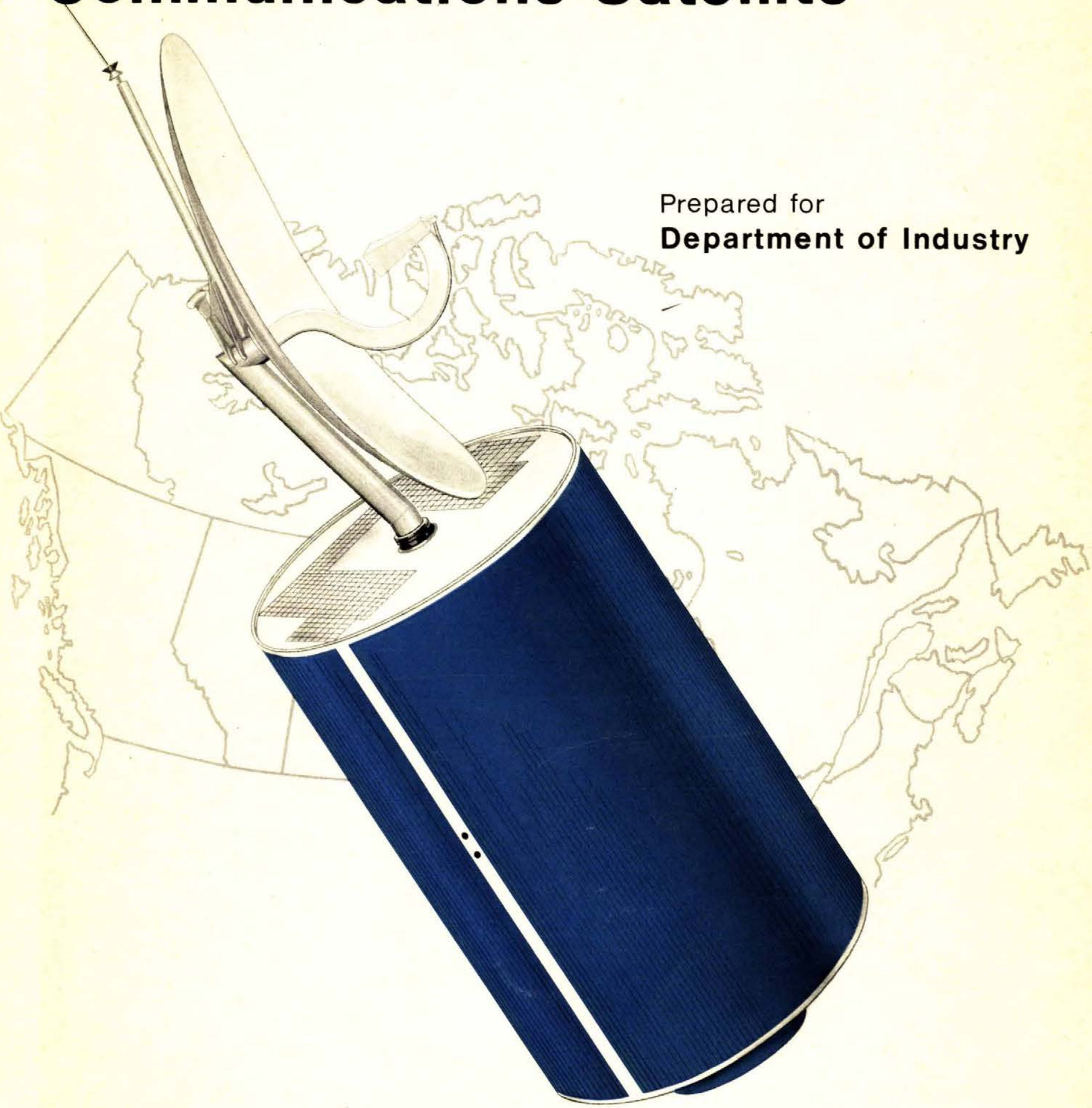
Study Program for
Canadian Domestic Communications Satellite



Prepared for
Department of Industry

Final Report
Volume One - Design Considerations

RCA Space Systems

Copy No. 46

STUDY PROGRAM

for the

DESIGN, DEVELOPMENT AND SUPPLY

of a

DOMESTIC SATELLITE COMMUNICATIONS SYSTEM

FINAL REPORT
VOLUME 1, DESIGN CONSIDERATIONS

Prepared for

DEPARTMENT OF INDUSTRY

by

RCA LIMITED, Space Systems
1001 Lenoir Street, Montreal

PROPOSITION No. 2220-I

NOVEMBER 25th, 1968

PREFACE

This report is submitted by RCA Limited to the Department of Industry in compliance with Section 4.2 of the Statement of Work forming part of D.O.I. Contract, File No. IRA. 9122-03-4.

The report is in six volumes, namely:

Volume 1	Design Considerations
Volume 2(a)	Spacecraft Design - Electrical
Volume 2(b)	Spacecraft Design - Mechanical
Volume 3	Technical Appendices
Volume 4	Program Plan
Volume 5	Program Costs

The information contained in the report is supplied to Her Majesty for use solely in connection with the design, development, manufacture, operation, repair, maintenance and testing of a Canadian Domestic Satellite Communication System.

SYSTEM ANALYSIS

DESIGN CONSIDERATIONS

Although the basic satellite parameters for the Canadian domestic satellite have been defined in the Department's work statement, RCA Limited has undertaken a number of system studies related to the overall performance and requirements which, in general, support the Department's choice of satellite parameters. However, some of these studies indicate that for optimum system performance, certain satellite parameters should be modified.

SUMMARY

The following conclusions have been drawn on the basis of this study.

a) The specified orbit inclination of $\pm 0.2^{\circ}$ should be reduced to $\pm 0.1^{\circ}$. This improved inclination control is easily attained and will result in lower degradation due to pointing errors for the small fixed antenna stations. This in turn will permit a smaller antenna size to achieve the required communications performance.

b) It appears feasible to use a 32 foot diameter "fixed" antenna with an uncooled paramp. of 100°K noise temperature to provide high quality television performance (55 dB video signal to weighted RMS noise) including a separate sound carrier. The G/T which can be achieved with this arrangement is about 29 dB which results in a downlink carrier-to-noise ratio of about 19 dB. This performance assumes satellite ERP of 34 dB W at the satellite antenna beam edge.

c) If large antennas are used to provide a G/T ratio of 40.7 dB, it is feasible to carry up to 1500 voice channels. A station with a G/T ratio of 36.7 dB can carry up to 1200 channels.

d) The cost differential between a fully tracking station and non-tracking station is very large.

1 SYSTEMS ANALYSIS

In establishing a synchronous satellite system, careful consideration must be given to total performance objectives of the system, i.e. traffic requirements, channel capacity, operational objectives, etc. Trade-off studies must be made to determine the most practical and economical system.

The following pages contain discussions on the various factors which influence the choice of overall system parameters. The discussions are centred largely on the satellite and the small receive-only stations since overall system performance is established mainly by these two elements. The main communication and control stations will have large diameter antennas, large G/T ratios and will be fully steerable. There will therefore, be no system performance limitations due to these stations.

1.1 Factors Affecting Performance

a) Satellite Drift

Drift Mechanisms: A satellite in orbit is subject to perturbations which cause the inclination, eccentricity and subpoint longitude of an initially perfect synchronous orbit to change. The dominant forces are radiation pressure and variations in the gravitational field. The variations in the gravitational field are caused by the asphericity of the earth and the gravitational fields of the sun and moon. The effect of radiation pressure is to cause a circular orbit to become eccentric and then circular again each year. Triaxiality of the earth causes the satellite to oscillate about the minor axis of the earth's equator. Points of stable equilibrium occur at 74° and 106° longitude, and the initial drift rate depends on the distance from the stable points, the maximum occurring halfway between the stable and unstable points.

The major effect of the sun and moon is to increase the orbit inclination. The sun causes an increase of 0.27 degrees/year, and the increase due to the moon varies with the inclination of the moon's orbit to the equatorial plane. The effect of the moon on orbit inclination varies from 0.48 degree/year to 0.68 degree/year, with an 18.6 year period. The effect of sun and moon together, in 1972, will be about 0.91 degree/year and will decrease to 0.75 degree/year in 1978, as discussed in Section 2 on Mission Analysis.

Effect on System Performance: As viewed from the surface of the earth, the motion of an inclined synchronous orbit is a "figure" eight", traced each sideral day. For an inclination of 0.1 degrees, the width of the "figure eight" is vanishingly small, and the height, as seen from the earth, depends on the latitude and relative longitude of the earth station and satellite subpoint. Eccentricity of the orbit also causes a diurnal variation, in this case, it is a longitudinal drift of $\pm$ 2e, where e is the eccentricity of the orbit. Longitudinal drift caused by triaxiality has no diurnal variation, and if uncorrected, would librate about 106°W longitude with a period (of the order of years) dependent on the initial position.

Assuming that the satellite can be kept on station within 0.1 degree in longitude and inclination drift, the largest angular excursion will be 0.156° from nominal position for most stations in Canada, decreasing somewhat for northern stations. For a tracking antenna satellite drift will not lead to a reduction in system gain, however, for a non-tracking antenna the gain reduction depends on the beam width. Figure 1-1 shows the reduction of antenna gain versus satellite inclination drift for a fixed longitudinal drift of $\pm$ 0.10°(see appendix 1). It can be seen that a drift of 0.10°in inclination can cause a gain reduction of 0.9dB. Figure 1-2 shows the attainable G/T (gain-to-noise temperature ratio) for conditions of maximum drift. Combinations of maximum longitude and inclination drift occur for a small fraction of the time and the degradation due to drift is treated as a fade in section 1.1(d).

b) Small Earth Station Pointing Errors

It is assumed that all earth stations except the large diameter main communication and control stations will have non-steerable antennas. These antennas will, however, be capable of manual positioning to provide coverage of the visible synchronous belt. These earth stations will thus be subject to degradations in receive C/T (carrier-to-noise temperature ratio) due to antenna pointing errors. The error due to earth station initial adjustment errors and environmental disturbances. The initial antenna positioning error is largely a matter of the time and effort expended; with a reasonable amount of effort it should be possible to position the antenna so that this initial set-up error is negligible.

Environmental conditions which affect pointing accuracy include:

i wind induced errors,
ii errors due to reflector surface deformations caused by ice and snow loads,
iii variations in refractive bending.

These errors are discussed in appendix 2 and are found to be small. Therefore, none of these contributions are expected to affect significantly the receive C/T with the possible exceptions of ice and snow loads. With antennas of about 30 ft. diameter wind induced errors can be kept very small without resorting to costly antenna structures. The refractive bending variations are maximum at the lowest elevation angles (5°) but even here they will be small compared to the antenna beamwidth. Degradation due to ice and snow loads may be significant in some parts of the country, particularly northern Canada, and design of earth stations for these regions must take cognizance of this environment. Special deicing and snow removal techniques may be required for

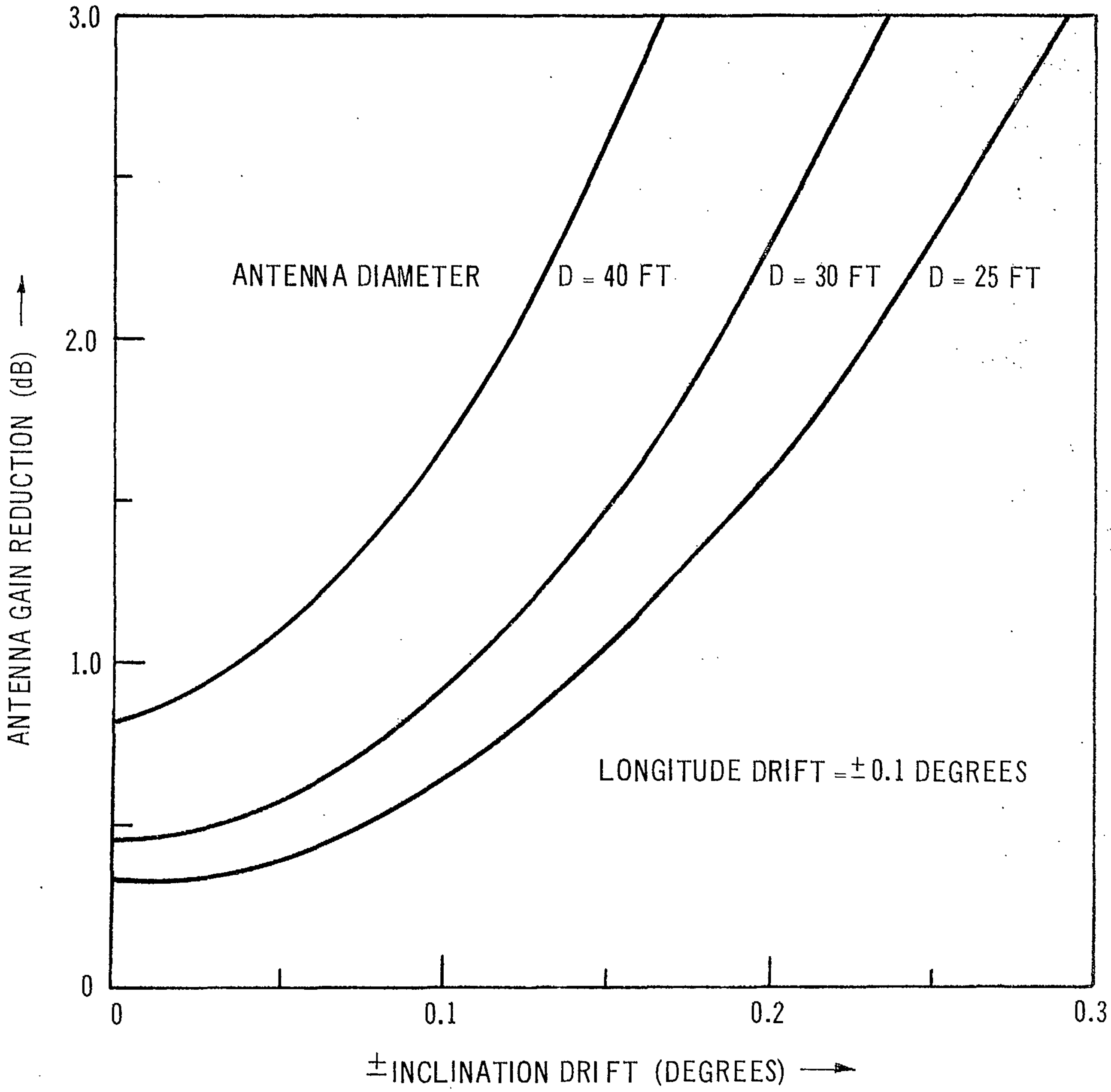


Figure 1–1 The Effect of Satellite Inclination Drift on Loss in Earth Station Antenna Gain

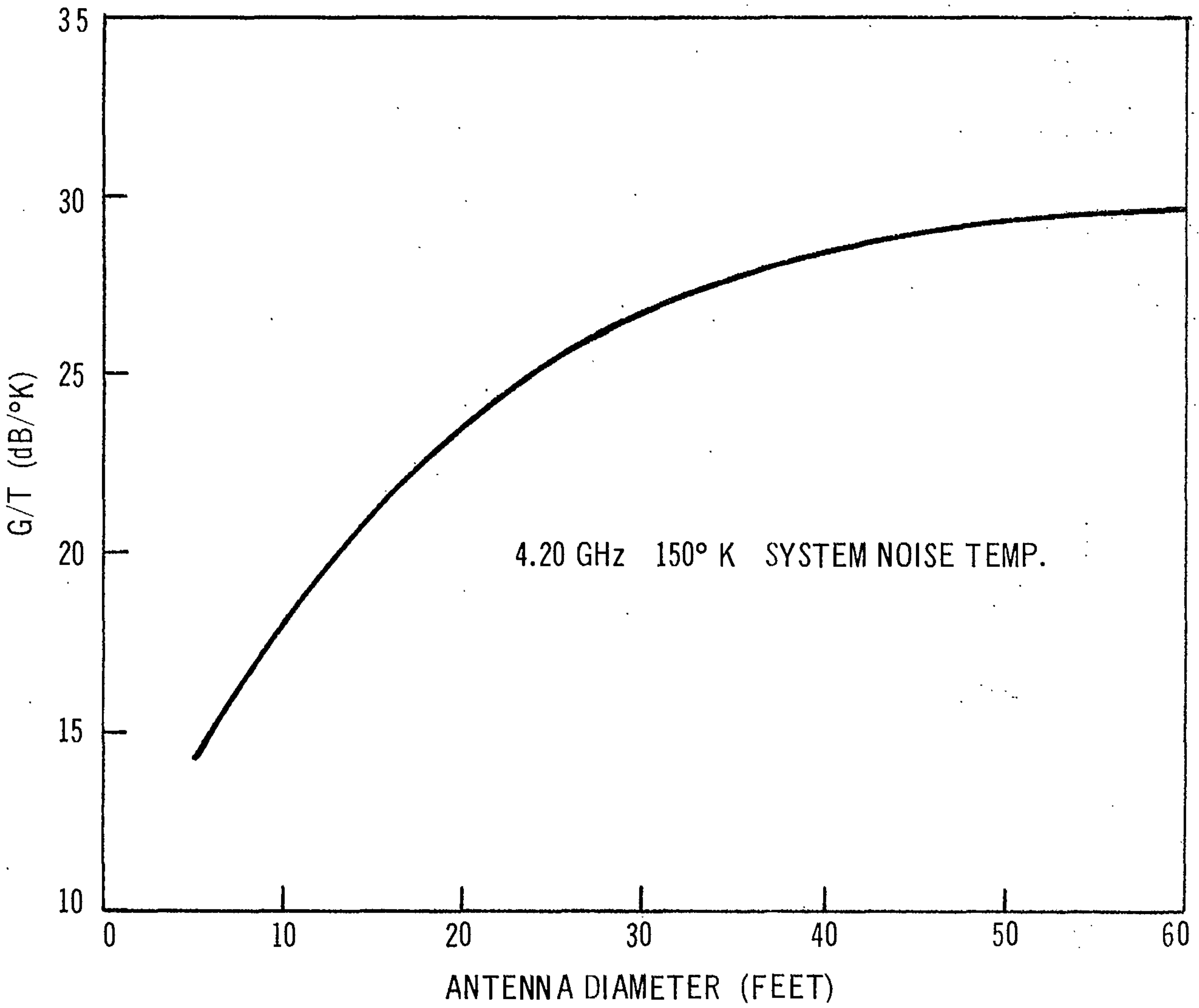


Figure 1–2 Minimum Effective G/T for Non-Tracking Antenna versus Size. Satellite station keeping ±0.1 degree in drift and inclination.

these stations. In summary, "small" earth station pointing errors will generally be small and will not significantly affect station performance.

c) Path Loss

Calculation of earth station performance at any particular site must take into account the path loss between the satellite and the earth station. Two components contribute to total path loss: free space attenuation, which varies with slant range to the satellite, and atmospheric attenuation, which varies with elevation angle. Both slant range and elevation angle are functions of earth station latitude and of the relative longitude of the earth station and satellite sub-point. Figure 1-3 shows the total path loss at 4 GHz a function of longitudes and latitudes in Canada and the derivation is shown in appendix 3.

Canada, with much territory at high latitudes, will experience considerable variation in path loss with site location. The peak difference will be about 0.7 dB so this factor must be taken into account when plotting the effective beam coverage of Canada. The combination of the projected antenna beam contours and path loss will result in contours of equal flux density. This is described in more detail in Section 1.7.

d) Fading Distribution

Calculations of station performance on a statistical basis must include C/T degradations due to all environmental conditions. The major contributor to fade, for both large and small stations, is rainfall. Obviously, this will vary widely across the country. Rainfall data shows that maximum annual rainfall occurs along the coast of British Columbia while maximum short term rainfall intensity occurs in southern Ontario. Central Alberta and the Atlantic coast also have relatively high short term rainfalls.

Prediction of station fade due to rainfall can only be estimated since the data available on total weather patterns for a given location is limited. Fade calculation required a knowledge of rainfall intensity distribution as well as the mean width and height of a rainstorm. In general, high intensity rainfalls occur over relatively small areas while low rainfalls or drizzle can be tens of miles in extent.

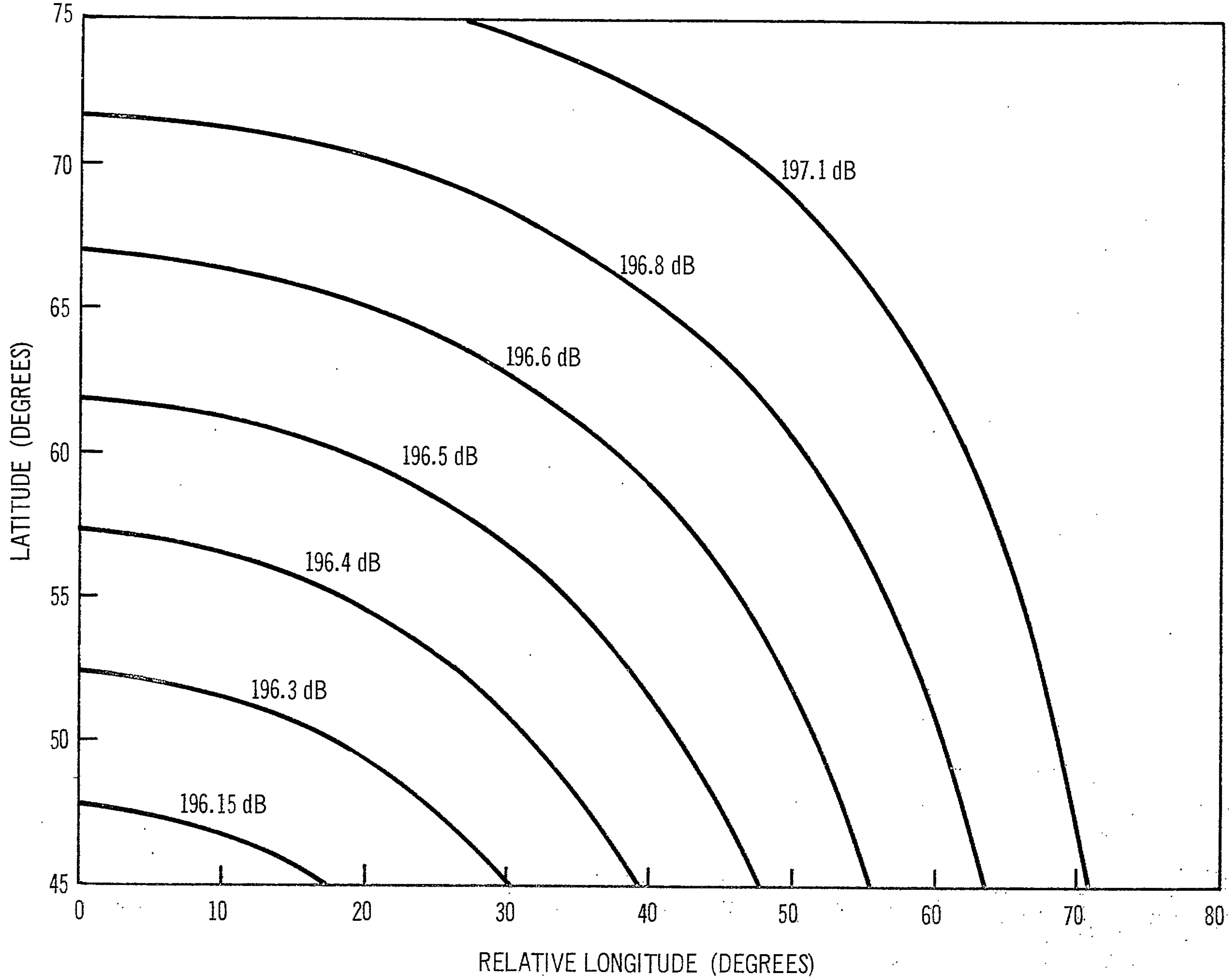


Figure 1-3 Total Path Loss in dB versus Site Location 4.0 Hz

The presence of a rainfield in front of an earth station antenna degrades system performance in two ways: the rainfield acts as an attenuator and directly reduces incoming signal strength, and this attenuation further deteriorates the system performance by increasing the overall receive noise temperature.

Figure 1-4 shows a fade distribution curve for a typical small earth station located in southern Canada. (See appendix 4) Assumptions made for the derivation of this curve include:

i Total annual rainfall approximately 50 inches.

ii Earth station noise temperature in clear weather - 150°K. (See section 1.1f).

iii Overall receive C/T is degraded 1 db due to up-link and satellite noise plus a further 1 db due to adjacent satellite and terrestrial interference which provides an effective overall system noise temperature of 235°K.

It can be seen that station fade due to rainfall is acceptable and will not impose any limitations on the design of the small earth stations. Fade due to wind or refractivity variations will be second-order compared to rainfall and is ignored.

In addition to the fade due to weather, the receive C/T at any particular location will experience fluctuations caused by inaccuracies in satellite position keeping and antenna pointing. Fading due to satellite antenna pointing error is discussed in appendix 5. The worst earth station location from an overall fade point of view will be for points at the nominal beam edge in the eastern and western regions. Overall receive system fade for such a station is shown in Figure 1-5, the reference signal level being based on a satellite EIRP of 34 dB W at the nominal beam edge which is taken as 0 dB fade. This curve has been derived by a convolution of the individual fading curves due to rainfall, satellite pointing inaccuracy and satellite position keeping accuracy. The overall fade distribution assumes the following:

i Satellite EIRP at the 8 1/2° x 3 1/4° beam edge is 34 dB W

ii Satellite antenna pointing accuracy is $\pm$ 0.5° each axis; 3-sigma gaussian distribution.

iii Earth station is located at a position 4.35° $\pm$ 0.5° (3-sigma) east or west of the nominal beam centre.

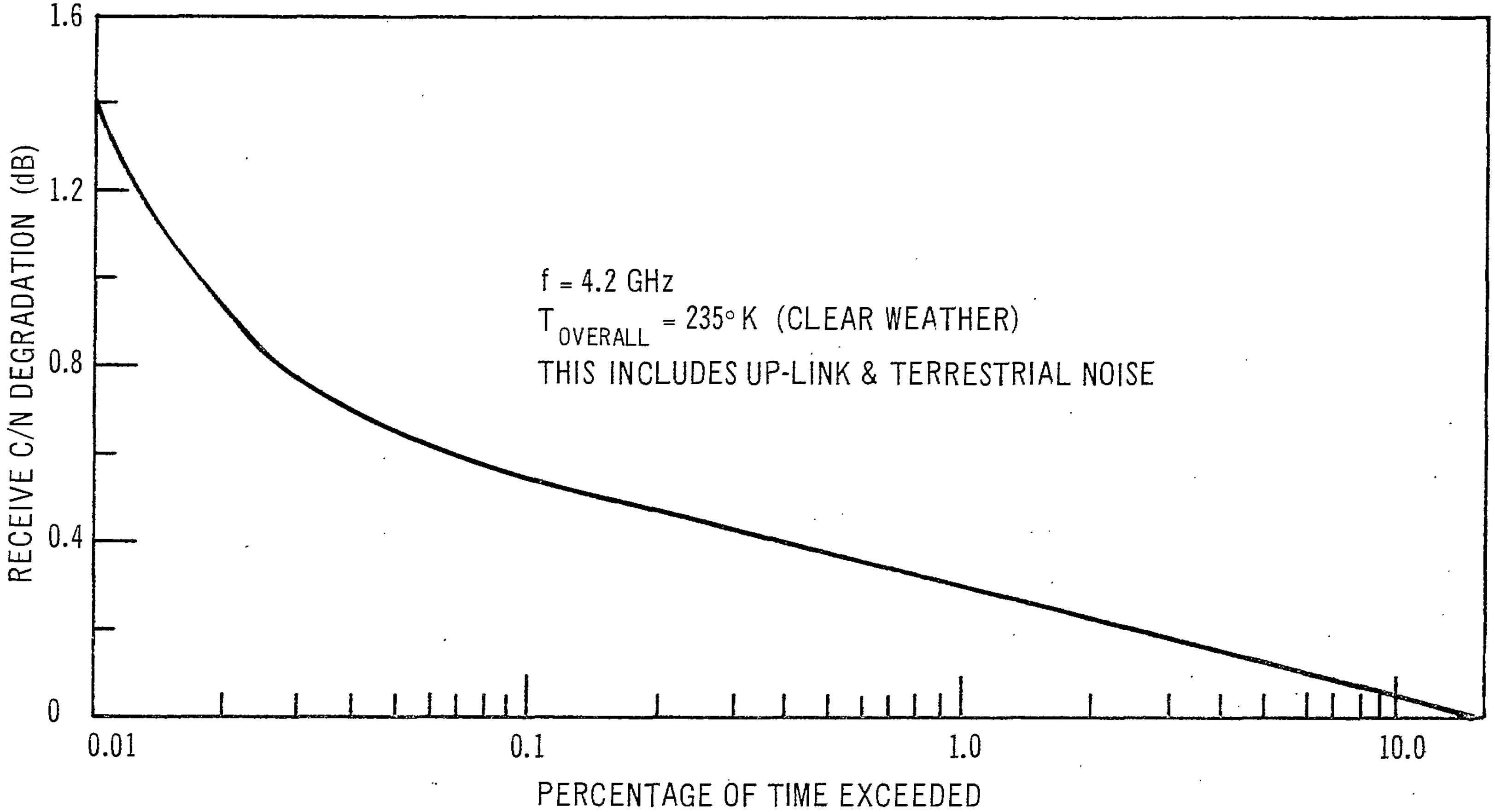


Figure 1–4 Typical Earth Station Fade for Southern Canada

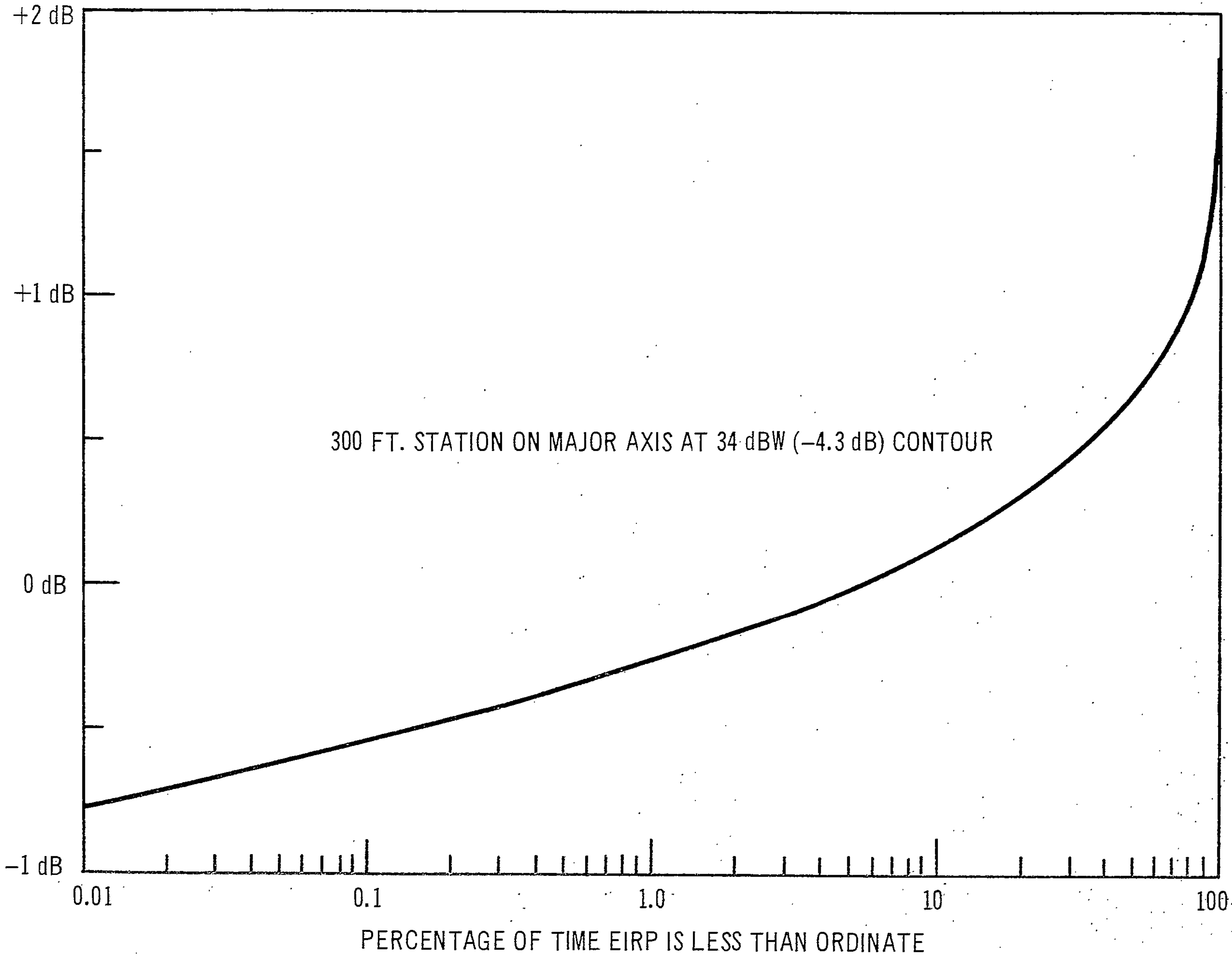


Figure 1–4 Small Station Fade or Variation in Overall C/T Ratio

iv Earth station antenna pointing error degradation (30 ft. diameter non-tracking antenna) assumes all satellite positions $\pm 0.1^{\circ}$ longitude and $\pm 0.1^{\circ}$ inclination, are equally probable.

v Rainfall fade distribution curve in Figure 1-4 applies.

The composite fade curve does not include the effects of initial earth station antenna positioning error, wind induced pointing errors, refractivity errors and errors due to snow or ice loads. As stated previously, however, these errors are expected to be of second order magnitude compared to the primary errors.

e) Noise Objectives and Budgets

In order to determine overall system parameters, the basic overall noise requirements must be defined. These requirements can be divided into three sections; telephony channel video and video program. The following requirements have been used as a basis for calculation.

Telephony	7500 pWp for the worst channel
Video	55 dB peak to peak for high quality (50 dB nominal quality)
Programme	57 dB psophometric at +9 dBmO

From these requirements the following noise budgets can be used:

Telephony:	Multiple Carrier Operation	Trunk Operation
Adjacent satellite interference	1000 pWp	1000 pWp
Terrestrial Link interference	1000 pWp	1000 pWp
Up-path Thermal	1000 pWp	1000 pWp
Intermodulation	1500 pWp	1000 pWp
Down-path Thermal	3000 pWp	3500 pWp
TOTAL	7500 pWp	7500 pWp

Video (High Quality):

Adjacent satellite interference effect in baseband	S/N = 65 dB
Terrestrial link interference effect in baseband	S/N = 65 dB
Satellite Intermodulation	S/N = 67 dB
Up-path thermal 6 db below down path thermal	S/N = 66 dB
Down path thermal noise	S/N = 57 dB
TOTAL	S/N = 55 dB

f) Earth Station Noise Temperature

Earth station system noise temperature objectives are dependent on the desired station G/T ratio. For the main communication and control stations with a G/T ratio of 36 to 40.7 dB it is necessary to use cooled parametric amplifiers to achieve the lowest possible noise temperature. Typically stations in this category can achieve system noise temperatures of about 70°K.

For the small TV receive-only stations with a G/T requirement of 27 - 30 dB uncooled paramps. can be used. Stations of this type would be expected to have system noise temperatures of about 150°K. These paramps. will of course be more reliable and simpler to maintain than the cooled paramps. These stations will also be less subject to rainfall fade since the increase in system noise temperature due to rainfall will be less significant than for a low noise station. (See appendix 6)

g) Sun Outage and Eclipse Operation

Solar Transits and Eclipses: Near the vernal and autumnal equinoxes the sun has a number of undesirable effects on both the space and earth segments of the system: the sun's apparent path causes it to pass behind the satellite as seen by an earth station, and to pass behind the earth as seen by the satellite. The transit of the sun through the earth station beam causes the receive system noise temperature to approach that of the sun. The passage of the sun behind the earth eclipses the sun from the satellite for which the present study envisages storage cells of sufficient capacity to enable three channels to operate during an eclipse. Fortunately eclipses occur during the hour near local midnight at the satellite subpoint and thus at a time when traffic would be relatively light for most of the country. The moon can also pass behind the satellite and cause an increase in receiver noise, however, the temperature of the moon is about 236°K and the increase in noise temperature should be within the station margin.

Total eclipse of the sun by the moon, as seen by the satellite will occur with about the same frequency as a total eclipse at a given spot on earth, but will last slightly longer. This is such a rare event that it can be considered negligible in comparison with eclipses of the sun by the earth.

The sun also causes system degradation by another mechanism. This occurs when the sun comes sufficiently close to the satellite's receive antenna beam to cause an increase in the up-link noise temperature budget. Assuming a satellite noise figure of 7 dB the worst case degradation gives a noise figure of 9 dB This occurs twice a year, just before the autumn eclipse and just after the spring eclipse. The degradation becomes progressively less on days preceding and succeeding the day of worst case occurrence. This type of degradation will have negligible effect on system performance.

Duration of Eclipses: Satellite eclipses by the earth are highly predictable, and tables have been computed to give in umbra, penumbra and time of occurrence for any arbitrary circular orbit[1]. In general, eclipses occur once per day for 43 days in the spring and 43 days in the autumn, with the longest eclipse (69 minutes) occurring on or about March 22 and September 22.

Duration of Solar Transits: Solar transits of the earth station antenna cause fading so deep, that it must be considered a communication outage. RF energy from the sun is not constant and during "disturbed" sun activity the RF level at 4 GHz can be 20 or 30 times greater than the "quiet" sun level. For the majority of the time however, "quiet" sun conditions prevail. At this time the 4 GHz flux level at the earth's surface will be approximately 2×10^{-20} watts/M^2-Hz. In this situation a 60 foot antenna with low sidelobe levels will experience a degradation greater than 12db for 6 minutes on the worst day, and 5 1/2 minutes and 2 minutes of the two days preceding and succeeding the

1 "Tables for Eclipse Times of an Earth Satellite of Arbitrary Circular Orbit", T. Szirtes, September, 1967, RCA Space Systems, Montreal

worst day. This will happen two months per year, and represents an outage on each of ten days per year. Figure 1-6 is a time distribution curve of the G/T degradation for 60 foot and 30 foot antennas with low sidelobe levels. It is unlikely that these outages would be acceptable to the satellite users. A number of methods for eliminating the outage or reducing it to an acceptable level are discussed below.

i Earth station antennas can be spaced approximately 500 to 1500 miles apart, depending on antenna size, so that the sun is not in the beam width of both antennas simultaneously. As this would require twice as many terminals, in addition to the microwave hops to connect them to the distribution centre, this solution has little to recommend it.

ii Two satellites can be orbited, and each station can have two antennas per receiver terminal. System costs will not be doubled by this approach since any system must have an orbiting backup satellite if long service disruptions due to satellite failure are to be avoided, and thus there are no extra costs in the space segment. For each receive station only an extra antenna and perhaps paramp are required. The remainder of the receive system can be common to both antennas. This, nevertheless represents an increase in system costs.

For low cost receive only stations, reduced outage time can be achieved by moving the earth station antenna to the second satellite when the sun comes into the region of the first satellite. A preliminary investigation reveals that acceleration rates of $2^{\circ}/sec^2$ and velocities of $2^{\circ}/sec$ can easily be obtained with the type of antennas envisaged with an unsophisticated positioning system. A very low cost servo is required as it is not a tracking system and can be pre-programmed. Mechanical brakes would hold the antenna at each position. For satellites 6 degrees apart this would cause a 5 second outage, and for satellites 12 degrees apart, this outage would be 7 seconds. If the station is initially using the westernmost satellite, the outages can be kept to one during the day and one at some convenient time at night.

1.2 Orbital Slot Utilization

Our views on using the stationary orbit efficiently have been published * and a copy is shown in appendix 7. The primary conclusion

* D. Jung, "Defining System Considerations for Canadian Domestic Satellites", Electronics and Communications, June 1968

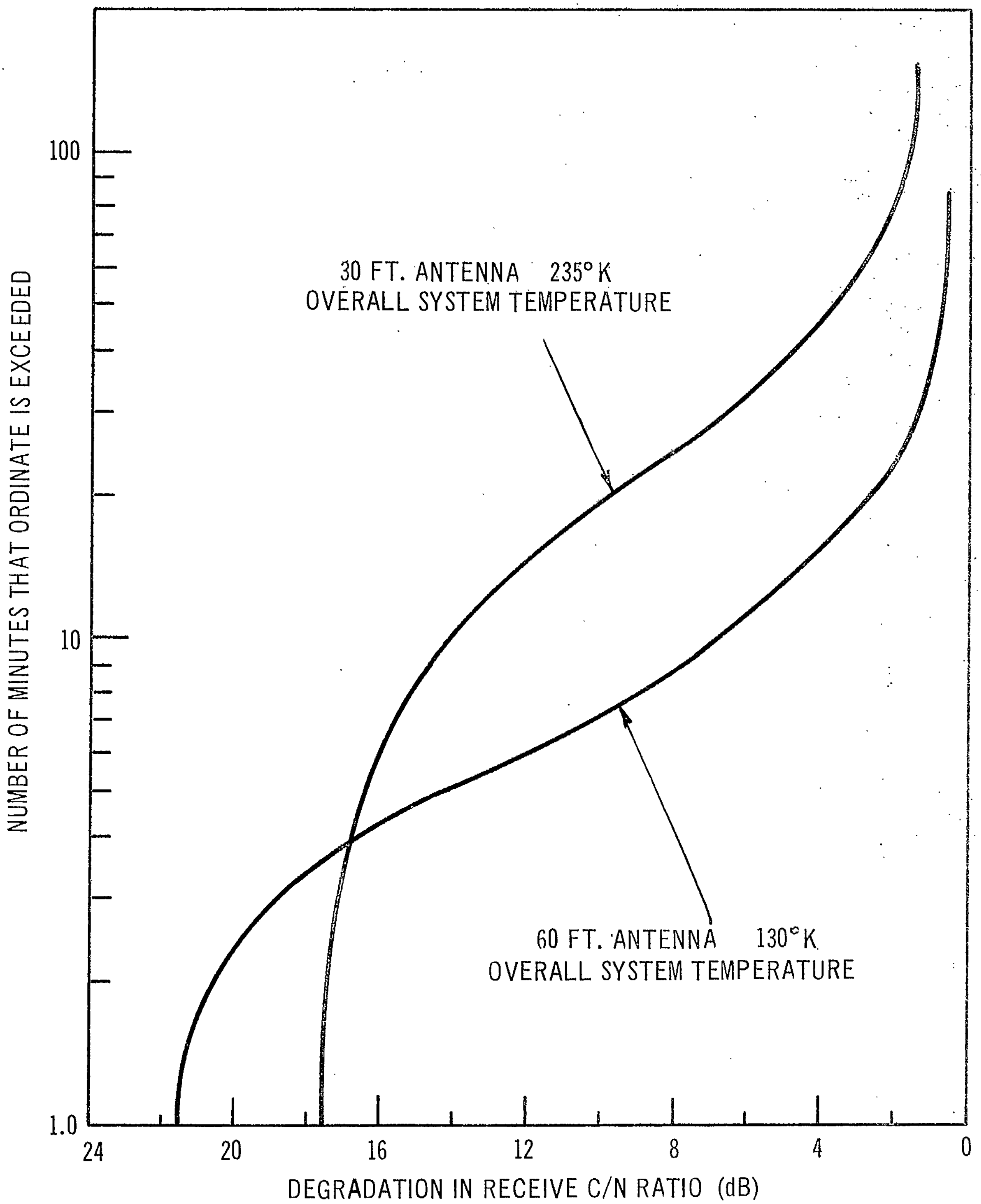


Figure 1-6 Deagradation Due To Solar Radiation – Worst Month

of the paper is that efficient use of the stationary orbit is of vital interest to Canada whose visible geo-stationary belt is only 35° wide in contrast to the United States proper (80°) and South America (105°). Since it is estimated that North America will run out of useful orbital space in the 4 and 6 GHz bands before 1980, there is a need to arrive at agreements regarding orbital slot allocations before the ITU World Administrative Radio Conference to be held about 1970-71. It is shown that satellites with identical RF channels can be spaced about 6° apart but if frequency interleaving is applied to adjacent satellites, the spacing can be reduced to about 3°. A proposed frequency plan is shown in Figure 1-7. No tolerances for satellite drift, antenna gain, ERP, etc., were allowed for determining the approximate spacing. However, for a practical system some allowance must be made and a factor of 7/6 appears satisfactory. This will provide a satellite spacing of 7° for the co-channel case and 3.5° for the interleaved case.

Interleaving can also be used for RF channels using multiple carriers for certain cases. For example, assume that 4 carriers, each carrying 60 channels are transmitted on RF Channel No. 1 of plan A, and assume 1500 channels are transmitted on RF Channel No. 1 of plan B. The worst case interference is caused by one of the 60 channel carriers interfering with the 1500 channel carrier at the worst frequency,but fortunately the satellite ERP of the 60 channel carrier is about 16 dB less than that of the 1500 channel carrier. Therefore, the interference in this case will be no more than that caused by 1500 channel carrier into an interleaved 1500 channel carrier. For other cases, such as transmitting two 600 channel carriers, interleaving does not seem to provide any significant advantage. This is due to the fact that we are only relying on the difference between transmitted powers since the receiver interference reduction factors (B-factors) are about the same between low channel capacity carriers and 1500 channel carriers for small frequency differences about 10 MHz. Some B-factors are shown in appendix 8. Interleaving multiple carriers requires some more study.

The spacing of satellites in the initial plan should be based on the co-channel case until more study and experience is available. This will also permit more flexibility for the system.

It should be noted that in the proposed frequency plan, it is found that adjacent RF channels should be crossed polarized in the up link in order to simplify the satellite design. For example, all the odd channels on plan A might be vertically polarized and the even number channels might be horizontally polarized.

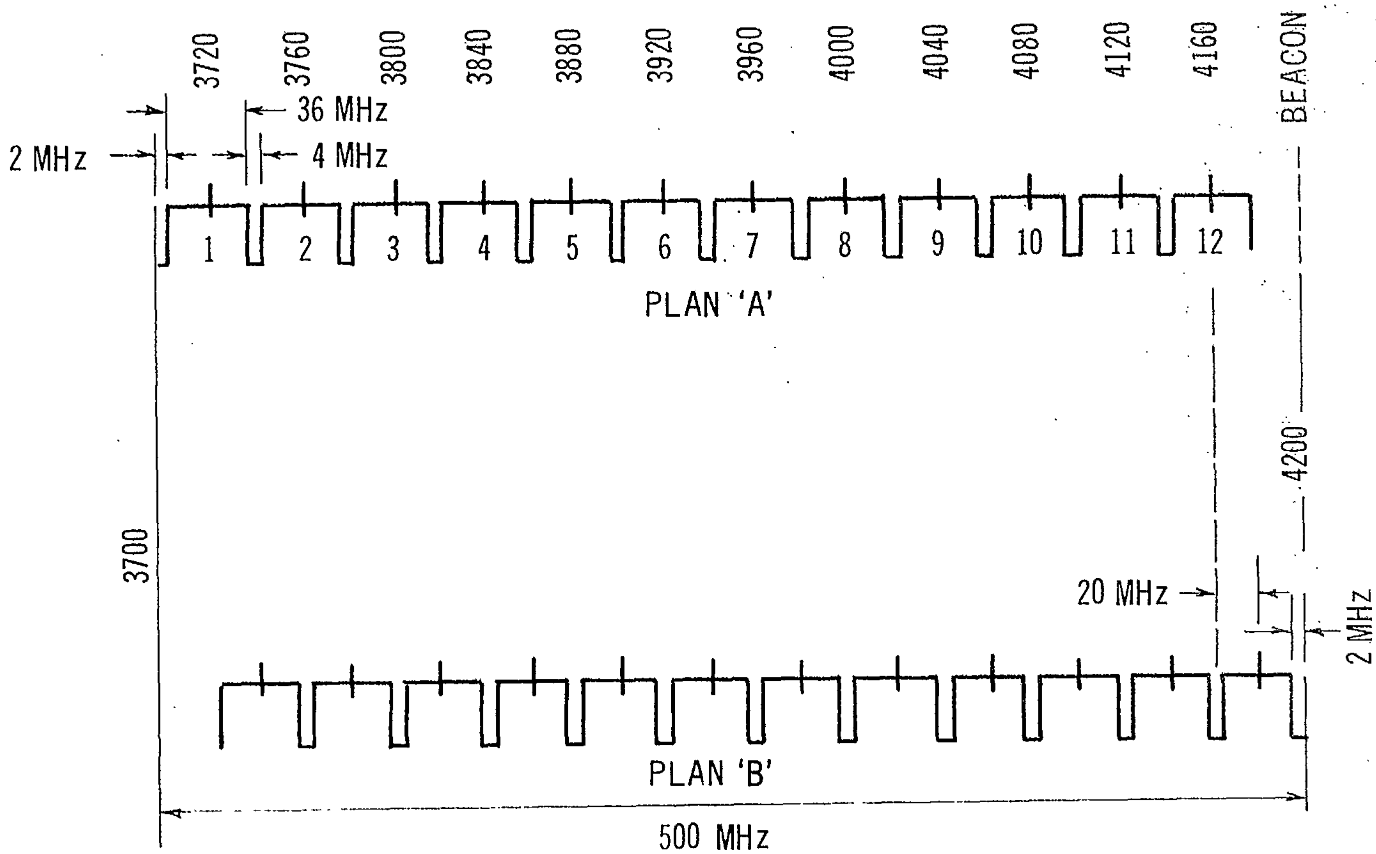


Figure 1-7 Proposed Frequency Plan for Adjacent Satellites

1.3 Overall Communications Performance

a) Television

The television performance can be calculated from the following formula:

$$S/N = 10 \log (3/2 \ C/kT \ \Delta f^2/f_m^{\ 3}) + 6 + 13$$

where $S/n = \frac{\text{peak-to-peak picture}}{\text{rms noise}}$ in dB

C/T	= carrier to noise temperature ratio
k	= Boltzman's Constant
Δf	= peak video signal deviation
f_m	= top video frequency
6(dB)	= rms signal to p-p picture conversion factor
13(dB)	= pre-emphasis and weighting factor

The C/T is determined from the following assumptions:

ERP at 3 dB beam edge	34 dB W
Path Loss at 5°	197.1 dB
Isotropic Power	-163.1 dB W

$$C/T = -163.1 + G/T \ (dB)$$

The deviation (Δf) is determined from the Carson's Rule bandwidth formula as follows:

$$B = 2(\Delta f + f_m)$$

where B is the receiving bandwidth.

The receive bandwidth has assumed two values 36 MHz and 32 MHz for this calculation. The smaller bandwidth would allow for a program channel to be transmitted adjacent to the video carrier which is the envisaged scheme. With an earth station of 29 dB G/T, this bandwidth would give a down path carrier to noise ratio of 19.5 dB. It should be noted that some deviation is required for energy dispersal which will affect the transmission bandwidth. This has been considered and is reflected in the results given here.

These parameters can now be substituted into the above equation from which the curves of Figure 1-7 of signal to noise vs. values of G/T were plotted.

b) Telephony

The telephony performance can be calculated from the following formula:

$$S/N = 10 \log \quad C/kT \quad \frac{f^2}{f_m^2} \quad \frac{1}{2b} \quad + P + W$$

where:

S/N = $\frac{\text{rms test tone}}{\text{rms noise}}$ in dB

C/T = carrier to noise temperature ratio

k = Boltzman's Constant

f = test tone peak deviation

f_m = top channel frequency

b = baseband channel bandwidth = 3.1 kHz

P = pre-emphasis advantage for top channel - 4 dB

W = psophometric weighting factor = 2.5 dB

Using the same ERP, path loss and bandwidth as used in the video calculation, we can derive the optimum number of channels that can be transmitted assuming a receive station G/T. If two carriers are transmitted via the same transponder the bandwidth should be reduced by half to 18 MHz and the ERP per carrier reduced accordingly. For these two cases it is assumed that earth stations with a G/T of 40.7 would be receiving the signal. For the case of telephony communication to northern stations it is assumed that multiple-carrier operation will be utilized with at least 4 carriers in the assigned RF channel. Furthermore, because of multiple carrier intermodulation, the spacing of the four carriers would have to be non-uniform thereby reducing the bandwidth available for each carrier. Therefore the maximum number of channels that can be carried has been calculated for a fixed bandwidth.

A summary of the telephony calculations is shown in Table 1.

Table 1

No. of Carriers per Transponder	No. of Channels per Carrier	Bandwidth per Carrier (MHz)	Station G/T (dB W/°K)
1	1500	36	40.7
2	600	18	40.7
4	60	4.5	27
4	72	5.4	27
4	84	6.4	27
4	96	7.6	27

c) Summary of Carrier-to-Noise Temperature Ratios

The following table outlines the required C/T ratios required for system operation. The uplink C/T ratio is that required at the satellite based upon 1,000 pWp for telephony and a signal to noise ratio of 66.1 dB for video. Similarly, the down-link contribution is based upon 3,000 pWp for telephony and a signal to noise ratio of 57.1 dB for video. The overall C/T ratio is based upon 7,500 pWp for telephony and a signal to noise ratio of 55 dB for video.

Up-path fades are not considered here because it is envisioned that some form of ground station ERP compensation will be available.

No. of Carriers per RF Channel	Capacity	C/T (dB W/°K) per carrier Up-Link	Downlink	Overall
1	TV	-125.0	-135.0	-137.1
4 to 6	60 channels	-135.1	-143.9	-147.0
2	600 channels	-123.2	-128.0	-131.9
1	1500 channela	-117.6	-122.4	-126.3

1.4 Typical TV Receive Earth Station Requirements

An examination of the data given in the preceding paragraphs enables the tentative establishment of small earth station parameters. Consider the high quality TV receive-only station where the receive S/N ratio objective is understood to be 55 dB or greater for 99 percent of the time. Calculations of the overall communications performance(Figure 1-8, Part 1.3)shows that this S/N ratio can be achieved with an earth station G/T of 28.2 dB. This is based on an EIRP of 34 dB W and a path loss of 197.1 dB (5° antenna elevation angle). The fading distribution curve (Figure 1.5, Part 1.1 d) shows that a station located at the beam edge will see an effective satellite EIRP of 33.75 dB W or greater for 99 percent of the time. It will therefore be assumed that the required earth station G/T is 28.5 dB. For a station with an uncooled parametric amplifier, the overall earth station noise temperature will be about 150°K. We have therefore:

Required Earth Station G/T ratio	28.5 dB
Earth Station Noise Temperature	150°K = 21.8 d./°K
Required Earth Station Gain	50.3 dB

With a shaped reflector and Cassegranian antenna optics, this gain can be achieved with a 30 foot antenna. A 32 foot antenna is chosen for the small receive stations in order to have a design margin of 0.5 dB on the required communications performance. The above calculations are based on the worst case condition at the satellite beam edge. All other stations located inside the beam will have extra margin on the required communication performance because of higher EIRP values.

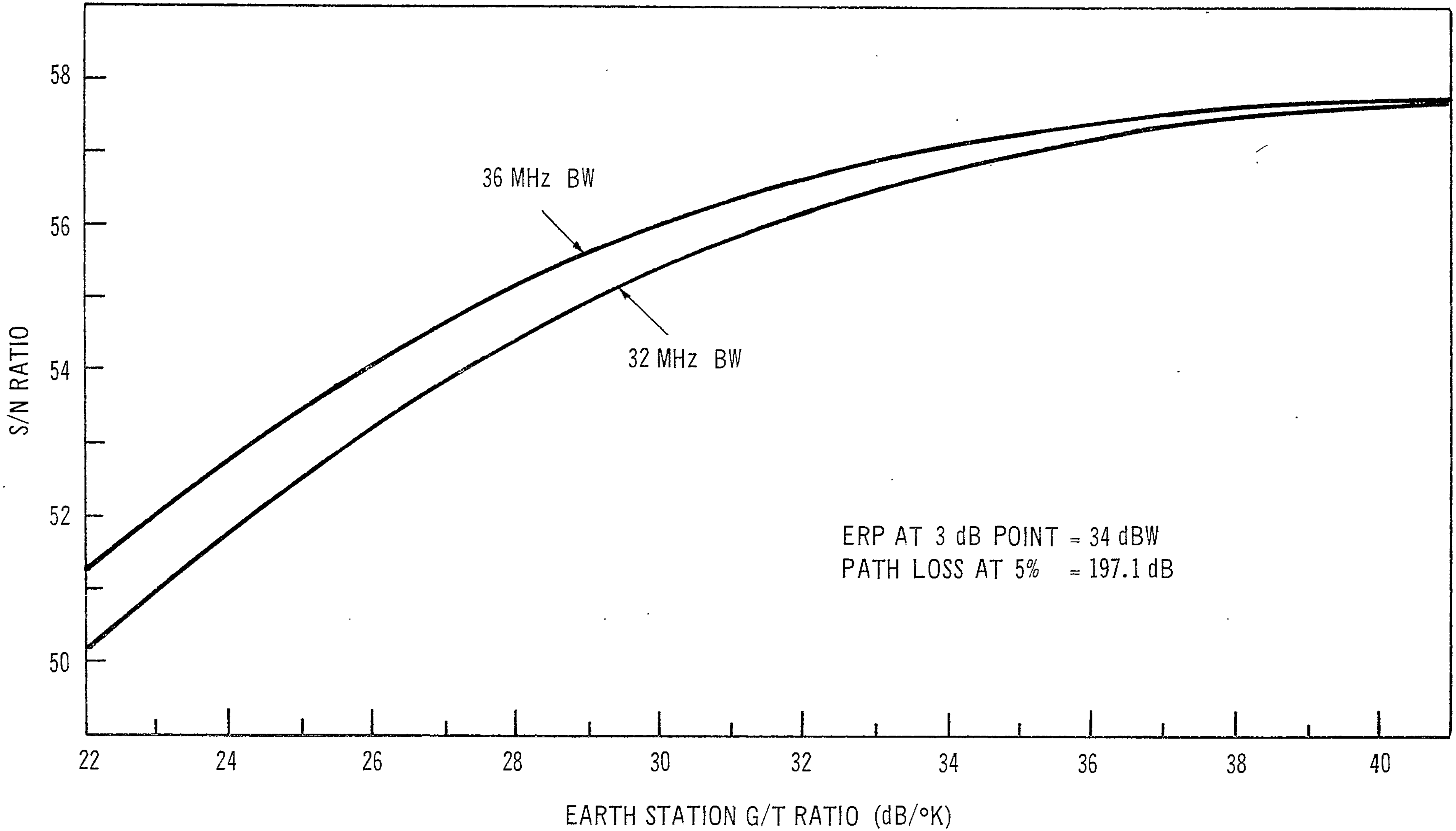


Figure 1-8 Video S/N ($\frac{\text{Peak - Peak Picture}}{\text{rms Noise}}$) versus G/T

A 32 foot antenna appears to be a near optimum choice for the TV receive station for several reasons, namely:

i This size antenna will enable high quality TV performance with 55 dB S/N ratio to be achieved with a 19 dB C/N ratio.

ii A 32 foot diameter is near the minimum acceptable size from the point of view of adjacent satellite interference if high density utilization of the synchronous orbit belt is to be achieved.

iii Fixed antennas with manual positioning capabilities can be utilized. Relatively small signal degradation due to satellite drift will be experienced.

For the nominal quality TV receive station, a simplification can be achieved through the use of tunnel-diode amplifiers at the receiver front end. These units are more reliable and lower in cost than parametric amplifiers. They also have a higher noise temperature but will still enable a receive S/N ratio of 47 dB to be achieved

1.5 Relative Cost of Earth Stations vs G/T

Earth station costs (not selling price) will cover a very wide range depending on the antenna size, G/T ratio, tracking or non-tracking antennas, the number of receivers and transmitters, site location and access, etc. It is beyond the scope of this study to make an accurate estimate of the station costs for the ground network. However, in order to obtain some indication of the probable costs for the entire system, budgetary costs for typical earth stations have been established.

Costs have been estimated for two types of station; the main communication; and control stations and the TV receive only stations. All stations are priced on the basis of a single antenna; if two antennas are used prices will increase by some factor less than 2 depending on the degree of redundancy desired. The equipment assumed for each type of station is as follows:

a) Main Communication Station (95 ft. or 60 ft. Antenna)

QTY.	
1	Antenna, fully tracking with autotrack facility
2	Cooled Parametric Amplifiers
6	Low Power (I K W) transmitters
8	Receivers (delivering video signals)
1	Control, Monitoring and Test Facility
1	Telemetry and Command System

b) TV Receive Only Station (40 ft. or 30 ft. Antenna):

Qty	
1	Antenna, manually positioned, partially steerable
2	Uncooled Parametric Amplifiers
3	Receivers

Not included in price estimates are:

. Cost of land

. Buildings and civil works

. Primary Power

. Deicing

Figure 1-9 shows the relative costs of the two types of station with a 30 ft. TV receive station cost normalized to 1.0. The specific antenna sizes and G/T ratios analyzed are:

Antenna Diameter	G/T Ratio
95 ft.	40.7 dB
60 ft.	36.7 dB
40 ft.	31.5 dB
30 ft.	29 dB

The absolute cost of the 30 ft. station installed will be in the order of 1/4 million. The remaining stations are scaled accordingly.

In summary, we emphasize again that these prices are "ball-park" values. The figures for the small stations may in fact be conservative since price reductions for large quantity supply of equipments should be achievable. Antenna suppliers have indicated that no special strengthening is required for Arctic locations although special steel, resulting in a 10% increase in antenna cost, might be necessary. Deicing might be required in most locations but this represents less than 5% of total station costs.

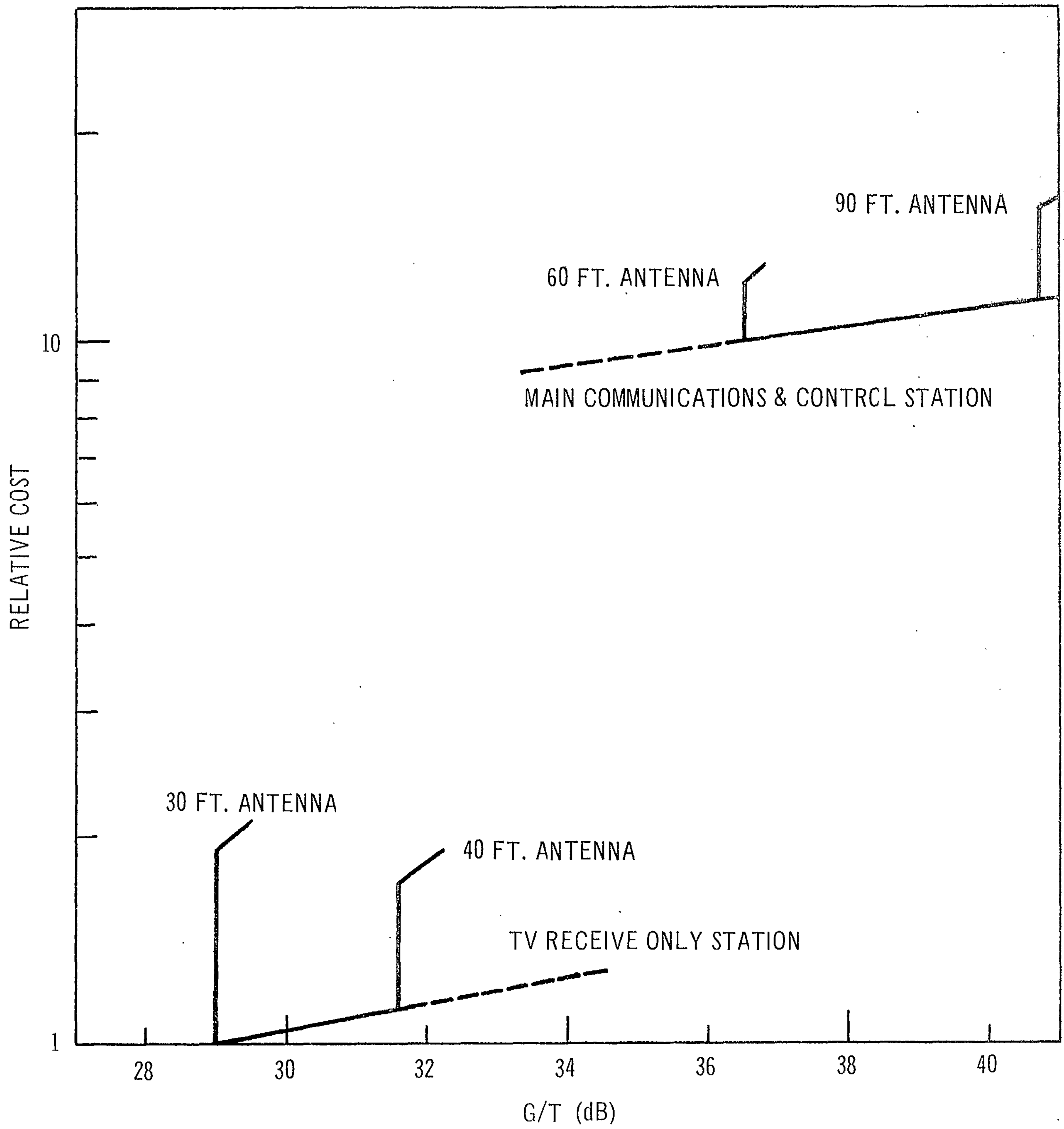


Figure 1–9 Earth Station Relative Cost versus G/T

1.6 DESIGN PHILOSOPHY

The large variation in the minimum carrier levels at the input to the transponder is a direct consequency of the diversity of Earth Station terminals used in this system. The proposed antenna sizes for earth stations range from 30' to 95' with G/T ratios ranging from about 29 to 40.7 dBW/°K. Assuming only telephone traffic and single r.f. carrier operation, the largest station has a C/T ratio 11.7 dB higher than the smallest station for the same r.f. flux density on ground. This is utilized to increase the telephone channel capacity for carrier to 1500 channels. However, to maintain the same ratio of uplink to downlink thermal noise, the uplink carrier at the input to the transponder would have to be raised by 11.7 dB .

The smallest stations receive only TV whereas the largest stations will work primarily for Telephony. The two types of services, TV and Telephony, have different bandwidth utilizations, overall performance criteria, and also different uplink/downlink thermal budgets. This represents about 0.3 dB increase in the uplink TV carriers and hence the difference between the uplink TV and uplink Telephone carriers turns out to be 11.4 dB .

Since the transponder output level is fixed, a variation in the input levels requires an adjustable gain transponder. The transponder is designed to provide sufficient gain to the lowest level single carrier in the system and then an attenuator introduced within to provide the gain adjustment.

In a strictly linear transponder gain adjustments pose no serious problems. However with any nonlinear elements present multicarrier intermodulation products limit the usefulness of the design. All amplifying devices as well as the translator used in the wideband portion of the transponder will generate 3rd order intermodulation products of the types $(2f_1 - f_2)$ and $(f_1 + f_2 - f_3)$. Also because of the equispaced frequency plan used, most of these products will fall back in channels in use.

In general the carrier to intermod ratio (C/I) is inversely proportional to the square of the operating power level of a device. The higher the C/I ratio for which the system has to be designed, the lower is the level at which the carriers have to be separated and amplified individually. This may force channelization after the 1st or 2nd stage of amplification and hence a large increase in the number of amplifiers in the transponder.

A tradeoff therefore exists between the system flexibility and the transponder complexity. System flexibility has two aspects: the range over which the carrier levels can vary and the mix of large (telephone) and small (TV) carriers.

Schematically the systems tradeoffs can be represented, as done in Figure 1-10 by satellite transponder operating ranges and relating them to the Earth Station parameters. Some of the schemes considered are:

(A) High gain, wide range:

This scheme imposes minimum uplink ERP requirements on all ground stations since the carrier level is matched exactly to the type of traffic carrier. The full brunt of flexibility is however placed on the transponder increasing its complexity.

(B) Medium gain, limited range & (C) High gain, Limited Range

By reducing the difference in the levels between various types of carriers a simpler transponder may be used. This however requires a 4 dB increase in the TV uplink carrier for scheme (B) or a decrease in the overall telephone channel capacity for scheme (C).

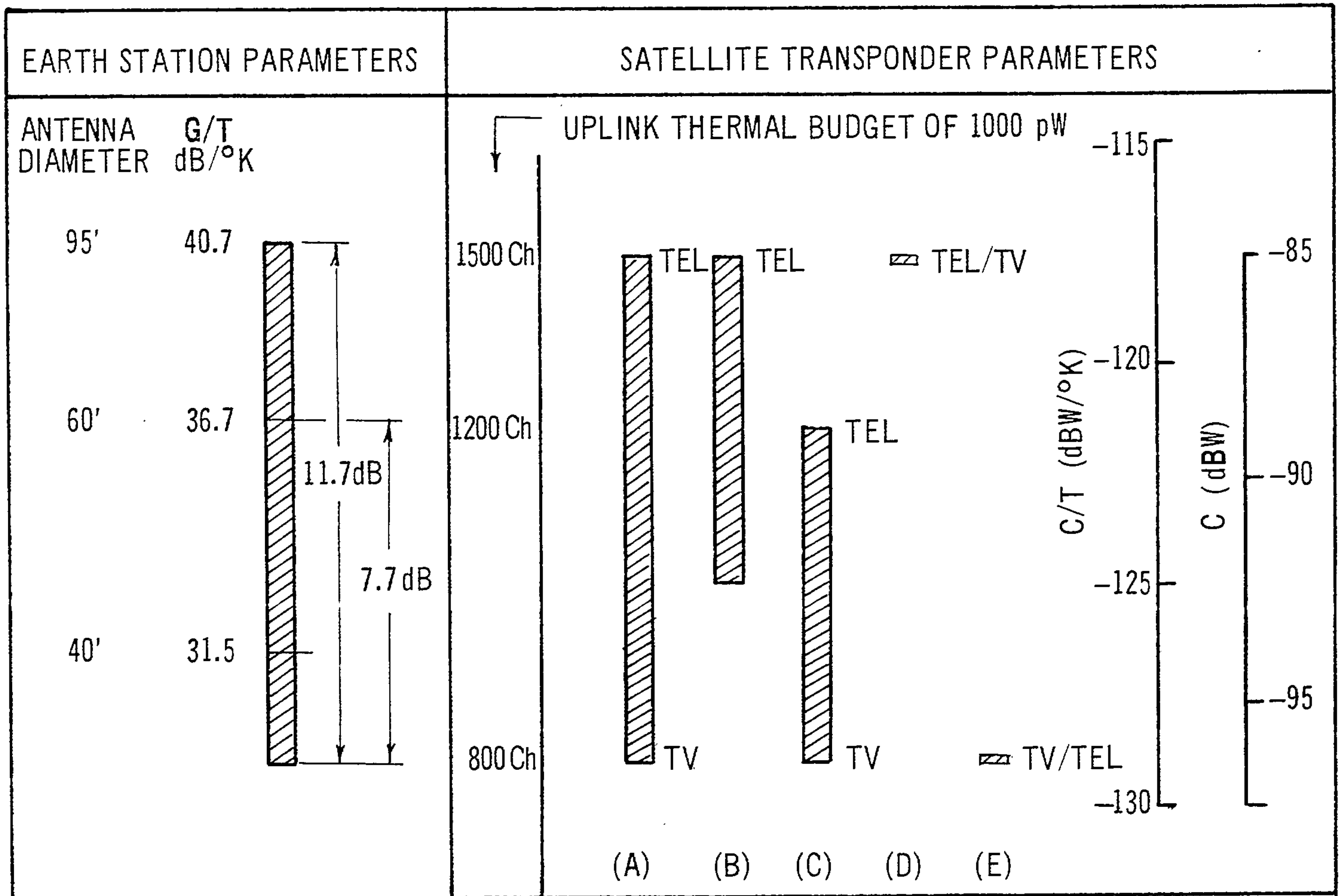
(D) Low fixed gain and (E) High, fixed gain

In the former, the TV carriers have to be increased 11.4 dB . In effect the system is designed around the largest ground stations to provide maximum available capacity while penalizing all other traffic. In the latter, the telephone carriers are dropped by 11.4 dB . The system is designed around the smaller ground stations and TV service while penalizing the total telephone capacity.

Scheme (E) results in the simplest transponder firstly because of the lack of gain control and also because of the lower C/I requirements for TV or equivalent service. The larger capacity telephone service (1500 ch) requires a C/I ratio about 10 dB higher than the TV only service leading to a stiffer design criteria for the transponder for scheme (D).

One of the major criteria in choosing a scheme of operations is the transmitter power required from earth stations. The calculated transmitted power required for various types of service from earth stations using 30', 60' and 90' antennas is given in Table 1. On some of the combinations the downlink or the uplink thermal noise exceeds its budget, these are so marked.

System B appears to be best since this case does not seem to impose severe requirements on the transponder and it will also meet the uplink thermal noise as allocated in Section 1.1e. Also the earth stations will not have to transmit excessive powers.



	(A)	(B)	(C)	(D)	(E)
	HIGH	MEDIUM	HIGH	LOW	HIGH
OVERALL TRANSPONDER GAIN (dB)	104.4	96.9	104.4	93.0	104.4
GAIN ADJUSTMENT IN TRANSPONDER (dB)	11.4	7.5	7.5	0	0

Figure 1–10 Modes of Satellite Communication Operation

TABLE 1

EARTH STATION TRANSMITTER POWER REQUIRED WITH VARIOUS TRANSPONDER SCHEMES

Antenna Diameter, Feet	30			60			90		
Receive G/T dbs/°K	29			36.7			40.7		
	Transmitter Power in DBW								
Transponder Scheme	TV	1200 ch Tel	1500 ch Tel	TV	1200 ch Tel	1500 ch Tel	TV	1200 ch Tel	1500 ch Tel
(A) Completely Flexible (High gain G= -11.4dB)	27.1 *	34.6 *	38.5 *	21.1	28.6	32.5 *	18.1	25.6	29.5
(B) Limited Flexibility (Medium gain G = -7.5 dB)	31.0	34.6 *	38.5 *	25.5	28.6	32.5 *	22.0	25.6	29.5
(C) Limited Flexibility (High Gain G = -7.5dB)	27.1	34.6 *	⌿	21.1	28.6	⌿	18.1	25.6	⌿
(D) Inflexible (Low gain G = 0 dB)	38.5	38.5 *	38.5 *	32.5	32.5	32.5 *	29.5	29.5	29.5
(E) Inflexible (High Gain G = 0 dB)	27.1	⌿	⌿	21.1	⌿	⌿	18.1	⌿	⌿

* Cannot support this capacity on downlink
⌿ Cannot support this capacity on uplink

Note: The transmit powers are based on a satellite antenna at the beam edge of 25 dB to be conservative. A more realistic estimate of the antenna gain is 26.85 dB (see section 1 of Vol. II)

1.7 EARTH COVERAGE BY SATELLITE ANTENNA

a) Philosophy:

For a satellite of a given weight whose main purpose is the distribution of television, the prime function of the antenna or indeed of the whole satellite is to cover the area of Canada with a suitable signal level on the largest possible number of channels. In a broad sense there is a weight tradeoff between antenna size and power system size that would give a minimum total weight if the communications function were to provide only point-to-point trunk communications. In the limit then, the antenna beam could be made as small as possible consistent with the attainable pointing accuracy. It is probable that the minimum weight would occur when the sum of solar array and battery weights nearly equalled that of the antenna and portions of the pointing system. For the problem of large area coverage it becomes clear that this weight optimization gives different results. With the present solar array and battery technology the weight of these components is greater than that of the optimum gain antenna that just covers the required area.

To achieve the highest antenna gain possible becomes then a problem of fitting the smallest practical elliptical antenna pattern, to a projection of Canada as seen from the orbital arc. The limits of this arc from 80^{o}W to 120^{o}W are set by the lowest practical ground station antenna elevation angle which is taken as 5^{o}.

The next steps in the choice of beam size and pointing depend on the satellite location in the orbital arc and on the attainable angular pointing accuracy.

Although a single satellite location may be optimum for best antenna gain the actual antenna design chosen should be acceptable for a reasonable range of locations, such as a 20^{o} arc, by appropriate beam twist about its axis and by suitable aiming. The angular dimensions of the beam determined in this way must then be increased by an amount set by the expected angular pointing error which is taken as $\pm 0.5^{o}$ in the N-S and E-W sense.

For this study the work statement specification of a 3 dB beam width of $4^{o} \times 8^{o}$ with beam centre EIRP 37 dBW was interpreted as follows:

The EIRP towards any point in Canada will not be less than 34 dBW.

For a more refined calculation of beam size, path loss variations due to slant range, and atmospheric loss are considered and the requirement interpreted in terms of a minimum flux density for the coverage area defined by the work statement.

b) Area Coverage Optimization

Figure 1-11 is a projection of Canada as seen from a satellite location at 90°W longitude and 0° latitude. On this projection, beam shapes appear as true ellipses and the selection of a minimum area ellipse can be made on a manual trial and error basis or by an iterative computer calculation. For the study program, a computer program was used to plot the coverage contours of selected beams onto a map of Canada. Various satellite and beam centre locations were used as inputs and a close approach to an optimum beam was selected by inspection for a best fit. Typical results of this are shown in Figure 1-12 which gives the projection of a 2.25° x 7.5° beam from 84°W longitude aiming at 89°W longitude and 51°N latitude.

This example represents very nearly the most Easterly satisfactory location determined by a 5° elevation angle at Inuvic and does not show any beam twist. A more satisfactory coverage can be gained by rotating the beam about its axis by about +5° (clockwise sense from the satellite) Figure 1-13.

Other more Westerly locations with different amounts of beam twist have been examined for coverage. While the foregoing shows the method of beam selection, the results are not necessarily final. There is perhaps a small advantage to be gained by elongating the beam 0.25° to give better E-W coverage from some satellite locations. The resulting change in gain would be negligible.

c) Effect of Pointing Error

To offset the effect of an anticipated ± 0.5 degree pointing error the minimum beam size must be increased by the amount of the error as a close approximation. A 3.25° x 8.5° beam is plotted in Figure 1-14, where the beam centre has been displaced from its nominal position by 0.5° in both N-S and E-W senses. The inner contour then represents the locus of the beam edge for the expected pointing error and is equivalent to the coverage area defined by the 2.25° x 7.5° undisturbed beam. The importance of beam pointing error on antenna gain is shown in Figure 1-15 where change in beam dimensions and the resulting change in gain is shown as a function of pointing error.

Satellite Subpoint 90° W. Longitude
Beam Pointing 50° N. Latitude, 90° W. Longitude

Figure 1–11 Canada as viewed from satellite

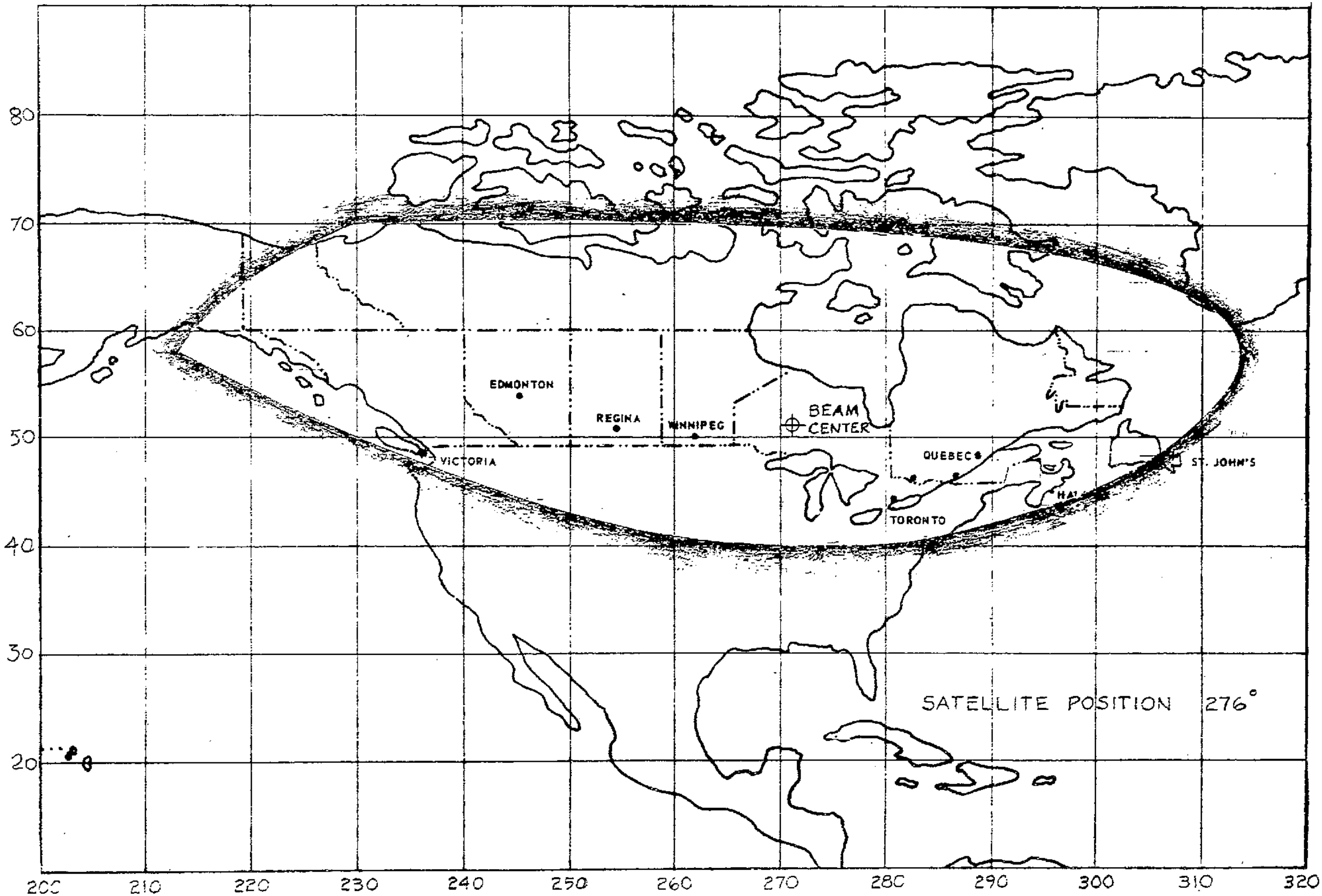
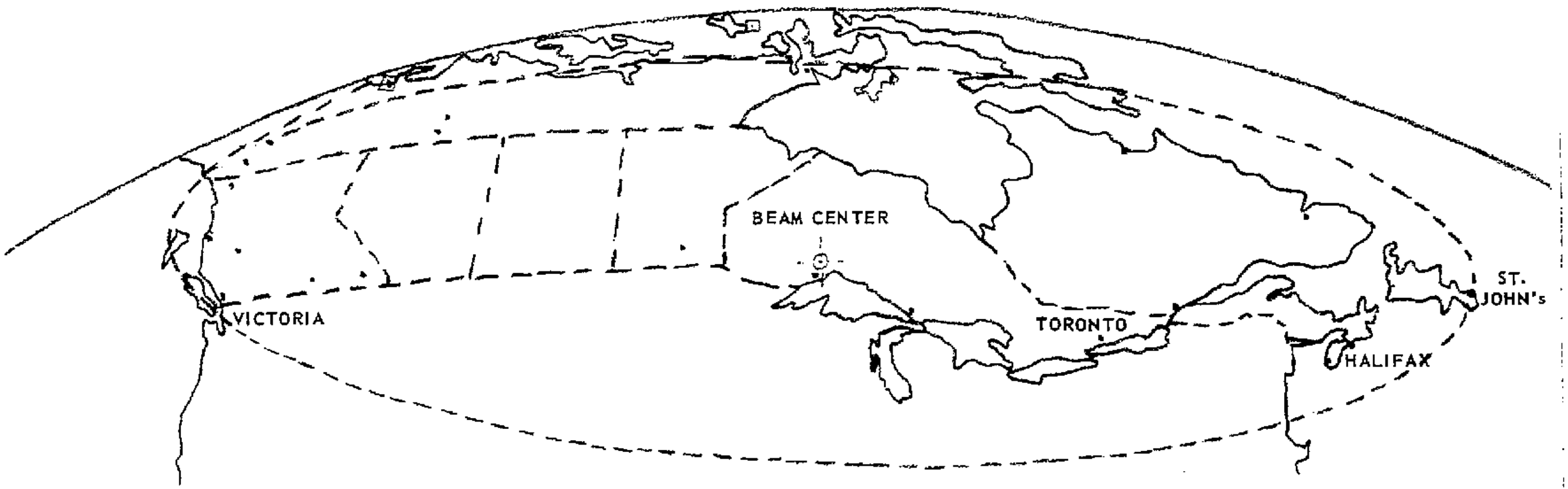


Figure 1–12 2.25 x 7.5° Beam Coverage



Satellite Subpoint 90° W. Longitude
Beam Pointing 50° N. Latitude, 90° W. Longitude
Beam Twist +5°

Figure 1-13 Canada as viewed from satellite

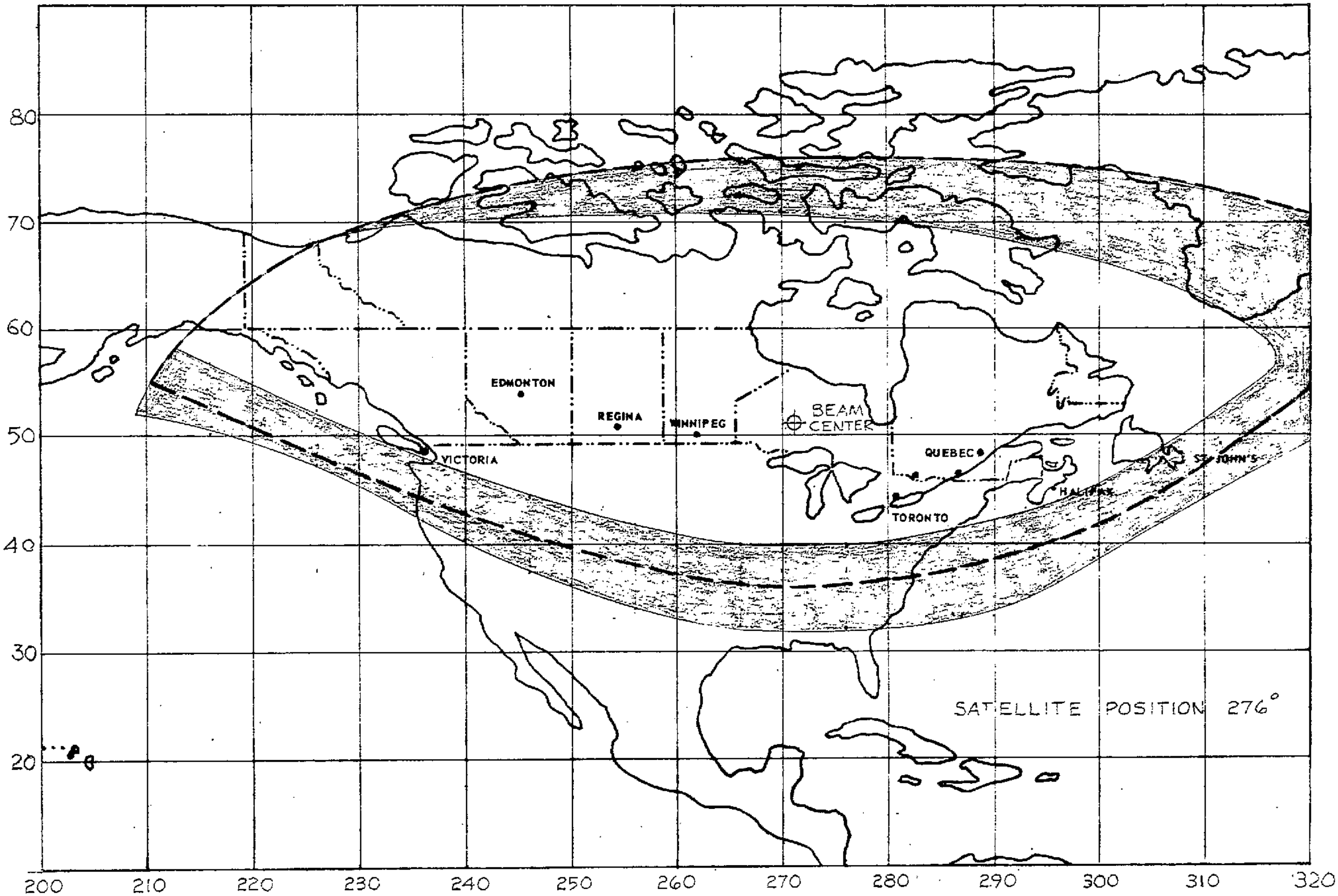


Figure 1–14 3.25 x 8.5° Beam Coverage, including Pointing Error

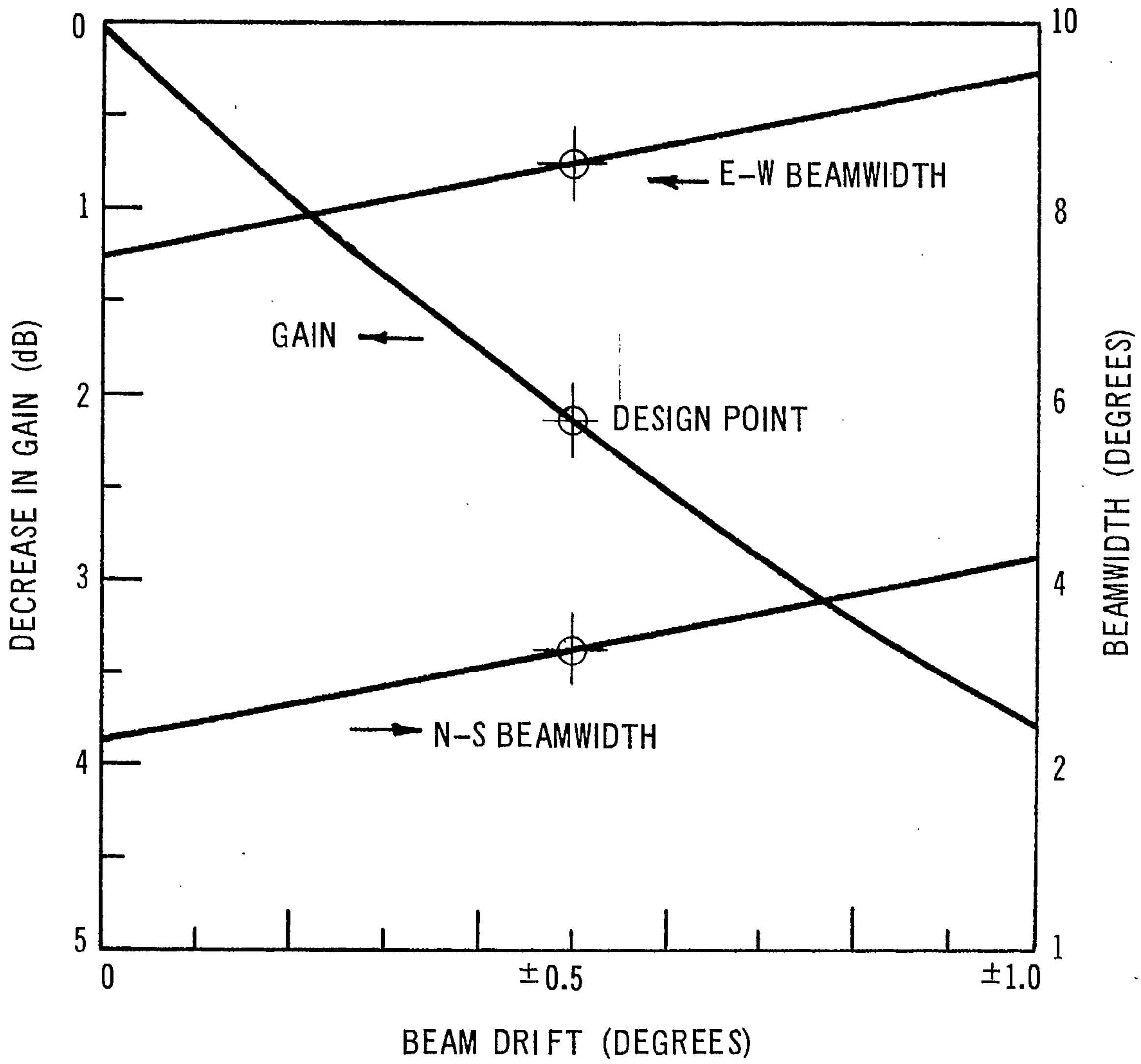


Figure 1-15 Antenna Gain as a function of Pointing Error

d) Optimization of Antenna Gain at the Beam Edge

When the minimum angular size of the beam has been selected it is necessary to decide what point is to be taken on the gain function of the antenna as the beam edge. This antenna gain function depends on the illumination function and two cases have been chosen for optimization, namely a cosine squared illumination and a lambda function illumination. The procedure is to write the expression for the gain as a function of angle and beam width and to differentiate the expression with respect to beam width and set to zero.

For comparison Figure 1-16 shows the gain as a function of angle of two cosine squared illumination antennas, with the 3 dB beam width of one equal to the 4.34 dB beam width of the other.

The ordinate of the curves may be interpreted as the EIRP produced by two antennas fed the same amount of power. For equal beam edge EIRP the narrower antenna is seen to produce a significantly higher ERP at beam center which may be useful when beam pointing error statistics are considered.

The optimization procedure is illustrated in Figure 1-17 which is a plot of the antenna gain vs the antenna beamwidth. The antenna beam shape in the main lobe was assumed to be parabolic-gain in dB $(\theta/\theta a)^2$ - and the half-neper beamwidth was normalized to the required angular coverage. Similarly the antenna gain is normalized to the beam center gain of the optimum antenna. Starting from a very small beam, Go, the gain at beam center is high but the edge of the coverage area sits very low on the beam pattern and has therefore very little gain (Ge). As the beam is widened, Go decreases at a rate of 2 dB/dB but Ge goes through a maximum of -4.34 dB (1/2 Neper). With further widening of the beam both Go and Ge decrease monotonically.

Similar curves can be drawn for other beam shapes created by different illumination functions, e.g., Ge has a peak value of -4.03 dB referred to optimum Go, for a lambda function illumination.

e) Beam Edge Optimization for Restricted Antenna Sizes

The foregoing section has shown the method of choosing an optimum beam size and the corresponding antenna size when no restrictions exist on antenna dimensions. When restrictions exist such as a maximum weight allowable within the existing Delta shroud, new size optimizations exist which are not obvious on first inspection.

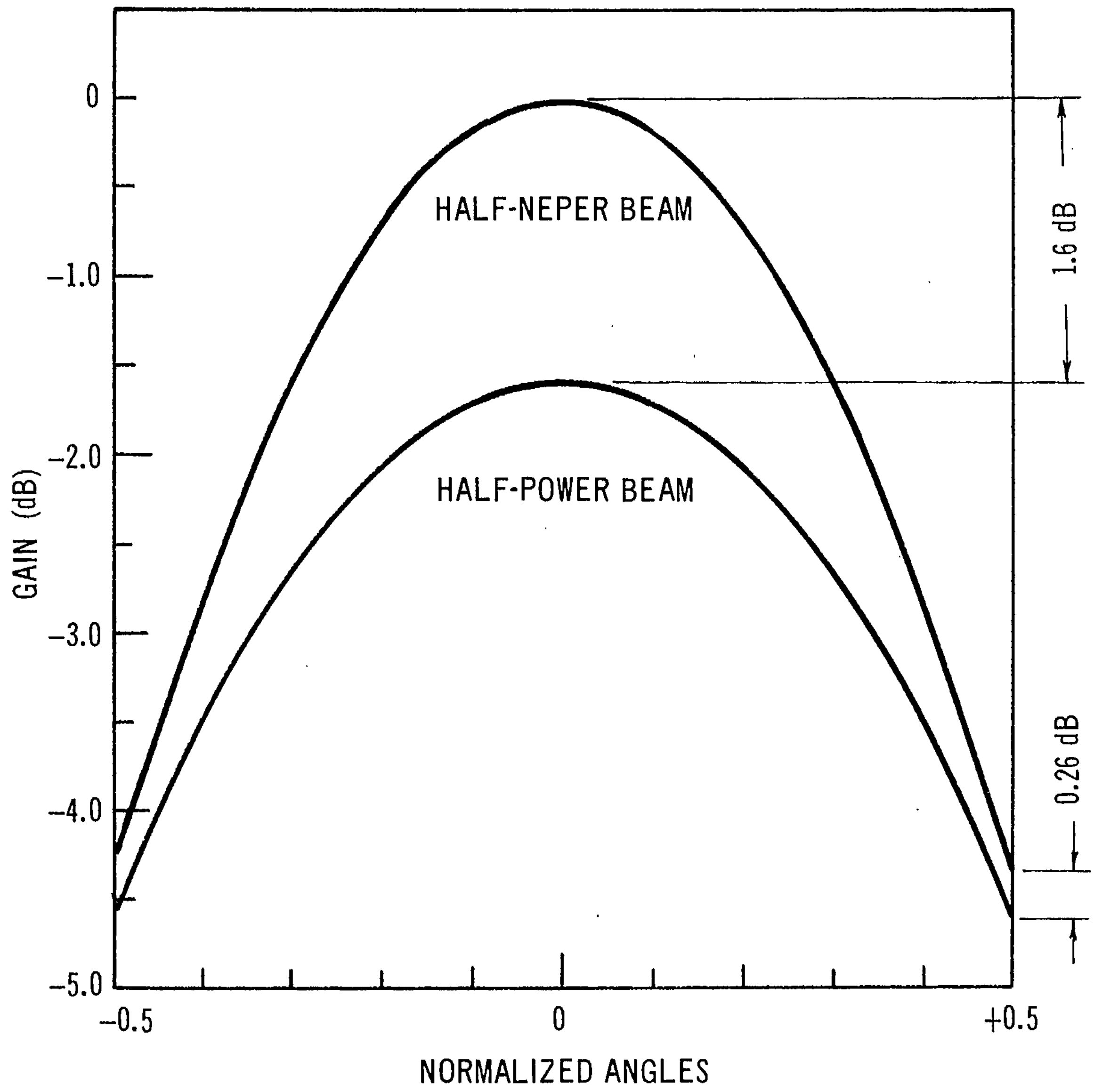


Figure 1-16 Comparison of Half-Power and Half-Neper Beamwidth

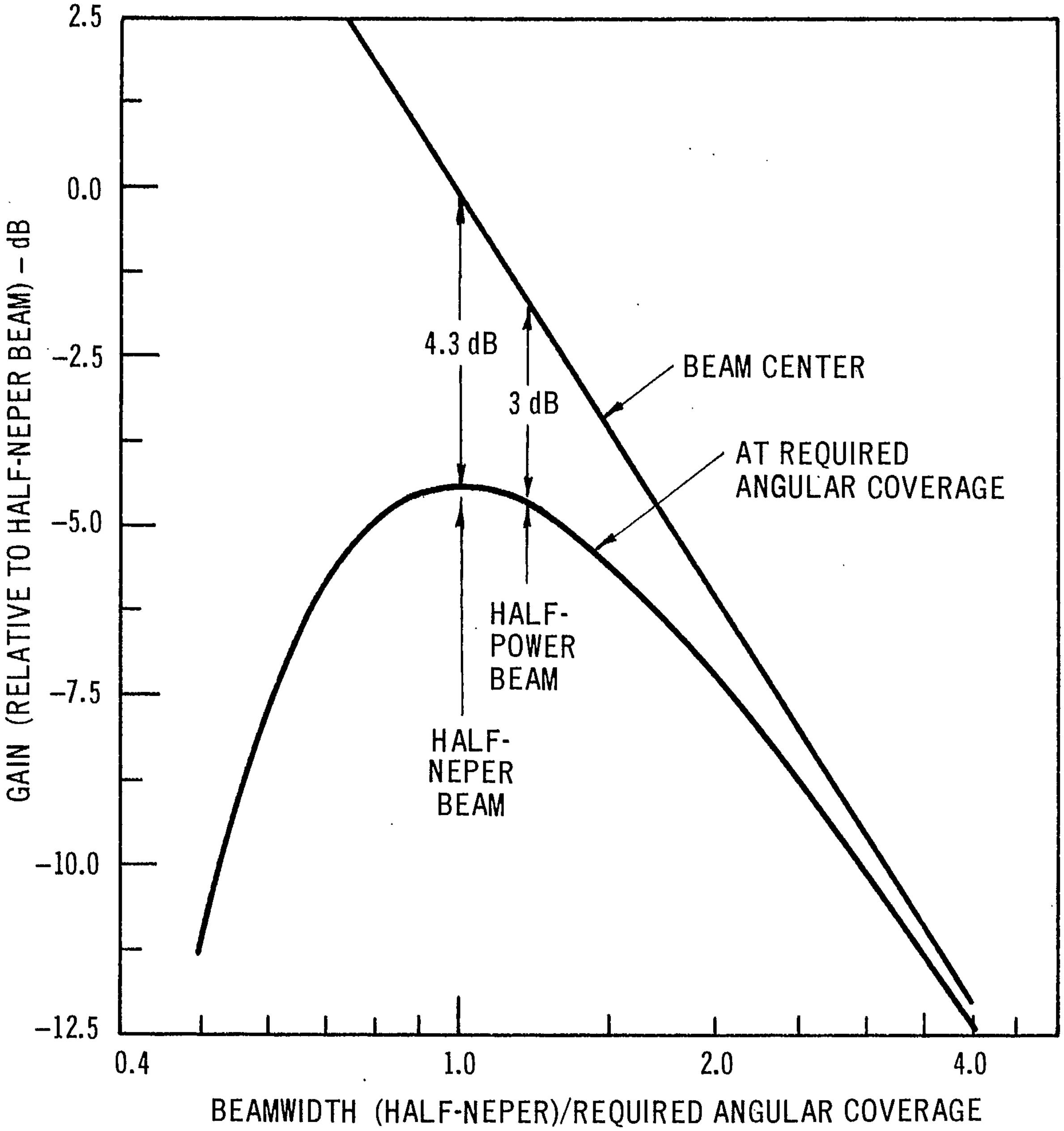


Figure 1-17 Optimization of Beam Edge Gain

Figure 1-18 is a useful plot of loss in optimum edge gain as a function of beam dimensions. The axis are beamwidths, normalized to the optimum beamwidth, and the curves represent the locus of all possible combinations of beam size that give the same loss in edge gain. The optimum antenna is represented by the point 1.0, 1.0 and the line sloping up at 45° represents the locus of the minimum size (weight) antenna for a given loss in edge gain. If we take antenna height as an independent variable the optimum path for best edge gain is, along the minimum size line until $\theta y/\theta_B = 1.4$ at which point the path is along a line of constant $\theta x/\theta$ or East-West beamwidth.

Referring to Figure 1-18, the letters A to E are explained by the following legend and represent the path of various trade-offs and optimizations.

(A) Fixed Available Power - No size or weight restrictions: Maximum ERP at beam edge.

(B) Fixed Available Power - Backoff beam edge ERP to reduce Antenna weight or size

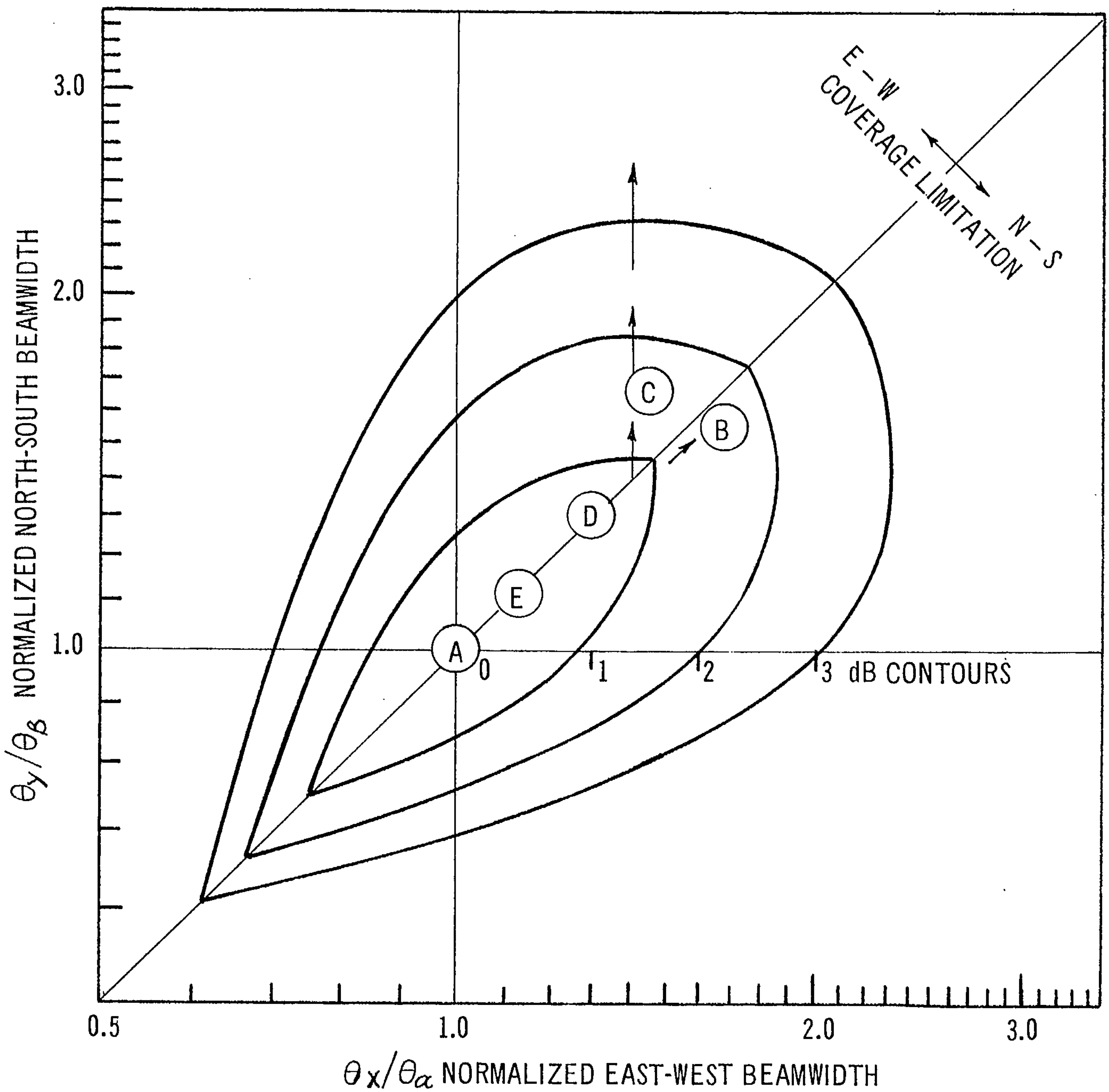
(C) Fixed Available Power - Backoff ERP to accomodate restricted Antenna height, no restrictions on weight or width.

(D) Satellite Height Restricted - Tradeoff antenna height vs array height

(E) Satellite Weight Restricted - Tradeoff antenna weight vs array battery and structure weight

f) Effect of Frequency on Beam Size

As the frequency of operation increases the beam size decreases while the gain on beam axis increases. It can be shown that these effects counterbalance each other if the beam size is chosen according to the optimization described above. This is true over the 500 MHz band from 3700 MHz to 4200 MHz, for which all the preceding discussions apply, but does not apply to the operation of the antenna as a 6 GHz receiving antenna. For this band it will be necessary to defocus the antenna or control the illumination in such a way as to produce a beam of nominally the same size as the 4 GHz beam.



NOTE: BEAMWIDTHS MEASURED AT HALF-NEPER-4.34 dB-POINTS

Figure 1–18 Decrease in Antenna Gain at the Edge of Required Area Coverage

g) Effect of Slant Range on Coverage

For each point on the earth within sight of the satellite, a particular slant range and elevation angle are associated which together define both the free space loss and the atmospheric loss. This information can be shown parametrically as a function of station longitude and latitude relative to the satellite (Figure 1-2). It can also be combined with an assumed antenna beam size and shape, such as that previously derived, to generate contours of equal flux density on the earth. This is shown in Figures 1-19 to 1-22 for different satellite locations, beam pointing, and beam twist for a beam pointing error of $\pm 0.5^{\circ}$. For purpose of comparison Figure 1-23 shows similar flux contours for a zero pointing error.

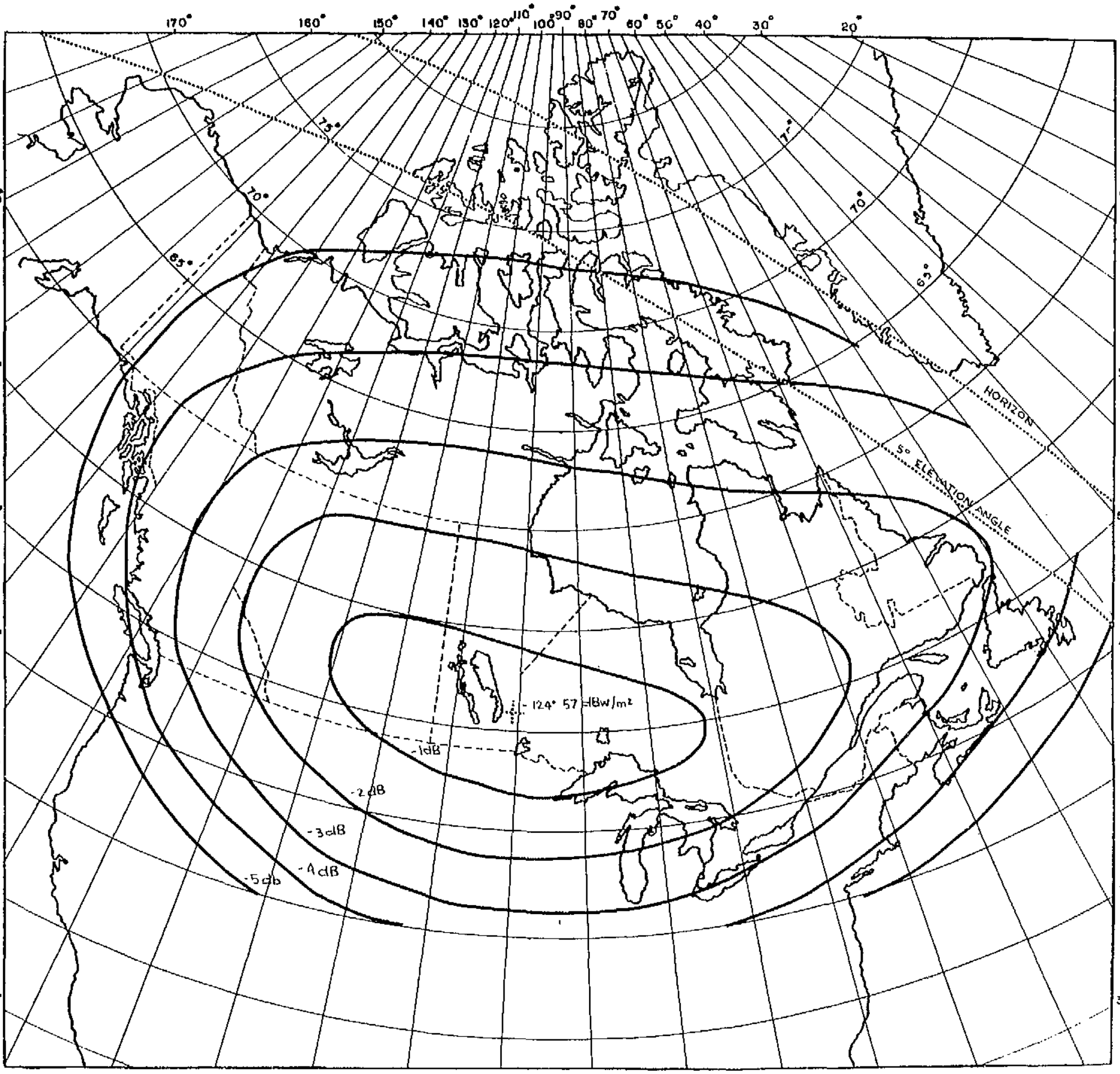


Figure 1–19 Flux Density Contours 3.25° x 8.5° Beam. From 249° E Longitude with 10° Twist. Pointing Error ±0.5°.

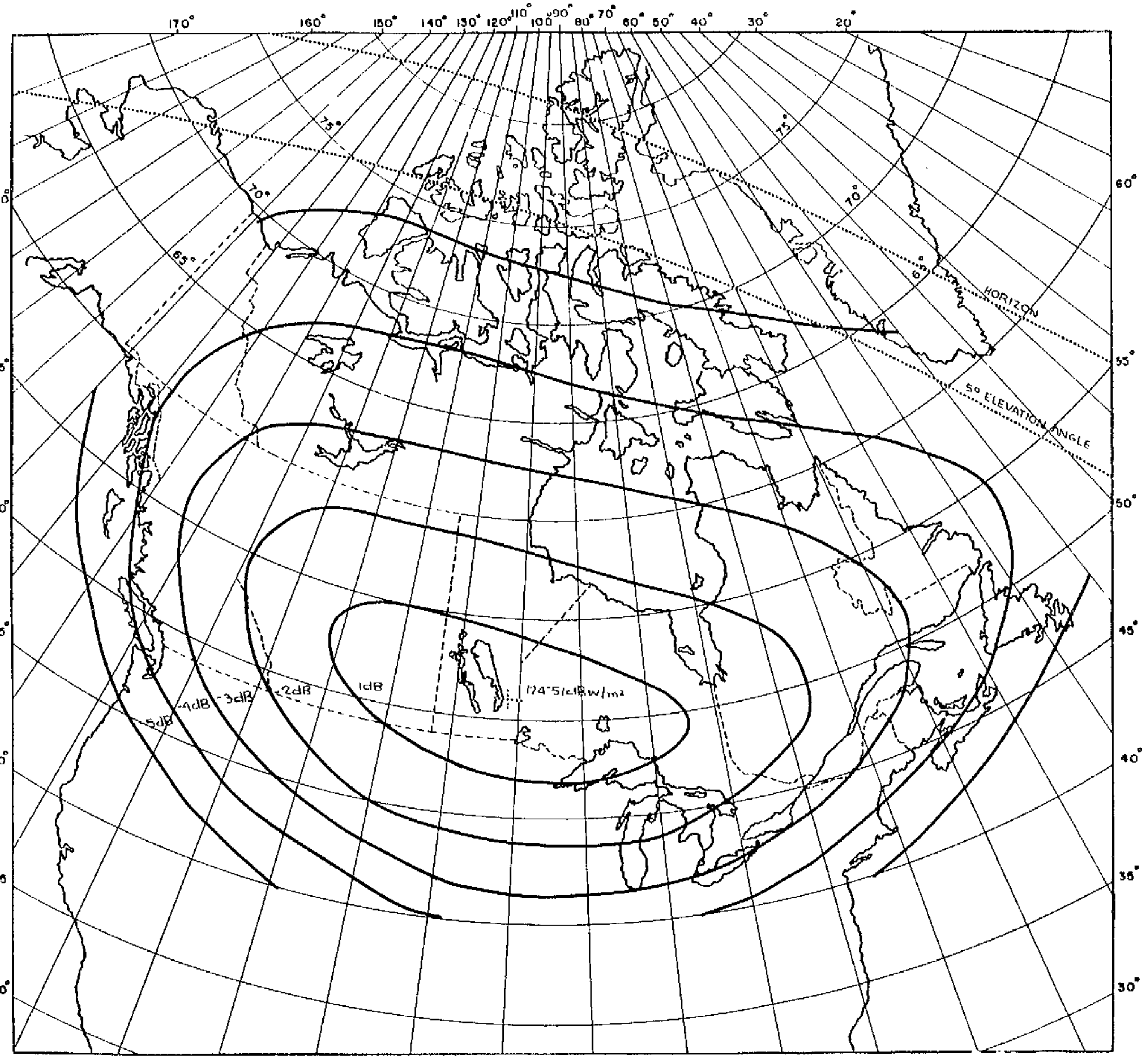


Figure 1-20 Flux Density Contours 3.25° x 8.5° Beam. From 249° E Longitude with 9° Twist. Pointing Error ±0.5°.

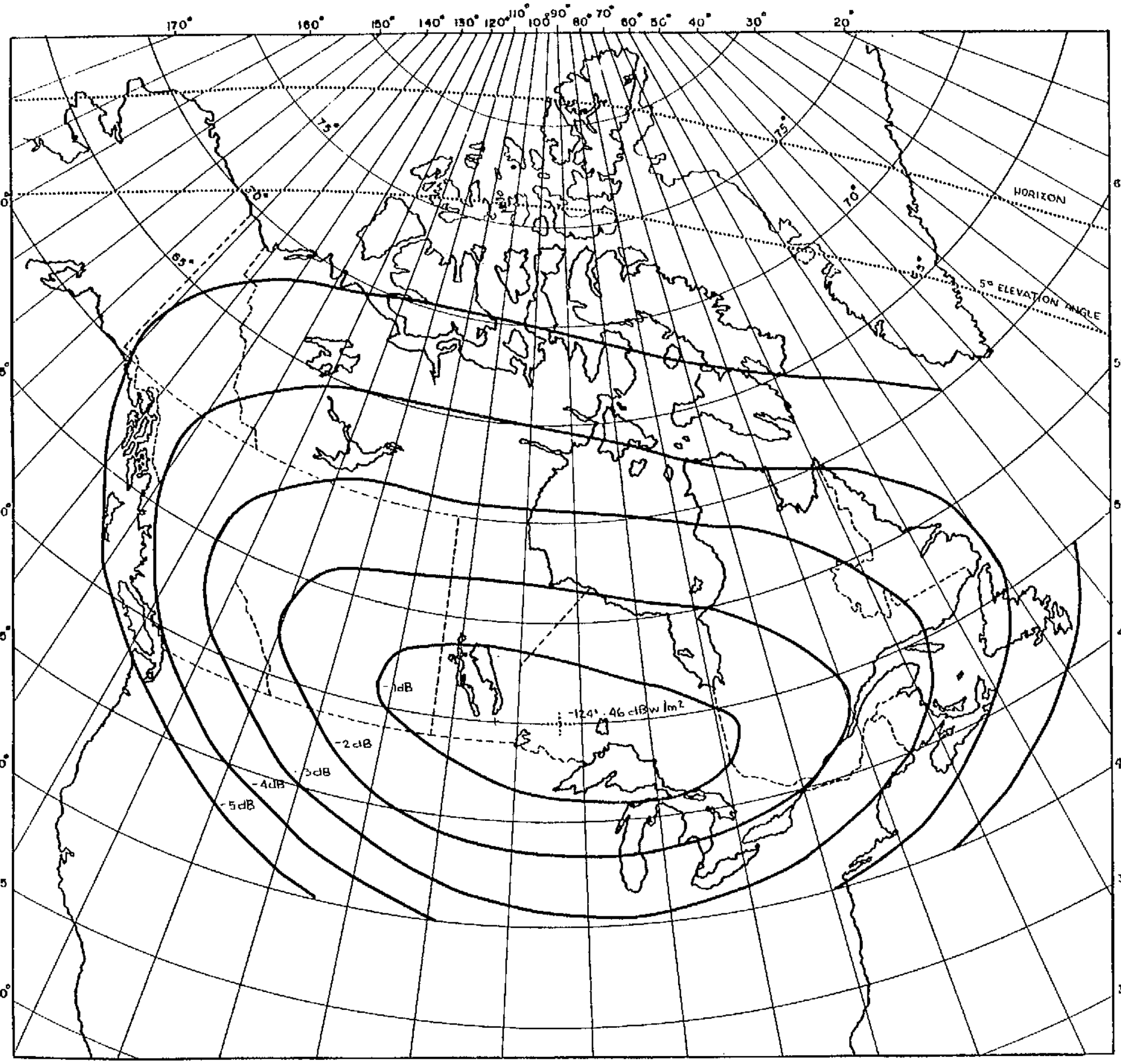


Figure 1-21 Flux Density Contours 3.25° x 8.5° Beam. From 261° E Longitude with 6° Twist. Pointing Error ±0.5°.

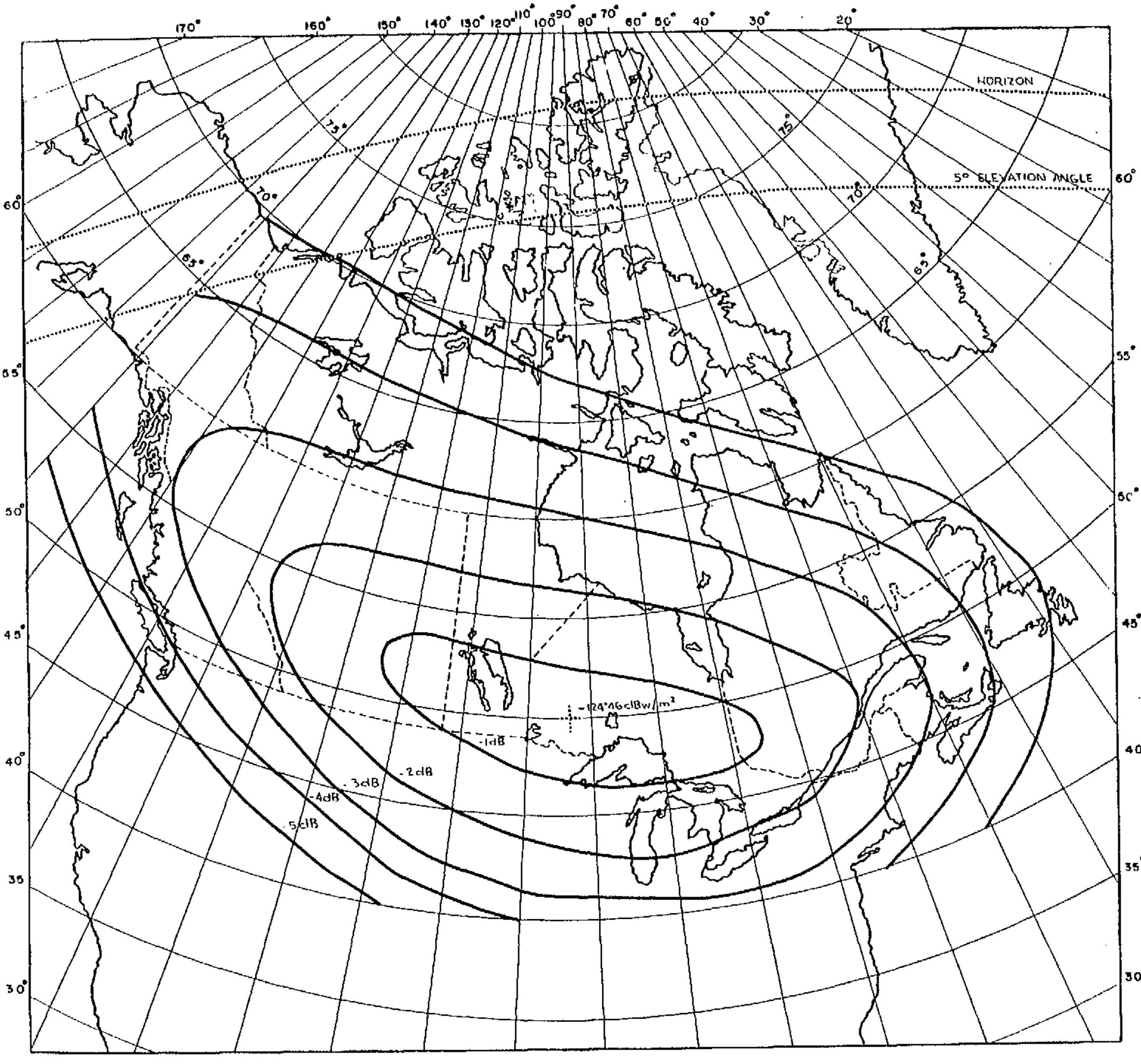


Figure 1–22 Flux Density Contours 3.25° x 8.5° Beam. From 276° E. Longitude with 5° Twist. Pointing Error ±0.5°.

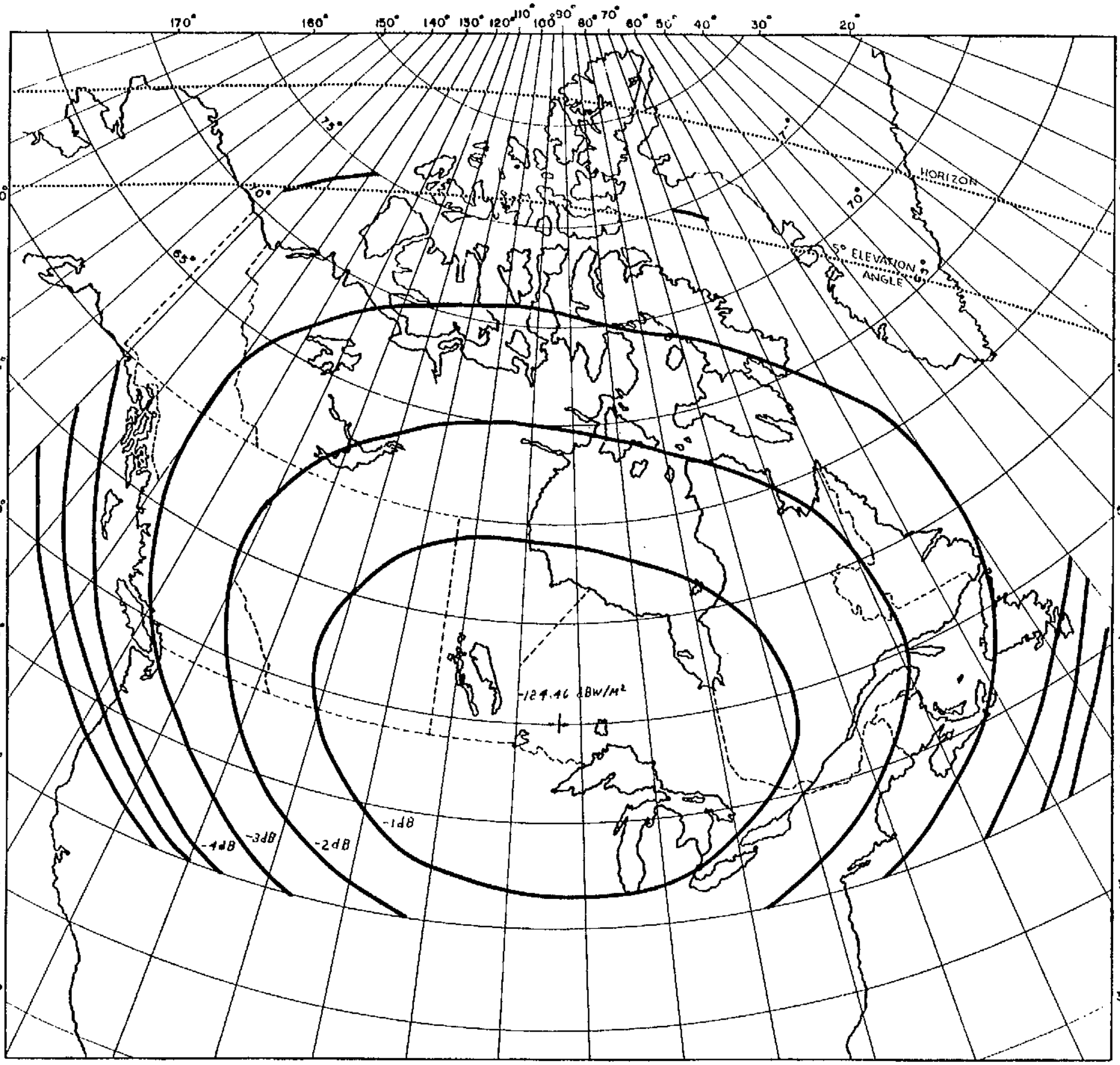


Figure 1–23 Flux Density Contours 3.25° x 8.5° Beam. From 261° E Longitude with 6° Twist.
Showing effect of zero Pointing Error.

MISSION ANALYSIS

2. MISSION ANALYSIS

2.1 LAUNCH VEHICLES

Two potential launch vehicles are available for the Canadian Domestic Communications Satellite Mission--one being the Thor-Delta (DSV-3L^2) built by McDonnell Douglas and the other being the Atlas/Burner II built by General Dynamics/Boeing. Both of these vehicles were examined in detail to determine their mission capabilities, fairing development status, and booster development status. The Description of each configuration is contained in the following sections.

2.1.1 Thor-Delta (DSV-3L^2)

Figure 2-1 shows the general configuration of the DSV-3L^2. The DSV-3L^2 is basically the same as the DSV-3L except that six strap-on solid motors are used on the booster instead of three. This configuration has recently been given a hardware contract go-ahead by NASA with the first launch scheduled in May, 1969. The total booster is made up of three stages. The first stage is the DSV-2L-1B with strap-on solids (6 Castor II). The second stage is the improved Delta, and the third stage is the TE-364-3 solid motor.

Douglas has performed a booster capability study to determine how much payload weight can be placed in a transfer orbit up to a synchronous equatorial apogee. Based upon their results, the maximum payload weight is 1005 pounds, including the interstage structure with all its separation equipment weighs a nominal 40 pounds. Therefore, the net payload weight into orbit is 965 pounds.

For the particular trajectory at which this booster capability exists, the orbit inclination is 28.7 degrees, which means that the payload must supply a ΔV capability of 6040 ft/sec to circularize the orbit at synchronous altitude. This particular trajectory does not completely satisfy all the present radio guidance constraints; however, it is expected that a modification will be made to the guidance antennas which will permit this trajectory to be flown. This type of modification has already been incorporated into Thor-Agena launch vehicles.

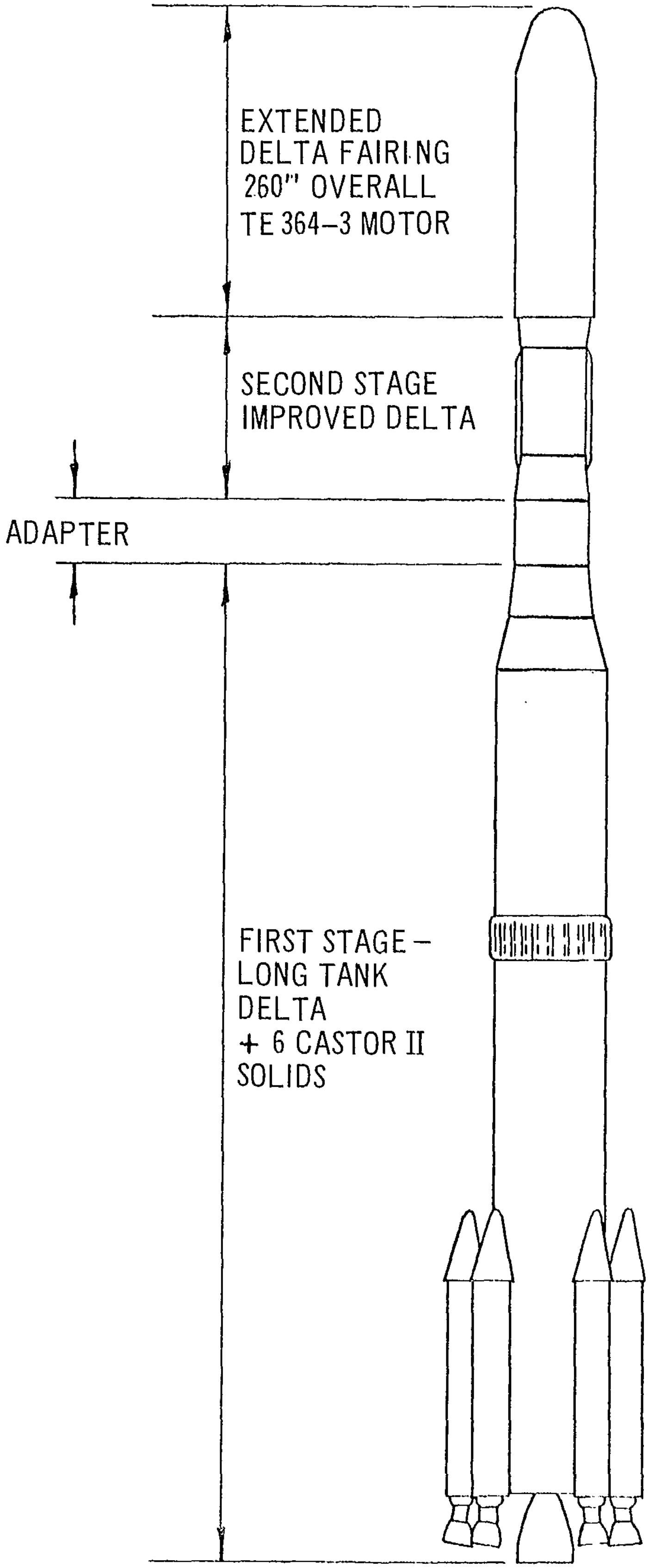


Figure 2-1 Thor-Delta Launch Vehicle DSV 3L[2]

The present fairing being used by the Thor-Delta is too short to incorporate the communications antenna and the required solar array for optimum six channel design. Douglas has proposed to develop a new fairing with the same diameter but approximately 263 inches long. An outline of this fairing toghether with the present one, is shown in Figure 2-2. The weight of the improved fairing is 675 pounds. This increased weight was included in the calculation of the net payload weight into orbit.

Another important booster performance parameter is the total orbit dispersion produced at synchronous altitude due to booster errors. This dispersion must be removed by the payload through the use of its own propulsion system. At the present time, only preliminary data is available to determine what this number should be. Based upon an error covariance matrix received from Douglas, the velocity dispersion could be as high as 500 ft/sec. However, it is expected that proper trajectory biasing and optimization will reduce this number. Therefore, a nominal ΔV of 300 ft/sec is currently being used in the design which corresponds to the Intelsat III dispersion, which flies on a similar type booster.

2.1.2 Atlas Burner II

The Atlas-Burner II is a relatively new booster combination which was flown for the first time in August, 1968. Because of a fairing malfunction, the flight was unsuccessful. This configuration is shown in Figure 2-3. The Atlas portion is the standard SLV-3A and the Burner II stage is the same stage flown on six Thor-Burner boosters, all of which were successful. The Burner II stage includes a strap-down inertial guidance with a programmer and velocity meter which enables this payload to be accurately injected into its transfer orbit. This is in contrast to the Thor-Delta which relies on the booster for attitude reference and injection into a transfer orbit by a spun-up last stage. This type of injection by Burner II produces a total booster dispersion equivalent to an error of 180 ft/sec at synchronous apogee which is considerably better than the Thor-Delta. This smaller booster dispersion is equivalent to about 8 pounds of booster payload because of the smaller quantity of hydrazine required.

The Atlas-Burner II is capable of placing a payload weight of 1005 pounds into a transfer orbit up to a synchronous equatorial altitude.

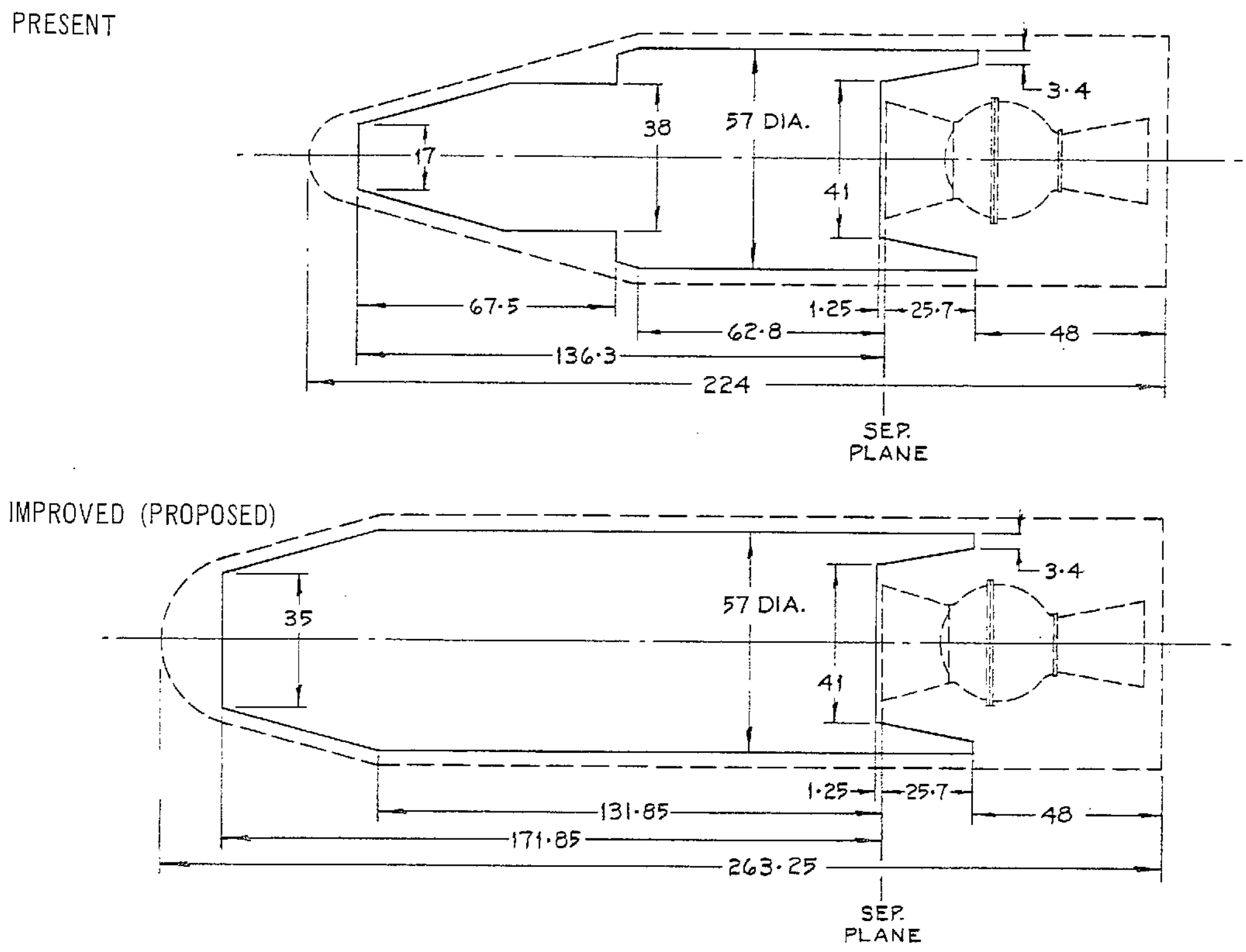


Figure 2–2 Thor-Delta Fairings

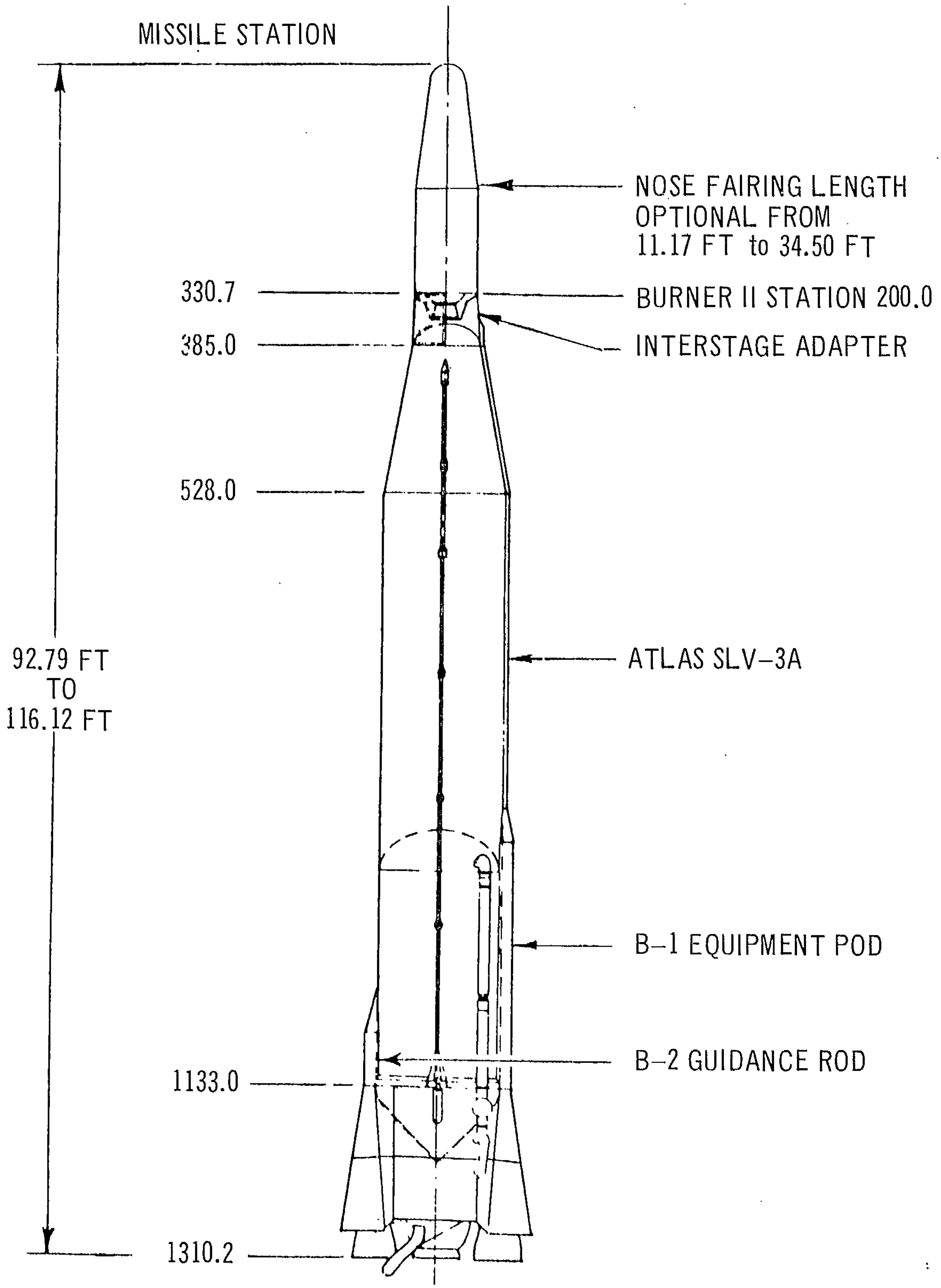


Figure 2–3 Atlas SLV–3A/Burner with 65-Inch Diameter Nose Fairing

This number is coincidently the same as the Thor-Delta except that the interstage structure has already been subtracted. For the type of trajectory flown by the Atlas-Burner II (a bi-elliptic orbit transfer), the required velocity to inject into a circular synchronous orbit is 6090 ft/sec.

Because of the size of this satellite, the standard Burner II fairing would have to be extended approximately 140 inches, thereby increasing the fairing weight up to 790 pounds. This type of modification is relatively simple; a similar type modification has already been made for the launch in August.

2.1.3 Performance Comparison of Thor-Delta and Atlas Burner II

	Thor-Delta	Atlas Burner II
Weight into transfer orbit	1005	1005
Interstage structure	40	0
Net weight into transfer orbit	965	1005
Solid motor weight	503	526
Hydrazine propellant	69	62
Net useful payload into orbit	393	417

Therefore, the net useful payload difference between the two boosters is 24 pounds.

2.2 LAUNCH CONSTRAINTS

2.2.1 Mission Sequencing Plan

A particular sequence of events will be followed in order to place the satellite into a synchronous, equatorial orbit at approximately 100°W longitude (midway between the eastern and western Canadian boundaries). The events are outlined below and discussed in more detail in subsequent sections.

- Lift-off

  Lift-off will take place from ETR. Two launch vehicles are currently being considered; Thor-Delta manufactured by Douglas and an Atlas/Burner II manufactured by General Dynamics and Boeing. (The Atlas configuration is SLV-3A.)

. Transfer to Synchronous Altitude

The Thor/Delta executes lift-off, coasts to the first southerly equatorial crossing and injects the satellite into a transfer ellipse with apogee at synchronous altitude.

Two ascent modes are available with the Atlas/Burner II configuration. The first is a Hohmann transfer from a 100 n.mi. circular parking orbit, and the second is a bi-elliptic ascent. Both of these modes (which are elaborated on in a later section) use a transfer ellipse with apogee at synchronous altitude.

. The Drift Orbit

When the satellite has coasted to apogee of the transfer ellipse, an apogee injection maneuver is performed to place it into a nearly circular, equatorial drift orbit. The drift orbit is required because, in general, the longitude at apogee on the transfer will not coincide with the desired stationary longitude. In order to obtain a more favourable initial longitude in the drift orbit, it may be desirable to delay the apogee injection maneuver until the second or third (or even later) apogee crossings. Each revolution in the transfer orbit takes about 10.5 hours and shifts the longitude of apogee westward by approximately 158°.

. Final Positioning

When the satellite reaches the proximity of the desired stationary longitude, a final positioning maneuver will be performed to circularize the orbit (remove the drift velocity), and take position at the desired stationary longitude.

. Station Keeping

Due to the ellipticity of the earth and the perturbations of the sun and moon, the satellite will drift in longitude and latitude from the location at which it was originally positioned. Thus periodic station keeping maneuvers will be required to maintain the spacecraft's position. These maneuvers will control longitudinal drift and orbit inclination build-up (latitude drift).

. Attitude Control

During its lifetime, the satellite will require periodic reorientation to its nominal attitude, due to the combined effects of magnetic moments, gravity torques, solar pressure torques, etc.

2.2.2 Launch Ascent Phase

Two types of elliptical transfer trajectories can be used to establish a synchronous equatorial orbit, depending on the flexibility of the launch vehicle. The usual mode of ascent is the Hohmann transfer (left side of Figure 2-4). The initial burn of the launch vehicle establishes a low altitude circular parking orbit. Upper stage burn at the desired nodal point provides the velocity to establish an elliptical transfer with apogee over the equator at synchronous altitude. The third burn is performed at apogee to simultaneously remove the inclination and circularize the final orbit.

If in the Hohmann transfer, a non-restartable launch vehicle is used with greater capability than required to reach the initial low altitude parking orbit, the remaining impulse is lost, since the booster cannot restart to augment the transfer vehicle burn at the node of the parking orbit. The bi-elliptic * non-coincident line of apsides transfer technique (right side of Figure 2-4) uses the full boost vehicle capability to establish an elliptical parking orbit with perigee located at boost burnout. The transfer burn occurs at the point where the elliptical parking orbit and the elliptical transfer orbit are tangent. A final burn is then used for plan change and circularization at synchronous altitude.

2.2.3 Positioning Sequence

In order to attain the desired altitude, the satellite will be injected into a transfer orbit, having a perigee of about 100 n.mi. (parking orbit altitude), and having nominal apogee altitude corresponding to synchronous altitude h = 19,323 n.mi. If a non-restartable launch vehicle is used, the bi-elliptic transfer mode discussed in section 2.2.2 may be adopted. For either of these techniques, the nominal inclination of the transfer orbit depends upon launch azimuth, the latitude of the launch site, and the specified plan change (if any) carried out at the epoch of injection into the transfer orbit.

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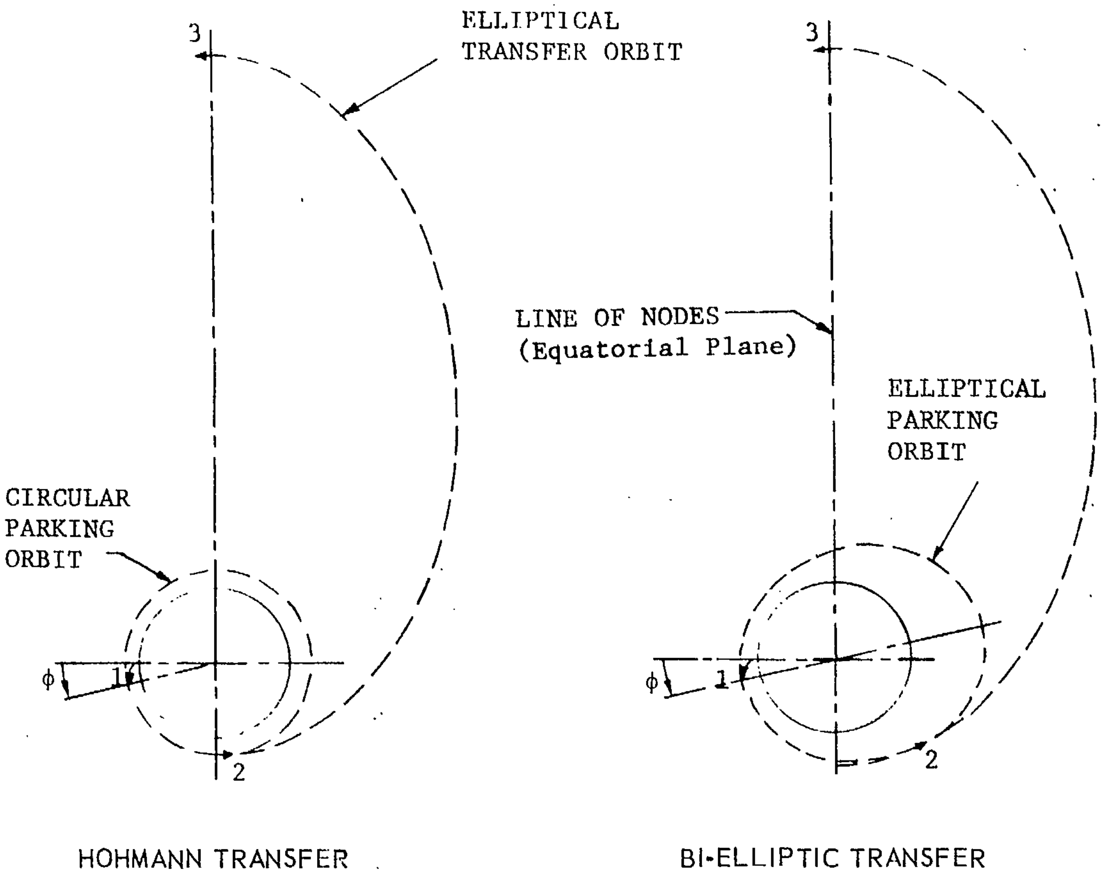
* Bi-elliptic is used here to mean cotangential transfer.



1 – First Burn to Establish Parking Orbit

2 – Second Burn to Establish Transfer Orbit

3 – Third Burn for Final Injection and Plane Change. ϕ = Boost Range Angle

Figure 2–4 Launch Ascent Profile for Synchronous Equatorial Orbits

Once the satellite has reached apogee on the transfer orbit, an "apogee burn" is carried out which serves the purpose of injecting the satellite into an equatorial drift orbit. The nominal longitude at which the apogee burn is carried out depends upon the longitude of the equatorial crossing at which the transfer orbit is initiated, the period of the transfer orbit, and the number of revolutions in the transfer orbit prior to performance of the apogee burn. Figure 2-5 shows the variation in longitude at which the apogee burn may be carried out for a launch azimuth 93° from ETR for the first northerly and southerly equatorial crossings and for the first six revolutions in the transfer orbit. Since the longitude at which the apogee burn is carried out will not, in general, correspond to the desired stationary longitude (in the vicinity of 100° W), the drift orbit will be designed to enable the satellite to drift to the desired stationary longitude in a specified time period.

The selection of the actual mission sequence will be predicted on the constraints that:

- Apogee motor firing as well as trim maneuvers due to launch vehicle injection errors, etc., and station keeping maneuvers will be carried out from a command station located in Canada in the close proximity of 80°W longitude and 40°N latitude.

- Tracking of the satellite for the purpose of station keeping as well as positioning (during the entire drift phase) will be carried out from Canadian based tracking stations.

These constraints impose the restriction that the apogee burn must be carried out at longitudes which are visible to the ground station. Acceptable regions for the apogee burn are contained within the vertical dashed lines of Figure 2-5.

In addition, the latter constraints require that the drift direction be such that the satellite can be acquired by the ground station at all times during the drift phase and station keeping phases.

Several possible ascent sequences are suggested by Figure 2-5. In general, it will be desirable to inject into the transfer orbit from either the first northerly or first southerly equatorial crossing. In fact, various launch vehicle operational constraints may require that injection be made at the first southerly crossing. When injection is made at the first southerly crossing, it follows from Figure 2-5 that a desirable sequence of events is as follows:

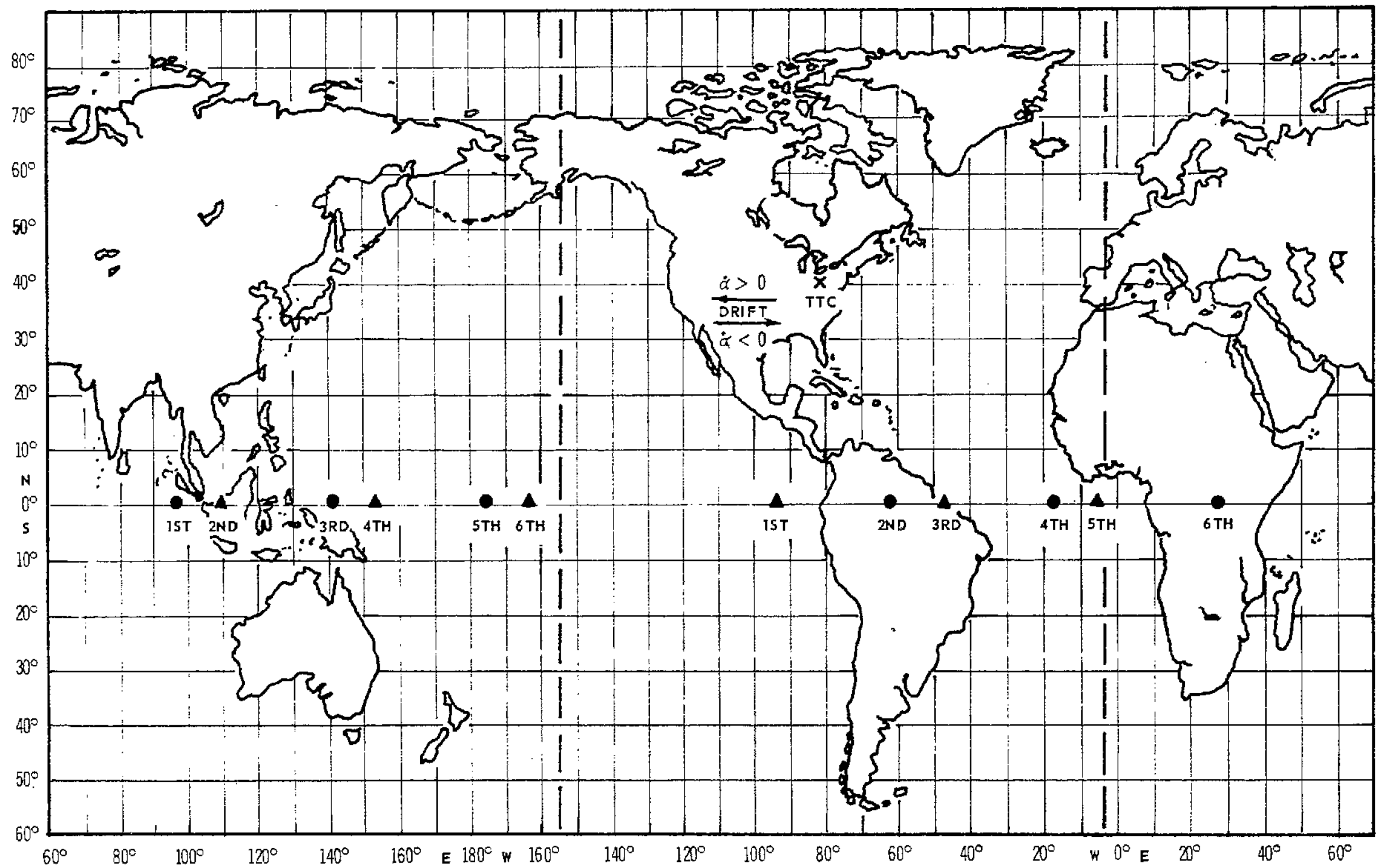


Figure 2-5 Successive apogees of a transfer ellipse resulting from perigee burn at first ▲ northerly ● southerly equatorial crossing

- Enter the transfer orbit at the first southerly equatorial crossing.

- Remain in the transfer orbit until arrival at the second apogee (in the vicinity of 60°W). Or alternately, remain in the transfer orbit until the fourth apogee is reached (in the vicinity of 20°W).

- Carry out a biased apogee burn at the second (or fourth) apogee on the transfer orbit which results in a drift orbit with westward drift.

- Upon completion of the apogee burn, the satellite will be tracked to determine its orbit. Subsequently, trim maneuvers will be carried out to remove any launch vehicle injection errors, etc.

- At a specified epoch during the drift phase, the final positioning maneuver will be performed which will position the satellite at the desired stationary longitude.

This mission sequence has the distinct advantage of a backup mode, i.e., if the apogee burn cannot be carried out at the second apogee, then an apogee burn at fourth apogee can be carried out without violating any mission constraints.

The apogee burn is characterized by an impulsive velocity change, ΔV, and firing angle, χ (defined as the acute angle between the apogee burn vector and the equatorial plane) which depend upon the inclination of the transfer orbit, i, and the drift rate, $\dot{\alpha}$. Figures 2-6 and 2-7 present values of ΔV and χ as functions of i and $\dot{\alpha}$. The values of ΔV and χ corresponding to values of $\dot{\alpha} = 0$ characterize the apogee burn required to enter an equatorial synchronous orbit. Values of ΔV and χ corresponding to values of $\dot{\alpha} > 0$ characterize the apogee burn required to enter a specified equatorial drift orbit (westward drift).

In particular, selection of an apogee burn which results in a drift rate $\dot{\alpha} = 5$ deg/day would enable final positioning at the stationary longitude (100°W) to be accomplished in about 8 days from the time of apogee burn at the second apogee, or in about 16 days from the time of apogee burn at the fourth apogee.

Figure 2-8 presents a graph showing the hydrazine required for final positioning at the desired stationary longitude. If the apogee burn is carried out at the second apogee, the longitudinal drift is

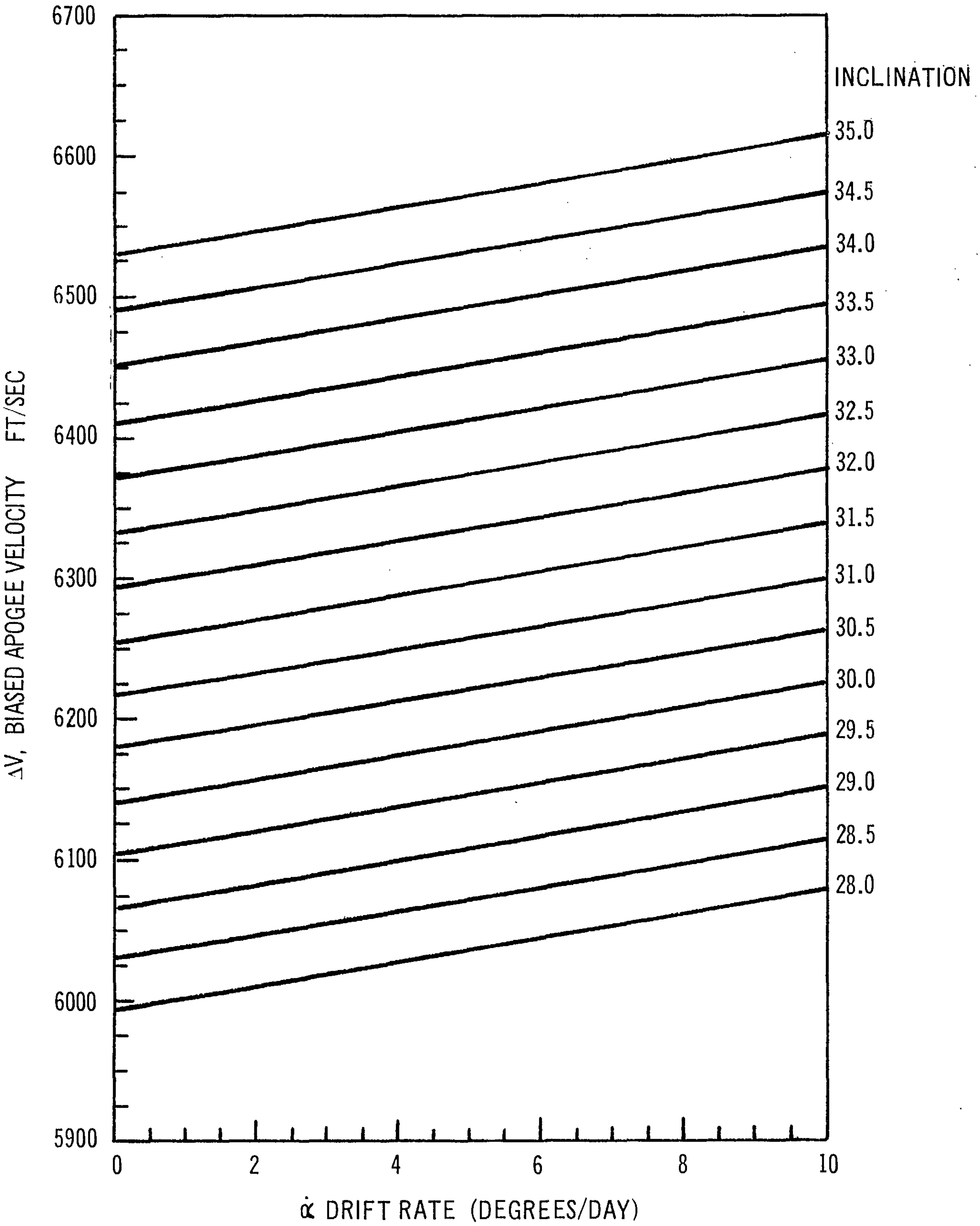


Figure 2.6 – Apogee burn ΔV versus drift rate

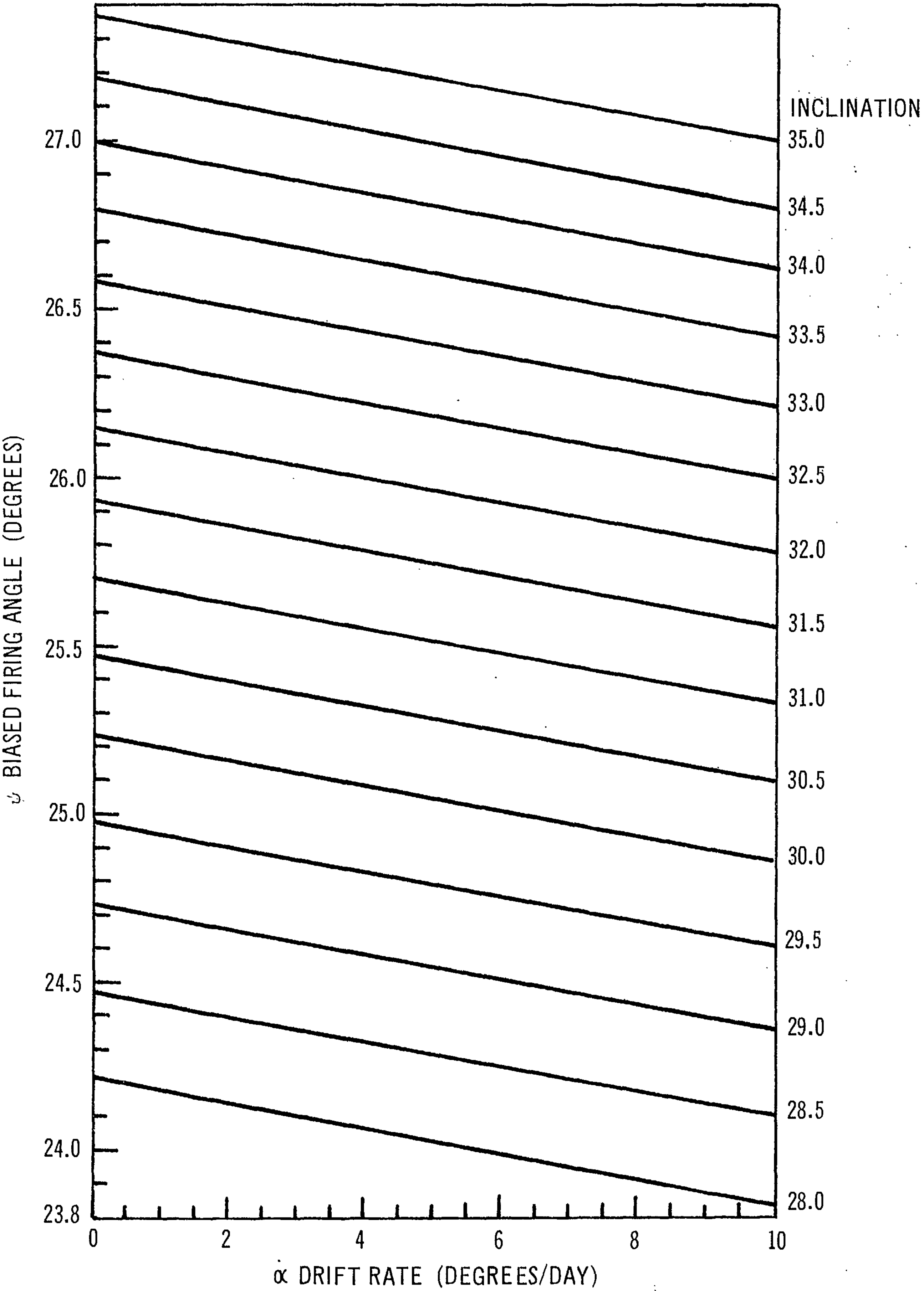


Figure 2.7 – Apogee burn firing angle versus drift rate

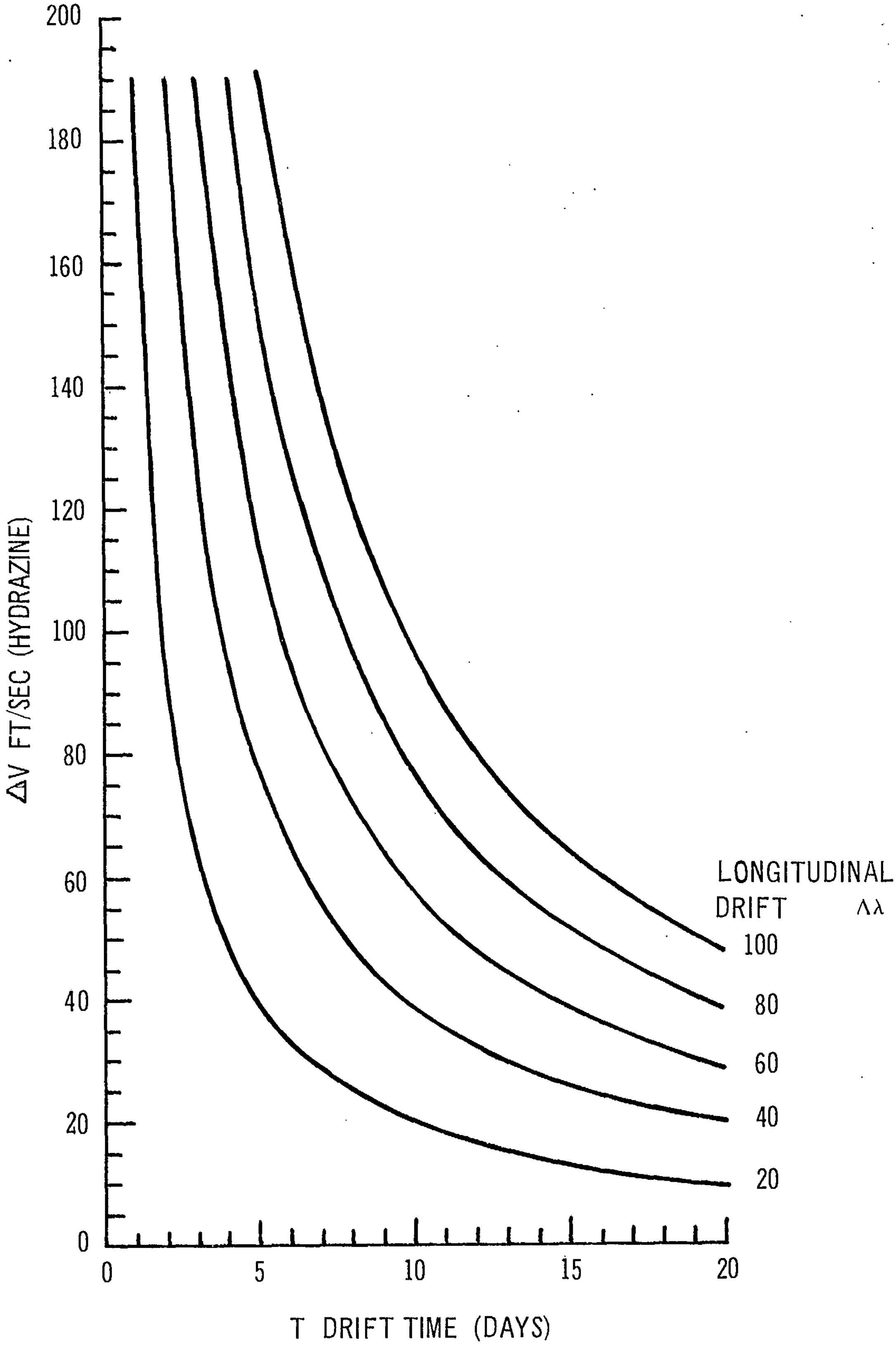


Figure 2.8 – Velocity required for final positioning maneuver

about 40° in 8 days ($\dot{\alpha}$ = 5.0). Similarly, if the apogee burn is carried out at the fourth apogee, the longitudinal drift is about 80° in 16 days ($\dot{\alpha}$ = 5.0). In both cases, the hydrazine required for the final positioning maneuvers is 48 ft/sec.

2.2.4 Station Keeping

Once the satellite has been positioned at its desired stationary longitude, in the vicinity of 100°W, it will tend to drift in longitude; primarily due to the ellipticity of the equator. In addition, it will drift in latitude; primarily due to the combined effect of luni-solar perturbations. The proposed station keeping capabilities for the satellite are $\pm$ 0.10° in longitude and $\pm$ 0.10° in latitude for a five year duration. This is an improvement over the original work statement requirements which is achieved at no penalty to the spacecraft and allows a higher effective ERP to be obtained for fixed pointing ground stations.

The longitudinal station keeping will be carried out by use of radial (hydrazine) thrustors. The satellite will be allowed to drift in longitude until the first extreme of the tolerance band is reached, at which time an impulse will be applied of magnitude ΔV_1 (see Figure - 2-9) which reverses the drift direction. The time required for drift from the stationary position to the first extreme of the tolerance band is T_1 (see Figure 2-10). The application of ΔV_1 enables the satellite to drift to the opposite extreme of the tolerance band and then return to the point of application of ΔV_1. The time required for drift through one complete cycle is T (see Figure 2-10). Once the satellite has reached the first encounter of the tolerance band limit, an impulse of magnitude ΔV is applied (see Figure 2-9) which enables it to drift through one complete cycle in T days. Subsequent applications of ΔV with periodicity T will maintain the limit cycle within the deadband of $\pm$ 0.10°. The total impulsive ΔV_T, required for five years of longitudinal station keeping, is shown in Figure 2-11.

The latitude of the satellite will vary over its operational lifetime due to the inclination build-up caused by the combined effect of lunar-solar perturbations and the oblateness of the earth. The rate at which inclination builds up is dependent upon the position of the moon's line of nodes and, hence, varies from year to year. A graph showing rate of change of inclination as a function of time is presented in Figure 2-12. The graph shows that, for the years 1971$\frac{1}{2}$ to 1976$\frac{1}{2}$, the rate of change of inclination of an initially equatorial orbit decreases almost uniformly from about $\frac{di}{dt}$ = 0.91 deg/yr to $\frac{di}{dt}$ = 0.79 deg/yr. The total inclination build up for the five year lifetime is the area under the $\frac{di}{dt}$ curve

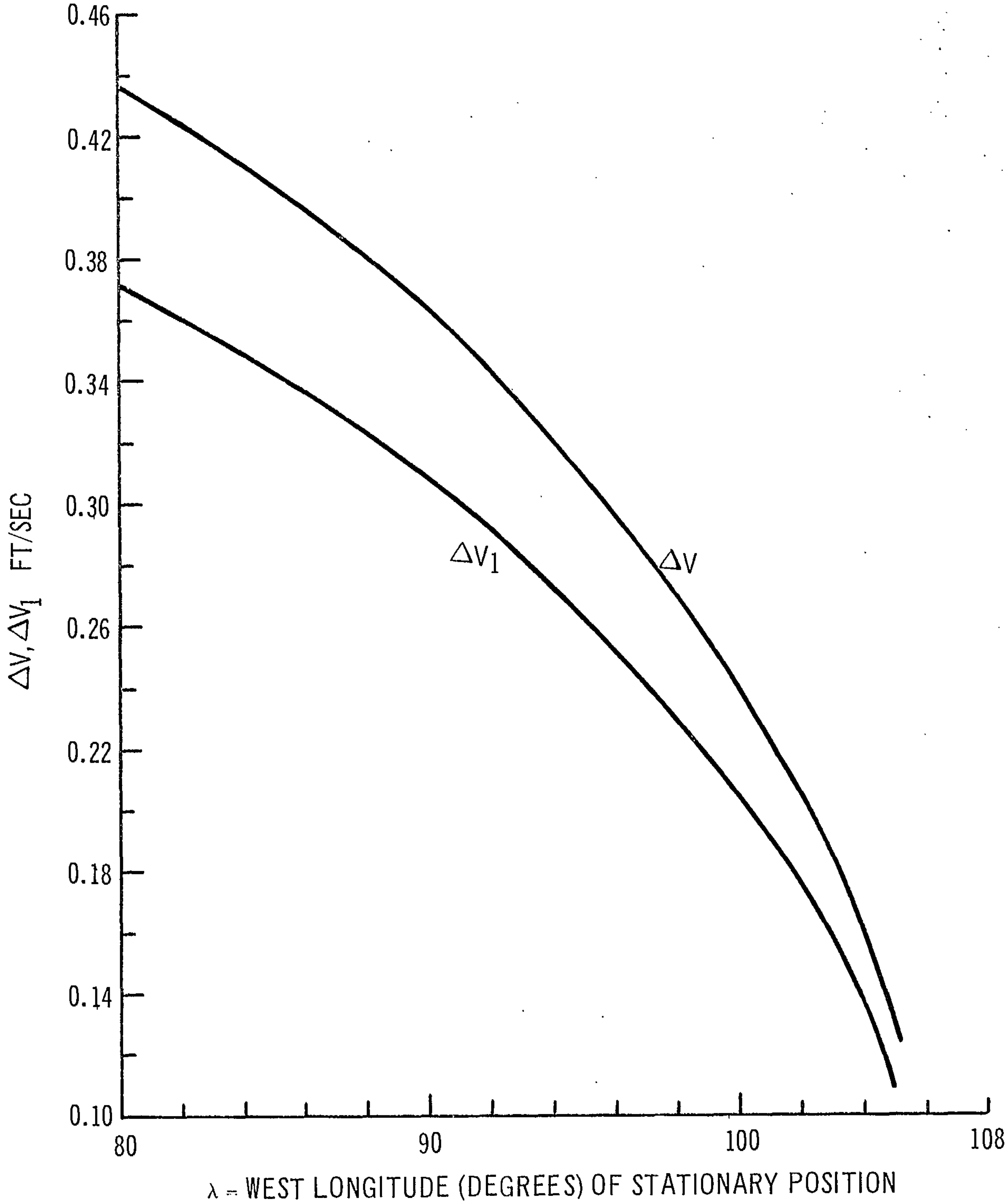


Figure 2.9 – ΔV_1 = Velocity impulse required for first longitudinal station keeping maneuver

ΔV = Velocity impulse required for subsequent longitudinal station keeping maneuvers, deadband = ± 0.10 degrees

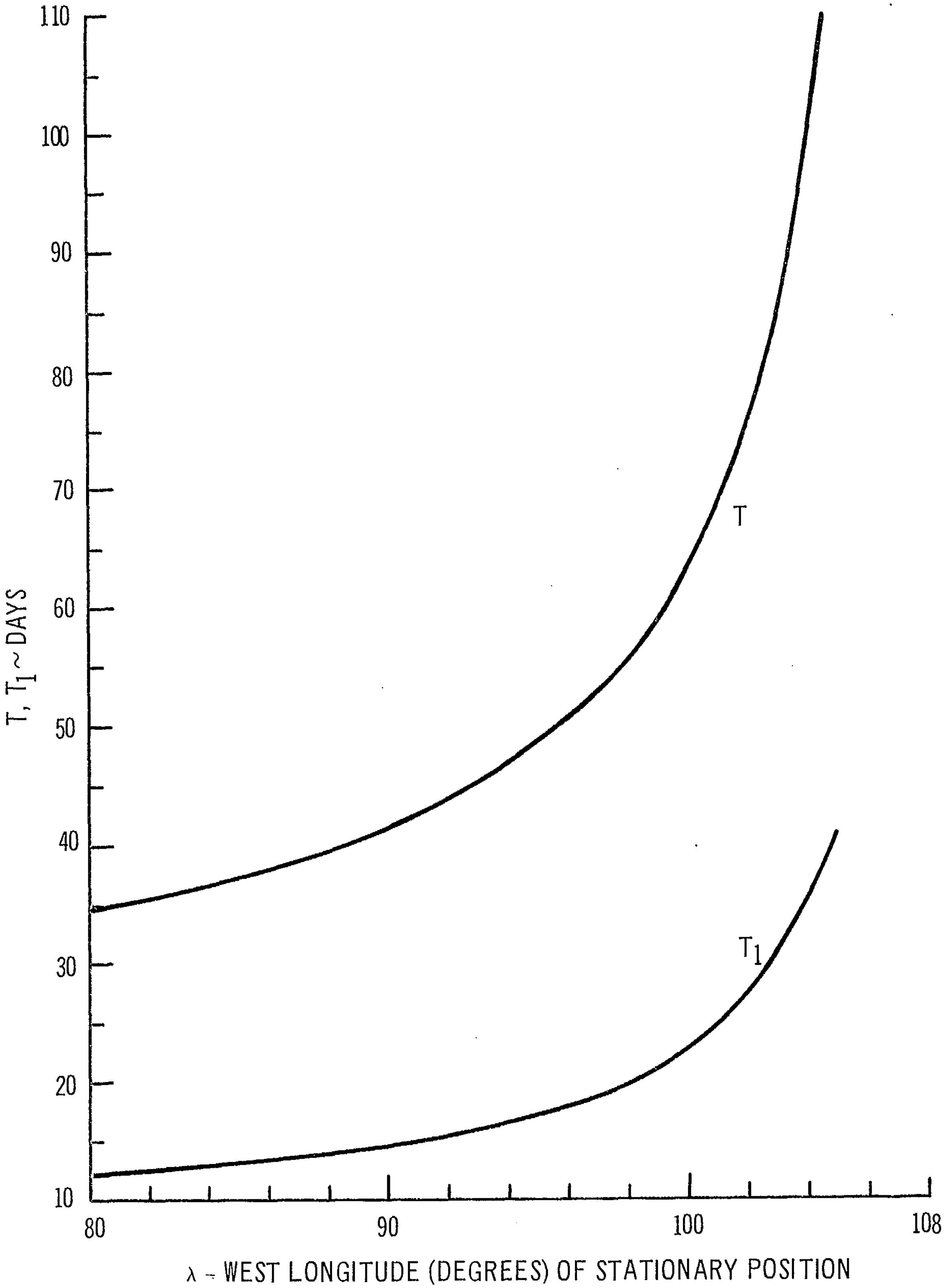


Figure 2.10 – T_1 = Drift time from stationary position to first longitudinal station keeping maneuever

T = Time between subsequent longitudinal station keeping maneuvers

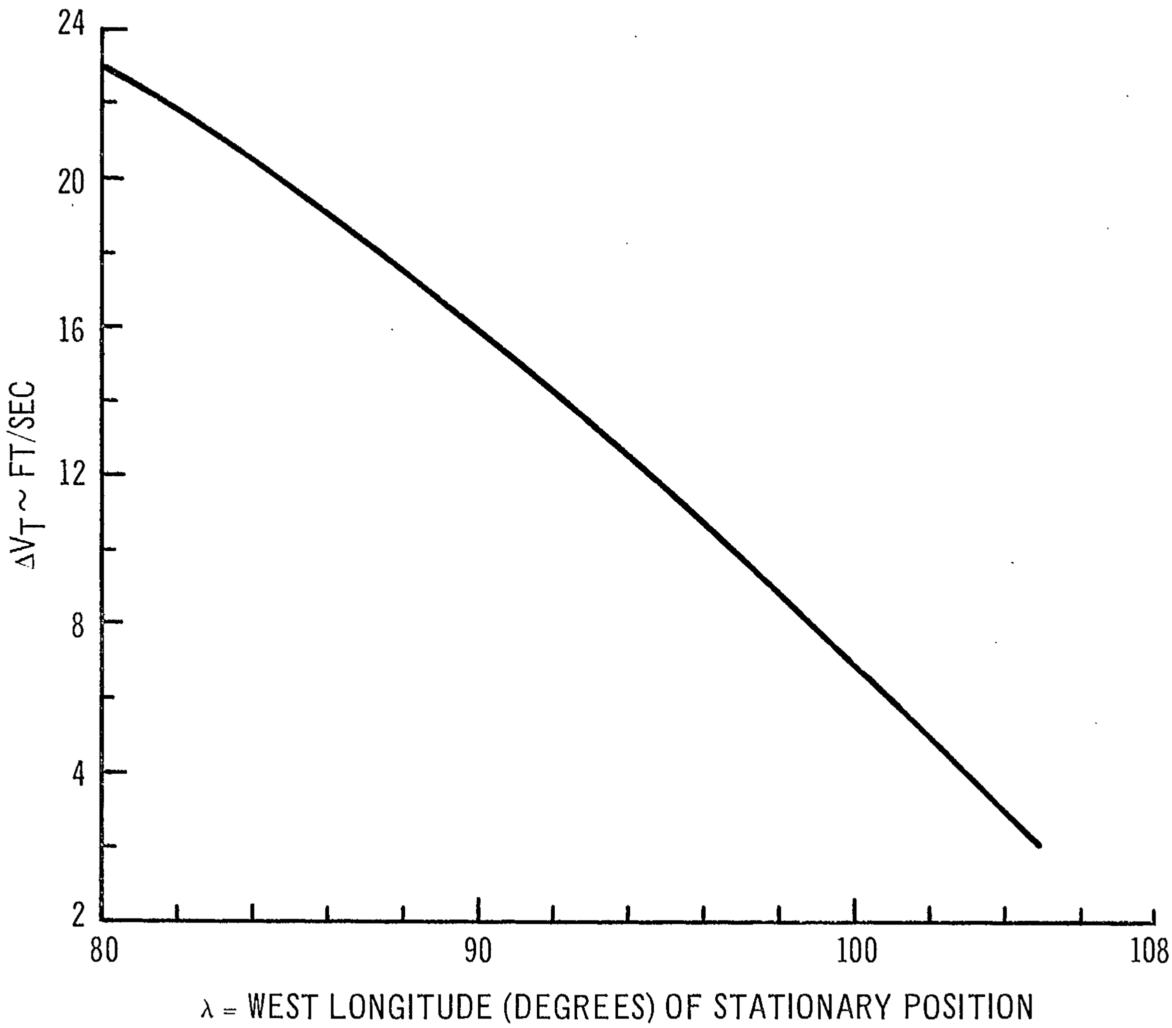


Figure 2.11 – Total ΔV required for five years of longitudinal station keeping, with deadband of ± 0.10 degrees

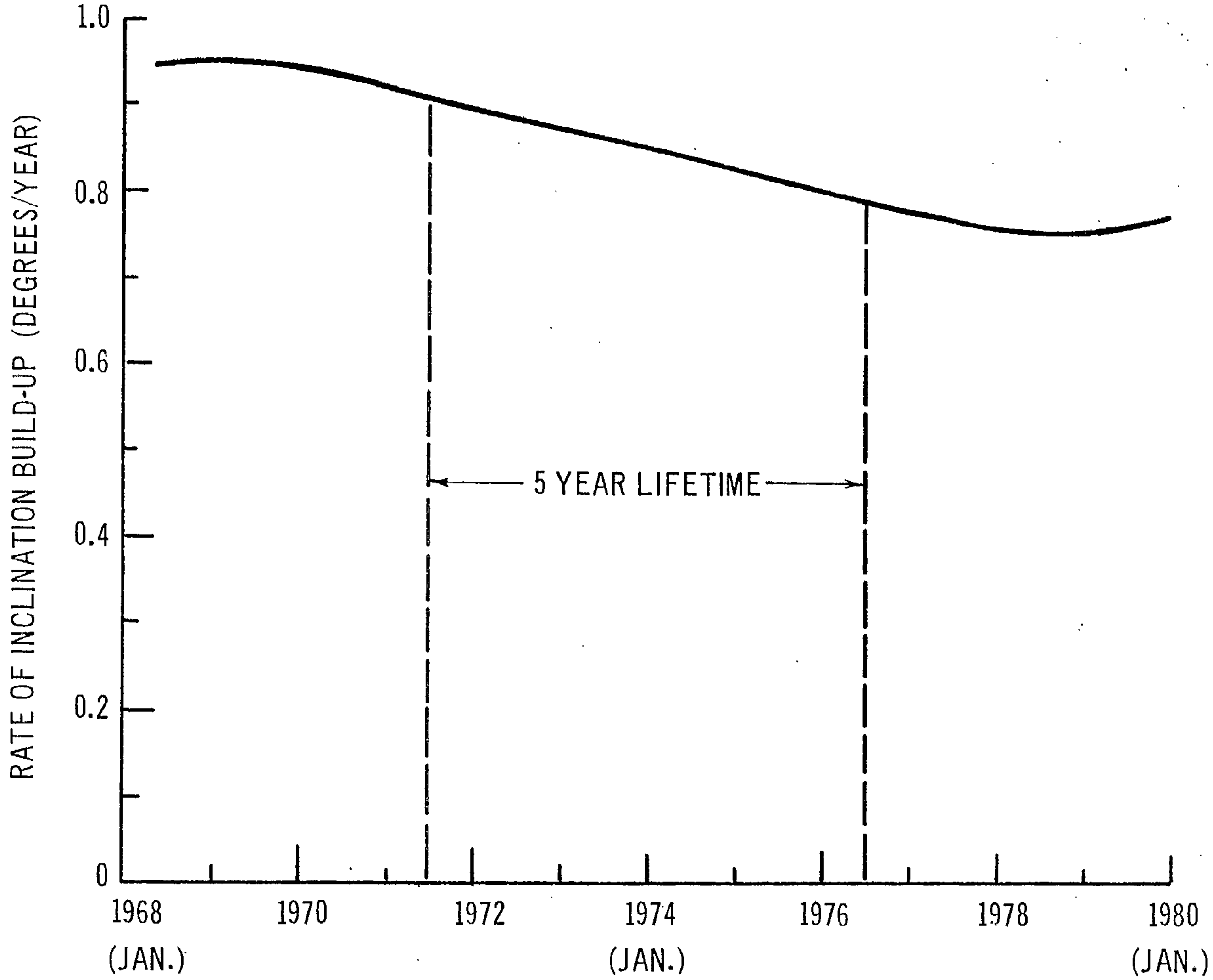


Figure 2.12 – Rate of change of inclination due to the combined effect of luni-solar and asphericity perturbations

between the dashed lines. If it is noted that a velocity increment of 176 ft/sec is required to remove one degree of inclination, it follows from Figure 2-12 that the total ΔV budget required to remove the five year inclination build-up ($1971\frac{1}{2}$ to $1976\frac{1}{2}$) is $\Delta V = 747$ ft/sec. The periodicity with which latitudinal station keeping maneuvers must be carried out depends upon the latitude tolerance of 0.10°. The time required for an inclination build-up of 0.10° varies from about 40 days in 1971 to about 46 days in 1976. Latitudinal station keeping will be carried out by use of the axial thrustors.

2.2.5 Attitude Reorientation Maneuvers

The spin axis of the satellite will require periodic reorientation to its normal position, which is perpendicular to the equatorial plan with the antenna end of the spacecraft pointing north. The first reorientation arises after apogee motor firing has occurred, and the desired drift orbit has been achieved. This spin axis reorientation will require an angular change capability of 114°. In addition, the spacecraft spin axis will vary, due to the combined effect of:

- Torque due to solar pressure: this torque is proportional to the distance between the center of mass and the center of solar pressure.

- Torque due to magnetic moment: great care is exercised to asure a maximum of magnetic cleanliness in the spacecraft.

- Torque due to gravity: since the gravity torque on the satellite is proportional to the angle between the spin axis and orbit normal and since the spin axis is kept within 0.50° from the orbit normal, the influence of this torque on the attitude of the spacecraft is negligible.

- Attitude error due to misalignment of the apogee motor, dynamic unbalance of the spacecraft, wobble and nutation.

The ΔV corresponding to a change of β radians in the attitude of the spacecraft spin axis is given by

$$\Delta V = \frac{I\omega}{mr_o}\beta$$

where I represents the moment of inertia of the spinning section of the spacecraft about its spin axis; ω is the spin rate; m is the mass of spacecraft; and r_o is radial distance from the spin axis to the axial thrustors. For this satellite design, the above equation takes the form

$$\Delta V = \begin{cases} 0.112^{\circ} & \text{at booster separation} \\ 0.178^{\circ} & \text{after apogee motor burn} \end{cases}$$

The initial reorientation maneuver of $\beta^{o} = 114^{o}$ will require ΔV = 20 ft/sec. Subsequent attitude reorientation maneuvers required to correct spin axis variation due to solar pressure, magnetic moments, gravity torques, misalignment of the apogee motor and dynamic unbalance of the spacecraft are estimated to require, at most, ΔV = 20 ft/sec. Hence, the total attitude control budget is estimated to be, at most, 40 ft/sec for the five year operational lifespan.

2.2.6 Total Hydrazine Requirement

In summary, the preliminary design indicates that the total hydrazine required to maintain the satellite within the specified tolerances for a five year operational lifespan is as follows:

ΔV required for:

. removal of 99% probable space vehicle injection errors	181 ft/sec Atlas-Burner II
	300 ft/sec, Thor-Delta *
. final positioning maneuver	48 ft/sec
. latitudinal station keeping	748 ft/sec
. longitudinal station keeping	8 ft/sec
. attitude control of spacecraft spin axis	40 ft/sec
TOTALS	
Atlas-Burner II	1025 ft/sec
*Thor-Delta	1144 ft/sec

2.3 TRACKING ANALYSIS STUDY

A study has been conducted to determine the accuracy with which the satellite can be tracked in a circular synchronous orbit. Two distinct tracking modes have been considered: angles only tracking and angle range tracking. Because the latitude determinations are about five times better than those for longitude, only the latter are discussed here.

The satellite was assumed to be placed at 100° W longitude and tracking stations were assumed to be placed as indicated in Table I.

Table 1

Station	Longitude	Latitude
P_1	135° W	43° N
P_2	152° W	23° N
P_3	158° W	15° N
Q_1	40° W	37.5° N
Q_2	65° W	43° N
Q_3	77° W	23° N

The stations labeled P are located west of the subsatellite point and those labeled Q are east. These stations have been used in pairs in various tracking combinations.

The error model shown in Table 2 is based upon information contained in Reference 1.

Table 2

Tracking Study Error Model

	Noise (1σ)	Bias (1σ)
Azimuth, Elevation	0.005°	0.02°
Range	40 ft.	60 ft.
Station Location		
Latitude, Longitude	---	0.28×10^{3} deg
Height	---	131 ft.
(Earth's gravitiational constant)	---	8.2×10^{-7} $\left(\frac{\sigma_{\mu}}{\mu}\right)$
C_{22}	---	2.15×10^{-9}
S_{22}	---	1.93×10^{-8}

The results of the tracking study are shown in Table 3. For comparison, both angle only measurements and angle range measurements were made at each ground station. Tracking periods of 2 days and 10 days at the rate of 2 points per hour for each measurement are shown. The entries in Table 3 are the 1σ error in the drift rate determination expressed in degrees per day, σ_D, and the 1σ error in the longitude determination expressed in degrees, σ_λ .

Table 3

Tracking Accuracy

Stations	Measurements	2 Days of Tracking		10 Days of Tracking	
		σ_D	σ_λ	σ_D	σ_λ
P_1, Q_1	A, E	.00325	.0138	.000651	.0133
P_2, Q_2	A, E	.00325	.0141	.000652	.0137
P_3, Q_3	A, E	.00275	.0121	.000650	.0118
P_1, Q_1	A, E, R	.00056	.00036	.00056	.00033
P_2, Q_2	A, E, R	.00056	.00033	.00056	.00030
P_3, Q_3	A, E, R	.00056	.00035	.00057	.00032

It is seen that the accuracy is nearly insensitive to the placement of the ground stations. These results may be interpreted in the following way: the quoted value is the standard deviation of the normally distributed parameter being estimated. For example, in the case of stations P_1 and Q_1, tracking for 10 days with angle measurements, the probability that the satellite longitude estimate differs from the true longitude by more than 0.0266° is 0.045. The probability of being more than 0.04° in error is 0.003.

It was concluded that angle only tracking was acceptable for this mission where long tracking periods were permissable. However, in discussions with D of I personnel, it was decided to incorporate range and angle tracking on the basis that:

- Only one station need track, and that for only a brief period.
- The software analysis was simpler
- The accuracy was greater
- The spacecraft penalty was minimal.

Accordingly provision has been made to pass ranging tones through the command receiver to the telemetry link.

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1. "Evaluation of Antenna Pointing Accuracy", D.C. Buchanan, RCA Internal Correspondence, 10 June 1968.

1-3
SYSTEM TRADEOFFS

3. TRADE-OFFS

3.1 CONSTRAINTS

The following list gives the various constraints, assumptions and guidelines that were used in the trade-off study.

	DSV3L[2]	Atlas Burner II
Payload weight in transfer orbit	965	1005
Shroud	existing or new with same diameter	new, based on existing
Solar array	Based on Intelsat III panels	Based on Intelsat III or new
Minimum No. RF Channels daylight	4	4
Minimum No. channels eclipse	2	2
Minimum EIRP at edge of country	34 dbw	34 dbw
Pointing error	$\pm 0.5^\circ$	$\pm 0.5^\circ$
Polarization	Down Link Linear E-W	Down Link Linear E-W

3.1.1 Spacecraft Trade-offs

The most important trade-off considered at the beginning of the study is the budgeting of the available payload weight to maximize communications capacity. As a starting point a transfer orbit payload weight of 950 pounds was used (as given in the work statement). As more up to date vehicle and subsystem numbers were generated the power and weight budgets were revised to maintain the largest possible fraction of payload weight dedicated to the basic communications function, while maintaining a reasonable margin for weight growth. Before the complete antenna beam optimization had been performed as described in Section 1.7 it was possible to make trial weight and power budgets on the tentative specifications of the work statement and the weights and powers of subsystems and components from the Intelsat III spacecraft. Workable if somewhat conservative assumptions of antenna gain and circuit losses were made to determine the TWT power requirements per RF channel, and DC power calculated using current TWT and power supply efficiencies.

To give flexibility in the weight evaluation of several candidate spacecraft configurations, TWT RF output per channel was used as a convenient parameter. This is consistent with the variations that may occur in the assumed antenna gain and circuit losses, and with the possible lowering of minimum EIRP depending on earth station assumptions.

3.1.2 Configurations Considered

1) Existing Delta Shroud

Within the existing Delta shroud the largest antenna of suitable beam size that will fit within the tapered portion has been described in Reference 1. The largest solar array can be taken somewhat optimistically as 61 inches long with a 55 inch diameter. The antenna gain at beam edge and the derived power level per channel indicates a maximum of four daylight channels. This configuration is space limited and some of the weight margin may be used to provide full eclipse operation and greater TWT redundancy.

2) Modified Delta Shroud

For this option no shroud restriction is placed on solar array height or antenna height, the array diameter is assumed fixed however at 55 inches. The largest solar array is used in conjunction with the highest gain antenna that covers Canada subject only to payload weight limit.

3) Atlas Burner II Shroud

For this option no shroud restriction was assumed and it is probable that the existing Intelsat III solar array diameter would be maintained and the length of array and antenna height subject to weight limits only.

Ref. 1 A Canadian Satellite to serve Canada's Domestic Communications Requirements RCA Space Systems, September 1967

3.1.3 Weight and Power

Figure 3-1 is a typical plot of array power as a function of TWT RF power output per channel for both a 6 channel and a 4 channel configuration.

Figure 3-2 is a plot of battery capacity in watt hours as a function of TWT RF power per channel with all other loads assumed constant and an eclipse of 72 minutes.

Combining the information from Figures 1 and 2 and using the best available estimates of solar array and battery specific weights we get a combined weight curve as shown in Figure 3-3 for various candidate configurations.

3.1.4 Beam Size and EIRP

Figure 3-4 is a graph showing combinations of NS and EW beam dimensions which give equal antenna gain according to the equation

$$Go = 45.85 - 10 \log \theta_{NS} \; \theta_{EW}$$

where Go = Beam centre gain

θ_{NS} = Beam width in N-S direction

θ_{EW} = Beam width in E-W direction

If we assume that the optimum beam will have a gain at the edge one half neper (4.34 db) below the beam center and further assume 2 db circuit loss between the TWT and antenna port, we can plot EIRP at beam edge with TWT output per channel as a running parameter. When the angular dimensions of the coverage area are shown as limits we can then see the minimum TWT output per channel required to produce a given EIRP at beam edge.

3.1.5 Eclipse Operation

To maximize the number of communications channels available during normal sunlit operation it is necessary to keep eclipse capacity to a minimum (2 ch.) thereby keeping battery weight to a minimum.

It is interesting to observe that considerable excess solar array capacity exists at equinox (eclipse season) because of perpendicular illumination, and is more than adequate to recharge batteries for

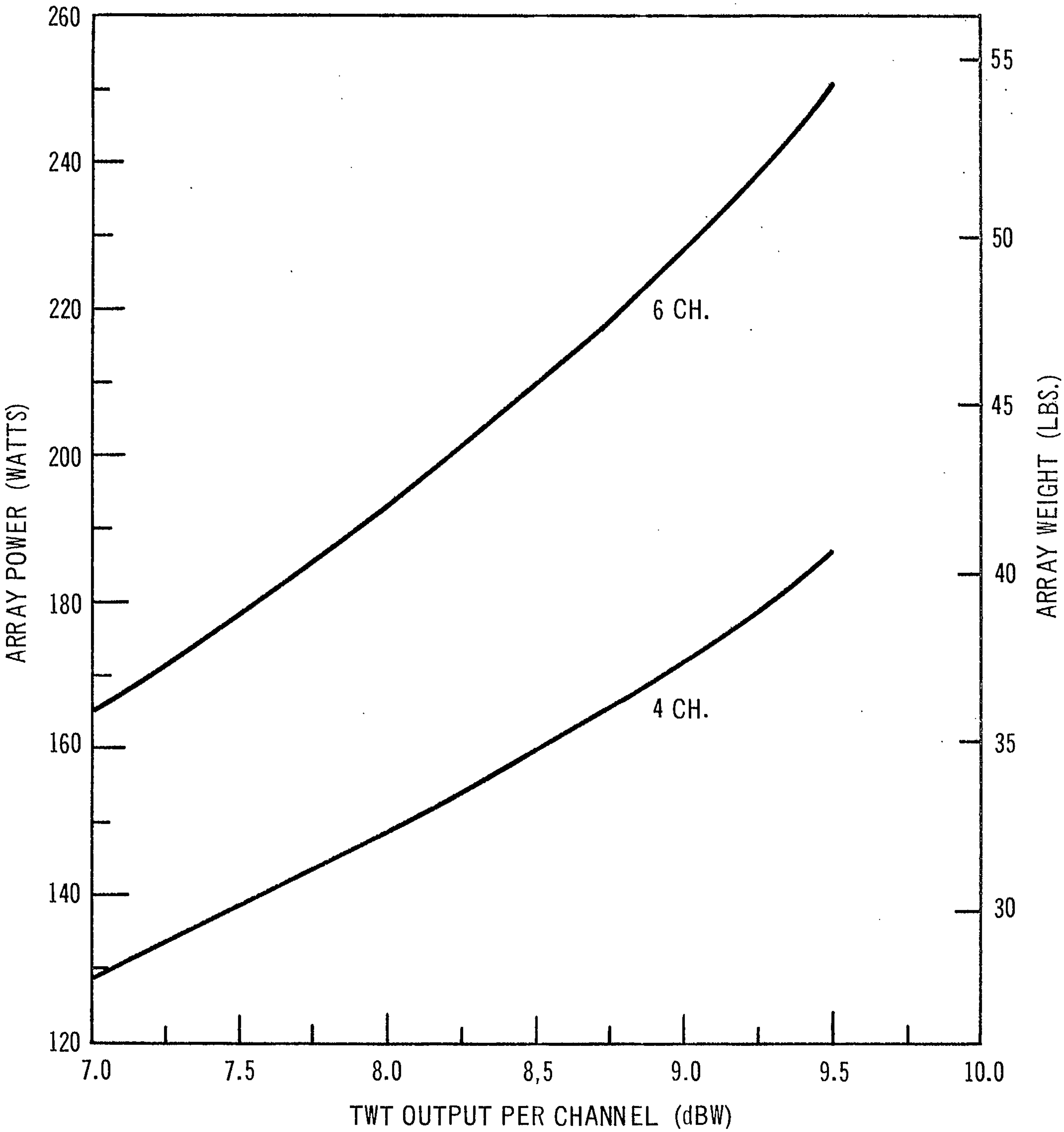


Figure 3.1 – Array Power versus TWT Output

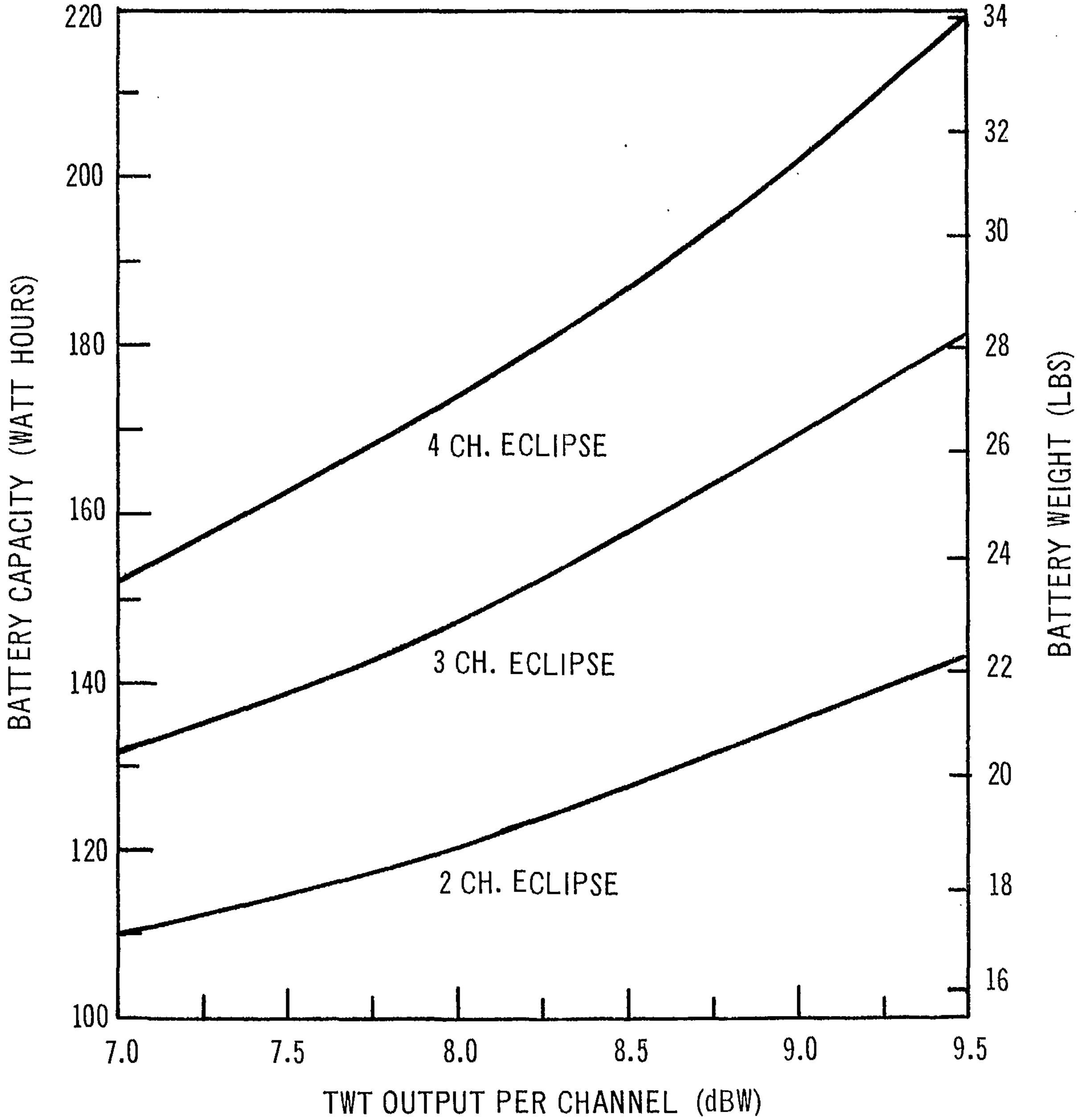


Figure 3.2 – Battery Capacity versus TWT Output
(Elipse time 72 minutes)

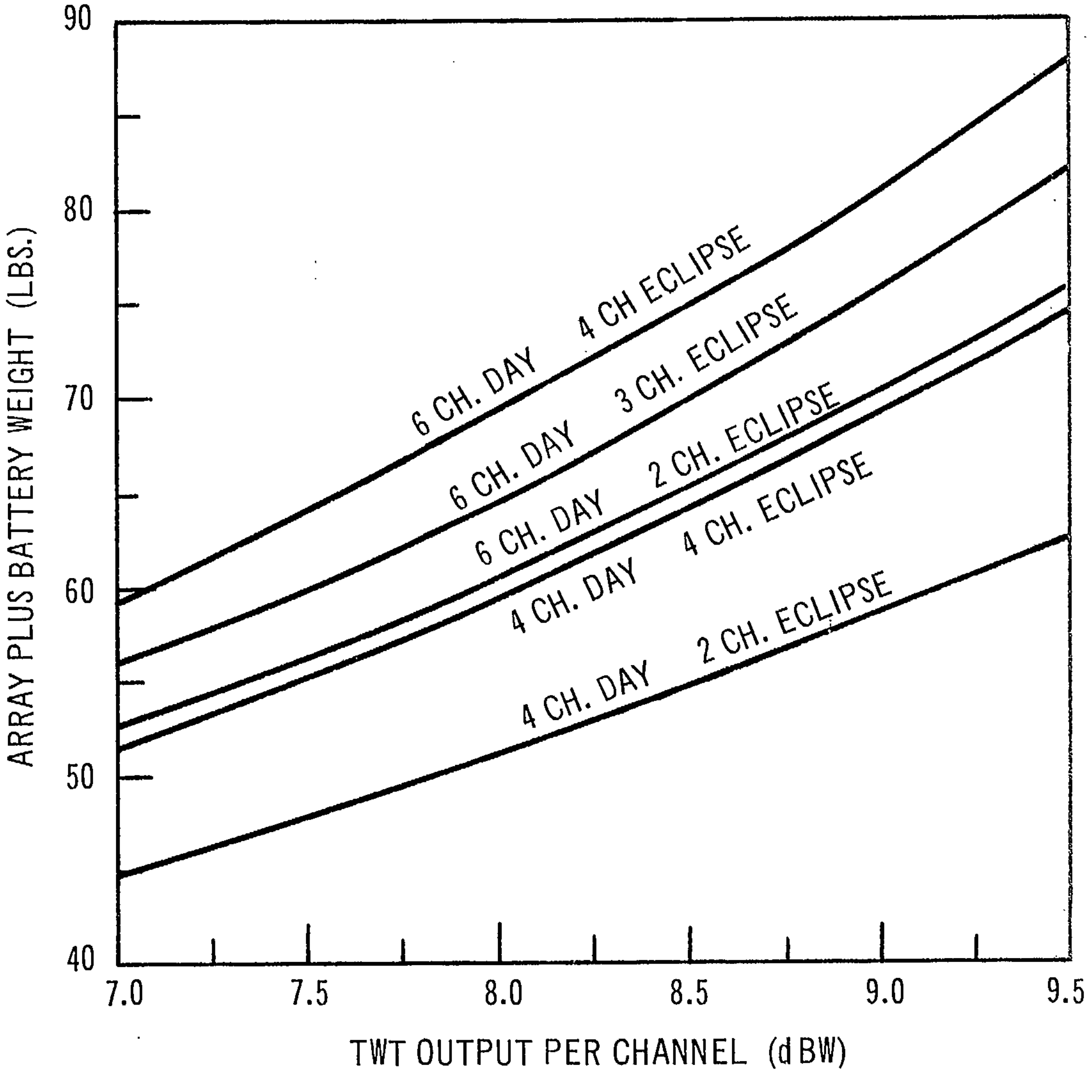


Figure 3.3 – Solar Array plus Battery Weight versus TWT Output per Channel

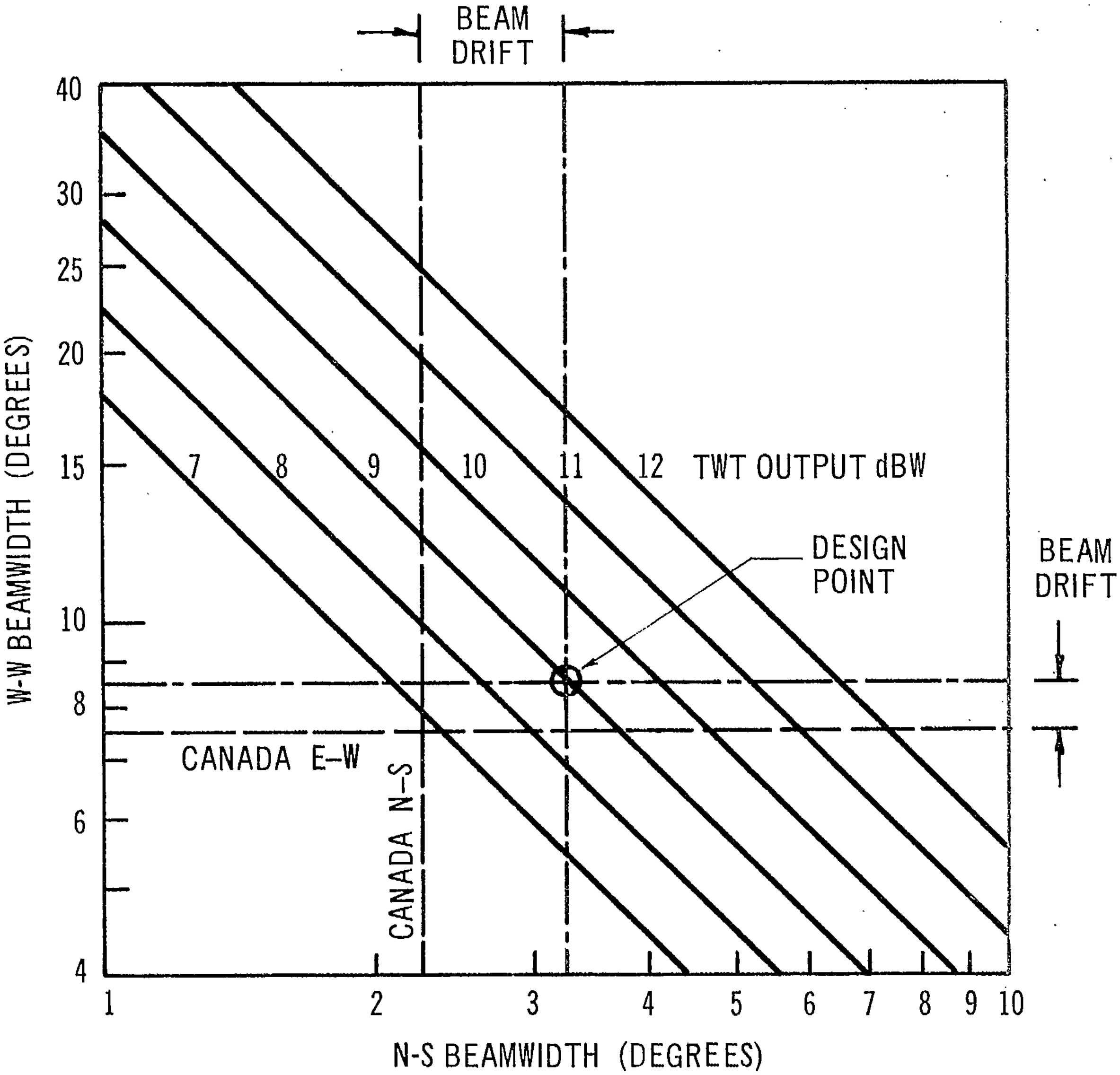


Figure 3.4 – Required TWT Power as a Function of Beam Size

Assumptions: Gain at beam edge = 41.51 – 10 log $\theta a \theta b$
EIRP at beam edge = 34 dBW

. the maximum number of eclipse channels. This can be explained by the long period available for recharge (20 hours). In practice the solar array size need only be increased by a small amount to account for a trickle charge to maintain battery charge and temperature during non eclipse times.

3.1.6 Station Keeping Considerations

No consideration is given to trading off fuel for longitude station keeping as this function, as well as spin axis maintenance, is essential. The largest portion of the secondary propulsion fuel is needed for correcting orbital inclination drift. This weight could be used alternatively for extra communications channels, increased ERP, operational lifetime, or for increased reliability through redundancy, if earth station designs could be economically adapted to the resulting satellite oscillations in latitude ($\pm 2.5^{\circ}$).

The drift of the orbital plane which is the cause of the problem is regular and predictable, so some consideration could be given to fuel savings by a combination of initial inclination biasing and a limited number of corrections. The pertinent expressions are:

$$V_N = V_T \cdot \frac{N}{N+1}$$

and

$$C_N = \frac{Ct}{N+1}$$

where V_N = velocity increment for N corrections

V_T = total velocity increment for complete removal of inclination

N = Number of corrections

C_N = Peak inclination error associated with N corrections

C_T = Peak initial inclination error (assumed one half of the total uncorrected drift over the satellite lifetime).

The technique is shown graphically in Figure 3-5. Figure 3-6 shows the total amount of secondary propulsion fuel required to stay within stated limits, with spacecraft lifetime as a parameter.

For the present earth station technology no economic solutions exist for the problem of limited steering capability to accommodate a satellite of uncontrolled or moderately limited inclination (say $\pm 2.5^{\circ}$). As can be seen from the figures, no significant fuel savings acrue when the inclination error is small and the number of corrections large. (Note 1) The ground station philosophy has been then to demand as accurate as possible satellite station keeping to permit use of relatively large (30') non steered antennas. No extra fuel penalty is paid for this accuracy and it becomes mainly a problem of obtaining accurate tracking information from the control stations. Without inclination correction there would be a small but significant penalty in the spacecraft antenna gain which depends on the peak orbital inclination because the beam would have to be increased in the NS sense by the amount of the peak to peak latitude excursion (5°). It can be mentioned parenthetically that much scope for ingenuity exists in the area of economical TV receiving stations for this satellite system.

3.1.7 Frequency and Polarization Considerations

From reasonably detailed considerations of spacecraft transponder filter design requirements, it is concluded that alternate channels should have opposite sense polarization. This is true whether all channels are in one satellite, as Intelsat IV, or half in one satellite and half in a second satellite occupying the same orbital slot as is proposed here. The same reasoning applies also for linearly or circularly polarized satellites. Where the adjacent channel protection is achieved by frequency filtering alone, a very difficult filter design must be assumed which becomes impractical at 4 GHz because of group delay and temperature problems. Even at an IF of several hundred MHz, filters are still troublesome. If polarization isolation at beam center can be used to give 12 to 15 dB of adjacent channel rejection, then filter problems are very much reduced and the prospects of an all RF design becomes more practical. This is considered advantageous because of the simpler configuration of the all RF transponder.

The transmission to ground would be with all channels of the same polarization. Adjacent satellite orbital slots would use the opposite sense of polarization taking advantage of any polarization discrimination that the ground station antenna sidelobes may provide on transmission or reception. For calculation of interference from

Note (1) Below 0-1° station keeping accuracy the fuel requirements would rise because short term fluctuations and measurement uncertainties would not average out and propulsion system efficiency decline because of the large number of small corrections called for.

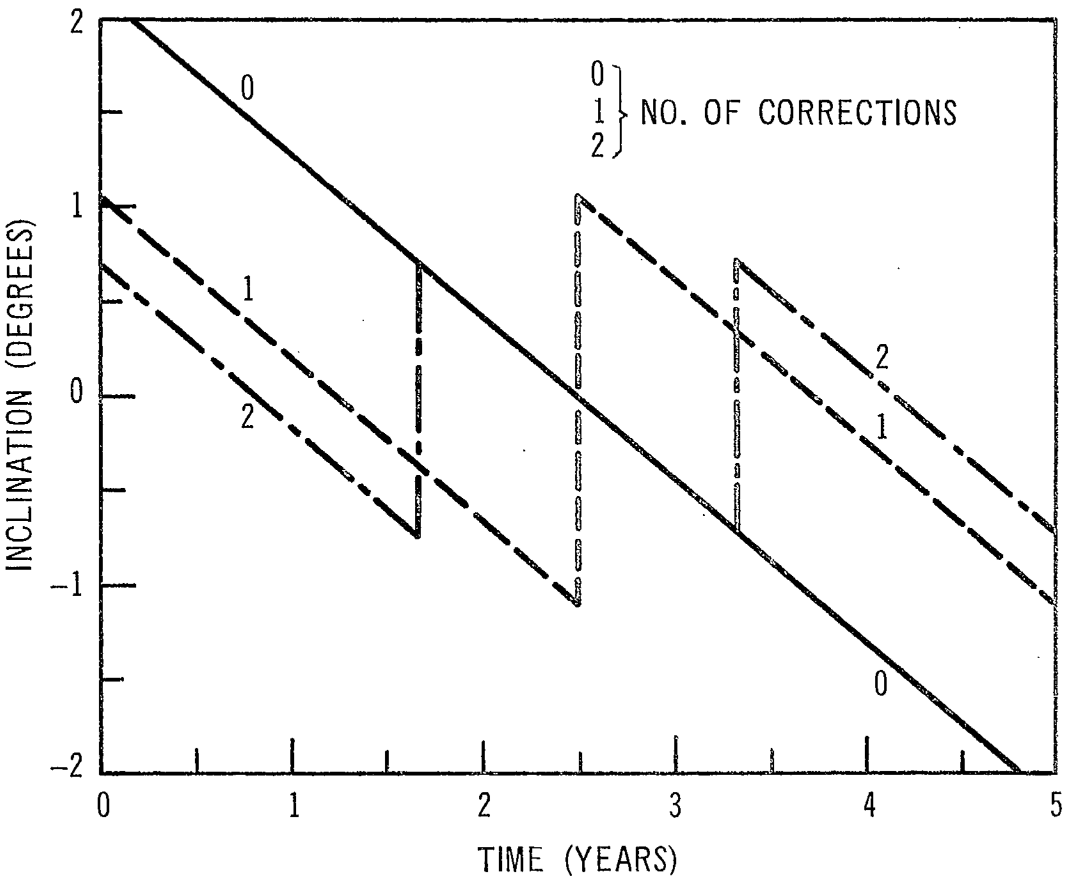


Figure 3.5 – Inclination History versus Number of Corrections

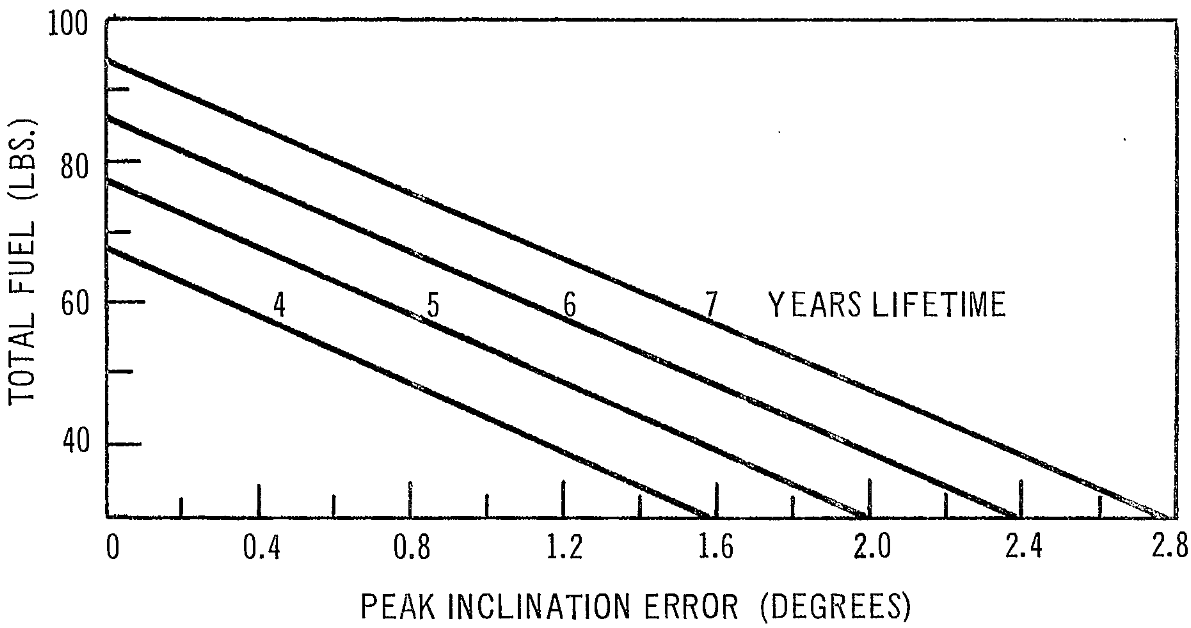


Figure 3.6 – Total Fuel Requirements versus Inclination Error and Lifetime

adjacent satellites into an earth station, which determines satellite spacing, it is assumed that all interfering satellites are illuminating the same area as the desired one. It is assumed that no advantage is gained by polarization discrimination in the ground station sidelobes but an isolation of about 15 dB would be obtained by offsetting adjacent satellite frequencies by one half a channel bandwidth.

Table 3.1.1 shows several possible combinations of frequency and polarization plans of which plan 3 is the preferred one. Here both up and down links of adjacent satellites would operate on interleaved frequency plans with opposite senses of polarization. This plan is optimum if no advantage is claimed for polarization discrimination. Other plans offer the same performance as #5 with all up links of orbit Position B of one polarization and orbit position A with up link channels cross polarized. The point to be made is that if polarization discrimination to adjacent satellites does not exist, then complete flexibility exists in a choice of polarization plans, and a plan is chosen which gives maximum benefit to satellite and ground station design without affecting system compatibility.

3.1.8 Ground Station Considerations

With all transmissions from one satellite location of the same polarization, the low noise front end of the ground receiver would be the same for any number of received channels. If reception from an oppositely polarized satellite is required it would probably be done by manually changing the connection between the low noise amplifier and the appropriate port of the feed assembly.

If it becomes necessary to transmit both polarizations from one antenna while receiving all channels, such as from a central station originating more than six programs, then it would be necessary to provide a transmit-receive frequency duplexer at one port of the orthogonal coupler. This would increase the system receiving temperature by a few degrees Kelvin.

3.1.9 Linear or Circular Polarization

Circular polarization has the advantage that a greater variety of antenna and feed systems are available as design choices. A rotating feed system is possible, that is one without the losses of a rotary joint, illuminating a despun mirror or combination of reflecting surfaces. Such a system is not possible with linear polarization as relative rotation between a linear feed and a mirror would produce

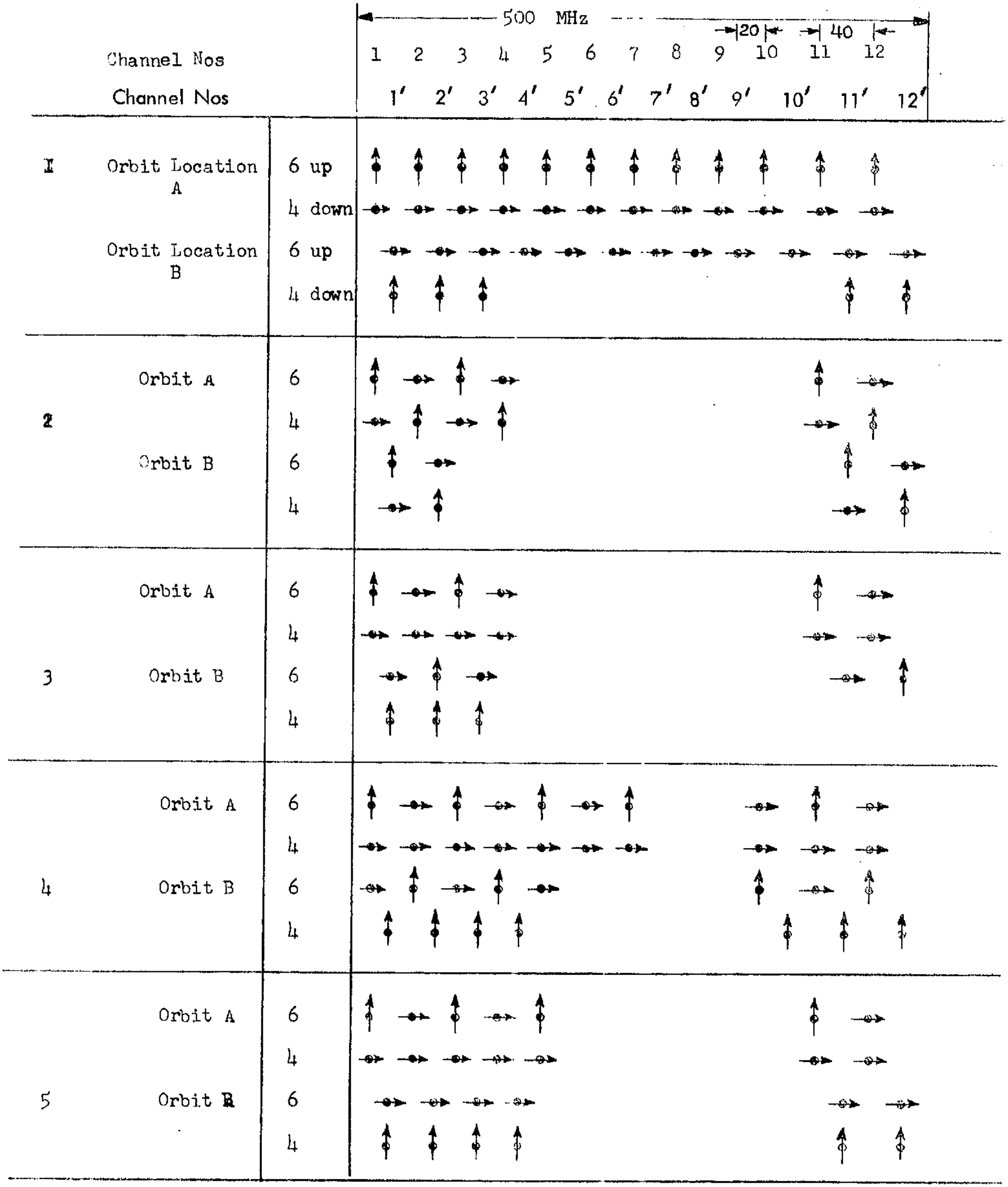


Table 3.1.1
ALTERNATIVE FREQUENCY AND POLARIZATION PLANS

rotating linear polarization. A rotating circular waveguide joint supporting orthogonal circular modes is attractive because of its low loss compared to a coaxial rotary joint. Figure 3-7 shows several methods of transferring signals between the spinning and despun portions of the satellite, for both odd and even channel satellites where it is assumed that;

(1) up and down links are linearly polarized

(2) all down links are vertically polarized for the initial Canadian system.

(3) there are six channels per satellite.

Figure 3-8 shows alternate feed methods for a circularly polarized satellite.

Construction of the satellite antenna feed system with circular polarization is more complex than linear because of the polarizer that is necessary to convert from linear to circular. At the same time the use of linear polarization prevents the possibility of using circular waveguide with a low loss waveguide rotary joint. This occurs because

(1) The orthogonal linear polarizations cannot pass through a rotating waveguide joint and

(2) It is impractical to use two polarizers to convert from linear to circular and back again after a waveguide rotary joint.

The polarization plans discussed here are naturally adapted to a six or three channel satellite configuration. The four channel minimum requirement configuration is not as well suited to the alternating polarization plan because one of the three satellites necessary to develop an orbital slot would have to carry both odd and even channels though not necessarily adjacent in frequency. See Table 3.1.2.

If linear polarization is used, method 2 Figure 3-7 using two coaxial rotary joints is preferred for the odd channel satellite. For even channel satellite method 6 using a single coax rotary joint is preferred. For circular polarization some question still exists about the possibility of achieving bends in a circular wave guide supporting the TE11 mode. If it is feasible, then the odd satellite would use method 7 and the even, method 9.

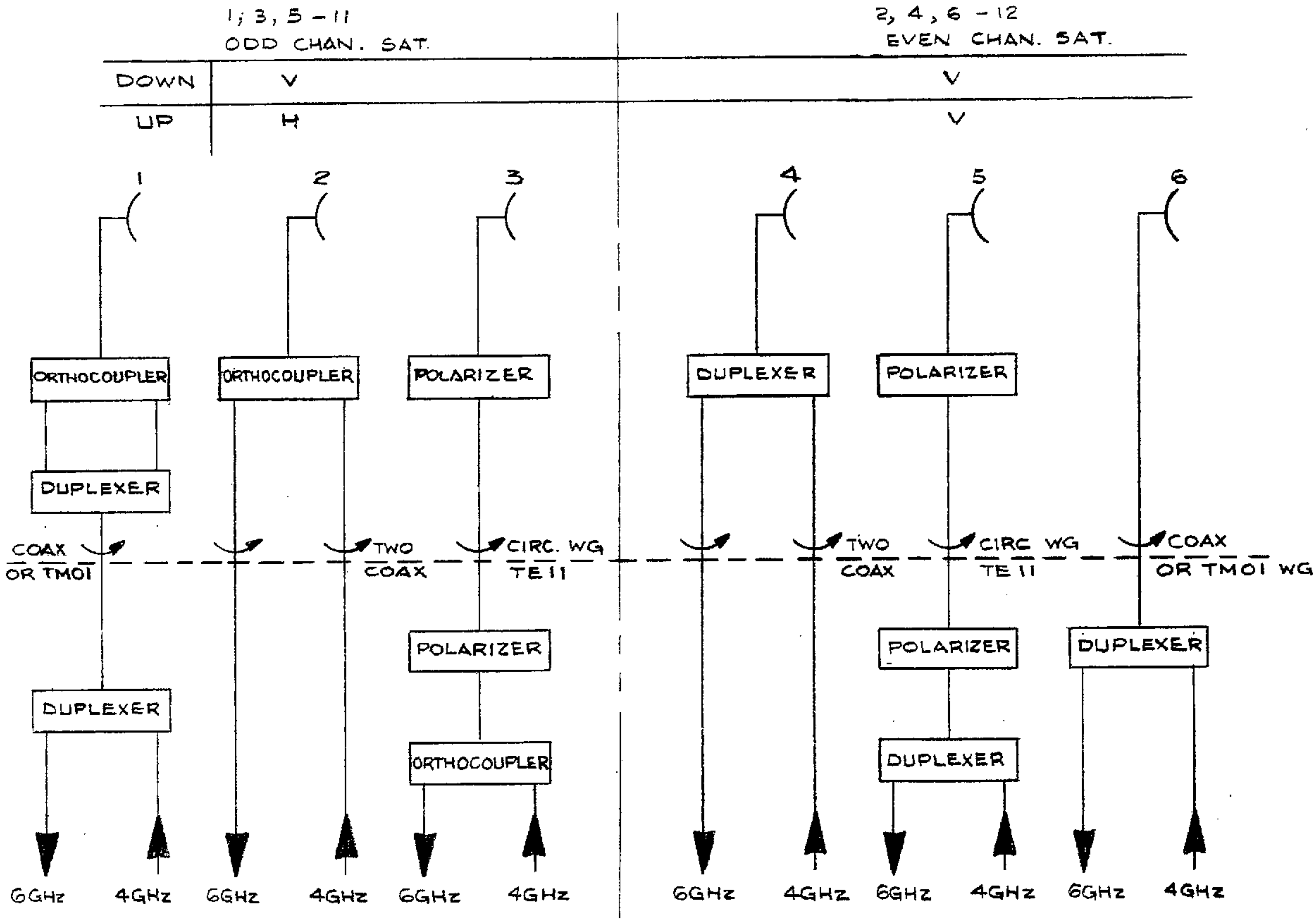


Figure 3–7 Alternative feed methods for a six-channel satellite with despun antenna (linear polarization)

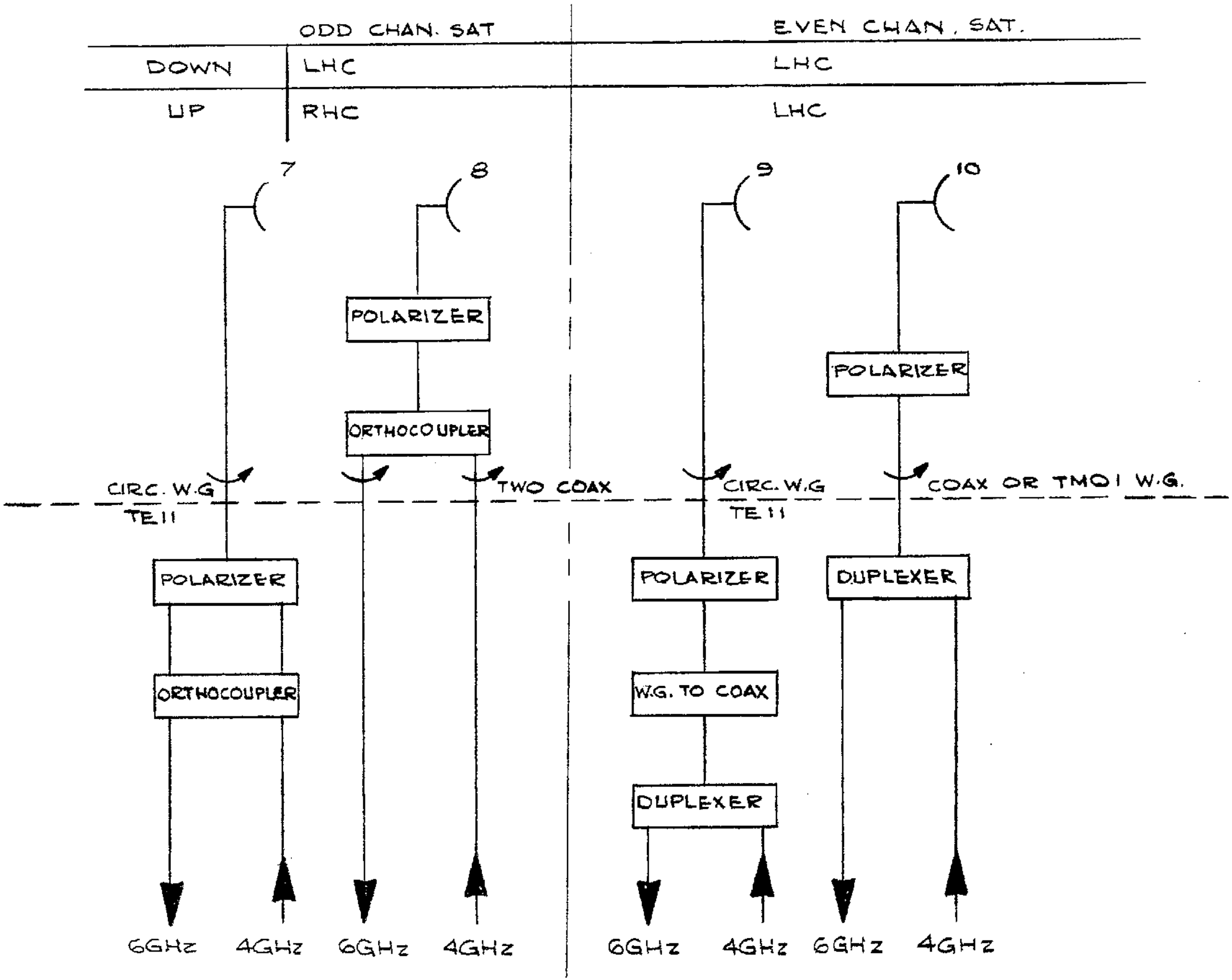


Figure 3–8 Alternative feed methods for a six-channel satellite with despun antenna (circular polarization)

CHANNEL NOS.	1	2	3	4	5	6	7	8	9	10	11	12	
SAT. 1.	↑						↑		↑		↑		
SAT. 2.		→		→		→						→	PLAN 1
SAT. 3			↑		↑			→		→			
SAT. 1	↑		↑		↑						↑		
SAT. 2						→		→		→		→	PLAN 2
SAT. 3		→		→			↑		↑				
SAT. 1					↑		↑		↑		↑		
SAT. 2		→		→						→		→	PLAN 3
SAT. 3	↑		↑			→		→					

TABLE 3.1.2

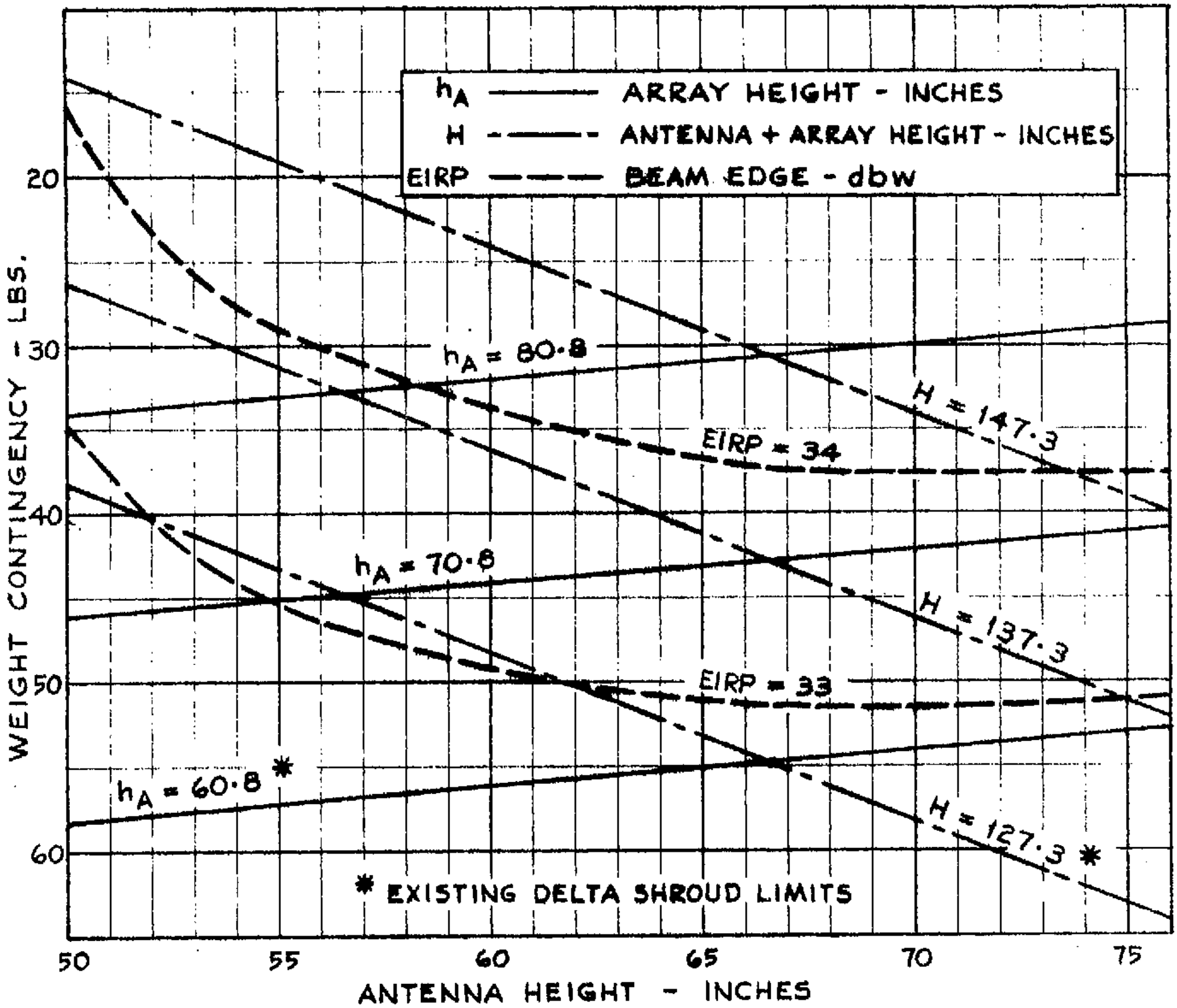
ALTERNATIVE FREQUENCY AND POLARIZATION PLANS FOR A FOUR CHANNEL SATELLITE

3.1.10 Weight Trade-offs for a Restricted Height Satellite

In conjunction with the study of an optimum antenna for the existing Delta shroud configuration the trade-off curves of Figure 3-9 and 3-10 were developed which can be applied to all configurations. In these graphs weight margin is shown as a function of antenna height and beam edge EIRP. Any point on the line labled EIRP represents a legitimate combination of solar array height and antenna height which will give equal EIRP towards the edge of the coverage area. It is somewhat surprising to note that the weight margin varies only slightly for a considerable range in antenna height below the optimum, when the beam optimization procedure of section 1.7 is followed. In contrast, if the antenna height only were varied keeping the antenna width constant, as was initially assumed, then the weight margin would decrease more rapidly with a reduction in antenna height. The curves shown for 6 day and 2 night, and 4 day and 4 night are not intended to be final but only to represent the method of optimization where overall height or array height restrictions exist. For the preferred configuration where no shroud height limitations exist it is still advisable to use the largest size antenna consistent with the area coverage requirement because of the extra EIRP at beam centre. It does indicate however that some flexibility exists in choosing antenna heights if structural and environmental problems prove unusually difficult for such a high antenna.

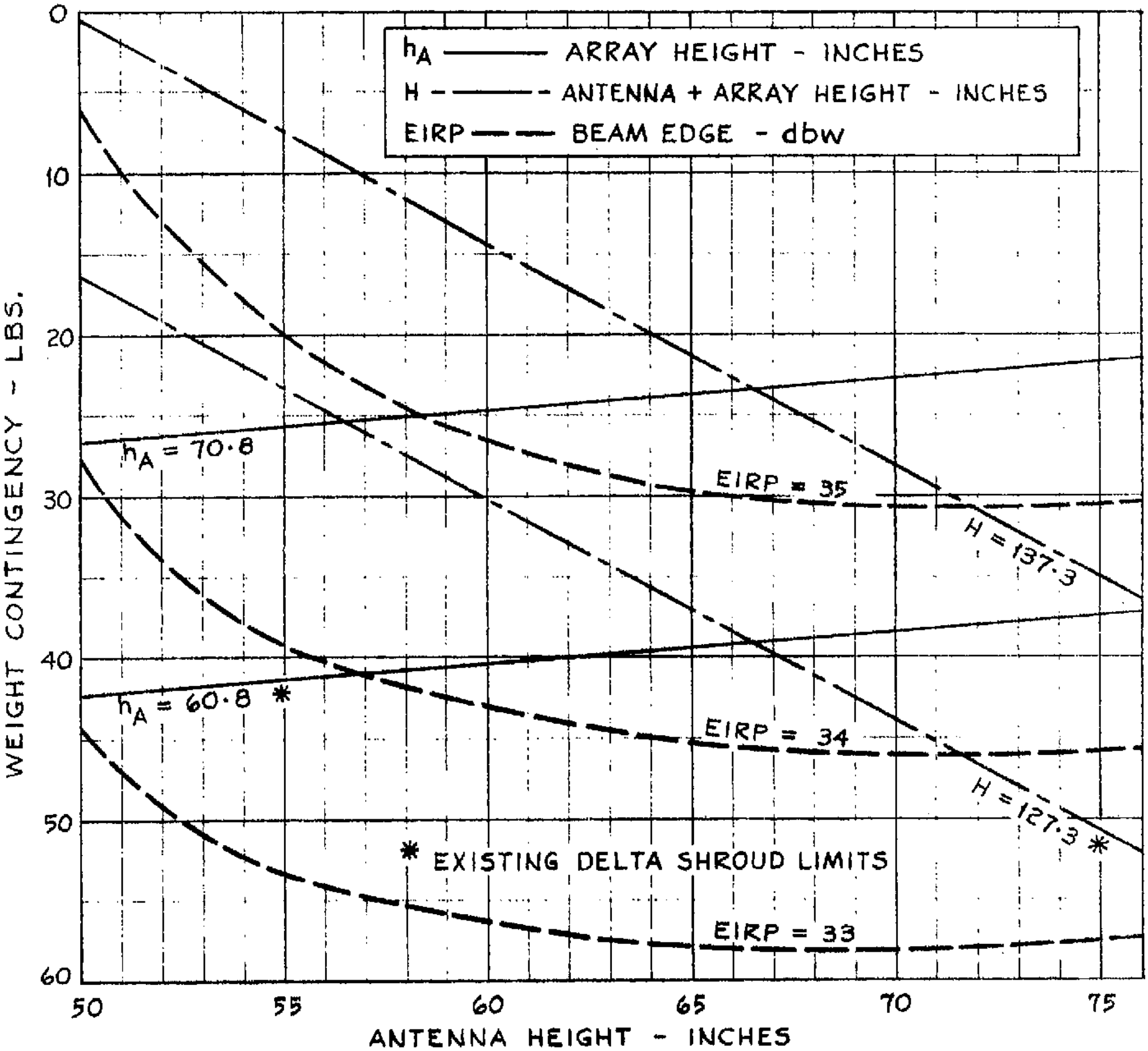
3.1.11 Summary of Spacecraft Configurations

Table 3.1.3 summarizes the important characteristics of several spacecraft configurations that were considered in the preliminary study phase. Following the presentation of the interim report, another configuration using 6 day and 3 night channels, with one for two output TWT redundancy was examined as the preferred configuration.



6 CHANNELS DAY, 2 CHANNELS ECLIPSE, 6 TWT'S, 5 YEARS LIFE

Figure 3–9 Spacecraft Design Parameters



4 CHANNELS DAY, 4 CHANNELS ECLIPSE, 6 TWT'S, 5 YEARS LIFE

Figure 3–10 Spacecraft Design Parameters

	Existing Delta Fairing		Modified Delta Fairing		Atlas Burner II		
RF Channels Day	4	4	6	4	6	6	6
RF Channels Eclipse	2	4	2	4	2	4	6
EIRP Beam Edge	34	34	34	34	34	34	34
Beam Size (3 dB)	3.3 x 8.6	3.3 x 8.6	2.7 x 7.0	2.7 x 7.0	2.7 x 7.0	2.7 x 7.0	2.7x 7.0
Array Weight lbs	39.7	39.7	49.5	37.5	49.5	49.5	49.5
Battery Weight	22.1	33.5	21	31.5	21	31.5	42
Lifetime Yrs	6	5	5	6	6	5	5
Number of OP TWTs	8	6	6	6	9	9	6
Prob of Survival 4 Ch.	.642	.683	.723	.618	.698	.741	.723
Prob of Survival Full	.642	.683	.344	.618	.591	.656	.344
Weight Margin	37	43.6	37.1	36.6	39.1	39.6	41.1
Limitation	DC Power	DC Power	Weight	Weight	Weight	Weight	Weight

Table 3.1.3

Summary of Spacecraft Configurations

I.4
SELECTED DESIGN

4. SELECTED SPACECRAFT CONFIGURATION

4.1 INTRODUCTION

During the second half of the study program, refined estimates of performance of the apogee motor gave a reduced dry weight (45.7 lbs vs 73 lbs) and a higher specific impulse requiring less propellant for the required ΔV. (452 vs 458 lbs). Detailed structural analysis indicated possible savings in weight of about 9 pounds. According to instructions given by the Department of Industry following the presentation of the interim report, any increase in weight margin above that considered adequate, should be used to achieve higher reliability, and more eclipse operation in that order of priority. Accordingly the improvement in weight margin of about 42 pounds was used to improve reliability by providing 3 more output TWTs in a one for two switched redundancy configuration and to provide extra battery capacity for operation of 3 channels in eclipse. The detailed weight breakdown given in section 12 reflects the best available estimates of component weights and the margin of 38 pounds shown is considered realistic. The margin shown is based on the DSV_3L^2 with modified shroud but the same spacecraft design could be used for the Atlas Burner II vehicle giving a corresponding increase in weight margin. No recommendation is made at this time on how the excess margin weight be used.

4.2 GENERAL CONFIGURATION

- A spinning cylindrical body 55" diameter by 72" high with a despun elliptical antenna 74" high and 28" wide mounted at the top of the cylinder. A solid fuel apogee motor is mounted within the bottom part of the cylinder. Equipment mounted on two platforms one at the top of the cylinder supported by a central conical structure and one near the middle. Thermal control is passive by design of structure, arrangement of heat loads and choice of surfaces. Attachment to the launch vehicle is by Marmon clamp at the bottom of the cylinder. Dynamic stability by means of viscous mutation damper on the despun portion.

- Max. weight in transfer orbit = 965 lbs.

- Design lifetime 5 year with 0.7 probability of full operation.

4.3 SUMMARY SPACECRAFT DESCRIPTION

4.3.1. Communications

Frequency plan	12 ch spaced 40 MHz
Receive Band	5.925 - 6.425 gHz
Transmit Band	3.7 - 4.2 gHz
No. channels daylight (all odd or all even numbers)	6
No. channels eclipse	3
EIRP Beam edge	34 dbw
Beam center	38 dbw
Transponder Type	Single translation all RF
No of output TWTs	6 active 3 spare
Useful channel bandwidth	36 MHz
Beam size (elliptical)	$3.25^\circ \times 8.5^\circ$
Beam pointing accuracy	$\pm 0.5^\circ$

4.3.2 Power

Energy Source	Silicon Solar cells N on P
No. of cells	20,000 mounted on 55" dia. cyl.
Array Capacity Summer Solstice	T = 0 294 watts T = 5 yrs 240 watts
Energy Storage	Nickle Cadmium cells 12 AH
No. of cells	20
System Configuration	Shunt limited bus for high loads, regulating converter for low loads.
Terminal Voltage	29 - 34 V DC
Daylight Load	235 watts
Eclipse Load	144 watts

4.3.3. Position and Orientation

Configuration	Total propellant equally divided between independant subassemblies connected by normally closed squib valve.
Propellant	Monopropellant Hydrazine
Pressurant	Gaseous Nitrogen
Weight Dry	16.7 lbs.
Loaded weight	84.9 lbs.
Total velocity increment	1144 ft/sec minimum
No. of axial thrusters	2
No. of radial thrusters	2

4.3.4. Attitude Control and Antenna Despin

Attitude sensing:	earth and sun sensors
Attitude control:	timed pulsing of axial thrusters
Inclination control:	continuous operation of axial thrusters
Position Control:	pulsed operation of radial thrusters
Antenna despin control:	earch sensor reference for servo loop or open loop control from ground control station.

4.3.5. Telemetry and Command

Telemetry frequency	At either edge of communications band 3.7 - 4.2 gHz
RF power	0.5 W
Modulation method	FM
System	PCM/PSK on 1 subcarrier plus analogue FM modulation on 2 subcarriers
Command frequency	At either edge of communications band 5.925 - 6.425 gHz
Modulation method	FM by subcarrier
System	Two step command-execute using digital FSK of subcarrier plus real time FM of subcarriers by analog signals
Antenna	Separate 40° Bicone antenna shared with telemetry
Tracking Beacon	136 MHz 4W with whip antenna mounted on top of bicone
Ranging	Multitone CW using microwave telemetry and command links

4.3.6 Injection Motor

Total impulse	To give ΔV 6040 ft/sec, about 140,000 lb-sec.
Thrust level	Between 3600 and 3960 lbs, not to exceed 10g.
Propellant	TP-4-3135 Polybutadiene
Configuration	Spherical titanium case with high expansion ratio nozzle of metal, plastic and carbon
Weight total	503 lbs.
Inert	50.6 lbs.