O. Roscoe

FXC73-1

STUDY OF MODERATE INCLINATION SATELLITE SYSTEMS

H.J. Moody
D. Bennett

FINAL REPORT
April 1974

Prepared for

DEPARTMENT OF COMMUNICATIONS
Ottawa, Canada

under

Contract No. SW1A 36100-3-0757
Serial OSW3-0462

RCA Research & Development

RCA Limited | Ste-Anne-de-Bellevue, Quebec, Canada, H9X 3L3

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Approved By

Dr. M.P. Bachynski
Director, R&D Laboratories

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1. INTRODUCTION

This is the final report on a study to determine the effect of imparting a moderate inclination to an orbitting multi-purpose UHF spacecraft on mission related factors such as spacecraft design and fuel requirements. The work was carried out for the Department of Communications, Ottawa under DSS Contract No. SW1A 36100-3-0757, Serial OSW3-0462. Mr. O. Roscoe was the DOC project officer.

The study is related to two previous studies. The first[1], under DSS contract OPL1-0005 "Consulting Services for Cost Studies of UHF Satellite Communications Systems" was completed in December 1971. Under this contract, system studies were carried out for both the ground and space segment for a satellite communication system using either 225-400 MHz or 1.5 GHz. The second study[2], under DSS contract PL3610-1-0622, Serial OPL2-0005 extended the system study to determine the feasibility of a hybrid system using two frequency bands 225-400 MHz and 2.5 GHz on the same spacecraft. In the present study the implications are investigated of placing one spacecraft in geostationary orbit to provide the primary service and the in-orbit spare in a moderately inclined orbit to provide service north of the primary service area.

In the remainder of the report, section 2 describes the mission concepts and assumptions, section 3 outlines the ground coverage that can be obtained and the percentage of the day that it is available under different combinations of inclination and elevation angle, while section 4 gives the time of day during which extended northern service is available. In section 5 the implications on the spacecraft weight budget are outlined while the implications on the availability of the primary service and of ENS service are delineated in section 6. Finally in section 7 the conclusions and recommendations are given.

2. MISSION CONCEPTS AND ASSUMPTIONS

The mission concept on which this study is based consists of two identical UHF spacecraft providing communication services in the 2.5 GHz and 300 MHz bands. The first spacecraft to be launched is placed in a near-geostationary orbit and provides a pilot service in both frequency bands. It is launched into a biased orbit with approximately one degree inclination and the appropriate right ascension so that the inclination build up is first towards zero inclination and then to one degree inclination with a right ascension differing by 180°. In this way the inclination will remain below one degree and require no fuel expenditure for the maximum possible time. The second spacecraft is launched about 2 years later and is placed in a moderately inclined orbit so as to provide extended northern service as well as back up for the primary service. The launching of the second spacecraft marks the beginning of the operational phase of the system. A third spacecraft would be held in reserve on the ground for launch at a later date or in the event of an early failure of one of the orbiting spacecraft.

Extended northern service (ENS) at 300 MHz is provided for a percentage of the day (depending on the latitude) by the inclined orbit spacecraft at latitudes out of reach of the geostationary spacecraft. Service from the inclined orbit S/C at 2.5 GHz is precluded by narrow beam non-tracking antennas. The inclined orbit spacecraft must maintain an orbit inclination greater than the specified minimum to ensure that the extended northern service will not degrade with time due to variation in orbital parameters. If necessary, station keeping fuel must be expended to accomplish this. Since the geostationary service is considered paramount, the prime purpose of the inclined orbit spacecraft is to serve as a back-up for this service while the secondary purpose is to provide ENS.

The first priority in this system concept is to maintain one spacecraft in geostationary orbit with minimum interruption of service. This spacecraft is used mainly for 2.5 GHz service and therefore must be placed in geostationary orbit. To ensure that this service is maintained, a spare spacecraft is placed in orbit adjacent to the operational spacecraft. Since the life time and reliability (of the spare) is not greatly altered by using the spare spacecraft, the concept has developed of using the spare spacecraft at 300 MHz and it is subsequently referred to as an operating spare. To further increase the usefulness of the operating spare, it is moved in the orbit until it is separated from the operational spacecraft by more than 18 degrees. With this separation the two spacecraft are not simultaneously eclipsed and a higher level of service can be maintained, though the period during which service is reduced is necessarily longer. If the operational spacecraft fails then the operating spare is moved 18 degrees in the orbit to take up the location and service of the primary spacecraft. To the extent that it will take a few days to move the spare in the orbit, the primary service is already compromised by making use of the spare satellite. In the present study in the event of failure, the operating spare would need to undergo an inclination change as well as be moved in orbit. To bring the operating spare into geostationary orbit in the event of a failure of the operational spacecraft will require not only an elapsed time of several days, but also the expenditure of a considerable amount of station keeping fuel. Thus the life time of the spare spacecraft will be reduced if it must be brought into service in the geostationary orbit. Thus placing the operating spare in an inclined orbit and instituting extended northern service has further compromised its availability as back-up for the primary service and the higher the inclination of the orbit the more severely it is compromised. The optimum can only be determined by the eventual customers who use the various services provided by the satellite. In this study a number of missions are evaluated which are in the range of providing an acceptable compromise between maintaining the primary service and providing an acceptable level of extended northern service.

Since identical spacecraft will be launched into the two orbits, the costs of establishing such a system will be minimized as only one design is necessary. Differences will be apparent between the two spacecraft only during pre-launch preparation. At this time, the Apogee Kick Motor (AKM) would be off-loaded on the spacecraft intended for the inclined orbit since its velocity increment requirements are less than those for the geostationary orbit. Off-loading would not be done before this time so that this spacecraft could be launched into geostationary orbit in the event that the geostationary vehicle fails before the entire system is established. Oversize hydrazine tanks for station-keeping will fly partly empty on the first spacecraft; these tanks will be filled on the second spacecraft to compensate for weight removed from the Apogee Kick Motor. The additional hydrazine is required to effect major inclination and right ascension changes.

The study is based on a 2000 pound transfer orbit weight which is the guaranteed performance objective of the 3914 launch vehicle currently being developed by the McDonnell Douglas Company for the U.S. Domestic Satellite program. The predicted performance is somewhat higher so that a higher transfer orbit weight may become a reality when the vehicle is fully developed. For the purposes of this study the guaranteed figure of 2000 lbs will be used. The apogee motor case design produced by Thiokol to match the 3914 vehicle is oversized so that 10% additional fuel could be accomodated, enough to take care of any projected increase beyond the 2000 lb transfer orbit weight.

3. GROUND COVERAGE FOR EXTENDED NORTHERN SERVICE

3.1 Coverage Contours

To evaluate the effectiveness of moderately inclined orbits as a means of extending northern coverage, it is useful to draw coverage contours as shown in Figs. 3.1 to 3.6. These contours are drawn on plain paper without a corresponding map of Canada. To use them a transparent polar projection of Canada and northern regions is included in this report (pocket in rear cover). The transparency is placed over a particular set of contours such that the vertical line which bisects the contours is co-incident with the desired sub-satellite longitude. The top end of the bisecting line in figures 3.1 to 3.6 is the location of the Pole. It must be made coincident with the north pole on the transparency.

Each set of contours gives ground coverage for various percentages of a day as a function of orbit inclination and minimum ground elevation angle. From North to South, the separate contours give visibility percentages of 0%, 25%, 50%, 75% and 100%. In addition the 50% line gives the coverage from a geostationary satellite for the corresponding elevation angles. Contour sets for inclinations of 5°, 7° and 10° are provided, each for ground terminal elevation angles of 0° and 5°. Contours may be interpreted as lines on which elevation angles to the spacecraft are equal to OR GREATER than the minimum of 0° or 5° for the corresponding percentage of the day.

Examining the set of contours for 7° inclination and 5° elevation angle (Figure 3.4) it is clear that for sub-satellite longitudes within $\pm 15^\circ$ of 105°W, a daily communications "window" of 6 hours duration exists half way up Ellesmere Island. However, Alert (not shown) is not covered at all. On the other hand, for a zero degree minimum elevation angle and 7° inclination it can be seen from the contour set in figure 3.3 that greater than 6 hours coverage exists even north of Ellesmere on the polar ice cap. Such low elevation angles might be useful on the ice cap but are unlikely to be useful on northern Ellesmere where mountain peaks extend to 9500 feet.

It is useful to note that for any inclination and 5° minimum elevation angle, the 50% contour passes essentially through Resolute Bay on Cornwallis Island

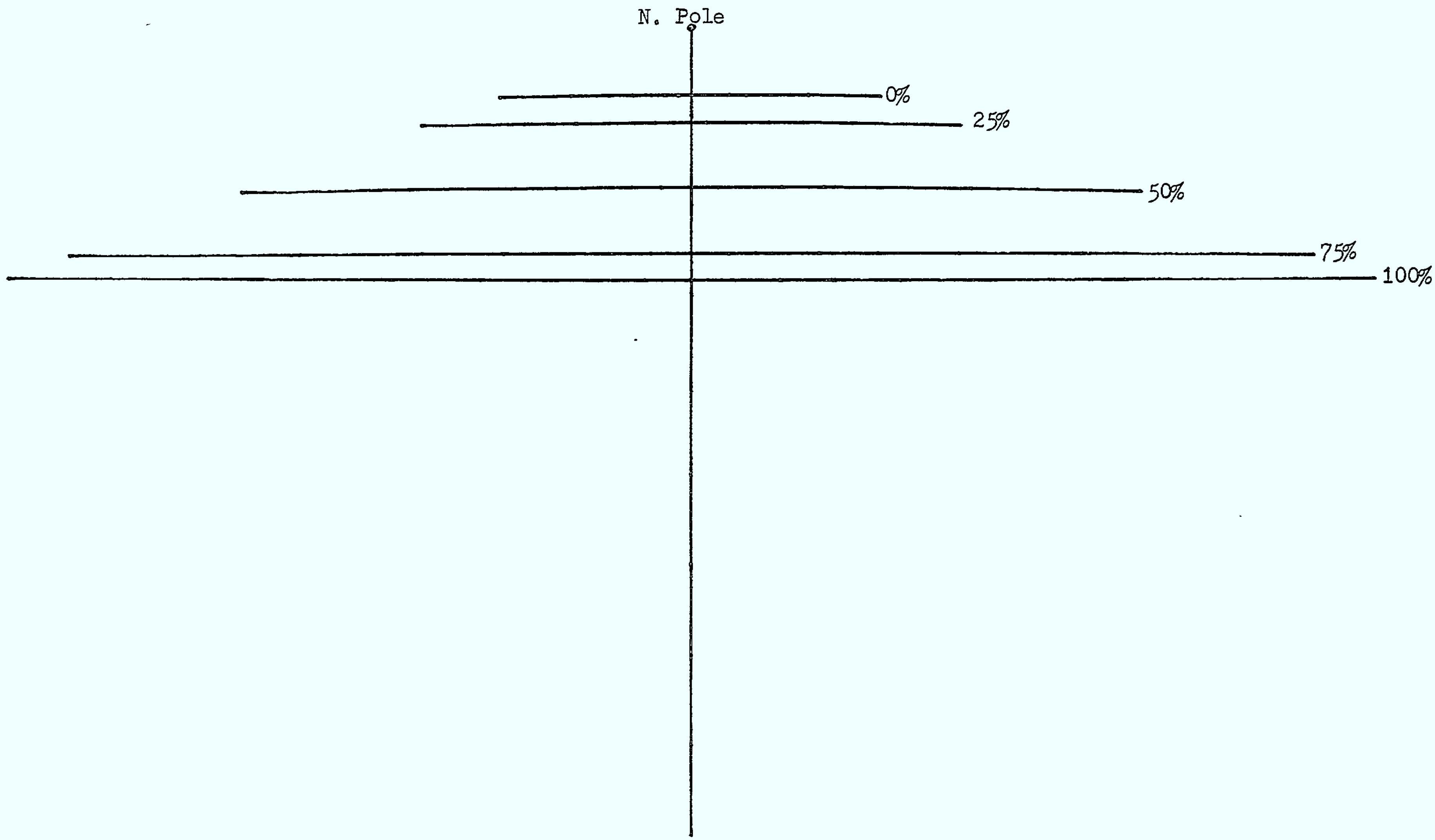


Fig. 3.1 Visibility contours for 5° inclination and 0° elevation angle.

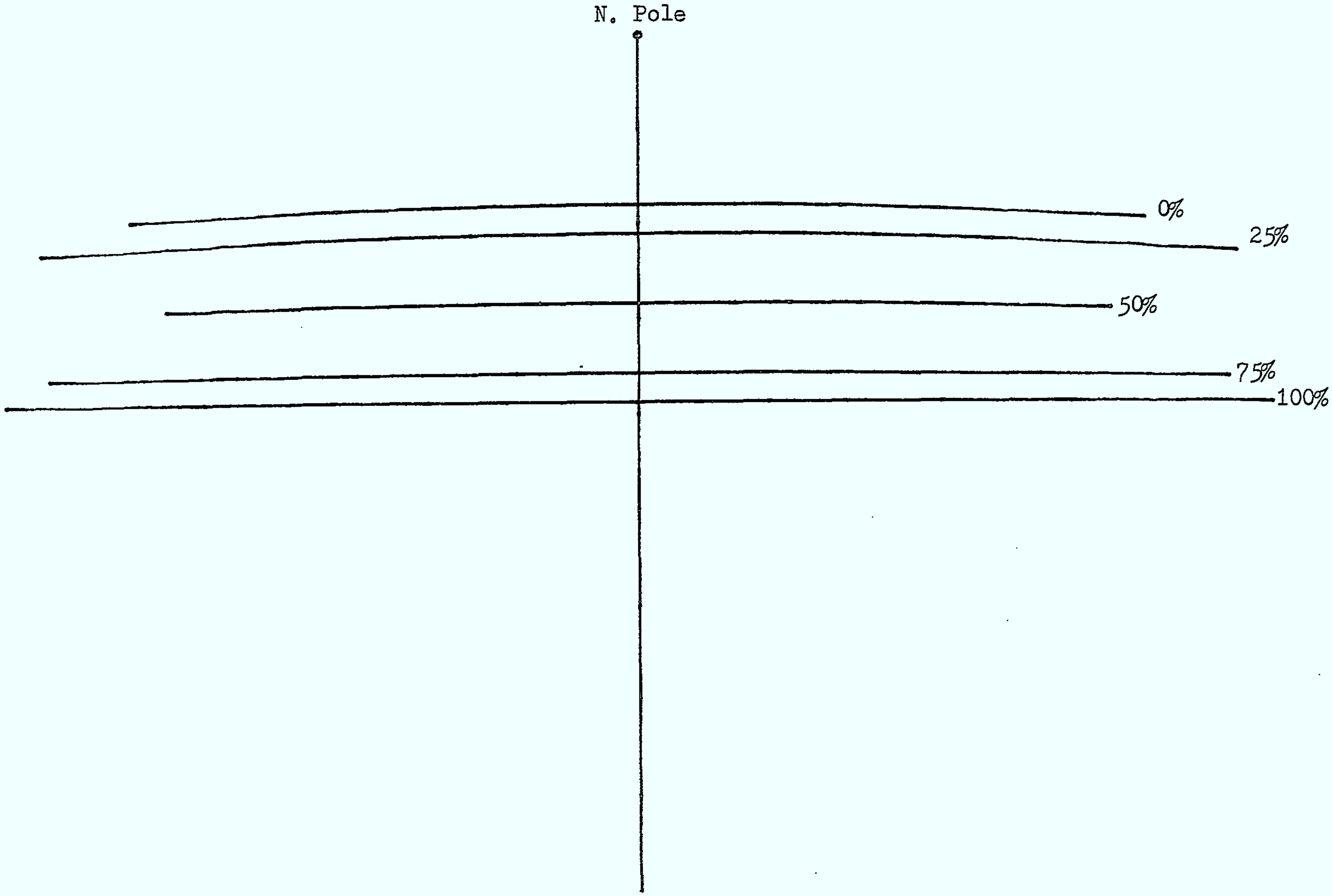


Fig. 3.2 Visibility contours for 5° inclination and 5° elevation angle.

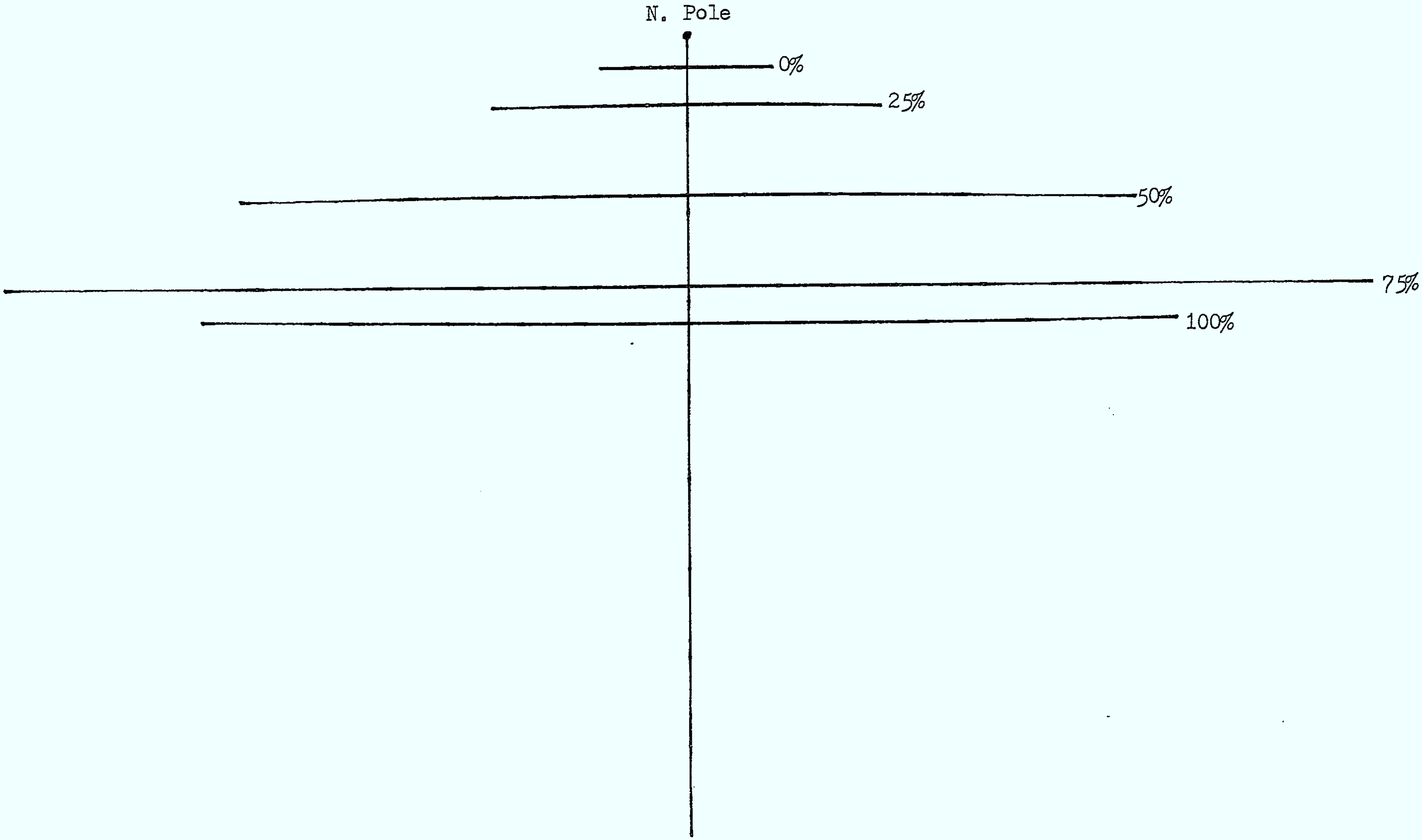


Fig. 3.3 Visibility contours for 7° inclination and 0° elevation angle.

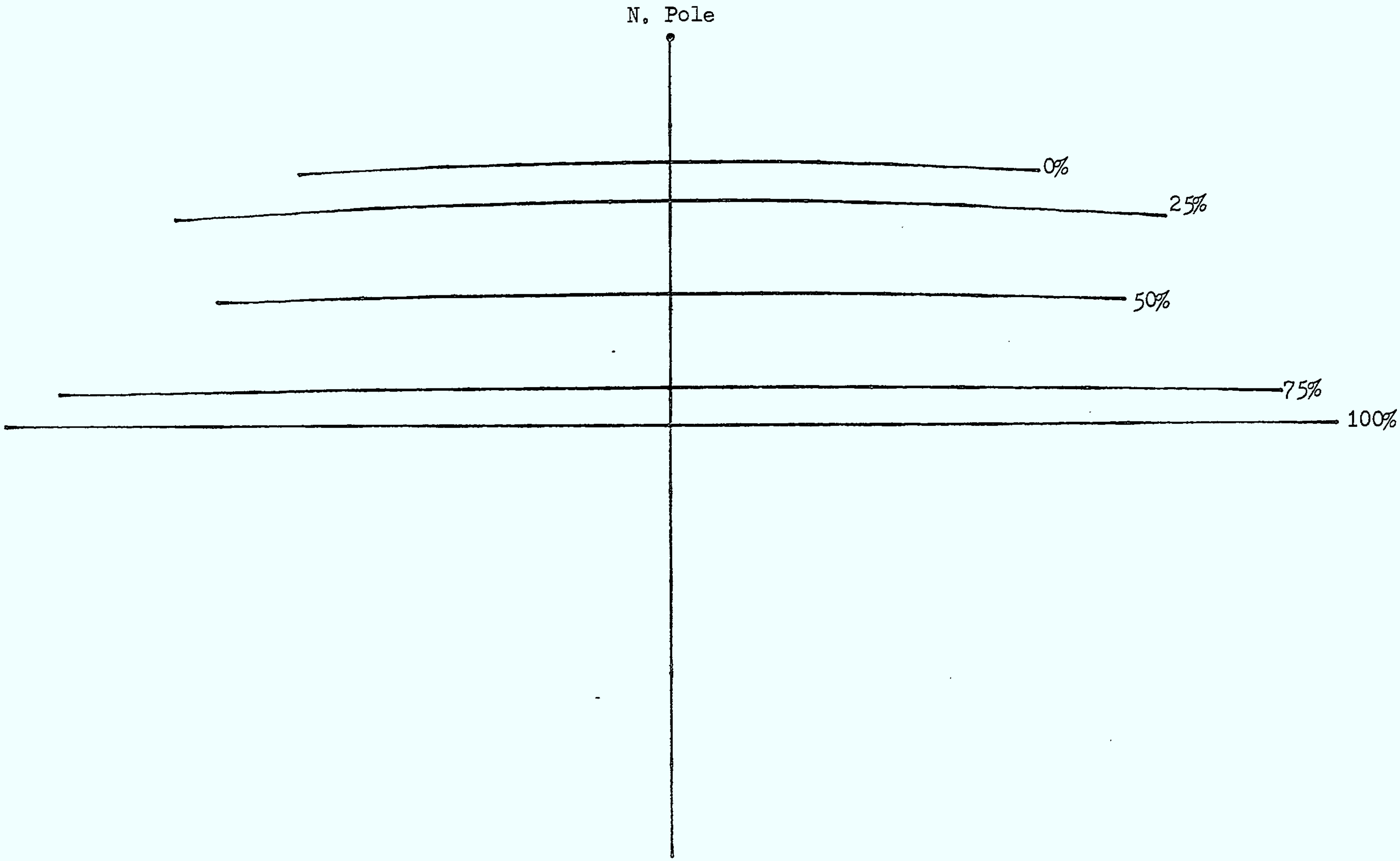


Fig. 3.4 Visibility contours for 7° inclination and 5° elevation angle.

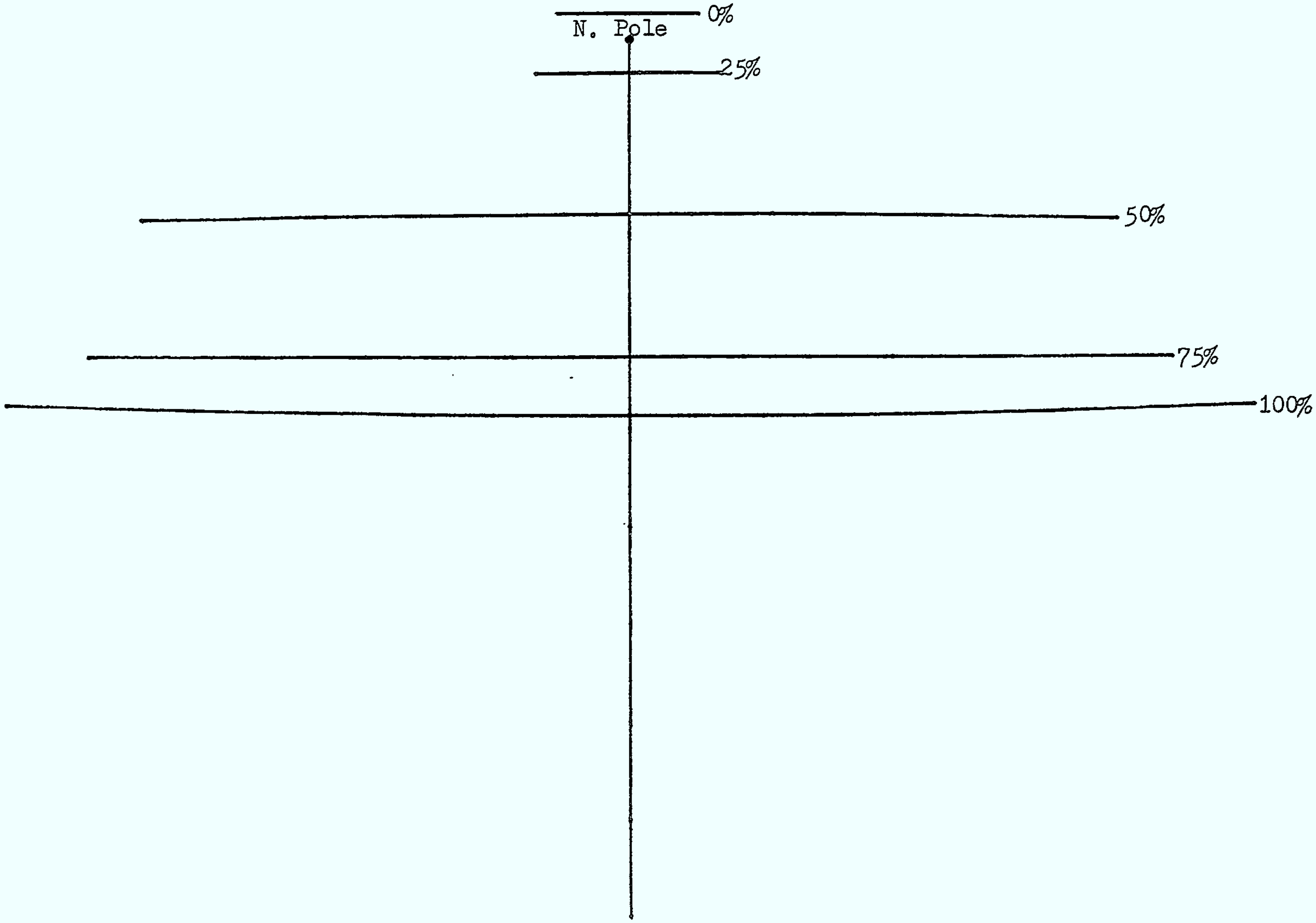


Fig. 3.5 Visibility contours for 10° inclination and 0° elevation angle.

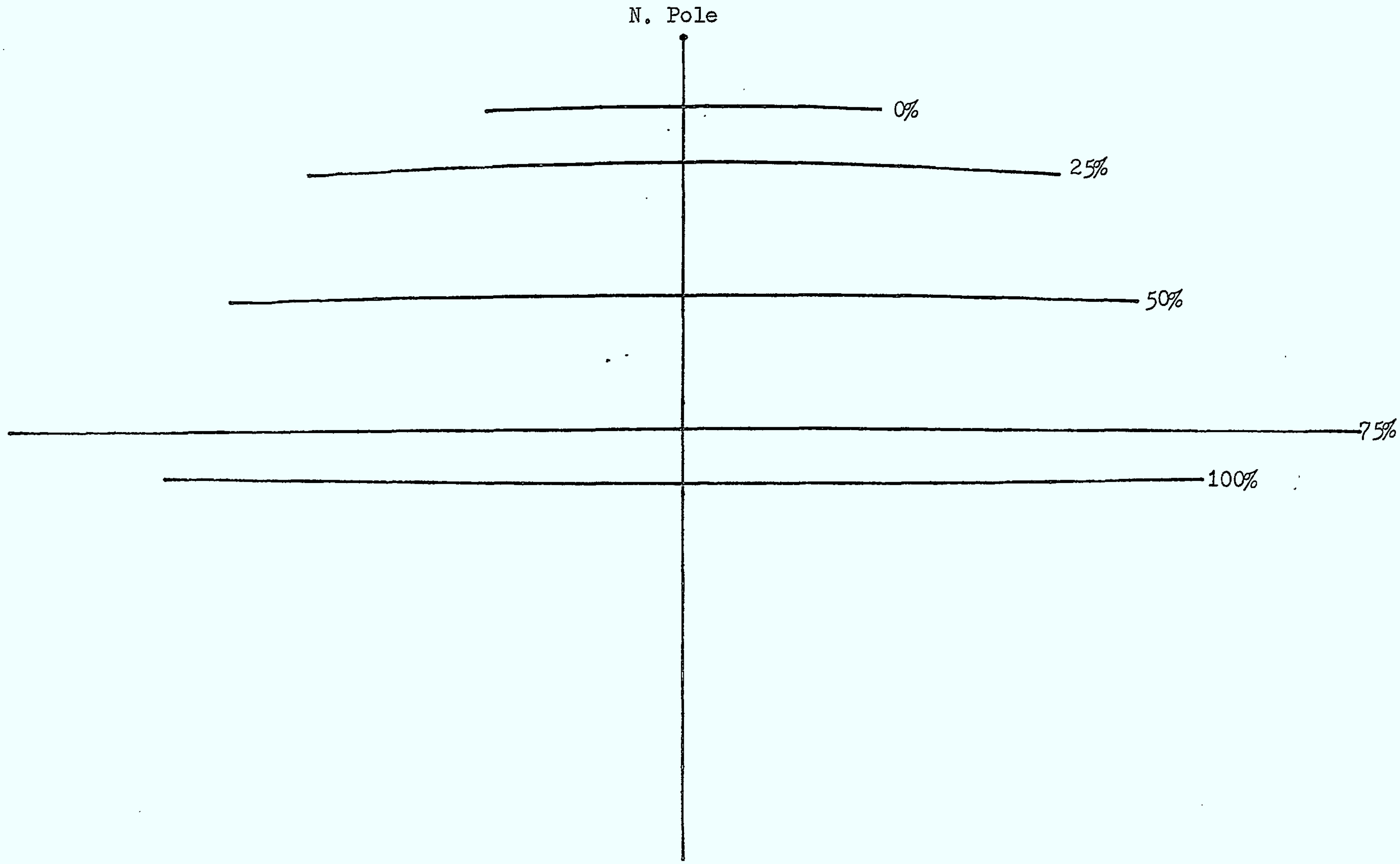


Fig. 3.6 Visibility contours for 10° inclination and 5° elevation angle.

for sub-satellite longitudes within 15° of 105°W. One might also note that for 5° minimum elevation angle, 25% or 6 hour coverage is available over the southern quarter of Ellesmere for 5° inclination, over the southern half of Ellesmere for 7° inclination and over the southern three-quarters for 10° inclination.

3.2 Coverage at Alert

To determine the criteria which could be used to define the minimum service acceptable, a further series of calculations at various inclinations has been carried out for Alert at the northern tip of Ellesmere Island. The results of these calculations are shown in Figure 3.7 where the percentage of the day that service is available is plotted against the orbit inclination with the minimum elevation angle as a parameter. It is seen that, at Alert, with a 7° inclination, the elevation angle is greater than 2.5° for 15% of the day (3.6 hours) and the maximum elevation angle that occurs at any time is about 3.5°.

Alert is located at the most northern point of land in the Canadian Arctic islands and has high mountains to the south. The elevation angle to the mountain peaks and the segment of the orbit they obscure is not known, however for the purposes of this report an inclination angle of 7° will be taken as the minimum that will give an acceptable level of ENS.

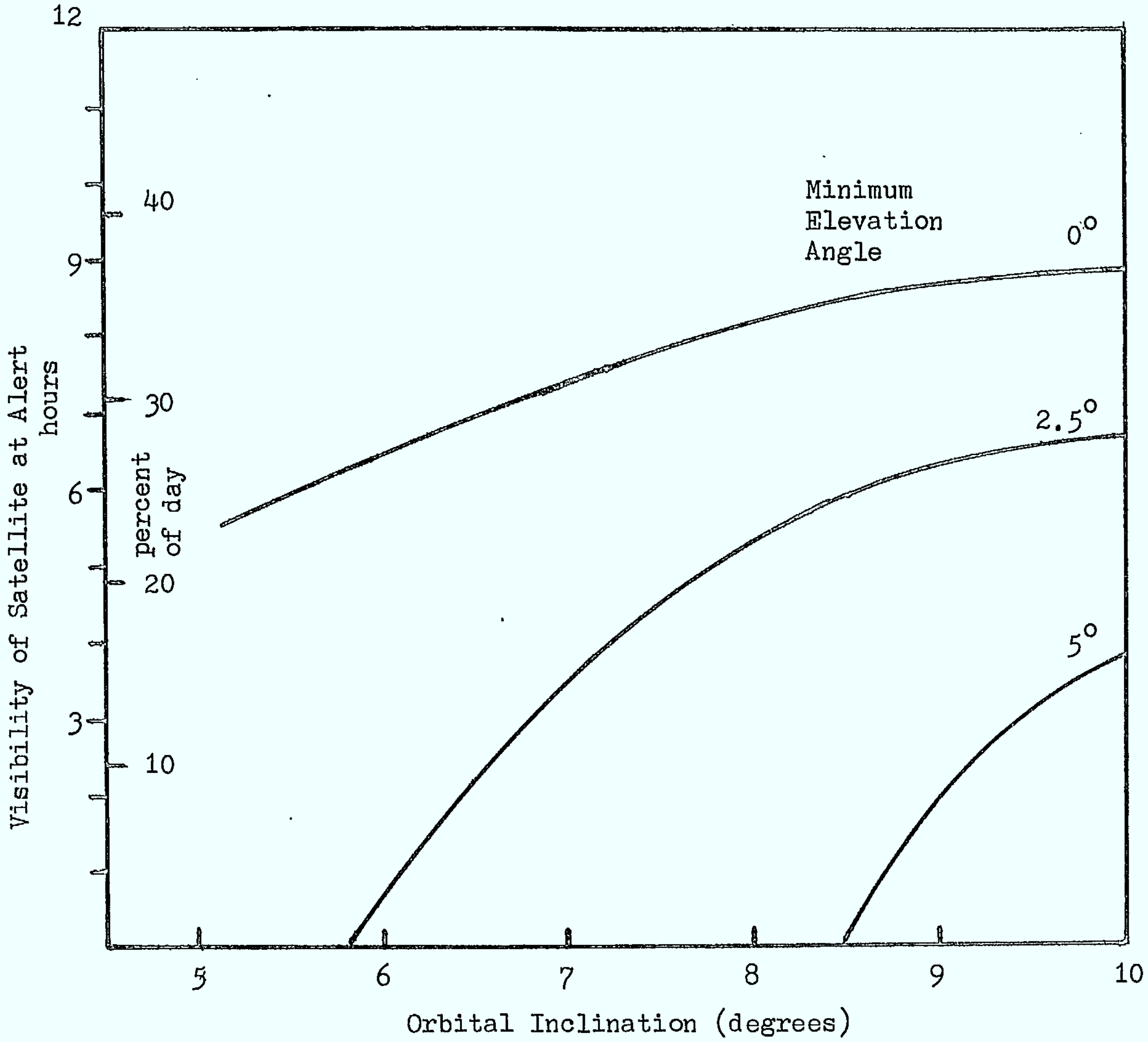


Fig. 3.7 Satellite Visitility at Alert on Ellesmere Island with Subsatellite Longitude of 97°W

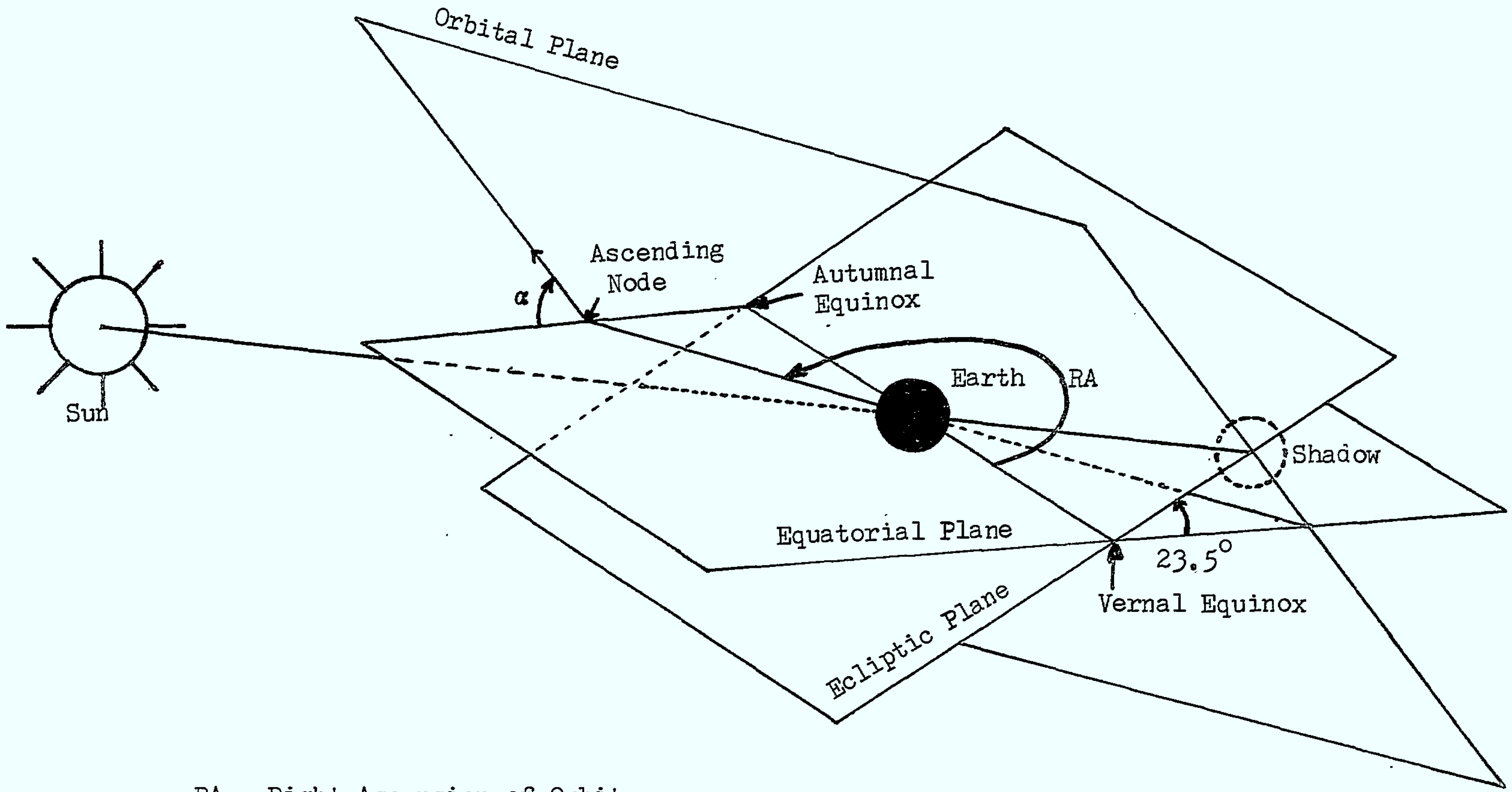
4. COMMUNICATION TIME OF DAY WINDOW

The ground coverage contours of section 3 indicate that the spacecraft is useful for communications for only certain percentages of a day at northern latitudes. The time of the day at which a communication "window" occurs is not fixed however. Any window of arbitrary length will open at an average time of 3 minutes, 56.5 seconds earlier than the day before. This means that for a window of say 6 hours per day, the window will advance through 24 hours in one year. Time of arrival of a window on a given day can be controlled by selection of the right ascension.

Figure 4.1 illustrates the concepts of orbital right ascension and inclination. Right ascension is the angle measured in the equatorial plane between the vernal equinox and the ascending node of the orbit. It is shown in Fig. 4.1 at approximately 210°. For zero inclination orbits (e.g. geosynchronous S/C) eclipses will always coincide with the two equinoctial points. However, for inclined orbits the eclipses may depart from the equinoctial points. If the right ascension of an inclined orbit is zero degrees, the eclipse times are unaffected by the inclination of the orbit, while for other right ascensions the eclipse time depends upon inclination. For high inclinations (near polar orbit) the time of year of the eclipse varies directly with the right ascension, appearing in mid summer and mid winter for right ascensions of 90 and 270°. For small inclination angles as used here, the dependence of eclipse time of year on right ascension is not great.

It has been defined in section 3 that for the purposes of this report an orbit of 7° inclination is the lowest inclination that will give acceptable service for some portion of the day to northern locations. It will be shown in section 6 that there are two plausible right ascensions (0° and 270°) for an orbit with a 7° inclination. The useful time of day for ENS and eclipse periods for these two right ascensions are shown in Fig. 4.2 and 4.3. The useful time is symmetrically placed about the time that the satellite reaches the maximum northern subsatellite latitude. The 25% window is that portion of the orbit occuring within ± 3 hours of maximum subsatellite latitude.

In figure 4.2, for a right ascension of zero degrees, this window opens on January 1 at 8 p.m. and closes on January 2 at 2 a.m. On July 1st, it opens



RA = Right Ascension of Orbit
α = Inclination of Orbit

Figure 4.1 Relationships of Orbital Plane to Equatorial and Ecliptic Planes. Sun is shown such that S/C is experiencing a spring eclipse of maximum duration. (72 min.)

at 8 a.m. and closes at 2 p.m. showing a continuous shift throughout the year. For a right ascension of 0°, spacecraft eclipse always occurs in March and September. As in the case of the geosynchronous spacecraft their maximum duration is 72 minutes, diminishing to zero times 35 days before and after either the vernal or autumnal equinox. Eclipse periods are also shown in Fig. 4.2. It is evident that they occur well outside the period of extended northern service and hence do not compromise communications traffic to northern regions at any time of year.

Window behaviour for an orbit with a right ascension of 270° is given in Figure 4.3. The principal effect of the change in orbit right ascension is to change by 6 hours the time of day during which the extended northern service occurs. An additional though less pronounced effect is to delay by 18 days the time of year at which eclipse occurs. As seen in Figure 4.3 the net effect of these changes is to place the autumnal eclipse within the 25% of the day during which the satellite provides extended northern coverage.

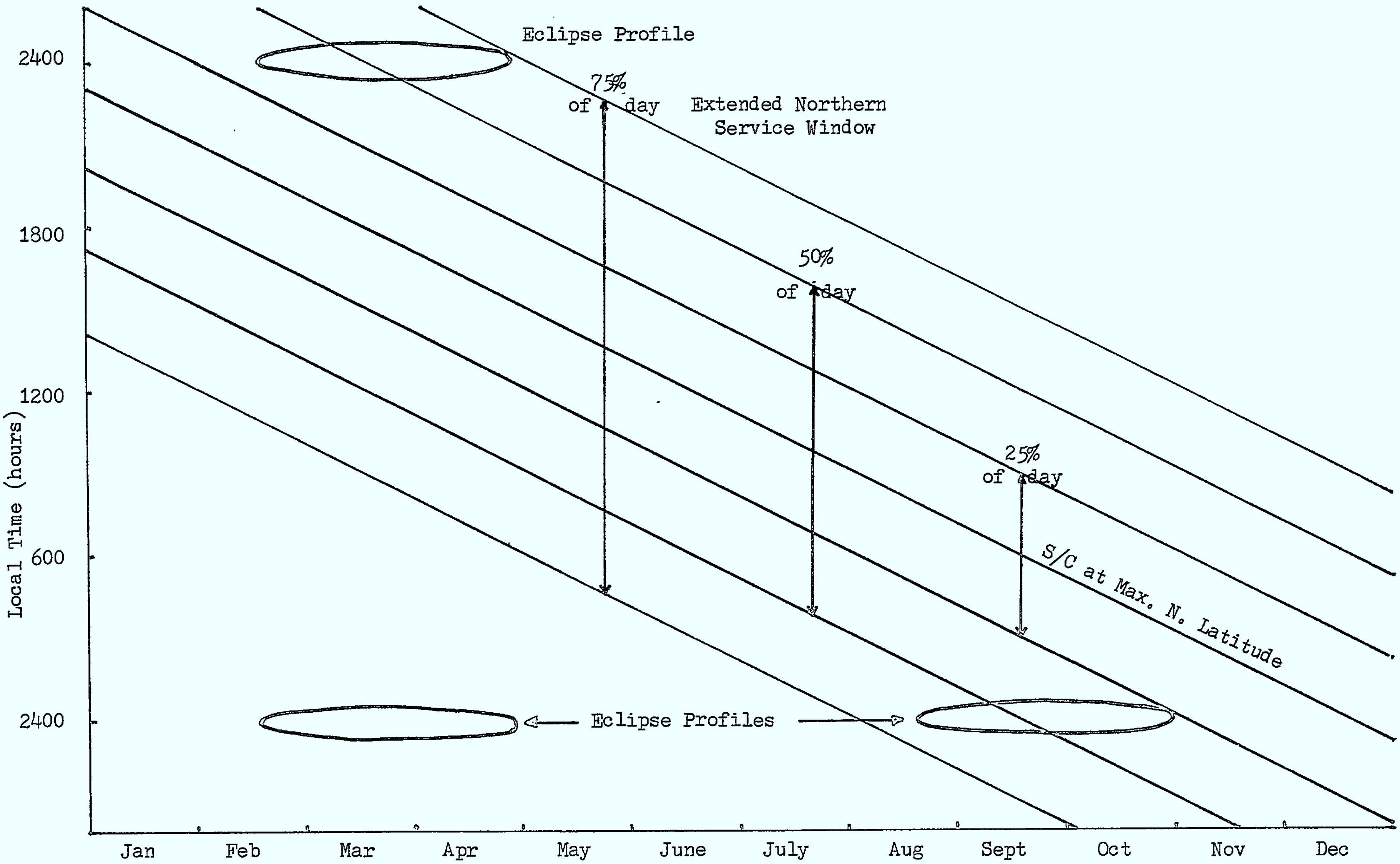


Fig. 4.2 Visibility window vs time of year for a synchronous circular orbit with 7° inclination and 0° right ascension.

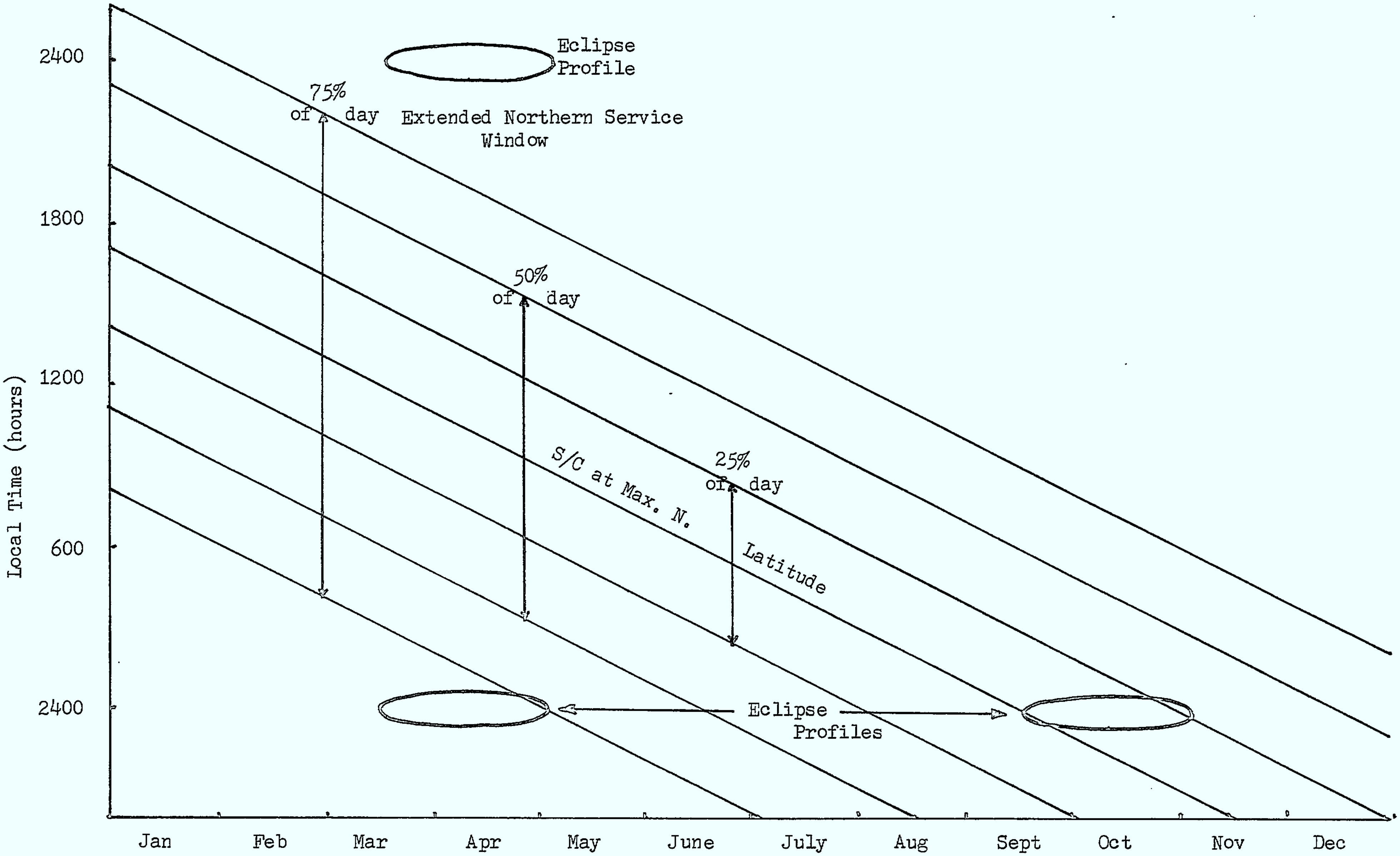


Fig. 4.3 Visibility window vs time of year for a synchronous circular orbit with 7° inclination and 270° right ascension.

5. IMPACT ON SPACECRAFT DESIGN

5.1 Spacecraft Configuration

In the previous study, dated December 1972, a comparison was made between a dual spin design and two 3-axis designs, one based on CTS and the other based on the RCA US domestic satellite design.* At that time the RCA design was somewhat preliminary and sufficient detail was not available to do a detailed comparison. However, at the present time RCA has initiated a program to produce a domestic satellite system for Globecom and detailed numbers are available and firm. For this reason a detailed comparison can be carried out at this time.

The main difference in design concept between the two 3-axis spacecraft is in the solar panel configuration; one has flexible roll out solar panels while the other design has rigid fold out panels. These panels fold up against the body of the spacecraft during launch and fold out on articulated members when deployed. These designs will be referred to as the flexible panel and the rigid panel designs in this report. The rigid panel design can supply prime power up to about twice that available from the dual spin concept. Higher powers could be provided by additional folds in the panels but the pounds per watt ratio becomes progressively higher. On the other hand, the flexible panel has a very low incremental pounds per watt figure and it is possible to generate several kilowatts using this concept. There is however, an initial weight penalty associated with roll-out mechanism which makes the flexible panel concept less attractive for low and intermediate power levels. For the present mission a minimum power level of about 400 watts is required. This is at the limit of the dual spin capabilities but fits easily within the range available from the rigid panel concept. On the other hand, it is well below the kilowatt range where the flexible panel design is optimum.

Other improvements have been made in the rigid panel design to make the US domestic satellite mission possible on a Thor Delta launch. The most evident is the apogee kick motor in which the fuel efficiency has been improved and the

* This is currently called AED SATCOM after the Aerospace Electronics Division which is producing it.

weight of the case has been reduced. These taken together have made a significant reduction in the weight of the apogee motor.

An additional major improvement is in the launch capability of the Thor Delta vehicle. McDonnell Douglas is currently developing the 3914 launch vehicle to have a planned capability of 2000 lbs into transfer orbit. This is 450 lbs higher than any previously available and is also higher than the figure of 1890 lbs proposed by NASA and used in the previous UHF satellite study.[2]

The AED Satcom design consists of a rectangular box with two fold out solar panels. Each panel is hinged in the centre and folded in half when stowed so that they fit against two opposite sides of the box. One half of each panel faces out and provides a limited amount of power during the transfer orbit when the spacecraft is spinning. When deployed the solar panel mounting shaft is rotated once per day to keep the solar panels oriented towards the sun while the body of the spacecraft is kept oriented towards the earth. The antennae are mounted directly on the body of the spacecraft and face down towards the earth. To accomodate the larger antenna diameter required at 300 MHz the spacing between the solar panels and the body would have to be increased. A possible alternative shown in figure 5.1 is a sketch of a 3-axis rigid panel configuration taken from the previous report.[2]

5.2 Apogee Kick Motor Offloading

A 2000 lb spacecraft can be launched into a 28.3° inclination transfer orbit by the 3914 Thor Delta launch vehicle. A 28.3° inclination results from a standard launch aximuth of 95° from the Kennedy Space Centre in Florida. No perigee vectoring (transfer orbit) is contemplated, hence the 2000 lb figure can be considered a firm minimum.

A kick motor similar in performance to that currently planned for the AED SATCOM program has been assumed. This motor has a specific inpulse of 291.4 seconds and an effective mass fraction of .952. An attach fitting of 75 lbs is used to secure the spacecraft to the Thor-Delta third stage motor. The attach fitting must be launched into transfer orbit and is therefore included in the 2000 lb transfer orbit weight.

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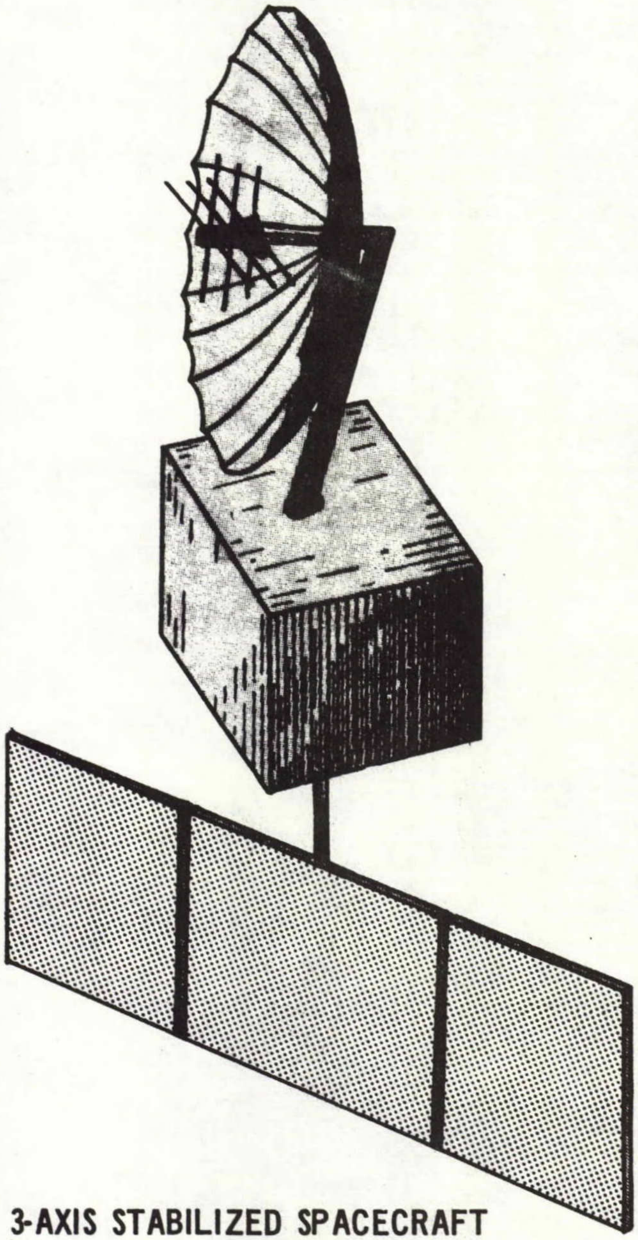


3-AXIS STABILIZED SPACECRAFT
RIGID PANEL SOLAR ARRAY

Figure 5.1

Weight calculations have been carried out for a 2000 lb transfer orbit weight using this kick motor to achieve geostationary and final orbit inclinations of 1°, 5° and 7°. Table 5.1 lists the results of the calculations. The weights listed for one degree inclination are those that apply to the first launch into geostationary orbit. This orbit is in fact a biased orbit with one degree inclination allowing the maximum time in space for a given hydrazine fuel load (as described in section 2.0).

Apogee motor fuel weights for 5° and 7° apply to the inclined orbit spacecraft. Two inclinations have been calculated allowing a comparison to be made between systems with these two orbit inclinations.

Table 5-1 APOGEE MOTOR PERFORMANCE

All Weights are in Pounds

Inclination	0°	1°	5°	7°
Total Spacecraft Weight (lbs)	2000	2000	2000	2000
Attach fitting	75.0	75.0	75.0	75.0
AKM case weight	48.5	48.5	48.5	48.5
Consummable INERTS	9.8	9.8	9.8	9.8
AKM Propellant Weight	908.5	891.3	862.8	849.4
Total AKM Weight	966.8	949.6	921.1	907.7
Useful Payload	958.2	975.4	1003.9	1017.3
Effective Mass Fraction	.952	.952	.952	.952
Velocity Increment (fps)	6011.1	5939.3	5668.5	5544.1
Firing angle (out of transfer orbit plane)	52.6°	51.1°	44.7°	41.3°

The propellant weight difference between the 1° and 7° orbits is 41.9 lbs. This amount is offloaded for the inclined orbit spacecraft and replaced by 41.9 lbs of hydrazine in the RCS system. Offloading of such an amount (4.7%) is well within current practice. If a different orbit inclination is planned such as 5° then a different amount of apogee motor fuel is offloaded and replaced by hydrazine fuel. In any case the amount of offloading is well within the limits of current practice.

5.3 Weight Budget

A comparison is made in table 5.2 between the weight budgets for a 3-axis rigid panel design and a dual spin design both for the 3914 launch vehicle. The 3-axis weights are obtained from the US domestic satellite design which is already sized for the 3914 launch vehicle, while the dual spin weights are derived from the budget for the 1890 lb version developed in the previous UHF satellite study.[2] These budgets are both for an unbiased orbit with zero degree inclination. Some changes have been made to both budgets in order to make them as directly comparable as possible. These changes include:

1) The apogee motor fuel and case weights for the 3-axis design have been used also for the dual spin spacecraft.

2) The eclipse power on the 3-axis spacecraft has been reduced to the 245 watts originally used on the dual spin design during the previous study.

3) The spacecraft margin has been made equal to 100 pounds (5% of launch weight) for both designs.

4) A nominal figure of 200 pounds has been allotted to the payload for both spacecraft. This is an increase of 38 pounds over that estimated in the previous study and as such represents some additional margin.

5) The structure for the dual spin estimate was increased from the previous study to take account of the increased transfer orbit weight.

One item that was not made identical in the two designs is the prime power. Thus the rigid panel has a prime power at end of life of 490 watts while the dual spin has 360 watts.

The resulting weight budgets for the dual and the 3-axis rigid panel designs are shown in Table 5.2. The main difference is in the power subsystem. This is mainly due to the fact that about three times as many solar cells are needed for the dual spin as only about one third of the total are illuminated at any instant. There are other variations, particularly the attitude control system. The 3-axis is heavier due to the use of magnetic torquing for attitude control as well as a back-up hydrazine jet control system. The net difference between the two budgets is about 32 lbs in favour of the

Table 5-2 WEIGHTS BUDGETS FOR THE GEOSTATIONARY ORBIT

	Dual Spin (lbs)	3-Axis (lbs)
Launch capability	2000	2000
Attach fitting	75	75
Apogee Motor Fuel and Consummables	908.5	908.5
Initial on orbit fuel S/C	1016.5	1016.5
Apogee Motor Case (retained)	48.5	48.5
Structure	116.0	131.9
Power, including batteries for 245W	225.0	153.5
Thermal	20	15.8
Attitude control	40	65.4
TT&C	24	33.1
Misc.	26	11.3
Reaction control (dry)	33	41.1
Spacecraft margin	100	100
Total bus weight (dry)	632.5	600.6
Communication payload	200	200
Reaction control propellant	184	215.9
Propellant for all but N-S control	57.9	57.9
Fuel for N-S control	126.1	158.0
Years of N-S control	6.2	7.9

3-axis design. This difference has been taken up by an increased station keeping fuel load giving an increase in life time of nearly two years.

6. MISSION PROFILE CONSIDERATIONS

6.1 Drift in Orbit Inclination

The Orbital Parameters of right ascension (Ω) and orbital inclination (i) of a satellite will be perturbed throughout the lifetime of the spacecraft by earth non-sphericity and by solar-lunar gravitational attractions. The long term effects of these perturbations are represented graphically with a high degree of accuracy in Fig. 6.1*. In this diagram the orbital inclination is represented by the radial distance from the center and the right ascension of the orbit is represented by the angle at the center referenced to zero degrees as datum. All right ascensions and inclinations are represented on the diagram and there is a one to one correspondence between points on the diagram and earth centered orbits. Since changes in right ascension result in an orbital plane change similar to that caused by changes in inclination, fuel must be expended to maintain the right ascension. For small inclinations the plane change caused by a change in right ascension is much smaller than the right ascension. Thus, for a 1° inclination, a change in right ascension by 180° will only cause a plane change of 2°, while a 90° change in right ascension causes a plane change of about 1°. It is evident then that for small inclination angles, such as those figure 6.1 is limited to, line lengths between two points, measured in radial units, give approximately the magnitude of the plane change between the two orbits represented by the two points.

An examination of Fig. 6.1 shows a number of curved lines each representing five years of drift. Each line also approximates an arc of a circle centered at C and subtending an angle of 34°. Orbits close to the "stable point" C have very low drift rates while those far from the stable point have high drift rates and take a lot of fuel to maintain in the same orbit. In effect, an instantaneous centre of rotation for the orbit normal can be identified and the inclination/right ascension time histories appear as arcs about this instantaneous center.[3] In time the arcs close to form circles. In other words the orbits precess about the stable point with a periods of about 53 years.

It is evident from Fig. 6.1 that i/Ω time histories depend on the initial right ascension Ω_o and on the initial inclination, i_o. Also, while it is assumed that the drift rates are constant for specific initial conditions, there are minor variations which

* This figure was produced by HITECH CANADA LIMITED, Ottawa, Ontario under subcontract to RCA Limited

change the pattern somewhat. Figure 6.1 has been calculated assuming that the launch occurred in the first quarter of 1978, drift behaviour for other launch dates can be expected to show the minor variations from that of Fig. 6.1 but would not invalidate the conclusions of this report.

Figure 6.1 shows plots for i_o of 5^o and 10^o and Ω_o of 0^o, 225^o, 270^o and 315^o. A special case of i_o 7.5 and $\Omega_o = 0^o$ is also shown to define more closely the location of the stable point. The initial conditions for the cases are listed in Table 6.1.

Table 6-1 INITIAL CONDITIONS FOR CASES OF FIGURE 6.1

Case	Initial Inclination (deg)	Initial Right Ascension (deg)
1	5	0
2	5	315
3	5	270
4	5	225
5	10	0
6	10	315
7	10	270
8	10	225
9	7.5	0

6.2 Candidate Inclined Orbit Missions

It has been defined in section 3.2 that for the purposes of this report an orbit inclination of 7^o is the minimum inclination that will give an acceptable level of extended northern service. In section 6.1 a particular combination of inclination and right ascension was identified which has minimum inclination and right ascension drift rates and therefore requires a minimum expenditure of station-keeping fuel. In this section two possible mission stratagies are identified while in section 6.4 the effectiveness of these two missions in providing service is determined.

In the case of both mission strategies, the first satellite would be launched with an inclination of one degree and a right ascension of 270^o. This allows the satellite to drift in orbit for the maximum time without expending fuel and without exceeding the one degree maximum inclination angle. In this way the expenditure of fuel is minimized and the anticipated life in space is maximized.

The second spacecraft is launched with 7° inclination but with either a 0° right ascension or a 270° right ascension. With zero degree right ascension the orbit has a very low drift rate and little or no fuel is required to maintain the inclined orbit. However if the inclined orbit spacecraft is required to replace the geostationary satellite it must undergo a plane change of 7°. On the other hand, with a right ascension of 270° the corresponding plane change is 6° but fuel must continuously be expended to maintain the inclined orbit. To determine the prefered strategy the more detailed comparison of section 6.4 is required. The initial parameters for the two missions are listed in table 6.1.

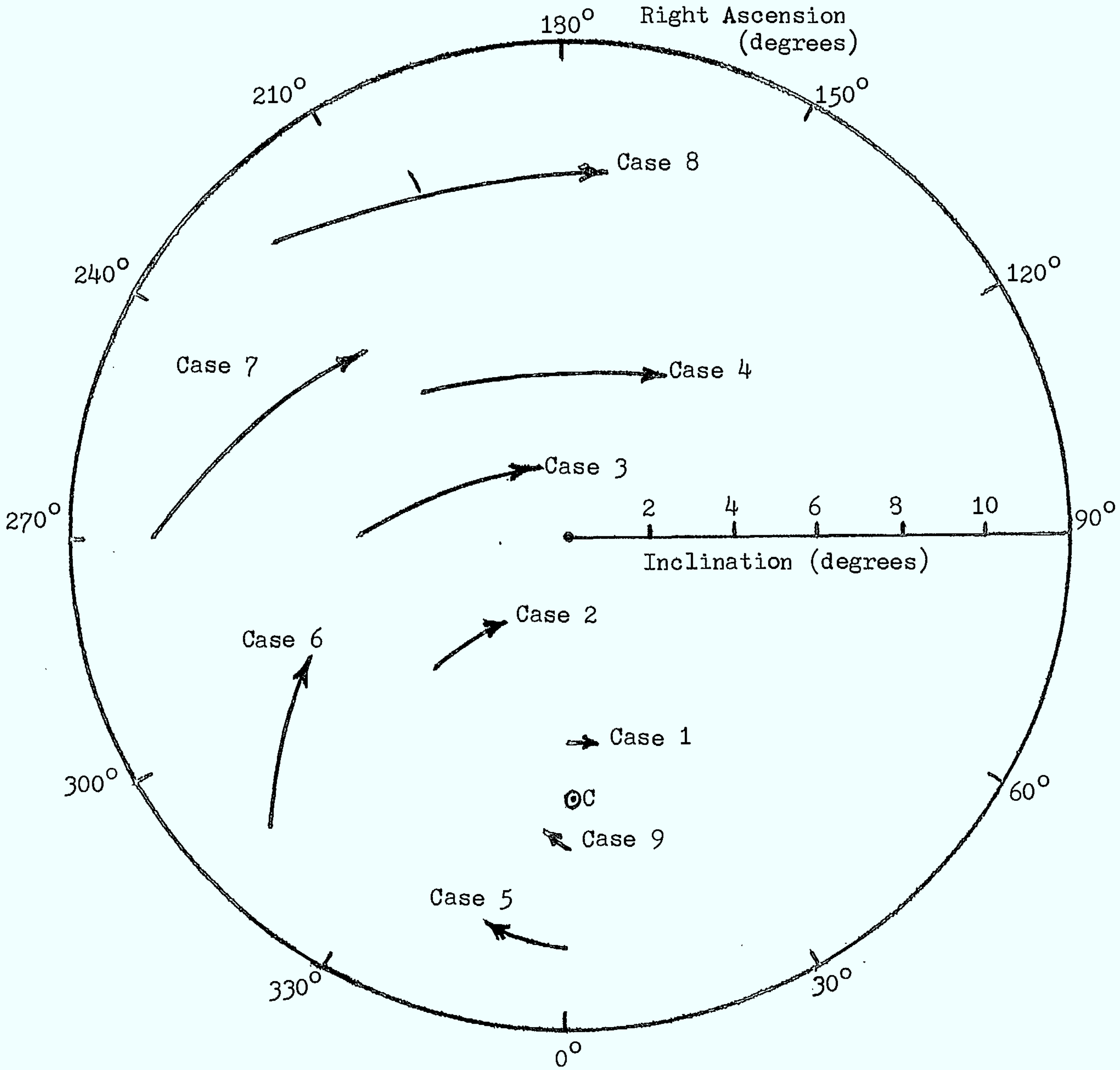


Fig. 6.1 Long Period Motion of Orbital Plane for 5 year Period Beginning 1st Quarter of 1978 for Various Initial Inclinations and Right Ascensions.

Table 6-2 SPACECRAFT INITIAL ORBITAL PARAMETERS

	Inclination	Right Ascension
Mission 1		
Satellite 1	1°	270°
Satellite 2	7°	270°
Mission 2		
Satellite 1	1°	270°
Satellite 2	7°	0°

These two missions are sketched in figure 6.2 with the initial orbit for satellite one and the satellite two initial orbits for the two missions shown as circles. Also shown is the drift path of the satellites through geostationary orbit. The dotted lines show the plane changes required to bring the inclined orbit spacecraft into geostationary orbit.

6.3 Fuel Budgets for Competing Missions

In the previous study Ref. 2 it was assumed that both the operational spacecraft and its "operating" in-orbit back-up spacecraft were in geostationary orbit. In the event that the 2.5 GHz transponder on the one spacecraft failed, the two spacecraft could be interchanged with minimum penalty in station keeping fuel. In this study the "operating" in-orbit back-up is in an inclined orbit and considerable station keeping fuel must be used up in order to bring it into geostationary orbit. Thus, by putting the spare spacecraft into an inclined orbit and providing some measure of extended northern service, the probability of having the primary service is reduced at some later date after the transfer from inclined to geostationary orbit. Under limiting conditions the inclination will be such that all the fuel will be used up in bringing the spacecraft to the geostationary orbit and it will be useful in that orbit only for the length of time it takes the satellite to drift outside the beam width of the ground stations. For inclinations greater than this limit the inclined orbit spare does not provide back-up for the geostationary satellite. Thus it is evident that, as the quality of extended northern service and the probability of having it increase, the probability of having the primary service decreases. The purpose of this study is to find a compromise

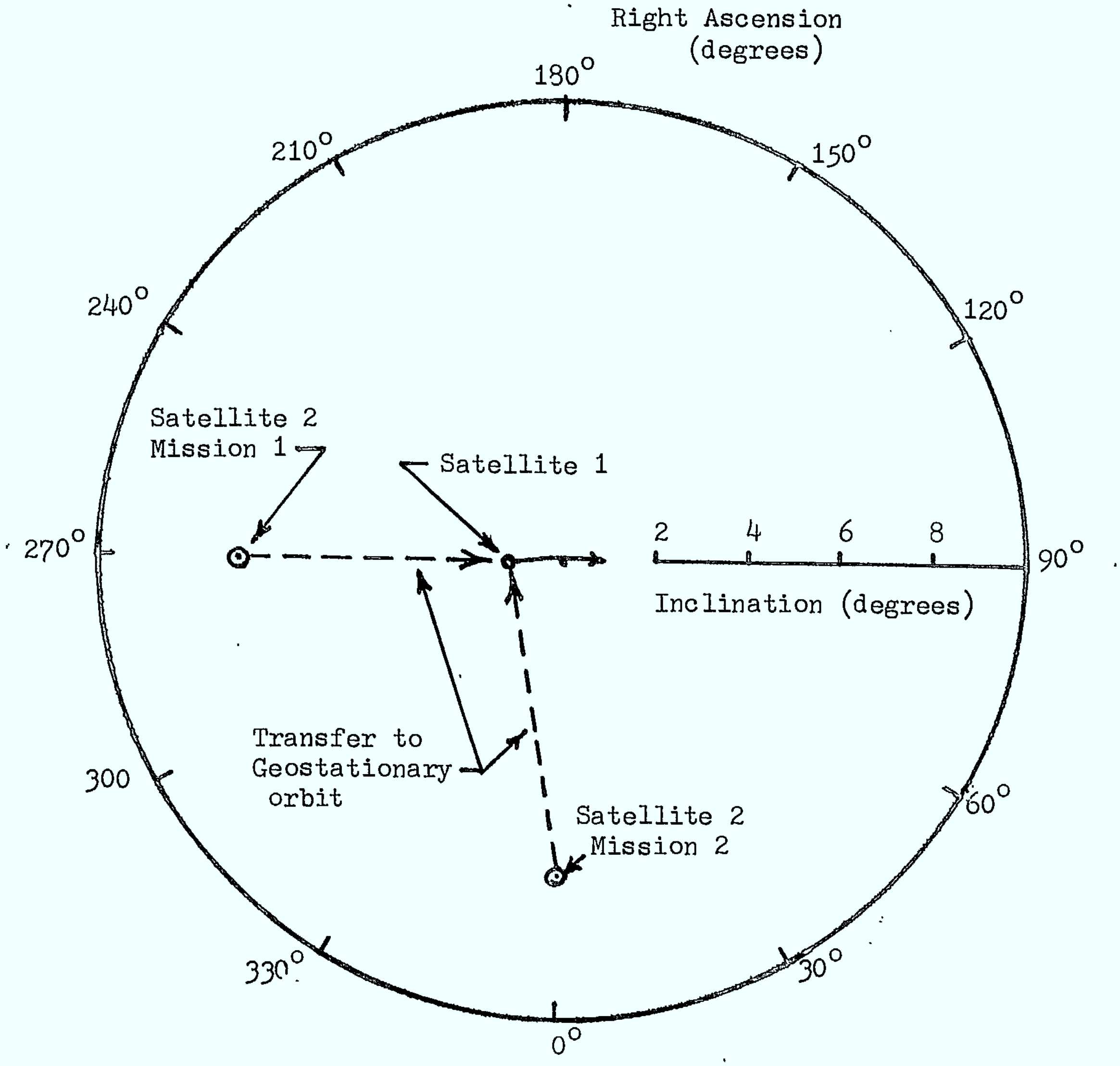


Fig. 6.2 Orbit Conditions for Two Possible Mission Profiles.

between a reasonable level of extended northern service and an acceptable degradation of the primary geostationary service. A related requirement of the study is to determine the best mission profile to maximize, as much as possible, the quality of both services and the probability of having them. In addition, the performance of the dual spin spacecraft is compared with that of the 3-axis rigid panel configuration for one typical mission profile.

Two mission profiles have been outlined in section 6.2, the first being to launch the inclined orbit spacecraft with a right ascension of 270° and the second to launch it with a right ascension of 0°.

The dual spin and 3-axis spacecraft are compared in table 6.2 for a strategy in which the second spacecraft is launched with an inclination of 5° and a right ascension of 270°. Fuel loads are shown for an orbit inclined at 5° and for a geostationary orbit (actually inclined at 1° with right ascension of 270 degrees). The station-keeping fuel and the portion of it to bring the inclined orbit S/C into geostationary orbit are converted to years of inclination control. For the inclined orbit spacecraft the difference between these times, called the "remaining years", may be used to keep it in the inclined orbit or to keep it in geostationary orbit after it is transferred to geostationary service. If it is not required for geostationary service during this time it can not be transferred at a later date as it will no longer have the necessary fuel load. Thus the fuel that would have been used to move it to geostationary orbit is available to keep it in inclined orbit for many additional years. Both spacecraft are assumed to have useful life during the time it takes to drift through the 1° inclination tolerance in geostationary orbit. No drift time is allowed in the inclined orbit because it is assumed that it is held at the minimum inclination that will provide acceptable extended northern service.

It is evident from table 6.2 that the 3-axis rigid panel configuration has a heavier load of station keeping fuel than the dual spin configuration and therefore has a potentially longer life in space.

In table 6.3 the 3-axis configuration is evaluated for the two mission profiles described in section 6.2. For mission 2, the inclined orbit drift rate is very low or non existant and the drift that does occur is drift in right ascension rather than drift in inclination. For this reason no fuel need be expended for inclination control.

Table 6-3 WEIGHT BUDGETS FOR TWO SPACECRAFTS

	DUAL SPIN		3-AXIS	
	1° Incl.	5° Incl.	1° Incl.	5° Incl.
Apogee Motor Fuel and Consummables	901.1	872.0	901.1	872.0
Initial on Orbit fueled S/C	1023.9	1052.4	1023.9	1052.4
Total Bus Weight (Dry)	632.5	632.5	601.6	601.6
Communication Payload	200	200	200	200
Reaction Control Propellant	191.4	219.9	222.3	250.8
Propellant for all but N-S Control	57.9	57.9	57.9	57.9
Fuel for N-S Control	133.5	162.0	164.4	192.9
Years of N-S Control	6.5	7.8	8.2	9.4
Years of Fuel to Bring to Geostat Orbit 1° ($.9^{\circ}$/year)		4.5		4.5
** Remaining years of Fuel		3.3		4.9
Years of Drift in Beost. Orbit	2.2		2.2	
*No. 1 Out of Fuel and Drift	8.7		10.4	
*No. 2 Out of Fuel and Drift	7.5	9.8	9.1	11.4

* Years from first launch
** Used up in inclined orbit

Table 6.4 WEIGHT BUDGETS FOR 3-AXIS SPACECRAFT

	Mission 1		Mission 2	
	1° Incl.	7° Incl.	1° Incl.	7° Incl.
Apogee Motor Fuel and Consummables	901.1	859.2	901.1	859.2
Initial on Orbit Fueled S/C	1023.9	1065.8	1023.9	1065.8
Total Bus Weight (Dry)	600.6	600.6	600.6	600.6
Communication Payload	200	200	200	200
Reaction Control Propellant	223.3	265.2	223.3	265.2
Propellant for all but N-S Control	57.9	57.9	57.9	57.9
Fuel for N-S Control	165.4	165.4	165.4	207.3
Years of N-S Control	8.2	10.1	8.2	10.1
Years of Fuel to bring to Geost-Orbit 1° (.9°/year)		6.7		7.8
Remaining years of Fuel		3.4**		2.3^x
Years of Drift in Geost. Orbit	2.2		2.2	
* No. 1 out of fuel and drift	10.4		10.4	
** No. 2 out of Fuel and Drift	7.6	12.1	variable	uses no N-S fuel

* Years from first launch
** Used up in inclined orbit
*** Available after transfer to geostationary orbit

Thus, when it is transferred to geostationary orbit the fuel load is a fixed quantity no matter how long after launch the transfer is made. When measured in years after launch, the time when this spacecraft runs out of fuel is uncertain but the probability of having service at any time can be evaluated.

6.4 Probability of Continuous Service

In order to properly compare the two proposed missions, it is necessary to calculate the probability that service is maintained utilizing the geostationary and the inclined orbit spacecraft. There are two services that are being provided by the two spacecraft, the primary service in geostationary orbit and extended northern service. The satellite in geostationary orbit is devoted to providing the primary service while the inclined orbit satellite is required to provide back-up to the geostationary satellite as well as supplying the extended northern service. Thus the two services are competing for the use of the inclined orbit satellite.

The operational life of the system can be divided into a number of periods. For mission 1 it is as follows. It is considered that the first spacecraft is launched into geostationary orbit and provides a pilot operation for a period of two years. This is called the first period of operation. At the end of two years, the second spacecraft is launched into the inclined orbit. In the event that the first satellite failed during the first two years the second satellite will be launched into geostationary orbit to replace the failed satellite. The second period of operation begins after the second spacecraft is placed in the inclined orbit and continues as long as full back up is available. The third period begins when the inclined orbit spacecraft no longer has sufficient fuel to bring it to the geostationary orbit and continues for the time it takes to drift through 1° tolerance in geostationary orbit. The fourth period continues until the first spacecraft would run out of fuel if it had continuously operated since launch.

For mission 2 only three periods are recognized. The first period is the 2-year pilot operation, the same as for mission 1. The second period continues until the inclined orbit satellite would expend its fuel if it had been transferred to geostationary orbit in the first days after launch. During the second period the geostationary service is fully backed up by the one in the inclined orbit. During the third period there is only partial back up as there is a finite probability that the inclined orbit satellite has been placed into geostationary orbit and has already expended its fuel. The third period continues until the first spacecraft launched would run out of fuel. At that time there is still a probability that the inclined orbit spacecraft remains available for geostationary service. However it would not be transferred at this time as the end of period 3 is a predictable event and plans would be made for launching a replacement.

An exact solution of the reliability of service would require a consideration of different failure rates for the two transponders, stand-by failure rates and variable rates of fuel expenditure; it is beyond the scope of this study. However, an approximate solution can be derived which shows the relative performance with different inclinations and for the two missions.

For this purpose the following assumptions are made:

a) The probability of survival of a single spacecraft is taken as exponential.

b) The failure probabilities of the two spacecraft are assumed equal even though different equipment is in service in each.

c) Station-keeping fuel expenditures are considered equal for the two satellites in mission 1 even though some saving could be effected by controlling only the inclination and not the change in right ascension.

d) No station-keeping fuel is used by the inclined orbit spacecraft in mission 2 even though some inclination drift occurs.

e) The failure rate of a single spacecraft is such that the probability of survival at 7 years is 0.7.

Let the probability of spacecraft survival be P_1 for the first satellite launched into geostationary orbit and P_2 for that launched into the inclined orbit where

$$P_1 = e^{-\lambda t}$$

$$P_2 = e^{-\lambda(t-2)}$$

$$\text{and} \quad e^{-\lambda 7} = 0.7$$

The probability of maintaining service will be designated by R_G for the geostationary service and R_N for the extended northern service. For period one, both missions, only one spacecraft is in orbit and $R_G = P_1$ and $R_N = 0$. For period two, both missions, the geostationary service is fully backed up and R_G is the probability that both spacecraft have not failed. Thus

$$R_G = 1-(1-P_1)(1-P_2) = P_1 + P_2 - P_1 P_2$$

Also there is extended northern service only if both spacecraft are operating, thus

$$R_N = P_1 P_2$$

During the third period the two missions differ. For Mission 1 the probability R_G consists of the probability P_1 that the first spacecraft survives plus the probability that it failed before the end of period two, times the probability that the second spacecraft still survives.

$$R_G = P_1(t) + P_2(t)\ (1 - P_1(T))$$

where T is the time to the end of period two measured from first launch. Similarly the probability of having extended northern service R_N is

$$R_N = P_1(T)\ P_2(t)$$

During the fourth period the satellite launched into inclined orbit will be out of fuel in geostationary orbit although it would still have fuel if it remained in inclined orbit. The probability of having service is $P_G = P_1$ and $P_N = P_1$ and $P_N = P_1\ (T)\ P_2(t)$.

For mission 2, period 3 is different. It is equal to the probability that spacecraft one is fully backed up reduced by the probability that the inclined orbit satellite has been transferred and has expended its fuel, but hasn't otherwise failed. This is

$$R_G = P_1 + P_2\ (1-P_1-P_1(2)(1-P_1(t-T)))$$

For mission 2 the inclined orbit satellite is always liable to be transferred to geostationary orbit so the probability of having extended northern service is unchanged from period 2. A summary of the equations is given in table 6.4.

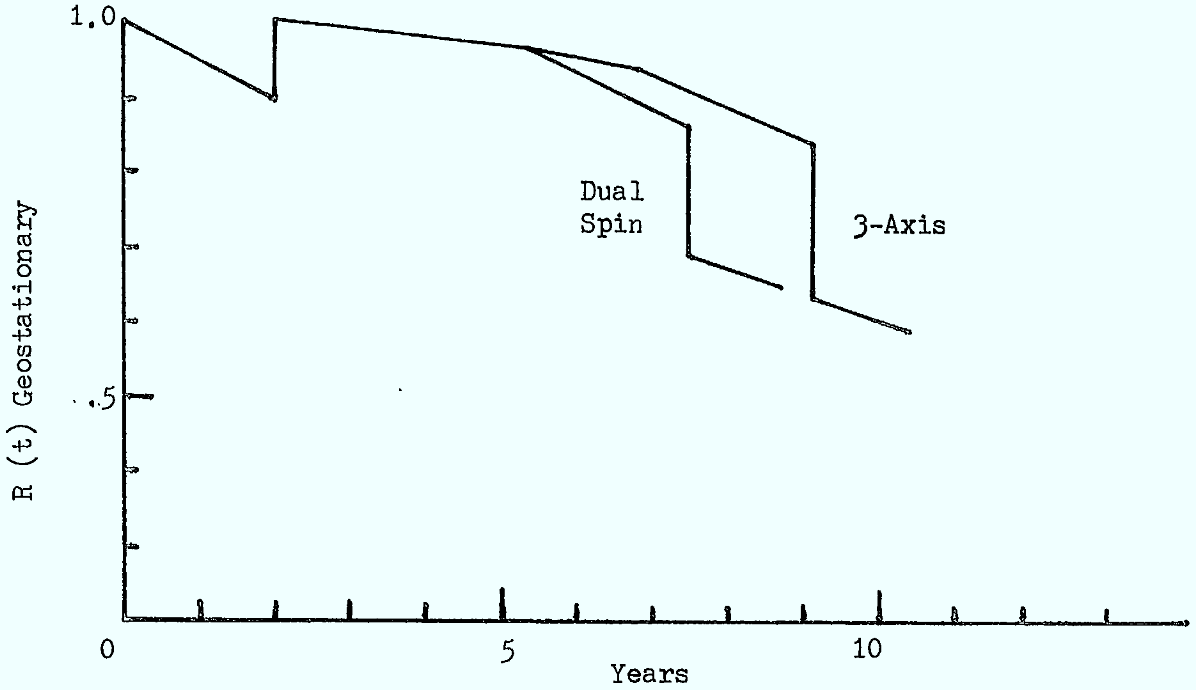
TABLE 6.5
EQUATIONS FOR RELIABILITY CALCULATIONS

Period R_G		
	Mission 1	R_N
1	$R_G = P_1$	$R_N = 0$
2	$R_G = P_1 + P_2 - P_1P_2$	$R_N = P_1P_2$
3	$R_G = P_1 + P_2(1-P_1(T))$	$R_N = P_1(T)\ P_2$
4	$R_G = P_1$	$R_N = P_1(T)\ P_2$
	Mission 2	
1	$R_G = P_1$	$R_N = 0$
2	$R_G = P_1 + P_2 - P_1P_2$	$R_N = P_1P_2$
3	$R_G = P_1 + P_2(1-P_1-P_1(2)(1-P_1(t-T)))$	$R_N = P_1P_2$

Using these equations, the probabilities of maintaining service have been calculated for various mission profiles for the purpose of making a comparison between different configurations or different missions and of selecting an optimum. The first comparison, shown in sigure 6.3, is between the dual spin and 3-axis rigid panel configurations (Table 6.2) launched into a five degree inclination mission 1-type orbit. The effect of having more fuel on the 3-axis spacecraft is to increase the probability of having geostationary service and to reduce the probability of having extended northern service in the later years of operation.

The second comparison, figure 6.4, is between a five degree mission 1 and a seven degree mission 1 profile using a 3-axis configuration. It is evident that going from five to seven degrees decreases the probability of having geostationary service and increases the probability of having extended northern service in the later years of operation. It also improves the coverage and extends the fraction of the day over which ENS is available.

The final comparison, figure 6.5, is between mission 1 and mission 2 both for seven degrees inclination and for a 3-axis configuration. It is evident from figure 6.5 that there



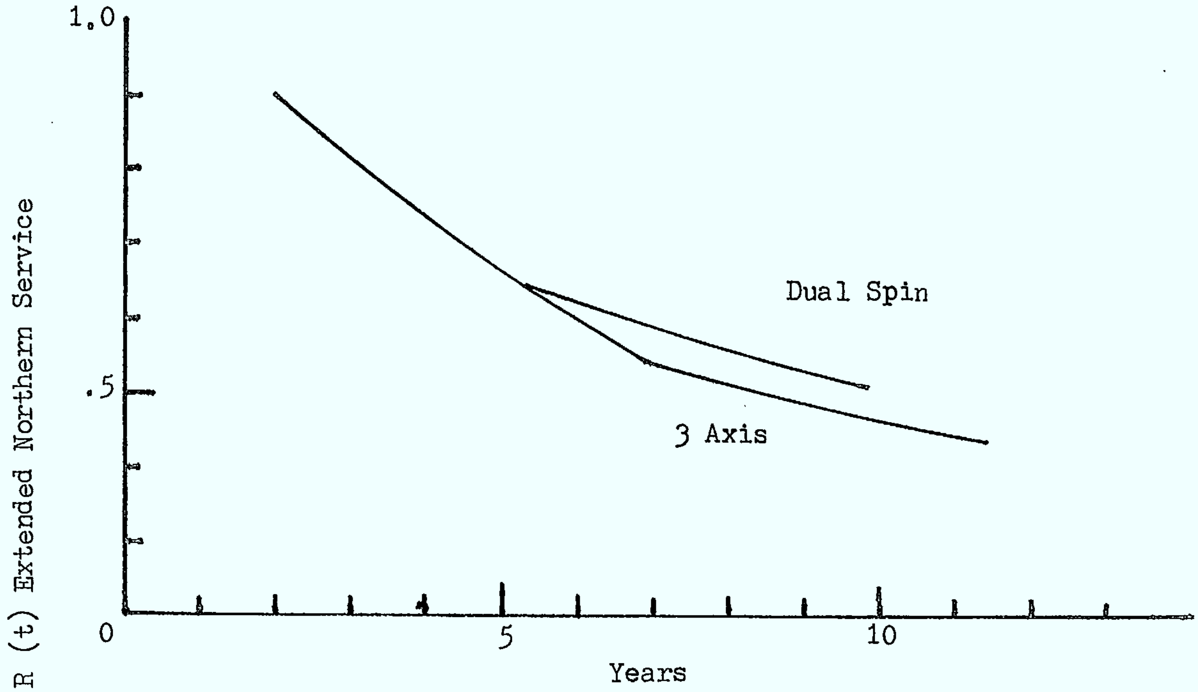
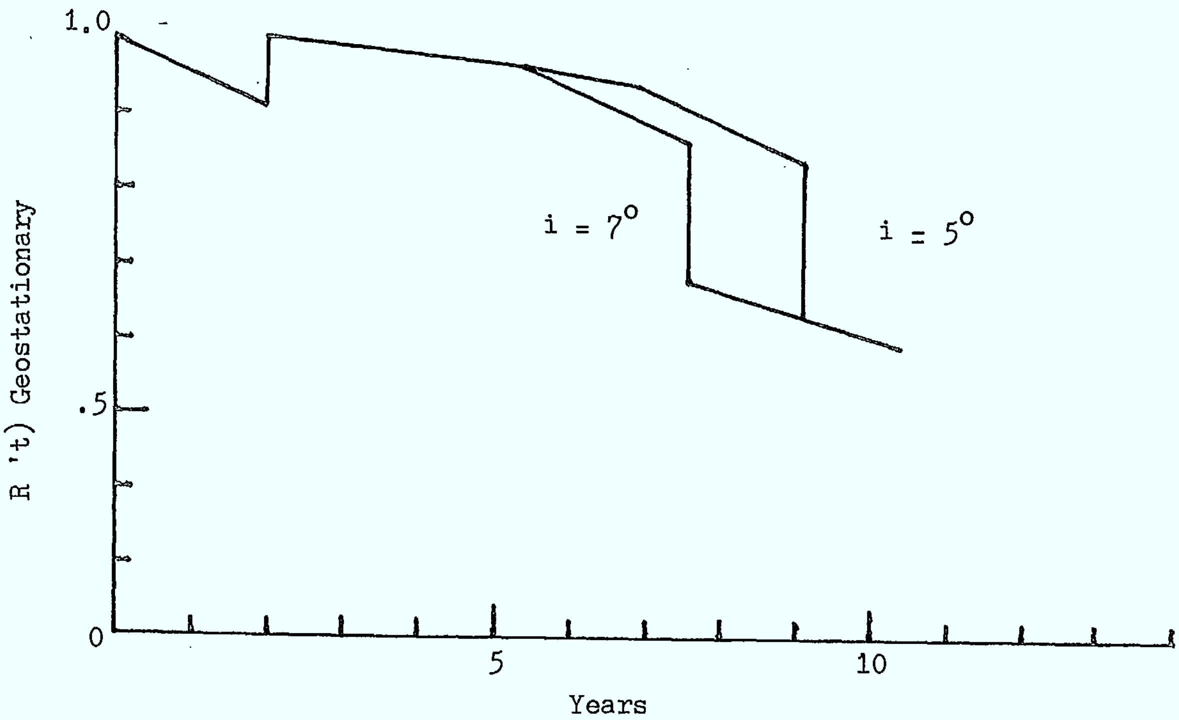


Figure 6.3 Probabilities of maintaining service for 5° inclination and 270° right ascension.



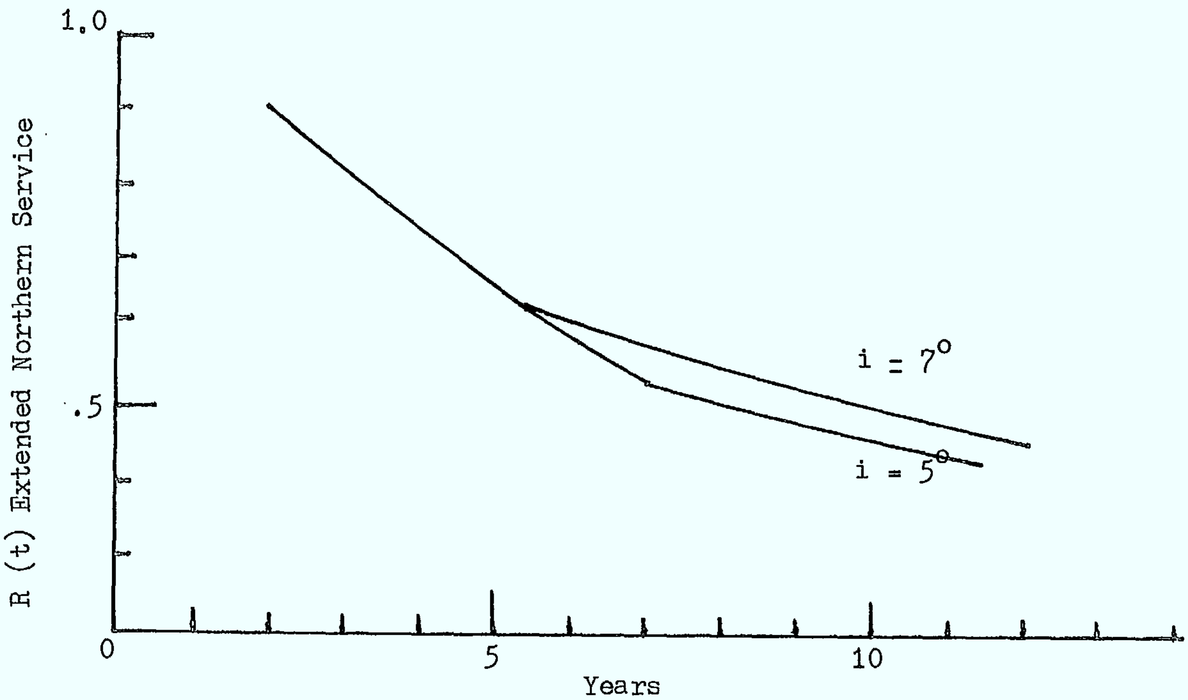
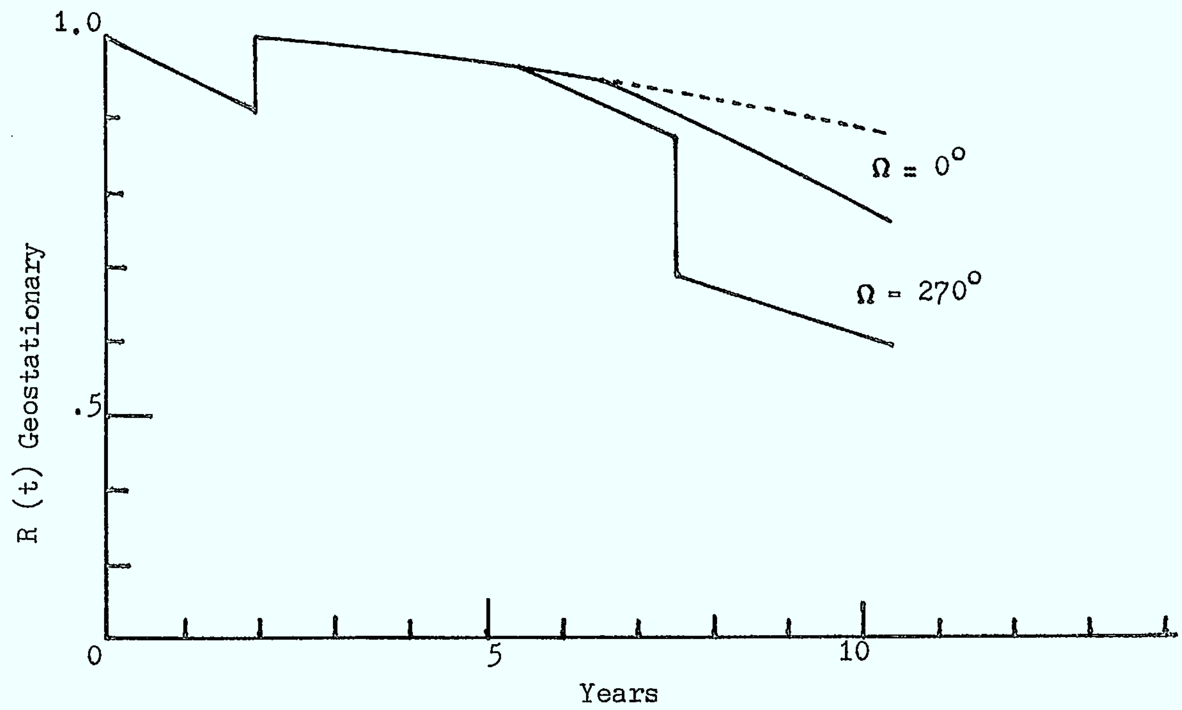


Fig. 6.4 Probabilities of maintaining service for three-axis spacecraft in orbits with right ascensions of 270° and inclinations (i) of 5 and 7°



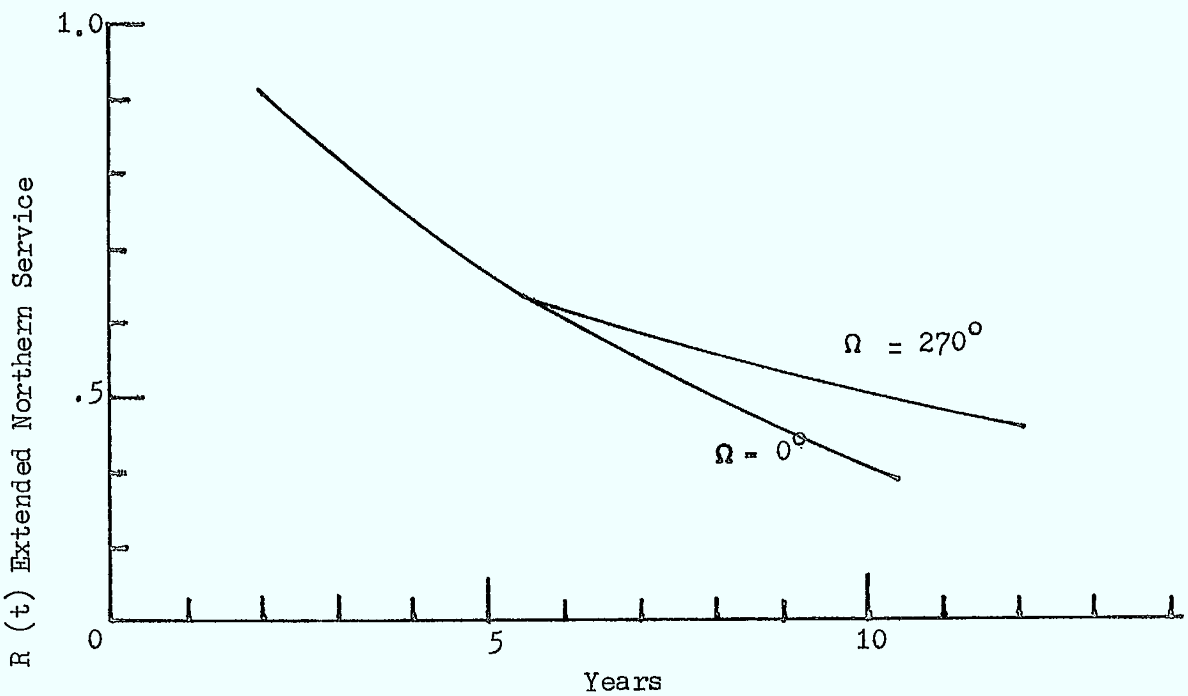


Fig. 6.5 Probabilities of maintaining service for three-axis spacecraft in orbits with 7° inclinations and right ascensions (Ω) of 0 and 270°.

is very little difference out to 7.5 years, but mission 2 gives a significantly higher R_G beyond this time, while R_N is lower for mission 2.

Of particular interest is the loss to the primary service caused by making use of the spare spacecraft for extended northern service. The probability of maintaining the primary service R_g for the case where the spare is kept in the same slot as the operational spacecraft is shown by the dotted line in figure 6.5. Thus there is no loss to the primary service until about six years after first launch. After six years there is a drop below the fully redundant system but by this time the third spacecraft would have been launched and would keep the probability of having the primary service at a high level. In exchange for a slight loss to the primary service after the sixth year there is a probability of having ENS which exceeds 0.5 during the four years that the primary service if fully backed up.

6.5 Discussion of Third Spacecraft Launch

In addition to the two spacecraft in orbit it is anticipated that a third spacecraft would be build and held in reserve on the ground to be launched at a later date. It is the purpose of this section to discuss some of the considerations in selecting a time for the third launch in order to optimize the overall performance of the system. In addition to selecting the time for the launch it is necessary to decide whether the satellite should be placed in geostationary or inclined orbit. Finally the decision about timing and orbit selection may differ for the two missions discussed in section 6.2 and depend upon what happens to the spacecraft or the traffic patterns during the first few years that the system is in operation. The events of the early years of operation can influence the launch because the decision about orbit inclination can be delayed until the spacecraft is being prepared for launch.

An examination of figure 6.5 shows that the primary service in geostationary orbit is maintained at a reasonable level of probability to about 6 or 7 years from the initial launch while the northern service starts initially at a probability of about 0.9 and drops rapidly. At 4.5 years after first launch the probability of having ENS has decreased to 0.7 and it seems unlikely that the third launch could be delayed beyond that time.

The question remains; into which orbit should the third spacecraft be launched. It would seem that it should be launched into the inclined orbit since the probability of having the ENS is lowest. But in fact, for a right ascension of 270° (mission 1) if both spacecraft are still operating at the time of the third launch, the launch should be into the geostationary orbit. The reason for this is that the excess hydrazine fuel available on the inclined orbit launch is not sufficient to take the spacecraft to the geostationary orbit. In other words, a spacecraft will have more fuel and and a longer life in space if it is launched directly into geostationary orbit than if it is launched into inclined orbit and then transferred to geostationary orbit.

For a right ascension of zero degrees, there is no need for inclination correction and it is possible to park the third satellite in the inclined orbit for many years with the expenditure of very little fuel. In this case, the life in space of the third satellite will be longer provided the transfer in orbit does not occur in the first two or three years.

An attractive possibility for the third spacecraft when mission 2 is used, is to launch into an intermediate orbit with an inclination of about 4°. The exact orbit would be the subject of an optimization study during implementation. Larger inclinations are indicated by the fact that orbit drifts are slower and fuel loads are higher for a higher inclination. On the other hand, if it is eventually used to replace a failed spacecraft in inclined orbit, the only fuel expended is that required to transfer it to the inclined orbit position and any parking orbit close enough to be transferred would be satisfactory. A possible strategy for the third spacecraft is shown in figure 6.6.

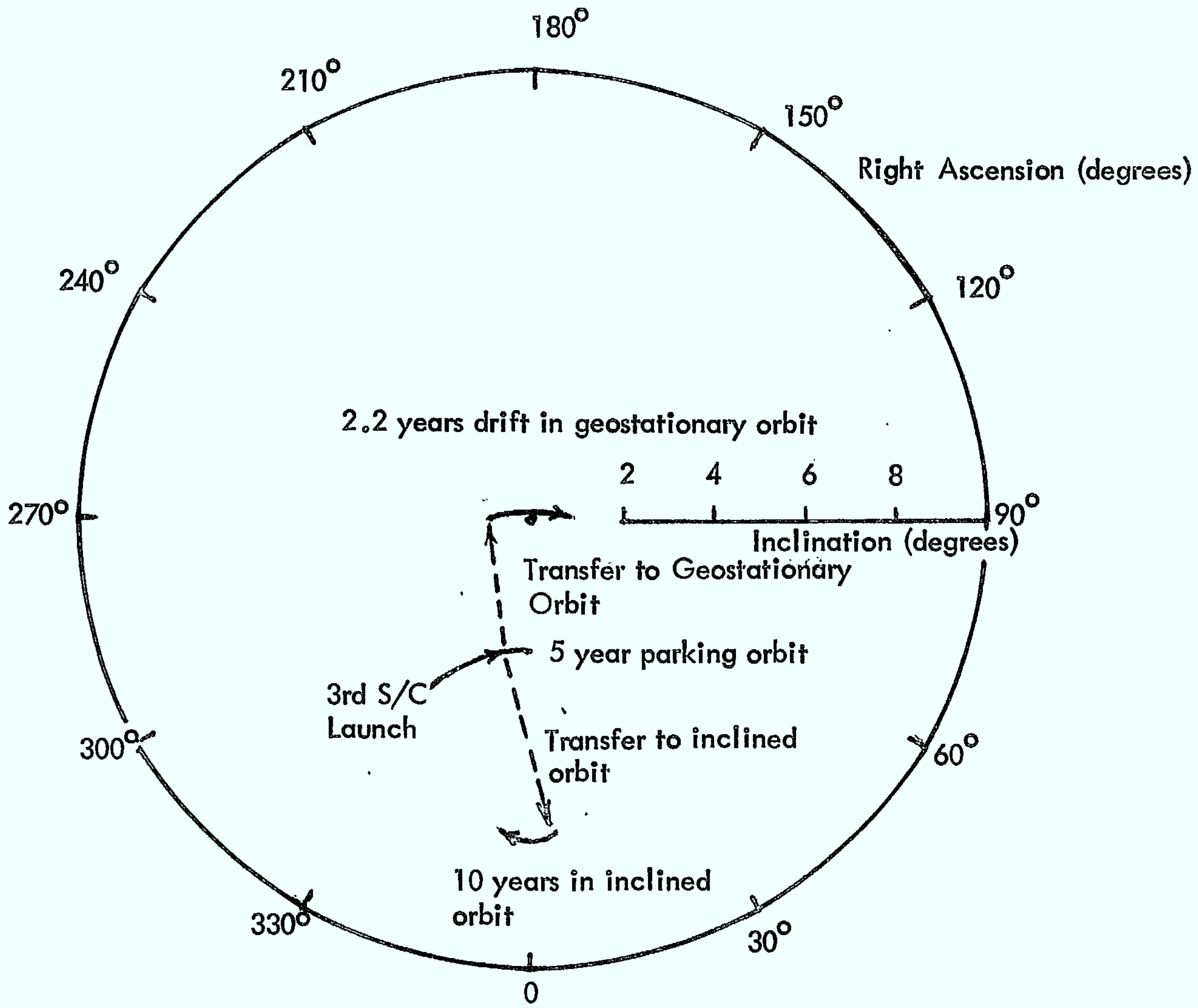


Figure 6.6: Mission Strategy for Third Spacecraft

7. SUMMARY AND CONCLUSIONS

It has been shown that a minimum level of extended northern service can be obtained with an orbit inclination of seven degrees. Two right ascensions for the 7° orbit have been investigated. A zero degree right ascension places the satellite in a stable or near stable orbit such that no fuel need be expended to maintain the orbit. A right ascension of 270° places the satellite in a position such as to minimize the fuel required to bring the inclined orbit spacecraft into the geostationary orbit.

The probability of maintaining the primary service has been compared for these two right ascensions and the zero degree right ascension shown to provide the highest probability of service. The performance has also been compared with a system that does not provide ENS. It is shown that there is no penalty to the primary service in the first six years of operation. The loss after six years would be compensated for by the launch of a third spacecraft. Considerations affecting the launch of the third spacecraft are discussed and it is shown that it should be launched into an orbit with an intermediate inclination of about 4 degrees.

The results of the study show that, with an orbit inclination of 7°, extended northern service can be provided for a reasonable portion of each day and with a probability that exceeds 0.5 for the first four years after the inclined orbit spacecraft is launched. This extended northern service is provided with only a slight penalty to the primary service which occurs beginning about six years after the first spacecraft is launched. This penalty occurs at a time when, in all probability, the third spacecraft would have been launched, provided additional back-up to the primary service. Thus the moderatly inclined orbit concept is a viable means of providing some service to the areas of the arctic not serviced by a geostationary satellite.

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APPENDIX

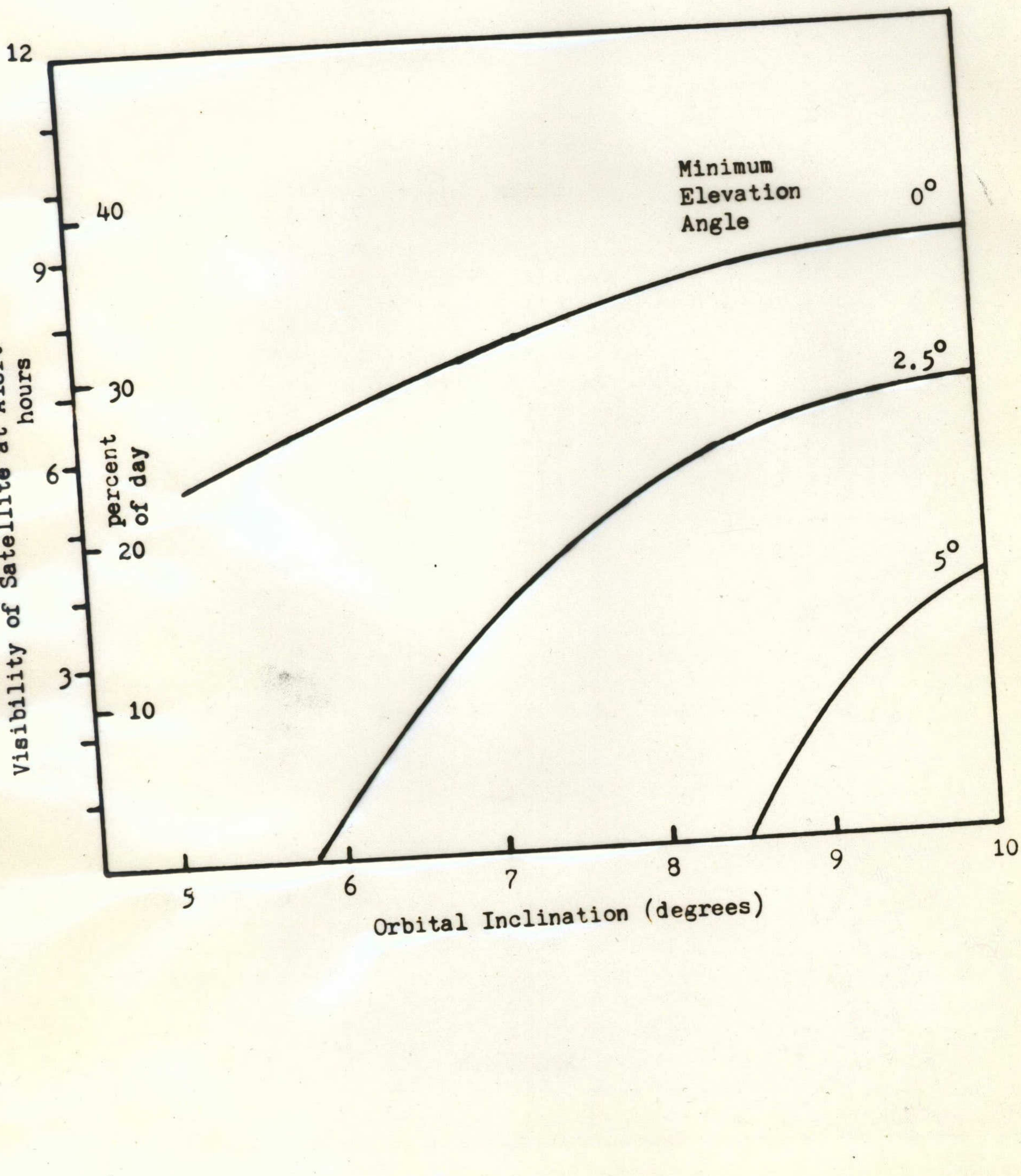
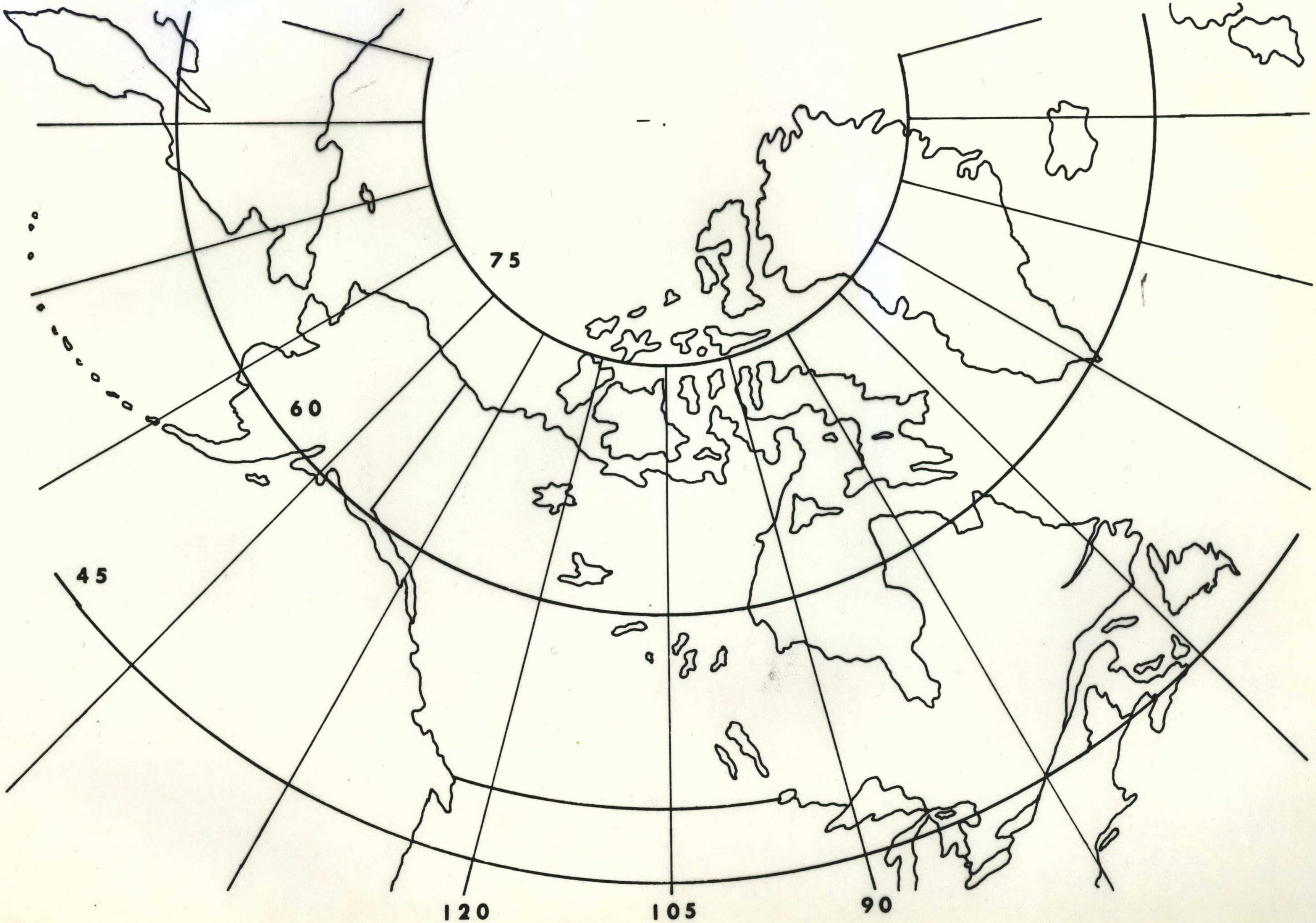


Fig. 3.7 Satellite Visitility at Alert on Ellesmere Island
with Subsatellite Longitude of 97°W

Inclination	5°		7°		10°	
Elev. Angle*	0°	5°	0°	5°	0°	5°
% Visibility — at Alert	25	0	30	0	35	10
— at Resolute	100	50	75	50	70	50

* Satellite longitude — 100°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