Case 14.

The third type of satellite orbit considered was the geostationary configuration. These are placed in equatorial orbit at an altitude (35,800 km) that matches satellite motion with Earth rotation, thus providing a stationary nadir point and continuous access to the same area of the Earth. While these orbits are ideal for many communication applications, their disadvantages (that of vehicle payload limitations, high transmit power requirements and size of antenna) are significant.

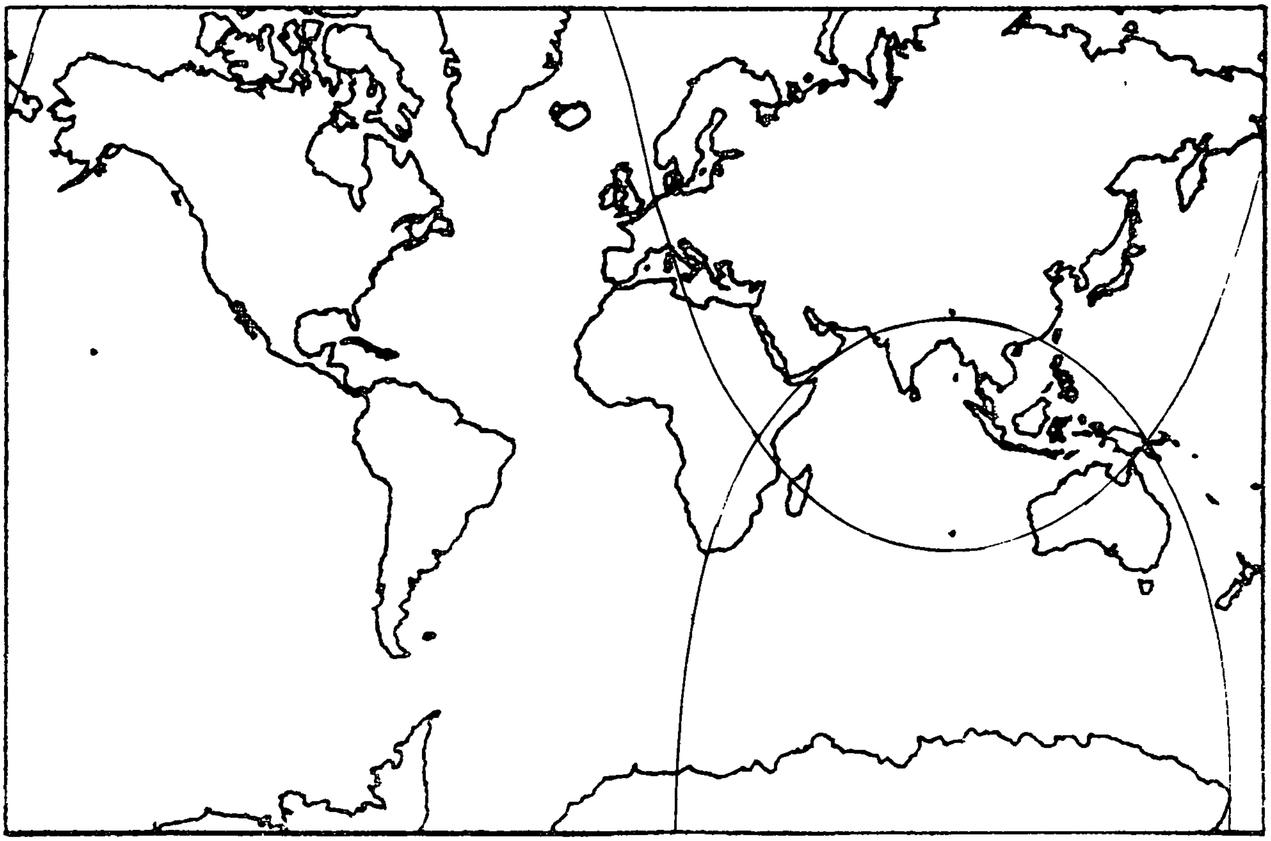


Figure 24. - Access of elliptical orbit at apogee and perigee.

3.1.4.2 Results of Analyses

A 24-hour simulation of the performance of the 23 combinations of satellite constellations and requirements listed in Table 14 provided the results shown in Table 13. The values shown are the coverage efficiencies described in the methodology section.

3.1.4.3 Temporal Effects on Satellite System Performance

Four physical phenomena that affect Earth satellites were analyzed to measure the potential degradation of system efficiency over periods of time:

1) Eclipse - the period when the satellite is in Earth shadow. This is important in considering systems powered by solar arrays.

2) Sidereal shift - the phenomena relating to the difference between a solar and a sidereal day.

3) Regression of nodes - a shift of an orbit's longitude of ascending node.

4) Rotation of line of apsides - a change in the apogee and perigee positions of an elliptical orbit.

Eclipse effects are dependent upon satellite altitude, inclination, and longitude of ascending node. Another factor is the time of year, with fall and winter eclipse periods being similar to spring and summer periods, respectively, because of the similarity in relative Earth/Sun/orbit-plane orientations.

Eclipse effects were examined for the cases listed in Table 14. It was assumed in this analysis that the VOA L-, VHF-, and HF-band satellite systems would not be able to transmit during a period of eclipse. For the relatively low altitude satellite systems considered (such as the 6-hour circular or- bits), the performance efficiency was generally degraded by only a small amount. Table 15 provides an example of this. Higher altitude (geostationary and high elliptical) orbits are affected even less.

TABLE 15. - EFFECTS OF SATELLITE ECLIPSE ON COVERAGE EFFICIENCY

Case	Orbit parameters	W/O eclipse	Eclipse (spring)	Eclipse (summer)
1	6-hr circular, 30°	0.45	0 44	0.43
8	12-hr circular, 45°	0.52	0.52	0.51

Sidereal shift is caused by the fact that the Earth rotates 360° in 23.934 hours, approximately 4 minutes less than a solar (24-hour) day. This minor difference has the effect of causing a satellite in a repeating ground trace orbit to arrive over the same geographical spot 4 minutes earlier each day when measured on a local (solar) clock. Because the VOA broadcast requirements are based on solar time, the sidereal time shift must be taken into account.

The greatest impact of the phenomenon occurs in the single satellite case, such as Cases 19-23 from Table 14. The Day 1 coverage pattern for the 8-hour circular orbit (Case 20) is shown in Figure 25. After 90 days, the access pattern has shifted to an entirely new set of zones requiring coverage (Fig. 26). After half a year, the trend begins to reverse itself and the satellite returns to its original time/position relationship with the Earth a year later.

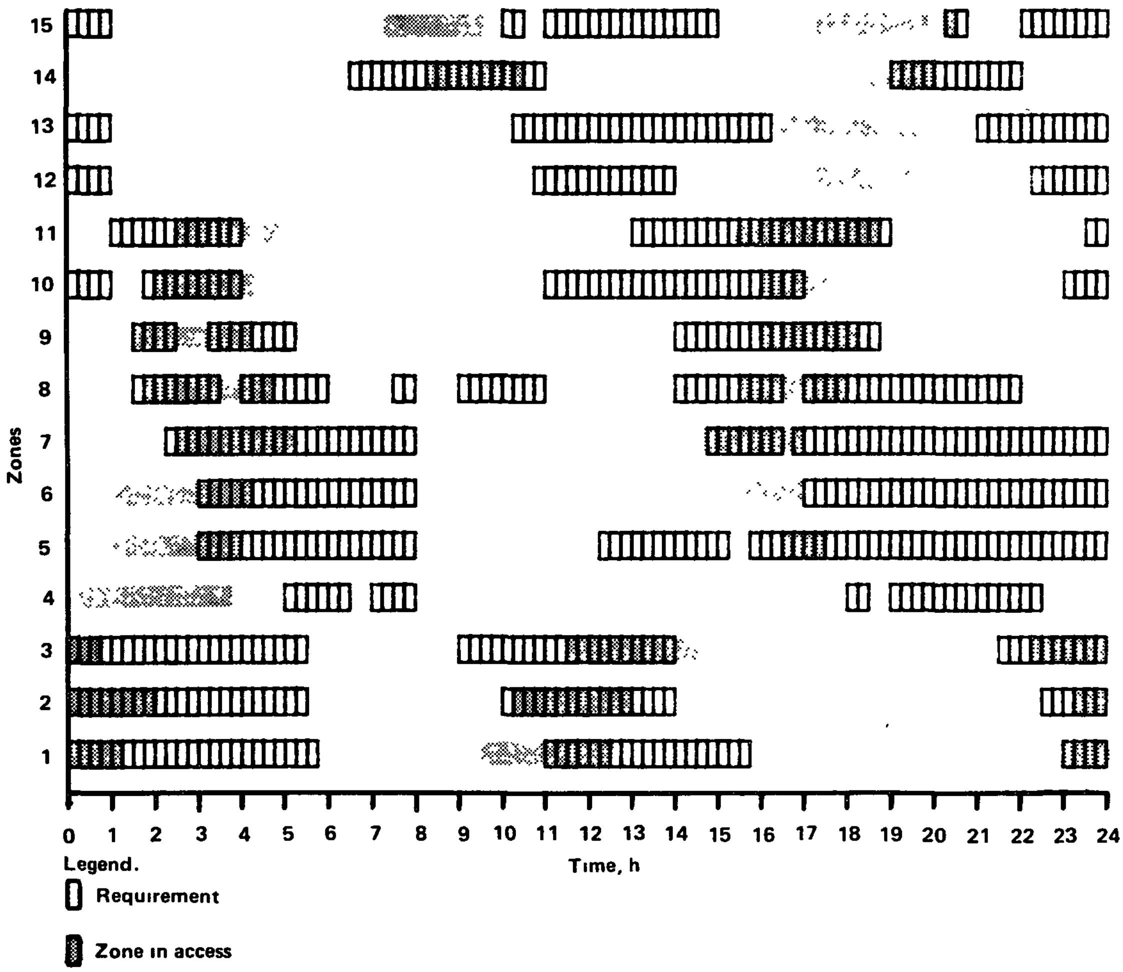


Figure 25. - One satellite access during Day 1.

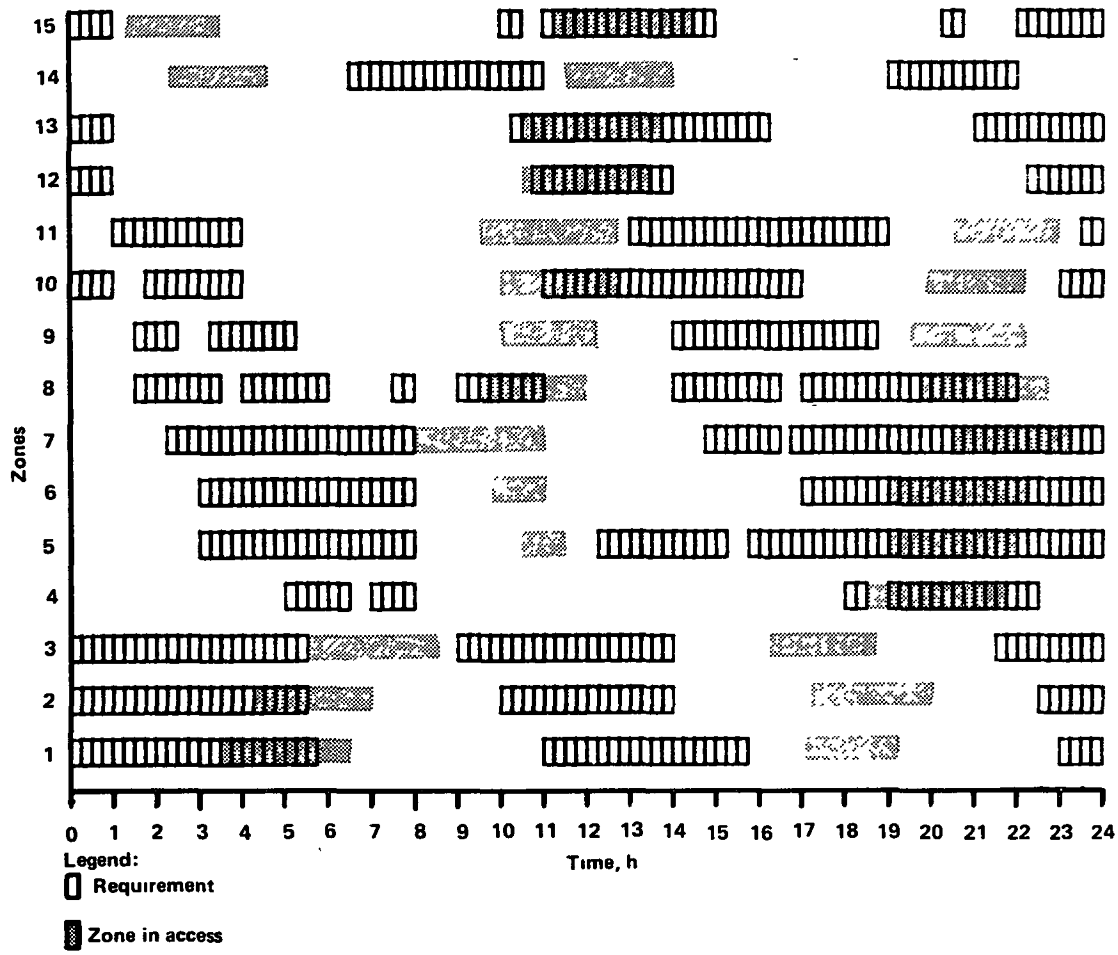


Figure 26. - One satellite access during Day 91.

This effect is significantly reduced in the multisatellite constellations. In an eight satellite constellation, for example, satellite tasking can be changed over periods of time so that requirements met by one satellite at one time are met by another after the sidereal shift begins to degrade performance. In constellations in which satellites repeat ground traces, after a time the relative satellite-Earth access patterns reconfigure. Table 16 gives an example of the effect of sidereal shift on the Case 1 configuration.

TABLE 16. - EFFECT OF SIDEREAL SHIFT ON COVERAGE EFFICIENCY

Case	Orbit parameters	Day	Day 15	Day 30	Day 45*
1	6-hr circular, 30°	0.45	0.44	0.43	0.45

*Reconfiguration of satellite occurs

Regression of nodes is a shift of the orbit plane because of Earth's oblateness. This shift is westerly for posigrade orbits (inclination less than 90°) and easterly for retrograde orbits. The rate of regression of the ascending and descending nodes is dependent on orbital inclination and period. For example, a 6-hour orbit at a 30° inclination will regress west approximately 0.3° per day.

For the applications considered in this study, nodal regression does not pose a serious problem. The reason for this is that a change in orbital period causes a change in ground trace that can be used to compensate for nodal regression. The west regression of 0.3° per day cited above, for instance, may be cancelled out by a slight decrease in orbital period. It should be noted however, that a decrease in orbital period in this case would increase the time shift discussed in the preceding section on sidereal shift.

Rotation of the line of apsides, the line joining the apogee and perigee in an elliptical orbital plane, is another effect caused by the Earth's oblateness. The rate of rotation is also dependent on orbital inclination and period (or altitude.) Rotation is in the direction of satellite motion if inclination is less than 63.4° or greater than 116.6°, and opposite between those values. For the elliptical orbits considered in this study (Cases 13-17 and 23), the rotation of the line of apside is approximately 8° per year in the direction of satellite motion.

Three methods of eliminating or compensating for this effect are suggested. First, no apsidal rotation is incurred at exactly 63.4- or 116.6° inclination. If these inclinations are suitable from other considerations, the problem can be eliminated. Second, choice of initial placement of the line of apsides can be made so that efficiency is maximized at satellite midlife with lower performance accepted at beginning and end of life. Lastly, satellite stationkeeping with onboard thrusters or orbit maneuvering vehicles could be done occasionally to return the orbital plane to its optimum orientation.

The triply-synchronous satellite orbit (Cases 10 and 18) is an interesting solution to the effects discussed above. This orbit's inclination, period, and eccentricity are chosen such that:

1) The orbital plane retains the same fixed orientation with respect to the Earth-Sun line by matching its retrograde orbit drift to the rate of motion of the Earth around the Sun (approximately 1° per day). This Sun-synchronous orbit can be positioned to eliminate or minimize eclipse, thus maximizing operation time of solar powered transmitters;

2) The satellite repeats its ground trace pattern daily;

3) The satellite repeats its time schedule each solar (24 hour) day; and

4) The apogee/perigee points (line of apsides) remain fixed.

3.2 PROPAGATION ANALYSIS

The propagation analysis consisted of determining the signal strength requirements and appropriate losses associated with transmission at each band of operation. Except for L-band, the exact signal strength requirements were specified in the SOW. For L-band, an analysis was performed to determine an appropriate signal strength to meet the SOW requirements. The propagation parameters were determined by reviewing literature on operating direct broadcasting satellites. The propagation parameters included the following:

1) Path loss,

2) Ionosphere attenuation and refraction,

3) Atmospheric and ionospheric scintillation,

4) Atmospheric refraction,

5) Atmospheric attenuation due to rain.

6) Spacecraft elevation,

7) Shadowing and/or attenuation due to buildings, trees and foliage.

3.2.1 Signal Strength Requirements

The four radio bands in the SOW and their specified signal or carrier levels on the ground are shown in Table 17. For the HF-band, the three lower required levels are shown for each modulation method; i.e., double sideband/amplitude modulation (DSB/AM) and single sideband/amplitude modulation (SSB/AM). The DSB/AM-required levels represent the carrier strength on the ground. The SSB/AM-required levels represent the signal strength on the ground. For the HF- and VHF-bands, the two upper-level signal requirements proved too severe for realistic DVBS applications, therefore a program decision was made to study only the lower levels. The L-band requirements were developed from a signal to noise requirement of 40dB for the P1 power level and 49dB for the remaining signal levels. The P1 signal level represents the signal required for portable or mobile re-

TABLE 17. - VOA TRANSMISSION REQUIREMENTS PER CHANNEL, REFERRED TO EDGE OF COVERAGE

Band	Specified EOC signal strength*	Modulation	Power density
HF (26 MHz)	50 μV/m	DSB/AM	-111.8 dBW/m²
	150 μV/m	DSB/AM	-102.2
	300 μV/m	DSB/AM	-96.3
	500 μV/m*	DSB/AM	-91.8
	1000 μV/m*	DSB/AM	-85.8
	35.4 μV/m	SSB/AM	-114.8
	106.1 μV/m	SSB/AM	-105.3
	212.1 μV/m	SSB/AM	-99.2
VHF (47 MHz)	150 μV/m	WBFM	-102.2
	250 μV/m	WBFM	-97.8
	1000 μV/m	WBFM	-85.8
	5000 μV/m	WBFM	-71.8
L (1.5 GHz)	P1 **	WBFM	-91 2
	P2—High	WBFM	-103.6
	P2—Low	WBFM	-116.1
	P3 **	WBFM	-100.1
Ku (12.2 GHz)	5.5 μV/m	WBFM	-131 0

*Assuming that EOC signal level is 3 dB less than center or maximum beam

**Power densities that were not studied due to severity of power requirement ceivers using a whip receive antenna. The P2-low signal level requirement represents an average estimate as to the ground receiver and outside ground antenna used to receive the signal. The P2-high signal level requirement represents a conservative estimate as to the ground receiver and outside ground antenna used to receive the signal. Finally, the P3 signal level represents an average receiver and inside receive antenna. The P1 and P3 signal levels were again too severe and were not studied. P1 signal requirements were derived from the CCIR Study Group 10/11, Report 955 (ref. 14). Detailed analysis of the P2 and P3 requirements is discussed below. The Ku-band specified power on the ground is in terms of power per 4 kHz, and is specified as a maximum in order to minimize interference with ordinary licensed services. Also shown in Table 17 are the corresponding power flux densities (PFD) on the ground. For all cases, the signal levels and PFD requirements are shown for edge of coverage.

3.2.1.1 L-band Analysis

The analysis performed for L-band system to determine the required PFD for each of the three signal levels, used the following requirements:

1) The signal to noise (S/N) ratio must be better than 49 dB,

2) Top modulation, frequency (FM) is 15 kHz,

3) Peak deviation, D, is 75 kHz,

4) Preemphasis is 75 µs.

For the first signal level, the receiver carrier to noise ratio, C/N, required to obtain a weighted S/N ratio of slightly better than 49 dB was estimated to be 19 dB in 250 kHz. This was conservatively estimated for a S/N ratio value of 40 dB corresponding to a C/N of 10 dB determined in CCIR Report 955. However, the preemphasis used in the CCIR Report was 50 µs. For the second and third signal level requirements, the change in preemphasis from 50 µs to 75 µs was also considered, which reduces the required C/N value to 10dB.

Additionally, the receiving antenna for each level was assumed to be 9 dBi, 14 dBi and 6 dBi respectively, and for the inside antenna case, signal level 3, an 11 dB building absorption loss was assumed.

To determine the flux density requirements, the first level assumed nominal RF receiver characteristics and the second and third levels accounted for recent improvements in receiver technology. A summary of results is shown in Table 18.

TABLE 18. - DETERMINATION OF POWER DENSITIES FOR L-BAND SYSTEM

Signal Level	P2-High	P2-Low	P3
Application	Outside antenna	Outside antenna	Inside antenna
Antenna gain	9 dBi	14 dBi	6 dBi
Transmission line loss	3 dB	3 dB	0 dB
Building absorption	0 dB	0 dB	11 dB
Equivalent system gain	+6 dBi	+11 dBi	-5 dBi
Equivalent antenna Effective area	0.013 m^2 or -18.97 dB-m^2	0.04 m^2 or -13.97 dB-m^2	0 001 m^2 or -29.97 dB-m^2
Receiver system temp	2000 K	700 K	700 K
Receiver noise in 250 kHz	-141.6 dBW	-146.1 dBW	-146.1 dBW
Equivalent G/T	-30 dB/K	-20.5 dB/K	-36.5 dB/K
Required C/N	19 dB	16 dB	16 dB
Required flux density*	-103.6 dBW/m^2	-116.1 dBW/m^2	-100.1 dBW/m^2

*Required flux density is for linear polarized signal A 3-dB polarization loss is included in the communications subsystem, assuming circularly polarized transmission and linear reception.

3.2.2 Propagation Parameters

For each radio band, propagation parameters that cause transmission losses were determined using available literature. Table 19 summarizes the resulting losses per band for each propagation parameter considered.

TABLE 19. - PROPAGATION LOSSES

Loss \ Band	HF	VHF	L	Ku
Ionospheric & atmospheric refraction	– – Neglected – –			
Ionospheric attenuation	2 dB	1 dB	0 dB	0 dB
Polarization	3 dB	3 dB	3 dB	0 dB
Atmospheric attenuation due to rain	0 dB	0 dB	0 dB	<0 5 dB
Spacecraft elevation	20°	20°	11.5°	11 5°
Atmospheric & ionospheric scintillation*	Not considered		0 dB	0 dB
Total loss	5 dB	4 dB	3 dB	0 dB

*Scintillation margin not included due to high power requirement.

Ionospheric refraction was neglected in this study since this effect is usually small. Even if it has a significant effect, neglecting refraction is conservative since the area served by the satellite will increase due to refraction rather than decrease.

Atmospheric attenuation due to rain was also neglected since this loss is also small. Figure 27 shows atmospheric attenuation due to rain as a function of frequency and availability at a constant 30° elevation angle (ref. 15). The availability of 90% represents the percent of time the satellite will be able to penetrate the rain if the loss associated with the availability curve is used in the link analysis. For the frequency range of interest in this study, 26 MHz to 12.2 GHz, it can be seen that rain attenuation has virtually no effect until the upper frequency of 12.2 GHz. For the Ku-band, assuming a 90% availability, the rain attenuation results in a loss of less than 0.5 dB (Fig. 27).

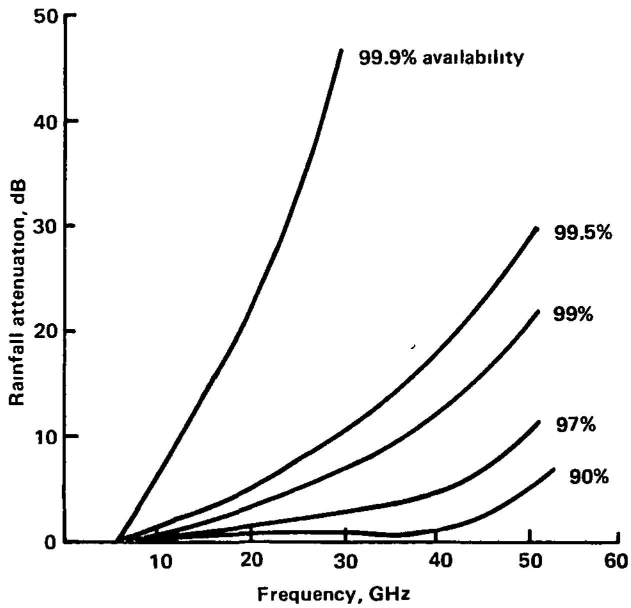


Figure 27. - Availability data for attenuation and frequency, elevation angle = 30°.

Ionospheric attenuation for the HF- and VHF-bands was determined by using the results from the 1984 NTIA technical memorandum on HF/VHF direct broadcasting satellites (ref. 15). For L- and Ku-bands the ionosphere produces no loss of signal.

Atmospheric and ionospheric scintillation has not been included in the loss margin for HF- and VHF-bands due to the severity of the loss that can occur. It is known that scintillation is essentially a nightime phenomenon and maximum losses up to 20 dB occur during maximum sunspot activity. The irregularities in the ionospheric electron density can cause rapid fading of the signal of such magnitude to render the signal useless. Adding scintillation into the link margin would have caused the power requirements to become excessive. L- and Ku-band frequencies are not affected by scintillation.

An additional loss not shown in the above table was path loss or spreading loss. Since spreading loss is dependant on the orbit and the satellite elevation angle, the spreading loss was calculated during the RF performance analysis for each band. The equation describing spreading loss is:

$$L_s = G_p/4\,\pi R^2 \tag{3-1}$$

where L_s = spreading loss in per square meters
G_p = 1.0 for free-space
R = Distance from satellite to receiver station

The distance from the satellite to the ground station, R, is always maximum at the minimum elevation angle of, for example 20 or 11.5° depending on the operating frequency. The minimum elevation angle of 20° for HF and VHF frequencies was chosen to assure signal penetration through the ionosphere 90% of the time.

3.3 PAYLOAD CAPABILITY ANALYSIS

In order to provide some measure as to which orbits were better candidates for VOA missions and to provide a basis for sizing the recommended satellite system concepts, a payload capability analysis was performed. Both the eastern test range (ETR) and the western test range (WTR) as well as a variety of OTVs were considered in the analysis.

3.3.1 STS Capabilities

As defined in the SOW, the STS shuttle was examined to determine the projected lift capability for the 1990's. Although the lift capability of 29483 kg (65000 lb) has yet to be achieved, it is anticipated that by the mid 1990's such a capability will exist.

The 29483 kg capability is not a lift-off limit but rather an abort condition limitation. Figure 28 shows the lift-off capability for the shuttle as a function of orbiter inclination and altitude. For ETR launch, the reduction in lift-off capability due to increased inclination can be offset by reducing the orbiter altitude. As an example, launching into a 45 deg, 370.4 km orbit the STS has a lift capability of approximately 27000 kg. However, by reducing the orbit altitude to 255 km allows the full STS capability of 29483 kg to be used. For the VOA study, the STS was placed to assure the full 29483 kg capability would be achieved. For WTR launches, a reduction in lift capability occurs after 60° of inclination. The orbit altitude of 185.2 km is generally considered the minimum altitude the STS must achieve.

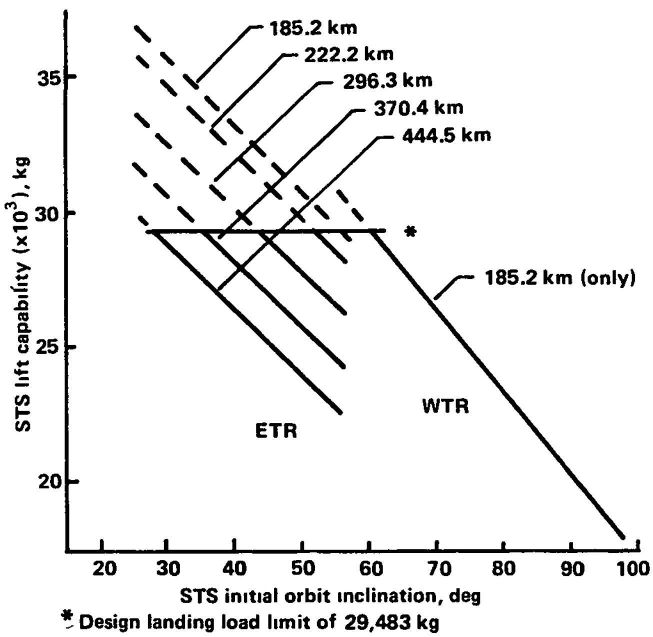


Figure 28. - Projected STS payload lift capability.

An ETR launch was considered for any final payload orbit less than 57° inclination. Additionally, the STS inclination matched the final orbital inclination when possible, e.g., for the 45°, 6-hour circular orbit, the STS was placed in a 45° inclination, 255-km orbit. This assured both maximum lift capability for the STS and a minimum plane change delta-v requirement for the OTV, thereby maximizing the payload capability to final orbit.

WTR launches were considered for any final payload orbit above 57° inclination (e.g, the triply-synchronous elliptical orbit). Unlike ETR launches, matching the final orbital inclination with the STS inclination reduces the lift capability of the STS. Even though this would minimize the plane change delta-v requirements on the OTV, it was not readily obvious that this was the best solution. Therefore, for each WTR launch, a number of STS inclinations were looked at to optimize the payload capability.

3.3.2 OTV Capabilities

In general, the present day OTVs, (PAM-D and IUS) do not have the payload capability from low-Earth orbit (LEO) that would be necessary for transferring proposed DVBS to final orbit. Therefore, this study looked at near-term vehicles that would provide such a capability, [e.g, the Transfer Orbit Stage/ Apogee Maneuvering Stage (TOS/AMS) and the Centaur G]. In addition, a Centaur derivative, Centaur B, was also examined that could be used for multiple STS launches. The Centaur B uses the entire STS lift capability by carrying additional fuel to provide a maximum capability to orbit. The satellite and Centaur B would be mated in orbit, thereby requiring a separate STS launch for the satellite. Both the TOS/AMS and Centaur G are expected to be operational transfer stages by 1986. The Centaur G is a single-stage system that uses a cyro-high energy thrust system while the TOS/AMS is a two-stage system with the first stage using a solid rocket and the second stage a storable bipropellant thrust system. Table 20 summarizes the parameters of each of the above OTVs.

TABLE 20. - ORBIT TRANSFER VEHICLES

OTV	TOS/AMS	Centaur G	Centaur B
Stage length, m	4 6	6.1	9 1
Stage loaded wt, kg	15,665	17,015	25,064
Max main propellant wt, kg	9,705/4,007	13,808	22,426
Effective I_{sp}, s	293/315	440 4	446.4
Average thrust, N	11,787/195,712	133,440	146,784
Shuttle ASE wt, kg	1,893	4,139.0	4,415 0
Typ P/L to Synch-Equatorial, kg	2,900	4,390	9,570

3.3.3 Determination of Payload Capabilility

To determine the final payload capability, three Martin Marietta computer programs were used. The first program determined the delta-v requirements to place a payload into a circular orbit from an STS park orbit, using a two burn transfer. Burn 1 was a perigee burn to transfer to the final altitude and includes a small plane change burn to minimize the total delta-v required. Burn 2 was the apogee burn used to circularize the orbit and complete any plane change. The amount of delta-v required for both burns was determined using a modified classical minimum energy technique, i.e., Hohman transfer. The modification was in a Newton-Raphson iteration technique that was used to compute the optimum amount of plane change that should occur during the first burn.

The second program was used to determine the delta-v requirements to place a payload into an elliptical orbit from an STS park orbit, again using a two burn transfer. Burn 1 was a perigee burn used to put the spacecraft into the transfer orbit. Burn 2 was an apogee burn to produce the final orbit and any plane change and adjustment for the argument of perigee. The minimum delta-v required from both burns was determined by iterating on the latitude at which the second burn occurred, then back solving for the minimum energy cotangential first burn.

The final program took the delta-v requirements from the previous programs and computed the payload-to-final-orbit capability using a specified OTV. This program automatically offloads fuel on the OTV to achieve the final orbit and meet shuttle lift capability.

Tables 21 and 22 show example outputs of the three programs. Table 21 is an example of the payload capability to the 6-hour, 30° inclination circular orbit. Table 22 is an example of the payload capability to the triply-synchronous elliptical orbit.

TABLE 21. - ORBIT TRANSFER ANALYSIS FOR 6-HOUR, 30° INCLINATION ORBIT

```
                PROGRAM--/ORBITC/
RUN DATE WAS: 02-05-1985                    AT TIME: 16:01:36
FOLLOWING IS A SUMMARY OF THE INPUT PARAMETERS:
*** ORBIT TYPE - CIRCULAR ***
INITIAL STS ORBIT INCLINATION = 30 degrees
STS PARK ORBIT = 407.44 km
FINAL P/L ORBIT INCLINATION = 30 degrees
FINAL P/L ORBIT PERIOD = 6 hours


RESULTING CALCULATIONS:
FINAL P/L ORBITAL ALTITUDE = 5607.594 n.miles
1 ST BURN DELTA V = 1480.685 m/sec ( 4857.891  ft/sec)
2 ND BURN DELTA V = 1174.493 m/sec ( 3853.325  ft/sec)
TOTAL REQUIRED DELTA V = 2655.178 m/sec ( 8711.216  ft/sec)
PLANE CHANGE (BURN #1) IS:-9.999999E-06
************************************************

                        PROGRAM--/PAYLOAD/
RUN DATE WAS: 02-05-1985                    AT TIME: 16:02:01

*** DELTA V REQUIREMENTS ***
Req. Delta-v Burn 1 =  4857.9 ft/sec   Req. Delta-v Burn 2 =  3853.3 ft/sec

*** Launch Site  ETR ***

***   STS/ACC/OTV DATA   ***           OTV NAME:CENTAURG
ASE (ACC) Weight:    0.0 Kg              ASE (SB) Weight: 4139.6 Kg
Service Support Weight:    0.0 Kg        STS Lift Capability: 29484.0 Kg

***   STAGE 2 DATA   ***
Max Propellant Weight: 13810.8 Kg       Structural (Dry) Weight: 3097.2 Kg
Misc Fluid (Res,He) Weight: 109.8 Kg    Propellant Margin: 113.4 Kg
Trapped (Unusable) Prop: 198.7 Kg       Misc. Fluid Use Before Burn 1:  9.1 Kg
Misc. Fluid Use Before Burn 2: 48.1 Kg  Main Propellant Losses Burn 1: 132.5 Kg
Main Propellant Losses Burn 2: 180.1 Kg Effective ISP (Sec): 440.4
Average Thrust:  133440 N

***   STS CAPABILITIES   ***
STS Cargo Capability (STS Lift Wt - All ASE & Adaptor): 25344.45 Kg
Vehicle Total Weight (Stages + P/L): 25344.4 Kg
Stage 1 Loaded Weight:     0.0 Kg    Stage 1 Main Propellant Weight:     0.0 Kg
Stage 2 Loaded Weight: 15351.6 Kg    Stage 2 Main Propellant Weight: 12144.6 Kg
Payload Weight Delivered to Final Orbit:  9992.9 Kg
                     Stage 1   Stage 2a  Stage 2b  Stage 2 Tot
Delta-V (ft/sec)     0         4857.891  3853.325  8711.217
Burntime (sec)       0         236.7428  136.0825  372.8253
************************************************
```

TABLE 22. - ORBIT TRANSFER ANALYSIS FOR TRIPLY-SYNCHRONOUS ELLIPTICAL ORBIT

```
                     PROGRAM--/ORBITE/
RUN DATE WAS: 02-05-1985                    AT TIME: 15:56:31
FOLLOWING IS A SUMMARY OF THE INPUT PARAMETERS:
*** ORBIT TYPE - ELIPTICAL ***
INITIAL STS ORBIT INCLINATION = 82 degrees
STS PARK ORBIT = 185.2 km
FINAL P/L ORBIT INCLINATION = 116.6 degrees
FINAL P/L ORBIT PERIOD = 3 hours
FINAL P/L ORBIT PERIGEE ALTITUDE = 521 km
FINAL P/L ORBIT ARGUMENT OF PERIGEE = 240 degrees
RESULTING CALCULATIONS:
FINAL P/L ORBIT APOGEE ALTITUDE = 7843.271 km

TRANSFER ORBIT PARAMETERS:
ARGUMENT AT PERIGEE = 221.1899 degrees
INCLINATION = 82 degrees
TRANSITION ALTITUDE AT PERIGEE = 185.2 km
TRANSITION ALTITUDE AT APOGEE = 6671.446 km

TRANSFER ORBIT INTERSECTION CONDITIONS:
TRANSITION ALTITUDE = 6036.122 km
LATITUDE = 14.73438 degree
VELOCITY VECTORS in ft/sec
(Before Burn)  VT2= 15928.12  PHI2= 11.76798  BETA2= 8.273887  NU2= 153.6926
(After Burn)   VT0= 16880.05  PHI0= 17.67792  BETA0=-27.57984  NU0= 136.5257

DELTA V REQUIREMENTS:
PERIGEE BURN DELTA V = 1196.739 m/sec ( 3926.308  ft/sec)
INTERSECTION BURN DELTA V = 3029.734 m/sec ( 9940.071 ft/sec)
TOTAL REQUIRED DELTA V = 4226.472 m/sec ( 13866.38  ft/sec)
*******************************************************

                        PROGRAM--/PAYLOAD/
RUN DATE WAS: 02-05-1985                    AT TIME: 15:57:23

*** DELTA V REQUIREMENTS ***
Req. Delta-v Burn 1 =  3926.3 ft/sec   Req. Delta-v Burn 2 =  9940.1 ft/sec

*** Launch Site  WTR ***

***   STS/ACC/OTV DATA   ***            OTV NAME:CENTAURG
ASE (ACC) Weight:    0.0 Kg               ASE (SB) Weight: 4139.6 Kg
Service Support Weight:    0.0 Kg         STS Lift Capability: 22130.7 Kg

***   STAGE 2 DATA   ***
Max Propellant Weight: 13810.8 Kg       Structural (Dry) Weight: 3097.2 Kg
Misc Fluid (Res,He) Weight: 109.8 Kg    Propellant Margin: 113.4 Kg
Trapped (Unusable) Prop: 198.7 Kg       Misc. Fluid Use Before Burn 1:  9.1 Kg
Misc. Fluid Use Before Burn 2: 48.1 Kg  Main Propellant Losses Burn 1: 132.5 Kg
Main Propellant Losses Burn 2: 180.1 Kg Effective ISP (Sec): 440.4
Average Thrust:  133440 N

***   STS CAPABILITIES   ***
STS Cargo Capability (STS Lift Wt - All ASE & Adaptor): 17991.12 Kg
Vehicle Total Weight (Stages + P/L): 17991.1 Kg
Stage 1 Loaded Weight:     0.0 Kg    Stage 1 Main Propellant Weight:     0.0 Kg
Stage 2 Loaded Weight: 14857.7 Kg    Stage 2 Main Propellant Weight: 11650.7 Kg
Payload Weight Delivered to Final Orbit:  3133.5 Kg
                   Stage 1   Stage 2a  Stage 2b  Stage 2 Tot
Delta-V (ft/sec)   0         3926.308  9940.071  13866.38
Burntime (sec)     0         139.8077  217.0339  356.8416
*************************************************
```

Figure 3.3-3. - Orbit transfer analysis for triply-synchronous elliptical orbit.

For each orbit studied in section 3.1, a payload capability was performed. A summary of results is shown in Table 23. Table 23 shows the launch site, shuttle orbit and OTV assumed for each orbital case along with the delta-v requirements for each orbit. The table shows that the payload capability increases as the orbital altitude decreases. Also, the payload capability increased as the inclination decreased for the same orbital period. This is attributed to the fact that the STS park orbital altitude is higher for the lower inclination. As an example, 7130 kg can be delivered to the 12-hour, 45° inclination orbit using the Centaur G. For the 12-hour, 30° inclination orbit, the payload capability is increased to 7236 kg. This is due to the fact that the orbiter park altitude was increased from 287.1 km to 407.4 km, thus reducing the delta-v requirements for the Centaur G.

TABLE 23. - ORBIT TRANSFER RESULTS

	STS park orbit			Delta-V			
	Altitude	Inclination	Launch-site	Burn 1	Burn 2	OTV	Payload capability
Triply-synch	185.2, km	86 0°	WTR	1209, m/s	2689 m/s	TOS/AMS	1,722 kg
Elliptical, 116.6°	185.2	82 0	WTR	1197	3030	Centaur G	3,134
Molniya, 63.4°	185.2	60.	WTR	2463	285	Centaur G Orbit	9,575
Geosynch ellip							
0 6/60°	185.2	59.0	WTR	2724	545	Centaur G	8,007
0.3/30°	370.4	30.0	ETR	2571	992	Centaur G	6,544
						Centaur B	13,054*
Triply-synch circular	185.2	85 0	WTR	1137	3552	Centaur G	2,127
Geostationary	444.5	28.5	ETR	2411	1781	TOS/AMS	3,013
						Centaur G	4,387
						Centaur B	9,572*
Sub-geosynch							
12 h/45°	287 1	45 0	ETR	2647	1421	Centaur G	7,130
						Centaur B	13,900*
12 h/30°	407 4	30.0	ETR	2011	1403	Centaur G	7,236
8 h/45°	287 1	45 0	ETR	1756	1316	Centaur G	8,745
						Centaur B	17,195*
8 h/30°	407.4	30.0	ETR	1720	1295	Centaur G	8,910
6 h/45°	287.1	45 0	ETR	1518	1198	Centaur G	9,804
6 h/30°	407 4	30 0	ETR	1481	1175	Centaur G	9,993
						Centaur B	21,770*

*Payload capability assumes separate STS launches for OTV & satellite.

3.4 TECHNOLOGY SURVEY

A survey was performed to assess the state-of-the-art (SOA) of subsystem technologies applicable to direct broadcast satellite systems. The survey sources included discussions with vendors in specific technology areas, data contained in the NASA Space Systems Technology Model, the NASA Systems Cost and Design Model, and information contained in the literature.

The objective of the survey was to identify applicable hardware that is space-qualified hardware that is being built for space use, and hardware that is or was intended for space application, but has not reached the qualification phase. The results of the technology survey were used to help identify critical technology areas, to identify and define system and subsystem requirements, and to enhance the ability to estimate size and cost of systems and subsystems. Figure 29 shows the scope of the technology survey and the information flow to other program tasks.

The technology survey was categorized by VOA satellite subsystem. Simplified block diagrams of some of these subsystems appear in Figures 30 through 32.

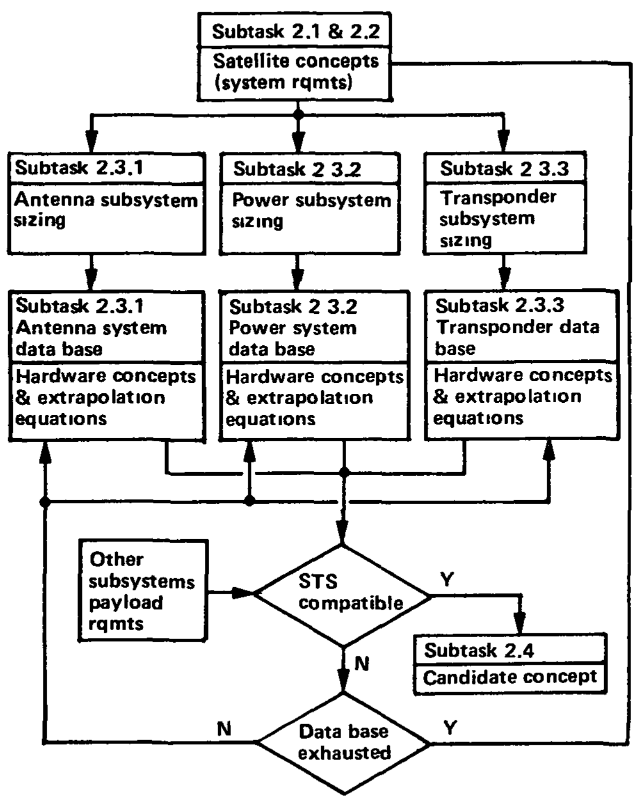


Figure 29. - Technology survey.

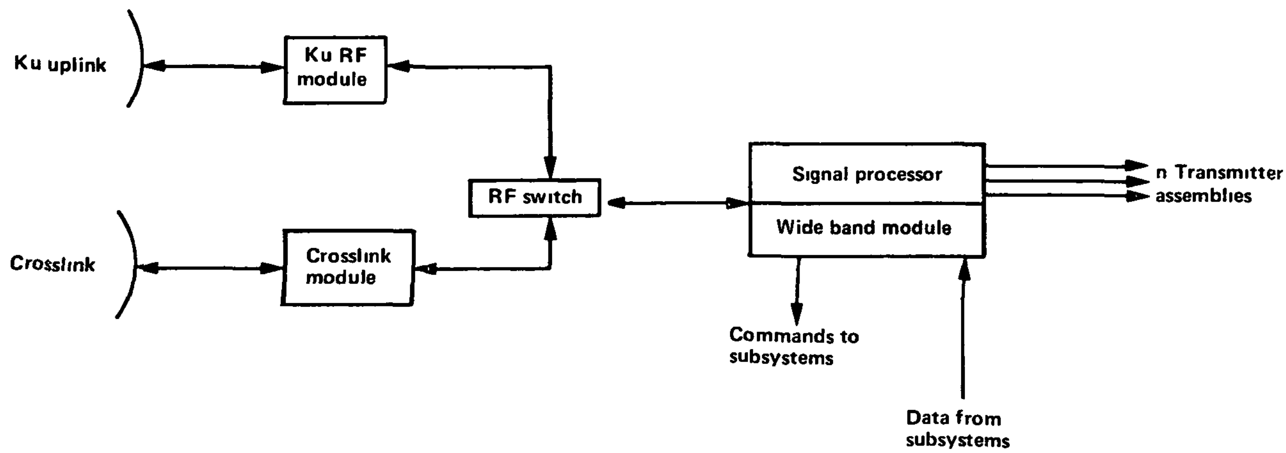


Figure 30. - Communications subsystem functional block diagram.

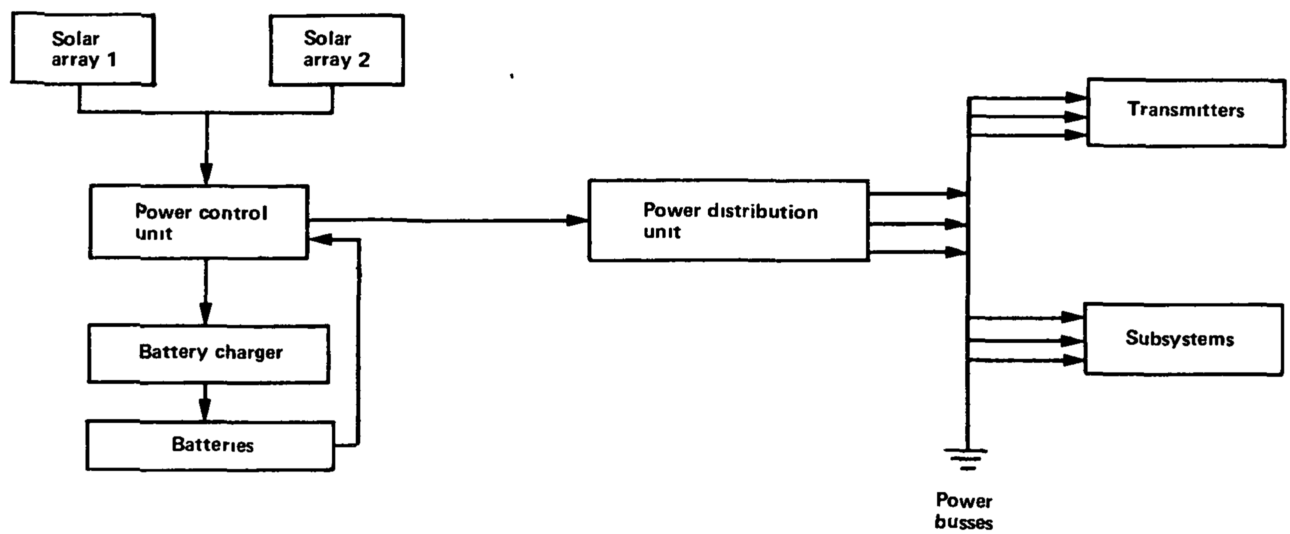


Figure 31. - Electrical power subsystem functional block diagram.

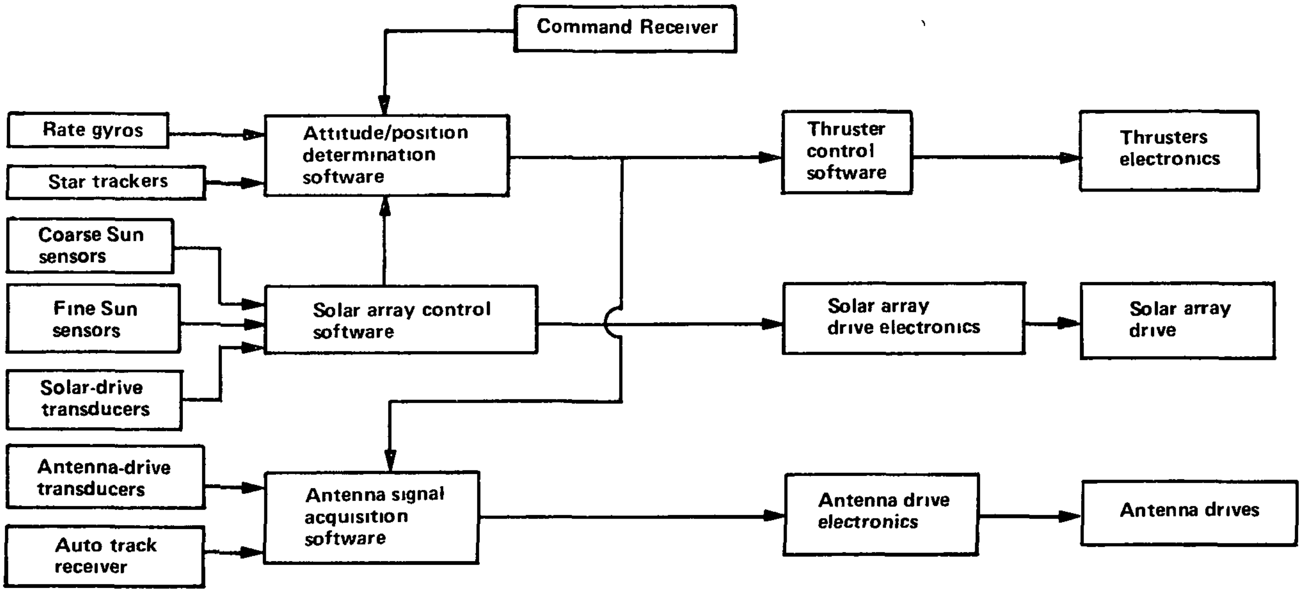


Figure 32. - ACS functional block diagram.

3.4.1 Antenna Technology

The antenna types considered in the VOA study fall into two classes: reflectors and arrays. Each class had advantages and disadvantages for DVBS applications, with the frequency of operation and aperture size of major importance. Reflectors had a distinct advantage from a simplicity standpoint. However, when large amounts of RF power were required, e.g, for HF-, VHF-, and L-band systems, an array had the advantage of allowing a distribution of the power over a number of element/transmitter pairs thus reducing the possibility of any arcing and multipacting effects from the generation of high RF power in space. This also proved to alleviate heat dissipation problems, and will provide graceful degradation if a transmitter should fail. Arrays also proved to have another advantage in that the pattern characteristics could be changed to scan the main beam and match the required coverage for the various zones. For this study, the aperture efficiency for any reflector system was considered to be 50% while an array system had an aperture efficiency of 80% for nonscanned array and 75% times the cosine of the scan angle for a scanned array. The 50% efficiency factor is generally used for reflector systems to account for the feed mismatch and circular polarization (CP) conversion losses, spillover loss and the losses due to reflector surface accuracy. The 80% array efficiency accounted for the losses associated with array mismatch, amplitude and phase control of the elements and CP conversion. For scanned arrays, an additional 5% reduction in the maximum aperture efficiency was used to account for the additional losses from phasors used to provide the scanning ability.

3.4.1.1 Parabolic Reflector Antennas

Parabolic reflector antennas in the 1-to 4-meter range are the most widely used on existing broadcast systems and proved to be acceptable candidates for the Ku-band system design. Antennas of this size are generally constructed of a single rigid dish and either a single feed horn or a small feed array. To increase structural and thermal stability, the most recent systems have used graphite composites in the reflectors and antenna support structures. An example of this technology is Intelsat VI (Fig. 33) which uses four graphite rigid dish parabolic reflectors. The smaller reflectors are fed with conventional horn feeds, while the larger 2- to 3-meter reflectors are fed using a honeycomb array feed.

For parabolic reflector antennas in the 4- to 20-meter size, two types of antenna systems, wrap-rib antennas and radial rib antennas, are most widely used. Because of aperature size, these reflectors are generally required to be stowed for STS launch.

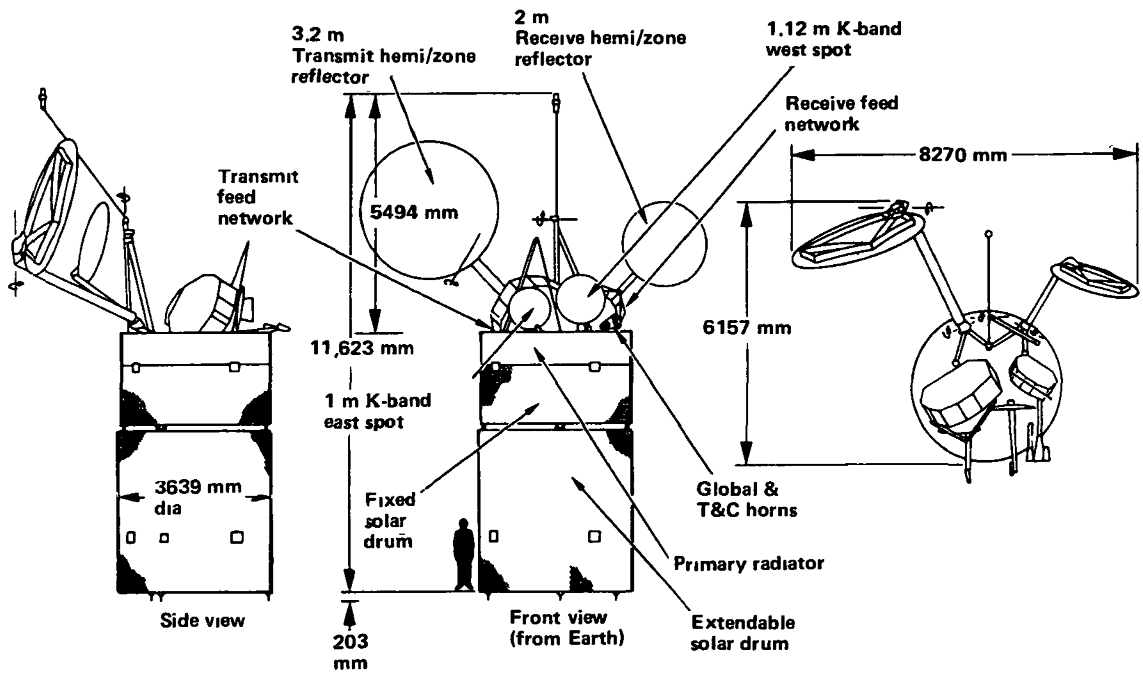


Figure 33. - Intelsat VI antenna farm dimensions.

Wrap-rib antenna. - The first system, the wrap-rib design, has been developed by Lockheed. The best known application is on the ATS-6 spacecraft, which uses a 9.1-meter parabolic, wrap-rib reflector antenna operating up to and above 8 GHz (Fig. 34). The ATS-6 antenna, made with aluminum ribs and conventional thermal blankets, represents a technology about 10 years old. Recent developments using this concept have resulted in a manufacturing capability for wrap ribs using structural composite materials with low coefficients of thermal expansion. New materials and manufacturing processes for reflector mesh have been developed recently, and the analytical capability for the detail design of the structure has been improved recently. These developments have made it possible to design, build, and predict antenna performance for lighter and more stable wrap-rib structures.

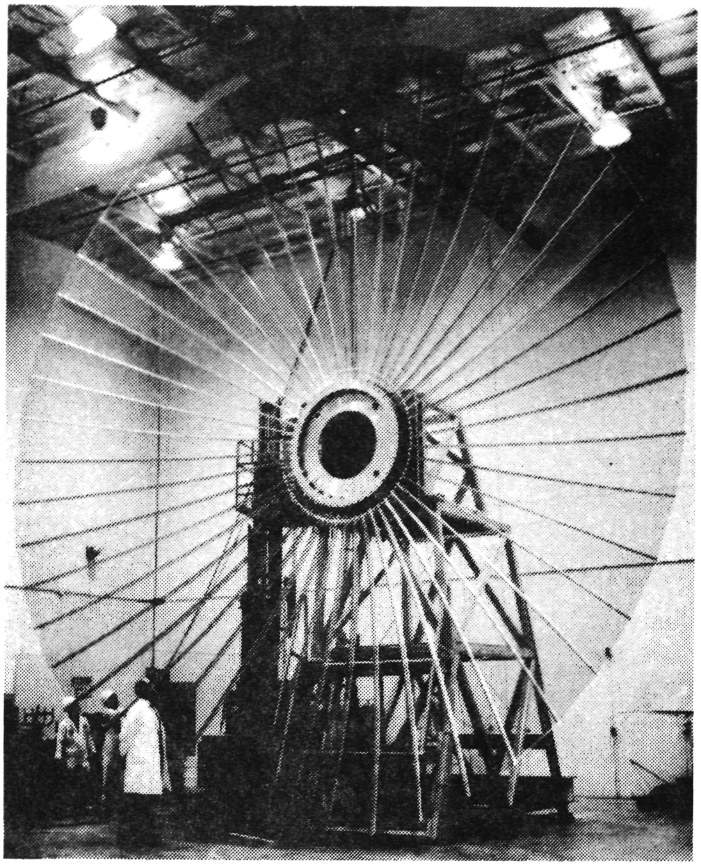


Figure 34.
LMSC ATS-6 flight antenna reflector.

Radial rib antenna. - The radial rib antenna has been used by both Harris Corporation and TRW on recent satellite systems. TDRSS uses two 6-meter diameter radial-rib antennas built by Harris (Fig. 35). Harris has also developed the radial-rib concept to the point of demonstrating that this technology qualifies for flight applications of antennas up to 18.3 meters in diameter and operation up to K-band. TRW's radial-rib antenna has been used on the FLTSATCOM satellite. The FLTSATCOM antenna is 4.9 meters in diameter and operates at 300 MHz. The reflector consists of a 2-meter diameter solid center dish and a deployable outer ring of rib-supported mesh (Fig. 36).

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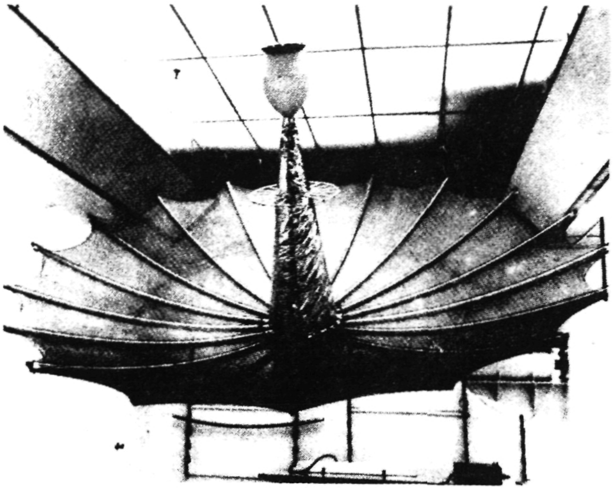

Figure 35. - TDRSS radial rib antenna.

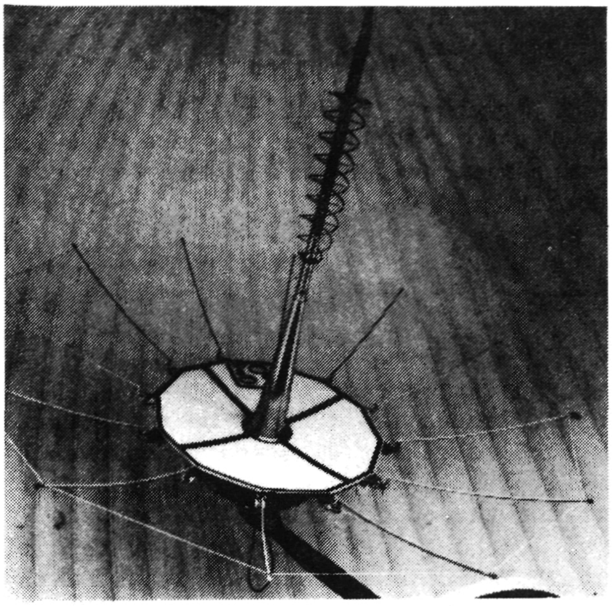

Figure 36.
TRW 5-m FLTSATCOM flight antenna concept.

For parabolic reflector antennas greater than 20 meters in diameter, no flight hardware subsystems exist; however, antenna systems of this size have been analyzed and prototype hardware is being built and tested that demonstrates the feasibility of these systems up to the 300-m diameter range. To meet the requirements of the HF and VHF-bands, antenna systems that use parabolic reflectors will require these large aperture reflectors. Because large aperture reflectors are orders of magnitude greater in size than the STS payload bay, the reflector support structures are required to fold and stow compactly for launch and then deploy with a high degree of reliability when in orbit with a typical stowed diameter-to-deployed diameter ratio of 0.04. Also, due to the focal lengths of large aperture reflectors, most large aperture reflector concepts attach the feed support onto the reflector support structure. The reflector support structures generally dictate the surface quality and make up the largest portion of the weight and deployed volume of the antenna system. For this reason, the stiffness, packaging efficiency, and weight of the various reflector support structure concepts is the major factor in the overall antenna system's weight, volume, and surface quality. The following candidate large reflector concepts were considered for the VOA study.

Lockheed wrap-rib reflector. - In addition to the smaller versions of the wrap-rib antenna reflector used on antenna systems such as ATS-6, Lockheed has been developing structural concepts for the wrap-rib design for reflectors up to 250 meters in diameter. This concept has the most efficient stowage density of all the radial-rib configurations, is the most mature in design development of large rib antennas, is capable of diameters to 250 meters, and is relatively light compared to other radial-rib systems. The concept can have a ring added at its circumference for increased stiffness. The wrap-rib antenna consists of a hollow, doughnut-shaped hub to which a series of radial ribs, formed to the shape of a parabola, are attached. A lightweight reflective mesh is stretched between these ribs to form the paraboloidal reflecting surface. Figure 37 shows four ribs of the 55 meter proof-of-concept hardware being built by Lockheed.

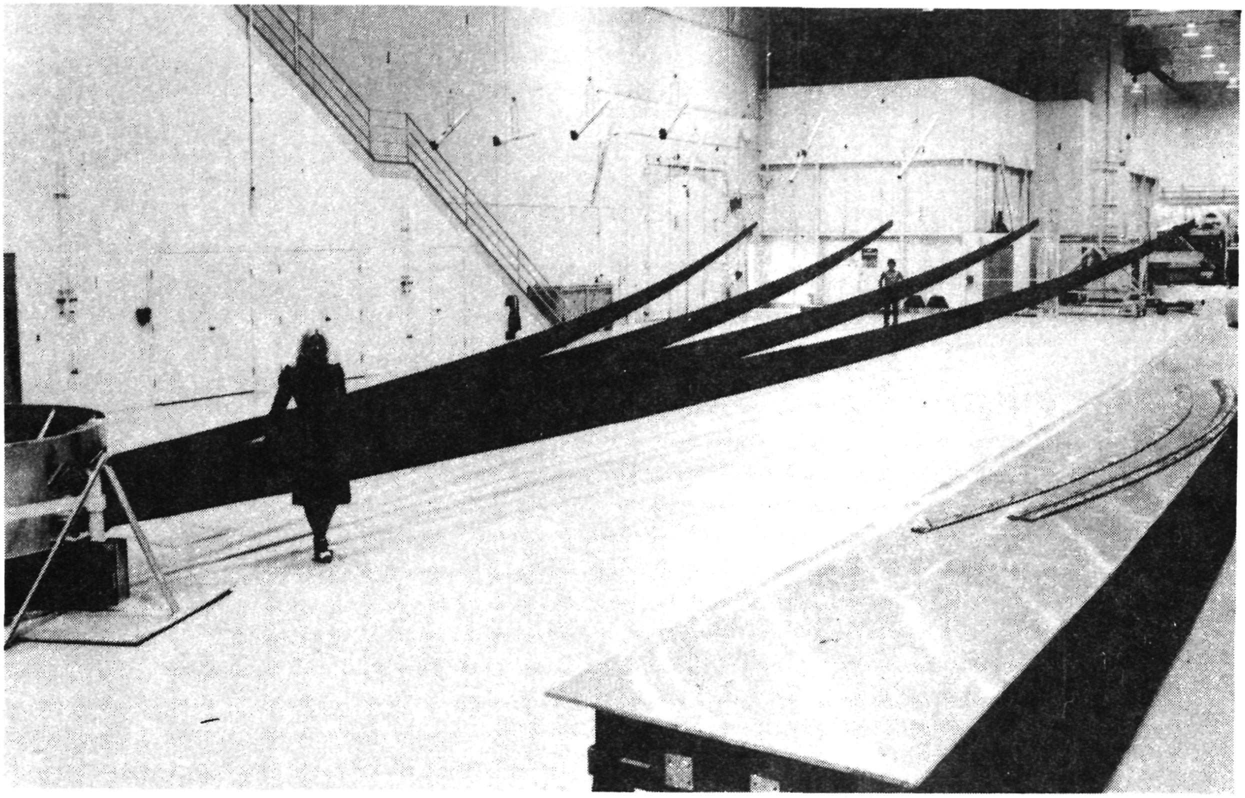

Figure 37. - Lockheed's proof-of-concept wrap rib reflector.

Articulated radial-rib reflector. - This concept (Fig. 38) is a logical extension of Harris' smaller radial-rib design, but has the flexibility to accommodate larger diameters and retain the same packaging efficiency. It consists of a central mast that supports the feed and to which rigid radial ribs are attached by privots at the base. Because of the antenna diameters under consideration and the constraint of the limited stowed volume available, it is necessary to put an articulation at the midspan of each rib. The ribs approximate a parabolic contour and have adjustable standoffs to which the reflective mesh is attached. The surface is shaped between the ribs by the secondary drawing surface technique. The concept is attractive from an experience standpoint, but there are serious packaging size limitations. The shortest stowed length with a single articulation in the ribs is one-quarter of the antenna diameter. For a 100-meter diameter antenna, this length would become prohibitive. Another articulation for each rib is possible, but the added mechanical complexity and probable mesh handling problems negate any potential advantages.

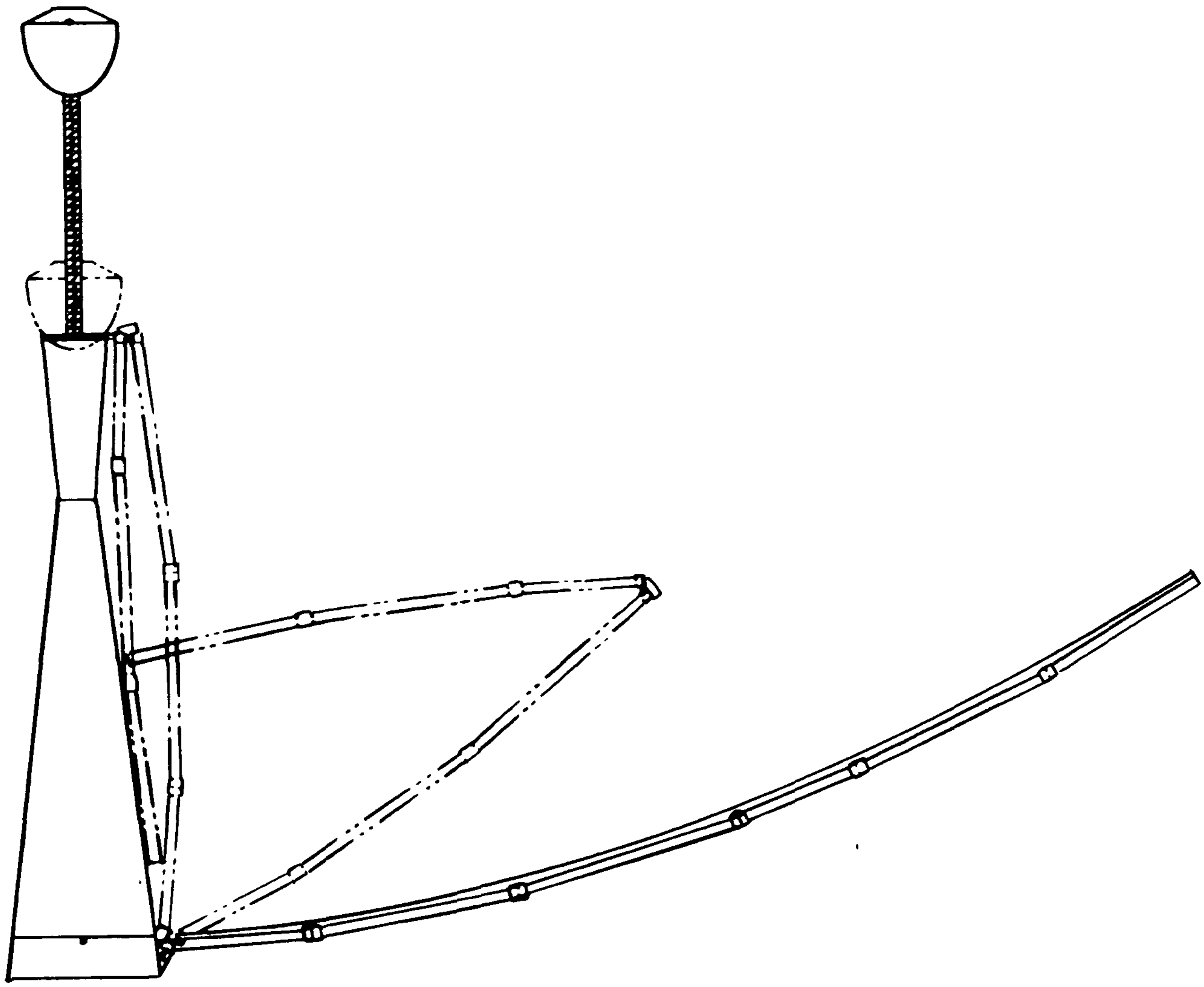

Figure 38. - Harris-articulated radial rib antenna.

Hoop and column reflector. - Both Martin Marietta and Harris have hoop and column reflector design concepts. The fundamental elements of the two concepts are similar but the mechanical design approaches are different. The Harris Corporation hoop and column reflector concept for self-erectable structures is intended for reflector design up to 250-meter diameter (Fig. 39). This concept has been developed to the point of a preliminary design for sizes up to 100-meter diameter; a 15-meter diameter conceptual demonstration model has been built. The fundamental elements of the support structure include the hoop and upper, lower, and center control stringers. The reflector consists of the mesh, mesh shaping ties, secondary drawing surface, and mesh tensioning stringers. The basic antenna configuration is a type of may pole with a unique technique for contouring the RF reflective mesh.

Contiguous truss reflector. - Two basic truss configurations, the Martin Marietta box truss, and the General Dynamics tetrahedral truss are the most widely used for designing contiguous truss reflectors. Components of the reflector structure use the basic bay design regardless of the number of bays. The selection of the number of bays for a given antenna size and application is a function of cost, reliability, weight, and surface tolerance.

The tetrahedral concept is a basic building block used in numerous combinations to achieve the desired shape and size of an antenna structure (Fig. 40). The basic element is a deployable tetrahedron hinged by spider links at each corner. Each tetrahedron forms one truss bay, and the number of bays can vary in number from four to 10 more across the major diameter of the reflector structure. This configuration forms the support structure for the reflective mesh. With tetrahedral trusses, the reflector outline is hexagonal rather than circular so the equivalent reflector diameter is about 10% less than the maximum point-to-point width.

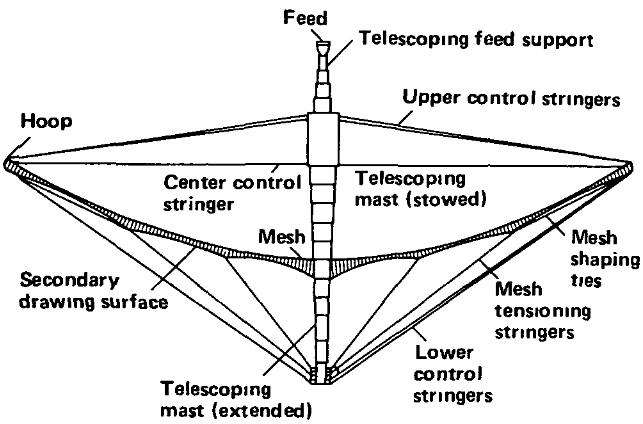

Figure 39 - Harris hoop/column reflector

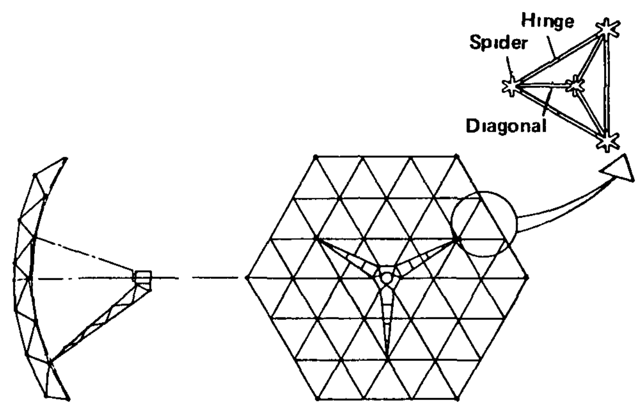

Figure 40 - Tetrahedral truss reflector.

Martin Marietta is developing a deployable box truss structural system applicable to parabolic antennas. Originally proposed for the Air Force's On-orbit Assembly program, we are currently developing the system with prototype beam and truss segments. The system features compact stowage, step-by-step deployment, high deployed precision and reliability, and adaptability to a wide variety of reflector sizes up 200-meter diameter. Figure 41 shows a recent concept of a 120x60-meter antenna system designed for NASA LaRC on the Advanced Earth Observation Spacecraft contract.

Truss ring antenna reflector. - The Martin Marietta box truss has also been used in box truss ring design. This box truss ring consists of box truss trapezoids forming a circular or racetrack ring. Figure 42 shows a box truss ring supporting an electrostatically controlled membrane mirror reflector. The application shown in the figure is a 100-meter diameter ring designed by Martin Marietta for the NASA LaRC Advanced Space Systems Analysis (ASSA) program. The box truss ring is a ultra-lightweight structure that has excellent stiffness. Stowed packaging is very efficient and allows packaging of subsystems in the center of the stowed truss.

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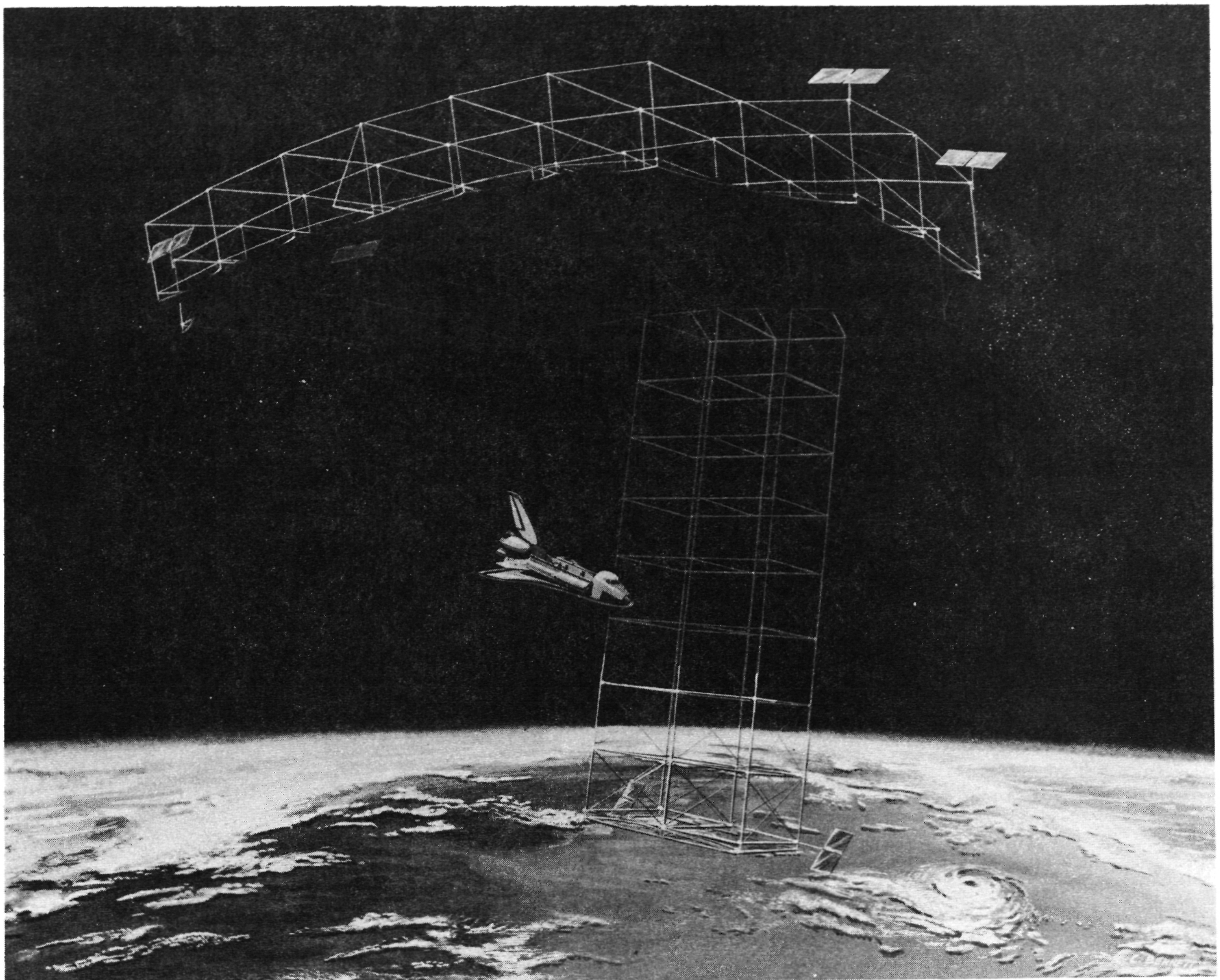

Figure 41. - Advanced Earth observation spacecraft using Denver Aerospace's deployable box truss structure.

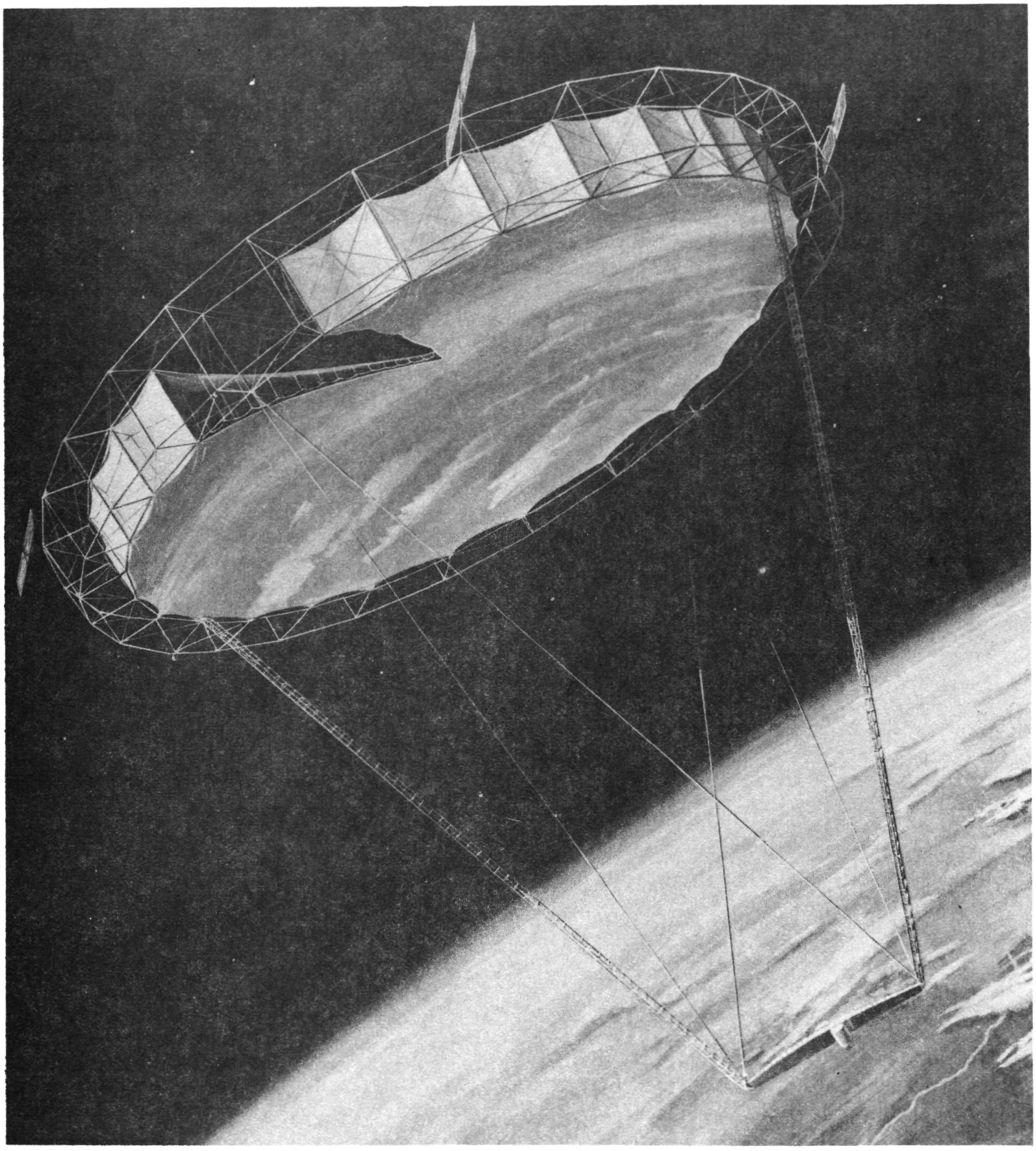

Figure 42. - ASSA using Denver Aerospace's truss ring reflector.

Inflatable reflectors. - Inflatable reflectors can offer low weight, compact stowage, reliable deployment, and adequate shape accuracy. L'Garde Incorporated and others are developing advanced antenna concepts using inflatable, thin-film bodies. These bodies incorporate surfaces of revolution such as a paraboloidal reflector capped by a cone (Fig. 43). Shape accuracies for

HF- and VHF-bands with 100-meter diameter apertures appear to be easily achievable. By operating at very low pressures (e.g, 10^{-6} psi), the weight of inflatant gas lost through micrometeoroid holes can be kept within practical bounds. A metallized surface forms the reflector and, to achieve adequate thermal control, the cone can be more than 90% metallized in a patchwork fashion; each patch being very small relative to the broadcast wavelength. In geostationary orbits, such reflectors could depressurize thermally and have degraded shape accuracy only during two periods per year (equinoxes) and, during those periods, only for short period (less than 3 hours) bracketing local midnight. It is also interesting to note that the feeds, with their high heat dissipation, can be located outside the inflatable body.

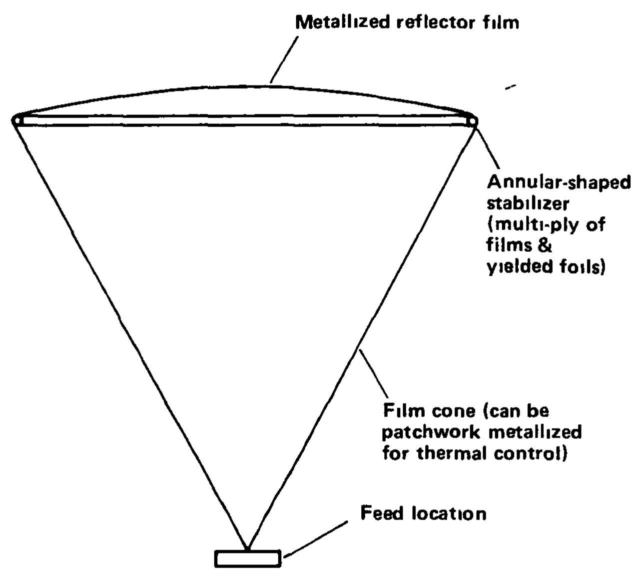

Figure 43
Typical inflatable reflector configuration.

3.4.1.2 Array Antennas

Similar to large reflector antenna concepts, array concepts require an exterior support structure. Unlike the lightweight reflective surface, the most mature array antenna technology uses large rigid panels containing array elements. For these systems, array panels dictate the weight and volume of the antenna system with the support structure accounting for only 10% of the total system weight and volume. Due to the fact that the rigid arrays must stow by folding, the array size is limited to moderate size apertures. Also, since the back surface of the panel is generally used as the ground plane, the panel thickness is dictated by the operating frequency of the array, i.e., generally $\lambda/4$ in thickness. This type of array proved acceptable for the L-band system. The HF and VHF systems, because of their long wavelengths, required a more advanced, yet unproven array systems using lightweight film material with thin array elements. For these systems, the support structure again dominates the total antenna system's weight and volume. The following are the various array system concepts studied under this contract. The first two concepts, the synthetic aperature radar (SAR) antenna and the gate frame truss phased array, are examples of technology that exist for the rigid panel type array systems. The remaining concepts show examples of membrane type array systems, capable of sufficiently separating the ground plane from the radiating surface allowing these concepts to be used for HF and VHF-band applications.

SAR antenna. - One of the key science instruments and a unique feature of the Seasat-I Spacecraft (launched during spring 1978) is the SAR antenna (Fig. 44). This antenna is a planar-phased array that measures 10.7 by 2.2 meters in the deployed configuration and operates at 1.275 GHz. The planar array consists of eight 1.3 by 2.16-meter rigid and structurally identical fiberglass honeycomb panels. The panels are hinged together in series, but are individually mounted to a deployable truss structure that provides support and alignment for the panels. The deployable truss, in turn, is supported by a deployable tripod structure whose function is to support and govern deployment of the truss and panels (Fig. 45). The tripod was attached to a biaxial actuator mounted directly on the Agena spacecraft.

The SAR antenna subsystem was developed by Ball Brothers Research Corporation. Lockheed Missiles and Space Company, was responsible for development of the flight hardware and integration of the SAR antenna with the Agena spacecraft. The weight of the antenna system is about 113.4 kg (250 lb).

Gate frame truss phased array antenna. - Figure 46 shows the gate frame truss structural system under development at Martin Marietta. It is specifically designed for compact stowage and high precision and stiffness when deployed with integral, rigid array panels. The support structure makes up about 5% of the total system weight and about 15% of the stowed volume.

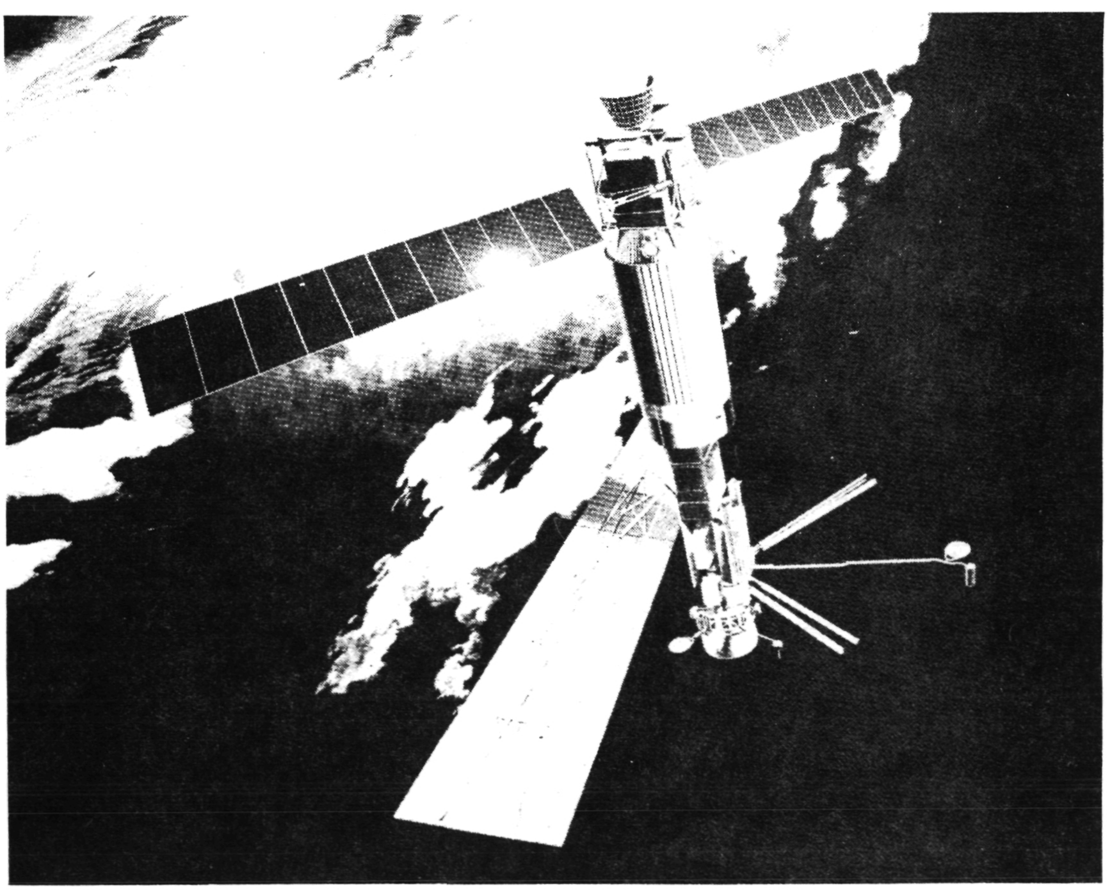

Figure 44. - Seasat-I spacecraft—cruise configuration.

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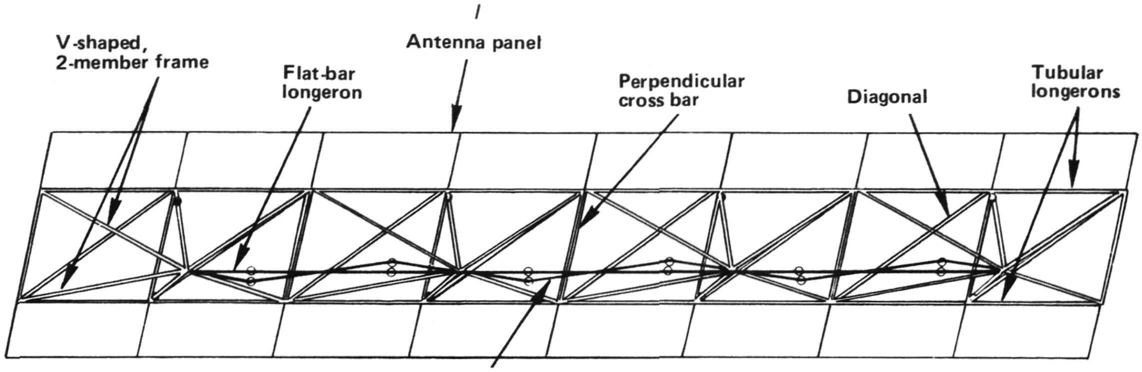

Figure 45. - Astro Research Corporation's deployable support truss for the panels of the Seasat-I SAR antenna.

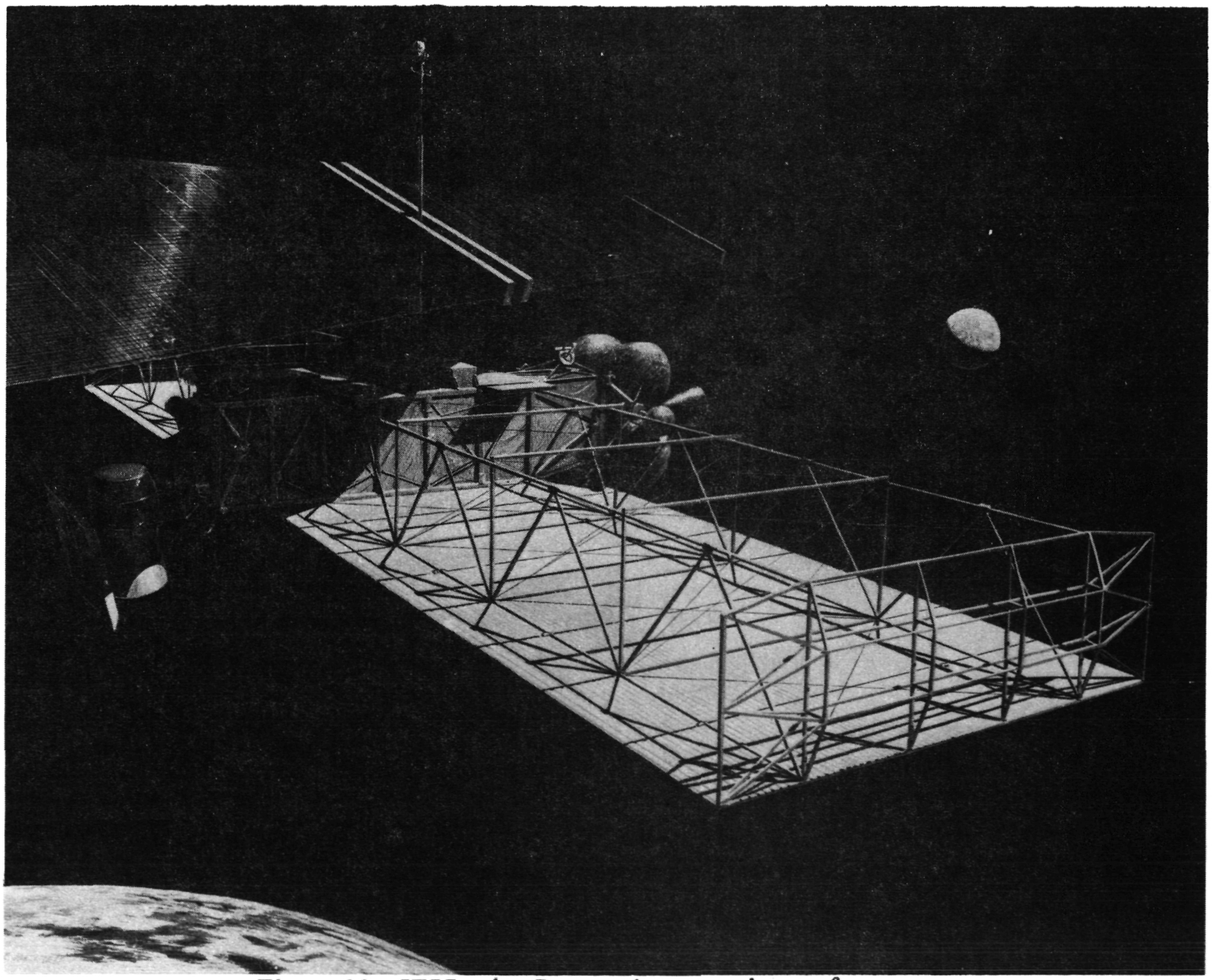

Figure 46. - ITSS using Denver Aerospace's gate frame truss.

Hoop and column phased-array antenna. - The Grumman space-fed, phased-array concept for self-deployable antennas is intended for designs up to 300-meter diameter and operation at L-band or lower (Fig. 47). Grumman developed this concept to the point of a preliminary design for a 60-meter diameter antenna and a 1.3-meter diameter mechanical model. The mechanical model was used to demonstrate and evaluate the basic mechanical conceptual design. The primary limitation of this concept is the complicated deployment and the low structural vibration frequency of the deployed membrane.

The Grumman antenna concept is a planar array whose basic support structure is a wire-wheel configuration. This concept development was centered around the design of 61-meter diameter and 300-meter diameter space-fed, phased-array antennas for operation at L-band. The phased array is composed of 32- to 72-gore panel assemblies and their tensioning devices.

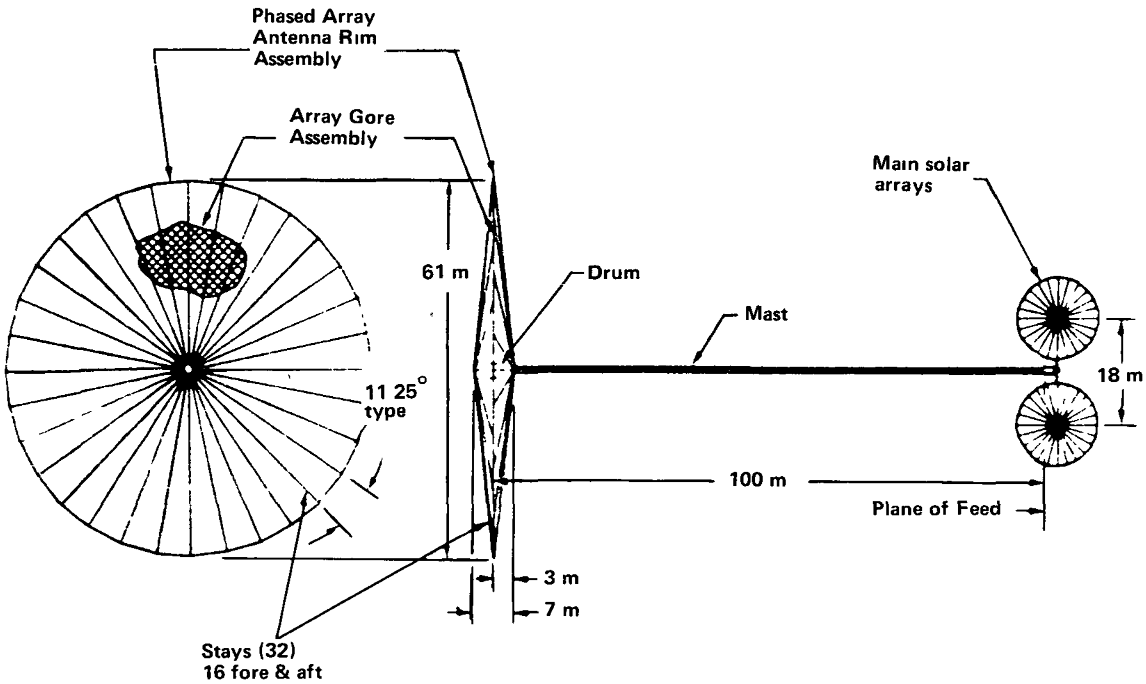

Figure 47 - Basic structural elements of Grumman phased-array concept.

Box truss ring phased array. - Similar to the Martin Marietta box truss ring reflector (see Fig. 42) a box truss ring phase array consists of box truss trapezoids forming a circular or racetrack ring. Instead of a membrane mirror reflector, the radiating surface is stretched flat across the ring with flexible bow-tie radiating elements attached on the surface. The ground plane is a second stretched membrane on the back surface, with the box truss depth being adjusted to provide the correct spacing between surfaces. The back surface also acts as a support structure for the antenna transmitters. Figure 48 illustrates this concept for an 80-meter array. This concept provides a lightweight and efficient method for producing a phased array in which the ground plane is required to be at a relatively large distance from the radiating plane. Also, since stowage of the system does not require the box truss

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to stow in depth, rigid coax can be used between the transmitters and radiating elements. (Rigid coax is desirable when high power is to be transmitted.) Another desirable feature of this system is the distribution of the transmitters over the back surface, providing a method to allow each transmitter to carry its own radiators to dissipate waste heat. This feature eliminates the need for a central radiator and a large and heavy system of heat pipes.

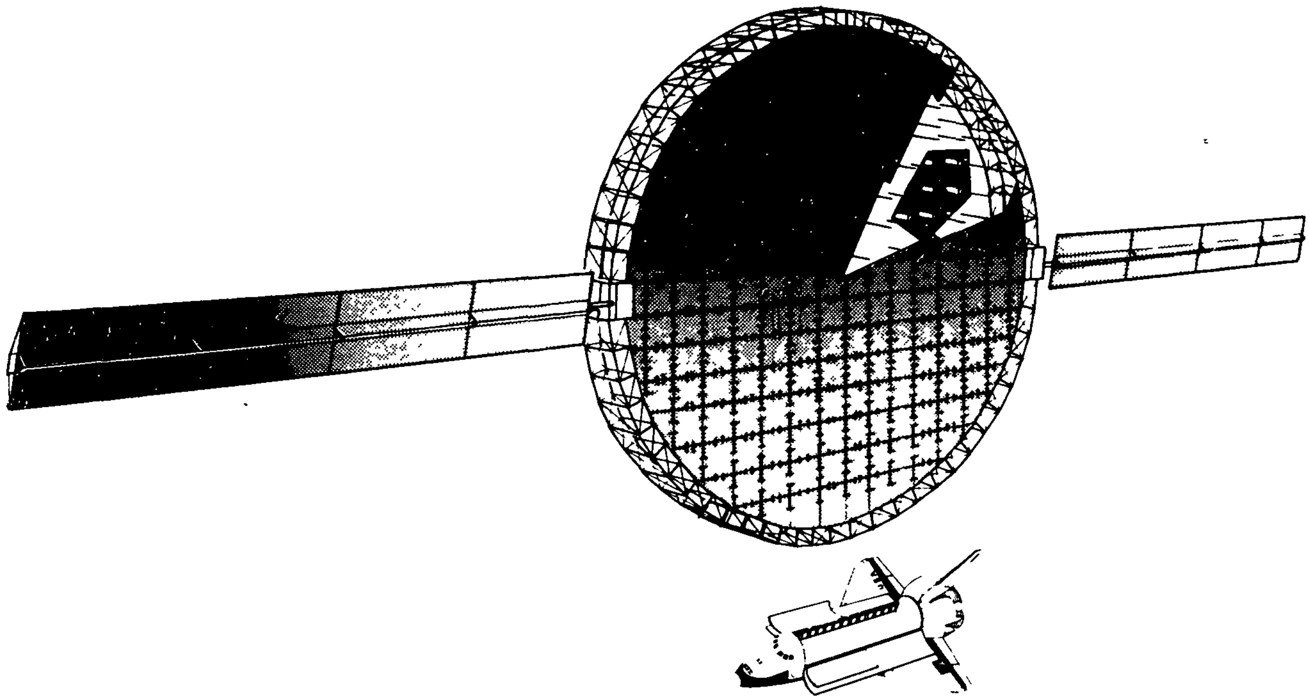

Figure 48. - Box truss ring-phased array.

3.4.2 Satellite Feeder Link Technology

The satellite uplink frequency bands allocated for the United States by the 1979 World Administration are summarized in Table 24. Also shown are intersatellite frequency allocations and unallocated bands. Technology for uplink, either for realtime or delayed retransmission, is SOA. The LANDSAT-D Ku-band technology will be adequate for VOA DVBS concepts. Figure 49 shows the basic concept and components required for the satellite feeder link. Earth station transmission is typically provided by high powered amplifiers such as traveling wave tubes (TWT) or Klystrons, again representing SOA technology. The specific Earth station design will depend on the spacing of satellites and whether or not intersatellite links are used. A later discussion presents a tradeoff of use of multiple Earth stations vs intersatellite links.

TABLE 24. - ALLOCATED SATELLITE BANDS FOR THE U S.

	Frequency band		Major uses in U.S.	Bandwidth
	Uplink	Downlink		
C-band	5 9-6.4 GHz	3 7-4 2 GHz	Fixed, point-to-point ground stations; nonmilitary	500 MHz
X-band	7 9-8 4	7 25-7 75	Mobile (ships, aircraft), radio relay, military only	500 MHz
Ku-band	14 0-14 5	11 7-12.2	Broadcast & fixed-point service; nonmilitary	500 MHz
Ka-band	27-30 30-31	17-20 20-21	(unassigned)	–
V-band Q-band	50-51	40-41 41-43	Fixed-point, nonmilitary Broadcast, nonmilitary	1 GHz 2 GHz
V-band	54-58 59-64		Intersatellite Intersatellite	3.9 GHz 5 GHz

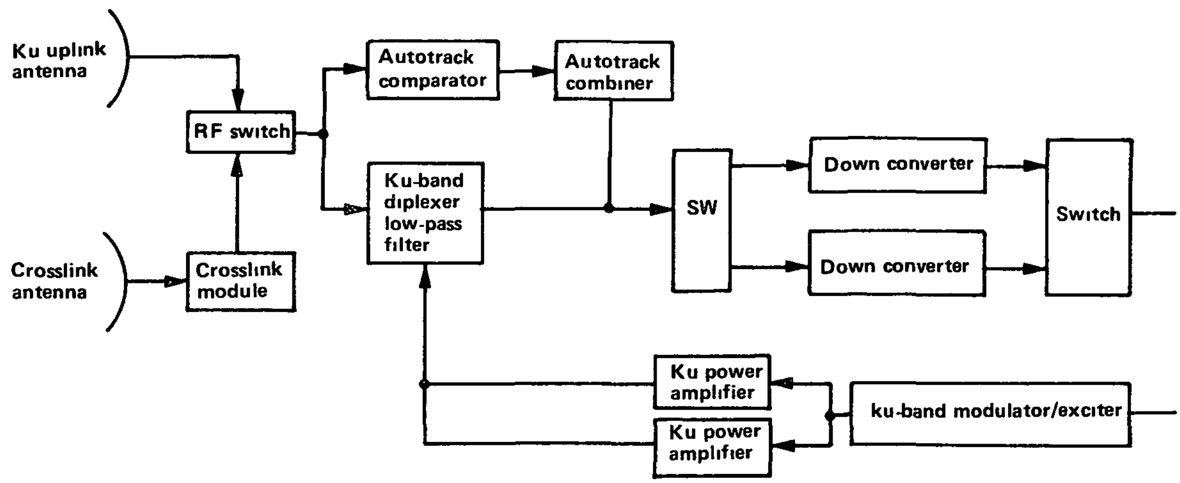

Figure 49. - Ku RF module block diagram.

3.4.3 Satellite Signal Processing Technology

The onboard signal processing generates the carriers for the downlink. Figure 50 shows options for satellite signal processing. All are existing SOA technologies. The method to be used will depend on the specific DVBS, considering band and number of carriers to be processed.

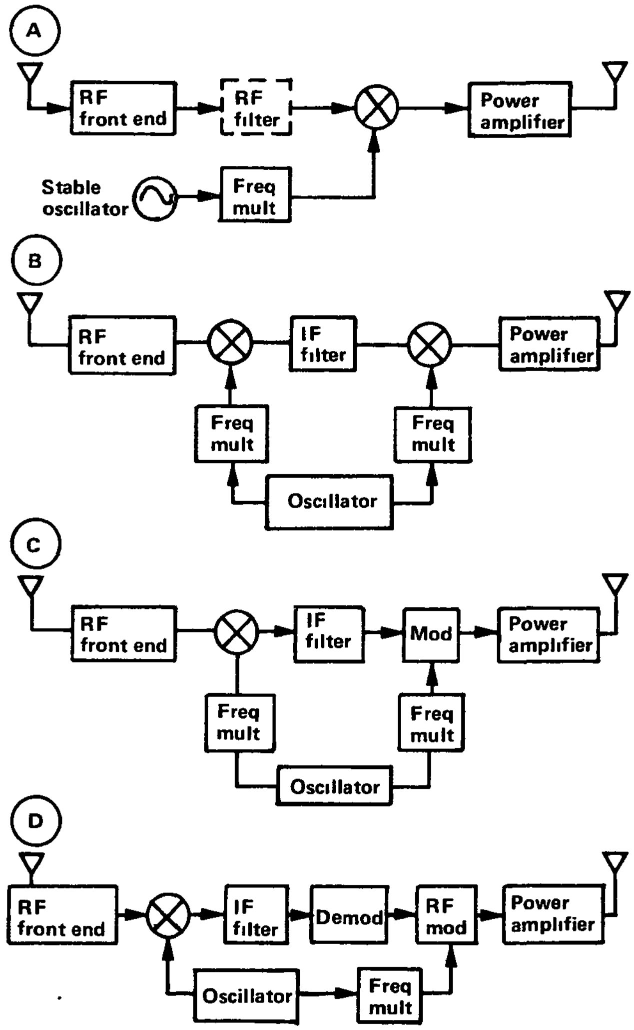

Figure 50. - Satellite processing models.

3.4.4 Transmitter Technology

The transmitter technology for Ku-band will use SOA TWTs. Figure 51 summarizes existing TWT technology. Power output levels on existing satellites are found between 5 and 250 watts. Higher power levels are considered easily attainable for use in space. Efficiencies of TWTs are typically about 35 to 50%.

For the L-band, TWTs are applicable for output power levels below 250 watts. Above these levels, solid state power amplifiers (SSPA) are an alternative. When used in phased array systems, multiple SSPAs can be used. 6-watt space qualified SSPAs exist in the L-band range. Terrestrial and airborne SSPAs are available up to 1000 watts.

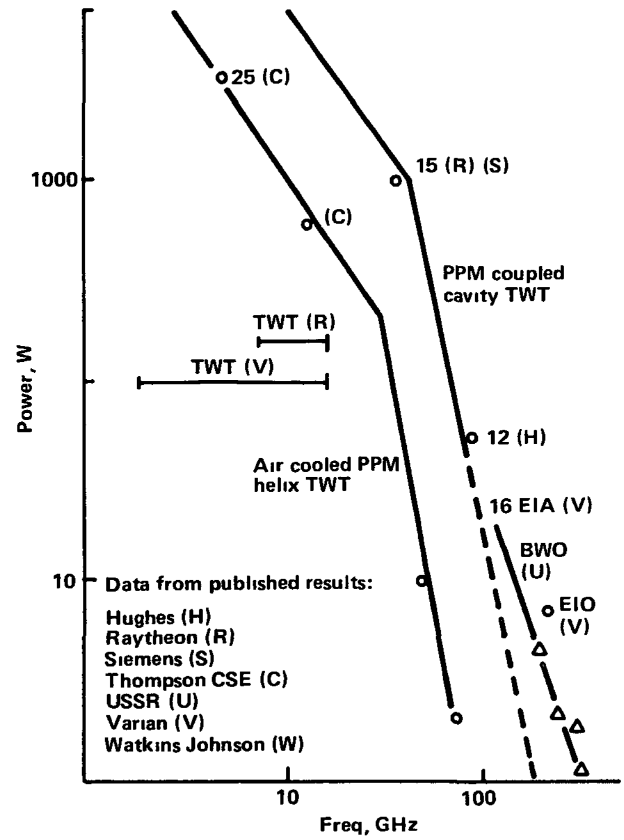

Figure 51. - Microwave power amplifiers technology.

HF- and VHF-bands SSPAs will require new design approaches. Double-sided printed circuit boards are projected to provide packaging densities of 700 kg/m^3 or greater. To meet a 7-yr lifetime requirement, designs will include internal redundancy and modularity. Current technology bipolar transistors are suitable for powers up to a 500 watts. A bipolar NPN transistor rated at 200-watts at 300 MHz has been developed. A 600 watt SSPA for low UHF-band is currently under development for space use. The same design is projected for growth to 1000 watts with minimal redesign. A 400 watt SSPA for operation from 100 to 500 MHz has also been developed for airborne military applications.

Use of power MOSFETs will make power levels of up to 20 kW possible in the 1990's. SSPAs using MOSFET technology will operate in a class-D mode and use dual single-ended, push-pull stages for powers greater than 1000 watts. A 200 Vdc power bus will be sufficient for powers up to 1.5 kW, but for higher powers (up to 20 kW) a higher bus (400 Vdc) will be required.

TABLE 25. - TRANSMITTER PARAMETRICS

Transmitter power	Mass	Volume
50 W	5.10 kg	0.0105 m^3
100	9.94	0.0177
200	12.81	0.0182
500	16.21	0.0219
750	19.61	0.0255
1000	23.02	0 0292
2000	36.62	0.0439

Predicted weights and volumes for SSPAs (including bipolar and MOSFET technologies) are summarized in Table 25. These weights and volumes assume a dc voltage bus on a satellite that results in limited required power conditioning. By using a regulated bus and with projected advances, efficiencies of 65% are predicted for SSPAs for DVBS in the HF-, VHF-, or L-bands.

3.4.5 Power Generation System Technology

Some VOA mission configurations considered require the energy source be capable of delivering power greater than 100 kW. The load power is a function of antenna array size and orbit altitude and optimum power levels will be determined through detailed parametric analysis. To qualify as a viable energy source, it was assumed that the source must be capable of supplying at least 100 kW, since the optimum power level will be near this value. All possible energy generation sources are diagrammed in Figure 52.

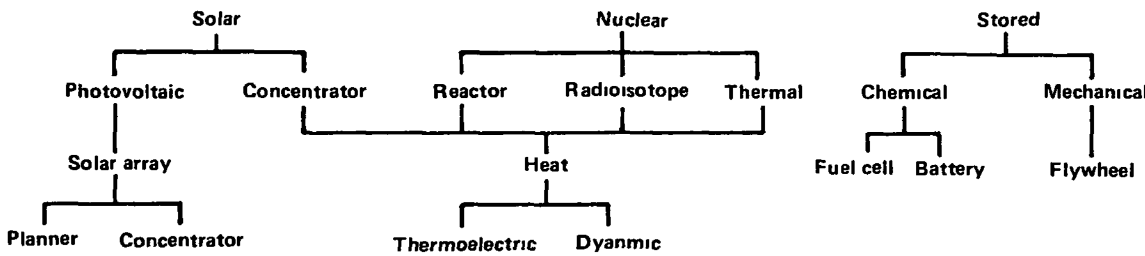

Figure 52. - Energy source choices.

Several choices may be eliminated immediately. All stored sources are secondary energy sources and thus are not applicable since they require periodic energy input to replenish their capacity. Radioisotope sources, such as radioisotope thermal generators (RTG) or radioisotope dynamic sources such as organic rankine cycles and brayton isotope power systems, have not been developed to produce the high power levels required. Dynamic systems of the type mentioned are promising sources (in the near term) in the 1 to 25 kW range only. Solar concentrator/ thermoelectric and solar dynamic sources are under early stages of development and although theoretically capable of high power output, the technological risk is high in the near term. These sources will be discussed further in Section 4.2, Technology Tradeoffs. Another source that presents difficult design challenges is nuclear reactor. This may be the only feasible source at some orbital altitudes (e.g., in the Van Allen belt). The most viable energy sources are solar photovoltaic. These latter two sources are discussed in Section 3.4.5.1.

3.4.5.1 Solar Photovoltaic Sources

Many advances have occurred in the photovoltaic field over the last 15 years and consequently a wide variety of choices now exist in the design of a solar array. Figure 53 graphically summarizes these improvements in terms of cost and efficiency. The types of solar cells currently available and under development are single crystal silicon cells, advanced vertical junction cells, gallium arsenide cells in a planar or concentrator array, multiband gap (MBG) cells, and three junction cascade cells. Advanced concepts under study to improve solar cell performance are the high concentration (100 suns) ratio miniature cassegrainian concentrator (MCC) concept, metal interconnects between cells, and the surface plasmon concept. The cell types together with the advanced concepts will be discussed here and in Section 4.2 with the aim of selecting optimum energy sources for use in the power system parametric equations.

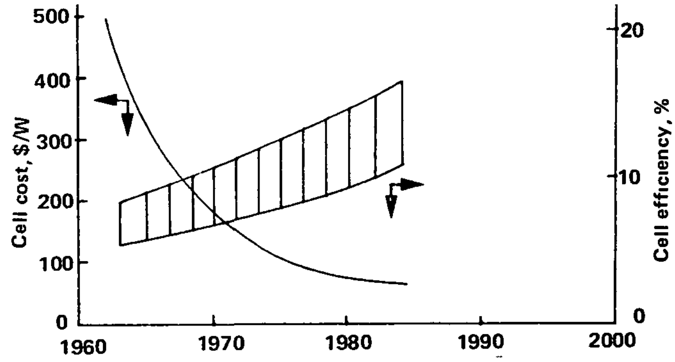

Figure 53. - Progress in solar cell technology.

Beginning with the FRUSA flight experiment in 1971 and the NASA developed solar electric propulsion (SEP) array in the early 1970s, significant improvements have been made in single crystal silicon solar cell technology. In the SEP, a 66W/kg flexible foldout design capable of supplying 25 kW beginning of life (BOL) was developed. A second generation SEP is currently under development and specific powers as high as 300 W/kg will likely be realized. Tantamount to achieving higher specific powers is the increase of the silicon cell conversion efficiency. Presently, state of the art efficiencies are near 14% for planar silicon arrays. It is not reasonable to expect efficiencies to increase much above this in the near future. Therefore, the use of higher efficient (up to 18%) gallium arsenide cells or concentrator arrays will be required to demonstrate very high specific powers of 400 W/kg or more. Although specific powers in the 300-400 W/kg range are not necessary to construct a 100 kW array, they result in lower array area which reduces spacecraft drag and decreases the propellent mass necessary for altitude maintenance.

Silicon solar cells have many advantages. They are flight proven, and relatively lightweight; advances in the ultrathin (50 μm) cells using advanced cell processing techniques such as back surface fields and reflectors, plus front surface texturing and multiple layer antireflection coatings, promise to push efficiencies to 15% by the mid-1990s. Silicon cells have the lowest cost; the use of larger cell sizes (6x6 cm and 8x8 cm) will reduce cost even more. Additionally, space station will probably use a planar nonconcentrated silicon array and thus incur much of the developmental cost for a high power (> 50 kW) system. The ultrathin cells and large cells have experienced problems with bowing but the use of a gridded pattern for the back contact configuration appears to have eliminated this with the added benefit of reducing cell weight. Silicon cells have the disadvantage of possessing the lowest efficiency of the new cell types, which means an array comprised of silicon cells will have the largest area. This translates into high inertias and highest effect from drag or solar pressures, which gives the highest orbit maintenance subsystem mass. Table 26 summarizes the advantages and disadvantages of silicon cell technology.

Gallium arsenide solar cells have been demonstrated to operate at efficiencies as high at 18% (20% with concentrators). This equates to a reduction of solar array area up to 30% of a silicon array. Also, gallium arsenide cells have a lower temperature coefficient than silicon cells and are more radiation resistant. The lower temperature coefficient makes gallium arsenide cells ideal for operating with a concentrator and the cassegrainian concept should demonstrate this. However, gallium arsenide cells are approximately two to three times the weight of an equivalently sized silicon cell and are presently expensive to manufacture. Reductions in cost and blanket weight are likely to occur in the future and with projected efficiencies of 20%, gallium arsenide will become very attractive. However, considering that a gallium arsenide array has yet to be flown, the technological risk and initial cost would be high to develop a gallium arsenide array for a mid-1990 launch. Table 27 summarizes the advantages and disadvantages of gallium arsenide cell technology.

TABLE 26. - SILICON SOLAR CELL TRADEOFF

Plus	Minus
Flight proven 15% Efficiencies possible Lightweight Low cell cost Large cell size possible Small development cost	Large array area — highest orbit Subsystem mass Array stiffness may be an issue

TABLE 27. - GALLIUM ARSENIDE SOLAR CELL TRADEOFF

Plus	Minus
Higher efficiencies vs Si	
Compatible with concentrators	High cost
Lower temperature coefficient vs Si	Not flight proven
Higher radiation resistance vs Si	Higher weight vs Si
Reduction in spacecraft inertias	Higher weight may negate smaller area

The most promising of the advanced cells are the three junction cascade cells with possible efficiencies of 30%. MBG cells, which use two materials with different bandgaps to convert more of the solar spectrum into electricity, show signs of full development in the near future. Vertical junction cells possess high radiation resistance but this is offset by a high solar absorptance that results in a higher operating temperature. Reaching higher specific powers by reducing the blanket weight through using thinner cover glasses and thinner cells (2 mil) should reach maturity in the near term. Although these technologies will not be available without high risk by the mid-1990s, they promise a bright future in solar array development. Table 28 gives a comparison of various solar cell designs.

TABLE 28. - BLANKET WEIGHT COMPARISONS FOR VARIOUS SOLAR CELL DESIGNS

	Present		Near Term			Far Term		
Cell description	Si 2 mil	Si 8 mil	Si 2 mil	Vertical Jcn	GaAs 7 mil	GaAs 2 mil	MBG 3 mil	3 Jcn Cascade
Cell efficiency, (% at 25°C	10.2	14.9	14	13 5	17-18	18	22	30
Operating temp, °C	78	73	75	84	65	68	66	71
kg/m^2	0.303	0 738	0.303	0.357	1.060	0.411	0 523	0.840
W/m^2, BOL	86	131	117	113	135	201	244	335
W/kg, BOL	284	116	385	312	252	493	375	399
Radiation degradation 1×10^{15} equiv, 1 MeV e/cm^2	29%	27%	22%	16%	20%	20%	20%	20%

3.4.5.2 Solar Array System Considerations

Some key issues that must be considered in designing a power generation system are mission duration, the existing and near-term technology, weight or system mass, radiation tolerance, size and interaction with spacecraft control, and orbital altitude. For solar photovoltaics, altitude is a key system driver. Altitude affects radiation level and thus the degradation of the solar cell performance. Altitude also determines the length of eclipse periods and thus determines the amount of power expended during occulation which must be made up during sunlight. As a point of departure, transmitter operation was constrained to be off during eclipse periods to avoid the large battery and solar array size increases. The seven year life requirement will be difficult to meet at some altitudes because high radiation levels will require large amounts of shielding weight to be added.

Several solar array configurations are possible. The solar cells may be arranged on panels either in a rigid foldout planar configuration or in a flexible rollout blanket configuration. The blanket configuration uses kapton as the flexible substrate and in addition to achieving high packaging densities, it is relatively lightweight. The cells may also be mounted to the spacecraft body. For gallium arsenide cells a concentrator system may be used, although development of this system is still required and concentrator reflector life is questionable for a seven year mission. The effect of orbital altitude on system weight for a silicon blanket array is shown in Figure 54. Because the transmitter was constrained to be off during eclipse, the effects of increased solar array size to account for power lost during eclipse was not included in the figure. As can be seen a severe weight penalty will be incurred with operation in the Van Allen belt.

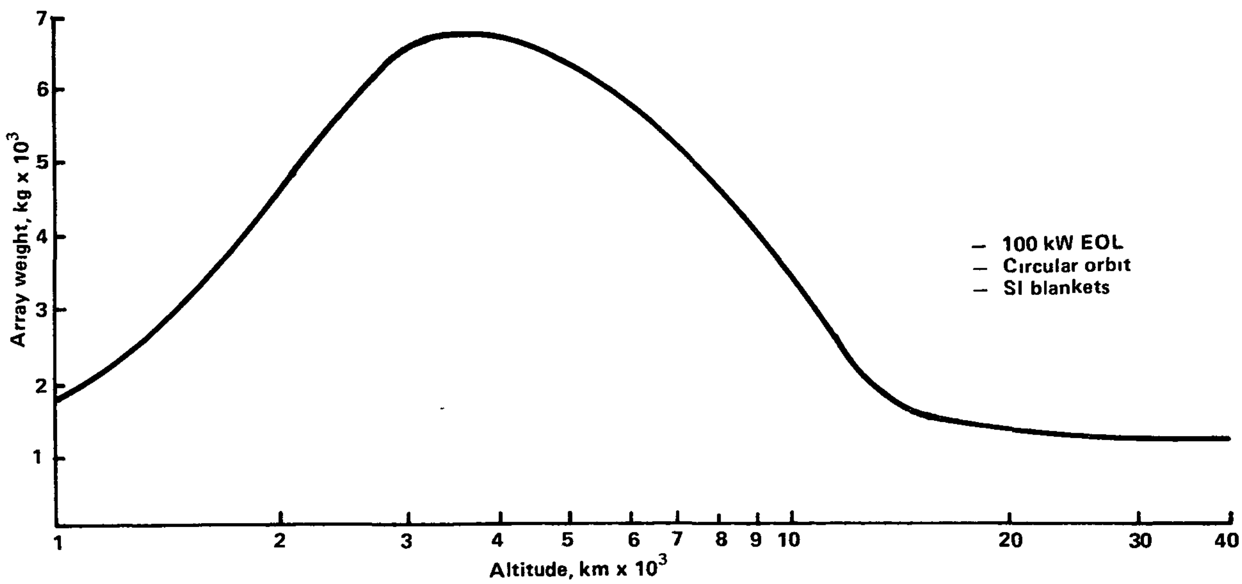

Figure 54. - Solar array weight versus altitude.

To fully use the power generation capability of the array, it must be Sun pointing. The technology now exists for solar arrays to track the Sun and maintain the array plane normal to the solar flux within $\pm 2°$. For a large power system it is impractical to design an array that does not track the Sun because the array size is proportional to the cosine of the solar incidence angle. A concentrator system also requires Sun tracking to direct the solar energy into the reflector. The disadvantage to a Sun-pointing system is that power transfer from solar array to the distribution bus is more difficult, especially at high voltage.

The technology with the lowest risk is the photovoltaic blanket planner array, utilizing silicon solar cells. If space station uses this technology as planned, then much of the engineering development effort for a 100 kW power system will be complete, making this system even more attractive.

3.4.5.3 Nuclear Reactor Source

The only space nuclear reactor power sources being developed is the SP-100 100 kW power plant. Table 29 highlights the features of one design concept. The power system described is a spacecraft power system that converts the thermal output of a nuclear reactor into electrical power for use by a spacecraft system. The normal output of the power system is 100 kW. The power system is compatible with launch in the STS. The power system consists of a reactor-converter subsystem, a heat transport subsystem, a power conditioning and control subsystem, and a structure subsystem. The nuclear-converter subsystem converts the fuel to heat energy by means of nuclear fission. The heat is converted directly to electrical power using in-core thermionic converters. Heat is removed from the reactor by the heat transport subsystem and transported to the radiator which rejects it to space. The raw electrical output of the nuclear-converter subsystem is regulated by the power conditioning and control subsystem and distributed to the spacecraft interface.

TABLE 29. - SP-100 DESCRIPTION

Type	In-core thermionics
Output	10 kW at 100 Vdc
Mass	300 kg
Dimensions	3.0 m diameter x 5.76 m length
Specific energy	33 7 W/kg*, 5 41 kW/m^3
Growth	Scales up to more favorable energy density cannot be scaled down

*Exclusive of mounting boom and distribution cabling

Thermionic conversion refers to a physical mechanism for the direct conversion of heat into electricity. A thermionic heat-to-electricity converter is an engine in which electrons are the thermodynamic working field. Electrons are emitted from a hot metallic surface (the emitter) maintained by nuclear or other heat sources at a temperature T_{em}, and absorbed on a second, parallel surface (the collector). The voltage developed between the two surfaces causes electrons to flow through an intervening electrical impedance, delivering electrical power to a load. The electrons also transport heat from one surface to the other. The collecting surface is held at the desired temperature, T_{em}, by thermal contact with a coolant. The temperatures, T_{em} and T_c, are approximately 1700 K and 1000 K respectively. Currently, there is widespread debate about whether the thermionic converter can meet a seven year mission at these high temperatures. The collector cylinder is contained within an electrical insulating sheath, which is in turn contained within a metallic sheath in contact with the liquid-metal coolant.

The SP-100 power system is projected for a demonstration flight by early 1990. For power levels above 100 kW and for orbits in high radiation belts the SP-100 system may be the only viable candidate for a mid-1990 mission. The advantages and disadvantages of this system are summarized in Table 30. There is still high development risk for this technology, making this system at present less attractive than solar photovoltaics. If development proceeds as planned however, this system may become a high power generation source of the future.

TABLE 30. - SP-100 POWER GENERATION TRADEOFF

Plus	Minus
– Produces power continuously	Nonproven technology-high development risk
– Not as susceptible to Van Allen belt	Nuclear thermal source
– Scales up readily	Does not scale down below 100 kW
– Low area and drag	Safety considerations–places lower limit
– Potential first flight 1990-1992	Onorbit altitude

3.4.6 Power Distribution Technology

Power distribution involves the transfer of power from the power source to the user loads. This entails selecting the type of cabling, the voltage level and the voltage frequency (ac vs dc), and the type of regulation and switching required. The voltage regulation or power conditioning is highly dependent on the user loads and their requirements and should be defined later in a detailed system design. The power conditioning technology exists now to be compatible with the power system selected. The issues that need to be addressed are cabling weight and type, voltage level, voltage frequency, and power transfer from solar array to bus.

3.4.6.1 Cabling

The cabling weight is a significant portion of the power system weight. For constant power loss, cabling weight increases with the square of the length and decreases with the square of the voltage. This relationship is not necessarily valid however, for shorter cable runs where the cable weight may be limited by the current carrying capabilities of the conductor. As can be seen, higher voltage levels can significantly reduce the cabling weight.

Several conductor types should be available in the near and far term. Besides the traditional copper conductors, aluminum conductors should be ready for near-term applications and in the far term, intercalated carbon fibers and sodium conductors should be available. These conductor types will be discussed further in Section 4.2.

3.4.6.2 Voltage Level

It was mentioned previously that high voltage levels can significantly reduce cabling weight. For large space power systems it is impractical to use low voltage levels. For example, a 100 kW system operating at 28 Vdc would produce currents in excess of 3500 amps. With these large currents, switch and semiconductor losses and the resultant heat increase, the efficiency of the system goes down, and a larger and more expensive energy source is required. Conversely, serious problems arise such as plasma charging and corona discharge if the voltage is too high. Another factor requiring careful consideration is the paschen breakdown voltage of various gaseous elements that may surround the spacecraft cabling and electrical components. The potential for breakdown exists as gaseous pressures are reduced during the spacecraft launch and ascent phase. Outgassing from various spacecraft materials may cause the potential breakdown conditions to exist long after the spacecraft has been placed in its orbit. Minimum breakdown voltages can occur at 300 V. In consideration of this, a safe upper limit of 250 Vdc should be selected. This is also below the voltage threshold where plasma arc discharging and power losses due to plasma charging are a factor. The plasma effects are primarily seen in low-Earth orbit, where the electron densities are relatively high. Figure 55 depicts the array cabling mass as a percent of total power system mass vs array voltage. This figure shows that for a 100 kW system, small weight savings is realized in increasing array voltage from 200 to 250 Vdc. Thus, to provide an added margin of safety between the Paschen breakdown voltage, a buss voltage of 200 Vdc would be optimum.

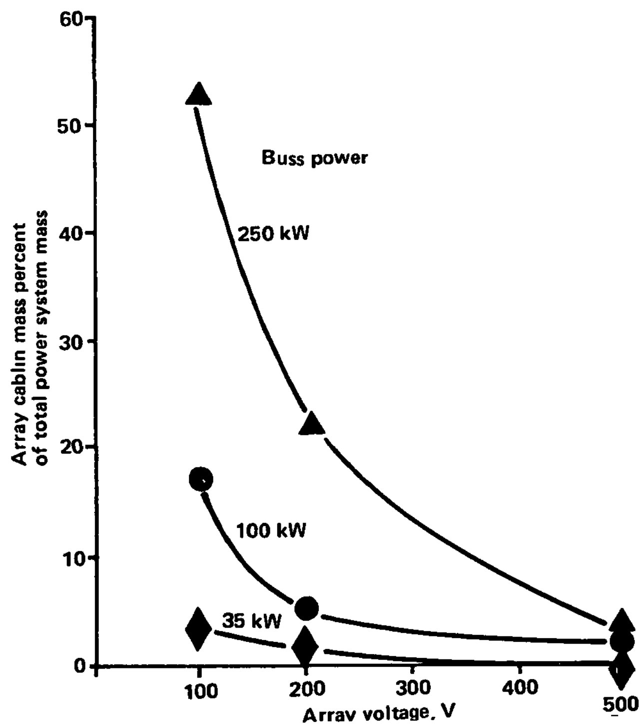

Figure 55. - Cabling mass versus array voltage for three buss power levels.

3.4.7 Energy Storage Technology

Because the transmitters will be off during eclipse for L-, VHF-, and HF-bands, the function of the energy storage portion of the power system will be to maintain only critical power on the spacecraft during deployment operations and during eclipse or occulation. The critical power load (primarily housekeeping loads) during eclipse has been assumed to be 500 W. This greatly simplifies the selection of an energy storage system.

Options available for energy storage devices include the traditional methods of nickel cadmium and nickel hydrogen batteries, and more creative methods such as regenerative fuel cells (RFC). To support housekeeping loads only, batteries are most effective. Regenerative fuel cells may be the only effective system if transmission is required during eclipse periods and power levels greater than 10 kW are needed. Nickel hydrogen batteries have performed very successfully on missions in GEO orbit and development programs are under way to qualify them for LEO applications. Nickel hydrogen batteries should be available at low risk for all applications by the early 1990's.

3.4.8 Power Control Technology

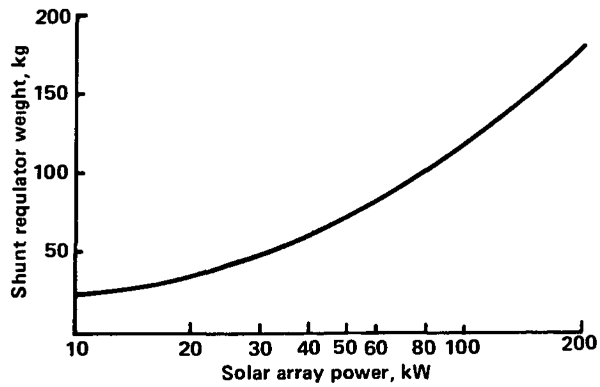

Figure 56. - Shunt regulator weight versus solar array power.

The objective of power control is to regulate voltage variations caused by source and load changes, and to dissipate excess power to prevent damage to the loads and electronic parts. The system must be capable of dissipating all but critical load power. Conventional designs use a shunt regulator or shunt resistor bank to dissipate the excess power. Figure 56 displays shunt regulator weight vs solar array power. the relationship is not linear because as the array power increases, increasingly large portions of the solar array can be controlled by single shunt regulator elements. The type of shunt regulator considered is an electronic regulator that shorts out solar array strings so that the power is never generated. Shorting the array strings will not harm the solar cells because short circuit current for a solar cell is typically only 5% above operating current. A resistor bank would be used in conjunction with the regulator to dissipate small amounts of power or fine tune the regulation. The design challenge is developing switching devices capable of handling large currents at high voltage. This system should be scaleable to 100 kW power systems at low risk.

3.4.9 Attitude Control, Stationkeeping and Maneuvering Technology

The attitude control subsystem (ACS) and auxiliary propulsion subsystem (APS) are considered together throughout the study, primarily because attitude control is performed by the APS system for most satellite concepts of interest. The requirement for multiple satellite and antenna beam steering in HF-, VHF-, and L-bands leads to a requirement for some degree of autonomous attitude control and orbit maintenance. Since adequate electrical power will be available during periods of no or lower RF transmission, electric propulsion can be considered as a viable alternative to chemical propulsion.

The basic elements of the ACS for HF- and VHF-bands were shown previously. Sun sensors, star trackers, and rate gyro units are all in use and space qualified. The onboard control electronics represents a new design for each satellite application, but the technology is well understood and the required components available. To meet the lifetime requirement for VOA DVBS, the designs should use high levels of redundancy and cross-strapping to enhance reliability, fault-tolerant operation modes, and distributed ACS data processing.

Control moment gyros (CMG) are used for attitude control on most satellite systems. Their technology is well understood, but for large antenna systems such as required for HF- and VHF-bands, they may not be desireable. Instead, electric thrusters can provide sufficient torque to counteract environmentally-induced torques. The use of small electric thrusters is possible because of the large moment arms with the large antenna structure.

The size of thrusters that can be used with the HF and VHF antennas is limited by the small allowable loads for these large deployable antennas. Thus, small thruster technology was surveyed to identify candidates for VOA DVBS. Table 31 contains a summary of small thrusters that might be applicable. Also shown is the status of each. Table 32 contains data for a pulsed plasma thruster (PPT) that might be used. Table 33 summarizes a mercury-ion thruster that has operating characteristics similar to the PPT. For geostationary orbits, chemical systems for APS and CMGs or reaction wheels for ACS have been used for some time. For VOA satellites in geostationary orbit, the ACS and APS technology is almost off the shelf (OTS). Figure 57 shows a block diagram representing a typical chemical APS.

TABLE 31. - MICROTHRUSTER CANDIDATES

System	Thrust	I_{sp}
Chemical		
Inert gas	10^{-4} to 1.0 N	35 to 275 s
Vaporizing liquid	10^{-5} to 0.05	50 to 100
Subliming solids	10^{-4} to 10^{-2}	40 to 80
Hydrazine direct catalyst	0 05 to 1000	100 to 225
Bipropellant (storable)	0 05 to 10^{4}	170 to 320
Electric		
Resistojet	0.01 to 5 0	175 to 860
Electrolysis	10^{-4} to 5.0	100 to 350
Pulsed plasma	10^{-6} to 10^{-3}	1000 to 5000
Ion (mercury)	10^{-3} to 0.5	2000 to 9000
Ion (noble gas)*	2.1×10^{-3}	5500 to 6400
MPD*	2.3×10^{-2} to $3 2 \times 10^{-2}$	2000 to 9000
Mass driver*	10^{-5} to 10	10^{4} to 5×10^{4}

*Not flight qualified.

C-2

TABLE 32 - PULSED PLASMA THRUSTER DATA

Manufacturer	Fairchild Republic Company
Status	
– Qualified	Yes
– Flown	LES 6, 8, & 9, TIP 2 & 3, NOVA
Life	
– Total impulse	320 000 N-s
– Total prop.	15 kg
Steady-state vacuum-specific impulse	2200 s
Propellant	Teflon
Power	170 W/milli-lb
Weight	23 kg

TABLE 33. - THRUSTER, GIMBAL, AND BEAM SHIELD UNIT

Thruster converts electrical power & propellant into thrust	
Thrust level	4.98×10^{-3} N
Specific impulse	2700 s
Weight	3 88 kg
Size	
– Beam diameter	8 cm
– Gimbal adapter diameter	7.6 cm
– External diameter	17 cm
– Length	22.6 cm
Power to thruster	125 W
Electrical efficiency	72%
Propellant efficiency	77%
Total efficiency	55%
Beam current	72 mA
Net acceleration voltage	1208 V
Thermal dissipation	35 W
Propellant flow rate	0 7 g/h
Temperature range	-20 to 80 °C
Design life	20,000 h 10,000 off/on cycles
Gimbal unit	
– Deflection in any azimuth	10°
– Motor steps per degree deflection	2421
– Time from 10 to -10-deg deflection	120 s
– Motor drive power (max)	7 W
– Mass*	15.0 kg
– Size	
Base	14 41 x 15.15 cm
Height	13 48 cm

*Includes mass of internal propellant feedline and manifold.

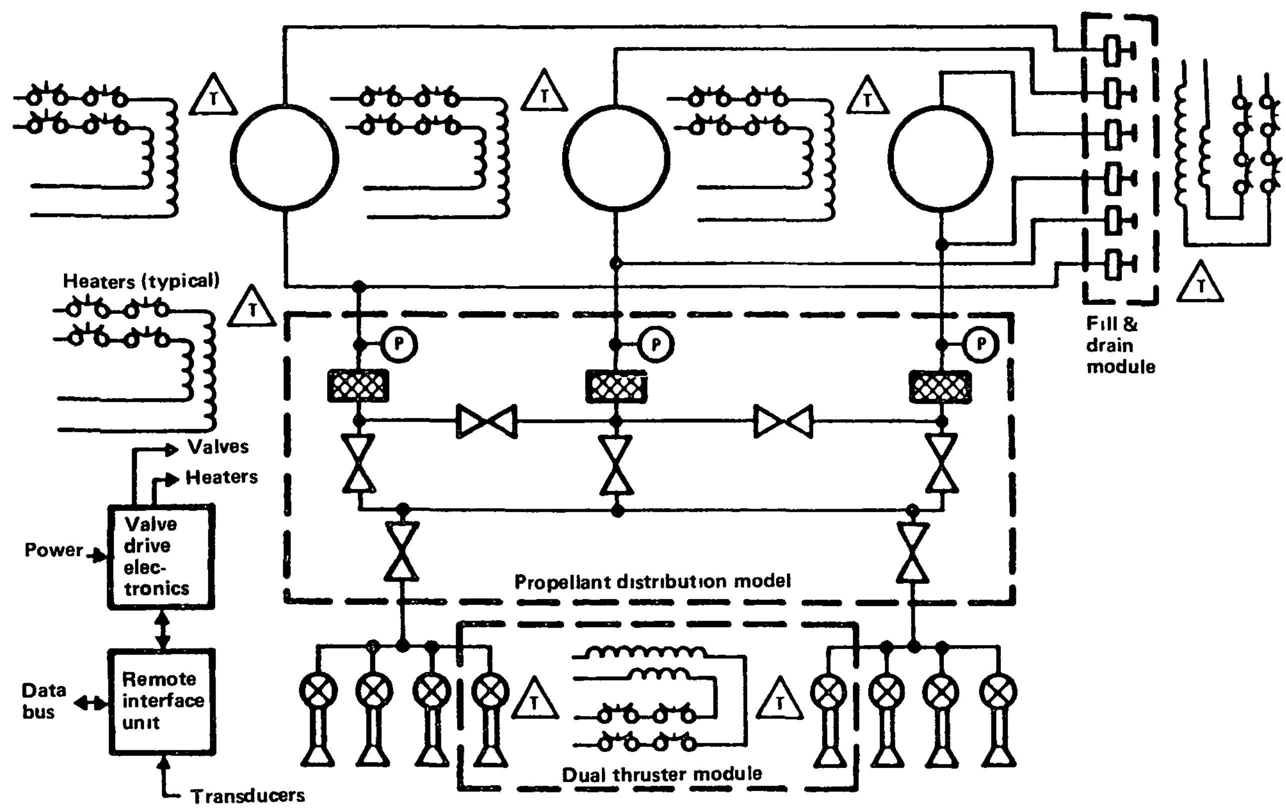

Figure 57. - Chemical APS block diagram.

The TT&C subsystem must provide the information to permit continual monitoring of satellite position and attitude. A turnaround ranging system is a common approach. This approach has the satellite instantly return an uplink ranging waveform to the transmitting Earth station. A functional block diagram of such a system is shown in Figure 58. The satellite range from the Earth station is computed as a function of the time delay between transmission and reception of the reflected waveform. The range computation is periodically repeated and the position compared to the desired orbit position to determine corrective maneuvers to maintain the orbit. This technology is in use and applicable to VOA satellite.

The other functions of the TT&C subsystem are to receive commands and the program information from the Earth station, decode the information for use onboard, encode satellite subsystem data, and transmit satellite data to the Earth station. The allocated frequency bands shown previously include downlink bands. For VOA satellite, the Ku-band link will include the frequency allocations shown in Figure 59. LANDSAT-D Ku-band hardware can be used for the frequency bands. As shown, an uplink can contain information for up to 6-HF or VHF satellites for clustered systems. Each satellite is assigned a bandwidth of 80 MHz. Within the 80 MHz a 36-MHz bandwidth could be used for command data on the uplink. Program data could use a 40-MHz bandwidth with additional separation for multiple channels. Each channel requires 450 KHz for high quality music. The 80-MHz bandwidth with 36 MHz for command is currently used on Intelsat systems.

For nonclustered systems, each satellite can make use of the entire uplink or downlink. Thus, up to 11 channels, each with-40 MHz bandwidth, can be supported per satellite. More channels could be provided by decreasing the bandwidth of each.

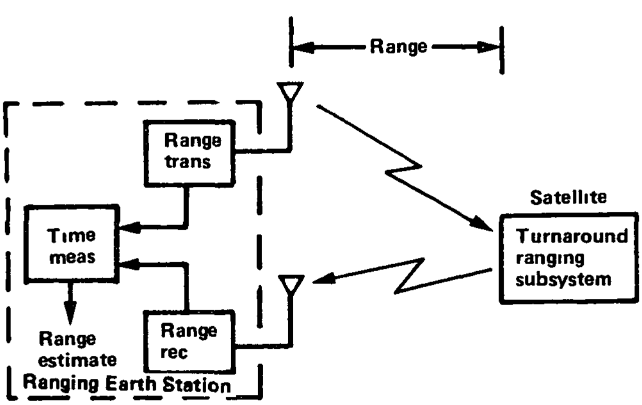

Figure 58. - Turnaround ranging system block diagram.

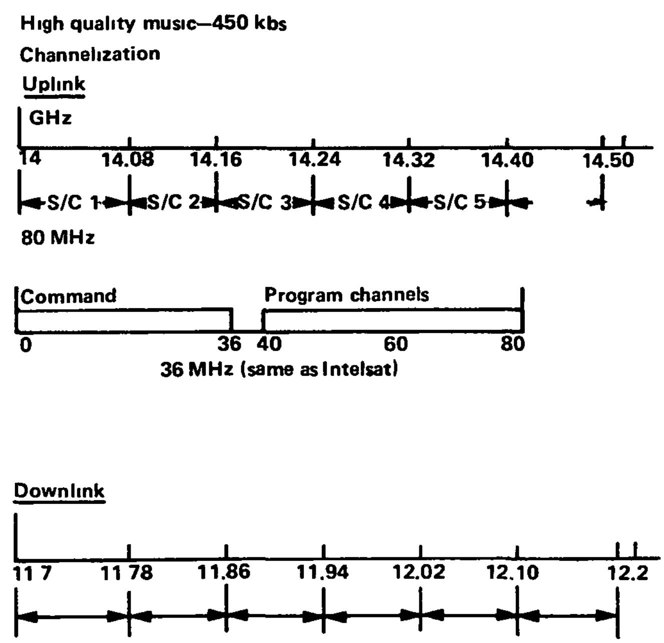

Figure 59. - Uplink/downlink frequency allocation

3.4.11 Thermal Control Technology

Thermal control of onboard components typically requires use of insulation, some means to transfer heat to and from components, and a heat dissipation method. Insulation may be a single, homogeneous material such as a low thermal conductivity foam or an evacuated, multilayer, insulation system in which each layer acts as a low conductance radiation shield and is separated by low conductance spacers. Electric resistance heaters are commonly used to maintain components above some minimum allowable temperature. Phase change material, such as electric salts, is used where components cycle on and off, storing heat when the component is on and providing heat when it is off. The result can be a relatively constant component temperature. Heat pipes have been used as a heat transfer method. Special radiative coatings have been developed with high emissivity and low absorptivity to enhance heat dissipation in space while decreasing heat absorbed from solar flux. Typical values of emissivity and absorptivity that can be expected are 0.8 and 0.2 respectively at satellite end of life. These technologies represent space-qualified and in-use methods and components.

Another heat transfer approach that may be useful for VOA HF and VHF satellites is a CPL that is in the development stage. The CPL differs from the heat pipe in that it is a continuous loop in which both the vapor and liquid flow in the same direction and that the condensor section can be made of smooth wall tubing. In a heat pipe, vapor and liquid flow in opposite directions, and the entire length of the pipe must be provided with a wick. Because of its smooth-walled condensor section, the CPL can have a distinct weight and volume advantage. Also, with a CPL system, a single evaporator (requiring a wick) can feed several condensor tubes configured in parallel.

A simplified schematic of the CPL is shown in Figure 60. The capillary pumping head is provided by the evaporator, which is an integral part of the cold plate. The high velocity vapor leaving the evaporator is at slightly higher pressure than the subcooled liquid entering the evaporator. The vapor head feeds the parallel condensor loops that correspond to an equal number of radiative surfaces.

In the condenser, the liquid moves in three successive flow regimes: as an annular film covering the wall of the tube near the entrance to the condenser; then, as a succession of slugs separated by condensing vapor bubbles, and finally, as an all-liquid phase. As indicated earlier, the fluid in the accumulator will be in a subcooled state during normal operation. Details not shown in Figure 60 may include provisions for trapping noncondensable gas in the system; and for priming the capillary in the evaporator by controlled flooding from the accumulator.

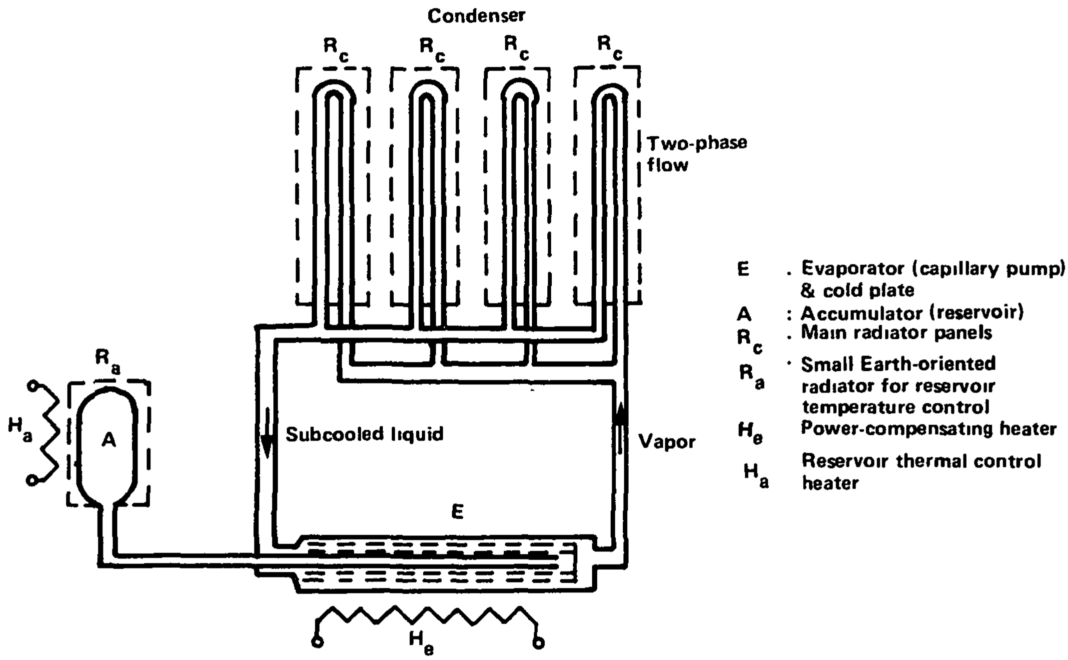

Figure 60. - Capillary pump loop flow schematic.

Top-level requirements established for the conceptual design of the SSPA thermal control subsystem are listed in Table 34. The similarity in their respective power levels indicates that a single basic thermal design can satisfy the requirements of both HF and VHF applications. The dual power levels imply the need for a large turndown ratio for the thermal control subsystem. The functional temperature limits of the transmitters were assumed on the basis of available data on similar equipment. Because of their effect on sizing and turndown ratio requirements, consideration must be given to the nominal as well as the limiting values of orbital environmental parameters (Earth IR, albedo, and solar radiation). The radiant interaction among transmitters and other elements of the system was evaluated on a preliminary basis. Although relatively small, these effects vary significantly with the location of the individual transmitters.

TABLE 34. - THERMAL CONTROL REQUIREMENTS

Heat dissipation, W
– 237 or 457 (HF-band) – 253 or 421 (VHF-band)
Equipment temperature limits
– -30 to + 55°C (functional)
Environmental criteria

– Orbital environments per NASA SP-8067 & TMX-64627
– Induced environments to include·
  – Mutual radiation blockage by transmitters
  – Transmissivity of tricot mesh
  – Radiation from solar panels

Orbital life· seven years
Constraints
– Weight-limited S/C – Prescribed stowed dimensions – No power during occultations

The 7-yr orbital life when translated into equivalent solar hours (ESH) can result in significant degradation of some thermal coatings at the end of the mission. The solar absorptivity of silvered teflon, for example, could increase from 0.07 BOL to 0.23 EOL. A more stable coating, although with somewhat higher beginning solar absorptivity would be preferred for this application. The 7-yr life also imposes constraints on the selection of thermal control equipment, e.g, components with no moving parts are preferred. It is assumed at the present that there will be no power available for thermal control purposes during occultation. This represents a significant challenge for the thermal designer, considering the high heat dissipation requirements during Sun-on operational modes, and the relatively long occultation periods of some of the proposed orbits.

3.4.12 Equipment Bay and Mechanisms Technology

The equipment bay is the structure that houses the electronics equipment for the various subsystems. The mechanisms for deployment, unstowing, and stowing of hardware other than the self-deployable antenna structure are also included in this subsystem.

The structure will use conventional aerospace materials and standard manufacturing processes. The mechanisms include those for radiator panel deployment, solar array stowage, uplink antenna boom stowage, and uplink antenna stowage. Any other gimbaled mechanisms should also be included here. Remote operation of all mechanisms is required. The technology for structure and mechanisms is state of the art, and no problems are anticipated to develop.

3.5 RECOMMENDED SATELLITE SYSTEM CONCEPTS

Upon completion of the orbit and coverage analysis, propagation analysis, payload capability analysis, and the technology survey, the preliminary results were evaluated to determine the most promising VOA systems that met or had the highest probability of meeting the VOA requirements. These candidates were then studied in depth to assess their performance, cost, schedule, risk, and technology development needs to achieve an operational system. Table 35 summarizes the parameters that were considered in the selection process. For HF and VHF systems, coverage, antenna diameter, and PFD were the critical parameters. For Ku- and L-band systems, coverage was the critical parameter since antenna size, power, and PDF did not drive the system design as they did for the HF and VHF systems.

TABLE 35. - RECOMMENDED SATELLITE CONCEPTS

Parameters considered in the selection process	
– Repeatable ground track – Coverage – Dwelltime – Power flux density	– Occultation – Radiation environment – Number satellites required – Antenna size
HF	
– Steerable array antenna – 6-h, 8-h, or 12-h orbits – Constellation of eight satellite clusters	
VHF	
– Steerable array antenna – 12-h or 24-h elliptical orbits – Constellation of eight satellite clusters (12 h) or four s ellite clusters (24-h el)	
L-band	
– Nonsteerable array antennas – Geostationary orbits (three orbital positions) – Eight satellites for -103 6 dBW/m^2 & three or five satellites for–116.1 dBW/m^2	
Ku-band	
– Rigid graphite/epoxy dishes or horns – Geostationary orbits (three orbital positions) – Three satellites–multiple antennas on each satellite	

3.5.1 Ku-Band System

The Ku-band system design is summarized in Table 35. Three satellites were placed in three geostationary slots to achieve the desired coverage of the 15 zones. Since each satellite covers multiple zones, multiple antennas (three or six) are placed on each satellite (see Fig. 124). Geostationary orbit was selected to insure 100% coverage. The low payload to geostationary orbit was not a factor since the Ku-band satellites are small with low power requirements.

3.5.2 L-Band System

The L-band system design is summarized in Table 35. Two power levels were evaluated, a higher $-103.6\ dBW/m^2$ requiring eight satellites and a lower $-116.1\ dBW/m^2$ requiring either three or five satellites. Geostationary orbit was selected because of the 100% coverage capability and the satellite masses did not exceed payload capabilities to geostationary orbit. Also, antenna sizes were not excessive in order to achieve the desired spot sizes. The L-band satellites (see Fig. 125) were placed in three orbital slots.

3.5.3 VHF-Band System

The VHF-band system design is summarized in Table 35. A constellation of eight satellites in a 12-hour circular orbit and a constellation of four satellites in a 24-hour elliptical orbit were selected. These two systems were selected for further study because both have their advantages and disadvantages. The 12-hour orbit produces higher PFD and smaller antenna diameters, but has lower coverage efficiency and requires more satellites (Fig. 61). The 24-hour orbit has higher coverage efficiency and fewer required satellites, but has lower PFD and requires a larger antenna (Fig. 61). The VHF-band satellite concept (see Fig. 126).

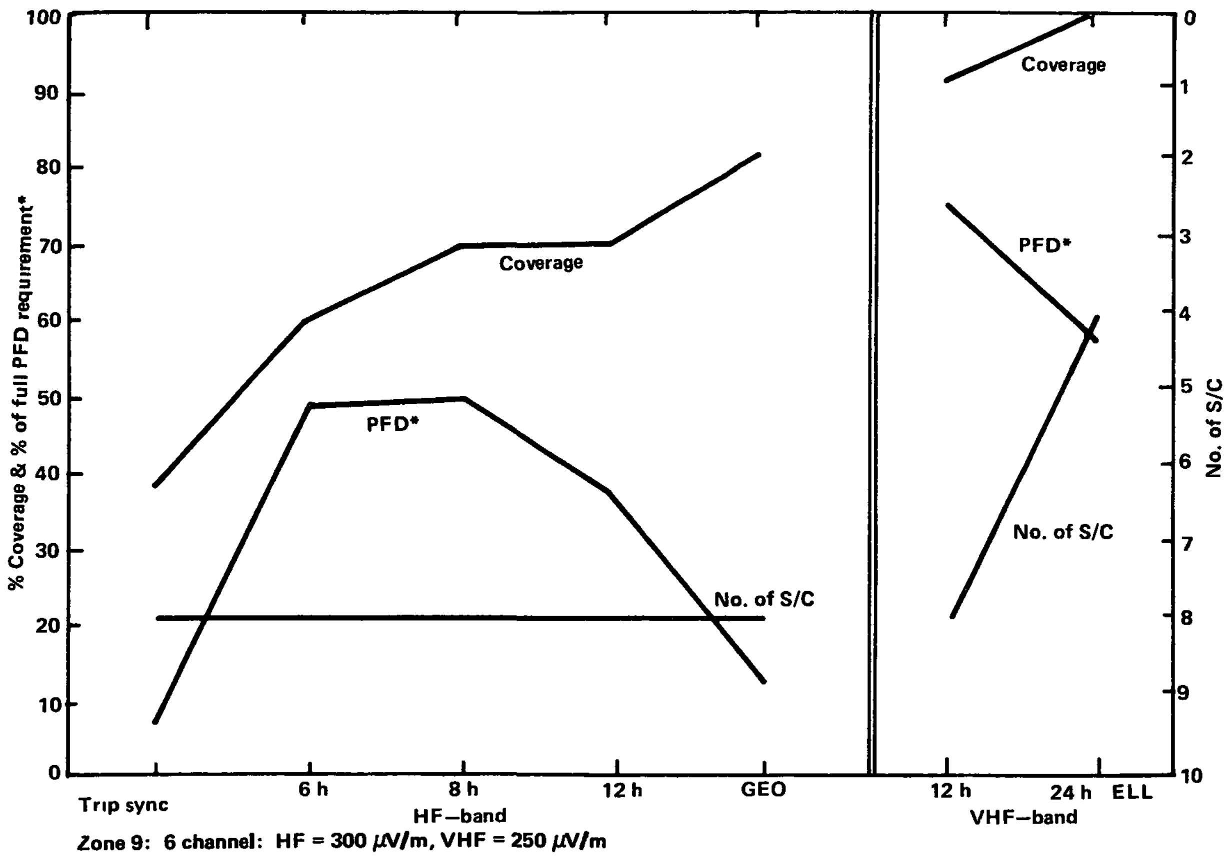

Figure 61. - Comparison of orbits for HF and VHF systems.

3.5.4 HF-Band System

The HF-band system design is summarized in Table 35. A constellation of eight orbital positions in the 6-, 8-, and 12-hour circular orbits were selected. Triply-synchronous elliptical and geostationary were eliminated because of their low PFD capability. The 8-hour orbit (see Fig. 61) was the most promising candidate. The 8-hour orbit had the same high coverage efficiency of the 12-hour orbit (higher than the 6-hour orbit). The 8-hour orbit also had the same high PFD as the 6-hour orbit (higher than the 12-hour orbit). Although the 8-hour orbit did look to be the best system, the 6- and 12-hour orbits were also evaluated to ensure completeness of the study. The HF-band satellite concept is shown in Figure 126.

4.0 SATELLITE CONCEPTS, SUBSYSTEM REQUIREMENTS DEFINITION, AND ANALYSIS

Each selected satellite system was evaluated to derive its specific system and subsystem requirements. The system requirements were derived from the mission (communications payload) requirements. A Martin Marietta Denver Aerospace computer program, Spacecraft Integrated Analysis Program (SCIAP) was used to model satellites and analyze their requirements. The SCIAP subsystems properties module (ref. the Appendix) was used to size and trade off candidate subsystem configurations. Methods were then developed to estimate weight and the cost of the satellites, ground control stations cost, and launch cost. Weight and cost estimates were made for each subsystem broken down by the following:

1) Communications subsystem:
    a) antenna system,
    b) transmitters,
    c) feeder link,
    d) signal processing,

2) Electrical power subsystem:
    a) power generation,
    b) power control,
    c) power distribution,
    d) energy storage,

3) Attitude control,

4) Auxiliary propulsion,

5) Telemetry, tracking and command,

6) Thermal control,

7) Equipment bay and mechanisms.

Following are discussions of the assumptions and methods used to estimate weight, volume, and cost of each of the subsystems. Also included are the assumptions and method used to estimate total program cost including satellite nonrecurring costs, recurring costs, launch costs, and ground control costs.

4.1 SUBSYSTEM WEIGHT AND VOLUME ESTIMATING METHOD

The first step in estimating subsystem weight and volume was to identify each subsystem's configuration and components. Then, an evaluation was made to determine if sufficient data was available to size components or subsystems as a function of a specific performance parameter (e.g, power, diameter, or orbit altitude). The TT&C subsystem and electronics associated with attitude control, communications uplink, and signal processing were felt to be relatively constant across the range of satellites to be analyzed. Thus, representative weights were obtained from components described in the NASA LaRC System Design and Cost Model (SDCM) data base. For all other subsystems and components, parametric relationships were used for weight and volume estimates.

4.1.1 Communications Subsystem Weight and Volume Estimating

The communications subsystem weight and volume estimates were made for four sets of components:

1) Antenna structure with reflector or array surfaces,

2) Transmitters,

3) Uplink/crosslink,

4) Signal processing.

The antenna structure and transmitters account for the majority of the weight and volume of the communication subsystem. Parametric sizing relationships were available or were developed to size the structures and solid state transmitters. Uplink/crosslink and signal processing estimates were based on state-of-the-art components described by the SDCM data base. Following are descriptions of the method used to estimate weight and volume of communication subsystem components. The overall flow of the method is shown in Figure 62.

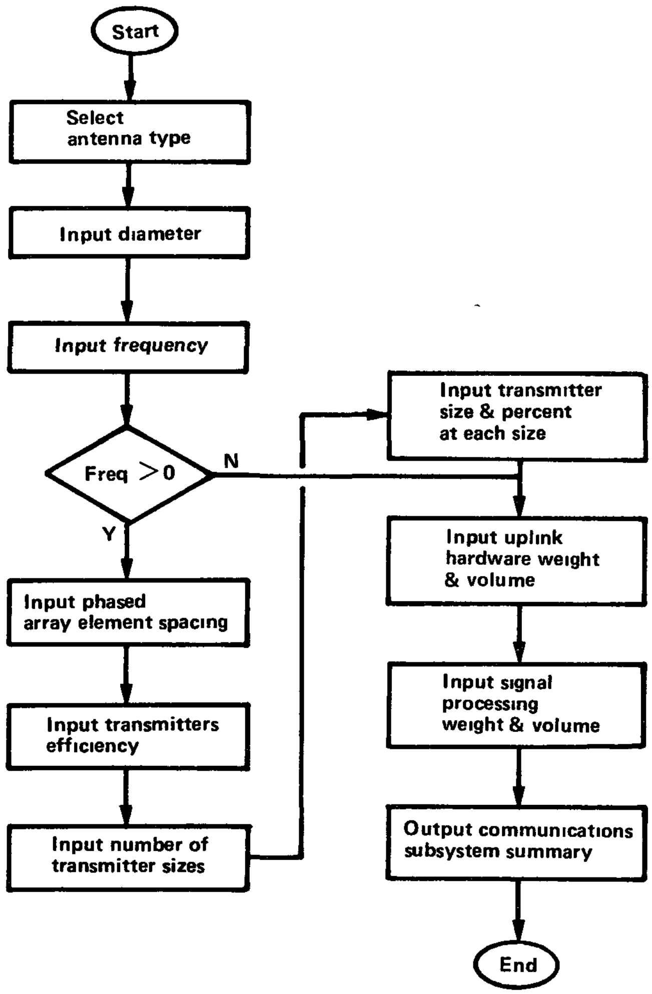

Figure 62. - Communications subsystem weight and volume estimating method.

4.1.1.1 Antenna Structure Weight and Volume Estimates

The antenna weight and volume estimates for HF and VHF satellites assumes a Martin Marietta box truss ring structure that has been studied previously (ref. 16). The transmitters and radiating elements are supported by a Kapton web and a tricot weave, gold plated molybdenum mesh ground plane as shown in Figure 63. The weight estimating relationship for this type of antenna structure is:

$$W_a = 1.26D + 0.02A + 0.266 \times L \times N \qquad (4\text{-}1)$$

where

W_a	=	antenna weight (kg)
D	=	antenna diameter (m)
A	=	antenna array area (m^2)
L	=	length of radiating dipole element (m)
N	=	number of radiating elements
0.02	=	area density of mesh ground plane (kg/m^2)
0.266	=	combined area density of Kapton support and elements (kg/m^2/element)

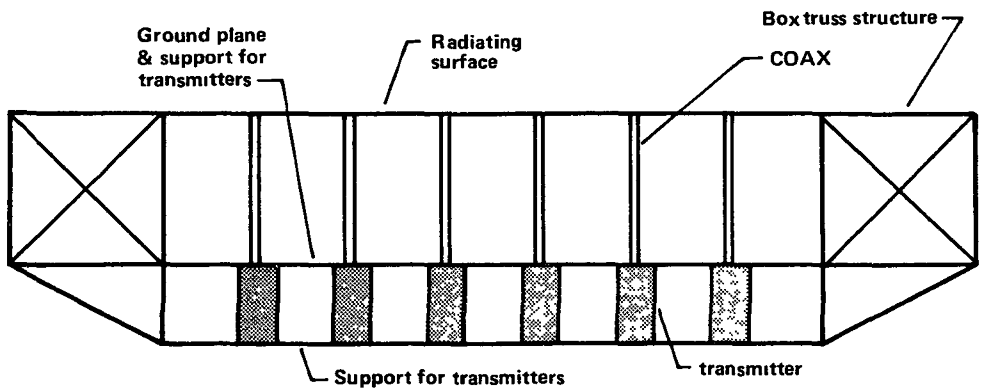

Figure 63. - Box truss ring array antenna configuration.

The first term in the weight equation estimates the weight of the ring structure. The second term gives the weight of the groundplane. The last term gives the weight of the two supporting surfaces of Kapton webs and the radiating elements, assuming a $\lambda/2$ spacing between radiating elements.

The volume of this antenna structure is given by: (4-2)

$$VOL = 14.19 \times \lambda/4$$

where

VOL	=	stowed structure volume (m^3)
$\lambda/4$	=	distance between ground plane and radiating surface (m)
14.19	=	STS payload allowable cross-sectional area (m^2)

The $\lambda/4$ term results from the design requirement for spacing between the ground plane and radiating surface for a front-fire array. A back-fire array would require $\lambda/2$ spacing, requiring a box truss depth of up to 6 meters for HF-band antennas. By using the front-fire design the maximum truss depth is kept at 3 meters, a value that provides good structural stiffness combined with effective packaging density.

For L-band antenna structures, fiberglass honeycomb panels were selected. The thickness of the panels is determined by launch constraints. The weight algorithm is:

$$W_a = 48A_p + 2(0.000381\ A_p)(1937.59) + 25 \quad (4\text{-}3)$$

where

A_p	=	panel area (m^2)
48	=	panel area mass density (kg/m^2)
0.000381	=	thickness of support and ground plane panels (m)
1937.59	=	fiberglass density (kg/m^3)
25	=	deployment mechanism and launch restraint (kg)

The L-band antenna volume algorithm is:

$$VOL = A_p \times \lambda/4$$

where

$\lambda/4$	=	distance between ground plane and radiating surface (m)

Ku-band antennas are assumed to be graphite epoxy parabolic reflectors similar to those used for LANDSAT-D. Each antenna has an estimated weight of 12.7 kg, including the feed electronics.

4.1.1.2 Transmitters Weight and Volume Estimating

As discussed in Section 3.4.4, data points were identified relating the weight and volume of solid state power amplifiers to their output power levels. Upon performing a regression analysis on the data points, two relationships were established, one for power levels less than 100 watts, and one for power greater than or equal to 100 watts. These relationships are:

$$W_t = 0.0136P_t + 9.41 \text{ (for } P_t \leq 100 \text{ watts)} \qquad (4\text{-}4)$$
$$W_t = 0.097P_t + 0.253 \text{ (for } P_t > 100 \text{ watts)}$$

where

W_t = transmitter weight (kg)

P_t = transmitter output power (watts)

The expressions used to estimate solid state transmitter volume is:

$$VOL = 1.47 \times 10^{-5}P_t + 0.0145 \quad \text{(for } P_t \quad 100 \text{ watts)} \qquad (4\text{-}5)$$
$$VOL = 1.441 \times 10^{-4}P_t + 0.003304 \text{ (for } P_t \quad 100 \text{ watts)}$$

where

VOL = transmitter volume (m^3)

For HF-, VHF-, and L-band satellites, many of the designs incorporate more than one size (power output) of transmitter. The use of different sizes permits beam steering and shaping to optimize PFD to each zone that might be served at different times by the satellite. This results in part-time operation of transmitters in a backed-off mode that has the added advantages of increased lifetime and reliability. The weight and volume estimates for these cases are based on the total number of transmitters required at each size.

For HF and VHF satellites, a nominal power per transmitter was identified assuming all transmitters operating simultaneously at the maximum available power consistent with power generator capability. Then, each required transmitter power level was used to determine a weight and volume per transmitter. Each weight and volume was then multiplied by the number of transmitters of each size to determine the total of transmitters' weight and volume. The power levels and percent of transmitters for HF and VHF satellites were 1.5 times the nominal power, 45%; 2.5 times the nominal power, 15%; 4.0 times the nominal power, 40%.

For L-band satellites, different transmitter sizes correspond to different array antennas. Thus, separate transmitter weight and volume estimates were made for each array antenna.

For the Ku-band, travelling wave tube amplifiers (TWTA) were assumed to have an average weight of 5.2 kg each. This value was obtained from the SDCM data base. Two TWTAs are assumed for each channel, resulting in a total transmitter weight of 10.4 kg times the number of satellite antennas. A volume of 0.05 m^3 was assumed for each pair of TWTAs.

4.1.1.3 Uplink/Crosslink Weight and Volume Estimating

The uplink/crosslink hardware weight was approximated by assuming the same weight as a LANDSAT-D Ku-band RF module (78 kg per ref. 17). The crosslink hardware estimated weight was 43 kg, since smaller TWTAs would be required and some of the original electronics would be used through switching and up/down conversion. Thus, the estimated weight for HF and VHF systems was 121 kg per satellite. The volume of the uplink/crosslink hardware was estimated at 0.5 m^3, much of which is from the reflectors, gimbal drives, and antenna booms.

For the Ku- and L-band satellites, the crosslink hardware would not be required. Also, the gimbaling hardware would not be required for the uplink system. The total weight estimate for these system's link hardware is reduced to 24 kg per satellite. The volume estimate is reduced by the same percentage to 0.2 m^3.

4.1.1.4 Signal Processing Weight Estimating

For HF and VHF satellites, the LANDSAT-D Ku-band wideband module weight and volume from ref. 18 are used (58 kg and 0.02 m^3 respectively). For Ku- and L-band, the signal processing weight is assumed at 9.5 kg per downlink antenna. The volume estimate is 0.01 m^3 per antenna.

4.1.2 Electrical Power Subsystem Weight and Volume Estimating

This section describes the space electrical power system from which power system sizing equations were derived. From load requirements for Sun and eclipse periods and distribution losses, required source power can be calculated. Knowing source power, the size of the solar array can be calculated. Then, the weights and volumes of the power system components can be calculated. Figure 64 summarizes the calculation flow of the power generation and distribution model. The various calculations are supported by a data base of performance factors, weight densities, and specific densities. The calculated power system weights and volumes reflect the state of technology expected by the early 1990 time period.

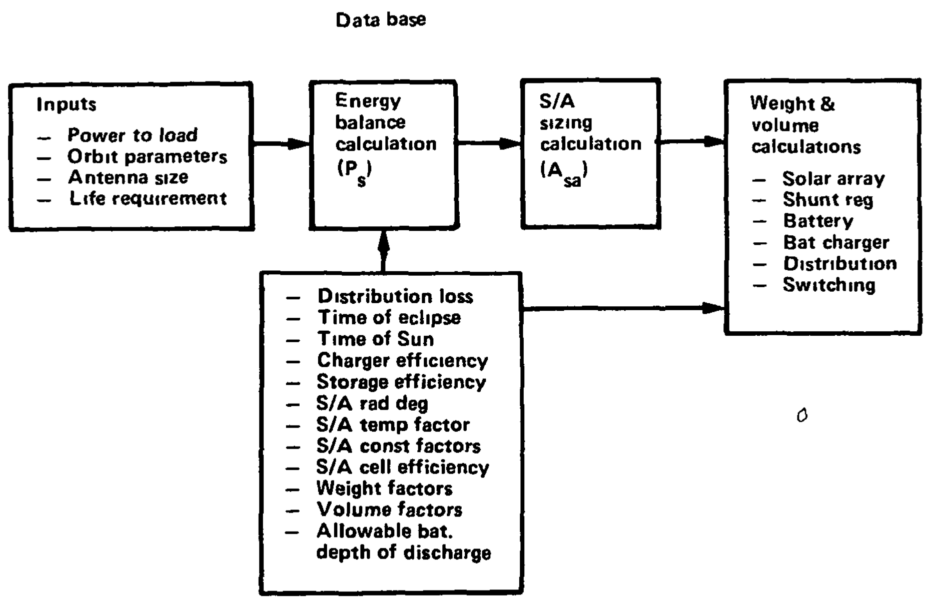

Figure 64. - Power distribution and generation calculation flow.

4.1.2.1 Energy Balance Equation

One simplification made in the power generation model is the assumption of a direct energy transfer system. This is a realistic assumption since the majority of spacecraft power systems use this form of energy transfer. This system implies a dc distribution network (accurate ac distribution models have yet to be developed) with an unregulated ($\pm$10%) bus and no conversion losses. Figure 65 illustrates a simplified direct energy transfer power system and also depicts the data base for the energy balance equation. Note that the distribution loss, N_r, is the sum of the wire (N_{ws} and N_{wr}) and diode (N_d) losses from source to bus and bus to load. Other energy transfer system approaches are possible; specifically, peak power tracking, a regulated bus, distributed power switching, and distributed inverters with ac distribution. These systems may have advantages over the system shown, however, the direct transfer system is viewed as optimum for modeling purposes.

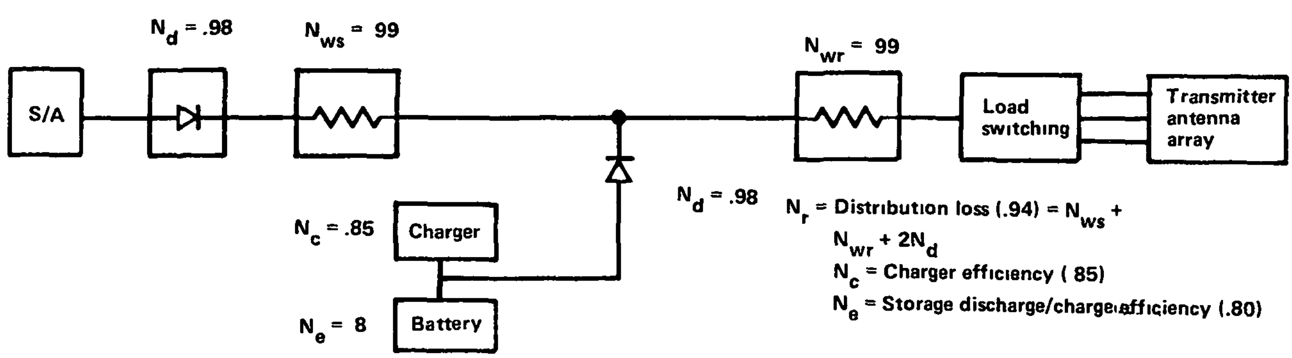

Figure 65. - Direct energy transfer system for unregulated (± 10%) bus.

The energy balance equation converts power required at the load into power required from the source. Because eclipse periods affect the size of solar-derived sources, the equation reflects the additional source power required to replenish any power taken from an energy storage device. If the load power, P_1 is the same during Sun and eclipse periods then the energy balance equation takes the form:

$$P_s = \frac{P_1}{N_r}\left[1 + \frac{T_e}{T_s}\left(\frac{1}{N_c N_e}\right)\right] \tag{4-6}$$

where

P_s = power from source

P_1 = load power

T_e = time of eclipse

T_s = time in Sun

N_r, N_c, and N_e are as defined in Figure 4.1-4

For non-Sun power generation (e.g, SP-100), $T_e = 0$. For those systems whose transmitters are assumed to be off during eclipse, the Sun and eclipse power levels will be different. In this case the appropriate balance equation is:

$$P_1 = \frac{P_{1_s}}{N_r} + \frac{P_{1_e}}{N_r} \quad \frac{T_e}{T_s} \quad \frac{1}{N_e N_c} \tag{4-7}$$

where

P_{1s} = Sun load power

P_{1e} = eclipse load power

Once P_s is known the solar array may be sized.

4.1.2.2 Solar Array Sizing Equation

The solar array sizing equation takes into account the major variables that affect solar array area. The variables dependent on altitude are solar cell radiation degradation (F_r), the effects of temperature on cell performance (F_t), and cover slide thickness weight factor (F_c). Figure 66 is a graph of these variables vs altitude. The figure shows how the radiation factor includes the benefits of increased cover slide thickness as the array passes through the Van Allen belt (centered about 4000 km). The weight of this increased shielding is reflected by F_c.

The variables (except solar cell efficiency) which are independent of altitude are multiplied together to give a solar array sizing constant, K. These variables are:

Array Orientation Factor	.98
Solar Cell Packaging Factor	.90
Cover Slide Transmissivity Loss	.97
Array Fabrication Loss	.98
Miscellaneous Loss Factor	.99
Solar Intensity	1353 W/m^2

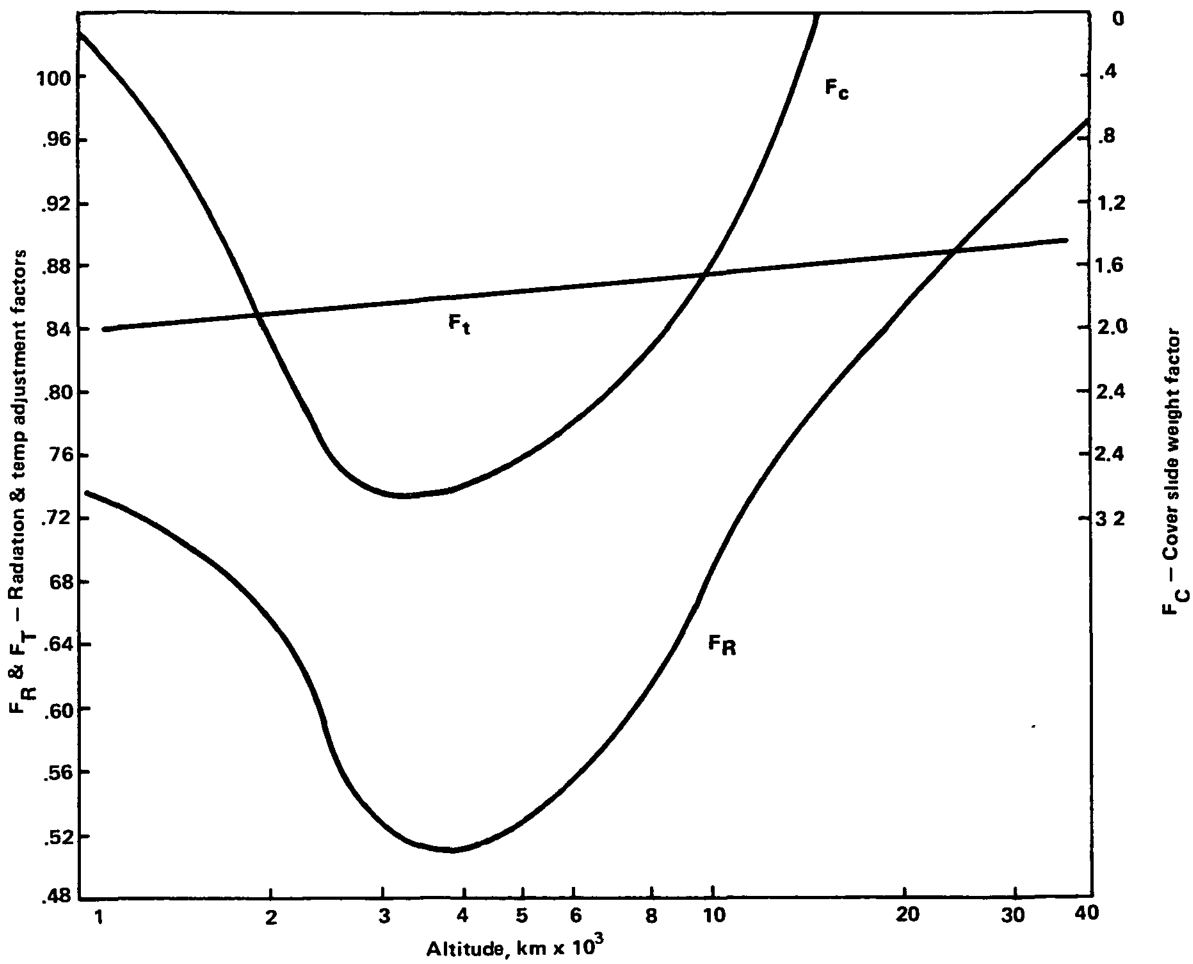

Figure 66. - Solar cell radiation, temperature, and cover slide factors, versus altitude.

The resulting value of K used to estimate area is 1123 W/m^2. Solar cell efficiency is the remaining variable. The efficiencies assumed in the model are 14% for 8-mil silicon and 18% for gallium arsenide. The array area sizing equation is:

$$A_{sa} = \frac{P_s}{(F_r)(F_t)(K)(N_i)} \tag{4-8}$$

where

P_s=power required from source (W)

A_{sa}=area of solar array (m^2)

F_r=radiation degradation factor

F_t=temperature adjustment factor

N_i=cell efficiency (14% or 18%)

K = sizing constant = 1123 W/m^2

4.1.2.3 Solar Array Weight and Volume Equations

Solar array weight and volume calculations rely heavily on the array panel weight and volume factors and the array structure factors shown in Table 36. Because the solar array can be such a large percentage of spacecraft weight, many studies and development programs have addressed means of reducing panel weight and volume. Table 36 reflects the results of these studies. Although the weight factors shown have not been proven by direct flight experience and attaining the factors may present design challenges, the factors are seen as being achievable in the early to mid-1990s. The weight and volume of the SP-100 reactor system is provided for comparison.

TABLE 36. - SOLAR ARRAY WEIGHT AND VOLUME ESTIMATING FACTORS

Source type	Panel wt factor	Structure wt factor	Panel volume factor	Reference
Si blanket (8 Mil)	0 75 kg/m^2	0.85 kg/m^2	0.008 m^3/m^2	SAFE Marshall's 25-kW array GE's 10-kW array Frusa DORA (frusa type) Space Station
Si panel	2.6	1.0	0 03	TRW FLTSATCOM TRW LRSA OTS Lewis Martin Marietta
Gallium arsenide panel	4.4	1 0	0.05	Lewis Martin Marietta
Body-mounted Si cells	2 6	0	0	
SP-100 in-core thermionic generator	3000 kg	58.2 m^3	100 kW net output EOL	

The solar array weight and volume estimating equations are given below.

$$W_{sa} = A_{sa}\,[\text{panel wt factor} + \text{structure wt factor} + \text{cover slide factor } (F_c)] \quad (4\text{-}9)$$

$$V_{sa} = A_{sa}\,(\text{panel volume factor})$$

where

W_{sa} = array weight

V_{sa} = array volume

A_{sa} = array area

F_c (determined from Fig. 66)

4.1.2.4 Battery Charger and Shunt Regulator Weight Equations

The battery charger and shunt regulator reflect state-of-the-art electrical design approaches. A battery charger weight factor was derived from current and near-term electronic power densities. Present power densities for low power are 61-122 W/kg and near-term power densities for high-power devices that employ active cooling are 244-305 W/kg. From these values, a realistic near-term power density of 175 W/kg was selected. This yields a charger weight factor of 5.7×10^{-3} kg/W.

The shunt regulator weight factor is shown in Figure 67. This factor reflects the material densities used in the construction of a shunt regulator. The weight factor is not linear because a sequential shunt regulator was assumed where increasingly large steps of array area can be shunted by a simple switch. The design challenge would be in the switching of large currents. The weight equations are:

$$W_{bc} = \left[P_1\,T_e / T_s\right] (5.7 \times 10^{-3}\ \text{kg/W}) \quad (4\text{-}10)$$

$$W_{sr} = P_1 \times F_{sr}$$

where

W_{bc} = battery charger weight (kg)

P_1 = load power (W)

T_e = eclipse time (hr)

T_s = time in Sun (hr)

W_{sr} = shunt regulator weight (kg)

F_{sr} is determined from Figure 4.1-6

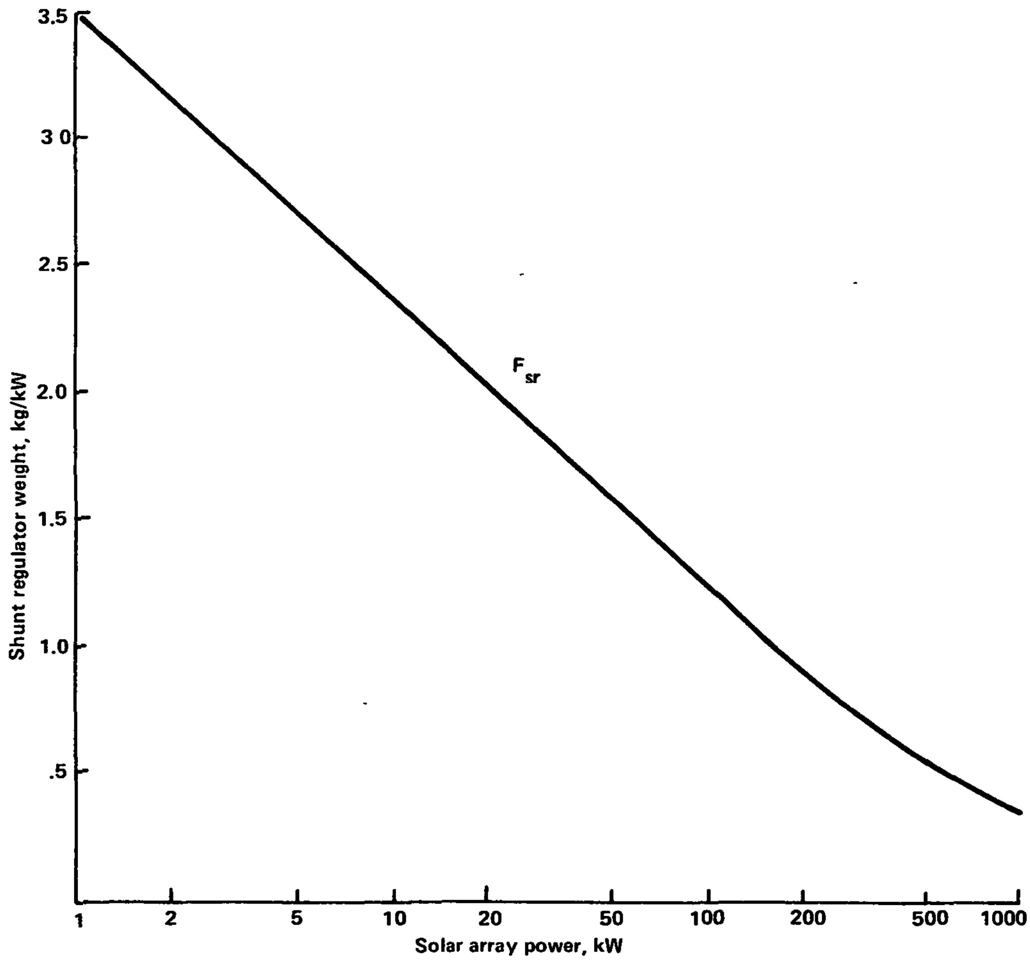

Figure 67. - Shunt regulator weight factor versus array power.

4.1.2.5 Battery Sizing Equations

Battery size is primarily dependent upon eclipse load power and orbital parameters. Orbital parameters affect the size in two ways. First, altitude and inclination affect the length (time) of eclipse (T_e) which determines battery discharge time. Second, the orbit period determines the number of discharge cycles per year. Battery allowable depth of discharge is also an important factor and depends upon the type of battery selected. Figure 68 displays the allowable depth of discharge (F_d) vs battery charge/discharge cycles for Nickel Cadmium and Nickel Hydrogen batteries.

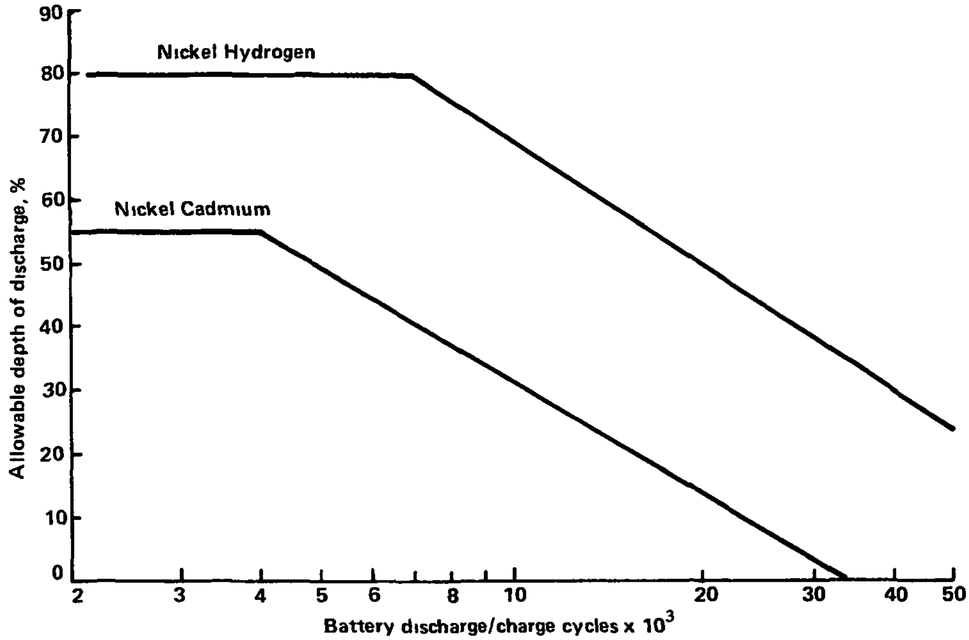

Figure 68. - Allowable depth of discharge versus charge/discharge cycles.

The remaining variables requiring definition to estimate battery weight and volume are energy densities and specific energies. Energy densities or battery weight factors that are achievable with low risk for nickel cadmium and nickel hydrogen batteries are 35 Wh/kg and 33 Wh/kg, respectively. Similarly, specific energies or battery volume factors achievable at low risk are 70 kWh/m^3 for nickel cadmium and 25 kWh/m^3 for nickel hydrogen. The battery sizing equations are given below.

$$W_b = \frac{(P_1)\ (T_e)}{(F_d)\ (F_{bw})} \qquad V_b = \frac{(P_1)\ (T_e)}{(F_d)\ F_{bv})} \qquad (4\text{-}11)$$

where

W_b = battery weight (kg)

F_d = allowable depth of discharge (from Figure 4.1.5)

F_{bw} = weight factor (35 Wh/kg for nickel cadmium or 33 Wh/kg for nickel hydrogen)

V_b = battery volume (m^3)

F_{bv} = volume factor (70 kWh/m^3 for nickel cadmium or 25 kWh/m^3 for nickel hydrogen)

4.1.2.6 Switching and Distribution Weight Equations

The equations for distribution weight are based on aluminum wire for power bus feeders. Handbook current ratings (70°C temperature rise) were used. These equations assume that cable weight is limited by the current rating of the wire rather than the allowable power loss. This is a valid assumption for shorter cable runs and may or may not be valid for longer cable runs, depending on the cost (in weight) of generating the additional power lost in the cable. Other assumptions made are 200 Vdc bus, hybrid electromechanical switches, and distribution losses of 3% from both source to bus and bus to load. The switching and distribution sizing equations are defined below.

Solar array to bus distribution weight:

$$W_d = (0.0243\ \text{kg/m}^2\text{-W})(A_{sa})(P_s) \qquad (4\text{-}12)$$

where

A_{sa} = solar array area (m^2)

P_s = power from source (W)

Bus to transmitters distribution weight:

$$W_d = (0.0243\ \text{kg/W-m})(L1 + L2/2)(P_1) \qquad (4\text{-}13)$$

where

L1, L2 = antenna dimensions (m)

P_1 = power to load (W)

Power switching equipment weight and volume:

$$W_{sw} = 2.18\ \text{kg} + (0.23\ \text{kg/kW})(P_1) \qquad (4\text{-}14)$$

$$VOL = \frac{W_{sw}}{1200\ \text{kg/m}^3}$$

For nongeostationary orbits, the assumed onboard equipment for attitude determination is summarized in Table 37. For geostationary orbits, the rate gyros could be eliminated as could most of the control electronics. Thus the ACS weight would be significantly reduced (13.1 kg).

TABLE 37. - ATTITUDE CONTROL

Component	Status	Number	Weight
Fine-Sun sensor	SOA/OTS	2	0.1 kg
Coarse-Sun sensor	SOA/OTS	2	0 2
Star tracker	SOA/OTS	2	12.8
Rate gyro	SOA/OTS	2	13.6
Electronics	SOA/NEW	1	26.0
Cabling/harness	SOA/NEW	1	5 0
Total			57 7 kg

The weight and volume of the APS was determined by running the SCIAP program. The inputs and outputs are summarized in Figure 69. When using SCIAP, the first step is to create a satellite mass and geometry model using an interactive model generator module. This data is then transferred automatically through the SCIAP data base to the mass properties module and then to the rigid body controls (RCD) Module. The RCD module computes forces and torques on the satellite as it travels around its orbit. The forces and torques are integrated to compute the total linear and rotational impulse per orbit. Allowable locations for thrusters are entered along with a nominal thrust per thruster. The RCD module then determines the optimum number of thrusters, thrusters' firing strategy, and the required propellant mass per orbit. The propellant per orbit is then extrapolated to the specified satellite lifetime. The RCD module permits definition of solar array orientation through the orbit, feathering of arrays in the occulted region of the orbit if so desired, specifying satellite orientation, and selecting from different density atmospheres. Effect of ACS limit cycle and maneuvers on propellant mass requirement are also included. The required propellant can be computed by assuming attitude control by either the APS thrusters or by momentum storage devices. In the latter case, the number of orbits between desaturation is an input and the propellant mass compulation includes propellant required for desaturation.

Following is an example of SCIAP run results for the HF-band triply-synchronous orbit. The RCD module inputs are summarized in Table 38. The altitude shown is the altitude at the start of the orbit. The orbit is actually defined later by interactive input of the orbit periapsis and apoapsis. Table 39 shows the satellite mass properties created by the model generator that were fed in through the data base. The results of the analysis are summarized in Table 40. Figures 70 and 71 show the total forces and torques on the satellite as a function of orbit anomaly angle.

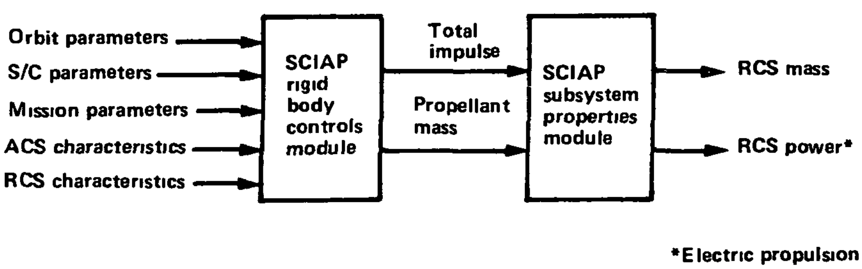

Figure 69. - Reaction control weight estimate.

TABLE 38. - RCD MODULE INPUT VARIABLES

HF SMALL S/C—TRIPLY SYNC ORBIT RUN NO. 2		
	RIGID BODY CONTROL DYNAMICS (RCD) INPUT	
5 21000E+5	1 H	– ORBIT ALTITUDE, m
2.00000	2 Inclin	– ORBIT INCLINATION, rad
0	3 PSIN	– ORBIT ASCENDING NODE, rad
0	4 SDAY	– NUMBER OF DAYS SINCE EQUINOX
7 00000	5 TFUEL	– TIME BETWEEN REFUELING, yr
21560 00000	6 ISP	– SPECIFIC IMPULSE, N s/kg
2.30000	7 CD	– AERODYNAMIC DRAG COEFFICIENT
2 00000	8 IE	– ORIENTATION FLAG (= .1 FOR INERTIAL OR = 2. FOR EARTH)
0	9 OPSI	– EULER ANGLES (3) DEFINING ORIENTATION OF SPACECRAFT FOR BOTH
0	10 OTHETA	INERTIAL AND EARTH, OPSI IS ROTATION ABOUT THE Z AXIS,
0	11 OPHI	OTHETA ABOUT THE NEW Y AXIS, OPHI ABOUT X, rad
1 00000E-05	12 WM3 (1)	– SPACECRAFT MANEUVER RATE REQUIREMENT X, Y, Z COMPONENTS
1 00000E-05	13 WM3 (2)	RESPECTIVELY, rad/s
1 00000E-05	14 WM3 (3)	
1.00000E-04	15 ALFAM3	– SPACECRAFT MANEUVER ACCELERATION REQUIREMENTS X, Y, Z
1 00000E-04	16 (2)	COMPONENTS RESPECTIVELY, rad/s^2
1.00000E-04	17 (3)	
0	18 NM	– NUMBER OF MANEUVERS PER ORBIT
2.00000E-04	19 E3 (1)	– INERTIAL ATTITUDE ACCURACY REQUIREMENT X, Y, Z COMPONENTS
2.00000E-04	20 E3 (2)	RESPECTIVELY, rad
2.00000E-04	21 E3 (3)	
0	22 UAS3 (1)	– CONTROL FLAG FOR ROLL CONTROL
0	23 UAS3 (2)	CONTROL FLAG FOR PITCH
0	24 UAS3 (3)	CONTROL FLAG FOR YAW
10.00000	25 BLANK	
10.00000	26 BLANK	
10.00000	27 BLANK	
0	28 KU	– CONTROL FLAG FOR TRANSIENT (1) OR STEADY STATE (0) ANALYSIS
10 00000	29 NORDES	– NUMBER OF ORBITS BETWEEN DESATURATIONS
10.00000	30 BLANK	
0	31 PLACS	– POWER REQUIREMENTS OF ACS EXCLUDING AMCD SPIN AXIS, W
5 00000E-04	32 LM (1)	– MINIMUM LINEAR INPULSE BIT WHEN CONTROLLING TORQUE,
5.00000E-04	33 LM (2)	X, Y, Z AXES RESPECTIVELY, N
5.00000E-04	34 LM (3)	
16.00000	35 NRCSGP	– NUMBER OF THRUSTER GRIDPOINTS (= NUMBER OF ROWS IN RCSMAT)

TABLE 39. - SATELLITE MODEL MASS PROPERTIES

HF small S/C—triply-sync orbit run no. 2		
	RCD category 2 input items	
4280.00000	1 TWRM	– Total weight of the spacecraft excluding RCD, kg
-214.49000	2 BXM	– Spacecraft center of mass for TWRM X, Y, Z coordinates
15.88800	3 BYM	Respectively, cm
15.42100	4 BZM	
7.01640E+05	5 XXM	– Moment of inertia XX for TWRM, kg-m^2
1.65308E+05	6 YYM	– Moment of inertia YY for TWRM, kg-m^2
8.65812E+05	7 ZZM	– Moment of inertia ZZ for TWRM, kg-m^2
7721.50000	8 PXYM	– Product of inertia XY for TWRM, kg-m^2
-6005.60000	9 PXZM	– Product of inertia XZ for TWRM, kg-m^2
444.86000	10 PYZM	– Product of inertia YZ for TWRM, kg-m^2
1.00000	11 KALKTK	– Prop tank M & A flag (> 0 user def, = 0 prop, < 0 auto)
0	12 NOPROP	– Number of propellant masses
0	13 NMAMCD	– Number of AMDC masses
0	14 ANBAYS	– Analysis, number of bays
0	15 NOGPAR	– Number of gridpoints in analysis (= no of rows in GP area)

TABLE 40. - RCD MODULE OUTPUT SUMMARY

HF small S/C—triply-synchronous orbit, run no. 2	
Linear impulse per orbit to orbit keep, Newton-s =	2.115E+02
— X component in spacecraft coordinates =	-2.071E+02
— Y component in spacecraft coordinates =	-9.519E-12
— Z component in spacecraft coordinates =	-4.381E+00
Linear impulse needed per orbit for desaturation, N-s =	—
Linear impulse for life to orbit keep, N-s =	.432E+07
— X component in spacecraft coordinates =*	- .423E+07
— Y component in spacecraft coordinates =	- .195E-06
— Z component in spacecraft coordinates =	- .896E+05
Linear impulse needed per orbit for desaturation, N-s =	—
Linear impulse for life for attitude control, N-s =	.290E+06
Linear impulse per orbit for attitude control, RCS only, N-s =	1.418E+01
Linear impulse per orbit for limit cycle, RCS only, N-s =	2.995E-04
Linear impulse per orbit for maneuvering, RCS only, N-s =	—
Total linear impulse per orbit for RCS only, N-s =	1.418E+01
Orbit radius, m =	6.899E+06
Orbit velocity, m/s =	8.821E+03
Orbit period, s =	1.080E+04
Propellant mass fix ratio =	1.000E+00
Propellant mass, kg =	2.141E+02
Spacecraft mass less propellant, kg =	4.280E+03
— X distance to center of mass in frame 4, m =	-2.145E+00
— Y distance to center of mass in frame 4, m =	1.589E-01
— Z distance to center of mass in frame 4, m =	1.542E-01

Mass moment of inertia, kg-m^2			Distance from center of gravity to center of pressure, m		
7.016E+05	-7.721E+03	6.006E+06	—	-8 745E-02	2.744E-01
-7.721E+03	1 653E+05	-4 449E+02	1.807E+00	—	3.208E-01
6.006E+03	-4.449E+02	8 658E+05	2.130E+00	-1.578E-01	—
Torque resulting from RCSMAT assuming all thrusters fire at nominal value					
				Inverse matrix	
2.288E-01	—	—	—	—	—
—	1.934E-01	—	—	—	—
—	—	2.288E-01	—	—	—
Eff radii for application of torque			Spacecraft projected areas		
6 356E+00	1.075E+01	1.271E+01	4.200E+01	4.000E+01	8.922E+02

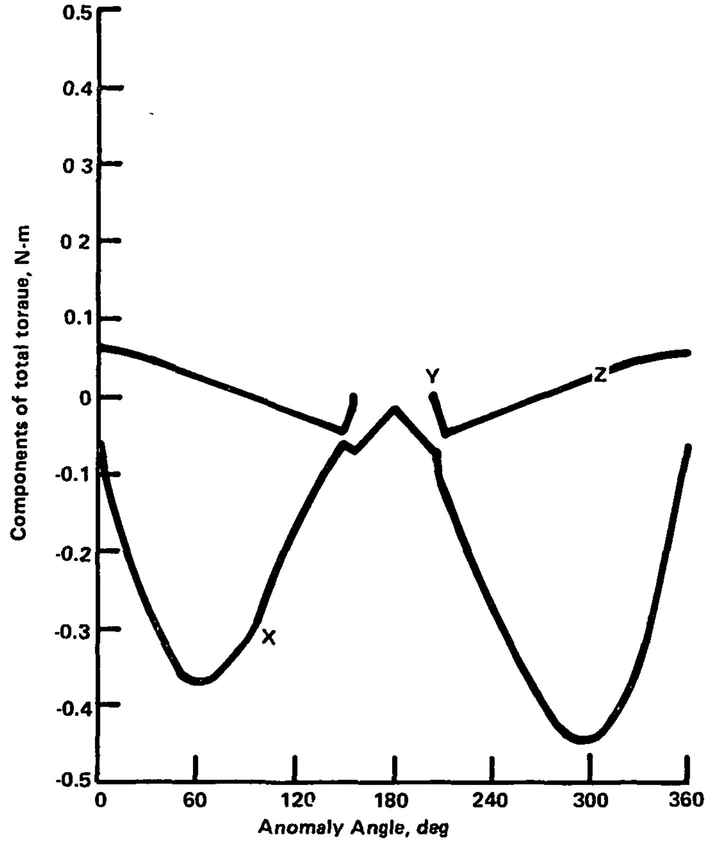

Figure 70. - Total force components versus anomaly angle.

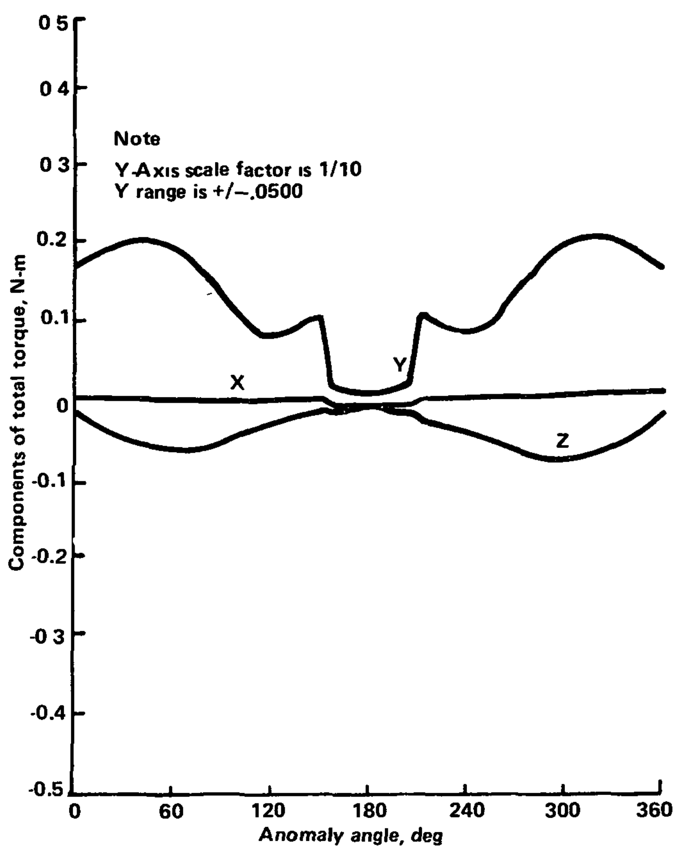

Figure 71. Total torque components versus anomaly angle.

Table 41 summarizes the required propellant mass for each satellite system studied during the program. The number and orientation of each required thruster was also determined for each case from the RCD module outputs. From the number of thrusters, the dry APS weight was then determined. For HF-, VHF-, and L-band systems, PPTs were assumed with a dry weight of 23 kg each and a volume of 0.275 m^3. A minimum of eight thruster modules was assumed for full 3-axis control and redundancy.

Ku-band satellites can use existing chemical thruster technology (e.g, hydrazine). The assumed dry weight of the chemical system is 91 kg. The higher propellant mass compared to electric propulsion makes the total weight comparable to the total weight of a PPT system.

TABLE 41. - VOA DVBS AUXILIARY PROPULSION SYSTEM REQUIREMENTS

System	Propellant mass		Reposition allowable	Dry mass*	Total APS mass	
	7 yr	10 yr			7 yr	10 yr
HF, 6-h orbit	12 kg	17 kg	70 kg	184 kg	266 kg	271 kg
HF, 8-h orbit	43	63	62	184	298	318
HF, 12-h orbit	44	63	50	184	269	288
HF, multi-launch, 6-h orbit	96	137	142	236	474	677
HF, multi-launch, 8-h orbit	102	146	120	236	458	654
HF, multi-launch, 12-h orbit	104	149	86	184	374	419
HF, inflatable reflector, array feed	1073	1533	86	368	1527	1987
HF, Molniya orbit	146	209	82	184	412	475
HF, small antenna, 6-h orbit	12	17	70	184	266	271
HF, small antenna, 8-h orbit	44	63	62	184	290	309
HF, small antenna, 12-h orbit	12	17	50	184	256	261
HF, small antenna, GEO	65	93	31	184	280	308
HF, small antenna, triply-synch	210	306	22	276	510	708
HF, small antenna, 24-h ellip	18	26	46	184	248	256
VHF, 12-h orbit	57	81	50	184	291	315
VHF, 24-h ellip orbit	160	229	46	236	442	511
VHF, multi-launch, 24-h ellip	180	257	80	276	570	644
VHF, inflatable reflector, 24-h ellip	1200	1714	46	368	1614	2128
L, GEO	32		16	184	232	
	318		152	91	561	
Ku, GEO	111		62			

*Mass of propulsion components only.

4.1.4 TT&C Weight and Volume Estimating

The TT&C subsystem includes the hardware required to sense and communicate satellite status and position information to a ground control station and to receive commands from a ground station. For this study, real-time access is assumed for all systems. Thus, no onboard data storage is required. The hardware assumed for HF- and VHF-bands TT&C subsystems is summarized in Table 42. The weight estimate used for HF and VHF systems was 32.1 kg. A mass density of 1000 kg/m^3 was assumed to compute TT&C volume. For L- and Ku-bands, the autotrack module would be eliminated, resulting in a weight of 26.4 kg.

TABLE 42. - TT&C EQUIPMENT LIST

Component	Status	Number	Weight
Onboard computer	SOA/NEW	(2)	8.3
Command receiver	SOA	(2)	6.6
Remote interface unit	SOA	(2)	2.1
Computer interface	SOA	(2)	1.0
Redundancy manager	SOA	(2)	2.7
Autotrack module	SOA	(2)	5.7
Cabling/harness	SOA	(1)	5.7

4.1.5 Thermal Control Subsystem Weight Estimating

The thermal control weight estimates are based on algorithms contained in the SCIAP subsystems properties module. Thermal control subsystem weight is computed as a function of radiative surface area using an assumed area mass density of 4.88 kg/m^2 (1 lb/ft^2). The required area is computed as a function of the BTUs of heat to be radiated (Q), maximum allowable temperature (T_{max}°R), end-of-life radiators emissivity (ε), end-of-life radiators absorptivity (α), and orbit altitude. The equations used to compute the required area are:

1) Altitudes below GEO:

$$\text{Area} = \frac{Q_{max}}{\sigma \varepsilon T^4_{max} - \alpha Q_s K} \qquad (4\text{-}15)$$

where

K is factor to account for orientation of radiators to Sun (0.707 for 45°)

σ is Stefan-Boltzman constant -0.1714 x 10^{-8} Btu/h ft^2 R^4

Q is solar constant 1353 w/m^2 (442 Btu/hr-ft^2)

2) Altitudes at GEO or above:

$$\text{Area} = \frac{2Q_{max}}{\sigma \varepsilon T^4_{max}} \qquad (4\text{-}16)$$

The volume is computed by assuming a radiator equivalent thickness of 0.05 meters (2 in.).

Usually, the heat to be rejected is computed automatically as the difference between the power into the load and the useful power out. However, HF and VHF satellites' thermal control subsystems could not be sized automatically because of the different sized transmitters and the requirement of some transmitters to be operated at different power levels depending on the zone to be covered (as discussed previously in Section 4.1.1.2). The power levels of the transmitters were estimated to fall into three categories: 45% at 1.5 times a nominal power, 15% at 2.5 times the nominal power, and 40% at 4 times nominal power. Thus, the radiative surface area for each transmitter must be increased appropriately. The weighted average of these transmitter size factors is 2.65 requiring an effective radiative surface area 2.65 times that required for the simple difference between maximum power available to the load and maximum power out.

Weight of other thermal control components such as insulation, cold plates, and heaters are included in the weight estimate of individual subsystems or components.

4.1.6 Equipment Bay and Mechanisms Weight Estimating

The relationships to estimate weight and volume of the equipment bay were taken from the SDCM. The equipment bay was assumed to be a conventional satellite structure applicable to a 3-axis stabilized spacecraft. The SDCM empirically derived weight estimate is:

$$W_e = K_d \quad W_s^{0.99} (L/D)^{0.24} \quad + 0.1 \, W_s \tag{4-17}$$

where

W_e = equipment bay weight (kg)

W_s = subsystem components weight (kg)

(L/D) = structural dimension ratio (1.0)

K_d = density coefficient (.129 for non-body-mounted solar arrays)

The equipment bay volume estimated is computed from the relationship:

$$V_e = 4.95 \quad V_s \tag{4-18}$$

where

4.95 = average volume sizing factor for satellites with solar arrays

V_s = volume of subsystem hardware to be contained in equipment bay

The mechanisms weight was estimated from data contained in the SDCM data base. For HF-, VHF-, and L-bands, the mechanisms weight was estimated at 102 kg. This includes the weight of all stowage and deployment mechanisms except for the deployable antennas (e.g, box truss ring structure, honeycomb panels). For Ku-band satellites, the mechanism weight was assumed to be 51 kg, primarily to account for the deployment mechanisms for the solar arrays.

4.2 TECHNOLOGY TRADEOFFS

The objective of the subsystem technology tradeoffs was to identify subsystem technologies that might enhance VOA DVBS missions, even though these technologies might be in an early development phase. This section will discuss why subsystem configurations were selected. The goals of the tradeoffs were to identify ways to minimize satellite weight, maximize reliability, increase lifetime, minimize cost, and reduce implementation time.

4.2.1 Antenna Structure Tradeoffs

The communications subsystem is the payload for VOA satellites and represents a significant part of the total cost, time to develop, and weight. Thus, the selection of the configuration and components will be critical to VOA satellite program success. Presented here are alternatives for the communications subsystem mode of operation and for subsystem components.

4.2.1.1 Antenna Technology Tradeoffs

Present antenna systems for Ku-band applications use solid parabolic reflectors in the order of 1- to 4-meter diameter. This technology is well established state of the art. However, larger antenna systems, particularly at the RF power levels needed for HF-, VHF- and L-band applications are relatively new technologies. The antenna types that make up the larger antenna systems fall into two types: reflectors and arrays. To use a reflector in such an application has the advantage of simplicity in the sense that the RF subsystem is not very complex. By comparison, an array requires a rather complex RF subsystem to assure that each element in the array is properly fed and phased.

Structurally, both large arrays and large reflectors are quite similar. Both must be stowable for STS launch and deploy reliably and accurately. Table 43 presents a comparison of maximum PFD available from an array and reflector for the same orbit and aperture. As shown, the PFD for the reflector is higher than for the array. In reality, the beamwidth of the reflector would be larger for the same diameter. Thus, for the same ground spot coverage, the PFD from the reflector would be still greater since a smaller reflector diameter would result in less weight for the structure. The reduced weight could be allocated to the EPS, increasing the available electrical power and thereby increasing PFD to the ground.

TABLE 43. - ARRAY VERSUS REFLECTOR PFD COMPARISON

Antenna	Aperture diameter	PFD
12 h—inflatable	168 m	290 μV/m
12 h—array	168 m	244 μV/m

The advantages of an array system, especially at the RF power levels and for the coverage requirements for VOA applications, is that power can be distributed over a number of element/transmitter pairs and the antenna pattern can be adjusted to meet coverage requirements. Distributing the power over a number of element/transmitter pairs helps to alleviate any arcing and multipacting effects from the generation of high RF powers in space. Also, heat dissipation problems and the risk of a single point failure are reduced. Unlike reflectors that require mechanical actuators to scan the antenna beam, an array has the capability to scan electronically by adding phasors at each radiating element. In addition, by selectively turning transmitters off and on, the antenna beam pattern of an array can be broadened or narrowed to meet zone coverage requirements. Combining phase control and selective activation allows an array to maximize power flux density on the ground and also provides extended coverage time for a given zone.

The above rationale was used in choosing an array over a reflector for the HF-, VHF- and L-band antenna systems. In the VHF-band system design that proposed a reflector (an inflatable reflector) an array was still used for the feed. The inflatable reflector concept resulted in a feed array of 26 x 26 meters and an inflatable aperture structure with extremely low mass. However, the large inflatable proved to have significant controllability problems and increased APS propellant requirements. For the array antenna concepts for HF- and VHF-bands, the box truss ring array was used because it provided lower weight and stowed volume than contiguous truss structures and provided a capability to attach both the radiating surface and ground plane surface onto the structure without adding additional structural elements. For the array antenna systems for the L-band, honeycomb panel arrays were used. These arrays are similar to existing technology which has been used in satellites such as the synthetic aperture radar antenna.

4.2.1.2 Transmitter Technology Tradeoff

For the HF-band, the requirements for DSB-AM require linear amplification if the carrier is modulated prior to uplinking and merely amplified in the satellite. Solid state power amplifiers (SSPA) typically are designed as linear devices while TWTAs have a nonlinear operating region. TWTAs require higher dc operating voltages than do SSPAs. While SSPAs can operate almost directly off a dc photovoltaic power bus, TWTAs require DC to DC conversion for power, thus increasing weight and complexity of the electrical power subsystem. Also, TWTAs have lower lifetime reliability than SSPAs, particularly when operated in a cyclic mode as is anticipated for VOA satellites.

As discussed in Section 3.3.4, powers above 1000 watts will require SSPAs using MOSFETs. Use of multiple SSPAs with array antennas will result in a weight penalty, but use of a transmitter per element provides high system reliability since failure of a small percent of transmitters should not seriously degrade performance. Also, use of SSPAs results in simpler packaging and permits distributed thermal control. Finally, since TWTA gain is proportional to wavelength, a TWTA at HF or VHF frequencies would be orders of magnitude larger than TWTAs for L- or Ku-bands if the same gains are required.

For Ku-band, TWTAs are in use and are selected for VOA satellites, based on their minimal required development cost and time and the availability of power levels needed for VOA satellites.

As an alternative to conventional space technology for the provision of single very high-power devices, we look to the technology that has provided the broadcast industry with high powered transmitters. This is the classical technology of triodes and tetrodes. Varian, for example, has developed a new tube called a klystrode, that combines the features of a klystron and a tetrode. It operates as a class-B linear amplifier in the manner of a tetrode, but with the reliability and high-power handling capability of the klystron. Such tubes have been operated in the 400 to 800 MHz region, and there are plans to extend operation to L-band. Powers available are up to 28 kW, with 30 kV beam voltage. The efficiency of this device when operating at full power can be 50%, and it has the unique feature that the dc power drawn is decreased as the input level is decreased. This feature could be particularly desirable for VOA operation, where power is at a premium. Peak power requirements from the primary power source could be reduced by the use of energy storage devices that can be charged during periods of low modulation, and by exploiting the average statistics of several channels aboard the same satellite. The average power requirements could also be reduced, since the power drawn would be low during periods of low modulation percentage. Use of this device would, of course, require a space qualification program, but might prove useful for L- or Ku-band satellites.

4.2.1.3 Array Antenna Phase Control

Signal processing for array phase control for steerable beams can be accomplished by splitting the signal from a common interface/intermediate frequency (IF) output through equal length and impedance cables to each transmitter. A separate phase shift circuit and command would then be used at each transmitter to perform shifting for steerable beams.

An alternative is to use minimum cable length to each transmitter, performing shifting in a separate IF stage for each transmitter. The shift for each would then be commanded as a function of desired beam steering with a built-in compensator relating the length of cable to the associated transmitter.

The common IF source approach has fewer electronics piece parts that should result in higher reliability and lower design and production complexity. The separate IF source approach requires less cable and therefore has lower weight (60 kg for largest satellite). However, for this approach, each IF stage must be shifted differently to compensate for the different cable lengths to each transmitter, making each stage design different.

Cost comparison is difficult to address, since the common source approach IF stage(s) must have higher gain to provide the same input power to the transmitters as the separate stage approach. This should result in higher cost per stage, but with significantly fewer stages, overall cost will probably be lower. Thus, the common IF source is recommended since it should be less complex, have higher reliability, and have lower cost.

4.2.1.4 Feeder Link Tradeoffs

The principal choice to be made in the feeder link system design is whether realtime uplink capability is required at all times. If so, then for low orbit satellites, up to perhaps seven or eight ground stations are required, spaced fairly evenly about the Earth. Higher orbits have greater visibility, and only two or three Earth stations are required. If realtime operation is not required, fewer Earth stations may be used, since low-orbit satellites will eventually pass over the United States and the programming may be uplinked during that pass. If jamming is a considera-tion, burst and store from the U.S. or nonhostile territory will provide protection. Finally, an intersatellite link can help to reduce the number of uplink stations, as well as provide protection against jamming. Although the relay satellite option shown is based on dedicated satellites, it may be possible to use the sound broadcast satellites themselves as intersatellite links to one another.

TABLE 44. - NUMBER OF FEEDER LINK STATIONS REQUIRED VERSUS SATELLITE HEIGHT

Satellite height	Great circle degrees of coverage, diam	No. of Earth stations for full coverage (assuming no overlap)
4,163, km	86.9	7.30
4,182	87.0	7.28
6,392	101.0	5.50
7,843	107.6	4.89
10,355	115.9	4.26
13,892	123 9	3.78
20,184	132 6	3.34
35,786	142.9	2.93
39,581	144.3	2.88
61,085	149.3	2.72

Table 44 summarizes the number of required ground stations vs orbit altitude. The recommended mode for HF- and VHF-bands systems is onboard crosslinks. Since there will typically be clusters of satellites spaced around an orbit, there will always be a realtime window from a ground station to any specific satellite. Since up to five ground stations would be required for some systems, the overall cost would be comparable to that for an intersatellite link. There would be an advantage with the intersatellite link option in that only two stations would be required, simplifying site acquisition and security.

As an example, Table 44 shows that four ground stations would be required for the 8-hour orbit (altitude of 13892 km). The estimated cost of each ground station is $56,500,000 for a 24-satellite system and a 20-yr operational lifetime. The estimated cost to develop the crosslink system is $9,000,000. The additional estimated first unit recurring cost per spacecraft for the crosslink system is $3,360,000. The two additional ground stations without intersatellite capability would cost $113,000,000. The estimated cost of the crosslink capability for a 20-yr lifetime (72 satellites) would be $126,000,000. Thus, there is no significant difference in cost between the two approaches.

A significant goal of VOA should be the transmission of maximum intelligibility in speech broadcasts, within the confines of power and energy available from the satellite. Techniques available can be used either to increase intelligibility for the power and energy available, to decrease the power and energy requirements for a fixed level of intelligibility, or to arrive at an intermediate compromise.

Intelligibility is increased with increased S/N ratio. One technique long used to increase the S/N ratio of voice transmission over the transmission channel is the use of compression and expanding, which may be applicable to the HF-band DSB-AM VOA system. Companding reduces the dynamic range of the signal carried over the transmission channel, so that high amplitude signals are transmitted at a relatively higher level. There is an accompanying reduction in quiet level noise and a notable increase in subjective or apparent S/N ratio. At the receiving end, the original dynamic range is restored, so that the listener receives a relatively faithful reproduction of the original. This restoration (as opposed to the compression alone) is particularly desirable in broadcasts of classical music and other programs that have a wide dynamic range, but even for voice it aids in maintaining natural sounding speech.

A disadvantage of conventional companding is that it requires a cooperative receiver with built-in expansion circuitry. Since the VOA system cannot depend on the presence of a population of such receivers, it is useful to ask whether there is an advantage in a system with compression only, without expansion at the receiver.

It turns out that there is such an advantage. By the amplitude compression of the voice signal, a higher average S/N ratio can be achieved. This in turn may allow usable reception using a reduced carrier power, and in fact, with reduced total power. This could be a significant factor for VOA systems that have such large power requirements. In effect, more of the primary power is diverted to the sidebands rather than to the carrier, which conveys no information, but is just an artifact of the classical AM system. However, the compression may also give the listener a feeling that what is heard is uninteresting and unrealistic. If the compression is applied only to that range of frequencies that influences intelligibility most, the natural variation to the lower and higher frequencies might convey the dynamics of a conventional transmission without defeating the objectives of compression.

The above technique increases not only the channel S/N ratio, but also the environmental or acoustic S/N ratio. This technique is somewhat similar to that used in Dolby B audio systems intended for a cooperative receiver. Listeners to these systems have in fact noted increases in intelligibility with the cooperative features of the receiver turned off.

Exploitation of the above and other power conservation methods demand that the system take all possible steps to maximize intelligibility. For example, attention must be given to ensuring an appropriate frequency response in the speech channel and properly equalizing to match microphone response. Consideration should also be given to differing equalization curves for different speaker's voices.

In addition, scrupulous attention must be given to the avoidance of nonlinear distortion in the speech channel, since this can seriously eat into the intelligibility budget. Random noise and unplanned level shifts must also be minimized. Level lineup must be maintained over a range of sources, including microphones, remote input channels, and prerecorded media. Operationally, this may call for more complete audition (medium preview) procedures than would ordinarily be required.

4.2.2 Electrical Power Subsystem Tradeoffs

4.2.2.1 Power Generation

Energy sources currently under development that possess the greatest potential for use in the spacecraft's power generation system are photovoltaic concentrator arrays, nuclear (SP-100), and solar thermal dynamic (using Stirling, Brayton, or Rankine cycles). Section 3.4.2 discussed the solar concentrator and nuclear sources. The solar dynamic source has many advantages, as can be seen in Figure 72 which compares specific mass and area of potential power systems. Note that a system with a relatively low specific mass may have a relatively high specific area, such as the silicon flexible blanket array and regenerative fuel cells (RFC) system. The solar dynamic Stirling system has the advantage of possessing both the lowest specific mass and area of the systems considered. The solar dynamic systems also have the potential for lower cost and higher reliability vs a photovoltaic system, and, with efficiencies of 23-25%, the solar dynamic collector area would be 35% of a silicon solar array. Table 45 summarizes the advantages and disadvantages of the power generation systems described here. If development continues at the present rate then a solar dynamic system should prove a viable candidate for high power applications in the far term. To ready this system for an early 1990 flight however, would require substantial engineering effort and would be high risk.

Development of 2-mil silicon cell technology would also result in improved performance of photovoltaic blankets. Table 46 shows the comparison of satellite subsystem weights, total power, and RF power for the maximum payload, 8-hour, HF baseline system. The increase in power (0.87dB) results in a 10.5% increase in PFD with the 2-mil technology as shown for a sample of zones.

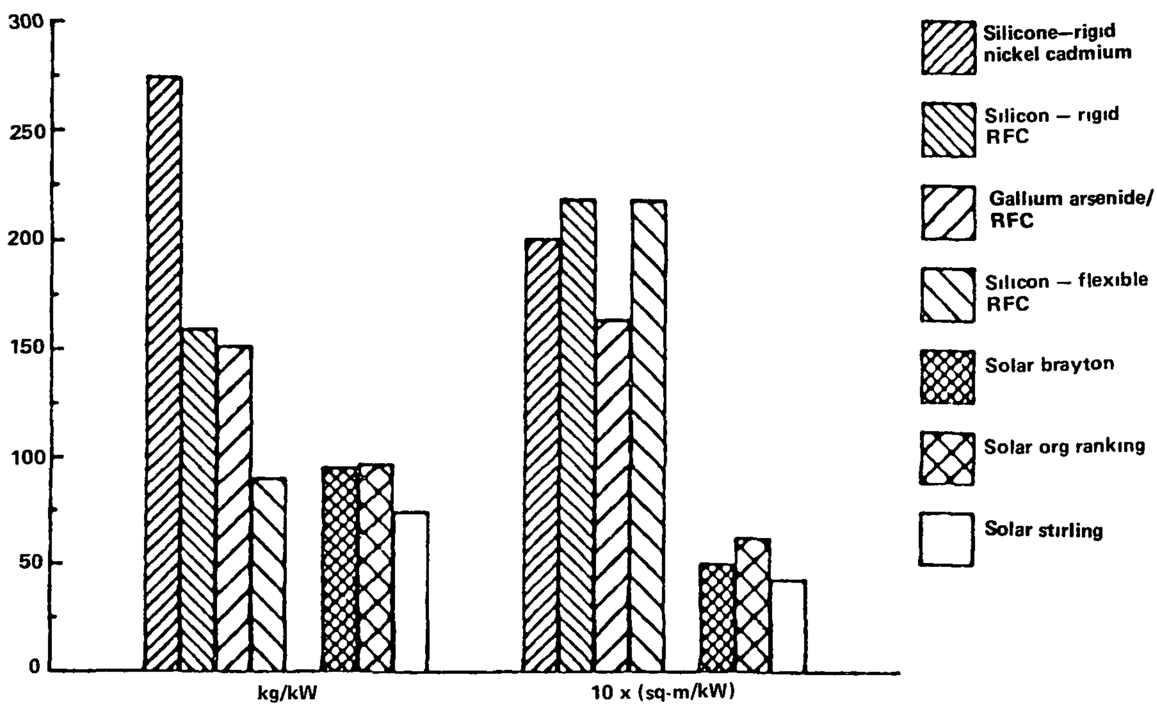

Figure 72. - Representative specific mass and area of potential power systems.

TABLE 45. - POWER GENERATION TECHNOLOGY TRADEOFFS

Technology	Advantages	Disadvantages
Photovoltaic planar solar array	– Low risk – Possible development by space station – Low cost	– Large area/drag – Low stiffness – Cell degradation high at some altitudes
Photovoltaic concentrator solar array	– Lower area/drag – Better performance	– Development required – Reflector life questionable – Need gallium arsenide cells (high cost)
Nuclear	– No sun tracking – No energy storage required – Lowest area/drag – Continuous operation – Direct ac generation possible	– Safety implications – High development cost/risk – High-temperature concerns – Dynamic interaction concerns – Reentry considerations
Solar thermal dynamic	– Low area/drag – No chemical storage – Direct ac generation possible	– Thermal storage concerns – High development cost/risk – Reflector life concerns – Accurate sun pointing required – Startup problems – Bearing life concerns

TABLE 46. - COMPARISON OF HF 8-HOUR BASELINE WITH 8- AND 2-MIL SI CELLS

Type of Solar Cell	Total EPS Power	RF power	PFD zone (one channel)		
			Zone 1	Zone 7	Zone 10
8-mil Si	93.1 kW	58 kW	320 μV/m	228 μV/m	282 μV/m
2-mil Si	108.9	70.8	354	252	312

Of the energy source technologies, a gallium arsenide concentrator array should be the first to be space qualified. Section 3.4.5 discussed the relative merits of this system and Figure 72 displays the savings in specific area vs silicon when paired with an RFC. Full development is not likely to occur by 1990 unless space station uses this technology, which is doubtful. Because the specific mass of a gallium arsenide concentrator is greater than a silicon flexible blanket array, this system may only be desirable for missions in orbits that experience the greatest effect from solar pressures and atmospheric drag. Table 45 compares this system to the other power systems.

In conclusion, the most viable near term (1990) energy source option is the photovoltaic flexible blanket array using the advanced silicon solar cell. While a solar dynamic system will have lower specific mass and area, for an unmanned satellite it will not be practical from the standpoint of maintainability. A nuclear source becomes attractive for very large power systems (200 kW) or for operation in areas of high radiation. If reductions in cost and blanket weight can occur for gallium arsenide arrays then that cell will be preferable to silicon for most applications.

4.2.2.2 Power Distribution

Areas in power distribution that will require optimization are cabling type, voltage type (ac or dc), power transfer mechanism (from solar array to conductors), voltage level, and voltage regulation for a dc system. A voltage level of 200 Vdc was previously selected as optimum (ref. Sect. 3.4.6). Considerations for selecting cabling type, voltage type, and power transfer devices are presented here.

4.2.2.3 Cabling

For cabling material, the product of electrical resistivity and density is the appropriate figure of merit for conductor lightness. For aluminum, the product is half that of copper (a 50% weight savings over copper can be realized). Some drawbacks of aluminum are its low tensile strength, poor flexibility, and poor crimp terminability. Poor crimp terminability can result in creep, causing looseness in the connector, eventual arcing, and an open circuit. Alloying aluminum with some other conductor such as copper should solve these problems and make aluminum available for large power systems in the 1990s.

Figure 73 illustrates the effects of increasing distribution voltage and type of conductor wire (aluminum or copper) on the combined distribution and power generation system weight. A 100 kW power level and 122 meter one-way length were assumed. The graphs are representative of system weight variance for a photovoltaic source. Also shown are the points for aluminum and copper wire at which the current in the wire exceeds its 70°C temperature rise limit. To further reduce conductor weight and thus increase distribution loss would drive the conductor to higher temperatures. The 70°C limit was used to provide a benchmark, the actual temperature limit being a function of the insulator material. It should be noted that the weight difference between copper and aluminum decreases as distribution voltage increases.

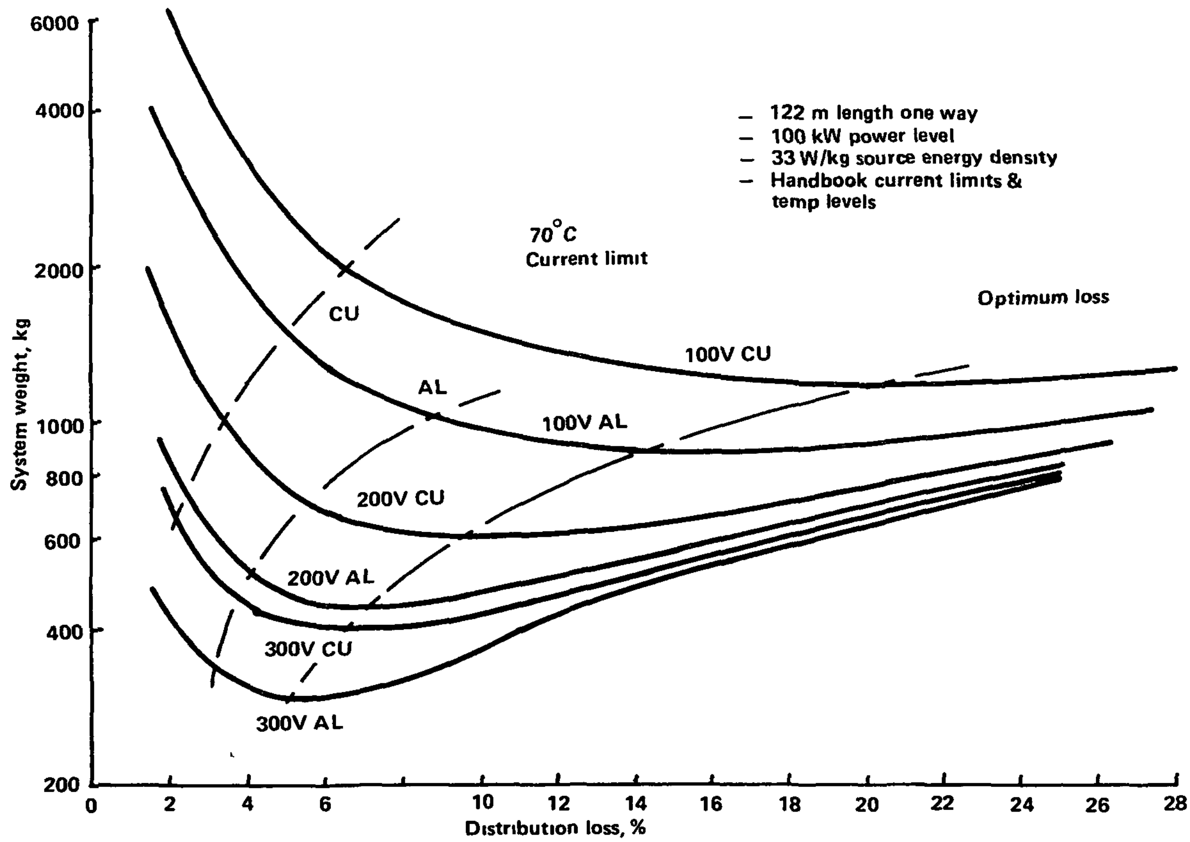

Figure 73. - System weight versus distribution loss.

Far-term options for conductor material are intercalated carbon fibers and sodium. A sodium conductor would be extremely lightweight, with a resistivity-density product 45% less than aluminum. The tensile strength of the conductor will depend upon the type of sheath used. The use of such a cable would be best suited for high current or extremely long distributors. The technological risk of sodium is high for the mid-1990s and it should not be considered as a viable option.

The application of intercalated carbon fibers is a possibility. Carbon fibers alone have a high resistivity (20 times that of copper), but doped with the proper elements, carbon can become a good conductor. The projected resistivity-density product of intercalated carbon is comparable to that of copper, so the advantage of using carbon is strictly cost. This technology is still experimental and should not be available by mid-1990.

4.2.2.4 Voltage Type (ac versus dc)

The advantage of high frequency ac to dc voltage is primarily the weight reduction with increasing frequency of energy storage devices such as transformers, capacitors, and other magnetic components. Also, filter sizes in power conditioning components decrease and voltage conversion is simpler and completed with fewer losses. With the development of high power switching and magnetic devices for use in high frequency ac systems, large spacecraft power systems will have the option of selecting an ac or hybrid ac/dc distribution and control network by 1990. Because cabling is not significantly affected by an ac system, the amount of weight savings realized may not be great enough when compared to the total system mass to justify the risk of developing an ac system. The type of voltage used, ac or dc is not seen as design critical to the power system.

4.2.2.5 Power Transfer

Power transfer assemblies currently in use have been designed for low voltage, low power satellite applications. They possess many design limitations that could prevent their use in high power, high voltage systems. Some of these limitations are a 200 V voltage limit due to critical pressure, a life limited by the brush wear in a vacuum environment, a current limited to the power dissipation in the interfacing elements of the electrical transfer mechanism, and a short circuit failure mode at wearout conditions. Alternatives to the conventional slip ring designs are roll rings, twist flex cables, and pressurized slip rings.

The roll ring is a new approach that incorporates a complex structure of mechanical parts. The device has a significant reduction in friction that may lead to instability if not damped properly. The roll ring is still susceptible to corona discharges at critical pressure, that could be significant considering that the broad array of elements between adjacent channels would increase the electric field strength in the gas media between rings. A roll ring would be an advantage in a high velocity control system that could benefit from the low friction.

The twist flex assembly has many attractive advantages. It is extremely simple and lightweight. The assembly permits power transfer through insulated wire bundles from one rotating disc to a second rotating disc, having a limited rotating angle of $\pm 205°$. The voltage carrying capability may be increased by increasing the dielectric strength of the wire and improved insulation can prevent corona discharge. Other advantages are minimal power loss, long lifetime and high reliability open circuit failure mode, and zero noise for signal handling. The major disadvantage is the residual torque generated by the twisted cables, which require additional power to overcome the loss. This is the preferred system assuming the Sun tracking requirement can be satisfied by the limited rotation.

For power systems with operating voltages greater than 250 Vdc, a pressurized slip ring assembly would be needed. A pressurized system can eliminate the corona discharge problem at high voltage in addition to allowing an increase in power density with improved heat transfer. The disadvantage is its complex design and early development stage.

4.2.2.6 Energy Storage

Because transmitters are assumed to be off during eclipse, the energy storage options are narrowed to nickel cadmium and nickel hydrogen batteries. Nickel hydrogen batteries have an advantage over nickel cadmium batteries in energy density and maximum allowable depth of discharge. Nickel hydrogen batteries may also prove to have a greater cycle life than nickel cadmium. The disadvantage of nickel hydrogen is that they are unproven in space at LEO, Although this should not be the case by 1990. Figure 74 depicts the allowable depth of discharge vs charge cycles for nickel hydrogen and nickel cadmium batteries. The figure indicates that for maximum depth of discharge, a nickel cadmium battery could not meet a seven year mission at LEO, unless a spare battery was carried to assume operation near the five year point. Also, at GEO the weight advantage of nickel hydrogen becomes small because of the relatively deep depth of discharge allowed for nickel cadmium. Figure 75 shows this, with its display of the power system weight needed to supply 500 watt average load vs altitude for nickel hydrogen and nickel cadmium batteries. Because the battery weight is only a small fraction of the total power system weight, the selection of a battery type is not seen as critical. If nickel hydrogen proves to have a greater cycle life than nickel cadmium then that battery should be chosen, especially for a mission at LEO. In conclusion, the energy storage technology will exist by 1990 to support the mission requirements with no adverse effects on system weight assuming transmitters are off during eclipse.

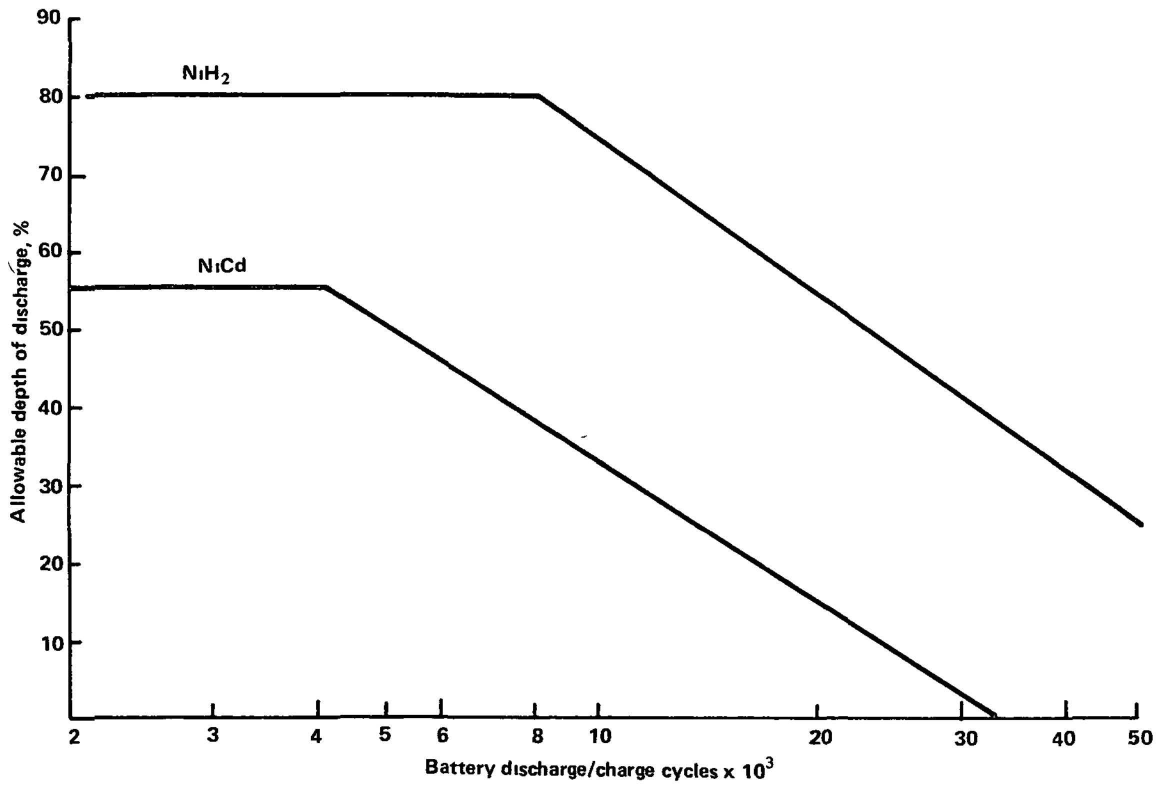

Figure 74. - Allowable depth of discharge versus battery discharge/charge cycles.

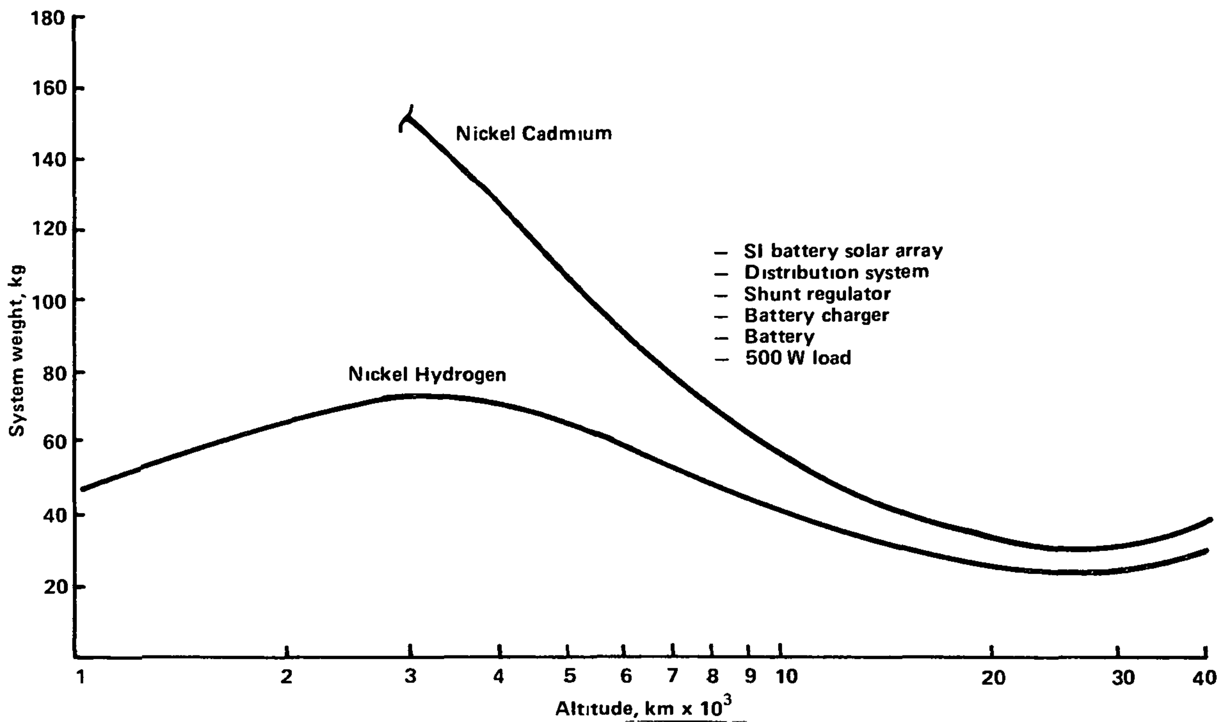

Figure 75. - Housekeeping power system weight versus altitude.

4.2.3 ACS/APS Technology Tradeoff

Table 47 summarizes advantages and disadvantages of candidate ACS/APS actuators. Table 48 shows the results of a tradeoff between APSs using chemical blowdown, chemical bipropellant, and PPT. The results shown are representative of the propellant mass and APS weight required for an HF multiple launch case and are about one-half of what would be required for the small HF design in the triply-synchronous orbit. Except for the Ku-band satellites, electric propulsion will result in a weight advantage, although for some L-band concepts there is little difference. For these L-band systems, a small added weight for a chemical system might be offset by lower cost for the chemical system.

TABLE 47. - CONTROL TORQUE ACTUATOR SUMMARY

Torque Actuator		Disadvantage
Thrusters	No cross-coupling with the vehicle motion to produce undesirable torques for which compensation must be made	
– Electric	High specific impulse, applicable to missions with a long lifetime	Low-thrust, high-power requirements; power already available for most VOA satellite concepts; well-suited for distributed system
– Chemical	High thrust, low risk, lower dry weight	Low specific impulse, no applicable to missions with a long lifetime; tankage volume & weight can be high; not well suited for distributed system, more complex thermal control
Momentum exchange devices reaction wheels & CMSs	Ideal when disturbances are cyclic with respect to an inertial reference frame & the secular component (bias torque) is small; could reduce the size & number of thrusters required for the mission	Cross-coupling with the vehicle motion to produce undesirable torques that require compensation; requires a desaturation control law scheme; number required could be prohibitive in terms of size, weight, & power required for large VOA antenna concepts
Magnetic torquers	Used in conjunction with momentum exchange devices for desaturation or momentum management purposes	Magnetic field of Earth is time variant & strongly altitude dependent, at any instant torque can be produced only along components normal to the local magnetic field vector; practical limitations in power supply & coil size make the generation of large torques impractical for VOA orbits

TABLE 48. - APS CANDIDATES WEIGHT TRADEOFF RESULTS

System	I_{sp}	Dry weight	Propellant for 7 yr	Total APS weight
Blowdown	230 s	220 kg	1033 kg	1253 kg
Bipropellant	400	220	594	814
PPT	2200	184	108	292

4.3 COST ESTIMATING PROCEDURE

The cost estimating procedure developed and used identifies the costs of each subsystem, total satellite recurring and nonrecurring costs, launch costs and ground operations costs. The end results of the cost estimating procedure are satellite system and program life-cycle cost (LCC) which can be used to estimate LCC per channel hour.

The first step was to select or derive cost estimating relationships (CER) that are applicable to VOA DVBS. The Denver Aerospace documented LCC methodology and cost analysis data base served as a starting point. We used the multiple independent cost estimating approach. This approach evaluates LCC estimates from several models, qualifying each by relation of hardware configuration assumed by the model to configuration defined for the satellite. The LCC estimating sources that were used included:

1) The Denver Aerospace 15-volume cost analysis data base;

2) the USAF Space Division model;

3) the NASA MSFC LSS model;

4) the NASA LeRC SDCM;

5) The U.S. Navy LCC Handbook.

In some cases, existing CERs were adjusted to reflect judgement as to applicability to VOA satellite concepts. Finally, new CERs were developed from information contained in recent literature, specifically for costs associated with electrical power subsystems that produce more than 20 kW, electric propulsion, and high power solid state power amplifiers. The CERs identified as candidates for estimating LCC of VOA satellite concepts were included in a computer program developed during this program. This program is documented in the Appendix and is delivered as part of the contract.

The second step was to estimate the performance parameters required by each CER. These estimates were obtained using the methods described in Section 4.1. Finally, a quantitative risk assessment method, including the risk assessment and management program (RAMP), was used to further qualify the LCC estimates obtained from the various CERs. The point LCC estimates presented in Section 5 are defined as the 50/50 values obtained by running RAMP. A further discussion of the risk analysis method is presented in Section 6. Following is a discussion of the approach that was taken to derive the base cost estimates for each subsystem and the overall DVBS.

4.3.1 Communication Subsystem Cost Estimates

The communications subsystem is broken up into: (1) the antenna structure, including array or reflector surface and ground plane; (2) the feeder link electronics and antenna(s); (3) the onboard signal processing; and (4) the payload transmitters. Since no prior data existed for LCC of antennas greater than 20 meters in diameter or for space qualified amplifiers (transmitters) in the HF- and VHF-bands, new CERs had to be developed. The CERs used for the communications subsystem are listed in Table 49.

TABLE 49. - COST ESTIMATING PROCEDURES FOR COMMUNICATIONS SUBSYSTEM

Component	Band	NRC CER	RC CER	Source
Box-ring truss planar array	HF, VHF	$NRC = 0.013W_t$	$RC = 0.002W_t$	MSFC, U S. Navy model, judgement
Honeycomb fiberglass array	L	$NRC = 0.013W_t$	$RC = 0.002W_t$	Same
Graphite/epoxy reflector	Ku	$NRC = 0.722W_t^{0.263}$	$RC = 0.0361W_t^{0.65}$	SDCM model
Solid-state transmitters	HF, VHF	$NRC = 0.746W_t^{0.7}$	$RC = 0.06N_t$	Space Division model
	L	$NCR = 0.746W_t^{0.7}$	$RC = 0.006N_t$	Vendor survey
Uplink	All	$NRC = 0.373W_t^{0.7}$	$RC = 0.051W_t$	Space Division model
Signal processing	All	$NRC = 0.373W_t^{0.7}$	$RC = 0.057W_t$	Space Division model

Legend:

N_t Number of transmitters

4.3.1.1 Antenna Cost Estimates

For planar array antennas, the U.S Navy LCC handbook recurring cost estimate includes the structure, dipole elements, elevation network, and ground plane. In 1984 dollars the CER for recurring cost is \$4,900 to \$9,800/m^2. The MSFC model CER for graphite epoxy structures is \$2,205/kg (\$1,000/lb). For a 60-meter diameter, 1,000 kg box truss ring structure, the Navy CER predicts a recurring cost of from \$13.9 to \$27.7M dollars while the MSFC CER predicts \$2.2M. Since the Navy CER assumes a continuous structure and the MSFC model is for graphite structures only, a compromise value of \$4,410/kg (\$2,000/lb) was selected for array antenna recurring cost. For nonrecurring cost, the MSFC CER at \$28,665/kg (\$13,000/lb) was considered adequate. These CERs were also selected for the L-band honeycomb fiberglass panel array antennas.

The Ku-band reflector antenna costs were best fit by the SDCM model. However, the SDCM data base also contains specific costs for graphite epoxy Ku-band reflectors used for the LANDSAT-D program. These costs were used for the VOA Ku-band systems. The total recurring cost from the SDCM data base is \$790,000 while the nonrecurring cost is estimated at \$1,700,000 for one antenna or \$850,000 per antenna for multiple antennas. There are up to eight reflectors (two uplink and six downlink) for a single Ku-band satellite concept.

4.3.1.2 Feeder Link Electronics and Antennas

A Ku-band feed link is assumed for all systems, augmented by a V-band crosslink capability for HF and VHF concepts. The Space Division model CER for communications electronics was used to estimate costs for all satellites based on the weight estimates described in Section 4.1. The costs for the L- and Ku-band satellites is considerably less than the HF- and VHF-satellites since no gimbaling, autotrack, or crosslink are required.

4.3.1.3 Signal Processing Cost Estimates

For all systems, the Space Division model communications electronics CERs were selected to estimate cost of onboard signal processing electronics. The weights used for the CERs were obtained using the estimating procedure of Section 4.1.

4.3.1.4 Transmitters Cost Estimates

The nonrecurring cost estimate for HF- and VHF-bands transmitters uses the Space Division Model CER for communications electronics. For HF and VHF satellites with different sized transmitters, the average transmitter weight was used as the input to obtain nonrecurring cost. For each of the four L-band systems, the largest transmitter size from all satellites in a system was used for the input to the CER.

The nonrecurring cost for the Ku-band system is based on a TWTA cost from the SDCM. TWTAs for Ku-band should require little development. A value of $37,500 was used for the development cost.

Recurring cost estimates for HF- and VHF-bands were obtained from a CER developed after conversations with vendors. Estimated cost to produce solid state amplifiers up to 1 kW was $60,000 per unit. For L-band, the Space Division model CER for communication electronics was selected. A learning factor included in the LCC estimating computer program reduces the overall average unit cost based on the number of transmitters. To estimate the transmitter's total recurring cost for each L-band system, the computer program was run for each complement of transmitter size and number. The total recurring cost for each set of transmitters (corresponding to each unique array antenna) was obtained and summed to get the total recurring transmitter cost for the system.

The Ku-band recurring cost estimates use an average cost for a 20 watt TWTA of $790,000 as obtained from the SDCM data base. This value is reduced by an 80% learning factor for multiple TWTA's required on each of the Ku-band system satellites. Thus the average unit cost is $632,000 per TWTA.

4.3.2 Electrical Power Subsystem (EPS) Cost Estimates

The EPS will provide power far in excess of that required for any existing satellite system. For this reason, the CERs in existing LCC models were considered inadequate for estimating EPS costs for HF-, VHF- and L-band satellites. A brief literature search was conducted to identify cost estimates of high power EPS. From information in ref. 17, 19, and 20, data points of cost vs power were identified as shown in Table 50. From the MSFC model, typical EPS components cost fractions were identified as shown in Table 51.

TABLE 50. - PLANAR SILICON ARRAY COST DATA, 1984 $M

Source	Power	Nonrecurring cost	Recurring cost
Lockheed MSFC Study (NAS8-32981)	206 kW 311	$21M 21	$67.2M 85.5
TRW LaRC Study (NAS1-17568)	12.5	12	22
JSC space station study	100	26	54

TABLE 51. - EPS COMPONENT COST FRACTIONS

Component	Nonrecurring cost fraction		Recurring cost fraction	
	MSFC	VOA	MSFC	VOA
Solar arrays	0.257	[0.38]	0.617	[0.70]
Batteries	0.126	[0.50]	0.091	[0.05]
Distribution	0 166	[0.17]	0.103	[0 10]
Control	0 451	[0.40]	0 189	[0.15]

From the data of Table 50, expressions for nonrecurring (NRC_{sa}) and recurring (RC_{sa}) cost of large solar arrays were developed as:

$$RC_{sa} = 0.3P + 20 \tag{4-19}$$

$$NRC_{sa} = 0.094P + 16$$

As indicated in Table 51, the solar arrays account for a fraction of the EPS cost. For conventional systems the fractions shown under the MSFC column would be used. However, for VOA high power satellites, the batteries will represent a very small part of the total EPS cost. For this reason, the new set of component cost fractions were identified for VOA high power satellites. Since the solar arrays are now assumed to account for 70% of EPS recurring cost, a new EPS recurring CER is obtained by dividing the cost for solar arrays by 0.7 resulting in:

$$RC_{eps} = 0.43P + 28.5 \tag{4-20}$$

Likewise, a new EPS nonrecurring CER is obtained by dividing the solar arrays development cost relationship by 0.38 resulting in:

$$NRC_{eps} = 0.247P + 42.1 \tag{4-21}$$

where

RC_{eps} = EPS recurring cost ($M 1984)

NRC_{eps} = EPS nonrecurring cost ($M 1984)

P = Required EPS power (kW)

A comparison of the MSFC and new VOA CERs at 20 kW shows good comparison for recurring costs ($35M and $37M respectively). However, the estimated nonrecurring costs from the MSFC model is $102M while the corresponding VOA EPS CER results in $47M. The VOA CER is considered better since it takes into account an economy of scale and, the MSFC model may include the multiplicative factor of 2.5 to 3.0 used by MSFC to reflect increased development cost for manned vs unmanned spacecraft. The EPS CERs are summarized in Table 52.

TABLE 52. - COST ESTIMATING RELATIONSHIPS FOR OTHER SUBSYSTEMS

Subsystem	NCR CER	RC CER	Source
Electrical power > 20 kW	$NCR = 42.1 + 0.247P$	$RC = 28.5 + 0.043P$	Lockheed multi-kW study, NASA space station, Boeing electric prop. NASA MSFC.
< 20 kW	$NCR = 14.3P^{0.656}$	$RC = 3.62P^{0.757}$	
Attitude control	$NRC = 2.923 + 0.2241W_T$	$RC = 0.0577W_T$	Space Division Model
Auxiliary propulsion	$NCR = 0.661 + 0.049W_T$	$RC = 0.0085W_T$	Space Division model & Boeing electric propulsion study
Chemical blowdown	$NRC = 0.36W_T^{0.6242}$	$RC = 0.0808W_T^{0.722}$	JPL (JPLD-972, 12/83)
– N_2H_4 pressurant	$NCR = 0.69W_T^{0.545}$	$RC = 0.141W_T^{0.65}$	Same
– N_2O_4/MMH	$NCR = 1.04W_T^{0.545}$	$RC = 0.211W_T^{0.65}$	Same
– LO^2/LH^2	$NCR = 3.2W_T^{0.545}$	$RC = 0.63W_T^{0.65}$	Same
– H^2 resistojet	$NCR = 1.64W_T^{0.545}$	$RC = 0.27W_T^{0.65}$	Same
– Cold gas	$NCR = 0.144_T W^{0.226}$	$RC = 0.36W_T^{0.206}$	Same

Legend:

P Power, kW

4.3.3 Attitude Control Subsystem (ACS) Cost Estimates

The satellite designs all use the reaction control system (RCS) for attitude control. Therefore, the attitude control subsystem consists of electronics alone. For this reason, the Space Division LCC Model CER for attitude determination was considered the best choice for both recurring and nonrecurring satellite costs. The ACS CERs are shown in Table 52.

4.3.4 Auxiliary Propulsion Subsystem (APS) Cost Estimates

The existing CERs are based on use of chemical APS, generally hydrazine blowdown or pressurized designs. For HF-, VHF-, and L-band satellites, electric propulsion was considered a better design because of the higher specific impulse and resulting lower propellant and subsystem mass. Proposed use of electric propulsion required development of a CER for electric propulsion. From ref. 21, a cost relationship was identified for recurring cost of an electric propulsion subsystem. For nonrecurring cost, an assumption was made that development costs would be about the same as for a chemical subsystem. As a result, the Space Division CER for nonrecurring cost was considered adequate. The CERs are shown in Table 53.

TABLE 53. - COST ESTIMATING RELATIONSHIPS FOR OTHER SUBSYSTEMS

Subsystem	NCR CER	RC CER	Source
Thermal control	$NRC = 1.84 + 0.172W_T^{0.66}$	$RC = 0.0361W_T^{0.65}$	Space Division model
TT&C	$NRC = 1.38 + 0.063W_T$	$RC = 0.66 + 0.056W_T^{0.93}$	Space Division model
Equipment bay	$NRC = 10.84 + 0.172W_T^{0.66}$	$RC = 0.361W_T^{0.65}$	Space Division model

4.3.5 TT&C Subsystem Cost Estimates

The CERs contained in most LCC models should be adequate for estimating TT&C subsystem costs. For consistency, the CERs from the Space Division model were selected. These CERs are shown in Table 53.

4.3.6 Thermal Control Subsystem Cost Estimates

The Space Division CERs were selected, again for consistency. The CERs are also shown in Table 53.

4.3.7 Equipment Bay and Mechanisms Cost Estimates

The equipment bay structure and the mechanisms for stowage and deployment will be similar to designs used on previous unmanned satellites. Thus, the Space Division model CERs should be adequate. The CERs are also shown in Table 53.

4.3.8 Other Program Costs

The nonrecurring and recurring costs estimated by the subsystem CERs do not include other generic program costs. The other costs that were included in the cost estimating approach follow:

1) Aerospace ground equipment costs;

2) Systems engineering, integration, and program costs;

3) Launch costs;

4) Ground operations costs.

4.3.8.1 Aerospace Ground Equipment Costs

These costs include tooling and investment costs for development, test, and production facilities. These costs are estimated by the Space Division Model at 11.3% of platform costs. Nonrecurring platform cost is the sum of the subsystem nonrecurring costs and likewise, recurring platform cost is the sum of subsystem recurring costs.

4.3.8.2 Systems Engineering, Integration, and Program Level Costs

Systems engineering and integration costs are expressed differently by different LCC models. The SDCM model uses the following factors:

1) 32% of design cost,

2) 27% of development and test cost,

3) 32% of production engineering cost,

4) 22% of first unit fabrication cost.

The MSFC LCC model assumes integration costs to be 15% of the nonrecurring platform cost and 10% of first unit production cost. Other program support costs (e.g, management, quality control) are estimated as a percent of total costs including integration. The SDCM model estimates these costs at 18% of costs before integration. Our approach is the sum of the subsystem's recurring and nonrecurring costs to obtain base recurring and nonrecurring costs. These base costs are then multiplied by 10 and 15% respectively to include integration cost and then by 23% to include program level costs.

4.3.8.3 Launch Costs

The launch costs for each satellite include the cost of the upper stage and the cost for STS launch computed as a percent of STS payload used times cost for a dedicated launch. The percent of payload used is the maximum of the total of satellite plus upper stage weight divided by 65,000 lb, or upper stage plus satellite length divided by 60 ft. This percent is then multiplied by the assumed dedicated STS launch cost. For point estimates, a subsidized total launch was assumed at a cost of \$100M for full payload in \$1984. The upper stage costs assumed follow:

1) TOS/AMS - \$15M,

2) Centaur G - \$25M,

3) Centaur B - \$50M,

The basic launch cost vs weight equations are then:

STS/TOS/AMS - Cost = \$15M + \$100M $(34{,}541 + W_s)/65{,}000$, (4-22)

STS/Centaur - Cost = \$25M + \$100M $(37{,}518 + W_s)/65{,}000$, (4-23)

STS/STS/Centaur B - Cost = \$150M + \$100M $(W_s)/65{,}000$, (4-24)

where

W_s = Spacecraft weight (lb)

The equivalent launch cost as a function of length equations are:

STS/TOS/AMS - Cost = \$15M + \$100M $(20 + L_{ss1})/60$, (4-25)

STS/Centaur - Cost = \$25M + \$100M $(23 + L_{ss1})/60$, (4-26)

STS/STS/Centaur B - Cost = \$150M + \$100M $(L_{ss1})/60$, (4-27)

where

L_{ss1} = Spacecraft stowed length (ft)

The maximum launch costs were limited to the following:

STS/TOS/AMS - $115M,

STS/Centaur - $125M,

STS/STS/Centaur B - $250M.

4.3.8.4 Ground Operations Costs

The ground control costs include the costs to maintain the operational orbit, to maintain the required satellite attitude, and to perform satellite on-orbit operations through the TT&C subsystem. The costs of a station include facility fabrication, ground equipment, procurement, and manpower costs. Although manpower cost is spread over the station lifetime, it is lumped into the total cost of each station. An eight person crew providing three shifts of coverage is assumed for each station with tracking capability as required for nongeostationary orbits. The eight persons include a working manager, two maintenance personnel, and five mission operators/analysts. The rate assumed for personnel was $75,000 per year. Total station cost for a 7-yr operation is $24,000,000, including manpower costs, equipment replacement, and spares. A further assumption is that the basic 8-person station can monitor and control up to eight satellites using satellite cross links for both spacecraft control and program feed. Each additional set of eight satellites will increase ground operations cost by $3.5M for each 7 years for three additional personnel and additional tracking equipment. The total 7-yr ground operations cost for two stations for nongeostationary orbit systems is then expressed as:

$$\text{Cost} = \$24\text{M} + \$7.0\text{M}\,(n) \qquad (4\text{-}28)$$

where

n = number of additional sets or partial sets of eight satellites

The cost of geostationary ground stations is estimated at $18,000,000 for each seven years of operation. The reduced cost results from requirement for fewer personnel and no tracking equipment. Also, none of the VOA geostationary systems have more than eight satellites; additional crews and equipment will not be required.

4.3.8.5 Learning Factor for Satellite Fabrication

To derive the total system life-cycle cost for multisatellite systems, a learning factor is assumed for satellite fabrication. The basic learning curve equations are:

$$C_{ave} = C_{u1} N^{-b} \qquad (4\text{-}29)$$

where

b = log(m)/log(2)

C_{ave} = average cost of N units

C_{u1} = first unit fabrication cost

N = number of units to be fabricated

m = slope of learning curve (0.9 assumed)

The total fabrication cost (C_t) is then:

$$C_t = C_{ave}\, N = C_{u1}\, N^{(1-b)} \qquad (4\text{-}30)$$

The total system recurring cost, including launch cost, excluding inflation factors for phase acquisition, is then:

$$RC_t = C_t + N \cdot C_l \qquad (4\text{-}31)$$

where

RC_t = total recurring cost

C_l = cost per launch

5.0 SATELLITE SYSTEM DESIGN AND ANALYSIS

The system concepts and satellite design options for the four bands are presented in this section. Satellite designs were generated and analysis performed to determine the key performance parameters of weight, volume, cost, and RF power flux density on the ground. These satellite designs were then compared to the payload capability of the launch vehicle (Centaur G, TOS/AMS, and a hypothetical large Centaur) to ensure they were launch compatible.

For HF and VHF satellite designs, the channel requirements and PFD requirements were quite severe. Where these requirements could not be met using a launch compatible satellite, the maximum capability satellite was determined and its reduced capability was calculated. These reduced capabilities were then compared to the VOA requirements to develop a system within reasonable cost parameters. Because the capabilities of satellites launched on a Centaur could not achieve VOA requirements, a larger satellite design was developed that would use a full orbiter bay and a hypothetical Centaur-type stage that would fill a second orbiter and produce a much greater capability on a single satellite.

The design and analysis flow used for the satellite system design and analysis is presented in Figure 76. Subsystem selection was based on the desire to use as much off-the-shelf hardware as possible, but still achieve the VOA requirements. This was feasible for the Ku-band and L-band systems. For the VHF and HF systems, both the antenna and electric power subsystem would require advanced technology. The power subsystem technology could be derived from space station development available in the early to mid 1990's.

Analysis was performed for all satellite system options to determine the ability of a constellation of the proposed satellites to cover the 15 zones at the VOA specified times.

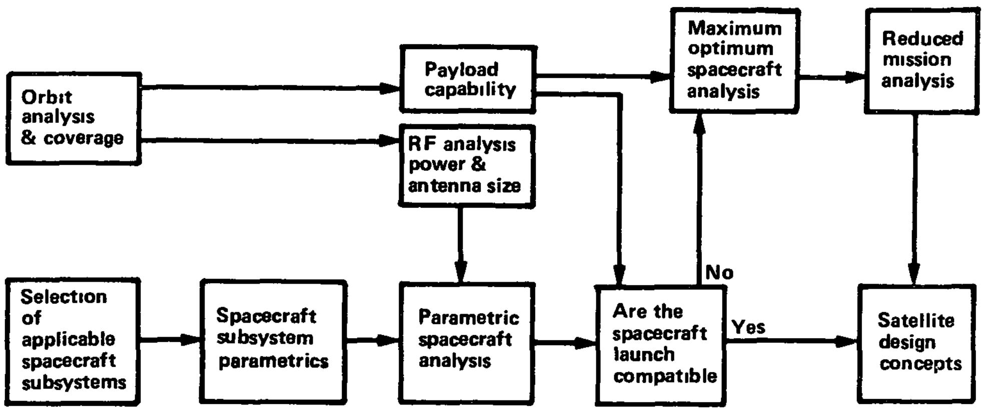

Figure 76. - Satellite system design and analysis approach.

5.1 KU-BAND SYSTEM

For Ku-band operation, the satellite system design resulted in a single point design to meet all VOA requirements. The system uses existing technology at reasonable power levels. The Ku-band system consists of three geostationary satellites in three orbit slots. Each satellite consists of a farm of offset fed parabolic dish antennas attached to a central spacecraft bus. The number of antennas per satellite varies depending on the number of zones covered.

The parabolic dishes use graphite composite technology to provide enhanced structural and thermal characteristics. Figure 77 illustrates the proposed satellite design. The following sections describe the results of the various design and analysis processes used to determine the satellite characteristics.

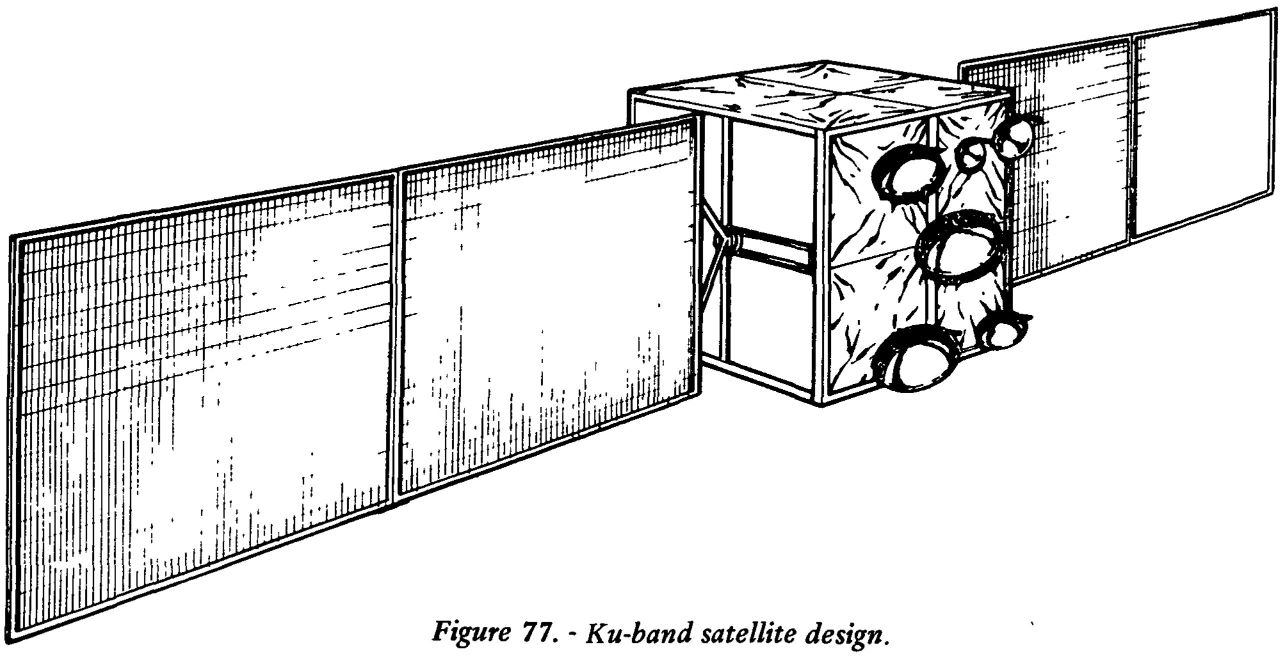

Figure 77. - Ku-band satellite design.

5.1.1 Ku-Band Systems Weight and Volume Estimates

The Ku-band satellites weight and volume are summarized in Table 54. The electric power subsystem requirements are based on a transmitter efficiency of 35% and housekeeping power of 500 watts. Also, unlike systems designed for the other bands, the Ku-band satellites are sized for operation during occultation. Table 55 shows the results of the sizing run for the first satellite of the system. This satellite has three payload antennas while the other two satellites each have six.

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TABLE 54. - KU-BAND SATELLITE WEIGHT AND VOLUME ESTIMATE

****

Ku Band S/C #1

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	228	2.05
Auxiliary Propulsion Subsytem	203.1	2.22
Telemetry, Tracking and Command	26.4	.02
Electrical Power Subsystem	72.08425	.3179024
Thermal Control Subsystem	4.625708	4.736863E-02
Equipment Bay Structure	83.58633	11.19356
TOTAL SYSTEM SUMMARY	617.7963	15.84884
	1362.241	(Lbs)

****

Ku Band S/C #2

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	399	3.1
Auxiliary Propulsion Subsytem	203.1	2.21
Telemetry, Tracking and Command	26.4	.02
Electrical Power Subsystem	78.04553	.3632764
Thermal Control Subsystem	5.550846	5.684232E-02
Equipment Bay Structure	103.6118	11.14723
TOTAL SYSTEM SUMMARY	815.7081	16.89735
	1798.637	(Lbs)

****

Ku Band S/C #3

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	405	3.1
Auxiliary Propulsion Subsytem	203.1	2.21
Telemetry, Tracking and Command	26.4	.02
Electrical Power Subsystem	75.46564	.3436316
Thermal Control Subsystem	4.810735	4.926337E-02
Equipment Bay Structure	104.2808	11.14587
TOTAL SYSTEM SUMMARY	819.0571	16.86876
	1806.021	(Lbs)

****

TABLE 55. - KU-BAND SIZING EXAMPLE

Ku-band Sizing Example

EPS SUMMARY

```
Power required from source = 1.320852 KW
Power at load, average = .76 KW
Orbital altitude = 35786 Km
Orbital period = 23.93445 hrs
Spacecraft lifetime requirement = 7 yrs
Total eclipse time per orbit= 1.156897 hrs
Solar array degradation factor due to radiation = .9538934
Solar array thermal adjustment factor = .8943101
Solar array cover slide weight factor = 0
Antenna Size = 1 m by 1 m


***** Power Generation Sizing *****

For Si Blanket Array
Area required = 9.847787 m2
Weight = 15.75646 Kg Volume = .0787823 m3


***** Shunt Regulator Sizing *****

Shunt Regulator Weight = 4.036412 Kg
 Shunt Regulator Volume = 4.576431E-03 m^3


***** Power Switching/Distribution Sizing *****

Power Switching Equip Wt. = 2.479833 Kg
Power Switching Equip Vol. = 2.811602E-03 m3

Distribution Weight; Source to Bus = .100516 Kg
Distribution Weight; Bus to Load = 3.203065E-02 Kg
Total Distribution Weight = .1325466 Kg
Total Switching and Distribution Weight = 2.61238
****** TOTAL SI SYSTEM WEIGHT W/O BATTERIES(kg) = 22.40525
****** TOTAL SI SYSTEM VOLUME W/O BATTERIES(M^3) = 8.617033E-02


***** Battery Sizing *****

For NiCd Batteries
Battery Capacity Required = 1051.725 Wh
Battery Weight - 29.81588 Kg Battery Volume = 1.489074E-02 m3


***** Battery Charger Sizing *****

For NiCd Batteries
Battery Charger Weight = .1675525 Kg  Battery Charger Volume = 1.899688E-04 m3

 TOTAL EPS MASS(kg) = 52.38868
 TOTAL EPS VOLUME(M^3) = .101251
```

TABLE 55. - CONTINUED

Ku-band Sizing Example

THERMAL CONTROL SUBSYSTEM SIZING SUMMARY

Maximum temperature(C)	55
Maximum heat radiated(W)	250
Radiative surface emissivity factor	.8
Radiative surface absorptivity factor	.2
Required radiator surface area(M^2)	.9473726

Thermal Control Subsystem mass(Kg)	4.625708	Lbs	10.19969
Thermal Control Subsystem area (M^2)	.9473726		
Thermal Control Subsystem Volume(M^3)	4.736863E-02		

Ku-band Sizing Example

EQUIPMENT BAY SUMMARY

Total mass of equipment bay(kg)	58.58634	Lbs	129.1829
Mass of deployment/stowage mechanisms(kg)	25		
Total mass(kg)	83.58633	(Lbs)	184.3079
Total volume of structure and mechanisms(m^3)	16.79034		

****

Ku-band Sizing Example

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	228	2.05
Auxiliary Propulsion Subsytem	203.1	2.22
Telemetry, Tracking and Command	26.4	.02
Electrical Power Subsystem	52.38868	.101251
Thermal Control Subsystem	4.625708	4.736863E-02
Equipment Bay Structure	83.58633	11.19354
TOTAL SYSTEM SUMMARY	598.1007	15.63218
	1318.812	(Lbs)

****

TABLE 55. - CONCLUDED

Ku-band Sizing Example

RF SUBSYSTEM SIZING SUMMARY
Data for Honeycomb Panel Array (Non-phased)

ANTENNA APERTURE AREA(M^2)	0
TOTAL ANTENNA STRUCTURE MASS(KG)	0
TOTAL ANTENNA STRUCTURE VOLUME(M^3)	0
WEIGHT OF UPLINK COMPONENTS(Kg)	119
VOLUME OF UPLINK COMPONENTS(M^3)	2
WEIGHT OF SIGNAL PROC. COMPONENTS(Kg)	109
VOLUME OF SIGNAL PROC. COMPONENTS(M^3)	.05
TOTAL RF SUBSYTEM MASS(KG)	228
TOTAL RF SUBSYSTEM VOLUME(M^3)	2.05

Ku-band Sizing Example

ATTITUDE CONTROL, STATIONKEEPING AND MANEUVERING SUMMARY

Total ACS subsystem mass(Kg)	13.1	Lbs	28.8855
Total ACS subsystem volume(M^3)	.02		
Total RCS subsystem mass(Kg)	190	Lbs	418.95
Total RCS subsystem volume(M^3)	2.2		
Total RCS/ACS subsystem mass(Kg)	203.1		
Total RCS/ACS subsystem volume(M^3)	2.22		

Ku-band Sizing Example

TT&C SUBSYSTEM SUMMARY

Total TT&C subsystem mass(kg)	26.4	Lbs	58.212
Total TT&C subsystem volume(M^3)	.02		

5.1.2 Ku-Band Coverage Analysis

To meet the coverage requirements in the SOW, the three satellites were placed in unique geostationary orbit slots. Figure 78 shows the location of each satellite. It was assumed that each satellite would carry multiple antennas targeted to the center of each zone to be covered. The first satellite was placed at 70° W. longitude and provided coverage for Zones 1, 2, and 3. The next satellite was placed at 15° E. longitude and provided coverage for Zones 4 through 9. The final satellite was placed at 110° E. longitude and provided coverage for Zones 10 through 15. To assure coverage up to 70° latitude, (to cover Zones 9, 10, 12 and 14) a minimum satellite elevation angle of 11.5° was required. The three geostationary satellites can provide 24 hour continuous service to each zone. Eclipse effects at geostationary orbit will be maximum at spring and fall equinox resulting in slightly more than one hour of eclipse time when the satellites could not transmit if battery storage had not been provided.

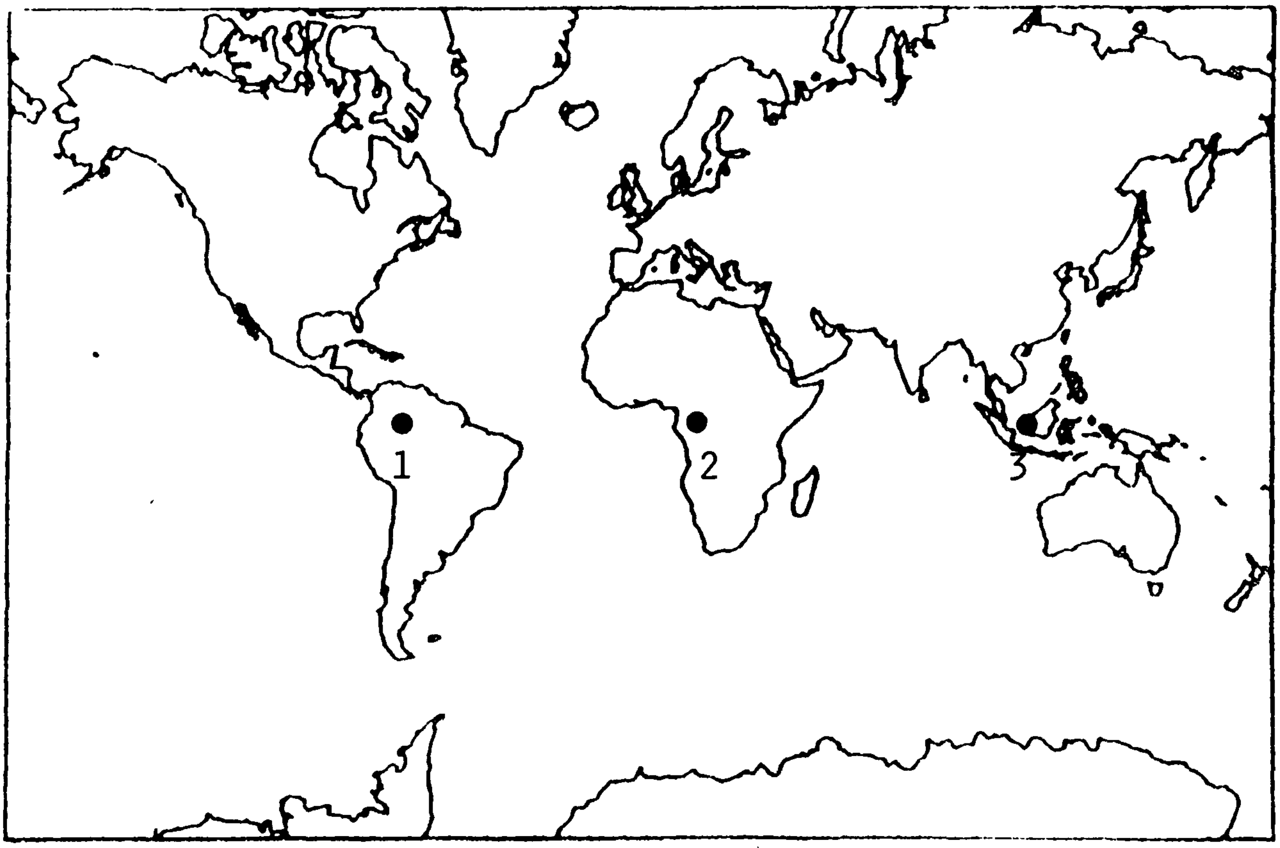

Figure 78. - Constellation for Ku-band—geostationary, three satellites.

5.1.3 Ku-Band System RF Performance Analysis

For each zone covered, an RF performance analysis was performed to determine the antenna size and power requirements needed to provide a PFD of -131 dBW/m^2 at edge of coverage. The RF analysis took the servicing satellite location and zone size into account when determining antenna aperture size. In addition, Ku-band reflector antennas were assumed to have a beamwidth equal to:

$$BW = \frac{70\lambda}{D} \qquad (5\text{-}1)$$

where

BW = beamwidth, degree
λ = wavelength, m
D = reflector diameter, m

To determine the PFD on the ground, a 3-dB polarization loss was assumed for circular polarization transmission and a linear polarized receiver. Spreading loss was calculated using the maximum slant range required for covering the zone (the slant range being the distance from the satellite to the ground). The maximum slant range occurs at the same point on the ground as the minimum spacecraft elevation angle.

A computer program (RFANAL - ref. the Appendix) was written to perform these calculations quickly and efficiently. Table 56 shows an example of the program output for Zone 9. The propagation loss of 0 dB used in this analysis is discussed in Section 3.2. The aperture efficiency of 0.5 was used to account for feed mismatch and circular polarization conversion loss, spillover loss, and losses due to reflector surface accuracy. The power is shown in watts required per channel for an EOC PFD of -131 dBW/m^2.

After each zones' aperture and power requirement were determined, the zones common to a single satellite were combined to determine the satellite's peak power requirement by combining the service requirements (showing how many channels for each zone are at any given UTC time were required to be transmitted) with the RF power required per zone per channel.

The results are shown in Table 57. Of the three geostationary satellites, the highest power requirement is 159 watts occuring at UTC 415. This was the result of simultaneously requiring 10 channels to operate in Zone 5, two channels in Zone 6, four channels in Zone 7, one channel in Zone 8 and six channels in Zone 9.

TABLE 56. - RF ANALYSIS FOR KU-BAND, ZONE 9

Zone covered	Zone 9 GEO Ku-band at 35786
Orbital altitude	35786 km
Operating frequency	12200 MHz
Satellite location	0 Lat. 15 Long.
Zone size	37x70 Lat., 20x53 Long.
Polarization—circular	
Aperture size	.659 m x .334 m using Tapered illumination
Antenna EOC gain	29.5 dBi
Antenna aperture efficiency	50%
Antenna beamwidth	2.613 x 5.146°
Ground coverage	3673.46 x 3673.46 km, 33 x 33°
Antenna scan angle	0
Polarization margin	3 dB
Propagation loss	0 dB
Min elevation angle	11.475
Spreading loss	163.13 dB
Slant range	40429.3 km
Power reqd/channel for EOC PFD of -131 dBW/m^2	3.61 W
EOC EIRP at EOC PFD of -131 dBW/m^2	35.13 dBW

TABLE 57 - KU-BAND ANTENNA DESIGN SUMMARY

Satellite	Zones	Aperture size	No. of channels	Max power per aperture
Satellite worst case design power at UTC 45 = 78 W				
1	1	0.49x0.32 m	2	9.7 W
	2	0 42x0.26	2	12.2
	3	0.17x0.17	2	56.0
Satellite worst case design power at UTC 415 = 159 W				
2	4	0.37x0.25	2	15.9
	5	0.55x0.24	11	66.4
	6	0.66x0.23	3	13.5
	7	0.18x0.24	4	62.6
	8	0.35x0.36	2	11.1
	9	0.66x0.33	6	21.7
Satellite worst case design power at UTC 1100 = 129 W				
3	10	0.43x0.37	3	14.8
	11	0.33x0.30	4	30.7
	12	0.85x0.23	2	8.2
	13	0.18x0 22	6	103.5
	14	0.77x0.34	2	6.1
	15	0.18x0 33	1	12.7

5.1.4 Ku-Band System LCC Estimates

TABLE 58. - KU-BAND SATELLITES LCC ESTIMATES, 1984 $

System Satellite	S/C NRC	Recurring cost	Launch cost	Ground* cost	Total cost
1	$76M	$174M	$213M	$17M	$480M
2	8	165	213	17	403
3	10	120	210	17	357
	94	459	636	51	1240

*20 years operational lifetime

The Ku-band three-satellite system LCCs are summarized in Table 58. Figure 79 shows the total cumulative cost as a linear function, and yearly cost, as a bar graph, for a 24-yr life cycle. Assuming four launches per year combined with a 7-yr satellite lifetime, three sets of launches are required starting after a 3-yr development program. This approach provides a 21-yr, full coverage Ku-band capability. The costs shown during the periods between launches include only the ground stations operating cost ($18M per seven years). The different recurring cost for each satellite results from different sized apertures, different power requirements, and different numbers of downlink apertures for satellite number one (three, vs six for satellites one and two). The launch costs, determined from payload weight, are similar since most of the STS payload is due to the TOS/AMS upper stage. Two ground stations are assumed for the system, although an intersatellite link using the middle satellite would permit real-time communication from a single ground station. The reduced cost for the ground stations would be offset by increased nonrecurring and recurring costs for the middle satellite. No learning factor was assumed in deriving the total cost.

Since the Ku-band satellites are capable of operation even during the occulted period of the orbit, the LCC per channel hour is easily computed from total LCC divided by total channel hours.

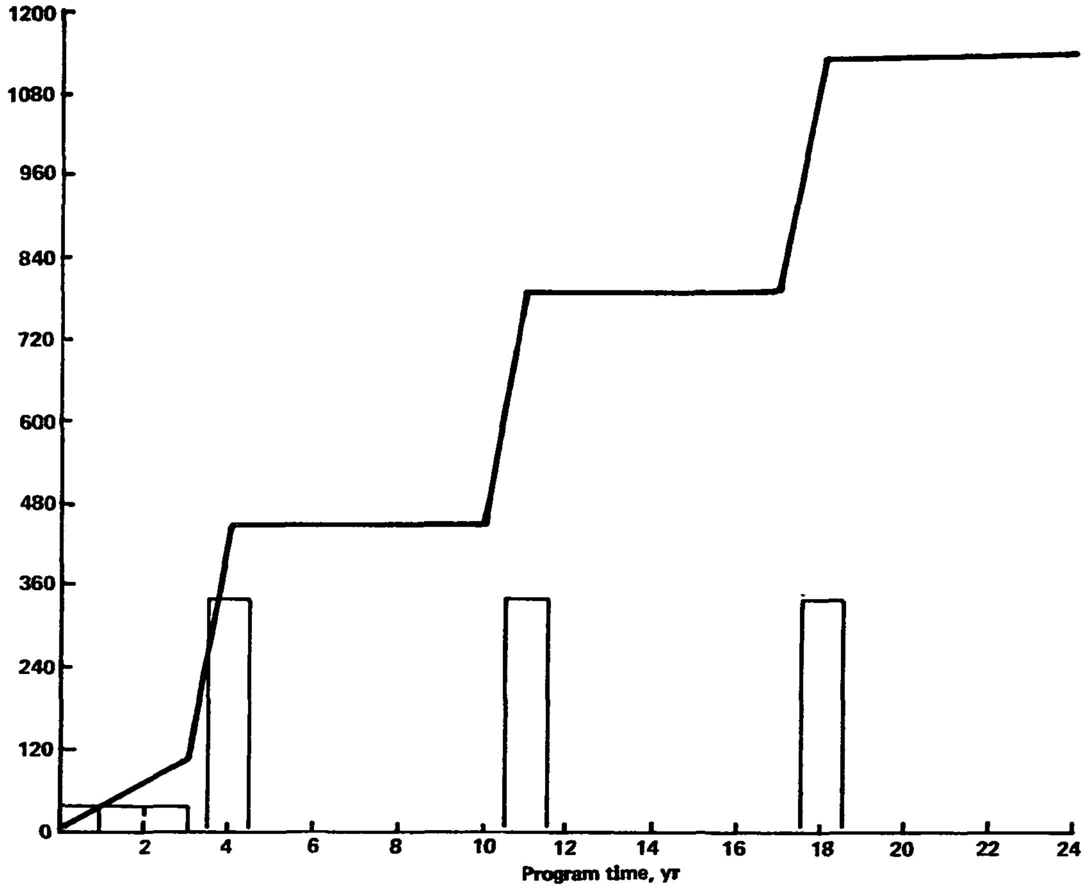

Figure 79. - Ku-band cumulative and yearly program costs, 1984 $M.

5.1.5 Ku-Band Summary of Results

The Ku-band system designed for VOA applications consisted of three satellites in geostationary orbit. Each satellite carried multiple parabolic dish antennas targeted to specific zones. This resulted in a 100% match in coverage with the VOA requirements. The frequency of operation for the Ku-band system was 12.2 GHz with sufficient RF power in each satellite to provide a power flux density of -131 dBW/m^2 at the edge of coverage. The orbit transfer stage used to deliver each satellite into orbit was the TOS/AMS, resulting in a payload capability to orbit of 3113 kg. This provided sufficient growth margin for satellites mass ranging from 618 to 819 kg.

Including a 3-yr development schedule, three sets of launches result in a 24-yr life cycle for satellite lifetimes of seven years. Total estimated cost for three sets of satellites is $1,189M. Ground operations cost for 20 years would be $51M. Thus, a 20-yr operational capability would cost $1,240M. Twenty years of operation would provide 2,184,523 channel hours resulting in a cost of $568 per channel hour. Since the system life is 21 years, one year of productive service is left after the 20-yr program.

5.2 L-BAND SYSTEM

For L-band operation, the satellite system design and analysis resulted in a single point design at the high RF power requirement and three system options at the low RF power requirement. Each system design meets all VOA requirements with the high RF power system designed to meet the PFD requirement of -103.6 dBW/m^2 and the low RF power systems designed to meet the PFD requirement of -116.1 dBW/m^2. All designs use existing state-of-the-art technologies for communications, ACS, stationkeeping, TT&C, thermal control, and spacecraft structure.

Although the electrical power subsystem uses existing technology, the size of the system is larger than present day satellites and will require projected improvements in the state of the art, e.g, those demonstrated in the solar array flight experiment (SAFE). The communication subsystem is made up of rigid honeycomb panel nonscanning arrays and solid state transmitters. This technology is similar to the communication subsystem found on the Seasat-I Spacecraft, Section 3.4.1.2.

The high RF power L-band system consists of eight satellites. Each satellite consists of a farm of rigid panel arrays attached to a central spacecraft bus. The number of arrays per satellite varies depending on the number of zones covered, with each array illuminating a single zone. Figure 80 illustrates the proposed satellite design for the high RF power system.

The first option, Option I, for the low RF power L-band system consists of five satellites. Again each satellite consists of a farm of rigid panel arrays and each array illuminates a single zone. The next two low RF power options consist of three satellites each. The first, Option II, has individual arrays per zone, while the second, Option III, has a single array for multiple zones. Option II of the three satellite systems has a lower power requirement than Option III but is heavier due to the number of arrays on each satellite.

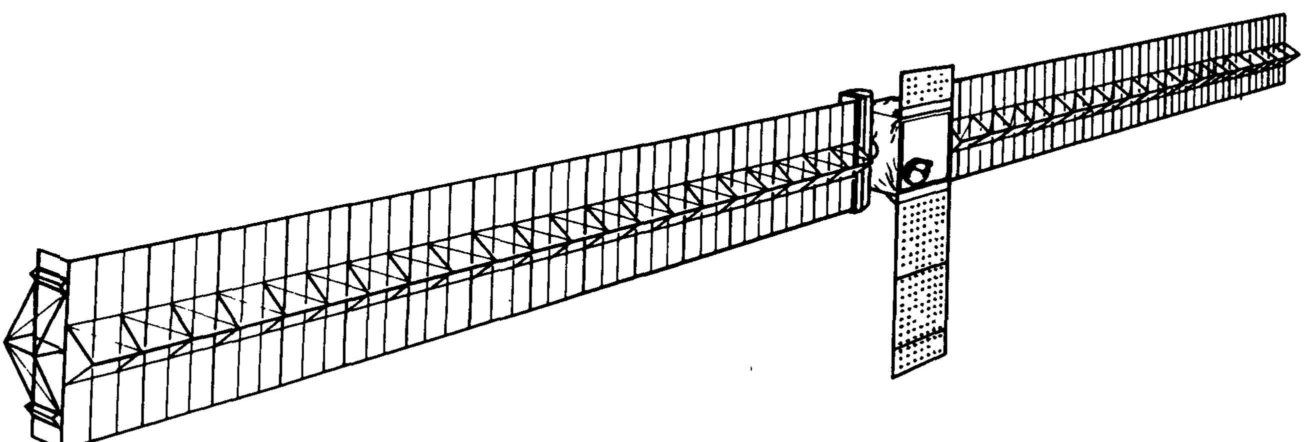

Figure 80. - L-band satellite design.

5.2.1 L-Band Systems Weight and Volume Estimates

Weights and volumes for the four L-band systems are summarized in Table 59. For sizing the electrical distribution weight from the bus to the load, the total area of all array antennas was computed and an equivalent area defined for the antenna diameter input to the EPS sizing program. Table 60 shows the sizing program results for satellite two of the high power system.

TABLE 59. - L-BAND SATELLITES WEIGHTS AND VOLUMES

System	Satellite number	Weight	Volume
High-power requirement design (-103.6 dBW/m^2) (eight satellites required)	1	3914, kg	26.3
	2	2444	23.3
	3	3710	24.5
	4	2981	21.7
	5	2821	22.4
	6	3373	23.2
	7	4249	28.4
	8	2651	9.4
Low-power requirement *option I (-116.1 dBW/m^2)* (five satellites required)	1	1323	18.0
	2	1472	19.0
	3	1460	19.2
	4	1707	22.2
	5	1176	5.2
Option II (-116.1 dBW/m^2) (three satellites required)	1	1252	18.0
	2	2150	23.8
	3	2150	24.2
Option III (-116.1 dBW/m^2) (three satellites required)	1	1065	16.2
	2	1893	18.5
	3	1835	17.8

Note:

Satellite locations by satellite number are shown in Figures 81, 82 & 83.

TABLE 60. - L-BAND SIZING EXAMPLE

L-band Sizing Example
S/C #2 @ -103.6 dB

RF SUBSYSTEM SIZING SUMMARY
Data for Honeycomb Panel Array (Non-phased)
Data for antenna no. 1
Antenna aperture (M) 2.8 x 1.9

Power per transmitter(w)	Number of transmitters	Output power(kw)
29.16667	216	6.3

ANTENNA APERTURE AREA(M^2)	5.32
TOTAL ANTENNA STRUCTURE MASS(KG)	63.6227
TOTAL ANTENNA STRUCTURE VOLUME(M^3)	.266
TOTAL RF TRANSMITTER MASS(KG)	672.4055
TOTAL RF TRANSMITTER VOLUME(M^3)	1.621494

Data for Honeycomb Panel Array (Non-phased)
Data for antenna no. 2
Antenna aperture (M) 4.1 x 1.8

Power per transmitter(w)	Number of transmitters	Output power(kw)
37.03704	324	12

ANTENNA APERTURE AREA(M^2)	7.38
TOTAL ANTENNA STRUCTURE MASS(KG)	84.60817
TOTAL ANTENNA STRUCTURE VOLUME(M^3)	.369
TOTAL RF TRANSMITTER MASS(KG)	1258.432
TOTAL RF TRANSMITTER VOLUME(M^3)	2.799696
WEIGHT OF UPLINK COMPONENTS(Kg)	78
VOLUME OF UPLINK COMPONENTS(M^3)	.5
WEIGHT OF SIGNAL PROC. COMPONENTS(Kg)	87
VOLUME OF SIGNAL PROC. COMPONENTS(M^3)	.03
TOTAL RF SUBSYTEM MASS(KG)	2244.068
TOTAL RF SUBSYSTEM VOLUME(M^3)	5.58619

TABLE 60. - CONTINUED

L-band Sizing Example
S/C #2 @ -103.6 dB

EPS SUMMARY

Power required from source = 30.0774 KW
Power at load, average = 28.65385 KW
Orbital altitude = 35786 Km
Orbital period = 23.93445 hrs
Spacecraft lifetime requirement = 7 yrs
Total eclipse time per orbit= 1.156897 hrs
Solar array degradation factor due to radiation = .9538934
Solar array thermal adjustment factor = .8943101
Solar array cover slide weight factor = 0
Antenna Size = 3.065372 m by 3.065372 m

***** Power Generation Sizing *****

For Si Blanket Array
Area required = 224.2461 m2
Weight = 358.7937 Kg Volume = 1.793969 m3

***** Shunt Regulator Sizing *****

Shunt Regulator Weight = 51.63594 Kg
Shunt Regulator Volume = 5.854415E-02 m^3

***** Power Switching/Distribution Sizing *****

Power Switching Equip Wt. = 9.007568 Kg
Power Switching Equip Vol. = 1.021266E-02 m3

Distribution Weight; Source to Bus = 10.92231 Kg
Distribution Weight; Bus to Load = 2.235811 Kg
Total Distribution Weight = 13.15812 Kg
Total Switching and Distribution Weight = 22.16568
****** TOTAL SI SYSTEM WEIGHT W/O BATTERIES(kg) = 432.5953
****** TOTAL SI SYSTEM VOLUME W/O BATTERIES(M^3) = 1.862725

***** Battery Sizing *****

For NiCd Batteries
Battery Capacity Required = 1051.725 Wh
Battery Weight = 29.81588 Kg Battery Volume = 1.489074E-02 m3

***** Battery Charger Sizing *****

For NiCd Batteries
Battery Charger Weight = .1675525 Kg Battery Charger Volume = 1.899688E-04 m3

TOTAL EPS MASS(kg) = 462.5787
TOTAL EPS VOLUME(M^3) = 1.877806

TABLE 60. - CONTINUED

L-band Sizing Example
S/C #2 @ -103.6 dB

ATTITUDE CONTROL, STATIONKEEPING AND MANEUVERING SUMMARY

Total ACS subsystem mass(Kg)	57.7	Lbs	127.2285
Total ACS subsystem volume(M^3)	.05		
Total RCS subsystem mass(Kg)	304	Lbs	670.3201
Total RCS subsystem volume(M^3)	2.2		
Total RCS/ACS subsystem mass(Kg)	361.7		
Total RCS/ACS subsystem volume(M^3)	2.25		

L-band Sizing Example
S/C #2 @ -103.6 dB

TT&C SUBSYSTEM SUMMARY

Total TT&C subsystem mass(kg)	32.1	Lbs	70.7805
Total TT&C subsystem volume(M^3)	.03		

L-band Sizing Example
S/C #2 @ -103.6 dB

THERMAL CONTROL SUBSYSTEM SIZING SUMMARY

Maximum temperature(C)	55		
Maximum heat radiated(W)	9855.269		
Radiative surface emissivity factor	.8		
Radiative surface absorptivity factor	.2		
Required radiator surface area(M^2)	37.34631		
Thermal Control Subsystem mass(Kg)	182.3497	Lbs	402.0811
Thermal Control Subsystem area (M^2)	37.34631		
Thermal Control Subsystem Volume(M^3)	1.867315		

L-band Sizing Example
S/C #2 @ -103.6 dB

EQUIPMENT BAY SUMMARY

Total mass of equipment bay(kg)	78.24596	Lbs	172.5324
Mass of deployment/stowage mechanisms(kg)	102		
Total mass(kg)	180.246	(Lbs)	397.4424
Total volume of structure and mechanisms(m^3)	17.49899		

TABLE 60. - CONCLUDED

****

L-band Sizing Example
S/C #2 @ -103.6 dB

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	2244.068	5.58619
Auxiliary Propulsion Subsytem	361.7	2.25
Telemetry, Tracking and Command	32.1	.03
Electrical Power Subsystem	462.5787	1.877806
Thermal Control Subsystem	182.3497	1.867315
Equipment Bay Structure	180.246	11.66599
TOTAL SYSTEM SUMMARY	3463.042	23.27731
	7636.009	(Lbs)

5.2.2 L-Band System Coverage Analysis

To meet the VOA coverage requirements, satellites for each L-band system were placed in three geostationary orbit slots. Depending on the system, the number of satellites in any one slot varied from four for the high RF power system to one for the low RF power system. Figures 81, 82, and 83 show the satellites locations for each system. The system designs call for each satellite to carry several array antennas with each antenna targeted to the center of a specific area to be covered. In the Option III design, the covered area consisted of multiple zones.

The first constellation of satellites was placed at 70° W. longitude and provided coverage for Zones 1, 2 and 3. The next satellite constellation was placed at 15° E. longitude and provided coverage for Zones 4 through 9. The final satellite constellation was placed at 110° E. longitude and provided coverage for Zones 10 through 15. To assure coverage up to 70° latitude for Zones 9, 10, 12 and 14, a minimum satellite elevation angle of 11.5° was required. The VOA coverage requirements were matched 100%.

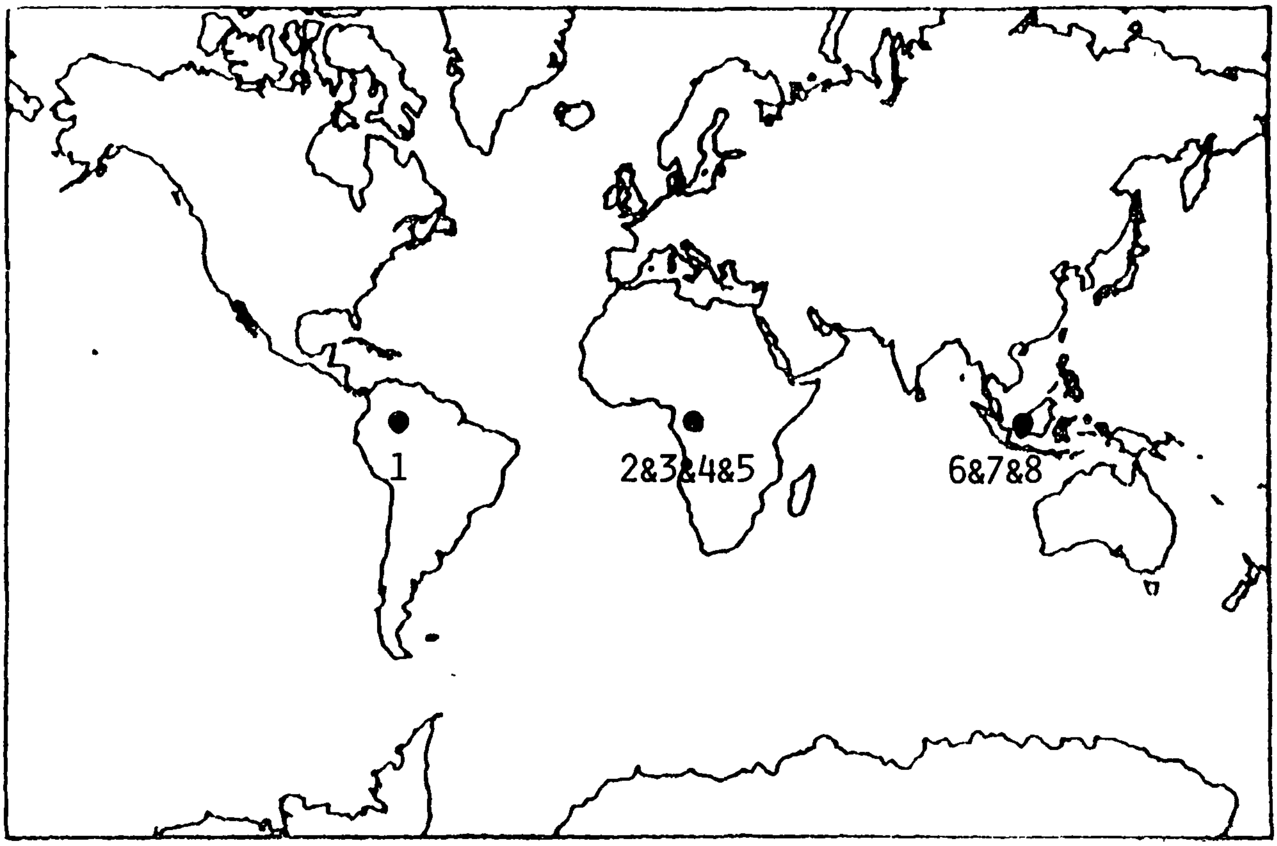

Figure 81. - Constellation for high RF power L-band system—geostationary, eight satellites.

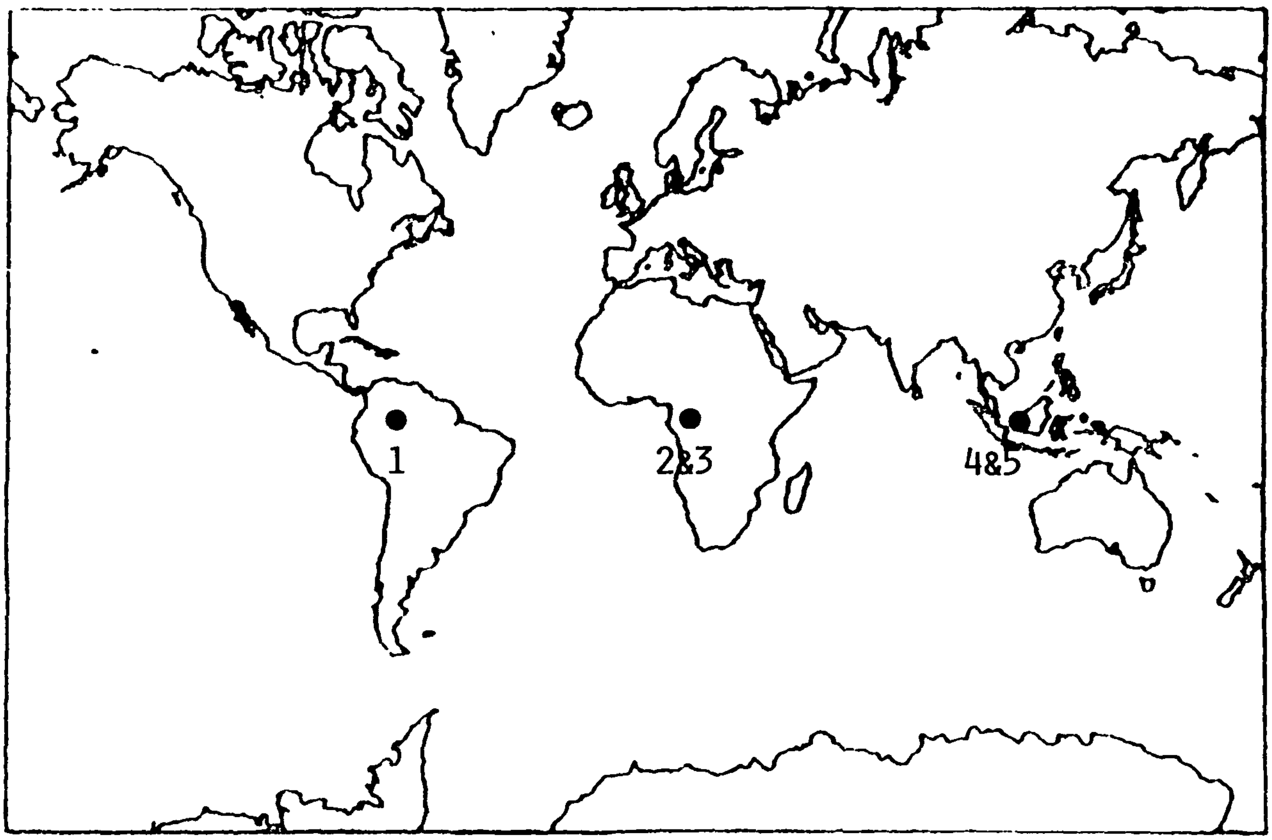

Figure 82. - Constellation for low RF power option I L-band system—geostationary, five satellites.

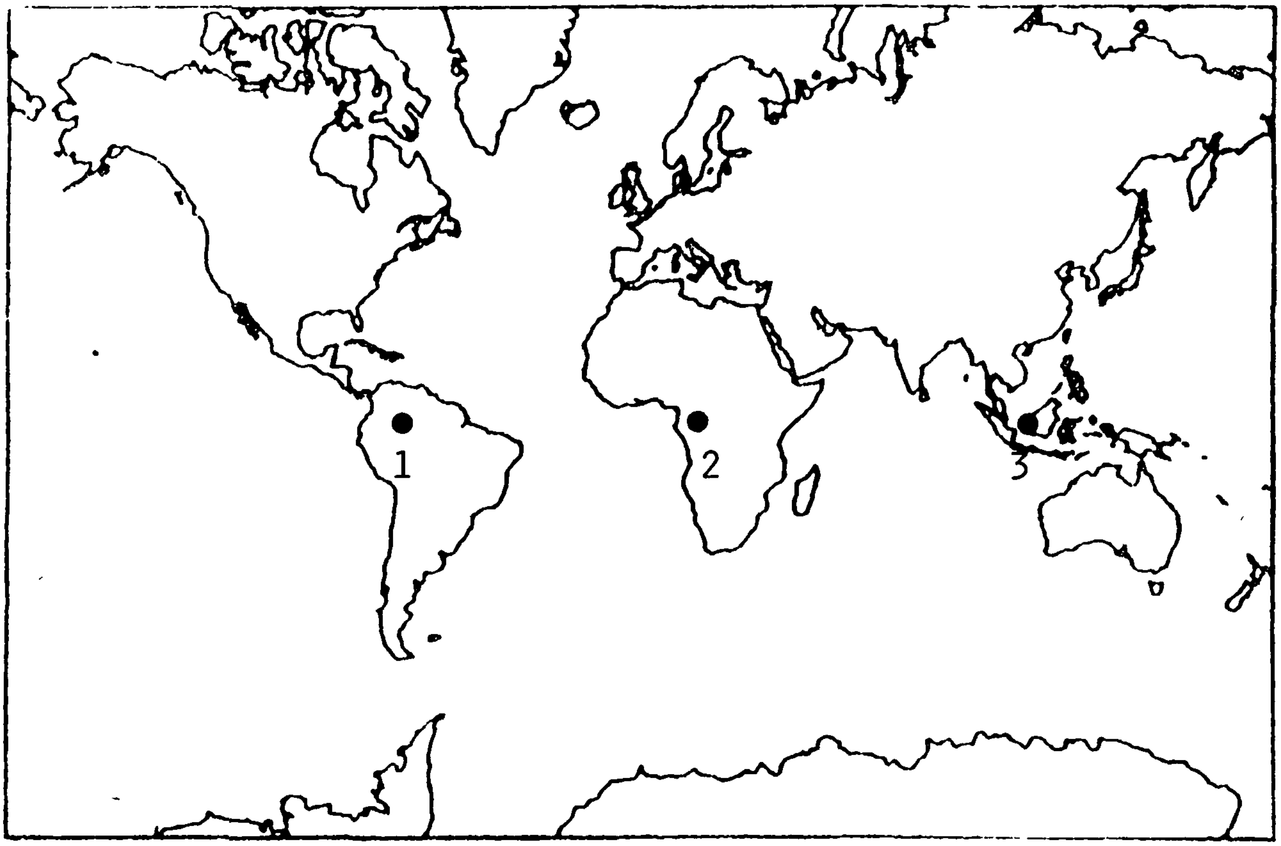

Figure 83. - Constellation for low RF power options II and III L-band system—geostationary, three satellites.

5.2.3 L-Band System RF Performance Analysis

For each L-band system, an RF performance analysis was performed to determine the antenna size and RF power requirements needed to provide the required PFD at edge of coverage. The RF analysis took the servicing satellite location and ground area to be covered into account when determining antenna aperture size. In addition, L-band nonscanning arrays were assumed to have a beamwidth equal to:

$$BW = \frac{65\lambda}{D} \tag{5-2}$$

where

BW = beamwidth, degree
λ = wavelength, m
and D = reflector diameter, m

To determine the PFD on the ground, a 3-dB polarization loss was assumed for circular polarization transmission and linear polarized reception. Also, the spreading loss was calculated using the maximum slant range required for ground coverage. The coverage area on the ground was equal to the zone covered for all but the last system, in which multiple zones were covered with a single aperture. A computer program was written to allow these calculations to be done quickly and efficiently. Table 61 shows an example of the program results for the high RF power system at Zone 9. Table 62 shows an example of the program results for Option III, the multiple zone per aperture case. The propagation loss of 0.0 dB used in the analysis is discussed in Section 3.2. The aperture efficiency of 0.8 was used to account for losses associated with array mismatch, amplitude and phase control of array elements and circular polarization conversion. The power per channel for the required PFD level is shown in watts.

TABLE 61. - RF ANALYSIS FOR HIGH POWER L-BAND, ZONE 9

Zone covered	Zone 9 GEO L-Band at 35786
Orbital altitude	35786 km
Operating frequency	1500 MHz
Satellite location	0 Lat 15 Long.
Zone size	37x70 Lat, 20x53 Long.
Polarization	Circular
Aperture size	4.974 x 2.53 m using UNIFORM illumination
Antenna EOC gain	30.94 dBi
Antenna aperture efficiency	80%
Antenna beamwidth	2 613 x 5.146°
Ground coverage	3673.46 x 3673.46 km, 33 x 33°
Antenna scan angle	0
Polarization margin	3 dB
Propagation loss	0 dB
Min elevation angle	11 47
Spreading loss	163.126 dB
Slant range	40429.3 km
Power reqd/channel for EOC PFD of -103.6 dBW/m^2	1439.191 W
Power reqd/channel for EOC PFD of -116.1 dBW/m^2	80 93 W
EOC EIRP at EOC PFD of -103.6 dBW/m^2	62.53 dBW
EOC EIRP at EOC PFD of -116.1 dBW/m^2	50.03 dBW

TABLE 62. - RF ANALYSIS FOR LOW RF POWER L-BAND, OPTION III ZONES 4, 6, 7, AND 8 COMBINED

Zone covered	Zone 4, 6, 7, & 8 GEO L-Band at 35786
Orbital altitude	35786 km
Operating frequency	1500 MHz
Satellite location	0 Lat 15 Long
Zone size	-35x44 Latitude, 40x60 Longitude
Polarization	Circular
Aperture size	1.05 m x .893 m using UNIFORM illumination
Antenna EOC gain	19.67 dBi
Antenna aperture efficiency	80%
Antenna beamwidth	12.38 x 14 556°
Ground coverage	8794.05 x 11131.71 km, 79 x 100°
Antenna scan angle	0
Polarization margin	3 dB
Propagation loss	0 dB
Min elevation angle	27.27
Spreading loss	162 78 dB
Slant range	38858.56 km
Power reqd/channel for EOC PFD of -103.6 dBM/m^2	17814.22 W
Power reqd/channel for EOC PFD of -116.1 dBM/m^2	10001.767 W
EOC EIRP at EOC PFD of -103.6 dBW/m^2	62.18 dBW
EOC EIRP at EOC PFD of -116 1 dBW/m^2	46.68 dBW

After each zones' aperture and power requirement were determined, the zones common to a single satellite were combined to determine the satellites peak power requirement. This was accomplished by combining the service requirements (showing how many channels for each zone at any given UTC time were required to be transmitted) with the RF power required per zone per channel. The results are shown in Tables 63 through 66. Table 63 gives the results of the high RF power L-band system made up of eight satellites. Table 64 gives the results of the five satellite low RF power L-band system, Option I. Tables 65 and 66 are the results of the three satellite low RF power L-band systems, Options II and III.

TABLE 63. - HIGH RF POWER L-BAND ANTENNA DESIGN SUMMARY

S/C*	Zone	Aperture size	No. of channels	RF output power/zone	Maximum S/C RF power
1	1 2 3	3.7x2.4 m 3.2x1.9 1.25x1.24	2 2 2	3.8 kW 4.8 22 3	31 0 kW-UTC 45
2	4 5	2.8x 1.9 4 1x1 8	2 5	6.3 12 0	18 3-UTC 530
3	5 6	4.1x1 8 5.0x1.7	6 3	14.4 5.4	19 8-UTC 530
4	7	1.4x1 8	4	24.9	24.9-UTC 415
5	8 9	2 6x2 7 5.0x2.5	2 6	4.4 8.6	11.6-UTC 430
6	10 11	3 3x2.8 2.5x2.3	3 4	5.9 12.2	16.2-UTC 300
7	12 14 15 13	6.4x1.7 5.8x2.6 1.4x2.5 1.4x1.7	2 2 1 3	3 3 2.4 5.1 20.6	25.7-UTC 1100
8	13	1.4x1 7	3	20.6	20.6-UTC 1100

*Satellite mass range-2651 to 4349 kg

TABLE 64. - LOW RF POWER L-BAND ANTENNA DESIGN SUMMARY—OPTION I

S/C*	Zone	Aperture size	No. of channels	RF output power/zone	Maximum S/C RF power
1	1 2 3	3.7x2 4 m 3.2x1 9 1.25x1.24	2 2 2	0.22 kW 0.27 1.25	1.75 kW-UTC 45
2	4 5 6	2.8x1.9 4.1x1 8 5.0x1 7	2 11 3	0 36 1.50 0.30	2.14-UTC530
3	7 8 9	1.4x1.8 2.6x2.7 5 0x2.5	4 2 6	1.40 0.25 0.49	2 0-UTC 415
4	10 11 12 14	3.3x2 8 2.5x2 3 6.4x1.7 5.8x2.5	3 4 2 2	0.33 0.69 0.18 0.14	0 9-UTC 300
5	13 15	1 4x1 7 1.4x2.5	6 1	2.3 0.28	2 6-UTC 1100

*Satellite mass range—1176 to 1707 kg

TABLE 65. - HIGH RF POWER L-BAND ANTENNA DESIGN SUMMARY—OPTION II

S/C*	Zone	Aperture size	No of channels	RF output power/zone	Maximum S/C RF power
1	1 2 3	3.7x2.4 m 3.2x1.9 1.25x1 24	2 2 2	0.22 kW 0.27 1.25	1 75 kW-UTC 45
2	4 5 6 7 8 9	2.8x1.9 4.1x1.8 5.0x1.7 1.4x1.8 2.6x2.7 5.0x2 5	2 11 3 4 2 6	0.36 1.50 0 30 1.40 0.25 0.49	3.4-UTC 430
3	10 11 12 13 14 15	3.2x2.8 2.5x2.3 6.4x1.7 1.4x1.7 5.8x2.5 1.4x2.5	3 4 2 6 2 1	0.33 0.69 0.18 2.30 0.14 0.28	2.9-UTC 1100

*Satellite mass range—1251 to 2150 kg

TABLE 66. - LOW RF POWER L-BAND ANTENNA DESIGN SUMMARY—OPTION III

S/C*	Zone	Aperture size	RF output power/aperture	Maximum S/C RF power
1	1&2 3	3.75x1.2 m 1.25x1.24	0.84 kW 1.25	2.1 kW-UTC 45
2	5&9 4&6&7&8	4.15x1.23 1.05x0.89	3.2 9.0	10.2-UTC 430
3	10&11 12&13&14&15	1.7x2.3 1.2x1.1	1.57 7.2	7.5-UTC 1100

*Satellite mass range—1066 to 1893 kg

5.2.4 L-Band Systems LCC Estimates

Each satellite of the L-band systems is generically the same. However, the unique coverage required from each results in different aperture sizes and different subsystem sizes. To estimate system LCC, each satellite's nonrecurring and recurring costs were estimated first. The system nonrecurring development cost was then obtained by comparing the subsystem development costs for each satellite, selecting the maximum for each subsystem and summing the maximums. The individual satellites' recurring costs were summed to estimate total system recurring cost. Because the satellites within each system have different antenna designs, different electrical power requirements, and different thermal control requirements, no learning factor adjustment for cost of fabrication was assumed. The LCC estimates for the L-band systems are summarized in Table 67.

TABLE 67. - L-BAND SYSTEMS' LCC ESTIMATES, 1984 $

System*	NRC	Ground costs*	RC	Launch costs	Total LCC
High power (-103 6 dBW/m^2)	130	$51M	1968	$3000M	$5149M
Option I (-116.1 dBW/m^2)	98	51	645	1095	1889
Option II (-116 1 dBW/m^2)	113	51	441	666	1273
Option III (-116.1 dBW/m^2)	135	51	504	663	1353

*20-yr operational lifetime

Figures 84 through 87 contain linear plots of total L-band system cost as a function of time, and bar graphs showing phased yearly costs for the four L-band concepts. The high-power option (Fig. 84) uses eight satellites, requiring six years for a full capability system (eight satellites), assuming four STS launches per year starting after a 4-yr development phase. This would provide full capability for 21 years, assuming 7-yr satellite lifetime. The other L-band options would also provide full capability for 21 years as shown on the graphs. The total L-band program length would be 25 years.

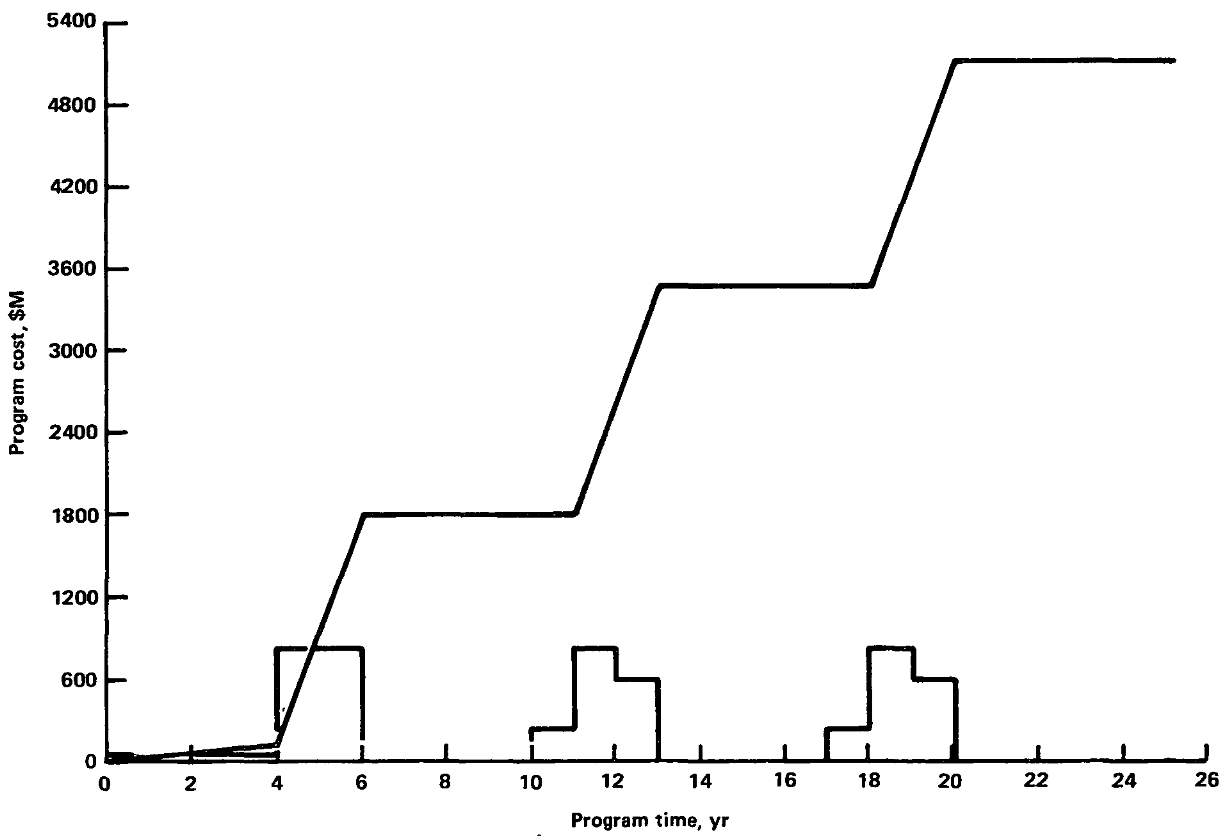

Figure 84. - L-band high power system program cumulative and yearly costs.

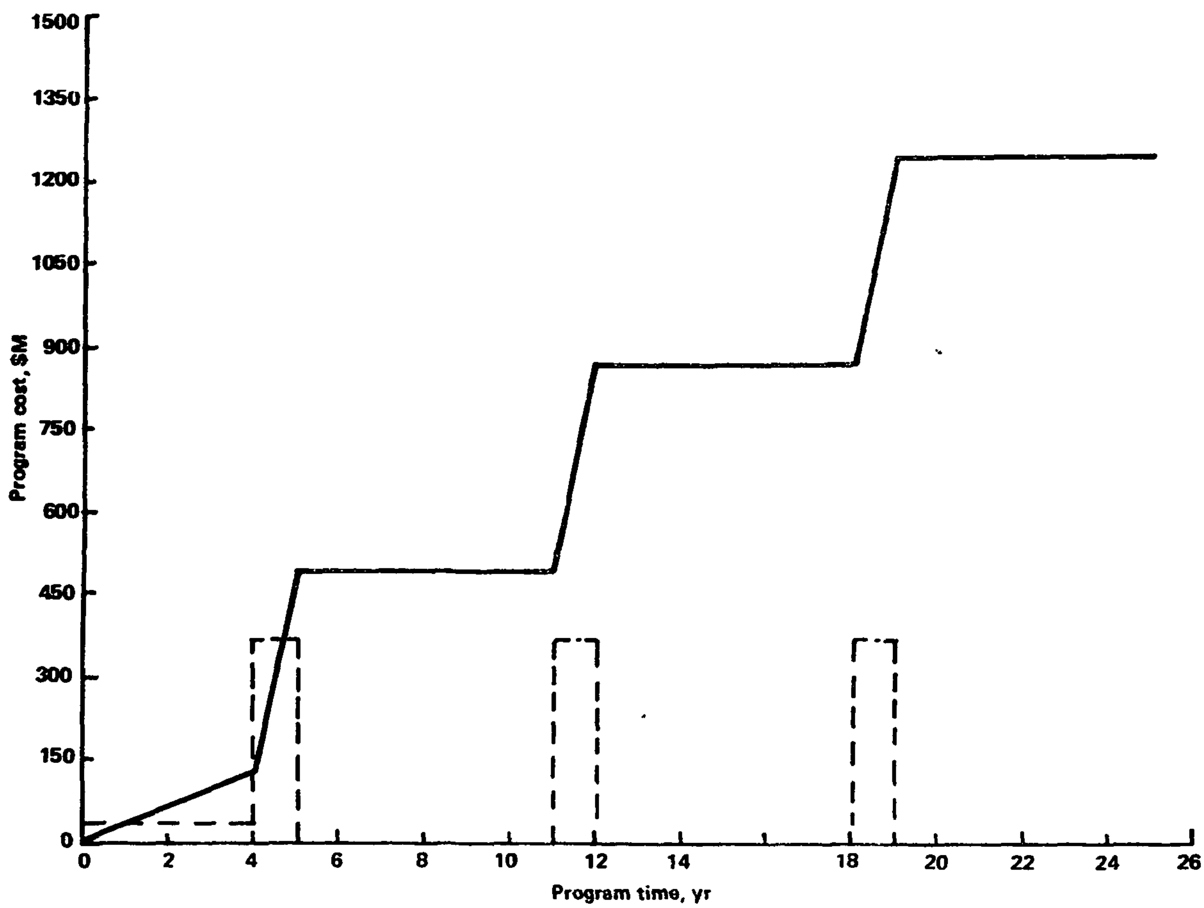

Figure 85. - L-band option I system program cumulative and yearly costs.

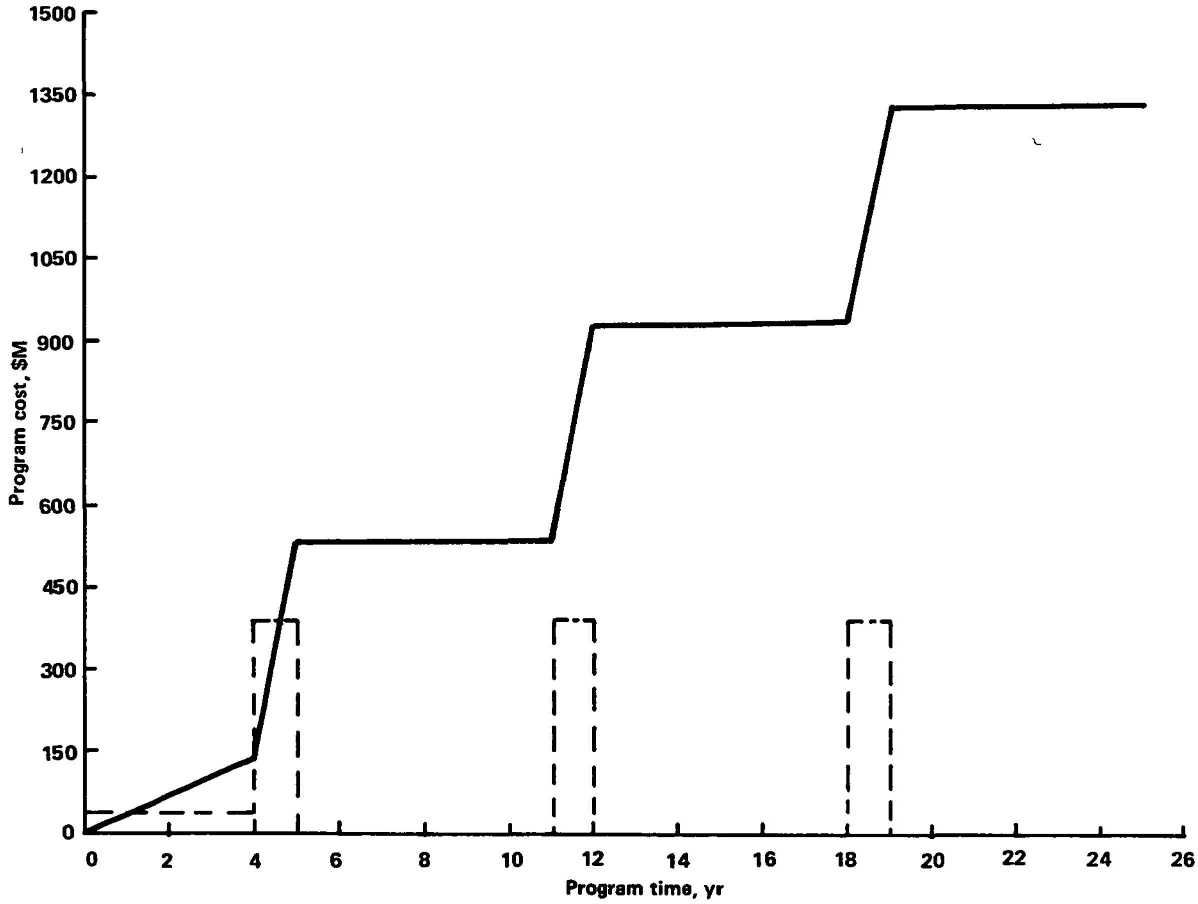

Figure 86. - L-band option II system program cumulative and yearly costs.

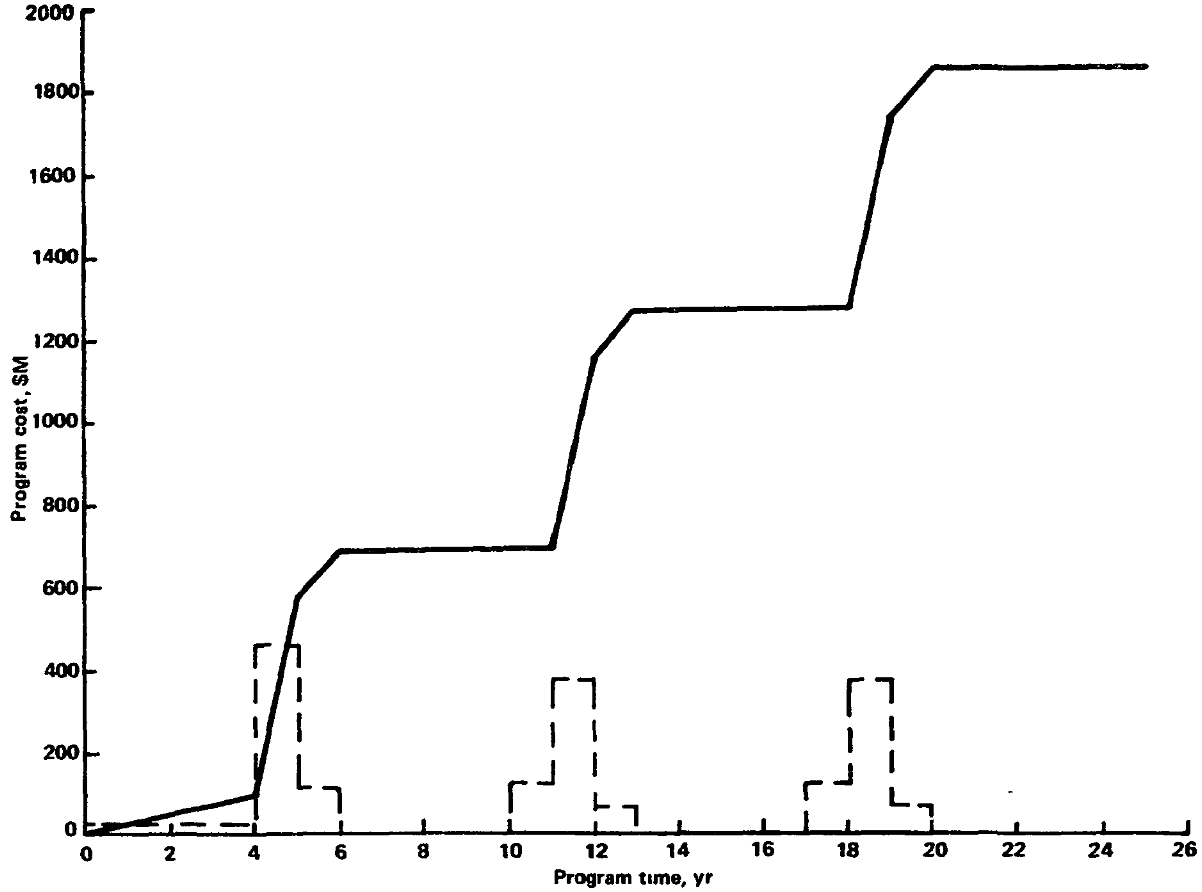

Figure 87. - L-band option III system program cumulative and yearly costs.

5.2.5 L-Band Summary of Results

The L-band systems designed for VOA applications consisted of four different system designs ranging from three to eight satellites in geostationary orbit. The first system design was for a high RF power level and required eight satellites in three geostationary orbit slots to achieve the required power and channel requirements. The next three system designs were for a low RF power level and require 5, 3 and 3 satellites. For the high RF power system and the first two low power systems, each satellite carried multiple nonscanning rigid panel array antennas targeted at each zone required to be covered. The last system design carried array antennas that could cover multiple zones. The frequency of operation for all the L-band systems was 1.5 GHz.

For the high RF power system, the orbit transfer stage used to deliver each satellite into orbit was the Centaur G, resulting in a payload capability to orbit of 4390 kg. The resulting satellite mass ranged from 2651 to 4349 kg. For the low RF power systems, the orbit transfer stage used was the TOS/AMS, resulting in a payload capability to orbit of 3113 kg. This provided a sufficient growth margin for the satellites mass which ranged from 1176 to 1707 kg, 1251 to 2150 kg and 1066 to 1893 kg for Option I, II, and III system designs respectively. The satellite power requirements range from 20 to 50 kW for the high RF power system and 2.5 to 17.2 kW for the low RF power systems.

For a 20-yr operational lifetime, ground operations would cost $51M. At the end of the 20-yr period, one year of full operating capability would be available with reduced capability available for up to three years depending on the system concept.

Total life-cycle cost for 20 years operation, including launch and maintenance costs, ranged from a high of $5149M for the high RF power system to a low of $1353M for the low RF power system, Option II. Tables 68 and 69 summarize the results for each L-band system design.

TABLE 68. - LOW RF POWER L-BAND SUMMARY OF RESULTS

L-band summary
– Geostationary orbit–three slots (70W, 15E, 110E) – Three and five satellite options – TOS/AMS orbit transfer vehicle (payload–3113 kg) – Array technology selected but reflectors could be considered due to lower power requirements. – Zones/channels–all/full VOA – Coverage efficiency–100%
Five-satellite design–option I
– Individual arrays per zone – Two, three, or four arrays on each satellite – Satellite mass range–1176 to 1707 kg – Satellite power range–2 5 to 5.2 kW – Total LCC–$1889M (20-yr operational lifetime)* – Total channel hours–2,184,526 – Cost per channel hour–$865
Three-satellite design–individual arrays per zone, option II
– Three, six and, six arrays on three satellites – System has lower power but more array weight than other three-satellite designs. – Satellite mass range–1251 to 2150 kg – Satellite power range–3.8 to 6.5 kW – Total LCC–$1273M (20 years)* – Cost per channel hour–$583
Three-satellite design–multiple zones per array, option III
– Up to four zones covered by one large array spot – System has only two arrays per satellite – System has lower array mass & overall mass but slightly higher power than other three satellite designs. – Satellite mass range–1066 to 1893 kg – Satellite power range–4.4 to 17.2 kW – Total LCC–$1353M (20 years)* – Cost per channel hour–$619

*1-yr full operational capability still available (total program life 20 years plus 4 years development = 24 years)

TABLE 69. - HIGH RF POWER L-BAND SUMMARY OF RESULTS

– Geostationary orbit–three slots (70W, 15E, 110E)
– Eight satellites are required to achieve power channel requirements.
– Each zone is illuminated by its own aperture.
– Array technology was selected due to high power requirements.
– Zone/channels–all/full VOA
– Coverage efficiency–100%
– Centaur orbit transfer vehicle (payload–4390 kg)
– Satellite mass range–2651 to 4349 kg
– Satellite power range–20 by 50 kW
– Total LCC–$5149M (20 years)*
– Cost per channel hour–$2357

*1-year full operationl capability still available (total program life 20 years plus 4 years development = 24 years).

For VHF-band operation, two candidate orbits were selected, the 12-hour circular orbit and the 24-hour elliptical orbit described in Section 3.2. As an option, a 24-hour orbit design was also evaluated that would use one STS launch for the satellite and another launch for the upper stage (an expanded capability Centaur design).

The basic design uses a deployable phased array configuration as illustrated in Figure 88. State of the art technologies are used for attitude control electronics, TT&C, signal processing, and communication uplink. The electric power system is state of the art with the exception of the solar arrays which must be larger than any flown experimentally or on operating satellites. However, space station requirements for large solar arrays will result in development of arrays that will be directly applicable to VHF satellites. The VHF-band design approach is summarized in Table 70.

The areas requiring further development for VHF-band systems are solid state power amplifiers using MOS-FET technology, thermal control for the amplifiers, V-band crosslink electronics, and the deployable antenna structure.

Table 71 presents a summary of VHF-based full and reduced system designs that meet VOA signal strength and channel capability requirements. As shown, the total number of satellite ranges from 4 to 16 for the various concepts considered.

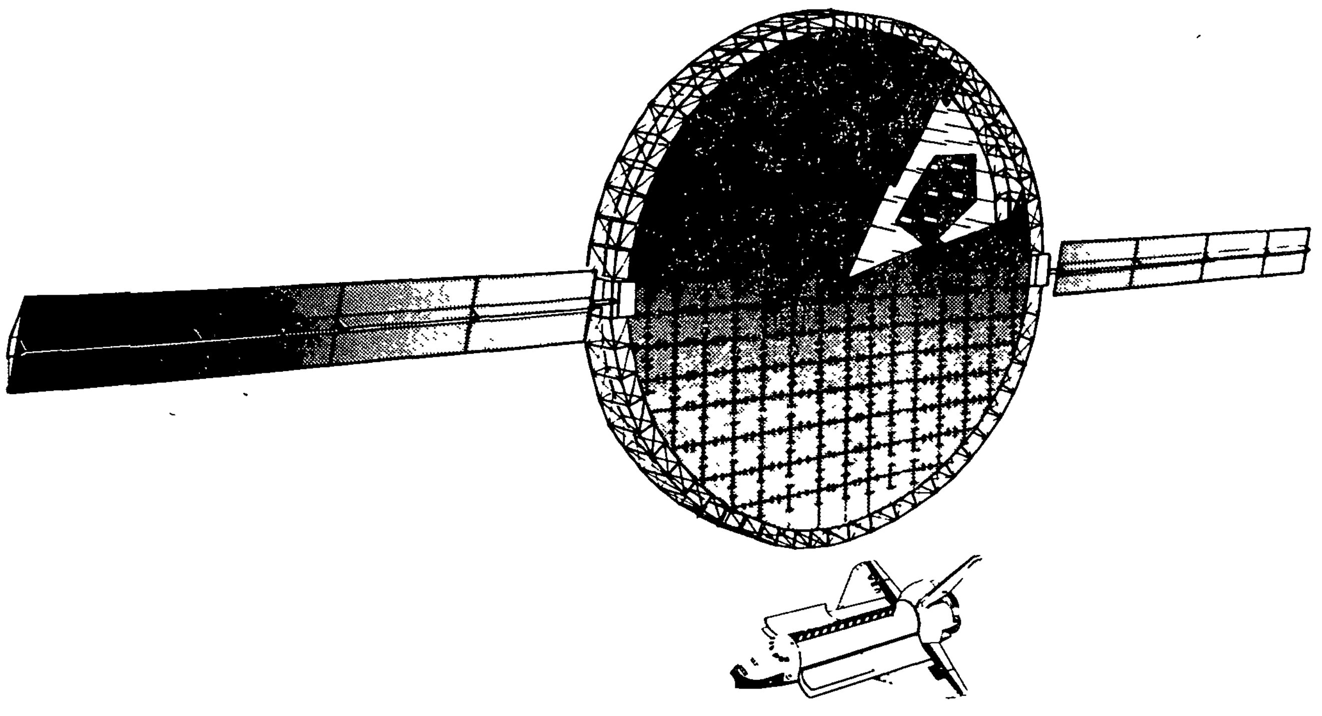

Figure 88. - Typical VHF and HF satellite.

TABLE 70. - VHF SYSTEMS' DESIGN APPROACH

- Antenna is deployable phased array.
- Power generation is by gimballed SI blanket solar array.
- Power distribution bus is at 200 Vdc
- Power storage is provided by nickel cadmium batteries.
- Transmitters' thermal rejection is by integral radiator panels.
- Attitude control is by three-axis stabilization
- Stationkeeping and maneuvering are by electric propulsion (high I_{sp}).
- Orbit transfer is by Centaur G.
- TT&C uses Ku-band uplink/downlink & V-band intersatellite link.
- Orbits are 12-h period, 45-deg inclination, circular, & 24-h period, 30-deg inclination, elliptical.

TABLE 71. - VHF-MATRIX OF DESIGN AND ANALYSIS CASES

Case	Orbit	μV/m	Channel required	Zones	No of S/C
1	12 h, 45 deg, circ	250	6, 3, 2, 2	9, 10, 12, 14	16
2	12 h, 45 deg, circ	150	6, 3, 2, 2	9, 10, 12, 14	8
3	12 h, 45 deg, circ	250	3, 2, 2, 2	9, 10, 12, 14	8
4	12 h, 45 deg, circ	200	5, 3, 2, 2	9, 10, 12, 14	8
5	24 h, 30 deg, ellip	250	6, 3, 2, 2	9, 10, 12, 14	12
6	24 h, 30 deg, ellip	(140 - 150)	6, 3, 2, 2	9, 10, 12, 14	4
7	24 h, 30 deg, ellip	250	2, 1, 1, 1	9, 10, 12, 14	4
8	24 h, 30 deg, ellip	250	6, 3, 2, 2	9, 10, 12, 14	4*

*Full shuttle satellite—large Centaur-type stage brought up in second shuttle & mated to satellite.

5.3.1 VHF Band System Weight and Volume Estimates

Weight and volume estimates for the three VHF systems are summarized in Table 72. The output of the sizing program for the 24-hour orbit, single STS launch is shown in Table 73. Note in the RF subsystem sizing summary the three different sizes of transmitters, the number of transmitters, and the total power for each set of transmitters. These total powers are for reference only since the satellite design never has all transmitters operating simultaneously. The three transmitter powers are 1.5, 2.5, and 4.0 times a nominal power level. This nominal power level times the total number of transmitters is the load power used to determine EPS size. For this VHF case the nominal transmitter power level was 180 watts.

TABLE 72. - VHF SYSTEMS SATELLITE WEIGHT AND VOLUME ESTIMATES

VHF stretched design-180 W/transmitter		
Configuration summary		
	Mass, kg	Volume, m^3
RF payload	4825.6	25.41
Auxiliary propulsion subsystem	507.7	3.34
Telemetry, tracking & command	32.1	.03
Electrical power subsystem	596.2	1.90
Thermal control subsystem	335.1	3.43
Equipment bay structure	213.8	17.12
Total system summary	6510.6	51.25
VHF baseline cast at 370 w/transmitter		
Configuration summary		
	Mass, kg	Volume, m^3
RFpayload	4204.1	26.94
Auxiliary propulsion subsystem	361.7	2.25
Telemetry, tracking & command	32.1	.03
Electrical power subsystem	1374.6	5.43
Thermal control subsystem	857.7	8.78
Equipment bay structure	206.7	12.00
Total system summary	7037.1	55.45
VHF stretched design for full STS & max Centaur—75.26 kW RF output		
Configuration summary		
	Mass, kg	Volume, m^3
RF payload	6851.	27.58
Auxiliary propulsion subsystem	627.	3.35
Telemetry, tracking & command	32.	.03
Electrical power subsystem	2170.	7.18
Thermal control subsystem	1321.	13.53
Equipment bay structure	290.	17.69
Total system summary	11293.	69.38

TABLE 73. - VHF SATELLITE SIZING EXAMPLE

```
[illegible]
Stretched Design -- 180 w/transmitter


EPS SUMMARY


Power required from source - 31.3111 KW
Power at load, average = 29.854 KW
Orbital altitude = 40000 Km
Orbital period = 27.6108 hrs
Spacecraft lifetime requirement - 7 yrs
Total eclipse time per orbit= 1.212519 hrs
Solar array degradation factor due to radiation - .9787761
Solar array thermal adjustment factor = .896
Solar array cover slide weight factor = 0
Antenna Size = 168 m by 168 m


***** Power Generation Sizing *****

For Si Blanket Array
Area required - 227.0803 m2
Weight = 363.3284 Kg Volume = 1.816642 m3


***** Shunt Regulator Sizing *****

Shunt Regulator Weight = 53.21464 Kg
 Shunt Regulator Volume = 6.033406E-02 m^3


***** Power Switching/Distribution Sizing *****

Power Switching Equip Wt. - 9.287619 Kg
Power Switching Equip Vol. = 1.053018E-02 m3

Distribution Weight; Source to Bus = 11.44194 Kg
Distribution Weight; Bus to Load = 127.5614 Kg
Total Distribution Weight = 139.0033 Kg
Total Switching and Distribution Weight = 148.291
****** TOTAL SI SYSTEM WEIGHT W/O BATTERIES(kg) = 564.8341
****** TOTAL SI SYSTEM VOLUME W/O BATTERIES(M^3) = 1.887507


***** Battery Sizing *****

For NiCd Batteries
Battery Capacity Required   1102.29 Wh
Battery Weight = 31.24937 Kg Battery Volume = 1.560666E-02 m3


***** Battery Charger Sizing *****

For NiCd Batteries
Battery Charger Weight   .1515221 Kg  Battery Charger Volume   1.717938E 04 m3

 TOTAL EPS MASS(kg)   596.235
 TOTAL EPS VOLUME(M^3) = 1.903285
```

TABLE 73. - CONTINUED

VHF Satellite Sizing Example
Stretched Design -- 180 w/transmitter

RF SUBSYSTEM SIZING SUMMARY
Data for Box Truss Ring Array (limit on diameter of 200 meters)
Antenna diameter (M) 168

Power per transmitter(w)	Number of transmitters	Output power(kw)
720	42	30.24
450	16	7.2
270	48	12.96

ANTENNA APERTURE AREA(M^2)	22167.08
TOTAL ANTENNA STRUCTURE MASS(KG)	2964.033
TOTAL ANTENNA STRUCTURE VOLUME(M^3)	22.63755
TOTAL RF TRANSMITTER MASS(KG)	1701.954
TOTAL RF TRANSMITTER VOLUME(M^3)	2.281131
WEIGHT OF UPLINK COMPONENTS(Kg)	121
VOLUME OF UPLINK COMPONENTS(M^3)	.5
WEIGHT OF SIGNAL PROC. COMPONENTS(Kg)	58
VOLUME OF SIGNAL PROC. COMPONENTS(M^3)	.02
TOTAL RF SUBSYTEM MASS(KG)	4844.987
TOTAL RF SUBSYSTEM VOLUME(M^3)	25.43868

VHF Satellite Sizing Example
Stretched Design -- 180 w/transmitter

ATTITUDE CONTROL, STATIONKEEPING AND MANEUVERING SUMMARY

Total ACS subsystem mass(Kg)	57.7	Lbs	127.2285
Total ACS subsystem volume(M^3)	.05		
Total RCS subsystem mass(Kg)	450	Lbs	992.2501
Total RCS subsystem volume(M^3)	3.3		
Total RCS/ACS subsystem mass(Kg)	507.7		
Total RCS/ACS subsystem volume(M^3)	3.35		

VHF Satellite Sizing Example
Stretched Design -- 180 w/transmitter

TT&C SUBSYSTEM SUMMARY

Total TT&C subsystem mass(kg)	32.1	Lbs	70.7805
Total TT&C subsystem volume(M^3)	.03		

TABLE 73. - CONCLUDED

VHF Satellite Sizing Example
Stretched Design -- 180 w/transmitter

THERMAL CONTROL SUBSYSTEM SIZING SUMMARY

Maximum temperature(C)	55		
Maximum heat radiated(W)	27226		
Radiative surface emissivity factor	.8		
Radiative surface absorptivity factor	.2		
Required radiator surface area(M^2)	68.82595		
Thermal Control Subsystem mass(Kg)	336.0544	Lbs	740.9999
Thermal Control Subsystem area (M^2)	68.82595		
Thermal Control Subsystem Volume(M^3)	3.441297		

VHF Satellite Sizing Example
Stretched Design 180 w/transmitter

EQUIPMENT BAY SUMMARY

Total mass of equipment bay(kg)	111.8368	Lbs	246.6003
Mass of deployment/stowage mechanisms(kg)	102		
Total mass(kg)	213.8369	(Lbs)	471.5103
Total volume of structure and mechanisms(m^3)	25.68732		

****

VHF Satellite Sizing Example
Stretched Design - 180 w/transmitter

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	4844.987	25.43868
Auxiliary Propulsion Subsytem	507.7	3.35
Telemetry, Tracking and Command	32.1	.03
Electrical Power Subsystem	596.235	1.903285
Thermal Control Subsystem	336.0544	3.441297
Equipment Bay Structure	213.8369	17.12488
TOTAL SYSTEM SUMMARY	6530.914	51.28815
	14400.67	(Lbs)

****

5.3.2 VHF-Band System Coverage Analysis

The VHF-band coverage analysis considered both 12-hour circular and 24-hour elliptical orbits. Figure 89 shows the satellite constellation and resulting ground trace for the 12-hour case. Eight satellite positions are shown, each position representing a satellite or satellites in one of eight unique orbital planes. For the 250 V/m requirement, each position shown must have two satellites to provide the required ground coverage and signal strength for full channel requirements. For reduced channels or signal strength, only one satellite is required for each orbital plane.

The 24-hour elliptical system ground trace is shown in Figure 90. Here, there are four orbital planes using three satellites each to meet full coverage requirements or one satellite each to provide reduced channel capability or signal strength.

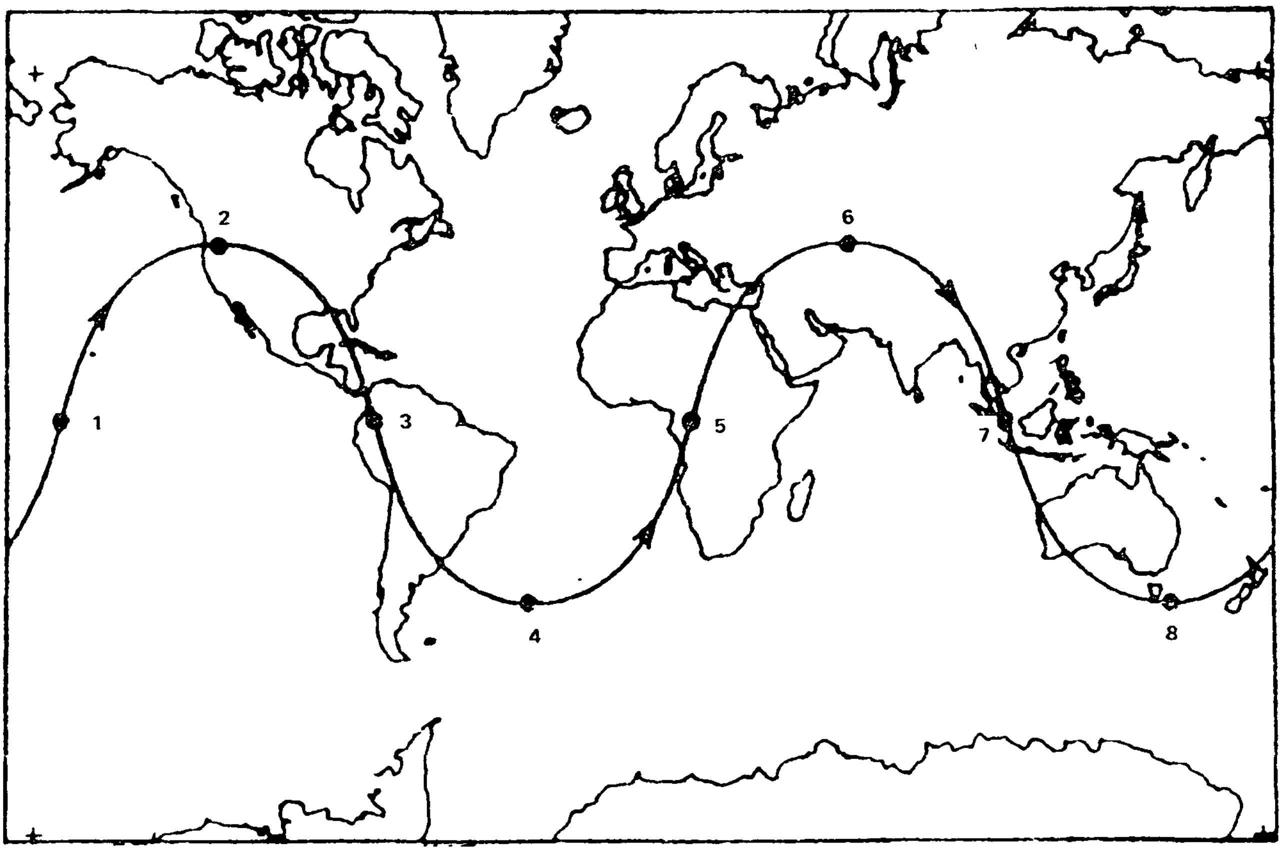

Figure 89. - Constellation for VHF system—12-hour, 45° inclination, circular, eight satellite clusters.

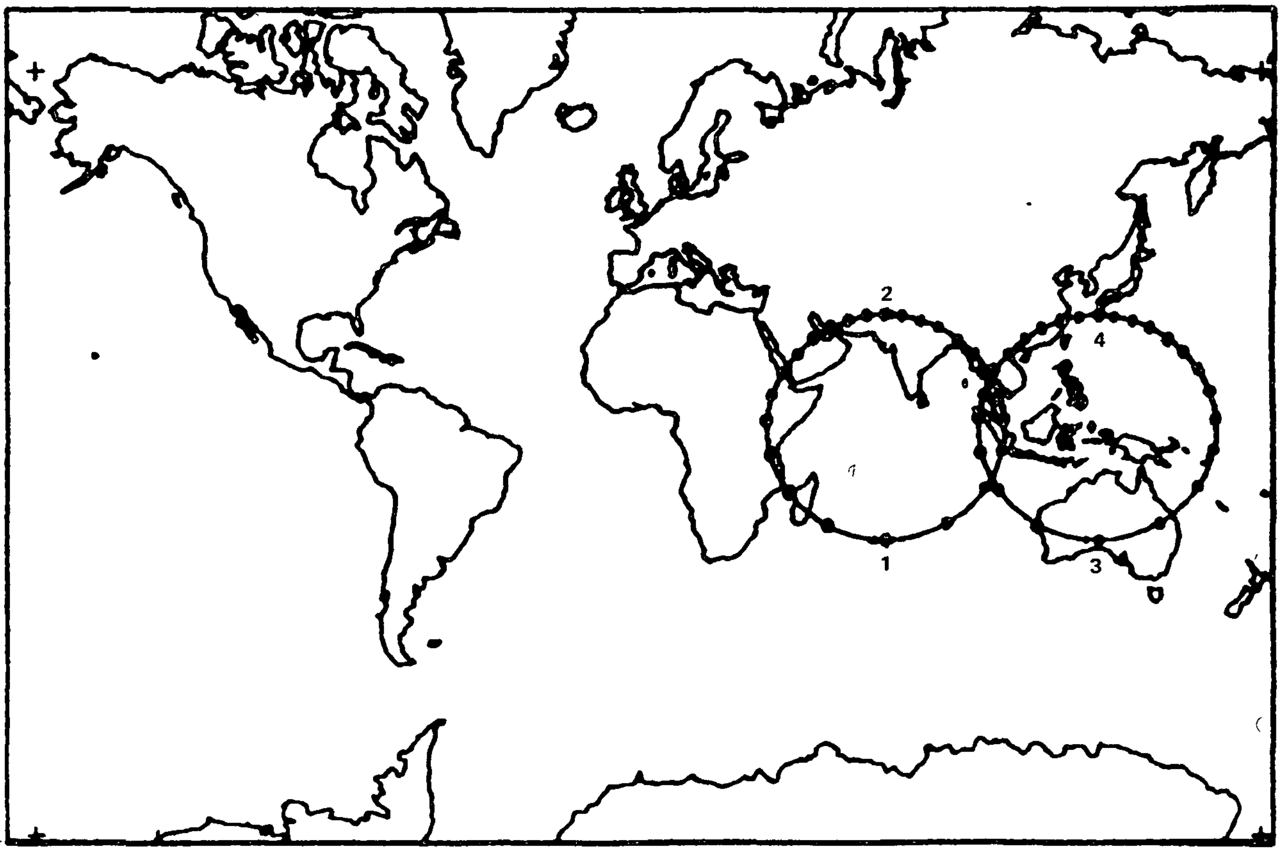

Figure 90. - Constellation for VHF system—24-hour, 30° inclination, elliptical four satellite clusters.

5.3.3 VHF-Band Systems RF Performance Analysis

For each VHF-band system, RF performance was analyzed to determine required antenna size and RF output power necessary to provide the required signal strength at edge of coverage. The RF analysis considered both satellite location and ground surface area to be covered to determine antenna aperture size. The VHF-band arrays can produce different beam shapes and sizes by selectively activating different sets of transmitters or by controlling phase. The beamwidth for VHF-band arrays was computed from:

$$BW = \frac{65\lambda}{D} \tag{5-3}$$

where

λ = wavelength (6.383 m)
D = antenna diameter (m)

The array spacing selected is between $\lambda/4$ and $\lambda/3$ to provide a compromise between adequate performance, number of transmitters, and transmitter size. A ground station antenna oriented for maximum gain was assumed for the analysis. In the ground PFD computation, a 3-dB polarization loss was assumed for circular polarization, along with a 1-dB propagation loss. For the 12-hour orbit, the antenna aperture size and beamwidth were calculated from nadir pointing to a 20° elevation angle. For the elliptical orbit, the size and beamwidth were computed using specific satellite locations. In both cases, the antenna aperture efficiency was computed from:

$$\text{efficiency} = 0.75 \cos(\alpha) \quad (5\text{-}4)$$

where

α = scan angle

The resulting maximum single satellite performance capabilities of the VHF-band satellites are summarized in Table 74. The table shows how many channels can be transmitted with the corresponding signal strength. For example, the first case shows that three channels at 254 μV/m can be transmitted to Zone 9, two channels at 256 μV/m to Zone 10, two channels at 298 μV/m to Zone 11, and two channels at 275 μV/m to Zone 12. Since the full capability calls for six channels at 250 μV/m to Zone 9, two satellites, transmitting simultaneously, would be required to provide the six channels. This leads to the requirement for 16 satellites, two in each of the eight orbital planes discussed in Section 5.3.2. The second case shows that each satellite can provide maximum signal strengths of 185 and 202 μV/m to Zone 9 and 10 respectively when transmitting the required number of channels.

TABLE 74. - VHF-BAND RF PERFORMANCE SINGLE-SATELLITE CAPABILITIES

VHF system	Zone 9		Zone 10		Zone 11		Zone 12	
	Ch	μV/m	Ch	μV/m	Ch	μV/m	Ch	μV/m
12-h orbit	3 6 5	254 185 203	2 3 3	256 202 202	2 2 2	298 298 298	2 2 2	275 275 275
24-h elliptical orbit, single STS launch	2 6	243 140	1 3	262 146	1 2	311 220	1 2	295 209
24-h elliptical orbit, two STS launches	7	258	4	260	6	252	5	262

For the 24-hour, elliptical orbit and single STS launch, only two channels can be broadcast to Zone 9 at 243 μV/m (slightly below the desired 250 μV/m). Thus, three satellites are required per orbital plane, or 12 satellites total for the system. The performance for full channel capability cannot provide 250 μV/m in any zone and can provide the required number of channels at 150 μV/m in Zones 11 and 12 only. As shown, the only satellite that can provide full channel and signal strength capability is the 24-hour system using two STS launches. To meet the ground coverage profile, four satellites would still be required, one for each of the four orbital planes shown previously.

5.3.4 VHF Band Systems LCC Estimates

The first step in estimating LCC for the VHF systems was to estimate the total nonrecurring and recurring costs for each satellite configuration. Based on the use of a single design for all satellites within a system, learning factors were then computed to derive an average recurring cost per satellite. Table 75 summarizes the costs for the VHF systems, using appropriate learning factors for a slope of 0.9. Table 76 shows the factors as a function of number of spacecraft for the VHF systems.

A further analysis of LCC could take into account the time required to launch all satellites for a system and the percent of coverage provided during the phased launches. However, this level of analysis was considered beyond the scope of the study. The basic objective was met: providing cost and performance data that would permit comparison between DVBS concepts and between DVBS and terrestrial broadcast methods.

TABLE 75. - VHF SYSTEMS' LCC ESTIMATES, 1984$

System*	1st Unit cost	No of S/C	Average unit cost	Average with launch	Total RC	S/C NRC	Ground cost	Total NRC	Total
VHF, case 1									
– 12-h orbit, 8 S/C	$156M	24	$ 97M	$222M	$5328M	$252M	$71M	$323M	$5651M
– 12-h, orbit, 12 S/C	156	36	90	215	7740	252	92	344	8084
VHF, case 2									
– 24-h orbit, 4 S/C	136	12	94	219	2628	256	71	317	2945
– 24-h orbit, 6 S/C	136	18	87	212	3816	256	71	348	4133
– 24-h orbit, 12 S/C	136	36	79	204	7344	256	92	348	7692
VHF, case 3									
– 24-h orbit, 4 S/C (multiple launch)	178	12	123	373	4476	304	71	375	4851

*20-yr operational lifetime (total program length is 20 years plus 6 years development = 26 years); 1-yr full capability remains

TABLE 76. - SATELLITE FABRICATION LEARNING FACTORS FOR VHF SYSTEMS

Number of satellites	Learning factor
12	0.69
18	0 64
24	0.62
24	0.58

5.3.5 VHF-Band Summary of Results

The VHF-band systems designed for VOA applications consisted of three system designs using from four to 16 satellites to meet VOA broadcast requirements. Each design uses a phased array and is capable of beam shaping and steering through combined phase control and selectively turning transmitters on and off. The characteristics of the three systems are summarized in Table 77.

Total estimated life cycle cost for a 20-yr operational lifetime plus a 5-yr development cycle, and VHF system capability summaries are presented in Table 78. As shown, the LCC per channel hour for VHF-band systems that were studied varies from a low value of $8,584 per hour to a high of $28,050 per hour.

TABLE 77. - VHF SYSTEMS' DESIGN SUMMARY

	12-h circular orbit	24-h elliptical orbit One STS launch	Two STS launches
Number of satellites	8 to 16	4 to 12	4
Antenna diameter, m	62.5	168	168
Antenna mass, -kg	7,037	6,511	11,293
Transmitter number/power, W	53/1480, 20/925, 59/555	43/720, 16/450, 47/270	43/2840, 16/1775, 47/1065
Total RF power, kW	49.8	19.1	75.3
Required EPS power (EOL), kW	78.5	31.3	120.4
Upper stage	Centaur	Centaur	Large Centaur derivative

TABLE 78. - VHF SUMMARY OF RESULTS

Case	Orbit	μV/m	Channels required	No. of S/C	Coverage efficiency	LCC	Total channel hours	Cost per channel hour
1	12 h	250	6, 3, 2, 2	16	84%	$8084M	343,100 h	$23,562* 28,050
2	12	150	6, 3, 2, 2	8	84	5651	343,100	16,470* 19,608
3	12	250	3, 3, 2, 2	8	84	5651	308,426	18,322* 21,812
4	12	200	5, 3, 2, 2	8	84	5651	337,626	16,737* 19,926
5	24	250	6, 3, 2, 2	12	100	7692	343,100	22,419
6	24	140 - 150	6, 3, 2, 2	4	100	2945	343,100	8,584
7**	24	250	2, 1, 1, 1	4	100	2945	251,851	11,693
8**	24	250	6, 3, 2, 2	4	100	4851	343,100	14,139

*Does not include coverage efficiency factor
**Multiple launch—hypothetical Centaur

5.4 HF-BAND SYSTEMS

For HF-band operation, two design approaches were followed. The first, similar to the VHF-band approach, identified designs required to meet full VOA requirements. The second approach identified and analyzed capabilities and LCC that would result with smaller apertures and lower powers. The basic satellite design for all HF-band concepts is similar to that shown pre viously for VHF satellites (ref. Fig. 88). In addition, an offset-fed inflatable reflector design was developed to be compared with array designs.

The inflatable reflector design uses a small deployable array antenna that serves as the feed for the reflector. All satellite subsystems would be mounted on the feed array antenna. The HF-band box truss ring array antenna design approach is summarized in Table 79. For the larger satellites, those with a goal to meet full VOA requirements, three orbits were selected for considera- tion. These orbits are defined in Table 79. For the small HF satel- lites, these orbits were considered along with a 24-hour elliptical orbit, a GEO, and a triply-synchronous orbit.

TABLE 79. - HF (26 MHZ) SYSTEM DESIGN APPROACH

- Antenna is deployable phased array.
- Power generation is gimballed SI blanket solar arrays.
- Power distribution (source to bus) is 200 Vdc.
- Power distribution (bus to transmitters) is 200 Vdc.
- Power storage is nickel cadmium batteries.
- Thermal rejection is passive radiator panels
- Spacecraft is three-axis stabilization.
- Stationkeeping & maneuvering is electric propulsion (high I_{sp}).
- Orbit transfer uses Centaur G.
- TT&C is Ku-band uplink/downlink, V-band crosslink.
- Orbits are 6-h, 30 or 45° inclination, circular; 8-h, 45° inclination, circular, & 12-h, 45° inclination, circular.

5.4.1 HF Band Systems Weight and Volume Estimates

The full capability of HF systems uses large aperture (60-115.5 m) antennas with multiple sized transmitters for beam shaping. The weight and volume estimating method for these systems is the same as for VHF systems described in Section 5.3.1. The weight and volume estimates for these satellites are summarized by the computer outputs shown in Tables 80 through 82.

TABLE 80. - HF WEIGHT AND VOLUME ESTIMATES–6-HOUR ORBIT

HF DSB 60-m antenna, 6-h orbit, single launch DSB carrier output power–38.1 kW DSB or SSB total RF output power–57.2 kW		
System Configuration Summary		
	Mass, kg	Volume, m^3
RF payload	4890	46.18
Auxiliary propulsion subsystem	323	2.25
Telemetry, tracking, & command	32	.03
Electrical power subsystem	3405	7.78
Thermal control subsystem	984	10.08
Equipment bay structure	208	12.10
Total system summary	9845 21710, lb	78 44
HF DSB 60-m antenna, 6-h orbit–multiple launch DSB carrier output power–97.5 kW DSB or SSB total RF signal power–146.3 kW		
System Configuration Summary		
	Mass, kg	Volume, m^3
RF payload	8052	49.56
Auxiliary propulsion subsystem	631	4.45
Telemetry, tracking, & command	32	03
Electrical power subsystem	8597	19.62
Thermal control subsystem	2518	25.78
Equipment bay structure	305	23 63
Total system summary	20137 (44404 lb)	123.09

TABLE 81. - HF WEIGHT AND VOLUME ESTIMATES–8-HOUR ORBIT

HF 80-m antenna, 8-h orbit, single launch DSB carrier output power–38.7 kW DSB or SSB total RF output power–58 kW		
System Configuration Summary		
	Mass, kg	Volume, m^3
RF payload	5329	46.26
Auxiliary propulsion subsystem	355	2.25
Telemetry, tracking, & command	32	.03
Electrical power subsystem	1787	7.03
Thermal control subsystem	1021	10.46
Equipment bay structure	217	12.10
Total system summary	8743 (19280 lb)	78.14
HF 80-m antenna 8-h orbit–multiple launch DSB carrier power–88.9 kW DSB or SSB total RF output power–133.3 kW		
System Configuration Summary		
	Mass, kg	Volume, m^3
RF payload	8070	49.20
Auxiliary propulsion subsystem	656	4.45
Telemetry, tracking, & command	32	.03
Electrical power subsystem	4040	15.94
Thermal control subsystem	2347	24.04
Equipment bay structure	308	23.55
Total system summary	15456 (34081 lb)	117.21

TABLE 82. - HF WEIGHT AND VOLUME ESTIMATES–12-HOUR ORBIT

HF 12-h orbit 115.5 m antenna–single launch DSB carrier output power–20.3 kW DSB or SSB total RF output power–30.5 kW		
System configuration summary		
	Mass, kg	Volume, m^3
RF payload	4995	45.19
Auxiliary propulsion subsystem	326	2.25
Telemetry, tracking, & command	32	.03
Electrical power subsystem	943	3.43
Thermal control subsystem	536	5 49
Equipment bay structure	198	11.81
Total system summary	7032 (15506 lb)	68.22
HF 12-h orbit 115.5 m antenna–multiple launch DSB carrier output power, 59.8 kW DSB or SSB total RF output power, 89.7 kW		
System configuration summary		
	Mass, kg	Volume, m^3
RF payload	7150	47.50
Auxiliary propulsion subsystem	431	2.25
Telemetry, tracking, & command	32	.03
Electrical power subsystem	2663	9 86
Thermal control subsystem	1579	16.17
Equipment bay structure	263	12.35
Total system summary	12120. (26726 lb)	88.18

The next set of HF systems are the small satellite reduced size and power concepts. These systems assume a single size of transmitter as opposed to three sizes assumed for the previous HF cases. For each of the applicable HF reduced size concepts, analyses were performed from 10 kW to the maximum power attainable with a fully dedicated STS launch. Figures 91 through 96 show the estimated weight and field strength as a function of total RF output power for each of the six orbits considered for small HF concepts. The field strength shown is for Zone 1 only.

The last type of HF system studied was the inflatable reflector concept with an array antenna feed. The weight and volume of the inflatable structure were obtained first, then the inflatable's mass was subtracted from the maximum payload limit and the weights and volumes for the small feed array and the rest of the subsystems were estimated. The total satellite weight is the sum of the two summaries shown in Table 83.

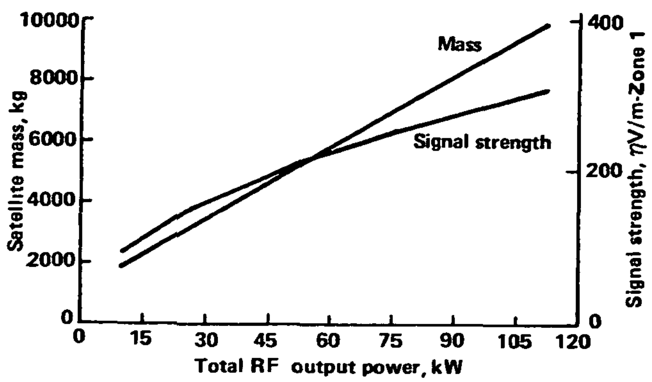

Figure 91. -
Small HF 6-hour satellite weight and signal strength.

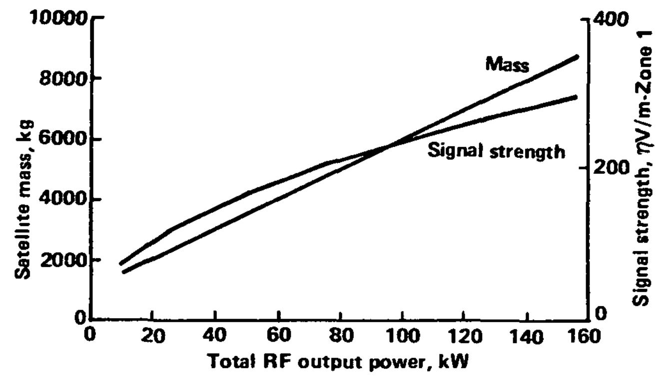

Figure 92. -
Small HF 8-hour satellite weight and signal strength.

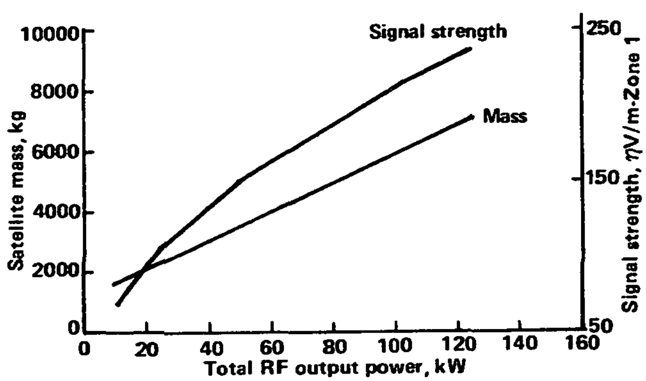

Figure 93. -
Small HF 12-hour satellite weight and signal strength.

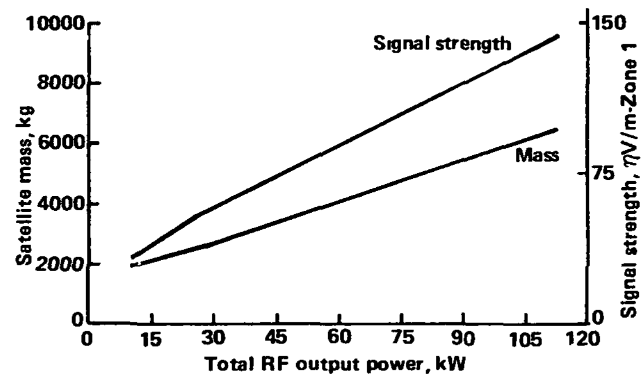

Figure 94. -
Small HF 24-hour elliptical orbit satellite weight and signal strength.

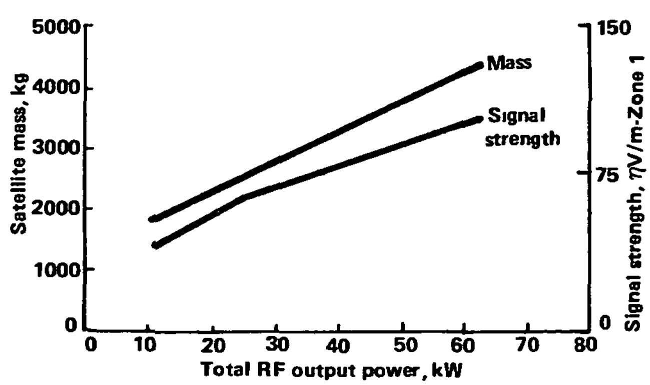

Figure 95. -
Small HF GEO satellite weight and signal strength.

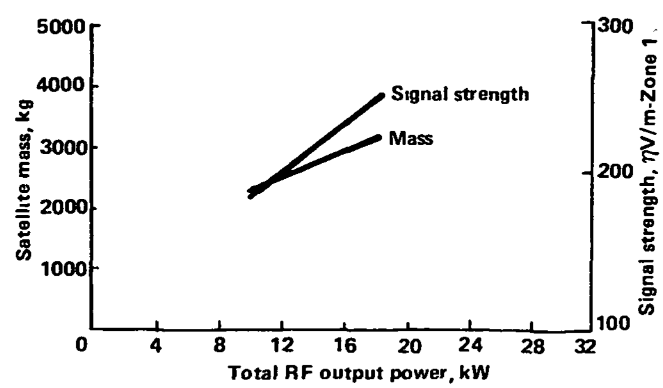

Figure 96. -
Small HF triply-synchronous orbit satellite weight and signal strength

TABLE 83. - HF INFLATABLE CONCEPT WEIGHT AND VOLUME ESTIMATES

HF inflatable structure		
RF Subsystem sizing summary Data for inflatable (limit on dia from 10 to 200 m)		
Antenna aperture area, m^2	22167	
Total antenna structure mass, kg	1663	
Total antenna structure volume, m^3	6 69	
Weight of uplink components, kg	–	
Volume of uplink components, m^3	–	
Weight of signal proc components, m^3	–	
Volume of signal proc components, m^3	–	
Total RF subsystem mass, kg	1663	
Total RF subsystem volume, m^3	6 69	
HF inflatable design 12-h orbit DSB carrier output power—38 9 kW DSB or SSB total RF output power—58 4 kW		
System configuration summary		
	Mass, kg	Volume, m^3
RF payload	1697	42.82
Auxiliary propulsion subsystem	1498	6 65
Telemetry, tracking & command	32	.03
Electrical power subsystem	1549	6 47
Thermal control subsystem	464	4.75
Equipment bay structure	161	.96
Total system summary	5404 (11916 lb)	61.69

5.4.2 HF-Band Coverage Analysis

The coverage analysis for the large HF satellite concepts was similar to that described for the VHF-band systems. Figures 97, 98, and 99 show the respective ground traces for the 6-, 8-, and 12-hour orbits. As shown in Figure 97, the three solid line paths represent the coverage trace for 30° inclined orbits while the dashed lines represent the trace for 45° inclined orbits. The complete system includes eight orbital planes with the number of satellites in each orbit dependent on the signal strength and channel capabilities of a satellite. The dots on the ground trace represent the relative satellite positions resulting from the eight orbital planes. Figures 98 and 99 show the ground traces and satellite positions for the 8- and 12-hour orbits. Again, eight orbital planes are used.

The five nongeostationary orbits for the small HF satellites were analyzed to determine the ground coverage that could be obtained from a single satellite. Figures 100 through 106 show typical coverages that could be obtained in each zone as a function of universal time. Figure 106 compared to Figure 105 shows the effect on ground coverage of the 4-minute daily sidereal shift over three months for the 8-hour orbit. Similar time shifts would result for the other nongeostationary orbits.

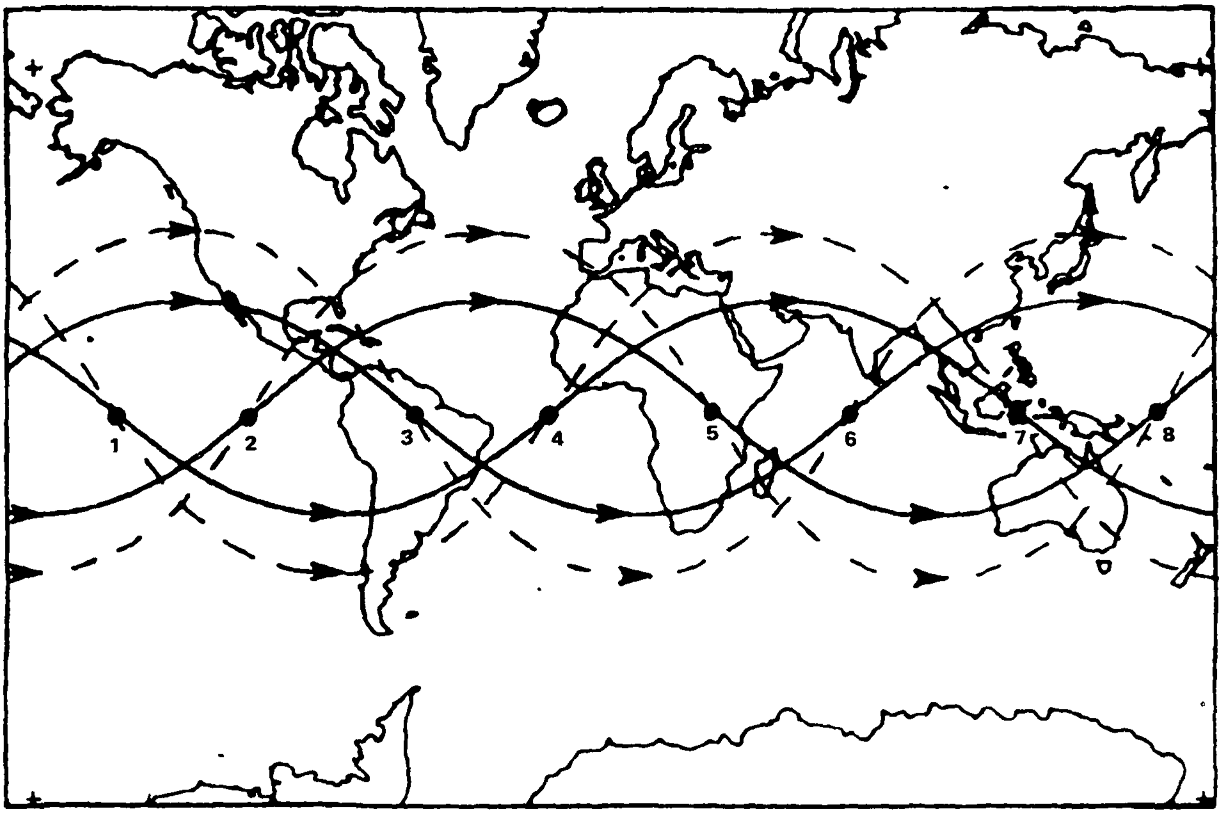

Figure 97. -

Constellation for HF system—6-hour, 30 and 45° inclination, circular, eight satellite clusters.

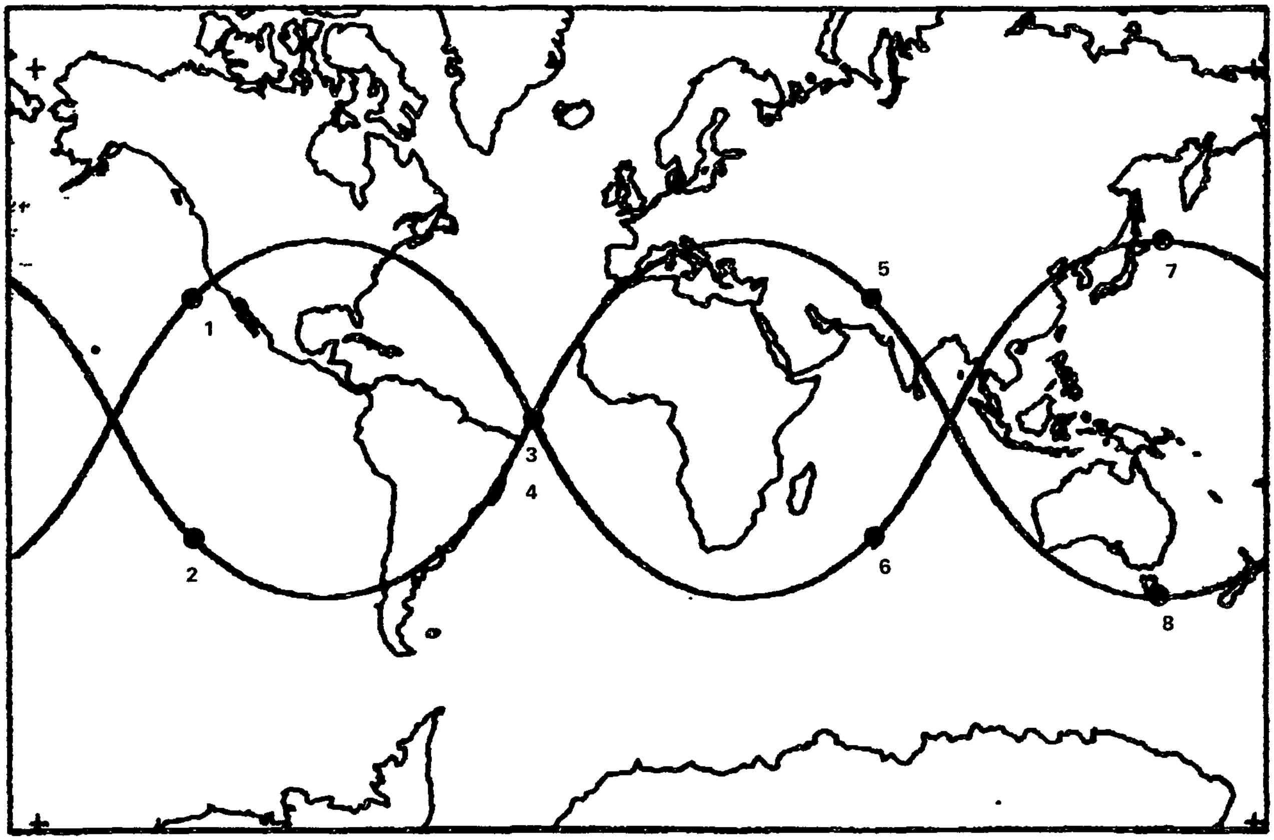

Figure 98. - Constellation for HF system—8-hour, 45° inclination, circular, eight satellite clusters.

C-3

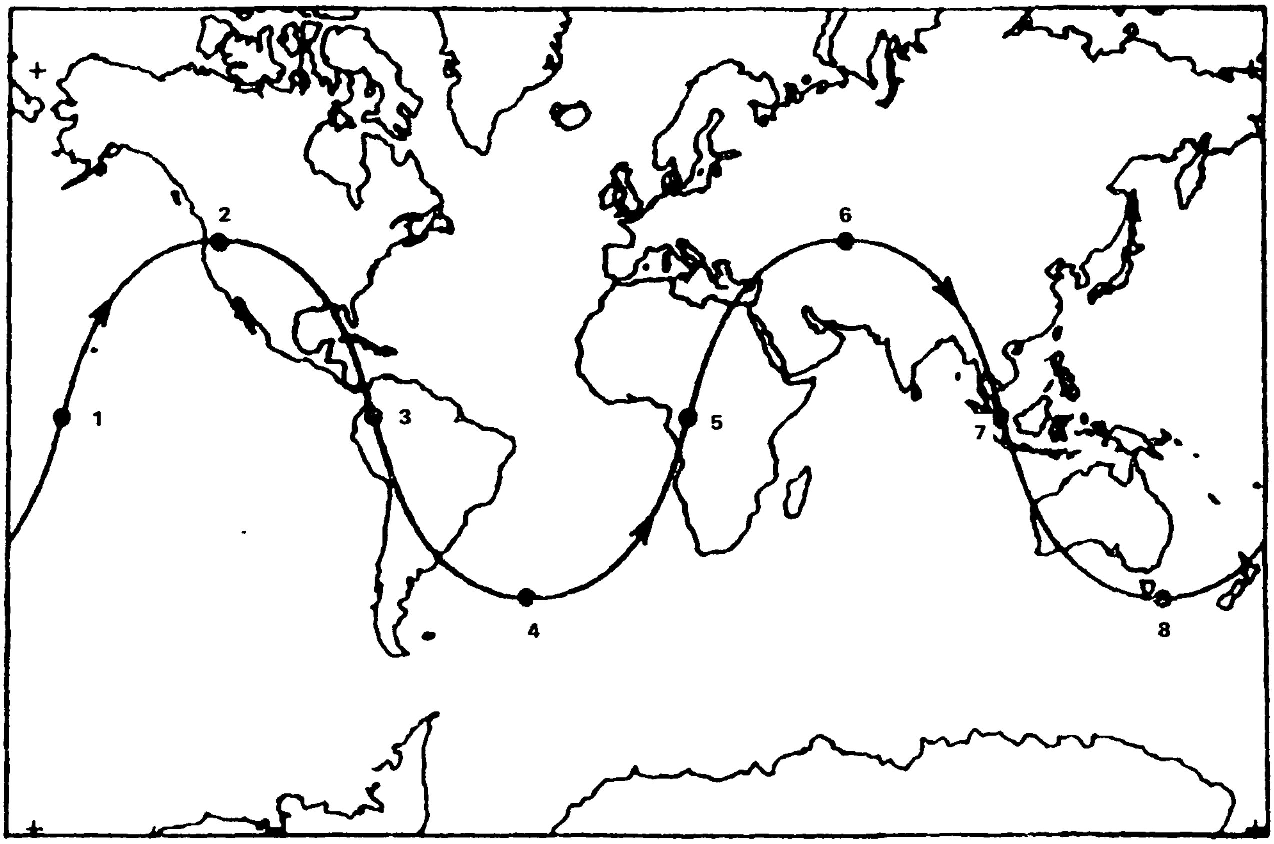

Figure 99. - Constellation for HF system—12-hour, 45° inclination, circular, eight satellite clusters.

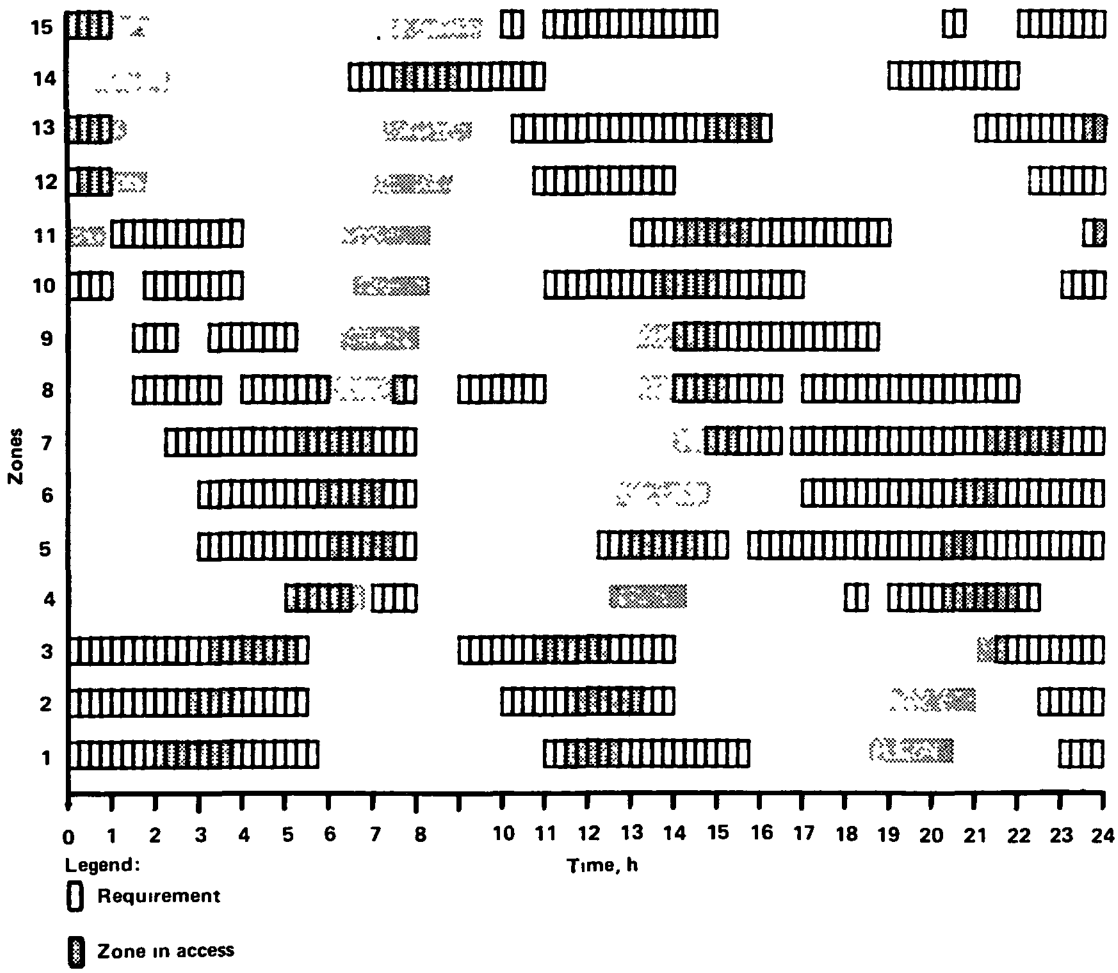

Figure 100. -6-hour orbital satellite coverage.

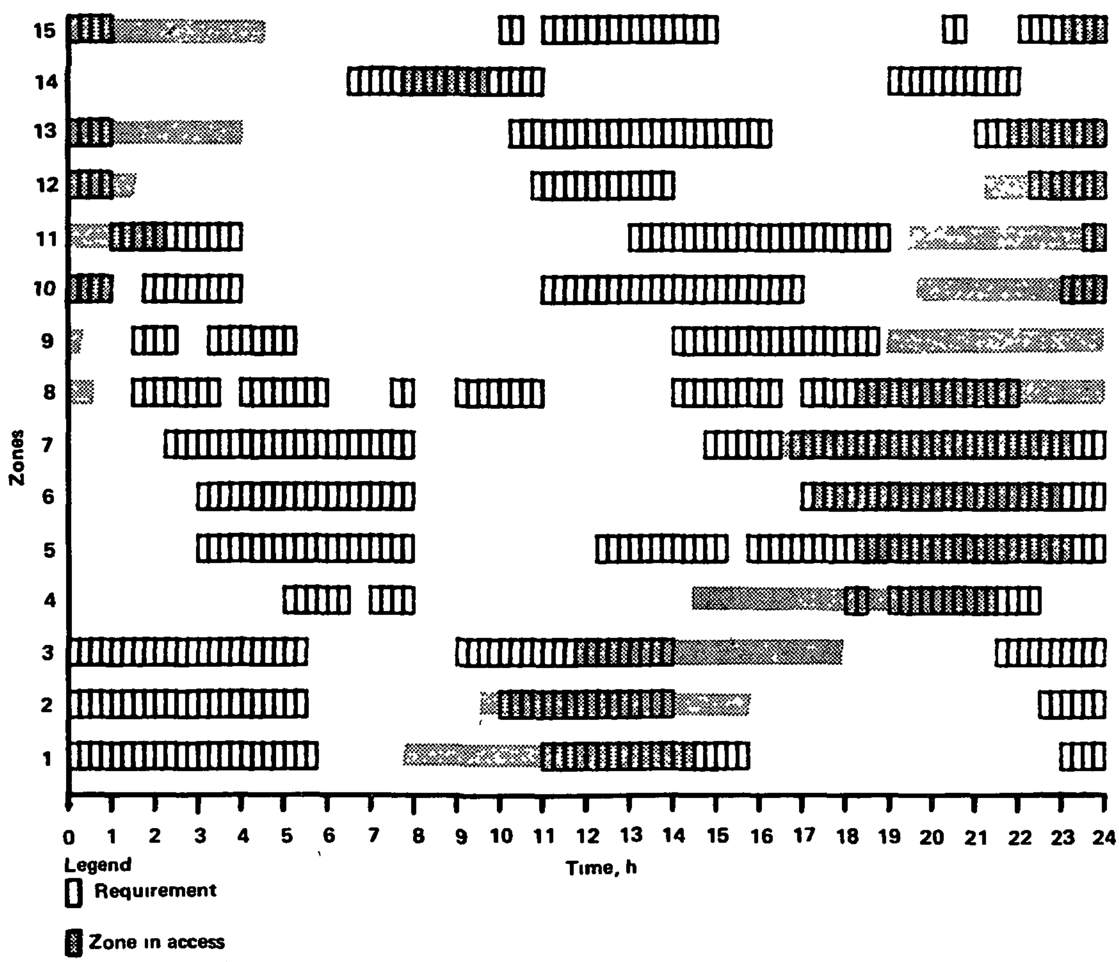

Figure 101. - Small HF satellite—12-hour orbital satellite coverage.

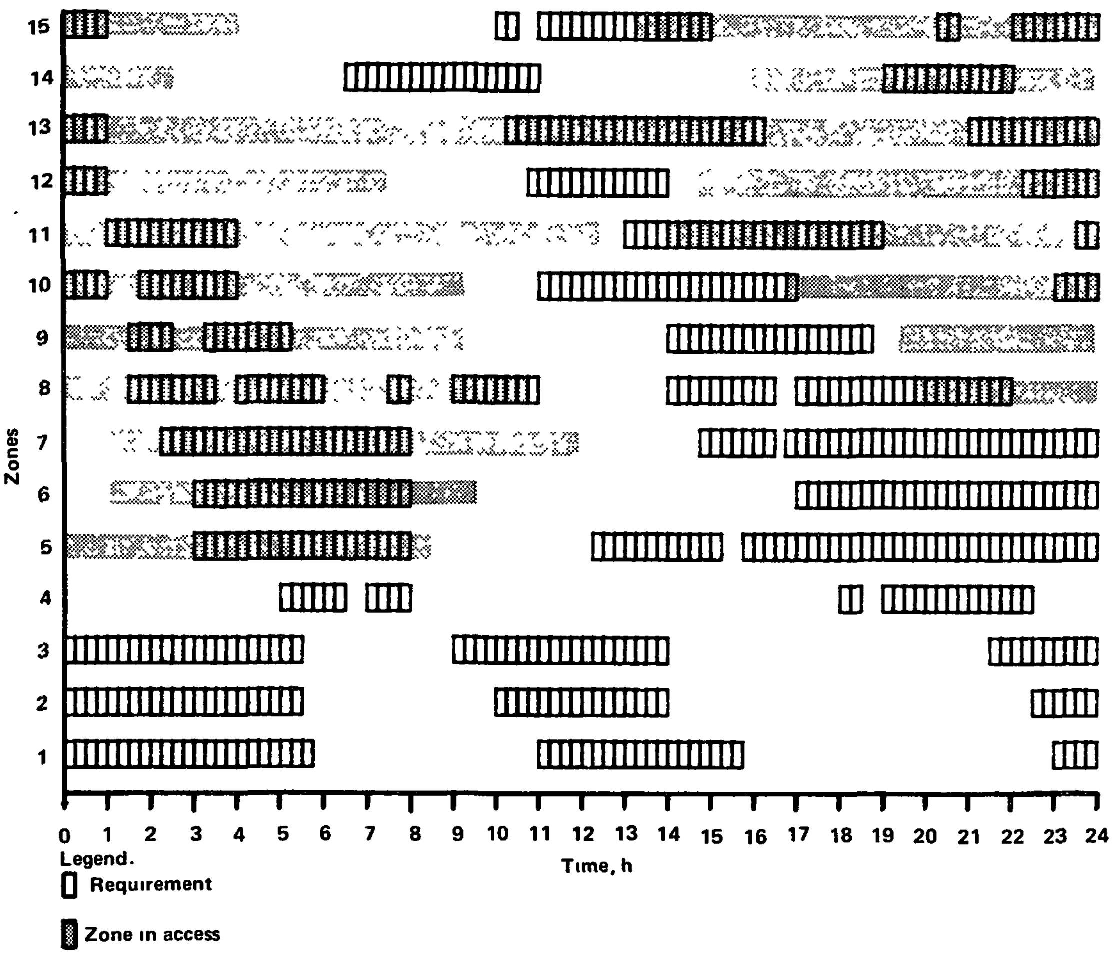

Figure 102. - Small HF–24-hour elliptical orbital satellite coverage.

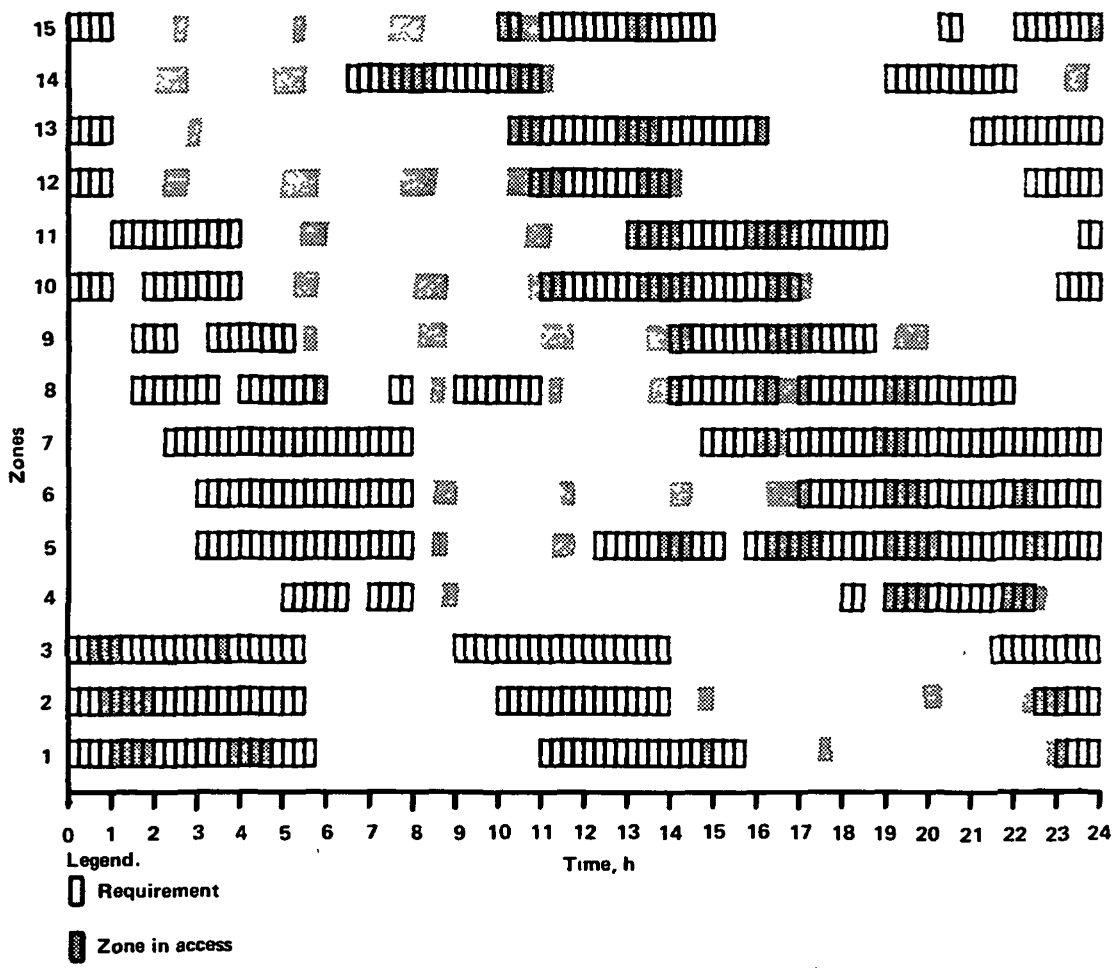

Figure 103. - Small HF–triply-synchronous satellite coverage.

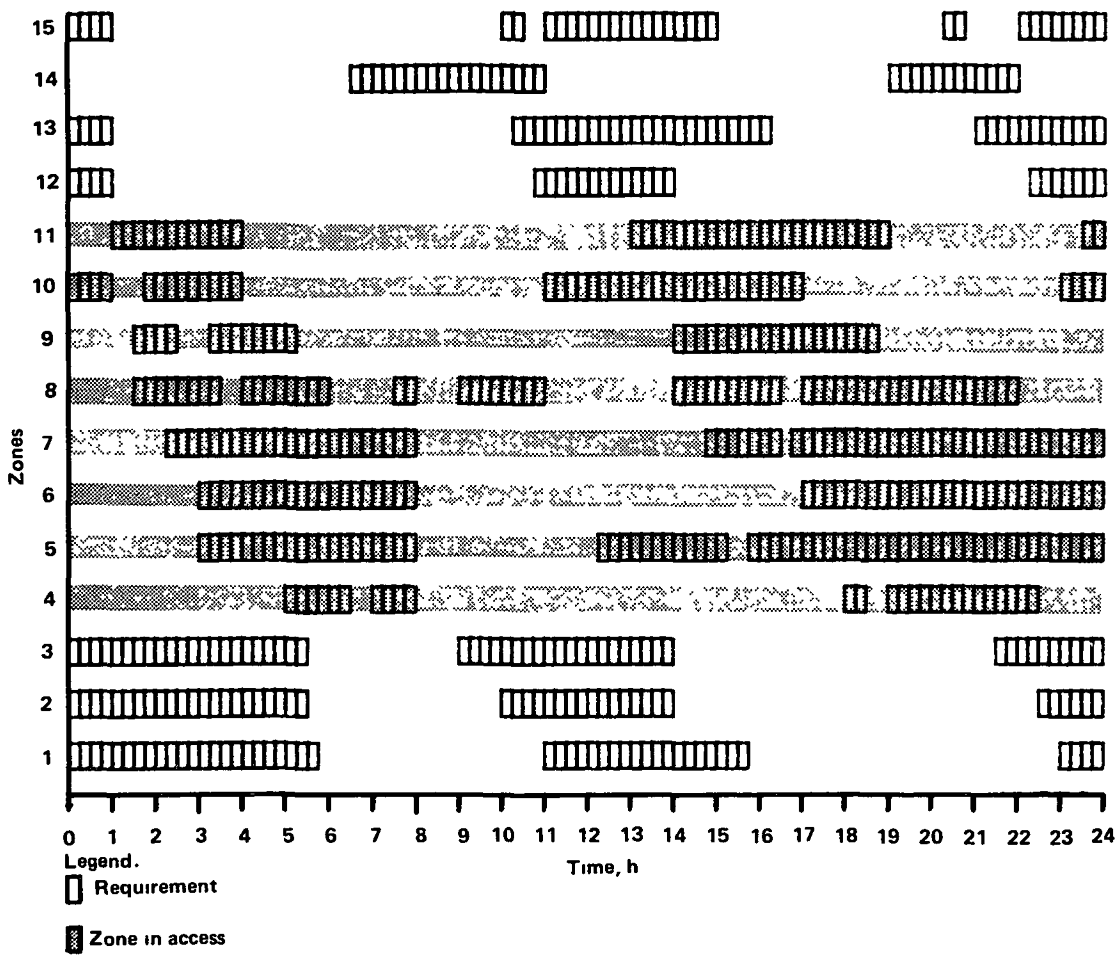

Figure 104. - Small HF—geostationary satellite coverage.

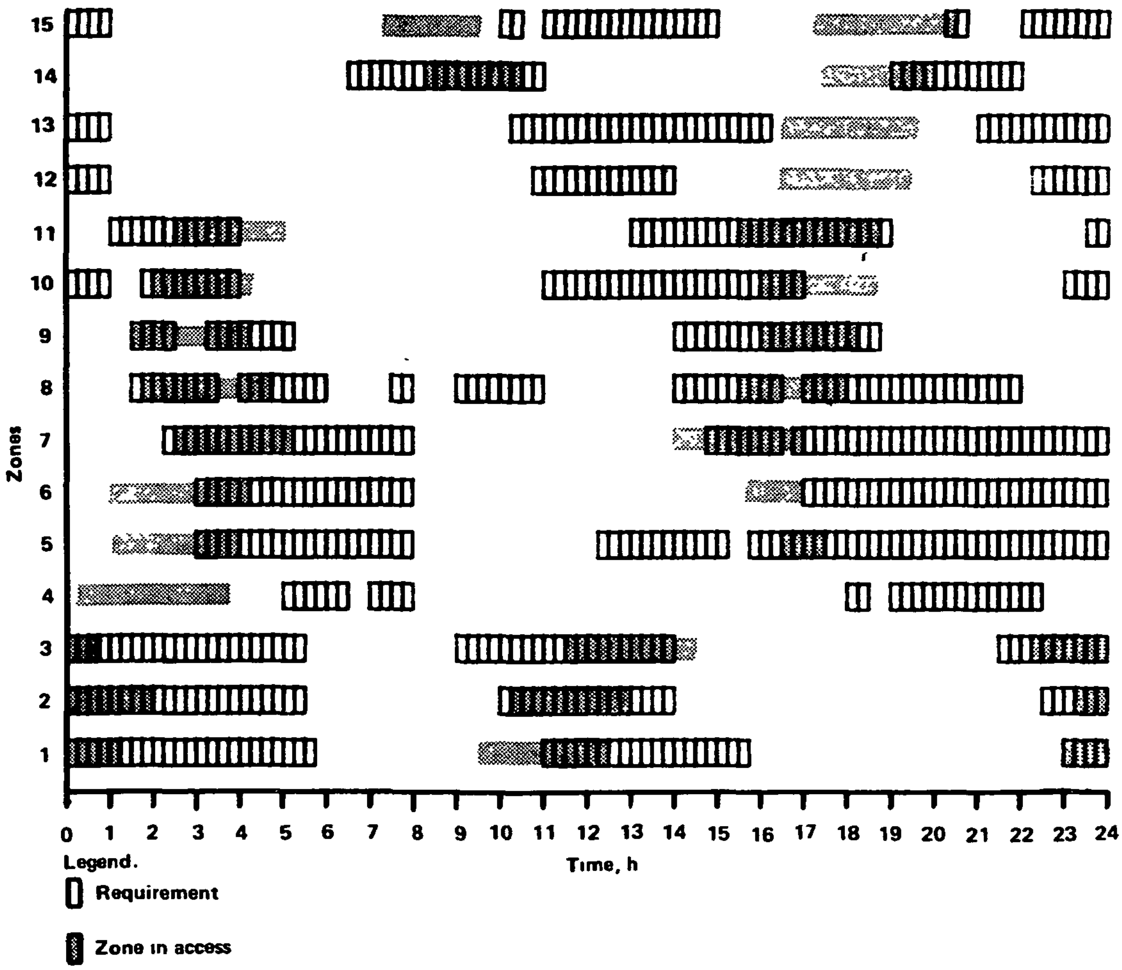

Figure 105. - Small HF—8-hour orbital satellite coverage.

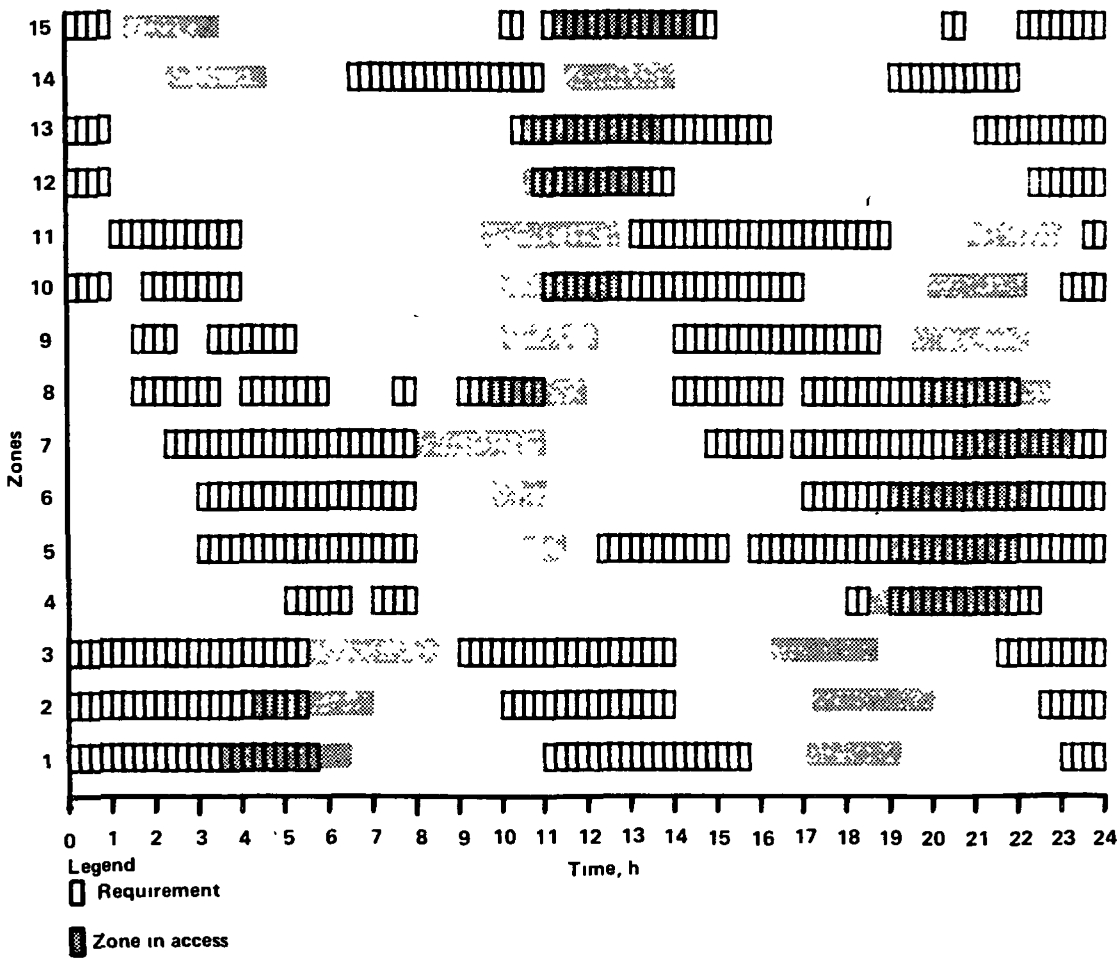

Figure 106. - Small HF—8-hour orbit: 3-month sidereal shift.

5.4.3 HF-Band Systems RF Performance Analysis

The approach taken to analyze HF-band satellites RF performance was similar to that described for VHF-band satellites. The only differences were a 2-dB propagation loss for HF instead of the 1 dB used for VHF, and element spacing from $\lambda/3$ to $3/4\ \lambda$ for HF arrays. Table 84 shows for a single satellite the maximum number of channels and signal strength available in each zone for the large HF systems. Signal strength for the small HF satellites was previously shown as a function of RF power (Figs. 91 through 96) along with satellite estimated weight.

TABLE 84. - HF SYSTEM MAXIMUM RF PERFORMANCE CAPABILITIES SUMMARY FOR A SINGLE SATELLITE

System	Zones															
	1	2	3U	3L	4	5	6	7	8	9	10	11	12	13	14	15
8-h orbit single launch	1–320	1–345	1–230	1–259	1–301	1–298	1–325	1–228	1–351	1–357	1–282	1–282	1–327	1–221	1–301	1–223
	4–160	5–154	2–162	2–183	4–152	4–149	4–163	2–161	5–157	5–159	3–157	3–158	4–163	2–156	4–151	2–158
8-h orbit Multiple launch	2–346	3–291	1–348	1–393	2–327	2–320	2–352	1–344	3–297	3–301	2–302	2–303	2–350	1–335	2–323	1–338
6-h orbit single launch	1–308	1–324	1–218	1–257	1–294	1–293	1–308	1–217	1–314	1–350	1–272	1–270	1–311	1–211	1–295	1–213
	4–162	4–159	2–154	2–182	3–165	4–147	4–154	2–154	5–153	5–157	3–152	3–151	5–156	2–149	3–165	2–150
6-h orbit multiple launch	2–399	3–292	1–350	2–291	2–334	2–332	2–350	1–347	3–304	3–314	2–308	2–306	2–350	1–337	1–335	1–341

5.4.4 HF-Band Systems LCC Estimates

The approach taken to estimate LCC for the HF-band systems was similar to that taken for VHF-band systems. A 20-yr operational lifetime was assumed after a 5-yr development cycle. A launch frequency was assumed that would result in a fully operational system in two years after completion of development. Obviously, this assumption may be unrealistic for those systems using a large number of satellites. Table 85 shows the LCC breakdown for the 6- and 8-hour HF systems. The number of satellites shown is for three launch sets. The average unit cost includes the learning factor for the number of satellites shown. The launch cost was $125M for the single STS launch cases and $250M for the multiple launch cases. The increase in ground costs as a function of number of satellites was described previously in Section 4.3.8.4.

TABLE 85. - HF FABRICATION COST LEARNING FACTORS

Number of satellites	Recurring cost learning factor
24	0.62
36	0.58
48	0.56
72	0.52
144	0.47
264	0.43

Figures 107 through 112 show satellite nonrecurring cost (NRC), recurring cost (RC), and total costs as a function of RF output power for each of the small HF concepts. Using a Centaur for the upper stage results in a dedicated launch for each satellite. Use of a different upper stage might be possible for a low-power satellite.

Estimated launch cost and ground operations cost should be added to the satellite costs shown in the figures to estimate a total system cost for the small satellite concepts. Also, use of multiple satellites would reduce recurring cost by the factors shown in Table 86.

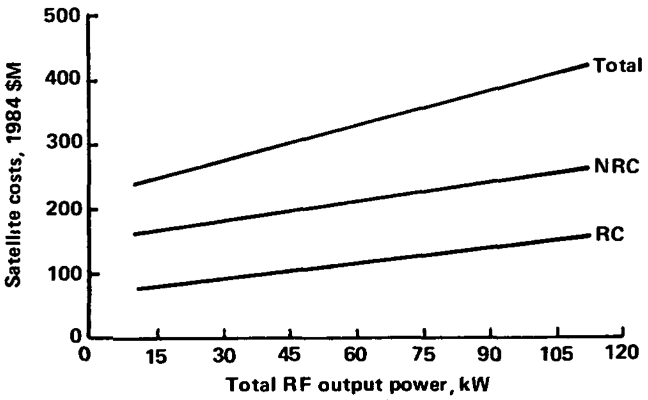

Figure 107. - Small HF 6-hour satellite costs

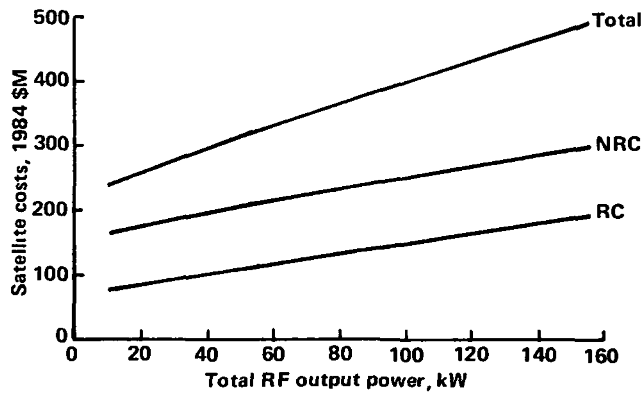

Figure 108. -
Small HF 8-hour satellite costs.

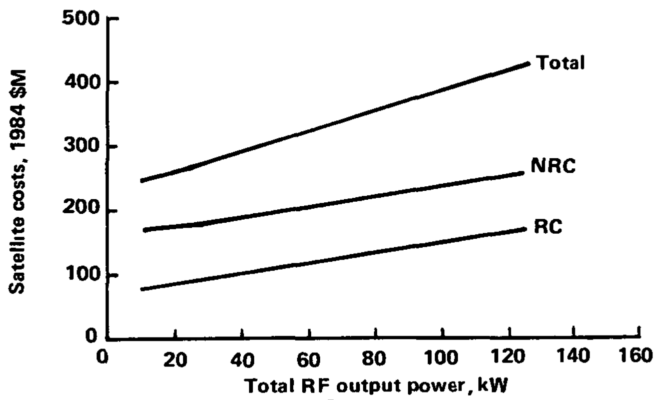

Figure 109. -
Small HF 12-hour satellite costs.

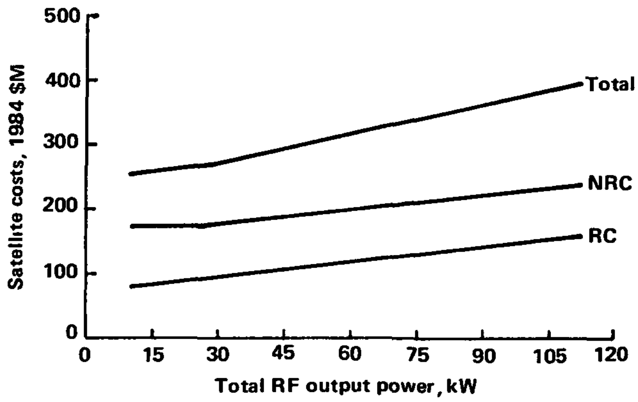

Figure 110. -
Small HF 24-hour elliptical orbit satellite costs.

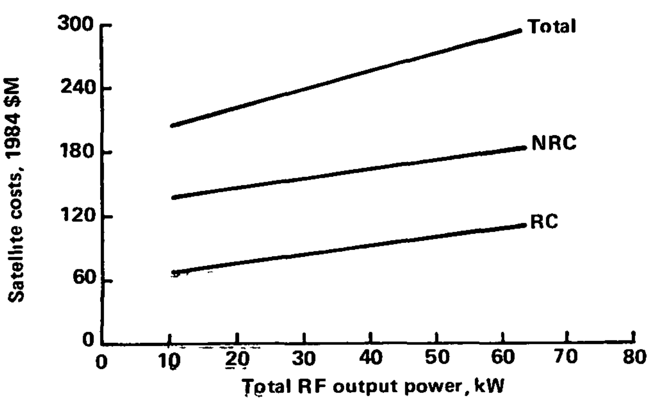

Figure 111. - Small HF GEO satellite costs.

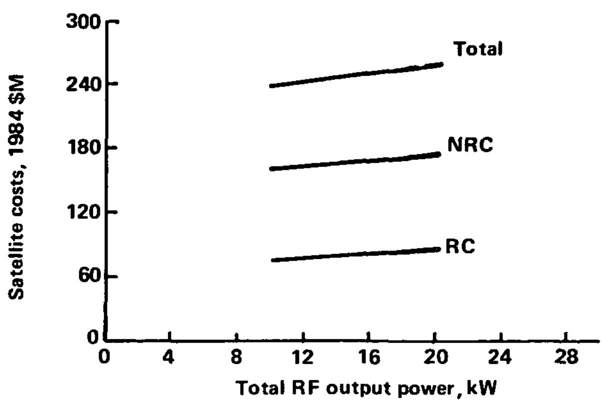

Figure 112. -
Small HF triply-synchronous orbit satellite costs.

TABLE 86. - HF SYSTEMS' LCC ESTIMATES

System*	1st unit cost	No of S/C	Average unit cost	Average launch	Total R	S/C NRC	Ground cost	Total NRC	Total
6-h orbit (single launch)	$161M	24	$100M	$225M	$ 5,400M	$247M	$ 71M	$318M	$ 5,718M
	161	48	90	215	10,320	247	92	339	10,659
	161	72	84	209	15,048	247	113	360	15,408
	161	144	76	201	28,944	247	176	423	29,367
	161	264	69	194	51,216	247	281	528	51,744
6-h orbit (multiple launch)	235	24	146	396	9,504	297	71	368	9,872
	235	48	132	382	18,336	297	92	389	18,725
8-h orbit (single launch)	170	24	105	230	5,520	271	71	342	5,862
	170	48	95	220	10,560	271	92	363	10,923
	170	72	88	213	15,336	271	113	384	15,720
	170	144	80	205	29,520	271	176	447	29,967
	170	264	73	198	52,272	271	281	552	52,824
8-h orbit (multiple launch)	230	24	143	393	9,432	305	71	376	9,808
	230	48	129	379	18,192	305	92	397	18,589

*20-yr operational life

5.4.5 HF-Band Summary of Results

The HF-band systems designed and analyzed for VOA applications included multiple satellite systems aimed at meeting all VOA requirements, and single small lower-cost satellites. The selected baseline system used an 8-hour circular orbit. The other full capability systems used 6- and 12-hour circular orbits. The 12-hour system was rejected due to its low power flux density on the ground. Table 87 summarizes the satellite designs for the 8- and 6-hour orbits. Tables 88 and 89 contain summaries of the 8- and 6-hour systems, showing required number of satellites and estimated system LCCs for a matrix of RF performance capabilities. As with the VHF systems, there would still be one year of full operational capability remaining after 20 years. The costs per channel hour and ground costs are each based on 20 years of operation. As shown, the LCC per channel hour for HF-band systems that were studied varied from a low value of $3,145 per hour to a high of $51,835 per hour.

TABLE 87. - HF SYSTEM DESIGN SUMMARY

	8-h Orbit, single launch	8-h Orbit, multiple launch	6-h Orbit, single launch	6-h Orbit, multiple launch
Antenna diameter, m	80	80	60	60
Satellite mass, kg	8,743	15,456	9,846	20,138
RF output power, kW	58	133	57	146
Required EPS power, kW	93	213	92	233
Transmitters, No.	177	177	177	177
Transmitter powers, W	492,819,1311	1130, 1882, 3012	485,808,1293	1240,2066,3306

TABLE 88. - HF-MATRIX OF DESIGN AND ANALYSIS CASES (8 HOURS)

Case	Required μV/m	No. of channels	Zones	No. of S/C	LCC	Cost/channel hour
1-8	300*	– Full VOA	All	88	$52,824M	$24,181/h
2-8	300*	– Six ch max	All	48	29,967	14,440
3-8	300*	– One ch max	All	8	5,862	5,082
4-8	300*	– Two S/C per cluster	All	16	10,923	6,268
5-8	150	– Full VOA	All	24	15,720	7,195
6-8	150	– Two S/C per cluster	All	16	10,923	5,168
7-8	150	– One S/C per cluster	All	8	5,862	3,225
8-8	300	– Two S/C per cluster	All	16**	18,589	9,451
9-8	300	– One S/C per cluster	All	8**	9,808	6,086

*Cannot achieve 300 μV/m for single channel Zones 3U, 3L, 7, 10, 11, 13, 15
**Full shuttle satellite—large Centaur-type stage in second orbiter

TABLE 89. - HF-MATRIX OF DESIGN AND ANALYSIS CASES (6 HOURS)

Case	Required μV/m	No. of channels	Zones	No. of S/C	LCC	Cost/channel hour
1-6	300*	– Full VOA	All	88	$51,744M	$23,686/h
2-6	300*	– Six ch max	All	48	29,367	14,140
3-6	300*	– One ch max	All	8	5,718	4,957
4-6	300*	– Two S/C per cluster	All	16	10,659	6,116
5-6	300*	– Six ch max	1, 2, 3U, 4, 6, 7, 8, 11, 13, 15	48	29,367	51,835
6-6	300*	– Six ch max	3L, 5, 9, 10, 12, 14	48	29,367	34,831
7-6	150	– Full VOA	All	24	15,408	7,053
8-6	150	– Two S/C per cluster	All	16	10,659	5,044
9-6	150	– One S/C per cluster	All	8	5,718	3,145
10-6	300	– Two S/C per cluster	All	16**	18,725	9,942
11-6	300	– One S/C per cluster	All	8**	9,872	6,127

*Cannot achieve 300 μV/m for single-channel Zones 3U, 3L, 7, 10, 11, 13, 15
**Full shuttle satellite—large Centaur-type stage in second orbiter

6.0 DVBS PLANNING SUPPORT

The objectives of this part of the study were to provide information for the selected satellite concepts that will identify the following:

1) What needs to be done to implement a DVBS satellite system,

2) What technologies need to be advanced to reduce technical risks or enhance DVBS satellite performance,

3) The cost and schedule risks that would be encountered in critical technology development plans,

4) The cost and schedule risks that would be encountered for complete DVBS satellite programs.

Following are discussions detailing the approach taken to meet these objectives and the results of analyses.

6.1 CRITICAL TECHNOLOGY DEVELOPMENT PLANS

The first step taken was to analyze performance risks for the four frequency bands of interest. The set of generic risks shown in Table 90 were selected and the risk level assessed for satellite systems, using the results of the technology survey described in Section 3.4, the technology tradeoffs described in Section 4.2, and analyses performed during the study. The risk assessment was expressed in qualitative terms as shown in the table. Using this risk assessment as a baseline, specific technologies were identified that must be addressed to reduce risk for VOA satellite systems.

TABLE 90. - DVBS PLANNING SUPPORT—PERFORMANCE RISK ANALYSIS

Risk	Ku	L	VHF	HF
State-of-the-art advance	Low	Medium	High	High
Physical properties	Low	Medium	Medium	Medium
Material properties	Low	Low	Low	Low
Radiation properties	Low	Low	Medium	Med/high
Material availability	Low	Low	Medium	Medium
Testing/modeling validity	Low	Low	High	High
Integration/interface (utilities)	Low	Medium	High	High
Program personnel	Low	Low	Low	Low
Software design	Low	Low	Low	Low
Safety	Low	Low	Low	Low
Security	Medium	Medium	Medium	Medium
Critical failure modes	Low	Low	Medium	Medium
Energy/environmental impacts	Low	Low	Low	Low

6.1.1 Critical Technology Identification

The HF and VHF satellite concepts require large deployable structures to provide the required aperture sizes for optimum performance. Since there are no deployable antennas developed or under development in the size ranges required for HF or VHF systems, a development program must be established to reduce uncertainty surrounding use of these types of structures. The objectives of this program follow:

1) To demonstrate the capability of the structure to deploy itself and the communication subsystem payload;

2) To demonstrate the capability of the deployed structure to withstand environmentally and internally induced stresses, forces, and torques;

3) To demonstrate the capability of the undeployed structure to tolerate ground handling, system integration, packaging, STS, and upper-stage launch loads;

4) To satisfactorily demonstrate structural element and overall structural fabrication methods;

5) To demonstrate applicability and validity of modeling, simulation, and analysis techniques to analyze large deployable structures.

Space qualified SSPAs are not presently available at the power levels required from the results of this study. Thus a transmitter development program is recommended to:

1) Demonstrate the capability of SSPAs to operate efficiently at powers of at least one kW at the HF- and VHF-bands, and up to 300 watts in L-band;

2) Demonstrate the ability of SSPAs to operate over the desired 7-yr lifetime for the duty cycles and operating environment projected for L, HF, and VHF systems;

3) Demonstrate capability of designing SSPAs to meet packaging and stowage requirements;

4) Demonstrate capability to manufacture SSPAs in the quantities that would be required for L, HF, and VHF satellites.

Another aspect of the HF-, VHF-, and to a lesser degree, L-band systems that involves uncertainty is the array antenna RF performance. A program is recommended to perform the following:

1) Demonstrate the capability to shape and steer beams using phase control at the high powers and over the long distances associated with HF and VHF satellites;

2) Demonstrate satisfactory performance of an array antenna with high-power radiating elements, both with respect to signal at the ground and noninterference with TT&C functions for HF-, VHF-, and L-bands;

3) Evaluate the potential of multipacting in the vacuum of space.

Thermal control for HF-, VHF-, and high-power L-band concepts requires a distributed approach such as the CPL described previously. A program paralleling SSPA development is recommended to:

1) Develop and qualify the CPL technology;

2) Demonstrate the capability to integrate a CPL thermal control system (or other thermal control technique) with the SSPAs and antenna support system, meeting thermal dissipation, packaging, and deployment requirements.

The last critical technology area is solar array technology. The high powers required for HF-, VHF-, and L-band concepts exceed any of current satellites. However, development under the space station program should be applicable to VOA satellite systems. If not, some development will be required to demonstrate the capability to package and deploy solar arrays capable of generating powers of 100 kW and greater; demonstrate the capability to efficiently fabricate the large solar arrays; and to demonstrate the capability of an EPS to regulate and control large powers in cyclic operating modes.

A final technology that is applicable to Ku-band satellites is the development of TWTA technology that decreases operating degradation when operated in a cyclic mode. Work is under way in this area and should be monitored for applicability to VOA Ku-band satellites that will probably operate in a cyclic rather than steady state output mode.

6.1.2 Critical Technology Plans

To meet the technology development goals and reduce uncertainty for antennas, SSPAs, thermal control, and electric power generation, the activities shown in Figure 113A & B should be started early in a VOA program. A flight test of deployable large structures has been proposed by other studies to verify deployment, test flexible body control approaches, and validate analysis techniques. The times shown in these two figures are considered reasonable estimates although they are subjective. As will be shown later, they are consistent with estimates obtained from the NASA SDCM data base. The estimated costs for critical technology development shown in Table 91 are based on CERs discussed in Section 4.3, with the cost range due to the range of CER input parameter values for the various satellite concepts.

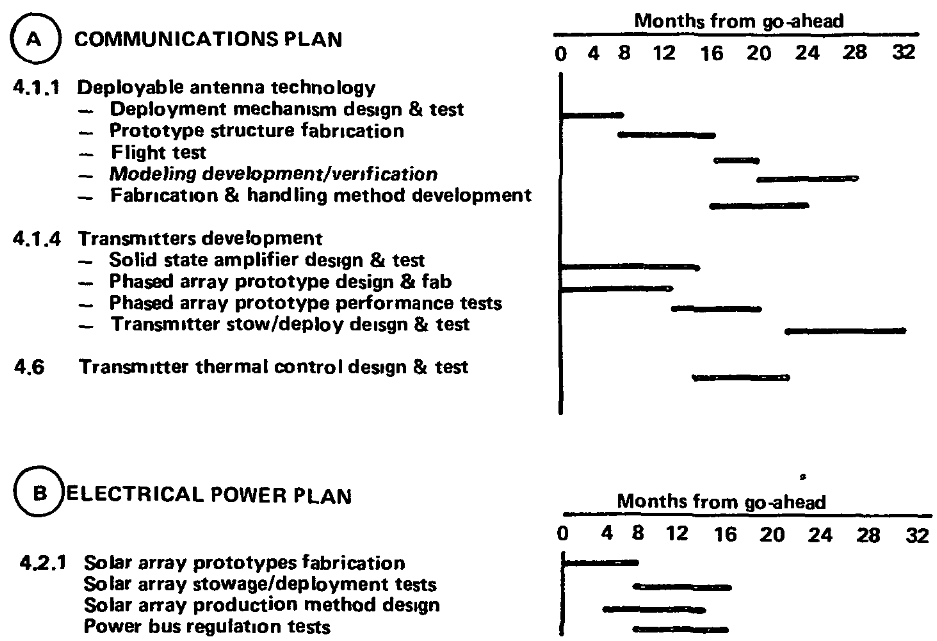

Figure 113. - Communications and electrical power subsystem development plans.

TABLE 91. - CRITICAL TECHNOLOGY DEVELOPMENT COSTS

4.1 1	Deployable antenna technology HF-band VHF-band	 $6 - $10M (+$25M for flight test) $10M - $13M(+$25M for flight test)
4.1.4	Transmitters development HF- & VHF-bands	 $12 - $20M
4.6	Transmitter thermal control development HF- & VHF-bands	 $4 - $10M
4.2	Electrical power subsystem technology HF-, VHF-, & L-bands	 $3 - $35M

6.2 SATELLITE SYSTEMS PROJECT PLANS

The set of top-level functions shown in Figure 114 were identified for a generic DVBS program. Also, the existence of a functional breakdown is required to facilitate a cost and schedule risk analysis using the Martin Marietta risk analysis methodology.

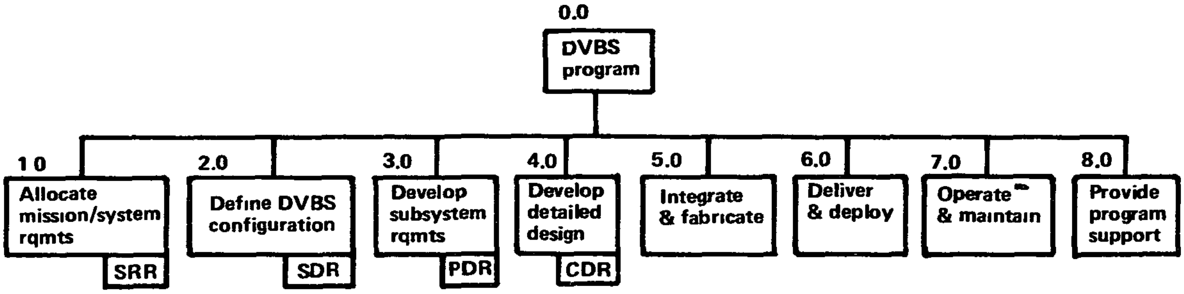

Figure 114. - DVBS planning support—satellite systems project plan.

6.2.1 Functional Decomposition

As an aid to developing estimates of cost and time required to perform the top-level functions, each was decomposed to identify the scope of activities or subfunctions under it. These subfunctions are shown in Figure 115. Program milestones, indicated where applicable, are: (1) System requirements review (SRR), (2) System design review (SDR), (3) Preliminary design review (PDR), (4) Critical design review (CDR).

1.0 Allocate mission & system rqmts

- Define mission rqmts
- Decompose functions
- Derive system rqmts
  - Draft system engineering mgmt plan (SEMP)
  - Draft A spec
- △ SRR

2.0 Define VOA DVBS system

- System modeling
- Tradeoff studies
- Subsystem allocation
- Design baseline
- Configuration drawings
- Interface definitions
- Preliminary B spec
- Final A spec
- Draft B spec
- △ SDR

3.0 Develop subsystem rqmts

- Communications
- Electrical power
- Attitude control
- Auxiliary propulsion
- Telemetry, tracking & command
- Thermal control
- Equipment bay & mechanisms
- Primary propulsion
- Ground control
- Program feed
- Perform trade studies
- Update interface definitions
- Perform subsystem prototype tests
- Draft C spec
- Prepare final B spec
- Update SEMP
- △ PDR

4.0 Develop detailed design

- Communications subsystem
- Electrical power
- Attitude control
- Auxiliary propulsion
- TT&C
- Thermal control
- Equipment bay & mechanisms
- Primary propulsion
- Ground control
- Program feed
- Continue trade studies
- Prepare final interface control document
- Maintain configuration control
- Draft flight operations plan
- Perform system modeling & testing
- Prepare facilities plans
- Prepare final C spec
- △ CDR

5.0 Integrate & fabrication

- Build & test components/subsystems
- Build & test satellite
- Build & test ground stations
- Prepare final payload integration plan
- Perform system acceptance tests

6.0 Deliver & deploy satellite

- Integrate with upper stage
- Install in STS & checkout
- Launch & separate
- Transfer to operational orbit
- Self deploy & checkout (HF & VHF systems)

7.0 Operate & maintain

- Perform onorbit functions
- Feed programs
- Monitor satellite health
- Monitor satellite attitude & position
- Maintain attitude & orbit

8.0 Provide program support

- Systems engineering
- Financial mgmt
- Schedule mgmt
- Risk mgmt
- Contract administration
- Quality control
- Documentation

Figure 115. - DVBS project plan—top-level functions 1.0, 2.0, 3.0, 4.0, 5.0, 6.0, 7.0 & 8.0.

6.2.2 Risk Assessment Procedure

The risk assessment procedure uses a quantitative risk assessment methodology developed at Martin Marietta in an IR&D project during 1982 and 1983. The method has been used on several programs to predict and monitor technical, schedule, and cost risks. The basic approach is not unique, using a Monte Carlo simulation approach. However, the method of implementation represents an advantage over approaches used by others within the industry and the government. The advantage lies in an easily used computer program, RAMP developed to facilitate management of risk data and to simplify access to the Monte Carlo simulation software. The steps involved in the RAMP-augmented risk analysis procedure follow:

1) Obtain risk data estimates (for VOA, low, most likely, and high values) that are used as values of a probability density function;

2) Document the rationale for the estimates;

3) Document the source of the estimates;

4) Enter the estimates and program identifiers into a RAMP data base unique to the program being evaluated;

5) Compute a cumulative probability function for the risk parameter of interest (e.g, cost, months).

Figure 116 shows a risk assessment form containing the three point antenna structure development cost estimate for the small HF design for geostationary orbit. The nonrecurring cost estimates for this satellite, as obtained from the LCC program are shown in Table 92. As indicated on the assessment form, the cost predicted by the LCC model is taken as the highest cost that might occur. Since this is a relatively small satellite (26-meter diameter) work currently in process or planned for deployable box truss structures should result in less required development. Based on past experience with this type of structure, estimates of the most likely cost are predicted to be $2,000 per pound and could conceivably be as low as $1,000 per pound.

Risk assessment can be performed assuming that program functions are independent of or dependent on each other. In the context of quantitative risk analysis, dependence between functions means that if one function occurs at a worst-case value then all dependent functions will also occur at their worst-case values. When functions are independent, the level of one function has no influence on the expected level of others. Since the objective of breaking up a program into a set of functions is to establish a set of relatively independent activities, this independence should be considered when estimating total cost or time to complete. Unfortunately, the practice of summing estimates is frequently used, resulting in over-conservative estimates of cost or time required.

The difference between risks resulting from the two methods is illustrated in Figure 117. For a given parameter value, the risk will be higher for an analysis that assumes dependence. The cost and schedule risk assessments presented here were performed assuming independence. However, a degree of dependence is automatically built in by summing values of recurring, nonrecurring, launch, and ground operations at the fixed risk values of interest (90, 50, and 10%). The steps for the cost risks performed in the study are outlined in Table 93. The steps for schedule risk assessment are listed in Table 94.

RISK ASSESSMENT LOG

Program: VOA small HF, GEO　　　Date: _________

Risk Analyst: ____________________

WBS/Function No 4 1 1

WBS/Function Title· Communications antenna development

Risk Type· Cost

Risk Description· The cost predicted by the MSFC planar array antenna and Navy graphite structure derived CER is considered the highest that might occur It is assumed that DDTE programs for deployable graphite structures will precede VOA satellite programs Thus, the most likely cost is predicted to be $2000 per lb with the lowest at $1000 per lb

Risk Parameter: $M-Nonrec (nonrecurring cost)

Most Likely Value/Level	1.65	Distribution	
Highest Possible	10.7	Type	Triangular
Lowest Possible	0 83		
Risk Weighting Factor	–		
Secondary Effects.			

Data Source MSFC CER, Navy LCC handbook CER, engineering judgement

Mitigation Approach:

Figure 116. - Example of risk assessment log.

TABLE 92. - HF SMALL SATELLITE NRC ESTIMATES

Nonrecurring design & development cost		
Subsystem	Cost, 1984 $	Cost source
Structure (equipment bay)	10.04 $M	Space Division CER
Thermal Control	22.21	Space Division CER
Electrical power	66.18	VOA CER
Communications antenna	10.74	Planar array antenna CER
Communications uplink	13.69	Space Division CER
Signal processor	11.13	Space Division CER
Communications transmitter	15.19	Solid state transmitter CER
Attitude determination	3.55	Space Division CER
Attitude reaction	20.54	Elect propulsion CER
Propulsion	0.00	Throughput
TT&C	5.05	Space Division CER
Aerospace ground equipment	6.58	Space Division CER
Total nonrecurring	184.91	

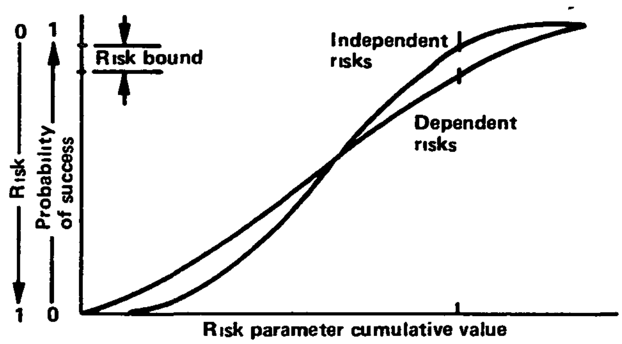

Figure 117. - Quantitative risk assessment interpretation.

TABLE 93. - DVBS PLANNING SUPPORT– COST RISK ANALYSIS APPROACH

LCC estimating program for point estimates
Three point estimates using values from LCC model as starting point – Most likely cost – Highest potential cost – Lowest potential cost Obtain cumulative functions – Nonrecurring spacecraft cost – Recurring spacecraft cost (with learning factor) – Aerospace ground equipment, integration, system engineering, & program cost factors – Launch costs – Ground operations cost Compute total cost risk – Sum four cumulative function results

TABLE 94. - DVBS PLANNING SUPPORT– SCHEDULE RISK ANALYSIS APPROACH

Identify program critical path – Estimates of components/subsystem design + test + qualification times – Select longest time for critical path Estimate program top function times including decision lags between phases Compute quantitative schedule risk

6.3 CRITICAL TECHNOLOGY RISK ANALYSIS

The cost risk assessment cumulative probability functions shown in Figures 118 and 119 are for the HF small design for geostationary orbit. The data estimates used for the antenna development include the estimate for antenna (ref. Fig. 116) and the 3-point estimate for transmitters development described in the risk assessment form of Figure 120. The output plots of Figures 116 and 117 are accompanied by a tabular risk printout (Table 95A & B). These tables identify the cost values corresponding to probability of success. For antenna technology development for this satellite design, there would be a 10% probability that the development would cost $10.8M or less, corresponding to a 90% risk if only $10.8M were allocated for development. The 90, 50, and 10% cost risks for antenna and EPS development, for this satellite, are shown in Table 96, along with the similar costs for the concepts selected as best for the HF-, VHF-, and L-bands. The Ku-band system is not included since it uses existing technology.

The corresponding schedule risk for antenna structure and transmitter technologies is summarized in Figures 121 and 122 and Table 97A & B. The times shown are considered reasonable across the range of satellite concepts studied. The list of component, subsystem, and system development times shown in Table 98 is presented for reference. These times were taken from SDCM documentation, and represent times required on previous satellite development programs.

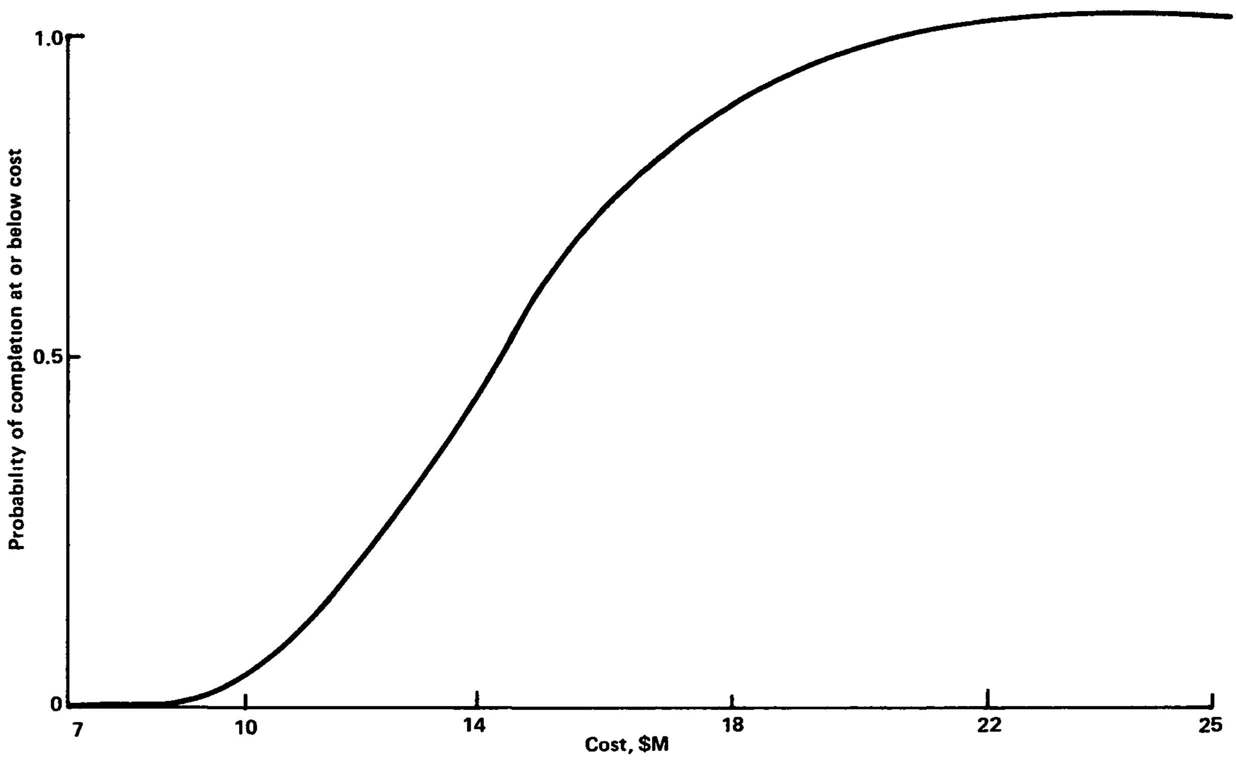

Figure 118. - HF small GEO case antenna technology development cost risk assessment.

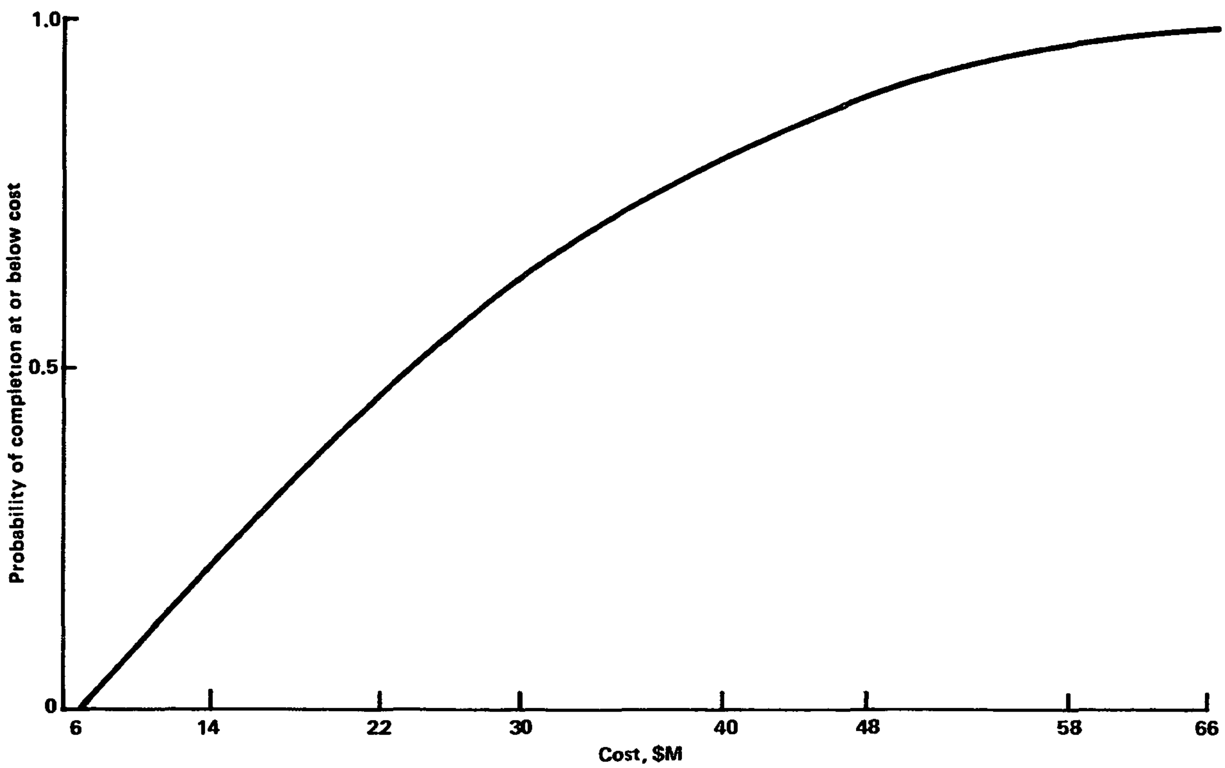

Figure 119. - HF small GEO case EPS technology development cost risk assessment.

RISK ASSESSMENT LOG

Program. VOA small HF, GEO **Date:** ______

Risk Analyst· ______

WBS/Function No. 4 1.4

WBS/Function Title: Transmitter development

Risk Type: Cost

Risk Description: The value predicted by the Space Division LCC model is used as the most likely value. Because higher powers are typically required than for other systems, the highest possible is assumed at 100% over the most likely value. The lowest value is taken to be 10% below the most likely value

Risk Parameter· $M-Nonrec (nonrecurring cost)

Most Likely Value/Level	7.6	**Distribution**	
Highest Possible	15.2	**Type**	Triangular
Lowest Possible	6.8		
Risk Weighting Factor	–		
Secondary Effects:			

Data Source: Space division LCC model, judgement

Mitigation Approach

Figure 120. - HF small GEO case transmitters cost risk log.

TABLE 95. - TABULAR OUTPUT FOR HF SMALL GEO CASE ANTENNA TECHNOLOGY AND EPS COST RISK

Probability	Parameter
0	7.630000
0 05	9.973318
0 10	10.78711
0.15	11.18905
0 20	11.81886
0 25	12.08865
0 30	12.69831
0 35	13.01544
0.40	13.26913
0.45	13.77533
0 50	14.04413
0 55	14.50329
0 60	14.88284
0 65	15.15558
0.70	15.57181
0.75	16 12199
0 80	16.76693
0 85	17.56044
0 90	18.36511
0 95	20.13933
1.00	25 90000

(A) ANTENNA TECHNOLOGY COST RISK

Antenna technology development cost, 1984, $M
- Risk parameter, $M—antenna development
- Independent risks
  - Number of passes, 200
  - Number of intervals, 20

Probability	Parameter
0	6.600000
0.05	8.109104
0.10	9.658474
0.15	11.25151
0 20	12.89214
0.25	14.58488
0.30	16.33506
0 35	18.14984
0 40	20.03403
0.45	21.99945
0 50	24.05643
0.55	26.21910
0.60	28.50565
0.65	30.94016
0.70	33.55573
0.75	36.39999
0 80	39.54607
0.85	43.11702
0.90	47.35281
0.95	52.87302
1.00	66.20000

(B) EPS COST RISK

EPS technology development cost, 1984 $M
- Risk parameter, $M-EPS development
- Dependent risks
  - Number of passes, 200
  - Number of intervals, 20

TABLE 96. - CRITICAL SUBSYSTEM TECHNOLOGY COST RISK

System	Antenna technology cost risk			EPS technology cost risk		
	90%	50%	10%	90%	50%	10%
HF small GEO	10 8	14.0	18.4	9 7	24.1	47 4
HF, 8 h	18 8	28 2	40.9	8 6	25.0	48.9
VHF, 24 h	25.2	45.2	73.3	8 2	18.7	35.7
L-band, option II	7 0	17 5	34.3	12.4	15 9	18 1
L-band, option III	10.6	12.1	14.3	13.6	33.8	66.5

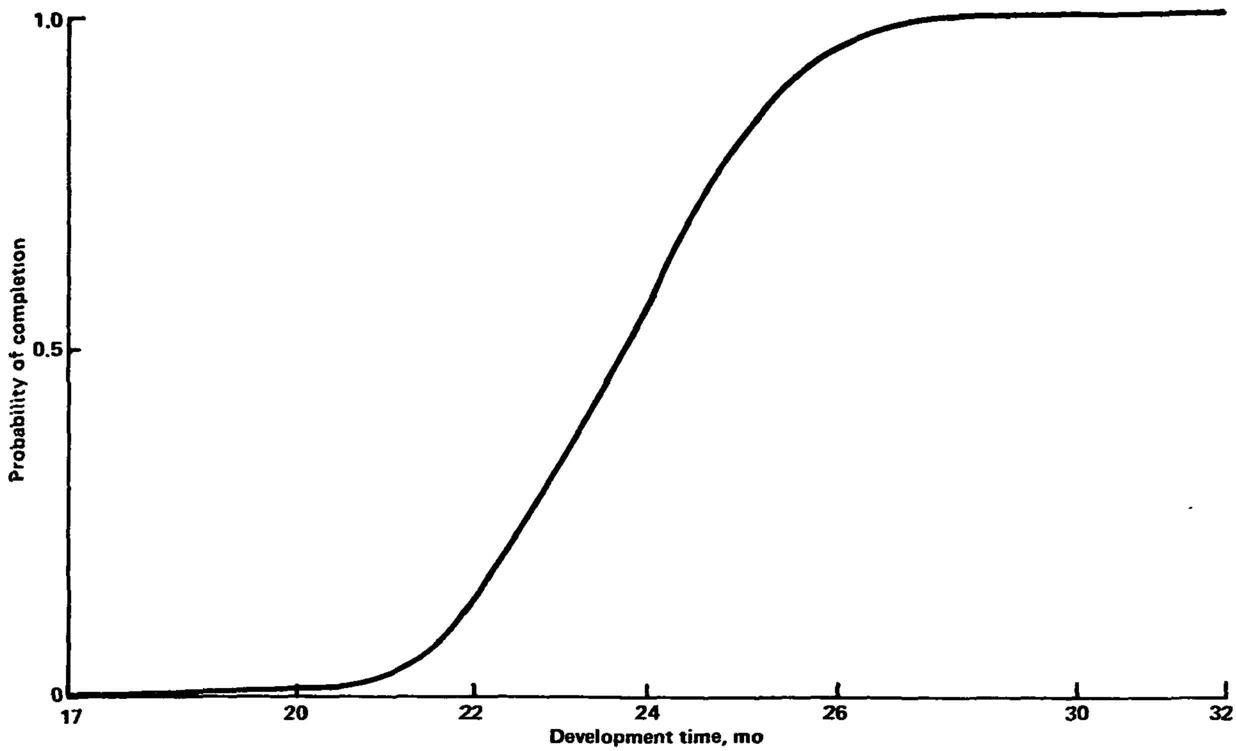

Figure 121. - Antenna technology development schedule risk assessment.

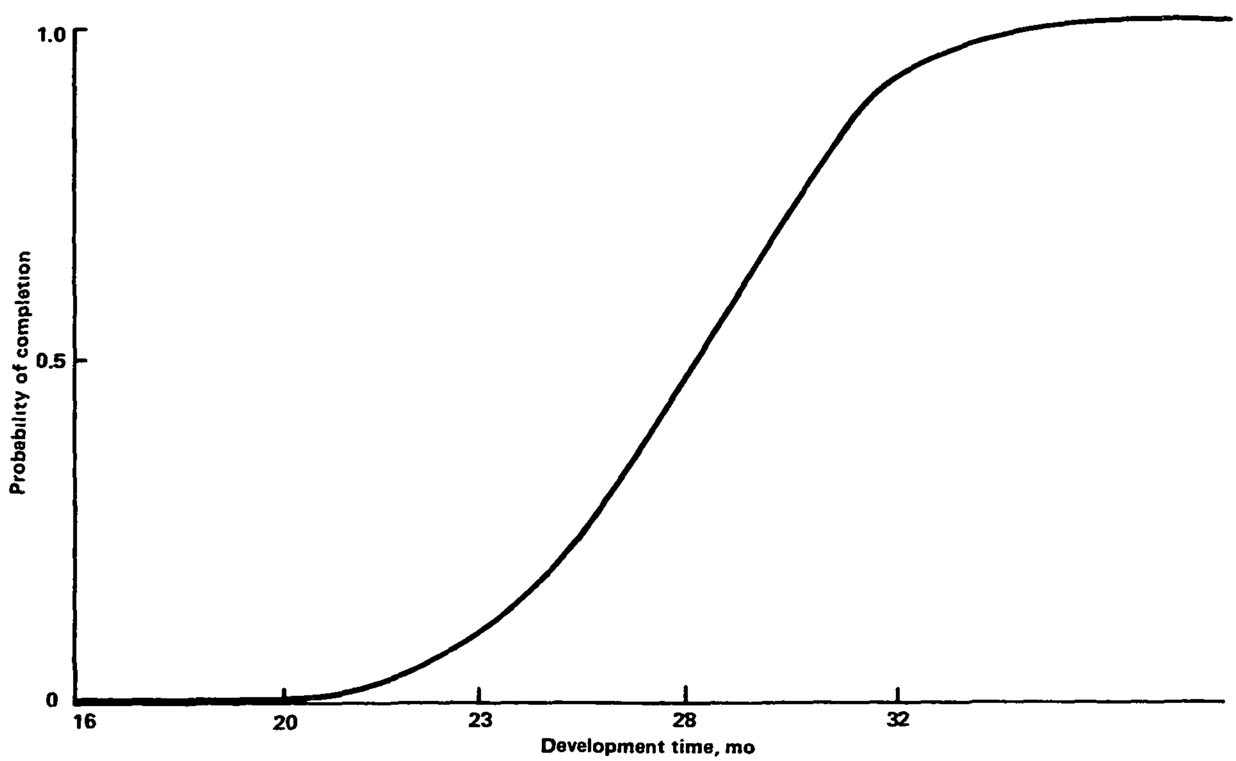

Figure 122. - Solid state transmitter technology development schedule risk assessment.

TABLE 97. - ANTENNA TECHNOLOGY AND SOLID-STATE TRANSMITTER DEVELOPMENT SCHEDULE RISKS

Probability	Parameter	Probability	Parameter
0	17.00000	0	16 00000
0 05	21.48229	0.05	23.0931
0.10	21.90552	0.10	23.93056
0.15	22.03702	0 15	25.04383
0.20	22.43647	0 20	25.71787
0.25	22.74920	0 25	26.13156
0 30	22 97231	0 30	26.81432
0 35	23 32456	0.35	27.43176
0.40	23.56778	0.40	27 65512
0 45	23.99574	0 45	28.08930
0.50	24.07534	0 50	28.49456
0.55	24.41297	0.55	29.00667
0.60	24.64095	0.60	29.50549
0.65	24.79995	0.65	29.84713
0.70	25.02822	0.70	30.58662
0.75	25.31348	0 75	30 79152
0.80	25.68262	0 80	31.62915
0 85	26.00193	0.85	32.19074
0.90	26.33169	0.90	32.72165
0.95	27 14575	0.95	33.44208
1.00	32.00000	1.00	40.00000

A ANTENNA TECHNOLOGY

Deployable array antenna schedule time, months
- Risk parameter, antenna months
- Independent risks
  - Number of passes, 200
  - Number of intervals, 20

B SOLID-STATE TRANSMITTER

Solid-state transmitter development time, months
- Risk parameter, SSPA months
- Independent risks
  - Number of passes, 200
  - Number of intervals, 20

TABLE 98. - COMPONENT AND SUBSYSTEM DEVELOPMENT TIME ESTIMATES FROM SDCM

Subsystem	T_{cd}*	T_{cq}*	T_{sd}*	T_{sq}*	T_{sysq}*
Communications			2.8	3 0	2.9
– Antenna	–	–			
– Transmitters	9.2	5.0			
– Feed link	9.4	4.5			
– Signal proc	8.0	5 0			
Electrical power	14 0	3 0	–	10 0	10 9
Thermal control	10.0	3 0	1 4	2 9	10 0
Attitude control	11 9	3.0	8.3	3 0	19 4
Auxiliary propulsion	5.7	2.5	8.2	3 0	10.3
TT&C	10.5	2 0	3.0	1 0	2.7

*T_{cd} – Component development time
T_{cq} – Component qualification time
T_{sd} – Subsystem development time
T_{sq} – Subsystem qualification time
T_{sysq} – System qualification time

6.4 SATELLITE SYSTEMS COMPOSITE RISK ANALYSIS

The final set of analyses are the cost and schedule risk assessments for the satellite systems studied over the course of the contract. Tables 99 through 102 summarize the cost risks for each systems' satellites (L-band uses an average cost per satellite as described in Section 4.3). In the tables, the 10/90 heading is a 90% risk level, while the 90/10 heading is a 10% risk level. All costs are in millions of 1984 dollars.

Table 103A, B & C, along with the corresponding Figures 123, 124, and 125 summarize the schedule risk assessment from phase B onward for the various systems. The 3-point time estimates shown in the tables are identified by the top-level function numbers described earlier. The similarities between the HF and VHF systems make it reasonable to use the same assessment for each. To summarize from the figures, the schedule risk values in months, starting with function 3.0 (Phase B) follow:

	10/90	50/50	90/10
Ku-band system	37.0 months	39.9	43.3
L-band system	50.5	54.6	61.4
HF/VHF-band system	59.4	63.5	69.1

TABLE 99. - KU-BAND SYSTEM COST RISK ASSESSMENT RESULTS

System	Nonrecurring costs			Recurring costs		
	10/90	50/50	90/10	10/90	50/50	90/10
Satellite No. 3	$ 64 M	$ 76 M	$ 94 M	$ 51 M	$ 58 M	$ 68 M
Launch No. 3				50	62	155
Ground operations	6	9	12			
Totals No. 3	70	85	106	101	120	223
Satellite No. 2	6	8	13	47	55	61
Launch No. 2				50	62	155
Ground operations	6	9	12			
Totals No. 2	12	17	25	97	117	216
Satellite No. 1	8	10	15	36	40	44
Launch No 1				50	62	155
Totals No. 1	8	10	15	86	102	199
System totals	90	112	146	284	339	638

TABLE 100. - L-BAND SYSTEMS COST RISK ASSESSMENT RESULTS, 1984 $

System	Nonrecurring costs			Recurring costs		
	10/90	50/50	90/10	10/90	50/50	90/10
Eight S/C at -103.6 dBW	$107 M	$130 M	$169 M	$ 568 M	$ 656 M	$ 760 M
– Launches	–	–	–	840	1000	1400
– Ground operations	12	18	24	–	–	–
– Totals	119	148	193	1408	1656	2160
Option I (5 S/C)	82	98	119	195	215	245
– Launches	–	–	–	292	365	510
– Ground operations	12	18	24	–	–	–
– Totals	94	116	143	487	580	756
Option II (3 S/C)	90	113	144	132	147	168
– Launches	–	–	–	187	222	311
– Ground operations	12	18	24	–	–	–
– Totals	102	131	168	319	369	479
Option III (3 S/C)	104	135	190	150	168	178
– Launches	–	–	–	150	221	310
– Ground operations	12	18	24	–	–	–
– Totals	116	153	214	300	389	488

TABLE 101. - VHF SYSTEM SINGLE SATELLITE COST RISK ASSESSMENT RESULTS, 1984 $

System	Nonrecurring costs			Recurring costs		
	10/90	50/50	90/10	10/90	50/50	90/10
12-h orbit	$218 M	$252 M	$306 M	$138 M	$156 M	$187 M
Launch				105	125	250
24-h orbit	217	258	336	121	136	162
Launch				105	125	250
24-h orbit, multiple launch	242	304	384	153	178	220
Launch				210	250	500

TABLE 102. - HF-BAND SYSTEMS SINGLE SATELLITE COST RISK ASSESSMENT RESULTS, 1984 $

System	Nonrecurring costs			Recurring costs		
	10/90	50/50	90/10	10/90	50/50	90/10
6-h orbit	$207 M	$247 M	$301 M	$138 M	$161 M	$195 M
6-h orbit, mult launch	237	297	376	201	235	276
8-h orbit	228	271	338	153	170	217
8-h orbit, mult launch	244	305	382	185	230	290
12-h orbit, mult launch	234	285	358	162	190	233
6-h small max pyld	220	265	332	140	170	221
6-h small 50 kW DSB	206	239	279	129	150	178
6-h small 50 kW SSB	197	231	270	115	133	159
8-h small max pyld	227	267	331	166	199	250
8-h small 50 kW DSB	208	241	291	127	147	176
8-h small 50 kW SSB	200	227	259	112	126	154
12-h small max payload	206	247	300	105	142	199
12-h small, 50 kW DSB	195	225	268	130	152	182
12-h small 50 kW SSB	180	210	249	116	131	151
Triply synchronous orbit small max payload	186	209	241	94	105	122
Geostationary orbit small, max payload	194	222	258	124	141	171
24-h elliptical orb small, max payload	209	245	293	149	172	217
24-h, 50 kW DSB	194	225	265	130	151	179
24-h, 50 kW SSB	180	199	246	117	134	157

TABLE 103. - KU-, L-, HF-, AND VHF-BAND PROJECT FUNCTION 3-POINT SCHEDULE ESTIMATES

(A) KU-BAND

Function/activity	Estimates, months		
	Low	Most likely	High
1.0 Allocate mission/system requirements (SRR)	3	4	5
Decision	0	1	1
2.0 Define DVBS configuration (SDR)	6	6	9
Decision	1	1	2
3.0 Develop subsystem requirements (PDR)	9	10	12
Decision	1	2	3
4.0 Develop detailed design (CDR)	9	12	18
Decision	1	1	2
5.0 Integrate & fabricate	6	9	12
6.0 Deliver & deploy	3	4	5

(B) L-BAND

Function/activity	Estimates, months		
	Low	Most likely	High
1.0 Allocate mission/system requirements (SRR)	3	4	5
Decision/RFP	0	1	1
2 0 Define DVBS configuration (SDR)	6	6	9
Decision/RFP	1	2	3
3.0 Develop subsystem requirements (PDR)	9	10	15
Decision	1	2	3
4.0 Develop detailed design (CDR)	15	18	30
Decision	1	1	2
5.0 Integrate & fabricate	12	15	18
6.0 Deliver & deploy	3	4	5

(C) HF- AND VHF-BANDS

Function/activity	Estimates, months		
	Low	Most likely	High
1.0 Allocate mission/system requirements (SRR)	3	4	5
decision/RFP	0	1	1
2.0 Define DVBS configuration (SDR)	6	6	9
decision/RFP	1	2	4
3.0 Develop subsystem requirements (PDR)	12	12	18
decision	1	2	3
4.0 Develop detailed design (CDR)	18	20	30
decision	1	1	2
5.0 Integrate & fabricate	15	20	24
6.0 Deliver & deploy	3	4	5

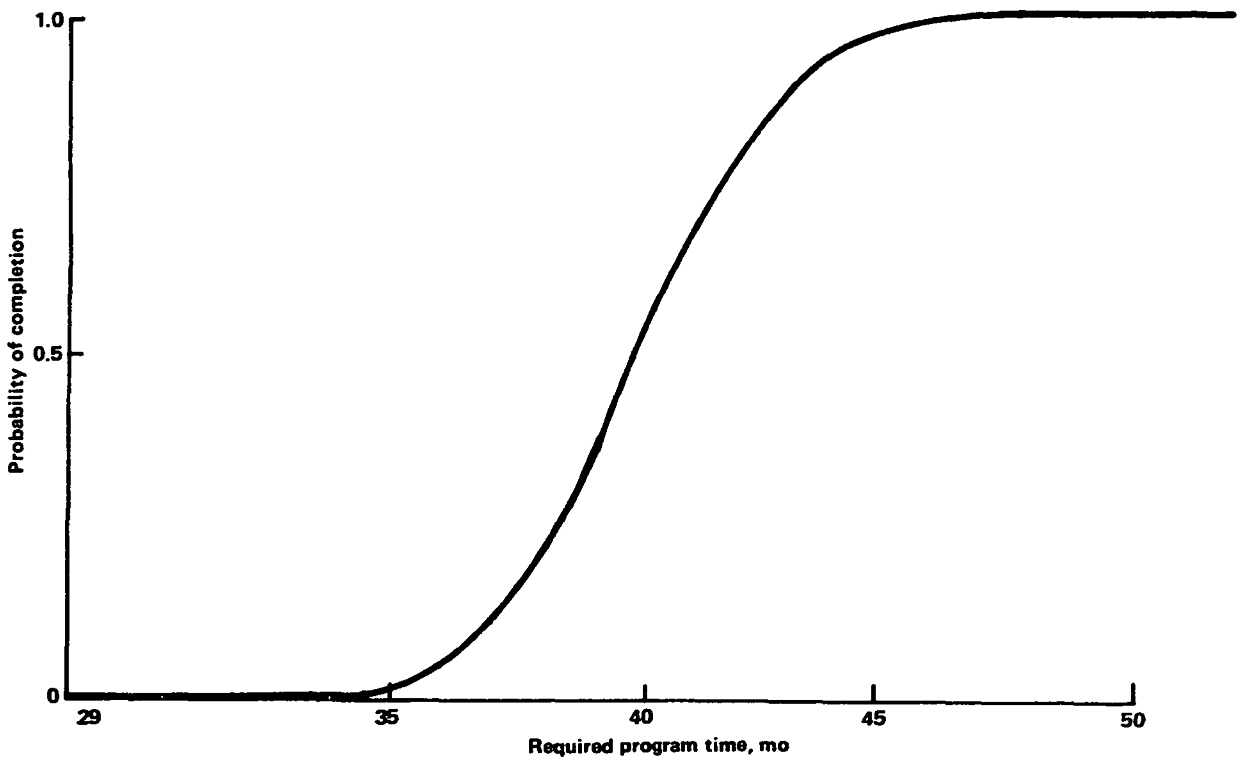

Figure 123. - Ku-band program schedule risk assessment results.

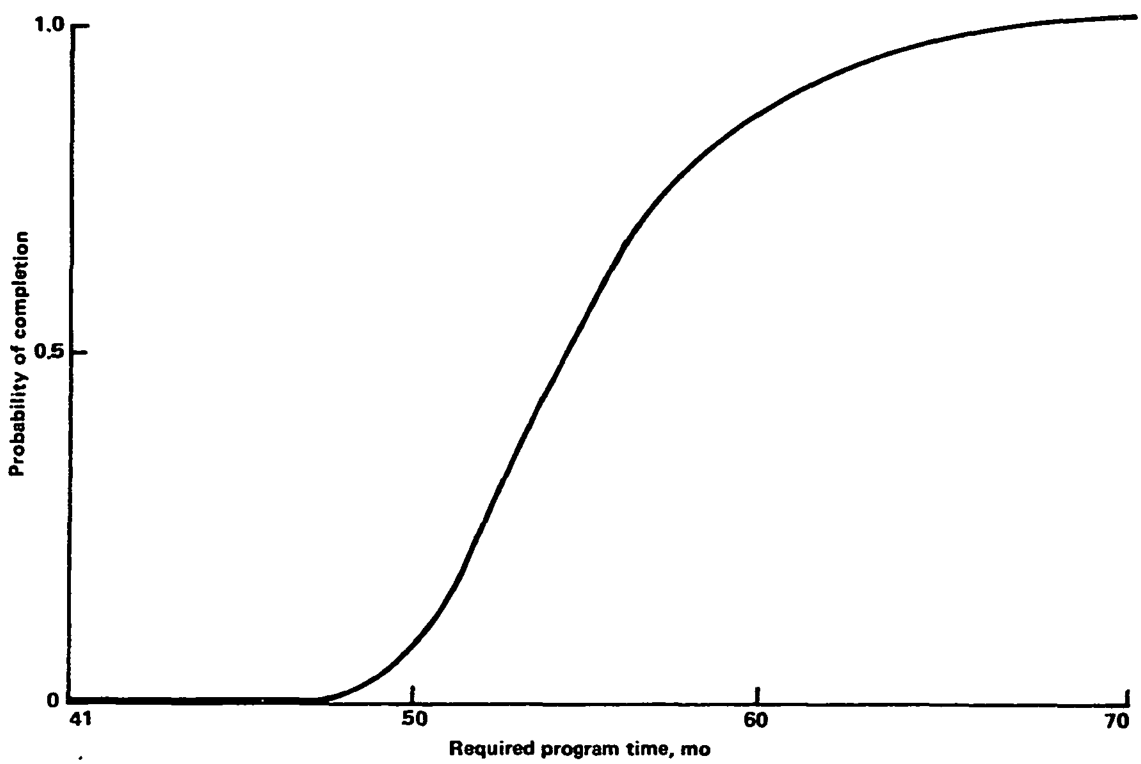

Figure 124. - L-band program schedule risk assessment results.

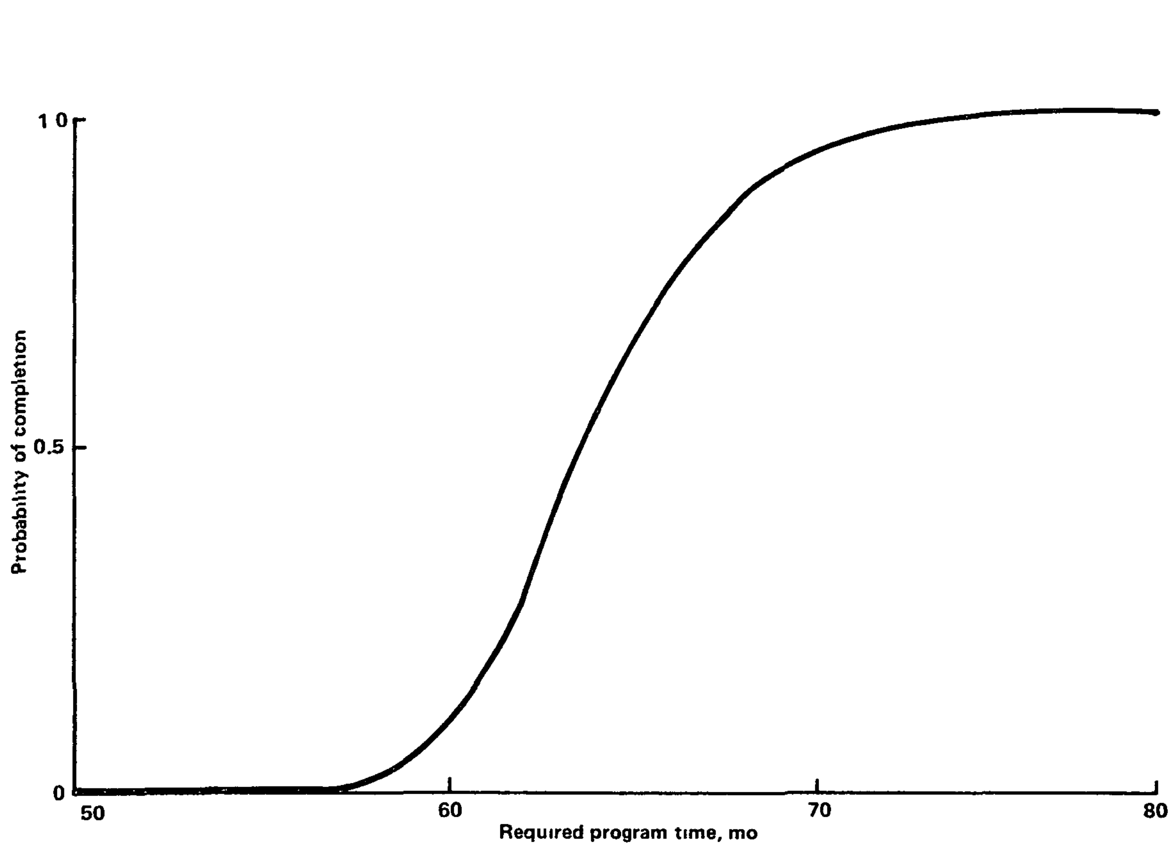

Figure 125. - HF and VHF program schedule risk assessment results.

7.0 CONCLUDING SECTION

Section 7.0 summarizes the key results of the Satellite Voice Broadcast System study which investigated the feasibility of direct voice broadcasting from space. Also presented are some conclusions that can be drawn from the results. It is anticipated that in the future as the results are studied further, additional conclusions will be drawn.

7.1 SUMMARY OF RESULTS

Nonorbital, nonterrestrial broadcast techniques are unable to meet the desired coverage even using large numbers of platforms. Both the numbers and resulting cost of the systems are excessive. Also, the nonorbital techniques evaluated are severely power limited and therefore, cannot penetrate into unfriendly territory as well as terrestrial systems.

Orbital techniques using derivatives of existing geostationary satellites can meet Ku-band requirements. L-band systems could be used at the lower power flux density (PFD) requirements of -116.1 dBW/m^2. VHF and HF do not exist either with aperture or power subsystems to meet even the minimum signal strength requirement. Table 104 summarizes the results of existing nonterrestrial broadcast techniques.

TABLE 104. - SUMMARY OF PROGRAM RESULTS FOR NONTERRESTRIAL BROADCAST TECHNIQUES

Summary of results
Nonorbital techniques – Not useful in unfriendly territory – Many (19 to 719) platforms needed for a single zone.
Orbital techniques – Practical systems operate only at geostationary orbit – Beam size in HF- & VHF-bands larger than Earth, and power is prohibitive. – L-band system could work with existing broadcast technique for minimum PFD requirement only – Ku-band systems may work with existing SBS type satellite technology.

The results of the Ku-band system design are summarized in Table 105. All VOA requirements could be achieved using existing technology and low program cost. Three satellites are required resulting in a total life-cycle cost of $1240M for a 20-yr operational lifetime with a corresponding cost per channel hour of $568. Figure 126 shows the proposed Ku-band satellite.

TABLE 105. - SUMMARY OF PROGRAM RESULTS FOR KU-BAND SYSTEM

3 Ku-band satellites in geostationary orbit meet program requirements. – Uses existing technology – LCC = $1,240M – Cost/channel hour = $568/channel hour – TOS/AMS launch vehicle – 100% coverage

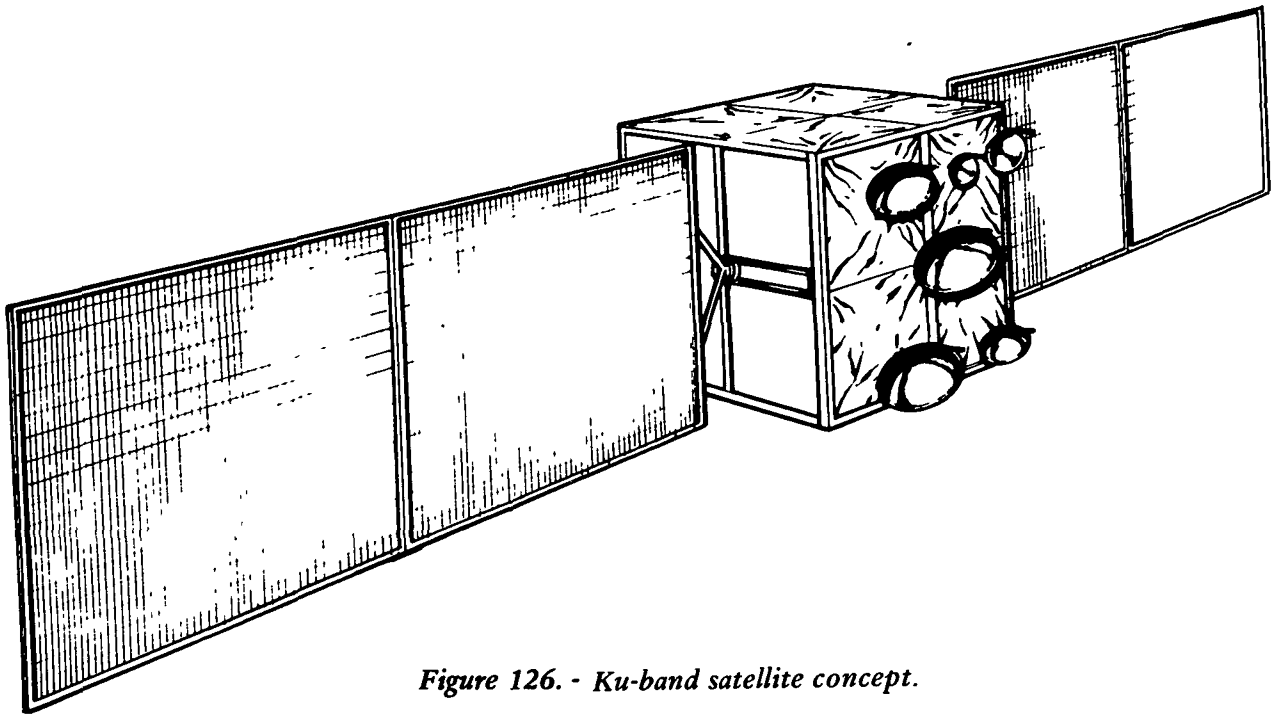

Figure 126. - Ku-band satellite concept.

The results of the L-band system design are summarized in Table 106. This design option uses the lower power requirement of -116.1 dBW/m^2. Each of three satellites has two antenna array apertures that cover multiple zones with one large spot. All VOA requirements could be achieved using existing technology and low program cost. The 17-kW power subsystem would require the use of SAFE array technology that was demonstrated by NASA. The three satellites have a 20-yr operational lifetime cost of $1353M and a cost per channel hour of $619. Two other low-power options and one high-power option were also studied. Figure 127 shows the proposed L-band satellite.

TABLE 106. - SUMMARY OF PROGRAM RESULTS FOR L-BAND SYSTEM

3 L-band satellites in geostationary orbit meet program requirements.
– Uses existing technology for -116.1 dBW/m^2
– LCC = $1,353M
– Cost/channel hour–$619/channel hour
– Requires 17 kW on satellite
– Two array antennas per satellite
– TOS/AMS launch vehicle
– 100% coverage

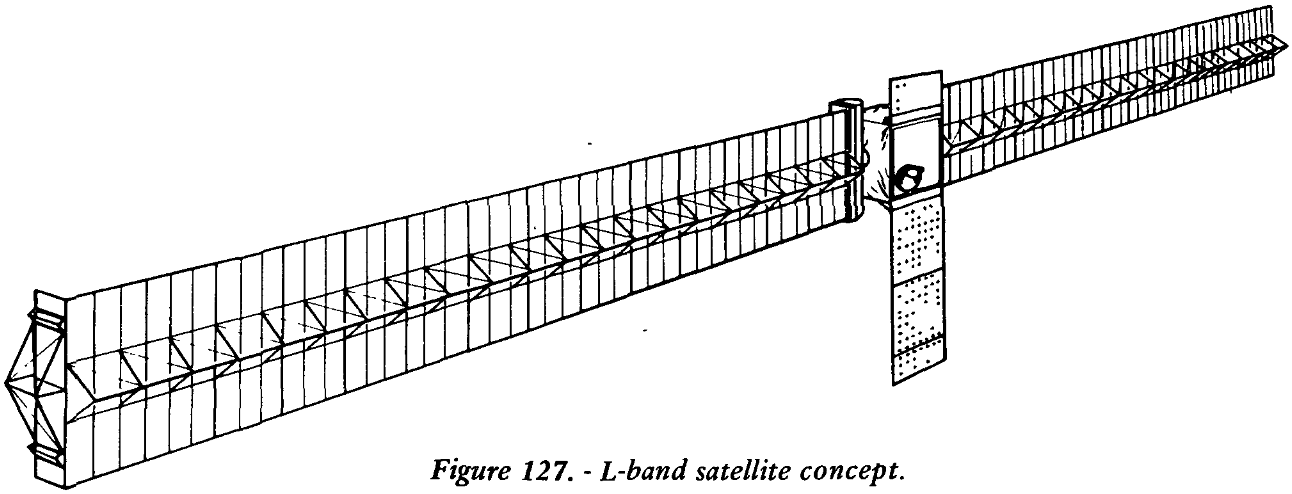

Figure 127. - L-band satellite concept.

The results of the VHF-band system design are summarized in Table 107. A four satellite constellation in a 24-hour elliptical orbit provides 140-150 μV/m performance at the minimum cost. The 12-hour orbit had a higher PFD (250 μV/m) but required eight satellites and had lower coverage efficiency. A deployable scanning array is used for the aperture and a deployable 31.3 kW solar array flight experiment (SAFE) type solar array was used. The four satellites have a 20-yr operational lifetime cost of $2945M with cost per channel hour of $8584. Figure 128 shows a proposed VHF-band satellite.

TABLE 107. - SUMMARY OF PROGRAM RESULTS FOR VHF SYSTEM

Four VHF-band satellites in a 24-h elliptical orbit can meet 140-150 μV/m requirement – New technology for 168 m array & transmitters – 31.3 kW satellite power existing technology (SAFE) – LCC = $2945 – Cost/channel hour—$8584/channel hour – Centaur G launch vehicle – 100% coverage

The results of the HF-band system showed that excessively large numbers of satellites are required (88) to meet all zone and channel requirements at 300 μV/m. By reducing requirements to 150 μV/m a constellation of eight satellites can provide the number of channels shown in Table 108. New technology is required for both the array antenna and the power subsystem. It is anticipated that space station will develop the technology for a 100 kW solar power system. The eight satellites have a 20-yr operational lifetime cost of $5862M and a cost per channel hour of $3225M. Although the total program cost for the HF system is the highest, the cost per channel is lower than the VHF system and only four times the cost of the Ku- or L-band systems. Figure 128 shows a proposed HF-band satellite.

Table 109 presents a cost comparison of the four systems. The HF system has the highest LCC, even at the reduced power level and reduced channel capability. The VHF system has the highest cost per channel hour due to the reduced zones being covered. The VHF spacecraft have the lowest satellite utilization factor.

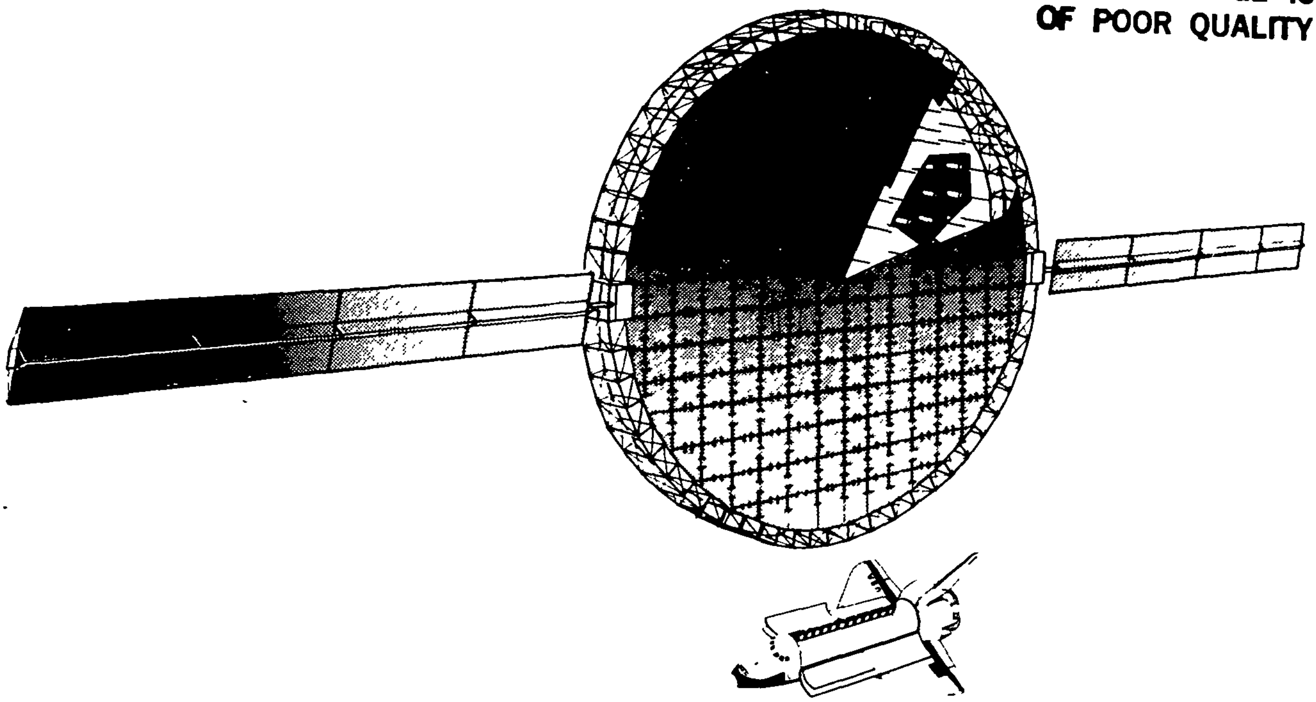

Figure 128.- VHF and HF satellite concepts.

TABLE 108. - SUMMARY OF PROGRAM RESULTS FOR HF SYSTEM

Eight HF-band satellites in an 8-h circular orbit can meet 150 μV/m requirement with reduced channel capability.
- New technology for 80 m array & transmitters
- 93.1 kW satellite power requires space station technology

Zone	1	2	3	4	5	6	7	8	9	10	11	12	13	14	15
No. of channels	4	5	1	4	4	4	2	5	6	3	3	4	2	4	2

- LCC = $5,862 m
- Cost/channel hour—$3,225/h
- Centaur G launch vehicle
- 68% coverage frequency

TABLE 109. - COST COMPARISON FOR KU-, L-, VHF-, & HF-BANDS

	No. of spacecraft	LCC	Cost/channel hour
Ku-band	3	$ 1,240M	$ 568/h
L-band (-116.1 dBW/m^2)	3	1,353	619
VHF-band (140 - 150 μV/m)	4	2,945	8,584
HF-band (150 μV/m, reduced channel capability)	8	5,862	3,225
HF-band (300 μV/m, full VOA)	88	52,824	24,181

7.2 CONCLUSIONS

The study conclusions are based only on this system study and do not include other significant considerations being evaluated by NASA LeRC and VOA (e.g, receiver population and distribution).

1) Terrestrial systems have the advantage of coverage over nonorbital, nonterrestrial systems because the high power and resulting signal skips for terrestrial systems is more significant in expanding coverage than raising altitude (increasing line of sight) to expand coverage. It is difficult if not impossible to achieve 100-500 kW power levels on either lighter-than-air or heavier-than-air vehicles. The nonorbital, nonterrestrial system does have potential to provide local coverage where existing fixed sights do not exist.

2) Orbital systems can expand coverage for VOA beyond existing terrestrial systems. Orbital systems can be used as an augmentation to the terrestrial system or as a potential replacement. Costs become significant when a full capability orbital replacement system for the HF terrestrial system is considered.

3) Cost of the system increases as the operation frequency decreases.

4) HF is desirable due to high ground receiver population, but antenna size is large and transmit power levels are high. Because of high power requirements at greater than 300 $\mu V/m$, future tests are desirable to determine if a reduced level (150 $\mu V/m$) could be received on the ground with adequate S/N ratio.

5) Both HF and VHF systems require technology development for both the power subsystem and the array antenna. The Ku-band system uses OTS technology. The L-band system uses OTS technology except for the array antenna and deployable solar array which are SOA.

6) HF and VHF array antennas have many advantages over reflectors including low power per transmitter, simple thermal control, electronic beam steering, higher reliability (graceful degradation if transmitter fails).

7) Parametric performance studies showed that power generation using deployable photovoltaic solar arrays were superior to other systems based on specific weight (kW/kg) packing volume (kW/stowed volume).

8) Optimizing the HF and VHF VOA coverage requirements for the selected orbit coverage characteristics can improve satellite utilization factors and provide more operational hours and thus reduce cost per channel hour. Also reducing the peak multichannel requirements can improve both the satellite utilization factor and cost per channel hour. Tasking of a satellite to cover more than one zone simultaneously with multiple beams (when satellite power and channel capability is available after covering first zone) can improve both coverage and satellite utilization factors producing lower cost per channel hour.

9) For subsynchronous orbits with multiple satellites, the use of two ground stations with satellite cross linking was more cost effective than additional ground stations and no satellite cross linking.

10) Two smaller HF systems: (1) two satellites in an 8-hour, 0° inclination orbit or (2) two satellites in a triply-synchronous orbit can provide VOA programming with reduced signal strengths but with repeating ground coverage times. This system has significantly lower cost than the full capability system and could be used as a low-cost startup system to augment the terrestrial system.

APPENDIX - COMPUTER PROGRAM DOCUMENTATION

This appendix contains descriptions of and user instructions for computer programs developed and used during the study of direct broadcast satellite systems. The first, life-cycle cost (LCC) estimating program (LCCOST), provides estimates of satellite LCCs. The second, RFANAL, performs calculations to estimate RF performance. A third set of programs is also provided although they were not developed during the study. These programs include SUBSIZ Voice of America (VOA), EPSSIZ VOA, and RFSIZ VOA. Together they provide estimates of weight and volume for satellite subsystems. These estimates and other parametric data can be used to automatically interface with LCCOST to estimate LCC. The three programs are derivatives of Martin Marietta Denver Aerospace programs that have been modified to reflect assumptions made for sizing VOA satellites as discussed in Sections 3 and 4 of the Satellite Voice Broadcast Final Report.

LCCOST

Following is a description of the LCC estimating program developed for use in the VOA satellite study. The LCCOST program was developed to run in a standalone mode or interface with outputs from SUBSIZ. This document discusses the standalone capabilities of LCCOST as provided to NASA.

The objective of the program is to provide an efficient means to estimate and tradeoff LCC of spacecraft concepts. LCC estimates are broken out by subsystem for:

1) Attitude control (attitude determination),

2) Stationkeeping and maneuvering reaction control system (RCS),

3) Primary propulsion,

4) Communication payload,

5) Electrical power subsystem,

6) Thermal control,

7) Telemetry, tracking, and command (TT&C),

8) Structure (equipment bay).

Following are descriptions of the LCC estimating approach for each of the subsystems required for VOA direct broadcast satellite systems. The LCC estimating algorithms and cost estimating relationships (CER) have been coded and tested on an IBM PC. This program provides an efficient tool for trading off different subsystems and for determining sensitivity of LCC to weight, stowed volume, and electric power for different system requirements.

LCC Estimating Procedure Overview

In the standalone mode, LCC estimates for each spacecraft configuration are determined from subsystem-specific input data obtained from some data base source (such as the NASA LaRC system design and cost model (SDCM) or IDEAS programs) and input manually. The input data are applied to user-selected CERs. The CERs contained in the program were described in Section 4. These CERs are used to compute cost at the subsystem level, with subsystem costs summed to provide total spacecraft nonrecurring cost and recurring first unit cost. Additional CERs are included to permit estimation of launch, ground support, and project management costs.

The output of the LCC program is a breakout of nonrecurring, recurring, launch, and ground costs by subsystem, by combinations of subsystems, or for a total system. In the standalone mode, sensitivity of cost to spacecraft design parameters is determined by manual input of new values for CER input parameters.

To estimate LCC for multiple spacecraft, it is necessary to determine an average unit cost (C) by using a learning factor. A commonly used approach uses the expression:

$$C = C_{u1}/N_u^b \qquad (1)$$

where: $b = \text{Log}(m)/\text{Log}(2)$

m is a learning factor slope expressed in %;

C_{u1} is estimated first unit cost;

N_u is total number of units to be produced

For spacecraft, a conservative value for m is 0.95, with an optimistic value of 0.8. The total recurring cost can then be obtained by multiplying C times N.

The CERs used to estimate spacecraft LCC include data obtained and/or refined during the VOA study contract and CERs based on cost data from the NASA SDCM, the NASA MSFC LCC data base, the REDSTAR LCC data base, and the USAF Space Division LCC data base. The CER contained in these models vary at the subsystem level primarily because of different assumed subsystem configurations in the different LCC models. The LCC program therefore includes the capability to select different CERs for different subsystems. Selection of the CER for a subsystem is at the discretion of the user. By using different CERs on subsequent runs, an estimate of cost estimating uncertainty can be obtained by comparing the differences in subsystem and system LCC estimates. If desired, the different results can be averaged to compute an average cost. The results can also be weighted manually to reflect the similarity (or dissimilarity) of each CER's subsystem configuration to the VOA satellite subsystem configuration.

Subsystem LCC Estimating Descriptions

Following are brief discussions of strategies for CER inputs recommended for specific subsystems and program activities.

1) Communication Subsystem--The communication subsystem includes the communications antenna(s), uplink components, signal processing components, and downlink components. For large space system (LSS) antennas, the downlink components are identified as the transmitters. If conventional (non-LSS) spacecraft costs are being estimated, downlink component weight should be distributed into the uplink and signal processing components. This is also the case if a passive LSS antenna (e.g. radiometer) is being considered. The VOA CERs for transmitters' recurring cost were obtained from discussions with potential vendors and are considered better for solid-state electronics than any of the other model CERs included in program LCCOST. A learning factor is built into the program to account for multiple transmitter assemblies for a single planar array antenna.

2) Electric Power Subsystem--For spacecraft requiring more than 10 kW from the source, the VOA CER should be used. For power requirements less than 10 kW, the CER derived from the MSFC LCC model may be better. Also, the VOA CER assumes a much smaller proportion of cost for batteries than is common for current spacecraft.

3) Reaction Control Subsystem (RCS)--The CER derived from the Space Division model should be used if the RCS uses chemical thrusters. If electric propulsion is used, the derived electric propulsion CER is applicable. This CER was obtained from results of electric propulsion studies found in the literature and by studies performed at the Denver Aerospace company.

4) Other Subsystems--For all other spacecraft subsystems, the Space Division model subsystem configuration assumptions are considered to provide the best fit with VOA or most other spacecraft. Thus, if CERs are used to estimate costs, the Space Division CERs are recommended. However, considerable data exists defining costs for the more common subsystem configurations. If costs of subsystem components are known from prior programs, the best approach is to use the throughput option of LCCOST.

5) Launch Costs--Launch costs are separated into STS launch cost and orbit transfer (upper stage) cost. The STS launch cost algorithm assumes non-subsidized costs, and computes cost as a function of required weight or volume fraction of the STS payload. The STS launch cost is computed as follows:

Cost = (dedicated launch cost + user fee + escalated user fee) times payload charge factor

where charge factor is smaller of 1.0 or load factor divided by 0.75; Load factor is larger of [spacecraft (S/C) + upper stage length] divided by 60 ft or (S/C + upper stage weight) divided by 65000 lb; dedicated launch cost is $210M; user fee is $4.3M; escalated user fee is $37M

For subsidized STS launch costs, a throughput value can be obtained by assuming $20,000 per pound for a Centaur G upper stage, or $19,000 per pound using a TOS/AMS upper stage.

6) Ground Costs--Aerospace ground cost, a nonrecurring cost, accounts for special tools, facilities, and test equipment necessary to support production and is computed as a function of total spacecraft weight. Ground operations cost includes the ground control station facility cost, a nonrecurring cost, and operations cost, which depends on the number of spacecraft to be monitored. No CER was identified that adequately represents this cost category. Therefore, use of the throughput option is recommended.

7) Other Costs--Program-level costs not directly associated with a specific subsystem must be included to account for such activities as program management, systems engineering, quality control, contract management, finance management, etc. The LCCOST program computes these costs as a user defined percentage of total nonrecurring and recurring costs. The last cost category included in the overall cost estimate is the program fee, again computed as a user defined percentage of total cost.

LCCOST User Instructions

The LCCOST program has two different execution modes. It can be linked to the executive program SUBSIZ and run automatically, or it can be run as a standalone program. When linked to SUBSIZ, all of the input parameters for the spacecraft CERs are input automatically. When run as a standalone program, these parameters must be entered manually in response to a prompt such as:

ENTER COMMUNICATIONS ANTENNA WEIGHT (LB): 2035.5

The required input parameters are summarized in Table 110.

TABLE 110. - LCCOST CER INPUT PARAMETERS

CER	Parameter
Structure (equipment bay)	weight, lb
Thermal control	weight, lb
Electrical power	power, W
Communication antenna	weight, lb
Communication signal processor	weight, lb
Communication transmitter	weight, lb
Attitude determination	weight, lb
Attitude reaction	weight, lb
Primary propulsion	impulse, lb-s
TT&C	weight, lb
Number of transmitters	unitless
Total S/C	weight, lb
Total S/C	length, ft
Ground operations	life, yr

An input of zero for any subsystem parameter will result in skipping cost computation for that subsystem. After parameter input is complete, the program will prompt for the CER source by displaying the available options for the subsystem. For example, the prompt for a communications antenna will be:

```
COST SOURCE FOR COMMUNICATIONS ANTENNA

          0 : THROUGH PUT
          1 : USER DEFINED CER
          2 : SPACE DIVISION CER
          3 : VOA CER
              SOURCE SELECTION -- ?

COST SOURCE FOR COMMUNICATIONS UPLINK

          0 : THROUGH PUT
          1 : USER DEFINED CER
          2 : SPACE DIVISION CER
              SOURCE SELECTION -- ?

COST SOURCE FOR SIGNAL PROCESSOR

          0 : THROUGH PUT
          1 : USER DEFINED CER
          2 : SPACE DIVISION CER
              SOURCE SELECTION -- ?

COST SOURCE FOR COMMUNICATIONS TRANSMITTER

          0 : THROUGH PUT
          1 : USER DEFINED CER
          2 : SOLID STATE CER
              SOURCE SELECTION -- ?
```

When a CER is chosen, the parameter value input previously for that element is used to calculate the cost. This process repeats for each subsystem in turn.

The throughput option allows direct input of recurring and nonrecurring cost without requiring an input parameter. The prompts for the throughput option have the form:

How Many Recurring Cost Elements for COMMUNICATIONS UPLINK? 1

Recurring Element Cost (1 of 1) ? 3.1

The user-defined CER option allows the user to define recurring and nonrecurring CER having the form:

A + B x (parameter)C

The prompts for the user-defined CER mode have the form:

ENTER A : ?
ENTER B : ?
ENTER C : ?

The user-defined CER then uses the previously input value of the CER parameter to calculate costs. The CER is not, however, retained after the module has terminated execution.

Exceptions to the above cost source input format are shuttle launch cost, program level, and fee. Program level and fee are input as decimal percentages. Program level is a percentage of the spacecraft total platform cost, and fee is a percentage of the program subtotal cost. Cost options for shuttle launch are either throughput or the shuttle launch algorithm described in Section 4. The algorithm requires information such as upper-stage weight, length, and cost (in FY 1984 $M).

Prompts for output options appear after calculations are complete. If the model is run as a standalone program a prompt will request an output title. This title is printed on paper printouts only. The next option is for display of the results on the screen or on the printer. After output is complete the user is given the option of printing again. A 'Y' response will return the program to the beginning of the output menu. A 'N' response results in program termination. The following three pages show an example of program LCCOST output.

NONRECURRING DESIGN & DEVELOPMENT COST

SUBSYSTEM	COST ($M) in 1984 $	COST SOURCE
STRUCTURE (EQUIPMENT BAY)	8.46	SPACE DIVISION CER
THERMAL CONTROL	9.22	SPACE DIVISION CER
ELECTRICAL POWER	42.11	VOA CER
COMMUNICATIONS ANTENNA	48.47	SPACE DIVISION CER
COMMUNICATIONS UPLINK	18.62	SPACE DIVISION CER
SIGNAL PROCESSOR	11.13	SPACE DIVISION CER
COMMUNICATIONS TRANSMITTER	7.79	SOLID STATE TRANSMITTER CER
ATTITUDE DETERMINATION	25.59	SPACE DIVISION CER
ATTITUDE REACTION	33.43	ELECT. PROPULSION CER
PROPULSION (AKM)	0.00	THROUGH PUT
TT&C	5.83	SPACE DIVISION CER
AEROSPACE GROUND EQUIPMENT	7.99	SPACE DIVISION CER
PROGRAM LEVEL	113.75	
FEE	33.24	
TOTAL NONRECURRING	365.64	

----- RECURRING PRODUCTION COST - - -

SUBSYSTEM	COST ($M) in 1984 $	COST SOURCE
STRUCTURE (EQUIPMENT BAY)	1.32	SPACE DIVISION CER
THERMAL CONTROL	1.46	SPACE DIVISION CER
ELECTRICAL POWER	28.51	VOA CER
COMMUNICATIONS ANTENNA	11.03	SPACE DIVISION CER
COMMUNICATIONS UPLINK	15.21	SPACE DIVISION CER
SIGNAL PROCESSOR	7.29	SPACE DIVISION CER
COMMUNICATIONS TRANSMITTER	3.58	SOLID STATE TRANSMITTER CER
ATTITUDE DETERMINATION	7.34	SPACE DIVISION CER
ATTITUDE REACTION	5.68	ELECT. PROPULSION CER
PROPULSION (AKM)	0.00	THROUGH PUT
TT&C	3.00	SPACE DIVISION CER
PROGRAM LEVEL	45.59	
FEE	13.00	
TOTAL RECURRING	143.01	

VOA TOTAL LIFE CYCLE COST

COST CATEGORY	COST($M) in 1984 $
TOTAL NONRECURRING	365.64
TOTAL RECURRING	143.01
STS LAUNCH	251.60
UPPER STAGE	25.00
GROUND OPERATIONS	12.00
TOTAL COST	797.25

RF PERFORMANCE PROGRAM (RFANAL)

Following is a description of the RF performance program developed for use in the VOA satellite study (contract NAS3-24233). The program, RFANAL, was designed to prompt the user for all necessary data required to run the program. The following are the program prompts and explanatory comments about each.

1) ENTER ORBITAL ALTITUDE IN km?

2) ENTER OPERATING FREQUENCY IN MHz?

   The program was designed to run in three separate bands of operating frequencies.

   a) 15 - 68 MHz,
   b) 1100 - 1300 MHz,
   c) 11000 - 13000 MHz.

   The user has the option of choosing any frequency within one of these three bands.

3) IS FEED SYSTEM CIRCULARLY OR LINEARLY POLARIZED (C/L)?

   The user should input a "C" for a circular polarized feed or "L" for a linear polarized feed. By inputting a "C" the program will add the appropriate 3 dB loss to the RF analysis.

4) ENTER SPACECRAFT LAT., LONG. POINT--SOUTHERN LAT. AND WESTERN LONG. ARE NEGATIVE?

   Enter the starting position of the satellite with respect to the coverage zone of interest. Southern latitudes and Western longitudes are defined as negative numbers. Therefore, latitudes are limited to $\pm$ 90° and longitudes are limited to $\pm$ 180°.

5) ENTER INCREMENT FOR LAT. LOCATION OF SPACECRAFT--ENTER 0 TO KEEP SATELLITE STATIONARY?

   This is used to allow the user to move the satellite along a constant longitudinal line at the increment specified. For each incremental step, the program will print out the RF analysis results. This will terminate when either the minimum elevation angle of 20° or the spacecraft's latitude limit (orbit inclination) is reached. This process will not be performed if the user inputs 0 for the increment.

6) ENTER INCREMENT FOR LONG. LOCATION OF SPACECRAFT--ENTER 0 TO KEEP SATELLITE STATIONARY?

This is similar to the above input, however the satellite will move along a constant latitude line. The program will again print out the RF analysis results for each incremental step. This will terminate when the minimum elevation angle of 20° is reached. This process will not be performed if the user inputs 0 for the increment.

7) ENTER SPACECRAFT LAT. LIMIT--I.E., THE INCLINATION OF THE ORBIT?

   This is used above to determine when the spacecraft has reached the latitude limit. This prompt will not appear if the user has entered 0 for prompt number 5.

8) ENTER NAME OF ZONE (LIMIT 80 CHARACTERS)?

   The user can input any combination of letters and numbers that will be used to title the output.

9) ENTER LOWER LAT. AND WESTERN LONG. OF SPOT--SOUTHERN LAT. AND WESTERN LONG. ARE NEGATIVE?

   Enter the lower most and left most boundaries of the zone. As with the spacecraft lat., long. point (prompt number 4) Southern latitudes and Western longitudes are defined as negative numbers.

10) ENTER UPPER LAT. AND EASTERN LONG. OF SPOT--SOUTHERN LAT. AND WESTERN LONG. ARE NEGATIVE?

    Enter the upper most and right most boundaries of the zone. As with the spacecraft lat., long. point (prompt number 4) Southern latitudes and Western longitudes are defined as negative numbers.

11) ENTER REFLECTOR DIAMETER LIMIT IN METERS?

    This is used to let the program know what the maximum size of the reflector can be. If the zone is too small requiring an antenna reflector larger than the user's defined limit, the program will flag the user and determine the actual zone covered by using the reflector diameter limit.

12) ENTER TOTAL TRANSMIT POWER IN WATTS?

This is the end of the program prompts. After each question has been answered the program will calculate and print out the RF analysis results. The following page shows an example of the program outputs.

```
*** USER INPUTS ****
ZONE COVERED -- EXAMPLE ZONE
TOTAL TRANSMIT POWER = 10000 WATTS   40 dBW
ORBITAL ALTITUDE = 10355 Km
OPERATING FREQUENCY = 26 MHz
SATELLITE LOCATION = 30  LAT 30  LONG
ZONE SIZE =-20  X  50  LATITUDE
           -20  X  50  LONGITUDE
POLARIZATION -- CIRCULAR
*** PROGRAM OUTPUTS ***
APERTURE SIZE = 22.9884 m X 22.9884 m using UNIFORM Illumination
ANTENNA EOC GAIN = 11.66537 dBi
ANTENNA APERTURE EFF = .7473356
ANTENNA BEAMWIDTH = 32.625 deg X 32.625 deg
GROUND COVERAGE = 7792.19 Km X 7792.19 Km
                  70 deg X 70 deg
ANTENNA SCAN ANGLE = 4.83
EOC EIRP = 51.66537 dBW
POLARIZATION MARGIN = 3 dB
PROPAGATION LOSS = 2 dB
MIN ELEVATION ANGLE = 18.85641
SPREADING LOSS = 153.6278 dB
SLANT RANGE = 13545.17 Km
EOC POWER FLUX DENSITY IN dBW/sq. m.
NO. OF CHANNELS         1       2       3       4       5       6       7
                  -106.97 -109.97 -111.74 -112.99 -113.96 -114.75 -115.42
EOC SIGNAL STRENGTH IN uV/m.
NO. OF CHANNELS         1       2       3       4       5       6       7
                   87.100  61.588  48.579  43.550  38.952  35.559  32.921
*************************************************************************
*** USER INPUTS ****
ZONE COVERED -- EXAMPLE ZONE
TOTAL TRANSMIT POWER = 10000 WATTS   40 dBW
ORBITAL ALTITUDE = 10355 Km
OPERATING FREQUENCY = 26 MHz
SATELLITE LOCATION = 30  LAT 30  LONG
ZONE SIZE =-20  X  50  LATITUDE
           -20  X  50  LONGITUDE
POLARIZATION -- CIRCULAR
*** PROGRAM OUTPUTS ***
APERTURE SIZE = 22.9884 m X 22.9884 m using UNIFORM Illumination
ANTENNA EOC GAIN = 11.66537 dBi
ANTENNA APERTURE EFF = .7473356
ANTENNA BEAMWIDTH = 32.625 deg X 32.625 deg
GROUND COVERAGE = 7792.19 Km X 7792.19 Km
                  70 deg X 70 deg
ANTENNA SCAN ANGLE = 4.83
EOC EIRP = 51.66537 dBW
POLARIZATION MARGIN = 3 dB
PROPAGATION LOSS = 2 dB
MIN ELEVATION ANGLE = 18.85641
SPREADING LOSS = 153.6278 dB
SLANT RANGE = 13545.17 Km
EOC POWER FLUX DENSITY IN dBW/sq. m.
NO. OF CHANNELS         1       2       3       4       5       6       7
                  -106.97 -109.97 -111.74 -112.99 -113.96 -114.75 -115.42
EOC SIGNAL STRENGTH IN uV/m.
NO. OF CHANNELS         1       2       3       4       5       6       7
                   87.100  61.588  48.579  43.550  38.952  35.559  32.921
*************************************************************************
*************************************************************************
```

PROGRAM ASSUMPTIONS

The program was designed to provide the necessary data for the VOA study. Therefore, the program automatically assumes certain RF parameters. First, for frequency bands 15-68 MHz and 1100-1300 MHz, the program calculates the antenna beamwidth using the following equation:

$$BW = 65 \times \lambda / D \qquad (2)$$

where BW = antenna beamwidth, deg
65 = 65° uniform illumination factor
λ = wavelength, m
D = reflector diameter, m.

For the frequency band of 11000-13000 MHz, the program calculates the antenna beamwidth by using:

$$BW = 70 \times \lambda / D \qquad (3)$$

where BW = antenna beamwidth, deg
70 = 70° tapered illumination factor
λ = wavelength, m
D = reflector diameter, m.

The difference in the two equations results because the first two frequency bands assume a phased array reflector system with uniform illumination, while the third frequency band uses a reflector dish and feed system with tapered illumination.

The next RF parameter built into the program is the aperture efficiency. For the first band, 15-68 MHz, the program uses:

$$Eff = 0.75 \times \cos(\Theta) \qquad (4)$$

where Eff = aperture efficiency
Θ = antenna scan angle off boresight

This equation assumes that the phased array antenna used for the first band design is an electronically steerable array. For the second band, the aperture efficiency is a constant 0.8, and for the third band, the aperture efficiency is a constant 0.5. These numbers assume a nonsteerable phased array and a dish reflector and feed, respectively.

The final parameter automatically calculated by the program is the propagation loss due to transmission from space. The program assumes a propagation loss of 2 dB if operating at less than 35 MHz, 1 dB if operating between 35 MHz and 68 MHz, and 0 dB for all other frequencies.

Program Listing

The following is the program listing for the rf performance program (RFANAL).

```
10 DIM PHI(4)
15 T$=STRING$(73,42)
20 CLS:INPUT "ENTER ORBITAL ALTITUDE IN KM";OT
30 INPUT "ENTER OPERATING FREQUENCY IN MHz";OF
40 INPUT "IS FEED SYSTEM CIRCULARLY OR LINEARLY POLARIZED (C/L)";A$
50 IF A$="C" OR A$="c" THEN PM=3:GOTO 80
60 IF A$="L" OR A$="l" THEN PM=0:GOTO 80
70 GOTO 40
80 INPUT "ENTER SPACECRAFT LAT,LONG POINT -- SOUTHERN LAT. AND WESTERN
                                        LONG. ARE NEGATIVE";LATSC,LONGSC
90 INPUT "ENTER INCREMENT FOR LAT LOCATION OF SPACECRAFT - ENTER 0 TO
         KEEP SATELLITE STATIONARY";SCLATINC
100 INPUT "ENTER INCREMENT FOR LONG LOCATION OF SPACECRAFT - ENTER 0 TO
          KEEP SATELLITE STATIONARY";SCLONGINC
110 LATSCI=LATSC:LOOP=0:LONGSCI=LONGSC:LATLIMIT=90
120 IF SCLATINC<>0 THEN INPUT "ENTER SPACECRAFT LAT LIMIT --- I.E.,THE
                                 INCLINATION OF THE ORBIT";LATLIMIT
130 INPUT "ENTER NAME OF ZONE (LIMIT 80 CHARACTERS)";NAM$
140 INPUT "ENTER LOWER LAT AND WESTERN LONG OF SPOT -- SOUTHERN LAT. AND
                                                    WESTERN LONG. ARE
                                                 NEGATIVE";LAT1,LONG1
150 INPUT "ENTER UPPER LAT AND EASTERN LONG OF SPOT -- SOUTHERN LAT. AND
                                                    WESTERN LONG. ARE
                                                 NEGATIVE";LAT2,LONG2
160 IF LAT1=LATSC THEN LAT1=LAT1-.0001
170 IF LAT2=LATSC THEN LAT2=LAT2-.0001
180 IF LONG1=LONGSC THEN LONG1=LONG1-.0001
190 IF LONG2=LONGSC THEN LONG2=LONG2-.0001
200 OLAT1=LAT1:OLAT2=LAT2:OLONG1=LONG1:OLONG2=LONG2
211 PHIL=90
220 INPUT "ENTER REFLECTOR DIAMETER LIMIT IN METERS";DIAL
230 INPUT "ENTER TOTAL TRANSMIT POWER IN WATTS";TTP
240 LAT1=OLAT1:LAT2=OLAT2:LONG1=OLONG1:LONG2=OLONG2:FL=0
250 SCAN=0:TAPER=0:ILLUM=65
260 IF OF>11000 AND OF<13000 THEN TAPER=1:ILLUM=70
270 WL=3E+08/(OF*1000000!)
280 AWLIM=ILLUM*WL/DIAL
290 TP=10*LOG(TTP)/LOG(10)
300 GC1=TAN((LAT1-LATSC)*3.1415927#/180)*(COS((LAT1-
        LATSC)*3.1415927#/180)*6378)
310 GC2=TAN((LAT2-LATSC)*3.1415927#/180)*(COS((LAT2-
        LATSC)*3.1415927#/180)*6378)
320 GC3=TAN((LONG1-LONGSC)*3.1415927#/180)*(COS((LONG1-
        LONGSC)*3.1415927#/180)*6378)
330 GC4=TAN((LONG2-LONGSC)*3.1415927#/180)*(COS((LONG2-
        LONGSC)*3.1415927#/180)*6378)
340 ALPHA1=ATN(GC1/(OT+(6378-COS((LAT1-LATSC)*3.1415927#/180)*6378)))
350 ALPHA2=ATN(GC2/(OT+(6378-COS((LAT2-LATSC)*3.1415927#/180)*6378)))
360 ALPHA3=ATN(GC3/(OT+(6378-COS((LONG1-LONGSC)*3.1415927#/180)*6378)))
```

```
370 ALPHA4=ATN(GC4/(OT+(6378-COS((LONG2-LONGSC)*3.1415927#/180)*6378)))
380 IF OF>15 AND OF<68 THEN
SCAN1=ABS(ALPHA1+ALPHA2)/2:SCAN2=ABS(ALPHA3+ALPHA4)/2:IF SCAN2>SCAN1
THEN SCAN=SCAN2:EFF=COS(SCAN2)*.75 ELSE SCAN=SCAN1:EFF=COS(SCAN1)*.75
390 IF OF>1100 AND OF<3000 THEN EFF=.8
400 IF OF>11000 AND OF<13000 THEN EFF=.5
420 ABW1=ALPHA2-ALPHA1:ABW1=ABW1*180/3.1415927#
430 ABW2=ALPHA4-ALPHA3:ABW2=ABW2*180/3.1415927#
440 IF ABW1<AWLIM THEN FL=1:GOSUB 1210
450 IF ABW2<AWLIM THEN FL=1:GOSUB 1350
460 A1=ILLUM*WL/ABW1:A2=ILLUM*WL/ABW2
470 ATG=(10*LOG(EFF*9.869605*A1*A2/(WL*WL))/LOG(10))-3
480 GCD1=LAT2-LAT1
490 GCD2=LONG2-LONG1
500 EIRP=TP+ATG
510 IF OF>=35 AND OF<68 THEN PL=1 ELSE PL=0
520 IF OF<35 THEN PL=2
540 PHIT=ABS(LAT1-LATSC):PHI(1)=90-PHIT-ABS(ALPHA1*180/3.1415927#)
550 PHIT=ABS(LAT2-LATSC):PHI(2)=90-PHIT-ABS(ALPHA2*180/3.1415927#)
560 PHIT=ABS(LONG1-LONGSC):PHI(3)=90-PHIT-ABS(ALPHA3*180/3.1415927#)
570 PHIT=ABS(LONG2-LONGSC):PHI(4)=90-PHIT-ABS(ALPHA4*180/3.1415927#)
580 FOR I=1 TO 4
590 IF PHI(I)<PHIL THEN PHIL=PHI(I)
600 NEXT I
610 IF PHIL<0 THEN PRINT "**** ALL OF ZONE CAN NOT BE SEEN FROM
                              SATELLITE ****":GOTO 1160
620 IF PHIL<20 THEN PRINT "**** CAUTION: ELEVATION ANGLE LESS THAN 20
                               DEGREES ****"
630 PHID=PHIL*3.1415927#/180
640 SR=SQR((6378*SIN(PHID))^2+2*6378*OT+OT*OT)-6378*SIN(PHID)
650 SL=-10*LOG(1/(12.566371#*SR*SR*1000000!))/LOG(10)
660 PFD=(TP+ATG)-(SL+PL+PM)
670 SS=(SQR(10^(PFD/10))*19.4165)/.000001
671 SS=INT(SS*100)/100
672 PFD=INT(PFD*100)/100
680 IF FL=1 THEN PRINT "*******************************"
690 IF FL=1 THEN PRINT "*  ANALYSIS IS LIMITED BY     *"
700 IF FL=1 THEN PRINT "*  USER INPUT MAXIMUM REFL.   *"
710 IF FL=1 THEN PRINT "*          DIAMETER           *"
720 IF FL=1 THEN PRINT "*******************************"
750 PRINT "APERTURE SIZE =";A1;"m X";A2;"m"
760 LPRINT "*** USER INPUTS ****
770 IF FL=1 THEN LPRINT "*******************************"
780 IF FL=1 THEN LPRINT "*  ANALYSIS IS LIMITED BY     *"
790 IF FL=1 THEN LPRINT "*  USER INPUT MAXIMUM REFL.   *"
800 IF FL=1 THEN LPRINT "*          DIAMETER           *"
810 IF FL=1 THEN LPRINT "*******************************"
820 LPRINT "ZONE COVERED -- ";NAM$
830 LPRINT "TOTAL TRANSMIT POWER =";TTP;"WATTS  ";TP;"dBW"
840 LPRINT "ORBITAL ALTITUDE =";OT;"Km"
850 LPRINT "OPERATING FREQUENCY =";OF;"MHz"
860 LPRINT "SATELLITE LOCATION =";LATSC;" LAT";LONGSC;" LONG"
870 LAT1=INT(LAT1*100)/100:LAT2=INT(LAT2*100)/100
   :LONG1=INT(LONG1*100)/100:LONG2=INT(LONG2*100)/100
880 LPRINT "ZONE SIZE =";LAT1;" X ";LAT2;" LATITUDE
```

```
890 LPRINT "                ";LONG1;" X ";LONG2;" LONGITUDE
900 IF PM=3 THEN LPRINT "POLARIZATION -- CIRCULAR" ELSE LPRINT
                         "POLARIZATION -- LINEAR"
910 LPRINT "*** PROGRAM OUTPUTS ***"
920 LPRINT "APERTURE SIZE =";A1;"m X";A2;"m";
930 IF TAPER=0 THEN LPRINT " using UNIFORM Illumination"
940 IF TAPER=1 THEN LPRINT " using TAPERED Illumination"
950 LPRINT "ANTENNA EOC GAIN =";ATG;"dBi"
960 LPRINT "ANTENNA APERTURE EFF =";EFF
970 ABW1=INT(ABW1*1000)/1000:ABW2=INT(ABW2*1000)/1000
980 LPRINT "ANTENNA BEAMWIDTH =";ABW1;"deg X";ABW2;"deg"
990 LPRINT "GROUND COVERAGE =";
    INT(100*ABS(LAT1-LAT2)*3.1415927#*12756/360)/100;"Km X";
    INT(100*ABS(LONG1-LONG2)*3.1415927#*12756/360)/100;"Km"
1000 LPRINT TAB(17) GCD1;"deg X";GCD2;"deg"
1010 LPRINT "ANTENNA SCAN ANGLE =";INT(100*SCAN*180/3.1415927#)/100
1020 LPRINT "EOC EIRP =";TP+ATG;"dBW"
1030 LPRINT "POLARIZATION MARGIN =";PM;"dB"
1040 LPRINT "PROPAGATION LOSS =";PL;"dB"
1050 LPRINT "MIN ELEVATION ANGLE =";PHIL
1060 LPRINT "SPREADING LOSS =";SL;"dB"
1070 LPRINT "SLANT RANGE =";SR;"Km"
1080 LPRINT "EOC POWER FLUX DENSITY IN dBW/sq. m."
1081 LPRINT "NO. OF CHANNELS       1       2       3       4       5
            6       7"
1082 LPRINT TAB(19);:LPRINT USING "####.##";PFD;
1083 LPRINT TAB(27);:LPRINT USING "####.##" ;PFD-3;
1084 LPRINT TAB(35);:LPRINT USING "####.##" ;PFD-4.771001;
1085 LPRINT TAB(43);:LPRINT USING "####.##" ;PFD-6.021001;
1086 LPRINT TAB(51);:LPRINT USING "####.##";PFD-6.9897;
1087 LPRINT TAB(59);:LPRINT USING "####.##";PFD-7.78151;
1088 LPRINT TAB(67);:LPRINT USING "####.##";PFD-8.451
1090 LPRINT "EOC SIGNAL STRENGTH IN uV/m."
1091 LPRINT "NO. OF CHANNELS       1       2       3       4       5
            6       7"
1092 LPRINT TAB(19);:LPRINT USING "###.###";SS;
1093 LPRINT TAB(27);:LPRINT USING "###.###" ;SS*.7071;
1094 LPRINT TAB(35);:LPRINT USING "###.###" ;SS*.557735;
1095 LPRINT TAB(43);:LPRINT USING "###.###" ;SS*.5;
1096 LPRINT TAB(51);:LPRINT USING "###.###";SS*.44721;
1097 LPRINT TAB(59);:LPRINT USING "###.###";SS*.40825;
1098 LPRINT TAB(67);:LPRINT USING "###.###";SS*.377964
1100 LPRINT T$
1110 IF LOOP=2 THEN GOTO 1140
1120 IF PHIL<20 OR ABS(LATSC)>=ABS(LATLIMIT) THEN
     LOOP=2:LATSC=LATSCI:LONGSC=LONGSCI+SCLONGINC:PHIL=90:GOTO 240
1130 IF SCLATINC<> 0 THEN LATSC=LATSC+SCLATINC:PHIL=90:GOTO 240
1132 IF SCLONGINC=0 THEN GOTO 1151
1133 LOOP=2:LATSC=LATSCI:LONGSC=LONGSCI+SCLONGINC:PHIL=90:GOTO 240
1140 IF PHIL<20 THEN GOTO 1151
1150 IF SCLONGINC<> 0 THEN LONGSC=LONGSC+SCLONGINC:PHIL=90:GOTO 240 ELSE
     GOTO 1151
1151 LPRINT T$:LPRINT CHR$(12)
1200 GOTO 20
1210 DELTA=AWLIM-ABW1
```

```
1220 DELTA=DELTA*3.1415927#/180
1230 IF ABS(LAT1-LATSC)>ABS(LAT2-LATSC) THEN ALPHA2=ALPHA2+DELTA
1240 IF ABS(LAT1-LATSC)<ABS(LAT2-LATSC) THEN ALPHA1=ALPHA1-DELTA
1250 IF ABS(LAT1-LATSC)=ABS(LAT2-LATSC) THEN
     ALPHA1=ALPHA1-DELTA/2:ALPHA2=ALPHA2+DELTA/2
1260 K1=(OT*TAN(ALPHA1)+6378*TAN(ALPHA1))/6378
1270 K2=(OT*TAN(ALPHA2)+6378*TAN(ALPHA2))/6378
1280 C1=TAN(ALPHA1):C2=TAN(ALPHA2)
1290 D1P=(2*K1*C1+SQR(4*K1*K1*C1*C1-4*(1+C1*C1)*
       (K1*K1-1)))/(2*(1+C1*C1))
1300 D2P=(2*K2*C2+SQR(4*K2*K2*C2*C2-4*(1+C2*C2)*
       (K2*K2-1)))/(2*(1+C2*C2))
1310 LAT1P=(1.570796-ATN(D1P/SQR(1-D1P*D1P)))*180/3.1415927#*
            ALPHA1/ABS(ALPHA1)+LATSC
1320 LAT2P=(1.570796-ATN(D2P/SQR(1-D2P*D2P)))*180/3.1415927#*
            ALPHA2/ABS(ALPHA2)+LATSC
1330 LAT1=LAT1P:LAT2=LAT2P
1340 ABW1=AWLIM:RETURN
1350 DELTA=AWLIM-ABW2
1360 DELTA=DELTA*3.1415927#/180
1370 IF ABS(LONG1-LONGSC)>ABS(LONG2-LONGSC) THEN ALPHA4=ALPHA4+DELTA
1380 IF ABS(LONG1-LONGSC)<ABS(LONG2-LONGSC) THEN ALPHA3=ALPHA3-DELTA
1390 IF ABS(LONG1-LONGSC)=ABS(LONG2-LONGSC) THEN ALPHA3=ALPHA3-DELTA/2:
      ALPHA4=ALPHA4+DELTA/2
1400 REM PRINT ALPHA3,ALPHA4
1410 K1=(OT*TAN(ALPHA3)+6378*TAN(ALPHA3))/6378
1420 K2=(OT*TAN(ALPHA4)+6378*TAN(ALPHA4))/6378
1430 C1=TAN(ALPHA3):C2=TAN(ALPHA4)
1440 D1P=(2*K1*C1+SQR(4*K1*K1*C1*C1-4*(1+C1*C1)*(K1*K1-1)))/
        (2*(1+C1*C1))
1450 D2P=(2*K2*C2+SQR(4*K2*K2*C2*C2-4*(1+C2*C2)*(K2*K2-1)))/
        (2*(1+C2*C2))
1460 LONG1P=(1.570796-ATN(D1P/SQR(1-D1P*D1P)))*180/3.1415927#*
             ALPHA3/ABS(ALPHA3)+LONGSC
1470 LONG2P=(1.570796-ATN(D2P/SQR(1-D2P*D2P)))*180/3.1415927#*
             ALPHA4/ABS(ALPHA4)+LONGSC
1480 LONG1=LONG1P:LONG2=LONG2P
1490 ABW2=AWLIM:RETURN
```

WEIGHT AND VOLUME ESTIMATING PROGRAM (SUBSIZ,EPSSIZ,RFSIZ)

The algorithms contained in the weight and volume estimating program were described in Section 4.0. The software provided uses these algorithms. Following is a summary of assumptions used to perform sizing for all subsystems, including those that were originally sized using other programs (e.g. SCIAP, SDCM).

1) Antenna sizing (RFSIZ):
   a) transmitter efficiency = 0.65 for frequencies > 3 GHz, where solid state power amplifiers (SSPA) are applicable;
   b) transmitter efficiency = 0.36 for frequencies $\leq$ 3 GHz, where traveling-wave tube antennas (TWTA) are applicable.

2) Feeder link electronics (RFSIZ):
   a) 121 kg for nongeostationary Earth orbits (GEO), where crosslinks are assumed;
   b) 78 kg for GEOs, where crosslinks are not required.

3) Signal processing electronics (RFSIZ):
   a) 58 kg for all cases.

4) Auxiliary propulsion (SUBSIZ):
   a) dry weight = 184 kg for non-GEOs (electric propulsion);
   b) dry weight = 91 kg for GEOs (chemical propulsion);
   c) fuel weight = 120 kg for both cases.

5) Attitude determination (SUBSIZ):
   a) weight = 57.7 kg for non-GEO cases;
   b) weight = 13.6 kg for GEO cases.

6) TT&C (SUBSIZ):
   a) weight = 32.1 kg for non-GEO cases;
   b) weight = 26.4 kg for GEO cases.

7) Thermal control (SUBSIZ)
   a) maximum temperature = 55°C
   b) minimum temperature = -12°C
   c) end-of-life absorptivity = 0.2
   d) end-of-life emissivity = 0.8

The EPS (program EPSSIZ) and equipment bay structure (SUBSIZ) sizes are computed from user inputs and outputs from the other subsystem sizings. The equipment bay structure sizing must not be performed until sizing is completed for all other subsystems.

Subsystem Sizing Example and User Instructions

The module, SUBSIZ, serves as the executive control, chaining to EPSSIZ, RFSIZ, or LCCOST as necessary in a mode completely transparent to the user. Subsystem sizing is started inserting the program diskette into the default drive and by typing:

```
BASICA SUBSIZ.VOA
```

or, if already in BASICA:

```
LOAD "SUBSIZ.VOA"
RUN
```

The following example shows the user inputs and resulting outputs for sizing an HF concept in an 8-hour circular orbit. First the main menu is displayed, listing the subsystems that can be sized.

```
INPUT TITLE FOR THIS ANALYSIS
? HF 8 hour orbit example
Input 2nd line if needed?
Input 3rd line if needed?
SUBSYSTEM SIZING MAIN MENU

1 - antenna/rf subsytem
2 - auxiliary propulsion subsystem
3 - primary propulsion subsystem
4 - TT&C
5 - electrical power subsystem
6 - thermal control subsystem
7 - equipment bay sizing
8 - All
9 - Stop execution

Your choice? 1
```

The choice made, 2, results in display of the menu from the antenna and RF sizing module. The choice made here, 12, is for the HF and VHF baseline design concept described in the report. Now the RF frequency, phased array element spacing, modulation mode (prompted only for HF cases), transmitter characteristics, and orbit class are defined by the user as shown.

[illegible]

```
RF SUBSYSTEM SIZING SUMMARY

Select antenna structural type

1 - Wrap Radial Rib (Lockhood's)
2   Hoop and Column (Grumman's)
3 - Hoop and Column (Harris')
4 - Box Truss Ring (Martin's)
5 - Inflatable (L'Garde's)
6 - Box Truss Reflector (Martin's)
7 - Cable Catenary Reflector (TRW's)
8   Tetrahedral Truss Reflector (GD's)
9 - Umbrella Radial Rib (Harris')
10 - Articulated Umbrella Radial Rib (Harris')
11 - Honeycomb Panel Solid Reflector (Ford Aero.)
12 - Box Truss Ring Array (Martin's)
13 - Honeycomb Panel Array (Non-phased)
Your choice? 12
Enter antenna diameter (m)? 80


Data for Box Truss Ring Array (limit on diameter of 200 meters)

For a reflector antenna, enter a 0 for transmit frequency.
For a phased array, enter the transmission frequency in Ghz.

FREQUENCY (OR 0)? .026


Enter the spacing between phased array elements
as a decimal fraction of mission wavelength
Fraction? .75
Calculated transmitter spacing - 8.653847  is this correct (Y/N)? y


Select modulation mode; DSB(D) or SSB(S) ? s
Input number of transmitter power levels? 1


TRANSMITTER TYPE  1
Enter power per transmitter(w)? 250
Input % of transmitters at this power? 1
Input number of channels per transmitter? 1
```

The element spacing input, 0.75 in the example, is used by the program to compute the number of transmitters as well as the number of radiating elements, assuming one transmitter for each element. However, for many designs it may be desireable to have multiple transmitters feeding a single element. In this case, the spacing must be defined to result in the desired number of transmitters. The required spacing to be input is computed from:

$$\text{spacing} = \sqrt{\frac{\pi R^2}{N-1}} \Big/ \lambda \qquad (5)$$

where R = antenna diameter (m)
N = number of transmitters required
λ = wavelength $f_{(m)}$

For example, for 177 transmitters on a 40-meter antenna at 26 MHz, the required spacing (as a fraction of wavelength) would be:

$$\text{spacing} = \sqrt{\frac{\pi(40)^2}{176}} \Big/ 11.538 = \frac{5.344}{11.538} = 0.463 \qquad (6)$$

The user prompts and inputs would be:

```
Enter the spacing between phased array elements
as a decimal fraction of mission wavelength.
Fraction? 0.463

Computed spacing from program is xxxx.xx.  Is
this correct (Y/N)? Y
```

The last prompt, "DO YOU WANT TO SIZE ANOTHER ANTENNA SUBSYSTEM(Y/N)?" transfers back to the executive after a "NO" input.

Main menu choices 2 or 4 result in the prompt to select the orbit category, either non-GEO or GEO. Depending on the choice, the appropriate weight and volume values are selected by the program.

```
SPECIFY ORBIT TYPE: 1 - GEO
                    2 - NON-GEO

YOUR CHOICE? 2
```

Main menu choice 5 results in the prompts and interaction necessary to select the EPS configuration and to then compute weights and volumes of EPS components. In the example, the required power is fed in automatically from the RF sizing section. If the EPS were sized first, a prompt would appear to input the required peak load power. The prompt for housekeeping power is for the average power required during occultation. The next inputs define the orbital altitude, lifetime, power generation component, and battery type. The last prompt permits another EPS sizing or transfers control back to the executive as for the antenna and RF sizing.

```
Input S/C housekeeping electrical power required for subsystems excluding Rf sub
system(Kw)? .5
Input orbit altitude (km)? 13892
Input Spacecraft lifetime requirement in YRS
? 7

SELECT POWER GENERATION TYPE

 1 Silicon blanket
 2-Silicon panel
 3-Existing GaAs panel
 4-Near term GaAs panel
 5-Future(long term) GaAs panel
 6-SP100

YOUR CHOICE? 1

SELECT BATTERY TYPE

 1-NiCd
 2-NiH

YOUR CHOICE? 1
```

Main menu choice 6 results in thermal control subsystem sizing. In the example, the difference between the power in and out of the load (transmitters) is shown and the user prompted to use this value or to input another. The user then selects an orbit category as shown, and further defines the thermal control subsystem configuration.

```
DIFFERENCE BETWEEN POWER TO LOAD AND POWER OUT IS  9.155254  Kw.

DO YOU WANT TO USE THIS VALUE TO SIZE THE THERMAL CONTROL SYSTEM(Y/N)? y
Input maximum operating temperature (C)? 55
Input minimum temperature(C)? -12

SELECT S/C OPERATIONAL ALTITUDE CLASS

 1 - GEO or greater
 2 - 926 Km(500 Nm) to GEO
 3 - less than 926 Km (LEO)

YOUR CHOICE? 2

Is phase change material included in the Thermal Control Subsystem(Y/N)
? n
```

Main menu choice 7 does not result in any further user interaction since equipment bay structure sizing is based on the weight of components sized for other subsystems.

Selection of main menu choice 9 results in output of the system weight and volume summary and display of a prompt to estimate life-cycle cost. If LCC estimates are desired, program LCCOST will automatically be executed.

The following three pages contain the sizing program outputs for the example inputs shown here.

ORIGINAL PAGE IS
OF POOR QUALITY

RF SUBSYSTEM SIZING SUMMARY
Data for Box Truss Ring Array (limit on diameter of 200 meters)
Antenna diameter (M) 80

Total DSR Output Rf Carrier Power(kw) 17
Total Output Rf Signal Power(kw) 17

Power per transmitter(w)	Number of transmitters	Total Rf power(kw)
250	68	17

ANTENNA APERTURE AREA(M^2)	5026.549
TOTAL ANTENNA STRUCTURE MASS(KG)	1242.438
TOTAL ANTENNA STRUCTURE VOLUME(M^3)	40.92173
TOTAL RF TRANSMITTER MASS(KG)	879.7908
TOTAL RF TRANSMITTER VOLUME(M^3)	1.236458
WEIGHT OF UPLINK COMPONENTS(Kg)	121
VOLUME OF UPLINK COMPONENTS(M^3)	.5
WEIGHT OF SIGNAL PROC. COMPONENTS(Kg)	58
VOLUME OF SIGNAL PROC. COMPONENTS(M^3)	.02
TOTAL RF SUBSYTEM MASS(KG)	2321.229
TOTAL RF SUBSYSTEM VOLUME(M^3)	42.67819

ATTITUDE CONTROL, STATIONKEEPING AND MANEUVERING SUMMARY

Total ACS subsystem mass(Kg)	57.7	Lbs	127.2285
Total ACS subsystem volume(M^3)	.05		
Total RCS subsystem mass(Kg)	184	Lbs	405.72
Total RCS subsystem volume(M^3)	2.2		
Total RCS/ACS subsystem mass(Kg)	241.7		
Total RCS/ACS subsystem volume(M^3)	2.25		

TT&C SUBSYSTEM SUMMARY

Total TT&C subsystem mass (kg)	32.1	Lbs	70.7805
Total TT&C subsystem volume(M^3)	.03		

HE 8 hour orbit sample case

EPS SUMMARY

Power required from source = 28.06161 KW
Power at load, average = 26.65385 KW
Orbital altitude = 13892 Km
Orbital period = 7.977999 hrs
Spacecraft lifetime requirement - 7 yrs
Total eclipse time per orbit= .812874 hrs
Solar array degradation factor due to radiation = .7847762
Solar array thermal adjustment factor = .8799453
Solar array cover slide weight factor - 0
Antenna Size = 80 m by 80 m

***** Power Generation Sizing *****

For Si Blanket Array
Area required = 258.4542 m2
Weight = 413.5267 Kg Volume = 2.067634 m3

***** Shunt Regulator Sizing *****

Shunt Regulator Weight = 49.00938 Kg
Shunt Regulator Volume = 5.556619E-02 m^3

***** Power Switching/Distribution Sizing *****

Power Switching Equip Wt. = 8.549986 Kg
Power Switching Equip Vol. = 9.693861E-03 m3

Distribution Weight; Source to Bus = 10.93997 Kg
Distribution Weight; Bus to Load = 54.43952 Kg
Total Distribution Weight = 65.37949 Kg
Total Switching and Distribution Weight = 73.92948
****** TOTAL SI SYSTEM WEIGHT W/O BATTERIES(kg) - 536.4656
****** TOTAL SI SYSTEM VOLUME W/O BATTERIES(M^3) = 2.132894

***** Battery Sizing *****

For NiCd Batteries
Battery Capacity Required = 1063.564 Wh
Battery Weight = 30.15152 Kg Battery Volume = 1.505837E-02 m3

***** Battery Charger Sizing *****

For NiCd Batteries
Battery Charger Weight = .5386365 Kg Battery Charger Volume - 6.10699E-04 m3

TOTAL EPS MASS(kg) = 567.1557
TOTAL EPS VOLUME(M^3) = 2.148563

HF 8 hour orbit sample case

THERMAL CONTROL SUBSYSTEM SIZING SUMMARY

Maximum temperature(C)	55		
Maximum heat radiated(W)	9155.255		
Radiative surface emissivity factor	.8		
Radiative surface absorptivity factor	.2		
Required radiator surface area(M^2)	27.69341		
Thermal Control Subsystem mass(Kg)	135.2178	Lbs	298.1553
Thermal Control Subsystem area (M^2)	27.69341		
Thermal Control Subsystem Volume(M^3)	1.384671		

HF 8 hour orbit sample case

EQUIPMENT BAY SUMMARY

Total mass of equipment bay(kg)	43.42271	Lbs	95.74708
Mass of deployment/stowage mechanisms(kg)	71.4		
Total mass(kg)	114.8227	(Lbs)	253.1841
Total volume of structure and mechanisms(m^3)	.7711403		

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HF 8 hour orbit sample case

SYSTEM CONFIGURATION SUMMARY

	MASS(KG)	VOLUME(M^3)
Rf Payload	2321.229	42.67819
Auxiliary Propulsion Subsytem	241.7	2.25
Telemetry, Tracking and Command	32.1	.03
Electrical Power Subsystem	567.1557	2.148563
Thermal Control Subsystem	135.2178	1.384671
Equipment Bay Structure	114.8227	.5140935
TOTAL SYSTEM SUMMARY	3412.225	49.00552
	7523.956	(Lbs)

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