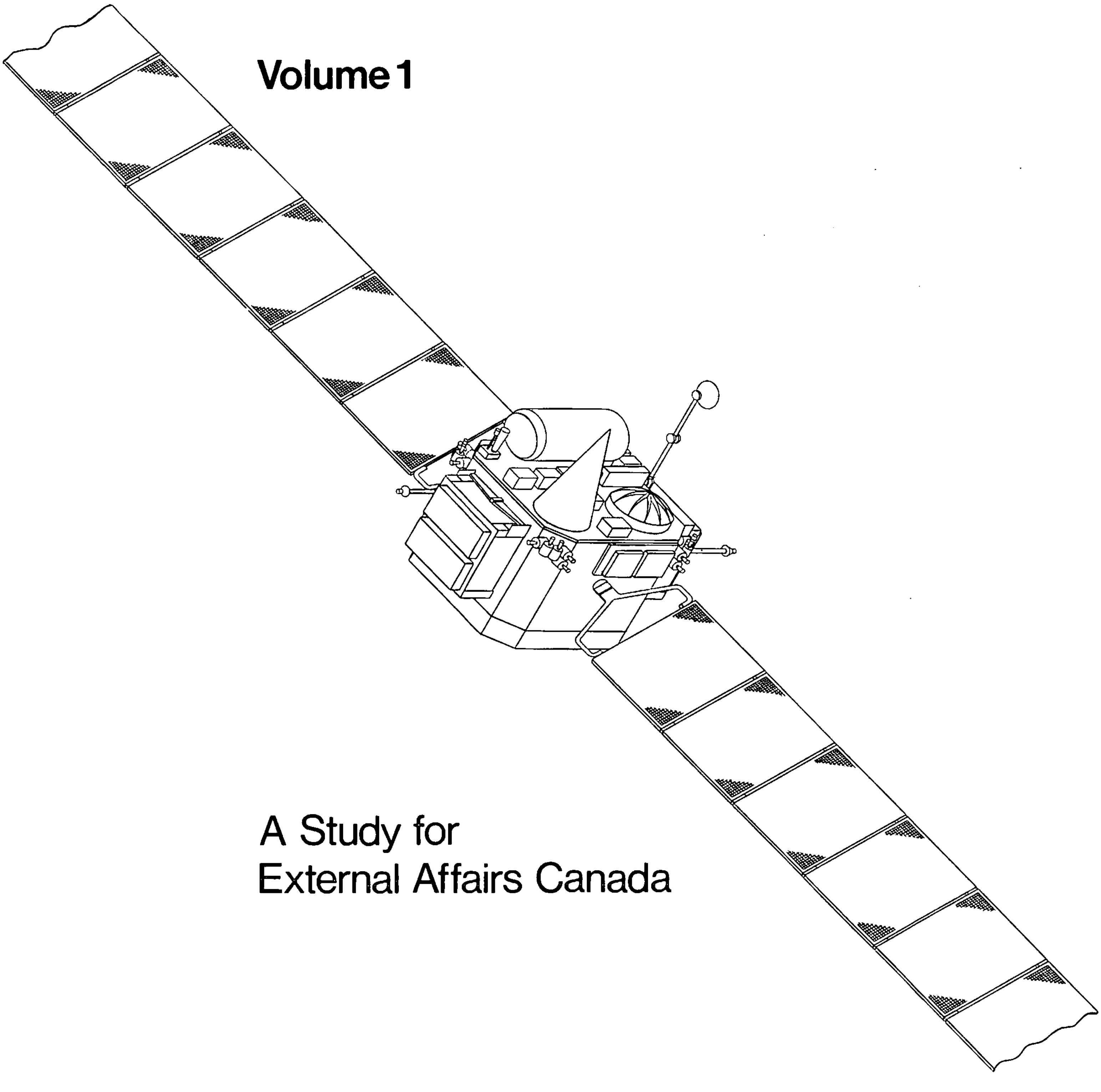
PAXSAT 'A'

Space Based Remote Sensing: Space-to-Space



Volume 1

A Study for
External Affairs Canada

Spar Aerospace Limited

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PAXSAT 'A'

SPACE-BASED REMOTE SENSING

SPACE-TO-SPACE

VOLUME I REPORT

REPORT NO.

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FOREWORD

This constitutes the final report under Contract No. 21ST-08011-4-2297, Serial OST84-00133, titled "Paxsat Concept for Arms Control and Disarmament Verification in Outer Space".

The contract was carried out by the Satellite and Aerospace Systems Division of Spar Aerospace Limited, with a major subcontract to Philip A. Lapp Limited who in turn were supported by the Canadian Center for Arms Control and Disarmament.

The report is presented in two volumes. Volume I is the main body of the report compromising of sections 1 through 10.0. Volume 2 is the appendix of the report and contains Appendices A through D.

The material on Space Assets and Weapons Analysis presented in Volume 1, section 2.0 and, on the operational aspects of the Paxsat concept presented in section 4.0 of this report, are the effort of Philip A. Lapp Limited. Additionally, the resources of Philip A. Lapp Limited generated the material on the ground based and space based optics capabilities presented in section 6.0. Section 3.0, the Political/Legal context for a Paxsat type mission is the effort of the Canadian Center for Arms Control and Disarmament. Remaining sections of the report including the Artificial Satellite Log of Appendix A were generated by the Satellite and Aerospace Division of Spar Aerospace Limited.

The contract was monitored for External Affairs Canada by Mr. Ron ClemInson and for Supply and Services Canada by Mr. Louis Cloutier. The monthly reviews and reports were made to an ad-hoc committee of DND, DEA, EM&R and DOC personnel chaired by Mr. J. Ray Marchand of the Interdepartmental Committee on Space.

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1.0 INTRODUCTION

This is the final report on a study for the Canadian Government, Department of External Affairs, regarding the feasibility of a spacebased remote sensing system designed to determine the presence of weapons in space.

The Paxsat A System Concept is based on the supposition that a properly configured set of observations in space can determine the function of an unknown satellite to an acceptably high degree of confidence, such that it can contribute to the determination and control of the presence of weapons in space.

The present study extends earlier studies in this field [Refs. 1,2] and is intended to develop a data base in respect to the Paxsat concept from which the Canadian Government may assess other similar related concepts or, develop a Canadian negotiating position in respect to an international forum. The study thus addresses three principle questions:

(a) Can space observations determine the role or function of an object in space?

(b) Are there one or more political/international agreements or treaty contexts in which observations could or would be carried out?

(c) Would the observational requirements and the political restraints of a governing treaty permit a viable Paxsat mission and design spacecraft?

The report discusses the concept and its implications under eight principal topics. Section 2.0 outlines the present distribution of assets in space, both civil and military, and considers the prospects for weapons in space.

Section 3.0 discusses the political considerations affecting an arms control agreement for outer space and suggests the limitations under which a Paxsat system might have to operate.

1.0 INTRODUCTION (Continued)

Based on the reality, options and limitations of the previous sections, Section 4.0 develops a political/technical scenario and plausible operational profiles which are analyzed in Section 5.0 as to their demands on the system performance and resources.

The basic sensor payload of the spacecraft is discussed in section 6.0, while the supporting subsystems and overall spacecraft concept are discussed in section 7.0. A typical program plan associated with this type of mission is presented in section 8.0.

The study conclusions are summarized in section 9.0.

Section 10.0 lists the references consulted dùring the course of this study.

Detailed data bases and analyses associated with various aspects of the report are appended in a separate volume.

2.0 THE OUTER SPACE SITUATION

2.1 Introduction

A weapon in space can have as its objective the destruction of, or the damage to another orbiting object, or the destruction of, or damage to targets situated on the earth. The former weapon's objective is accomplished in the space environment while the latter weapon's objective may be accomplished either directly from space or subsequent to a re-entry through the earth's atmosphere. The current debate over the Strategic Defense Initiative (SDI), a spacebased ballistic missile defense concept more widely known as the 'Star Wars' concept, marks a third function for a space weapon: namely the destruction of suborbital ballistic missiles during flight.

The review of weapons in this section of the report is developed in the context of a Paxsat A system operating to verify a treaty agreement with the verification taking place in space. Legitimate candidate weapons for Paxsat investigative scenarios are weapons placed in stable orbits with the aim of being used at some future time. Weapons like the Fractional Orbit Bombardment System (FOBS) developed by the Soviet Union in the late 1960's, the current generation of antisatellite weapons under development and testing within the Soviet Union and the United States, and the familiar strategic and tactical ballistic missiles of the current day, are not candidate weapons in the Paxsat scenario. These weapon systems spend far too limited a portion of their flight time in the space domain for space-to-space investigation.

In the case of the FOBS, a nuclear warhead can be fired into an orbit of 160 km altitude and then slowed down by retro-rockets to re-enter the earth's atmosphere and fall on the target before the completion of its first orbit. This approach makes it possible to attack Western targets by the 'back door', travelling three quarters of the way round the world via the South Pole, instead of the traditional 30 minute ballistic missile trajectory over the North Pole. Such a roundabout trajectory would last approximately one hour.

2.1 Introduction (Continued)

The two antisatellite weapons (ASAT's) currently under development in the Soviet Union and the United States are also not verifiable in the Paxsat scenario because they are not in space for a sufficient length of time to enable an investigation to be undertaken. The Soviet Union has successfully tested and put into operation a ground launched weapon while the United States is currently testing an air launched ASAT weapon. Since these weapons seek out and engage targets within hours or even minutes of their launch, there is no question of their presence being be verified by a Paxsat spacecraft based in space. Verification of these weapons would have to be done while the weapons were still on the ground. However, it is envisioned that the next, or second generation of ASAT's would employ alternative methods to destroy or disable the targets from the current impact method, and be based in stable orbits to carry out their mission. The Paxsat system would be attuned to the verification of these types of weapons in space.

The review of weapons in space conducted in this section of the report is presented in three parts. Section 2.2 addresses the targets in space and the space weapons likely to be deployed against them. Section 2.3 addresses targets on the earth and weapons likely to be deployed against them. Section 2.4 summarizes the preceding analyses to tabulate the threats relative to the earth and space assets, and defines the weapons sytems most likely requiring verification by the Paxsat system.

2.2 Space-to-Space Weapon Situation

No known operational spacebased weapon system for space-to-space operation has yet been deployed in space. Thus, there is still a considerable amount of uncertainty as to how these systems would be configured for optimal performance. What is known about the situation in space however, is the location and distribution of potential targets in space.

2.2 Space-to-Space Weapon Situation (Continued)

Potential targets for a spacebased space-to-space weapons system can be divided into two distinct classes based upon the orbital parameters of these targets. These classes are:

(a) Sub-orbital projectiles

(b) Orbiting artificial satellites.

The sub-orbital class of targets encompasses such vehicles as Intercontinental Ballistic Missiles (ICBM's), Submarine Launched Ballistic Missiles (SLBM's) and Fractional Orbit Bombardment Delivery Vehicles (FOB's). These targets spend but a brief time in the space environment ranging from approximately 8 minutes as in the case of SLBM's to approximately 60 minutes for FOB's and do not in any case fully complete an orbit about the earth. It is this class of targets that the proposed Strategic Defence Initiative (SDI) is addressed.

The Strategic Defense Initiative as proposed by president Reagan in March of 1983, is generally envisioned to be complex system employing a series of orbiting satellites using exotic technologies to shoot down ballistic missiles during their flight. A primary emphasis has been placed upon disabling the missiles during the boost phase of their flight and a variety of technologies are proposed for this concept, including chemical rockets, hypervelocity rail guns, lasers and particle beams. Current research and development activities appear to be slanted towards directed energy weapons like lasers and particle beams for the boost phase intercept portion of the layered defense system. Technical and economic concerns over the viability and effectiveness of the concept is the current debate, since ICBM's may be 'hardened' to counteract the destructive mechanisms of the directed energy weapons. Even if the system was to fail against the robust missile targets, the SDI concept would make an effective antisatellite weapon since satellites are much more fragile than missiles and are far easier to target. In fact, it is regarded that, "virtually any putative BMD (Ballistic Missile Defense) system will be an effective

2.2 Space-to-Space Weapon Situation (Continued)

ASAT long before it achieves any significant ABM (Antiballistic Missile) capability". [3] Additionally, given the intrinsic vulnerability of spacebased systems, the domination of space by ASAT measures would be a prerequisite to the reliable ballistic missile defense of an entire nation. Thus, the more immediate concern for the placement of weapons in space are weapon systems designed to carry out antisatellite activities and it is in this direction that the report ensues.

Since the launch of Sputnik I by the Soviet Union on October 4, 1957, to the end of 1983 over 2,500 known successful space launches have occurred, hurtling over 14,400 objects, consisting of artificial satellites, rocket fairings, expent rocket casings, etc. into the domain of space. [4,5]. Many of these orbiting objects have been placed into low earth orbits where the drag exerted by the earth's rarified atmosphere has resulted in their firey return such that approximately 5,000 objects remain in space today.

A database of all known satellite launches compiled from a variety of sources [3-17] for the period 1980 to 1983 inclusive, indicates the intensity of space activities by the world's nations in recent history. Appendix A documents this satellite listing. During this period, over 476 successful space launches for an average annual rate of 119, have placed a total of 585 artificial satellites into outer space. Table 2-1 illustrates this level of space activity. Of these artificial satellites, the USSR and the US are the predominant owners accounting for 80% and 13% respectively. Approximately 70% of all satellites launched during the period serve a military function with approximately 80% of all Soviet satellites serving military roles and approximately 50% of all American satellites performing military activities. Certain of these satellites while launched for military use, serve a double purpose as part of the arms control verification process between the Superpowers. Therefore, certain military uses of space are essential from the arms control aspects. Nevertheless, as increasing numbers of single-purpose military assets are placed into orbit, an increasing militarization of space will result. But, as mentioned

TABLE 2-1 CIVILIAN/MILITARY SATELLITE DISTRIBUTION LAUNCHED 1980-1983 AD

COUNTRY	YEAR 1980	1981	1982	1983	TOTAL
UNITED STATES					
Military	11	7	8	13	39
Civilian	4	10	9	13	36
SUBTOTAL	15	17	17	26	75
SOVIET UNION					
Military	90	100	101	92	383
Civilian	20	23	18	24	85
SUBTOTAL	110	123	119	116	468
NON-SUPERPOWER					
Military	0	0	1	1	2
Civilian	4	17	6	13	40
SUBTOTAL	4	17	7	14	42
TOTAL WORLD					
Military	101	107	110	106	424
Civilian	28	50	33	50	161
SUBTOTAL	129	157	143	156	585

TABLE 2-2 TYPES OF ANTISATELLITE TARGETS

SATELLITE FUNCTION	DATABASE FUNCTIONAL ABBREVIATION (APPENDIX A)	MILITARY EMPLOYMENT	CIVILIAN EMPLOYMENT
Antisatellite	ASATT	X	
Communications	COMMU	X	X
Early Warning	EARLY	X	
Electronic Intelligence	ELINT	X	
Experimental (Technology Development)	EXPTL	X	X
Earth Resource Monitoring	ERSAT		X
Interplanetary	INTER		X
Manned Missions	MAN'D	X	X
Meteorological	METEO	X	X
Navigation	NAVIG	X	X
Ocean Surveillance	ORSAT	X	
Radar Calibration (Minor Military)	RADAR	X	
Photo Reconnaissance	RECON	X	
Scientific (Pure and Applied)	SCIEN	X	X
Spacebased Weapons	WEAPO	X	
Targeting	TARGE	X	

2.2 Space-to-Space Weapon Situation (Continued)

previously, there are no known and currently deployed weapons in space, such that the present concern is over the weaponization of space.

Though they vary in importance from trivial to strategic as targets, all satellite systems, civilian and military are, by virtue of their very presence in space, potential targets for a weapon. Table 2-2 lists 16 types of application satellites ranging from conventional communications satellites through the sophisticated surveillance satellites to the most esoteric ASAT weapon platforms. Figure 2-1 illustrates the distribution of the satellites launched between 1980 and 1983 according to these functional classifications.

The first generation antisatellite weapons developed thusfar are to be based upon the earth. The currently operational Soviet system requires a large booster rocket to lob its kill vehicle into a phasing orbit about the earth. The kill vehicle of this system can require up to two complete earth orbits to align itself with the target and terminate its mission with a close proximity explosion. Thus, if an American antisatellite system were to have a response time on the order of minutes, the Soviet system could itself become a target of an antisatellite system. This is in fact the apparent design philosophy of the American antisatellite system currently undergoing testing. Launched from fighter aircraft, the smaller American antisatellite weapon is much more versatile than its Soviet counterpart. Time from launch to impact of its target is on the order of minutes since the kill vehicle directly ascends into the flight path of its intended vehicle. Consequently, even first generation, ground based, antisatellite weapons are targets for themselves.

Early warning satellites can, by recognizing the infrared radiation from an ICBM launch, provide about 30 minutes warning of an attack. This effectively doubles the time available from ground based radars to make crucial decisions. It has been postulated that if early warning satellites can be disabled quickly, a nation can be rendered blind, being unable to detect launches during the early phases of a confrontation. However, it is also argued that such an attack on early warning

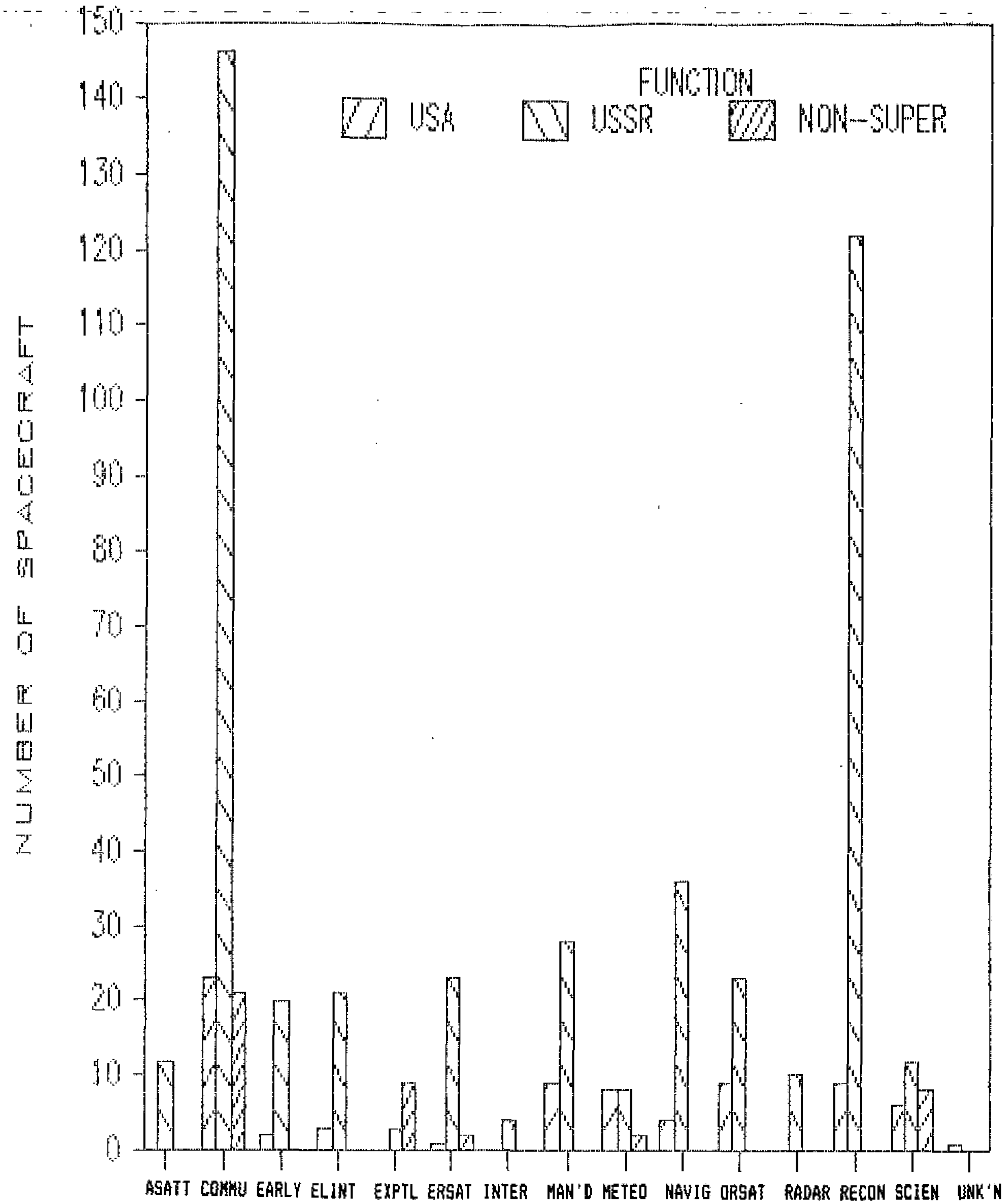


FIGURE 2-1 FUNCTIONAL DISTRIBUTION OF SATELLITES LAUNCHED 1980-1983 AD

2.2 Space-to-Space Weapon Situation (Continued)

satellites would merely serve notice of an impending nuclear strike. Nontheless, early warning satellites remain potential targets for antisatellite weapons in strategic war game scenarios.

Electronic Intelligence satellites (ELINT) are electronic 'ears' recording radio and radar transmissions from areas of military activity. They provide data about missile tests, missile defenses and early warning systems and thus serve an important role in the monitoring of ABM treaty articles. On the darker side of intelligence activities, ELINT satellites may even monitor government and civilian communications providing Communications Intelligence (COMINT) data for which the code cracking computers of the intelligence communities constantly hunger. Thus by their nature, ELINT satellites become potential targets for antisatellites.

Ocean surveillance satellites are satellites designed specifically to monitor military naval activities upon the high seas. To fulfill this function, two types of ocean surveillance satellites have evolved. EORSAT, an acronym for Electronic Ocean Reconnaissance Satellites, operate similar to the passive ELINT satellites listening for the telltale signatures of shipborne radars and communications signals. RORSAT, an acronym for Radar Ocean Reconnaissance Satellites, are active satellites employing radar to detect the presence of ships in all weather conditions. Soviet ORSAT's are of such an effect, that US Naval officials worry that they could facilitate attacks on US ships. Thus, ocean surveillance satellites can be expected to be high priority targets for any antisatellite weapon system.

Photo reconnaissance satellites or 'spy' satellites are major components of a nation's National Technical Means (NTM) providing irreplacable intelligence on the military and strategic activities of hostile nations. Their capabilities are shrouded in secrecy but are hypothesized to be able to discern an object on the order of 15 cm in diameter on the surface of the earth from their low earth orbits [15]. The US maintains three photo reconnaissance systems and the USSR two.

2.2 Space-to-Space Weapon Situation (Continued)

The importance of these systems to the intelligence communities of the superpowers mark them as high priority targets for antisatellite weapons.

A combination of the current surveillance satellite systems, ocean reconnaissance, photo reconnaissance, communications and navigation satellites can provide near real-time data for targeting purposes of associated weapon systems. Future satellites dedicated for real-time targeting are likely to evolve and become an integral part of the weapons system itself. These targeting satellites will utilize sophisticated technologies both to locate itself, and hence, its remotely sensed targets, and to process the data into a form that is immediately useable by the aiming or the guidance portion of the weapon system it supports. Such an exotic system would be a formidable weapon and a high priority target for an ASAT system.

The other satellite applications in Table 2-2 are self-explanatory and will not be discussed further.

The energy required and thus the cost of placing spacecraft in orbit is such that the spacecraft design and its orbit must be highly optimized in terms of its required function. The result of this constraint is that all spacecraft whether scientific, remote sensing, experimental, commercial, or of military application are found in several specific volumes of space defined by orbital parameters. These orbits are illustrated in Figure 2-2. Most application satellites are found in one of the four orbit regimes identified. Notable exceptions are the interplanetary spacecraft who employ particular trajectories to escape from the gravitational pull of the earth. As such, these satellites do not orbit the earth.

The geosynchronous orbit (GEO) is a particular circular orbit above the equator of the earth with a unique feature; the period of the orbit is equal to the period of the earth's rotation about its spin axis. Such a characteristic translates into the fact that there does not exist any relative motion between points on the ground and the orbiting satellite. From a point on the ground then, the satellite appears to remain at one spot in the sky. Thus, receiving stations are greatly

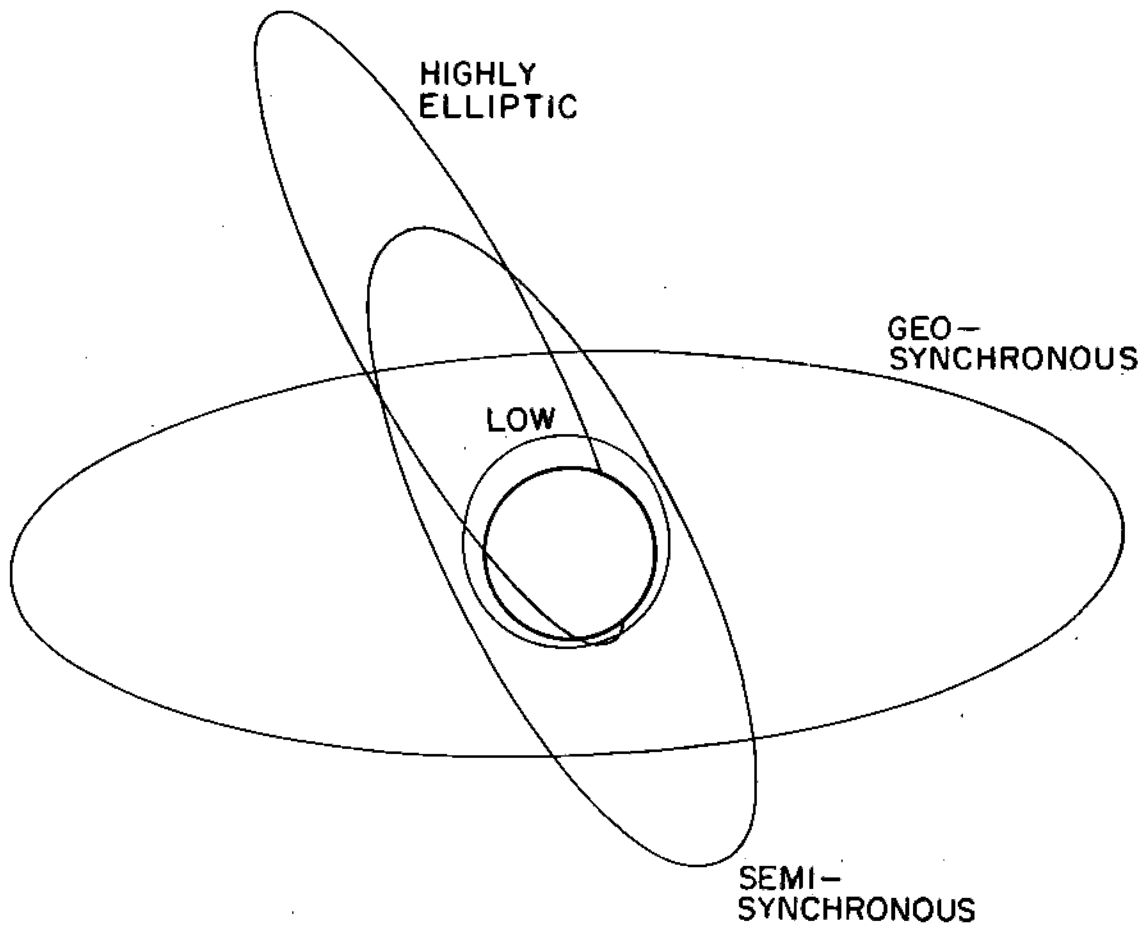


FIGURE 2-2 FOUR MOST UTILIZED ORBITAL DOMAINS

2.2 Space-to-Space Weapons Situation (Continued)

simplified by the fact that they do not need to track the satellite as it crosses the sky. In addition, the high altitude of the orbit enables the satellite to view all but the extreme edges of the hemispherical disk of the earth. Three equally spaced satellites about the earth's equator are required to view all of the earth except for the extreme polar regions.

Because the geosynchronous orbit does not provide clear line of sight to installations in high Arctic regions, an alternative orbit is employed. A highly elliptic orbit inclined at approximately 63° to the earth's equator with its apogee positioned over the Northern Hemisphere, permits 8 or more hours of its 12 hour period to be within a clear line of sight of the North Polar region. An inclination of 63° is critical to the maintenance of the apogee above the Northern Hemisphere, as gravity anomalies caused by the non-sperical shape of the earth tends to disturb the orbit from this optimal alignment. Ground stations in this case need mechanisms to steer the communications antenna as it follows the motion of the satellite in the sky.

Another circular orbit with a 12 hour period is utilized exclusively by navigation satellites. This semi-synchronous orbit is also inclined at approximately 63° to null the effect of the earth's gravitational aberration. A constellation of 6 satellites equally spaced in three such orbits also equally positioned about the earth, enable a number of satellites to be visible to an observer on the ground at any one time. This multiplicity of observable satellites, enables an observer to calculate his position in three dimensions to a high degree of accuracy. The American Global Positioning System (GPS) enables a position fix to be calculated with an error less than 10 meters.

The fourth orbital domain is the range of orbits classified as Low Earth Orbit (LEO). An orbit is defined to be a LEO orbit simply if the altitude of the orbit is less than 3,000 km. However, most satellites of interest to antisatellite weapons in this domain lie between the inclinations of 50° to 105° and altitudes between 160 km and 1,500 km.

2.2 Space-to-Space Weapons Situation (Continued)

Orbits in the LEO domain can also be defined as either 'prograde' or 'retrograde' orbits. Prograde orbits are orbits with inclinations between 0° and 90° while retrograde orbits are orbits with inclinations between 90° and 180°. The distinction is based on the fact that satellites with inclinations greater than 90° rotate about the earth in the direction opposite to the rotation of the earth on its axis, hence, the term retrograde. Conversely, prograde satellites rotate about the earth in the same direction as the earth's rotational motion. The term is of significance only in that there exists a special class of retrograde orbits that are known as sun-synchronous orbits. Because the earth is not a true sphere, gravitational forces cause the orbit plane of a satellite to precess in inertial space. Here inertial space is simply a reference frame to which all motions can be described relative to the orientation of its composite axis system. The precession rate of the orbit depends upon its inclination and altitude above the earth. If these parameters are selected carefully, an orbit can be established that exhibits a special rate of precession whereby the plane of the satellite orbit rotates once per year in inertial space. To an observer on the ground, a satellite covers the same track in the sky at the same time each day because the precession rate of the orbital plane just matches the day to day change in the earth's relation to sun as the earth moves around the sun. This orbit is referred to as sun-synchronous and is of particular interest to satellites carrying optical instruments like photo-reconnaissance and remote sensing satellites since the angle between the sun and the surface of the earth is relatively constant for all observation points along a particular latitude.

Figure 2-3 illustrates the distribution of the satellites for the past four years according to the orbital parameters, inclination and semi-major axis. For a circular orbit, the semi-major axis is simply the altitude of the orbit above the surface of the earth plus the mean radius of the earth measured from its geometric center. The three dimensional plot excludes 4 civilian interplanetary, 4 civilian highly elliptic astronomical and one military satellite for which the orbital elements have not been published. Figure 2-4 focuses on the low earth orbit satellites in the

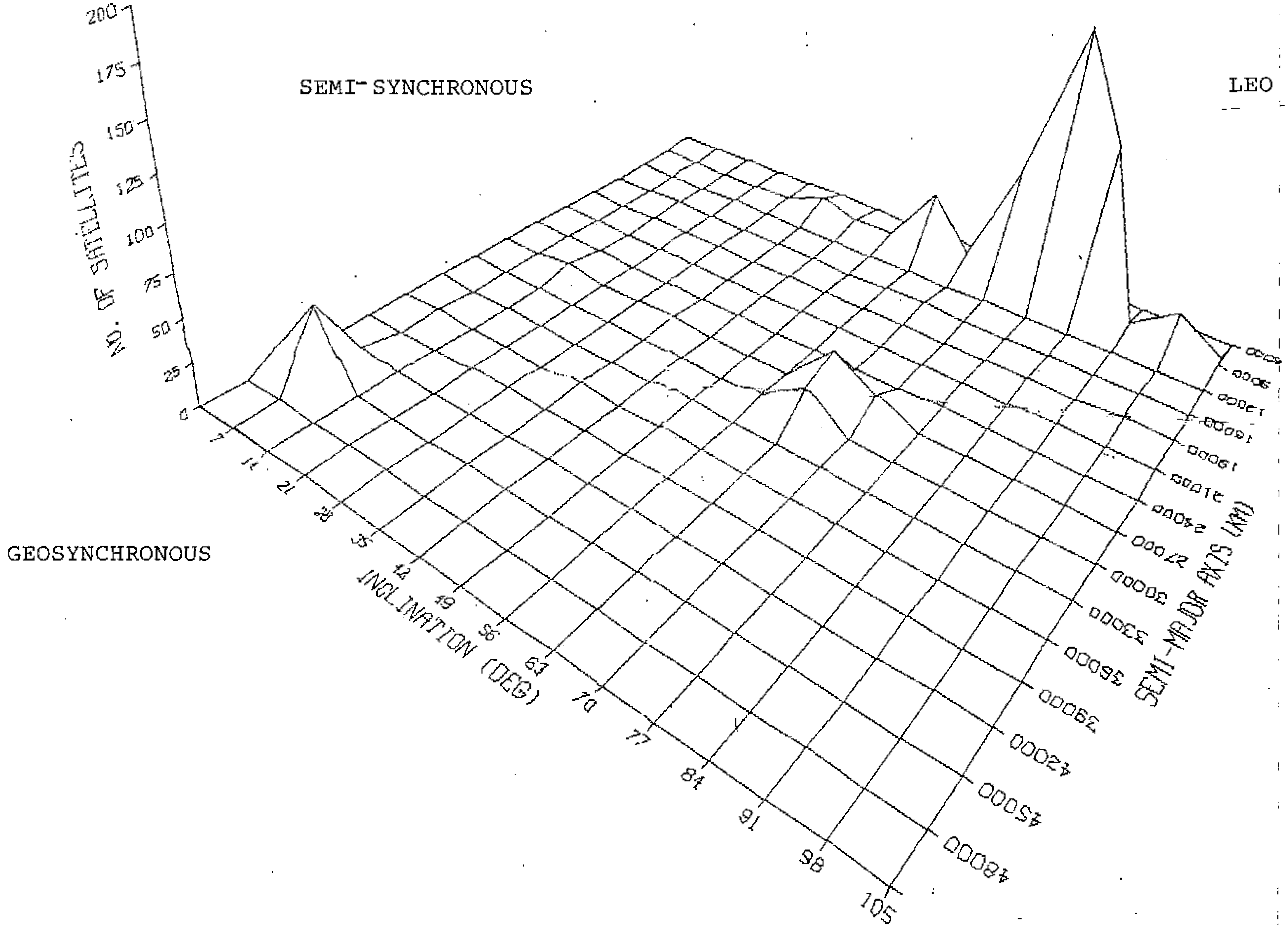


FIGURE 2-3 ORBITAL DISTRIBUTION OF SATELLITES LAUNCHED 1980-1983 AD

2.2 Space-to-Space Weapons Situation (Continued)

database with a cross-sectional view of the earth and LEO orbits. These two figures illustrate the degree to which satellites are employed in quantized orbital bands.

Figure 2-4 illustrates the entire population of satellites launched in the past four years into the LEO domain, while Figure 2-5 only illustrates the military launches. Comparison of these figures illustrates that both civilian and military launches utilize low earth orbits and that the LEO military satellites occupy a narrowband of inclinations between 50^{o} and 105^{o}. Both civilian and military satellites utilize the four principle orbit domains illustrated in Figure 2-2. Tables 2-3 and 2-4 specifically identify the orbits of American and Soviet military satellites found in employment today. Figure 2-6 illustrates the typical distribution of these satellites in the four characteristic orbits. Evident from these tables and figures is the degree to which satellite missions exploit the advantages of the specific orbit regimes.

In terms of vulnerability from an antisatellite weapon, a satellite is more or less at risk in terms of the type of satellite it is, the orbit into which it is placed, and the type of mission it is to carry out.

Delicate optical sensors for remote sensing or on-board altitude control of satellites can be readily burnt-out by a powerful laser beam. The maneuver to bring a damaging beam into the field of view of an optical sensor can be made very difficult but once accomplished, it only requires a momentary exposure to cause irreparable damage through permanent blindness.

All satellites are easily damaged by physical contact. Their light weight construction entails the use of structural members that are just adequate to withstand a launch environment. The only exception might be small bomb which, because of its size, could be physically robust in the space environment.

Solar arrays can also be damaged by powerful lasers because like other optical sensors they are tuned for maximum absorption of visible light. Under threat of an

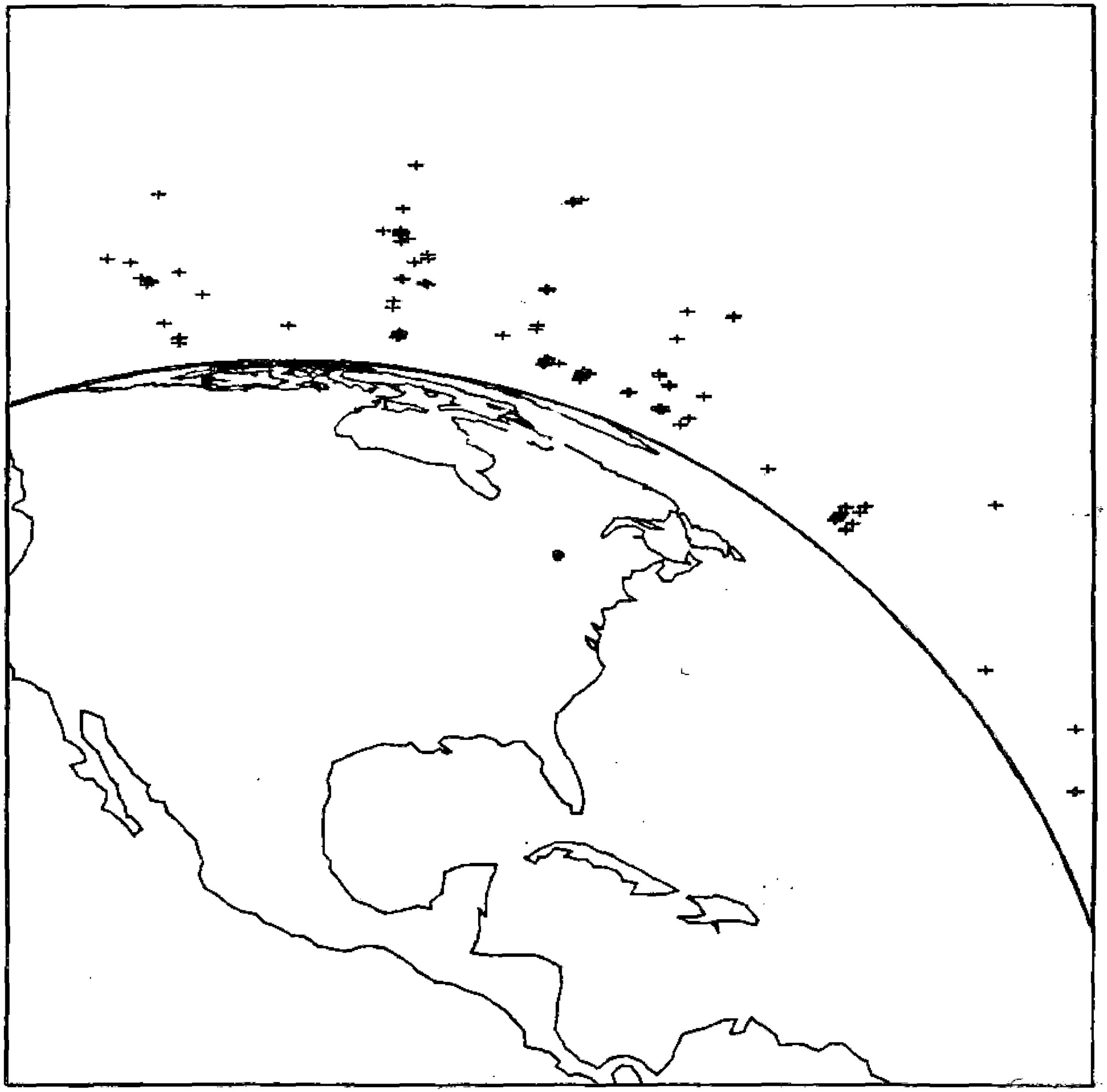


FIGURE 2-4   ORBITAL DISTRIBUTION OF LEO SATELLITES LAUNCHED 1980-1983 AD

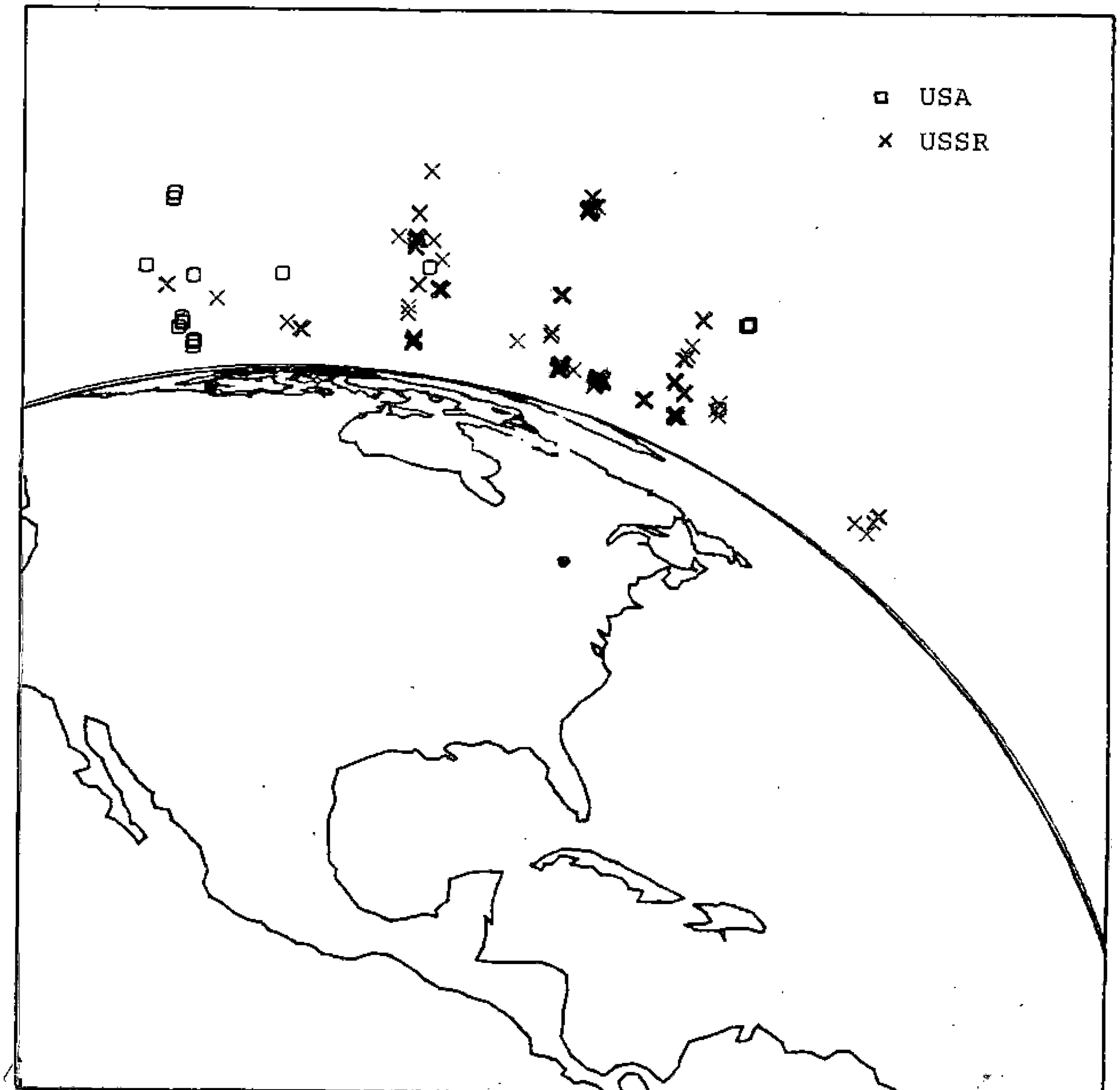


FIGURE 2-5 ORBITAL DISTRIBUTION OF LEO MILITARY SATELLITES LAUNCHED 1980-1983 AD

AMERICAN MILITARY SATELLITE SYSTEMS

FUNCTION	SYSTEM NAME	INITIAL YEAR	1980-1983 LAUNCHES	SYSTEM COMPLEMENT	ORBITAL REGIME	INCLINATION (deg)	PERIOD (min)	PERIGEE ALTITUDE (Km)	APOGEE ALTITUDE (Km)	MISSION LIFETIME
COMMUNICATIONS	DSCS II	1971	1	4+2	GEOSYNCHRONOUS	2.0	1432.2	35644.0	35776.0	5 YEARS
	DSCS III	1982	1	4+2	GEOSYNCHRONOUS	2.4	1432.2	35644.0	35776.0	10 YEARS
	SDS	1971	3	2	HIGHLY ELLIPTIC	63.4	708.1	375.0	39063.5	2 YEARS
	FLTSATCOM	1978	3	4+1	GEOSYNCHRONOUS	2.6	1433.6	35185.0	39000.0	5 YEARS
	NATO	1970	0	3	GEOSYNCHRONOUS	2.8	1436.0	35784.0	35784.0	7 YEARS
	MARISAT	1976	0	3	GEOSYNCHRONOUS	2.5	1436.0	35784.0	35784.0	10 YEARS
EARLY WARNING	DSP	1970	2	3	GEOSYNCHRONOUS	1.9	1430.5	35637.0	35717.0	2-5 YEARS
ELECTRONIC INTELLIGENCE	FERRET	1980	3	1	LOW EARTH ORBIT	96.7	111.8	1304.5	1308.0	UNK'N
	RHYOLITE	1973	0	4	GEOSYNCHRONOUS	0.2	1436.0	35784.0	35784.0	3-6 YEARS
METEOROLOGY	DMSP	1971	2	2	LOW EARTH ORBIT	98.7	101.3	812.5	827.0	3 YEARS
NAVIGATION	TRANSIT	1964	0	5	LOW EARTH ORBIT	90.0	105.0	1075.0	1100.0	3 YEARS
	NAVSTAR	1978	3	18+3	SEMI-SYNCHRONOUS	62.8	713.8	19879.3	20279.3	5-7 YEARS
	NOVA	1981	1	*	LOW EARTH ORBIT	90.7	109.0	1170.0	1187.0	6 YEARS
OCEAN SURVEILLANCE	WHITECLOUD	1976	9	12	LOW EARTH ORBIT	63.4	107.3	1055.1	1159.8	>3 YEARS
PHOTOGRAPHIC RECONNAISSANCE	KH-8	UNK'N	0	1	LOW EARTH ORBIT	96.5	(86.2)	52.0	119.0	<6 WEEKS
	KH-9	UNK'N	2	1	LOW EARTH ORBIT	96.9	89.6	136.0	367.5	6 WEEKS
	BIG BIRD	1971	4	1	LOW EARTH ORBIT	96.5	88.6	154.6	268.8	3-5 MONTHS
	KH-11	1976	3	2	LOW EARTH ORBIT	97.0	92.1	253.7	496.3	>2 YEARS
SCIENTIFIC	HILAT	1983	1	1	LOW EARTH ORBIT	82.0	100.9	767.0	834.0	UNK'N

* NOVA SATELLITES ARE BEING INCORPORATED IN THE TRANSIT SYSTEM.
** DURING 1980-1983 ONE SPACECRAFT LAUNCHED FROM ETR COULD NOT BE IDENTIFIED.
( ) INDICATES THAT THE DATA IS UNCERTAIN.

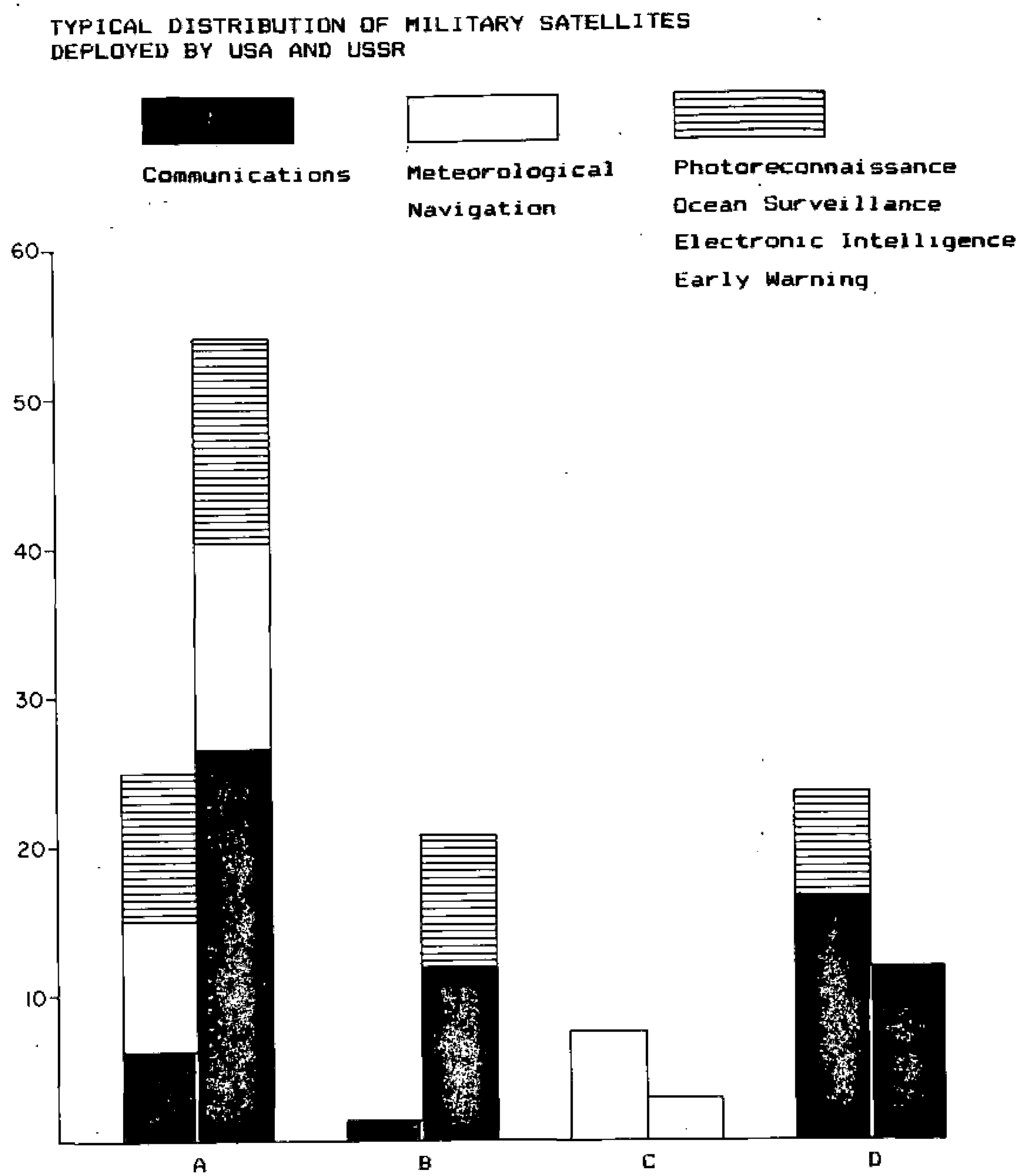
TABLE 2-3 AMERICAN MILITARY SATELLITE SYSTEMS

SOVIET MILITARY SATELLITE SYSTEMS

FUNCTION	SYSTEM NAME	INITIAL YEAR	1980-1983 LAUNCHES	SYSTEM COMPLEMENT	ORBITAL REGIME	INCLINATION (deg)	PERIOD (min)	PERIGEE ALTITUDE (Km)	APOGEE ALTITUDE (Km)	MISSION LIFETIME
COMMUNICATIONS	COMM 1	1970	10	3	LOW EARTH ORBIT	74.0	100.8	788.1	826.0	17 MONTHS
	COMM 2	1970	80	24	LOW EARTH ORBIT	74.0	115.4	1451.3	1524.0	5 MONTHS
	MOLNIYA-1	1965	14	8	HIGHLY ELLIPTIC	62.8	721.1	539.4	40003.8	2 YEARS
	MOLNIYA-3	1974	11*	4	HIGHLY ELLIPTIC	62.8	736.0	547.9	40738.7	2 YEARS
	RADUGA	1975	8	2+2	GEOSYNCHRONOUS	0.7	1464.5	36346.6	36346.6	2 YEARS
	GORIZONT	1978	5	6+2	GEOSYNCHRONOUS	1.0	1458.2	36217.0	36217.0	2-3 YEARS
	COSMOS-1366	UNK'N	1	UNK'N	GEOSYNCHRONOUS	1.5	1437.0	35820.0	35820.0	UNK'N
EARLY WARNING	EW-1	1972	20**	9	HIGHLY ELLIPTIC	62.8	712.7	625.7	39877.5	20 MONTHS
ELECTRONIC INTELLIGENCE	ELINT 2	1970	13	6	LOW EARTH ORBIT	81.2	97.5	629.5	672.9	20 MONTHS
	##	##	3	(1)	LOW EARTH ORBIT	74.0	94.9	493.0	541.0	UNK'N
	##	##	2	(2)	LOW EARTH ORBIT	83.0	109.1	407.0	1982.0	UNK'N
	##	##	3	(3)	LOW EARTH ORBIT	82.5	97.8	642.0	672.0	UNK'N
METEOROLOGY	METEOR 2	1975	4	3	LOW EARTH ORBIT	81.3	102.0	838.0	903.8	UNK'N
			1#	(2-3)	LOW EARTH ORBIT	82.5	104.2	954.0	976.0	UNK'N
NAVIGATION	NAV 2	1974	27	6	LOW EARTH ORBIT	83.0	104.9	976.4	1023.7	16 MONTHS
	NAV 3	1976	##	4	LOW EARTH ORBIT	##	##	##	##	3 YEARS
	GLONASS	1982	9	9-12	SEMI-SYNCHRONOUS	64.6	674.3	19118.0	19118.0	UNK'N
OCEAN SURVEILLANCE	EORSAT I	1979	9	2	LOW EARTH ORBIT	65.0	93.3	432.7	456.6	6 MONTHS
			5	(2)	LOW EARTH ORBIT	82.5	97.8	647.6	678.2	UNK'N
	RORSAT	1967	9	2	LOW EARTH ORBIT	65.0	90.0	276.0	296.3	3-4 MONTHS
PHOTOGRAPHIC RECONNAISSANCE	VARIOUS	1962-1975	##	4x(1)	LOW EARTH ORBIT	##	##	##	##	2-6 WEEKS
	##	##	1		LOW EARTH ORBIT	50.6	90.0	209.0	377.0	
	##	##	15		LOW EARTH ORBIT	64.9	89.5	188.0	341.0	
	##	##	17		LOW EARTH ORBIT	67.2	89.7	181.2	371.9	
	##	##	31		LOW EARTH ORBIT	70.4	90.0	208.8	373.1	
	##	##	43		LOW EARTH ORBIT	72.9	90.0	207.5	375.1	
	##	##	15		LOW EARTH ORBIT	82.3	89.0	213.3	272.1	
SCIENTIFIC	GEODETIC	1968	2	(1)	LOW EARTH ORBIT	82.6	116.0	1495.0	1526.5	UNK'N
	##	##	1	(1)	LOW EARTH ORBIT	73.6	116.1	1497.0	1537.0	UNK'N
	(IONOSPHERE)	UNK'N	2	(1)	LOW EARTH ORBIT	83.0	106.2	353.5	1767.0	UNK'N

* INCLUDES 1 UNSPECIFIED MOLNIYA SATELLITE FAILURE.
** INCLUDES 2 EARLY WARNING SATELLITE FAILURES.
\## DISTINCTION AMONGST SATELLITE SYSTEM CLASSIFICATIONS DIFFICULT WITH INFORMATION AVAILABLE. CLASSIFICATION SHOWN IS BASED ON ORBIT PARAMETERS.
\# 1ST METEOR 2 SATELLITE IN THIS ORBIT REGIME.
( ) INDICATES THAT THE DATA IS UNCERTAIN.
@ 12 ANTISATELLITE RELATED LAUNCHES & 8 MINOR MILITARY (PRESUMABLY RADAR CALIBRATION) LAUNCHES WERE LOGGED BETWEEN 1980-1983 AD.

TABLE 2-4 SOVIET MILITARY SATELLITE SYSTEMS



	LOW (circular)	HIGHLY ELLIPTIC	SEMI-SYNCHRONOUS (circular)	GEO-SYNCHRONOUS (circular)
Altitude	150-2000Km	Apogee 40,000Km Perigee 600Km	20,000Km	36,000Km
Inclination to Equator	65 -115	63	63	0 -6
Period	1.5-2 Hrs	12 Hrs	12 Hrs	24 Hrs

FIGURE 2-6 AMERICAN/SOVIET MILITARY SATELLITE DEPLOYMENTS

2.2 Space-to-Space Weapons Situation (Continued)

attack by an ASAT, solar arrays of the future may need to be designed for repeated deployment and retraction to assure survival of a satellite's primary power source.

Communications satellites are susceptible to jamming and temporary loss of function. Although anti-jamming technology is reaching ever increasing levels of sophistication, the threat is also becoming more adaptive and more sophisticated.

From the point of view of orbits, a satellite in a low earth orbit is at a high risk for the simple reasons that:

(a) It is easier to place a heavy weapon in low orbits

(b) A weapon in low earth orbit has a much higher selection of potential targets, is more effective in terms of the number of potential kills and is therefore, more likely to be found there.

Since the current population of satellites reflects an optimization for effectiveness assuming no threat, it must be presumed that at some future time under the threat of attack by ASAT's on satellites, the optimum deployment of application satellites, will see the gradual introduction of hardening, redundancy and unconventional orbits as a defence against complete loss of function or service. The nature and timing of this new optimization will be driven by the pace of events in the weapons arena: an arena affected in turn by the changing nature of the target.

In an early report [1], optional generic forms of ASAT's were reviewed in some detail. Drafting from that report, a weapon in space whose prime function is to destroy or permanently damage another satellite can accomplish its objective in one of six ways. It can:

(a) Collide - possibly many times

(b) Explode - with a conventional fragmentation or pellet warhead

(c) Explode - with a nuclear warhead

2.2 Space-to-Space Weapons Situation (Continued)

(d) Launch smaller rockets with warheads, or with an intercept and collision capability

(e) Launch electromagnetic or particle beams

(f) Jam and spoof command, communications and sensors on a satellite.

Depending on how it achieves its objectives, an ASAT weapon can be classed as a close range weapon or a stand-off weapon. The stand-off weapons are further subdivided into weapons that are destructive and weapons that cause temporary disorientation or improper functioning of the satellite without experiencing destructive effects in the long term. The various classifications are set out in Table 2-5.

From information available in the current unclassified literature, the trend and the outlook in weapons technology is the eventual use of beam weapons. Launch and intercept weapons using physical impact or explosive warheads are the first generation of ASAT weapons. Lasers in space would be the second. The X-ray laser is included on the list of potential lasers, although its deployment is a special case because of the present ban on nuclear explosives in space. Particle beam weapons are likely to be the third generation of ASAT weapons, with a capability to attack targets on the earth from space. Certain classes of lasers may also have wavelengths suitable for penetrating the earth's atmosphere from space. Spacebased weapons for ballistic missile defense would be (more) complex derivatives of the second and third generations of antisatellite beam weapons.

A reading of the unclassified literature of the past three to five years leaves little doubt that both the USSR and the US have well advanced conceptual options for protecting their space assets from space. The current generation of ASAT weapons using the launch, seek, maneuver and kill sequence is at least partially in place now and could, by the end of the decade, be in full deployment and readiness. As has already been mentioned, Paxsat has no role in the scenario of these first generation weapons.

TABLE 2-5 KILL RANGE OF ASAT WEAPONS

A. CLOSE RANGE GROUP

1. Collision - Zero range, requires contact
2. Conventional explosives - a few tens of meters
3. Small rockets - a few kilometers

B. STAND-OFF GROUP, DESTRUCTIVE

1. Nuclear explosives including EMP - long range area or volume weapon
2. Visible/Infrared lasers - medium range, to 500 km
3. Short Wavelength lasers - long range, a few thousand kilometers
4. X-ray laser - medium range, possibly long range

C. STAND-OFF GROUP, DISORIENTING

1. Jamming and Spoofing - long range

2.2 Space-to-Space Weapons Situation (Continued)

Decades of the 1990's and the 2000's would see the next generation of ASAT weapons being developed and deployed. These weapons would be based on beam technology, being either electromagnetic (laser) beam or particle beam. For deployment in space, laser beam technology is the most promising and hence would be the first of the two to be used. Extensive testing for at least ten years preceding deployment of any of the three generations is assumed.

While it is beyond the scope of this study to envisage all of the possible configurations for an effective ASAT, given the range of targets against which they might be directed, there are certain general observations that would hold true for most systems.

Damage or destruction of satellites in geostationary orbit can be accomplished by ASAT's with short range capability since an ASAT weapon drifting slowly in or near geostationary orbit will eventually come within a few kilometers of all of the satellites in that orbit.

Damage or destruction to satellites in Molniya orbits or 12 hour circular orbits would require an ASAT with a stand-off capability. The only alternative would be to employ a close range weapon and place the weapon platform in a co-orbit with the target satellite, clearly a provocative act requiring no further verification.

A stand-off weapon with a range of several hundred kilometers could be effective against many of the low altitude reconnaissance satellites shown at inclinations between 60° and 80°. A low altitude satellite with a nuclear warhead would be particularly effective against targets in this range of orbits. Satellites with weapons to be used against earth targets, if optimally deployed, would be found in the same low altitude range.

By similar reasoning, a satellite with a range of a few hundred kilometers at an inclination corresponding to sun-synchronous operation could present a threat to the military photo reconnaissance satellites and civilian remote sensing satellites operating in the sun-synchronous orbits.

2.2 Space-to-Space Weapons Situation (Continued)

In August 1981, the Joint Chief-of-Staff (JCS) of the US issued an ASAT requirements document which set out the perceived requirements in the United States for a US antisatellite weapons system. This document contains a threat list of Soviet spacecraft at high altitude and at low altitude. Soviet satellites shown on the threat list were divided into four priorities including passive and active satellites.

The first priority for the US ASAT system is Soviet weapons systems on satellites and Soviet satellite surveillance systems capable of real-time targeting against US forces.

The number two priority is surveillance systems capable of targeting US forces, but not in real-time.

The third priority is Soviet support system directly supporting weapons platforms, i.e. communications and navigation satellites.

The fourth priority is satellites supporting Soviet forces that indirectly support the weapons platforms, i.e. national and major headquarters level communications.

The Deputy Secretary of Defense in the U.S. has directed the United States Air Force to develop an ASAT system capable of negating priority one and priority two passive threat satellites at low altitude. Soviet space weapons are active satellites and would not be included in this request.

The targets for a US ASAT system in order of priority are summarized in Table 2-6

In the August 1981 ASAT requirements document referred to above, the US Joint Chiefs-of-Staff estimated that the USSR will have six orbiting high-energy laser ASAT's by 1990, designed for the same mission as the first generation of launch, seek, maneuver, and kill ASAT weapons.

TABLE 2-6 TARGETS FOR US ASAT's IN ORDER OF PRIORITY

1. Soviet weapons systems and surveillance satellites capable of real-time targeting against US Forces.

2. Soviet surveillance satellites for non-real-time targeting.

3. Soviet navigation and communications satellites supporting weapon platforms.

4. Soviet satellites indirectly supporting weapons, e.g. H.Q. Communications Satellites.

2.2 Space-to-Space Weapons Situation (Continued)

A review by the US Government Accounting Office of the Defense Department plans for performing the ASAT function has revealed a USAF ASAT concept utilizing relatively low power maneuverable laser weapons. In the concept, seven weapons would be placed in low altitude waiting orbits and eight others would be placed near the geostationary orbit. The 15 satellites would meet the stated ASAT mission requirements.

An alternative US ASAT system would see a constellation of high power, long range laser weapons in fixed orbits. This alternative system would also have a capability for targets other than ASAT's.

2.3 Space-to-Earth Weapon Deployment

Targets in space for a spacebased weapon can be enumerated and ranked according to some priority, however primitive. Earth targets for a spacebased weapon are much more difficult to enumerate because they are a diverse assembly of strategic objects and locations, the destruction of which has meaning in the context of a military objective. A reading of the current literature reveals that seats of government, military and industrial complexes and large civilian population centers, though not necessarily in that order, are prospective earth targets. Isolated space support installations, for example, a control center for surveillance and tracking satellites, would also seem to be logical candidates. The location of these targets is well known and they are all immovable. That being said, further detailing of their size, numbers and location is not useful in the context of a conceptual study save to note that they are distributed around the globe.

Given the immense size of some of these targets as compared to a single satellite or even a cluster of satellites, the choice of effective weapons to be parked in space for eventual deployment against them is more limited than in the case for satellite targets.

A nuclear explosion in space is known to be an effective weapon in that the ensuring Electromagnetic Pulse (EMP) would cause the destruction of communications and other electrical apparatus, perhaps even in primary power systems over many thousands of square miles. A nuclear

2.3 Space-to-Earth Weapon Deployment (Continued)

device in a satellite is therefore a legitimate option, technically speaking. Similarly, a nuclear explosion following re-entry of a satellite is an effective weapon of mass destruction. Its tactical or strategic value might be questioned when it is compared to, for example, a submarine launched missile with a nuclear warhead, if for no other reason than that while in orbit, its capability to re-enter at the appropriate place on earth is dictated by its orbital characteristics. Delays of 12 hrs or more to achieve the right sub-orbit location might be a necessary, but an unacceptable, restriction. Nevertheless, a re-entry nuclear device is an option.

A chemical/biological weapon for use against earth targets must re-enter and be placed at a specific location before it is activated. Successful deployment of such a weapon could cause havoc in a heavily populated area. As for the re-entry nuclear device, operational questions arise because of restricted useable time windows and the alternative of a submarine launched missile. But the option does exist.

Beam weapons attacking earth targets from space are a very future oriented concept. First of all, to be effective, the target or some key component of it must be small in size, comparing perhaps to a satellite. Second, most earth targets can be hardened against a beam attack, so the effectiveness of the weapon comes into doubt. In the light of these two difficulties, the range of acceptable earth targets may be so small in numbers that the spacebased beam weapon is suboptimal when compared to other options. A final problem with a beam weapon is the effect that the earth's atmosphere and magnetic field have on it, from the point of view of absorption and beam bending.

Without benefit of sophisticated (and classified) operational research analysis, a simple ordering in likelihood from most probable to least probable is as follows:

(a) A high altitude nuclear detonation from space creating an EMP

(b) A de-orbited chemical/biological area weapon

2.3 Space-to-Earth Weapon Deployment (Continued)

(c) A de-orbited nuclear device

(d) A laser beam weapon in space

(e) A particle beam weapon in space.

The first three candidates are technically feasible, conventional technology, and the remaining two require technical feasibilities to be established over the next one or two decades.

Verification of the first three candidates by Paxsat could require that Paxsat be maneuvered to within a few kilometers of the satellite carrying or believed to be carrying the weapon. Close-in remote sensing of nuclear decay products or chemical leakage would be a key measurement. Physical features as observed optically might be quite innocuous.

The remaining two candidates would be more easily verified because of the large dimensions and unique appendages on the satellite.

2.4 Summary of the Space Weapon Environment

The nature of the targets in space and on earth, and the qualitative dimensions of the threat to these targets from weapons on satellites have been examined in sections 2.2 and 2.3 of this section. In this final part, the results are combined in a single Paxsat system framework.

The highest priority targets in space are military satellites for targeting and tactical surveillance and, of course, other satellites carrying weapons.

Should targeting satellites be placed at very high altitudes, in the order of 100,000 km for example, a practical weapon system will also have to be placed at high altitude, certainly within 10,000 km of its intended target. Successful verification by Paxsat for such a weapon is probably not practical, although the very presence of the satellite may serve to cast suspicion on its mission. A jammer would be classed as a weapon in this context.

2.4 Summary of the Space Weapon Environment (Continued)

Military satellites for surveillance of terrestial activities or even activities within the atmosphere, for example aircraft operations, must be in relatively low orbits to obtain highly detailed information. All of these satellites are vulnerable even with hardening, so they are legitimate targets. Weapons satellites would be placed in the same general region of space as these satellites, the actual separation between the weapon and the target depending upon whether the weapon was in the stand-off or close range class. Paxsat would have a meaningful role in all cases.

Weapons to be deployed against targets on earth would be stationed in relatively low orbits for reasons of cost effectiveness. Whether used directly from space or used after re-entry, they are all legitimate objects of Paxsat verification.

Difficulties in successful verification of earth directed weapons as well as space directed weapons pertain mostly to verifying the first generation of unsophisticated close range weapons and the stand-off nuclear EMP device. Both require close inspection to confirm the presence or absence of a weapon payload. The other stand-off weapons, essentially the beam weapons have more distinguishing features and are therefore harder to disguise.

A summarization of the four categories of assets (i.e. targets) and the six potential spacebased weapon systems is shown in Table 2-7. Relevance between weapons systems and targets are signified in this table by the eight cases marked 'yes'.

Collision weapons, because they are limited to close range encounters, are effective against the category 1 targets, designated Space Assets in Table 2-7, but have no role against targets in categories 2, 3 or 4. These weapons are relatively inexpensive, a re-usable weapon could require re-fuelling in space, and the technology exists to build such a weapon now. Such a weapon could be difficult to verify if it also served some peaceful role.

TABLE 2-7 WEAPON SYSTEM THREAT SCENARIO RELEVANCE

TARGET CATEGORY	WEAPON SYSTEM					
	COLLISION	EXPLOSIVE	NUCLEAR	ROCKET	LASER BEAM	JAMMING
SPACE ASSETS	YES	YES	YES	YES	YES	UNLIKELY
TERRESTIAL	NO	NO	YES	NO	NO	NO
BALLISTIC MISSILES	NO	NO	UNLIKELY	NO	POSSIBLE	NO
SPACEBASED WEAPONS	UNLIKELY	UNLIKELY	YES	UNLIKELY	YES	UNLIKELY

2.4 Summary of the Space Weapon Environment (Continued)

Conventional explosion weapons, like collision weapons, are effective at close range only. Thus, they too are effective against category 1 targets and ineffective against targets in categories 2, 3 and 4. These weapons are relatively inexpensive, they could be built and deployed now, and they could be difficult to verify.

Nuclear explosion weapons are relatively inexpensive and could be built and deployed now. They are classified as long range area or volume weapons. The radiation from a nuclear explosion in space is effective against category 1 targets, present and future, and against the terrestial targets and spacebased weapons of categories 2 and 3. The Electromagnetic Pulse (EMP) from a nuclear explosion in space is most effective against category 2 (terrestial) targets on a continental basis.

Space launched rockets are effective in space only. They are very effective against the current generation of military and commercial assets which are highly vulnerable and without defences. Rockets would have limited effectiveness against a spacebased weapon system, which is assumed to be 'intelligent' in a threat situation. Spacebased rockets are inexpensive although the spacebased launching platform is a highly intelligent system. Such a system could be successfully deployed in the next decade. A rocket platform would be less difficult to verify than an exploding device.

Lasers or particle beam weapons in space are a threat to any object in space or an object approaching or leaving space. Hence, they are effective against category 1, 3 and 4 targets. The potential effectiveness of these weapons against point targets on earth has yet to be established. These systems are very expensive to build and deploy and they may be expensive to maintain operationally capable. They are not difficult to verify.

Spacebased jammers for use in Electronic Warfare (EW) and Electronic Countermeasures (ECM) are ineffective against categories 2 and 3 targets and have very limited effectiveness against category 1 targets as a substitute for earthbased jamming sources. A spacebased jammer has

2.4 Summary of the Space Weapon Environment (Continued)

doubtful effectiveness against a category 4 target because these targets are assumed to be very sophisticated and highly optimized for countermeasures. An additional barrier to long term use of spacebased jammers is the continuing development of new EW technology and hence the need to continually upgrade a jammer's capability.

To take this analysis of the space weapon environment a step further, the population of satellites in Category 1 has been broken down into seven groups in Table 2-8. Studying the most threatened groups serves to focus the Paxsat scenario more clearly.

Bearing in mind that a space wars concept assumes a very short (perhaps only a few hours) and intense conflict, it follows that all targets must be pre-selected and pre-targeted and will (because of the complexities of this process) be only the most crucial targets. To be cost effective, the spacebased weapons will be highly optimized, therefore they must be pre-programmed and pre-positioned. In the space environment of orbiting targets and weapons at various inclinations, and at different points in the orbit cycle, only a limited number of targets can be attacked in a coordinated action occuring within a very short time frame. The indiscriminate widespread, target-of-opportunity, attack of space assets becomes an unlikely scenario. In examining the roles and capabilities of the space population groups, it is concluded that the groups at the high end of the risk scale are the second and third groups, military navigation and military surveillance and reconnaissance groups. These groups include what is often referred to as targeting satellites. At this point in time, some of the military satellites of these types are at considerable risk, they are visible, accessible and vulnerable. It must be assumed that succeeding generations of such space assets would be less vulnerable through hardening, repositioning and redundancy measures.

TABLE 2-8 CURRENT SPACE ASSET CATEGORIES

1. COMMUNICATIONS
2. NAVIGATION
3. RECONNAISSANCE/SURVEILLANCE
4. METEOROLOGICAL
5. REMOTE SENSING
6. MANNED MISSIONS
7. SCIENTIFIC

2.4 Summary of the Space Weapon Environment (Continued)

In the matter of terrestial targets of category 2 (Table 2-7), there are two classes:

(a) Area targets

(b) Point targets

An area target might be a city or military/industrial area. A point target might be a missile launch site or a military/government command and control center. Taking into account the features of spacebased weapons and cost effective alternatives, it is concluded that area targets would be more rewarding than point targets.

Hence the nuclear radiation/EMP risk could be relatively high. A bateriological/chemical risk is valid to consider but the necessary optimal target conditions, including local weather conditions could present some difficulty in such a scenario.

Concerning the category 3 target in Table 2-7 , ballistic missiles, little more can be said at this time. The destruction of ballistic missiles in flight in the context of current US thinking places a weapon system in space, where none now exist. This event could trigger the generation of the category 4 targets, another weapon.

3.0 POLITICAL CONSIDERATIONS AFFECTING AN OUTER SPACE ARMS CONTROL AGREEMENT

3.1 Introduction

In constructing a politically plausible scenario for the operation of a Paxsat A system, two connected sets of issues must be addressed. Taken together, these issues and their implications form the general context which will determine the degree to which the operation of a Paxsat system is both plausible and valuable in arms control terms. These issues also have specific implications for whatever criteria of technical 'sufficiency' are to be applied in defining and evaluating the operational parameters of a Paxsat system.

The first set of issues concerns the 'participatory status' of an arms control agreement relating to the weaponization of outer space. In particular, will the arms limitation regime which Paxsat will assist in verifying be bilateral (US-Soviet) or multilateral in nature? In other words, what role can be envisioned for non-superpower states? This question may, in turn, be broken into two component parts:

(a) The participatory status of the negotiations process (bilateral versus multilateral).

(b) The nature of the resulting agreement, (bilateral versus multilateral) and the degree, if any, of multilateral involvement in the administration of the agreement, including its verification and compliance provisions.

The second set of issues concerns the precise nature of the treaty administration and verification regime associated with an outer space arms control agreement. In particular, what purpose could a Paxsat system serve in constituting or contributing to the verification requirements or provisions of a plausible accord, and what standards, if any, can be identified to assist in defining verification 'sufficiency' for Paxsat? And lastly, should a credible verification role be identified for Paxsat, what general operational parameters can be identified which might serve as the

3.1 Introduction (Continued)

broad administrative and decision-making structure governing the use of Paxsat in relation to the total treaty context?

3.2 Multilateral Verification and Arms Control Agreement

A central question in constructing a plausible scenario for the operation of a Paxsat system is the issue of whether or not an outer space arms control regime can be expected to be bilateral or multilateral in nature. If outer space arms control negotiations and the resulting agreement were to be a purely intramural superpower exercise, the task of defining a credible verification role for Paxsat would be entirely different from that present in the case of a multilateral treaty containing provisions for multilateral treaty administration and verification. Put somewhat starkly, the former would imply that Paxsat operate outside the treaty context, relying essentially on self-contained technical resources, and doing so possibly even against the wishes of the Superpowers. In the latter case, a more benign scenario can be envisioned, with US and Soviet acquiescence in and support of third party involvement in the mechanisms of treaty verification. Verification in this case might constitute, as is discussed in more detail below, a more cooperative multi-tiered activity, with far less onerous technical requirements for Paxsat in the context of a carefully constructed multilateral verification framework.

Prior to proceeding with this question, it will be noted that a central assumption underlying this discussion is that a legal framework is required to legitimate and direct the functioning of a Paxsat system. This is based on several factors. If the proposed purpose of Paxsat is to assist in verifying the existence or otherwise of certain classes of military activities or systems, it is difficult to envisage what role Paxsat would have in the absence of a treaty or agreement relating to these systems and/or activities. Without a treaty, Paxsat would merely be 'verifying' the occurrence of activities, or the existence of systems, which were sanctioned by international law. Moreover, were such a role be envisioned, several somewhat troublesome questions would arise. If Paxsat were to simply observe what was allowed, what would be the

3.2 Multilateral Verification and Arms Control Agreement (Continued)

purpose of such an activity? Since questions as to the legality of activities would not arise, by definition, the observation of legally sanctioned events would either be pointless, or would simply serve to enhance the accuracy or breadth of information available to those with access to the output of the Paxsat system. In the latter case, it is likely that those states whose systems and activities would be under surveillance (primarily the Superpowers) would be opposed to such a development.

This, in turn, might pose certain problems:

(a) the operation of Paxsat might itself be perceived as the hostile act, acting to increase precisely those tensions which the system was presumably designed to reduce.

(b) As a matter of practical politics, it is questionable whether states allied with either Superpower would seek to engage in activities which were opposed by the US or USSR.

(c) Should the only purpose of Paxsat be intelligence gathering, it is doubtful whether the states whose resources would be required to put Paxsat in place would consider the expense justified.

(d) Depending on the identity of the states involved in operating Paxsat, there would arise real questions as to the willingness of these states to share intelligence, and practical problems as to who would direct the system to look at whose activities.

In conclusion, such a scenario over and above the specific issues just outlined, is not an arms control or verification scenario. Simply put, to postulate the operation of Paxsat without reference to a specific arms limitation regime is to postulate the development of a system whose only role could be to gather information about military activities sanctioned by international law.

3.2 Multilateral Verification and Arms Control Agreement (Continued)

It is possible, however, to envisage other scenarios according to which Paxsat might operate without immediate or formal linkage to a treaty. One would involve the development and deployment of Paxsat in anticipation of an arms control agreement which it could then assist in verifying. It could be argued that this option has several advantages.

(a) Given the lead time involved in the development and deployment of a satellite verification system, deployment in advance would allow for the immediate utilization of the system once a treaty is signed.

(b) To the extent that it is accepted that the existence of Paxsat would enhance confidence in treaty adherence, it could be argued that deployment in advance would encourage negotiation of the agreement itself.

Other factors, however, would seem to speak against the viability of this option.

(a) Unless the system is designed to simply lie dormant pending the signing of an agreement, the same questions concerning what exactly the system is verifying would arise as in the case discussed above.

(b) It is unlikely that states would consider the expense and effort involved in deploying such a system warranted in the absence of an assured role.

(c) The optimum technological and operational characteristics of the systems are likely to be dependent on the precise nature of the restrictions embodied in the arms control agreement. Deployment in advance would preclude this design optimization, and might result in a system inappropriate to the agreement.

3.2 Multilateral Verification and Arms Control Agreement (Continued)

Another scenario would envisage the development and deployment of Paxsat as an adjunct to existing agreements, rather than as a verification asset at the disposal of a new accord relating to military activities in outer space. According to this option, Paxsat could assist in verifying adherence to such agreements as the ABM Treaty, the Outer Space Treaty and the Moon Treaty.

Once again, however, there would appear to be serious problems with this scenario.

(a) Insofar as it is intended that Paxsat be accorded a formal verification role, the treaties as negotiated would have to be amended to this end.

(b) Given that these agreements were negotiated in the absence of the sort of capability represented by Paxsat, it is at least questionable whether or not such a capability is needed in order to ensure adequate confidence in adherence. Moreover, in some contexts, such as the Outer Space Treaty, it is arguable that certain provisions of the agreements are themselves unverifiable, given the nature of the technologies and activities prohibited. The utility of Paxsat in these contexts is, therefore questionable.

Specific problems arise in contemplating a role for Paxsat in the context of existing bilateral accords such as the ABM Treaty. Were Paxsat and its associated administrative mechanisms to be formally associated with the Treaty, renegotiation would be necessary. Moreover, if Paxsat is seen as a verification asset at the disposal of a group of states other than the Superpowers, the ABM Treaty itself would have to be made into a multilateral accord. This, in turn, would pose serious problems since key provisions of the existing treaty are incompatible with its multilateralization.

There is one final scenario which would involve the operation of Paxsat in other than a multilateral treaty context. This would envisage Paxsat as a 'stand-alone' verification asset at the disposal of a state or group of states, designed to verify the adherence of other

3.2 Multilateral Verification and Arms Control Agreement (Continued)

states to particular arms control agreements. For example, this could involve the policing of the ABM Treaty by a group of states who are not parties to that agreement. The rationale for such a scenario would presumably be based on a desire to ensure that the parties are adhering to the agreement, on the assumption either that existing verification assets are inadequate, or that there might occur some sort of collusive violation of the agreement by the signatories.

However, problems abound with this scenario.

(a) Since by definition neither Paxsat nor the states involved in its operation would be legally linked to the treaty or treaties to be verified, there would be no legally established means for discussing compliance issues arising from the operation of Paxsat. Although Paxsat itself and its associated administrative structure might have international legal standing by virtue of agreements signed between the participating states, there would be no legal linkage between this structure on one hand and the agreements to be verified on the other.

(b) It is generally accepted that verification and compliance processes are conditioned by the political context in which they operate. What in one political context might be construed as a violation of an agreement might be seen in another as an activity which is either allowed by the agreement or a violation insufficiently serious to warrant a major political conflict.

A stand-alone verification capability operating outside the context of the political relations between the treaty parties would interfere with this relationship between verification and politics, and might well create unnecessary political problems between the parties. In addition, since compliance issues arising between the parties are subject to private consultations, through such mechanisms as the Standing Consultative Commission created by SALT I, independent 'findings' could well interfere with this process, generating more problems than they would solve.

3.2 Multilateral Verification and Arms Control Agreement (Continued)

Over and above these questions, however, is the issue of practical politics. It is doubtful if a state or group of states allied with one or other of the Superpowers would find it politically prudent to police agreements between the Superpowers which they themselves have pledged to adhere to and verify. The US and USSR would find this representative of a lack of trust, and would see it as constituting interference, if not an attempt to secure intelligence on Superpower military programs.

It would seem, therefore, that the most logical scenario for the operation of a Paxsat system is in the context of an arms control agreement which is multilateral in nature, involving the Superpowers and other states, with Paxsat as a verification asset formally legitimated by the treaty itself. It is to the plausibility of this scenario that the discussion now turns.

3.3 Bilateral Versus Multilateral Outer Space Arms Control

The question of the 'participatory status' of an outer space arms control process and agreement may be divided into three distinct parts. First, to what extent can it be anticipated that outer space arms control negotiations will be conducted bilaterally between the superpowers, or multilaterally among a group of interested states? Second, if this process is bilateral in nature, can it be anticipated that the resulting agreement will be opened to other parties for signature? And third, if states other than the Superpowers are allowed to participate in the negotiations and/or the resulting agreement, what role can be envisioned for those states or other institutions in the implementation of the treaty provisions?

3.3.1 The Negotiation Process

There is a clear historical pattern to superpower perspective and behavior relating to bilateral approaches to arms control. In general terms, neither the US nor the USSR has demonstrated a willingness to negotiate over critical central-strategic issues except through direct bilateral channels. Central strategic

3.3.1 The Negotiation Process (Continued)

issues may be defined as matters whose importance to the security of states is perceived by both to be paramount. This approach has been favored for several reasons.

(a) Given the importance of the issues and assets under discussion, direct participation by others in the negotiation process has been considered imprudent by both Superpowers. In certain cases, consultations with Allies take place when their territory or interests are implicated, but a formal negotiating role for them is eschewed.

(b) Since such negotiations generally relate to assets which are solely owned by the Superpowers, and are deployed on, in or over Superpower or international territory, there is, in a strictly legal sense, no requirement to seek the acquiescence of other states in agreements concerning these systems.

(c) As a result of the nature and importance of the systems in question, national security data and intelligence have been an inevitable part of these negotiations; disclosure of such information to third parties has been considered unwise.

(d) It is widely recognized that the process of bargaining between two parties with very different strategic programs, interests and perspective is in and of itself sufficiently delicate and difficult to make the involvement of other states, with their own interests and perspective, undesirable in terms of managing the negotiating process and producing a successful outcome. The most important area where these considerations have applied is strategic nuclear arms control where all negotiations in this field have been purely bilateral in nature.

Multilateral arms control negotiations, on the other hand, have been pursued under a different set of conditions. In general terms, a multilateral approach has been adopted when the issues under discussion have demanded it, or when the probable impact of an agreement is sufficiently secondary in a military sense, to allow

3.3.1 The Negotiation Process (Continued)

far more open and less controlled (or controllable) negotiations. In addition, multilateral negotiations have been favored in conditions where one of the aims of the talks is the improvement of general political relations between a particular set of states.

For example, the ongoing talks on conventional force reductions in Europe (the MBFR negotiations) are of necessity multilateral (although conducted on a bloc-to-bloc basis) given that the territory and troops of states other than the Superpowers are under negotiation. Similarly, the recently convened Conference on Disarmament in Europe (the CDE) is by its very nature multilateral both in substance and geographic scope. Moreover, a primary purpose of both sets of negotiations is political, rather than strategic, and narrowly defined. This political effect is to a large extent a function of the multilateral nature of the talks themselves in terms of broad East-West dialogue and confidence building.

There is in addition an alternate hybrid pattern of negotiation which has been followed in some contexts. In situations where the subject matter(s) under negotiation has a clear multilateral dimension (as a result of prevailing or projected deployment patterns, territorial considerations, etc.), but where the US and USSR determine that central strategic interests are at stake, a 2-track process has developed: formal, multilateral talks have been supplemented by private bilateral approaches.

Chemical weapons arms control illustrates this pattern. The proliferation of existing or projected capabilities requires the involvement of states other than the Superpowers in negotiating any meaningful chemical weapons arms control measures. At the same time, a high proportion of the chemical weapons stockpile is owned by the Superpowers and the strategic implications of these assets for deterrence in Central Europe are considered central by both states. Hence, while multilateral negotiations proceed in Geneva at the CDE, the US has sought private, bilateral talks with the USSR.

3.3.1 The Negotiation Process (Continued)

On the basis of this historical pattern, it can be plausibly asserted that it is unlikely that serious negotiations concerning outer space arms control will be conducted on a multilateral basis. This assessment is based on several factors.

(a) The Superpowers are the primary owners and operators of satellite systems for military use.

(b) Weapons for use in outer space are currently deployed and under development only by the Superpowers. This is likely to remain the case for the foreseeable future.

(c) Because satellites and other space systems are considered to be national territory, and since space itself is considered to be international in nature, the involvement of other parties is legally unnecessary.

(d) Satellites and weapons technologies and systems are considered by both Superpowers to be critical to their central strategic interest.

(e) The sensitive nature of these systems in terms of technological characteristics and capabilities may be such as to make the Superpowers reluctant to disclose such information through a process of multilateral negotiation.

(f) The issues confronting outer space arms control negotiations, ranging from differing interest to problems of definition and verification are sufficiently difficult so as to create a lack of interest in third party involvement, which might be seen as unhelpful interference.

Historical precedent would seem to support this assessment. The most significant restrictions on Superpower military activities in space are embedded in bilateral agreements which were negotiated between the two parties, in particular the ABM and SALT I and II accords. Restrictions contained in multilaterally arrived at agreements, such as the Outer Space Treaty, are widely considered to be less significant in terms of their consequences.

3.3.1 The Negotiation Process (Continued)

In addition, the only previous set of arms control negotiations concerning antisatellite weapons, the focus of much of the current military activity and arms control debate, were bilateral in nature. More recently, the US has opposed the multilateralization of outer space arms control negotiations at the CDE in Geneva, partly on the grounds that the most productive and prudent approach would be private US-USSR talks. And the current bilateral approaches between the Superpowers concerning possible other space arms control negotiations are premised on a strictly bilateral negotiating process.

The most plausible scenario for the negotiation of an other space arms control agreement is, therefore a set of bilateral talks. The implications of this for the plausibility of Paxsat as a multilaterally operated verification capability are as follows:

(a) The precise context of an outer space arms control agreement, together with associated verification and compliance arrangements are likely to reflect a mixture of US and USSR interests, rather than those of third parties.

(b) It follows that the Superpowers themselves will have to be convinced of the value of a multilaterally operated Paxsat system if this system is to exist in organic connection with an arms control agreement.

(c) This process will require the multilateralization of a bilaterally negotiated agreement in order to formally link third parties to its provisions.

(d) The parameters for the operation of Paxsat will have to be embedded in the agreement itself, and therefore the Superpowers themselves must integrate this system into their verification discussions at a relatively early stage. This is not to suggest that specific verification technologies require identification in the agreement, but rather that the legal framework for

3.3.1 The Negotiation Process (Continued)

the existence and operation of a multilateral verification structure will have to be established in order to avoid subsequent treaty renegotiation.

3.3.2 The Participatory Status of an Outer Space Arms Control Agreement

The above analysis suggests that it is unlikely that an outer space arms control agreement will be negotiated on a multilateral basis. If the most plausible scenario for the operation of Paxsat is in the context of a multilateral treaty containing provisions for multilateral treaty administration and verification, the implications of this assessment are serious indeed. However, it does not necessarily follow that a bilaterally negotiated agreement need result in an agreement which is bilateral in terms of participation. As noted earlier, the accession of other states to bilaterally negotiated conventions has been sought in situations where their participation is seen as enhancing the effectiveness of the accord. In particular, when the Superpowers have an interest in ensuring that a prohibition on certain activities does not apply solely to them, multilateralization may be sought. Such interest usually reflects a desire to avoid treaty circumvention through the transfer of technology and capabilities to non-signatory states and to preclude the proliferation of capabilities through indigenous production by non-signatory states.

In addition, states other than the Superpowers may have an interest in encouraging the multilateralization of bilaterally negotiated accords. In areas where military developments adverse to the security interests of states other than the Superpowers are occurring or anticipated, those states may encourage the Superpowers to negotiate an arms control limitation agreement. Such persuasion may take the form of a willingness to sign an agreement which one or both Superpowers might deem inadequate in the absence of assurances that proliferation or circumvention could not take place.

Taken together, these considerations may well apply in the area of outer space arms control.

3.3.2 The Participatory Status of an Outer Space Arms Control Agreement (Continued)

(a) From a Superpower perspective, both the USSR and the US are likely to be sensitive to the possibility of a threat to spacebased assets resulting from third party activities and programs. In the outer space context, this concern may be heightened by the increasing reliance of both Superpowers on satellite systems and the inherent vulnerability of those systems to attack, even from relatively minor ASAT capabilities. There may, therefore be an incentive for multilateralization arising from a fear of medium to long term proliferation or treaty circumvention.

(b) Non-superpower states who rely on satellite systems for various tasks may fear that these assets will become vulnerable to attack from Superpower outer space weapons. In addition, certain states such as Britain, France and China may fear that the development of spacebased Ballistic Missile Defense (BMD) systems may erode the credibility of their independent nuclear deterrent. Hence, apart from a general assessment that the weaponization of outer space may be destabilizing, there is also a feeling that such a development would accentuate the asymmetry in strategic power between the Superpowers and other countries. Recognizing that a fear of proliferation or treaty circumvention may be a barrier to successful negotiations, an offer by certain non-superpower states to sign an outer space accord may enhance the chances of an agreement. In addition, the application of political pressure from allies, who in general terms are opposed to the weaponization of outer space, may encourage the multilateralization of an outer space arms control accord.

There are two final considerations which may be seen as increasing the prospects of a multilateral agreement:

(a) On the assumption that the essential features of an agreement will be negotiated bilaterally between the Superpowers, as argued above, and that the Superpowers would retain the prerogative to

3.3.2 The Participatory Status of an Outer Space Arms Control Agreement (Continued)

consult bilaterally on questions of compliance and verification, there are a few, if any, costs to the multilateralization of an outer space arms control accord. Insofar as the essential features of an agreement would reflect a mutually agreeable set of constraints, and insofar as both Superpowers could subsequently approach each other at will on treaty-related matters, neither need fear the intrusion of 'extraneous' interests or constraints.

(b) Given the prevailing state of technology and deployments, it is probable that an outer space arms control agreement will involve a prohibition on certain types of technologies, activities and/or deployments. Since prohibitions are absolute restrictions as opposed to limitation on already existing deployments or technologies, the multilateralization of an arms control agreement becomes considerably easier. For example, in the case of SALT I and II, the agreements involved specific limitations on existing US and USSR forces in terms of quantity and quality. Multilateralizing these accords would be impossible given the very nature of the treaty provisions.

However, in the case of prohibited activities, technologies and deployments, there are no legal or logical problems in opening up an accord for multilateral signature. This is demonstrated by existing multilateral treaties and negotiations which are prohibitory in nature, such as the Geneva Protocol, the Antarctic Treaty, the Limited Test Ban Treaty, the Outer Space Treaty, the Biological Weapons Convention and the ongoing chemical weapons arms control negotiations.

Based on the above considerations, therefore, it is at least plausible that an outer space arms control accord may be multilateral in nature. As noted earlier, this is of relevance to the operation of Paxsat since the formally sanctioned participation of non-superpower states in the treaty itself will be required for

3.3.2 The Participatory Status of an Outer Space Arms Control Agreement (Continued)

political if not legal reasons. However, the multilateralization of an other space arms control accord is not in and of itself sufficient to mandate or justify the existence of a multilateral verification mechanism. It is to this question that the discussion now turns.

3.3.3 Multilateral Participation in Treaty Administration and Verification

Although it is plausible that the Superpowers may seek to multilateralize an outer space arms control agreement, it is by no means clear that such an agreement would thereby sanction a multilateral verification capability. Indeed, the only multilateral arms control treaty currently in force which contains provision for a specific multilateral organization designed to ensure compliance with treaty provisions is the Treaty for the Prohibition of Nuclear Weapons in Latin America. Other multilateral agreements such as the Biological Weapons Convention, the Partial Test Ban Treaty, the Antarctic Treaty and the Outer Space Treaty simply bind the parties to adhere to the terms of the agreement, and to verify compliance with the agreement through their own national resources. Questions of compliance are to be resolved on an ad hoc basis through consultation between the parties. There are no ongoing administrative mechanisms or verification assets at the disposal of the signatories as a group.

Therefore, the involvement of third parties in an arms control regime for outer space may require only that these states ensure that their national policies and programs conform to the provisions of the agreement, with verification of compliance consisting of the application of so-called National Technical Means (NTM's). In such a case, a legally sanctioned relationship between a multilateral verification capability such as Paxsat and a multilateral arms control regime would not exist. It must be asked, therefore, whether or not there exist incentives for the Superpowers and other states to sanction the inclusion of non-superpowers, and non-national verification assets in a compliance regime associated with an outer space arms control agreement.

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

This question may in turn be broken into two logically distinct components. To what extent is it plausible to envisage a multilateral organization empowered to administer the treaty and discuss questions of compliance? And to what extent is it plausible to envisage such an organization possessing and operating verification assets such as Paxsat for the purpose of ensuring compliance?

In considering these questions, the following points bear consideration. First, the interest of the Superpowers (and other states) in creating a multilateral verification capability is likely to be related to the degree of difficulty anticipated in verifying the provisions of the treaty, together with the potential significance of violations should they occur. Simply put, an agreement which is easy to verify using existing NTM's is unlikely to prompt interest in a multilateral verification capability. Similarly, an agreement which is unlikely to be violated (for reasons of prudence or military logic), or whose provisions are strategically inconsequential is equally unlikely to prompt such interest.

However, significant all encompassing outer space arms control agreements may be difficult to verify and the consequence of violations extremely serious. For example, the sensitivity of an ASAT arms control regime to small numbers of violations is considered to be high. A small number of concealed ASAT tests may be sufficient to develop confidence in an ASAT system adequate to contemplate operational deployment. Similarly, a relatively small number of deployed ASAT weapons may be sufficient to constitute a serous threat to key satellite communication, navigation, early warning and intelligence assets. The 'elasticity' which exists with regard to nuclear weapons, where a small number of warheads or launchers in excess of agreed limits would not create fundamental asymmetries or instabilities, does not exist with ASAT's.

In addition, it is widely agreed that a satisfactory verification regime for an outer space arms control agreement will be difficult to negotiate and to implement. This arises from the following factors:

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

(a) The small number of tests and deployed systems deemed sufficient to constitute a significant threat.

(b) The possibility of ground tests or tests of component systems in space in a mode difficult to detect.

(c) The multi-functional nature of certain technologies and launchers (e.g. rocket boosters, aircraft, lasers).

(d) The relative ease of concealment of certain destructive mechanisms (especially conventional or nuclear explosives).

Indeed, these considerations have been put forward by the current US Administration as an argument against the pursuit of a comprehensive ban on weapons in outer space.

Compounding this situation is the fact that certain cooperative verification techniques of relevance to other arms control areas are not possible in the outer space realm. On-site inspection, for example, although available as an adjunct to NTM's for certain terrestial activities, is of little relevance to certain space related activities. If the system in question are spacebased, on-site inspection may be impossible, unless the parties are willing to contemplate system retrieval by other states for the purpose of examination.

Based on these considerations then, there is a prima facie case for the maximization and multiplication of verification assets at the disposal of the signatories to an outer space arms control regime. This need not, however, imply either a multilateral treaty administrative and compliance body or verification assets under the control of such a body. Other options include an increase in NTM's, or a simple assessment that what is available is adequate, though less than ideal. It is therefore necessary to outline

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

considerations which may turn a recognition of the difficulty of verification and the potential seriousness of violations into a desire to multilateralize the verification process.

First, it must be recognized that given the significance which both Superpowers attach to outer space military activities, neither is likely to acquiesce in an agreement where the verification technology and the verification authority resides solely with a multilateral organization. Both Superpowers possess technology more sophisticated in its ability to verify an arms control agreement than that at the disposal of third parties. This is unlikely to change in the immediate future, though both France and China are currently experiencing R&D effort in this area. Moreover, national control of this technology allows verification activities to proceed according to national interests priorities, unimpeded by an international political process. These activities can also proceed in secret, a factor which avoids international political controversy and allows observation without notification. In addition, the technology involved in verification is sufficiently sophisticated that neither Superpower would be willing to expose its sources. This stems from both the quality of technology involved and the desire to maintain some uncertainty in terms of the knowledge of potential adversaries as to the activities and capabilities of verification systems. And finally, given that verification technology yields sensitive information about the military programs of other states, neither Superpower would be willing to grant others unimpeded access to such data. To do so would amount to exposing sensitive intelligence data to all parties to an arms control accord, and would inform each Superpower of the precise state of knowledge concerning the other's military program.

Based on these factors, it can be concluded that neither Superpower would be willing to rely exclusively on multilaterally controlled systems for the verification of an outer space arms control agreement. For the Superpowers, primary reliance for verification in the outer space arms control area, as with nuclear weapons, is likely to remain with National Technical Means.

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

A second factor concerns the political process associated with the verification of arms control agreements. Historically, both Superpowers have sought to resolve compliance issues bilaterally and secretly, either through ad hoc channels or through institutions such as the Standing Consultative Commission created by SALT. This approach has been preferred for several reasons. Confidential bilateral approaches generally avoid the politicization of compliance questions which would be more likely to result from an open process of discussion conducted either bilaterally or multilaterally. A bilateral approach also avoids the sharing of sensitive information with others.

In addition, since most compliance questions relate to activities which are ambiguous either in terms of their nature or their relationship to specific interpretations of treaty languages, confidential bilateral approaches allows the process of clarification to take place on a routine basis without unnecessary international political scrutiny or interference. This in turn, reflects the fundamentally political nature of verification. As noted earlier, what is or is not perceived as a questionable activity or an outright violation of an agreement depends to a large extent on the overall context of Superpower and East-West relations. Historically, in a period of detente, different criteria have been employed in evaluating treaty adherence than in a period of tension. Taking compliance questions out of this bilateral political context and placing them in a less controllable multilateral context might do violence to this delicate contextual relationship between verification and politics.

Based on these considerations, the Superpowers are likely to resolve most compliance issues associated with outer space arms control on a closed, bilateral basis. Therefore, it can be concluded that in terms of verification technology, the Superpowers will continue to rely on National Technical Means, and in terms of political process for resolving verification issues, the current pattern of bilateralism will be maintained. While this may be seen as arguing against the

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

plausibility of a multilateral verification capability and process, this need not be the case. These factors may simply suggest certain technological and political parameters for the operation of such a system. Indeed, a multilateral verification process and capability may complement rather than conflict with primary reliance on national verification capabilities and bilateral Superpower relations for the resolution of compliance issues.

In particular, the following points emerge from this analysis:

(a) Since the Superpowers will insist on relying upon existing verification capabilities and approaches, a multilateral verification system for outer space will be unacceptable unless there is no interference with the continued operation of this verification system. The use of NTM's for the verification must be legitimized by any outer space arms control agreement, as must the right of individual states to resolve compliance issues bilaterally. In addition, for the reasons outlined earlier, there can be no obligation to either share data derived from national verification assets or to bring compliance issues before a multilateral body.

(b) Given that the Superpowers will retain existing verification assets, there is no requirement for the creation of a multilateral verification capability which duplicates the technology currently at the disposal of the US and USSR. Any new verification assets must be seen as supplements rather than replacements. And since the Superpowers will continue to monitor each other using NTM's, there is no need to create a similar (and redundant) capability at the multilateral level. This implies that the quality of the technology at the disposal of a multilateral treaty verification body must be judged not in isolation, but in relation to the total verification assets at the disposal of the parties to the agreement.

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

In a more positive sense, however, what case can be made for the requirement for a multilateral verification regime? First, since in general terms the Superpowers will be extremely sensitive to the possibility of any violation of an outer space arms control agreement, given the possible strategic consequence, additional verification assets of either an organizational or technical nature should be welcome, if configured in a manner which does not militate against the Superpower prerogatives noted earlier. Second, given the difficulties associated with verifying an outer space arms control agreement, it is possible that compliance issues may arise more frequently than is the case with existing accords in other areas. The availability of additional capabilities and avenues for the resolution of these problems might well be perceived as advantageous by the parties to an agreement.

(c) It is arguable that the Superpowers and others would see some value in an institutional mechanism for the exchange of data, discussions concerning activities and programs, and the resolution of compliance issues through debate and/or study. The existence of a properly configured, multilateral 'court of appeal' would provide a legal framework for the conduct of activities relating to compliance and verification questions which could not be resolved on a bilateral basis. The 'deterrent' effect of a legally constituted multilateral forum for the arbitration of disputes could be seen by the parties to an agreement as an important asset in ensuring treaty adherence, a confidence building measure for outer space arms control.

The support of non-superpower states for a multilateral verification body would arise from additional factors. Since most states for the foreseeable future are unlikely to be economically or technically capable of mounting and sustaining a national, stand-alone capability to verify an outer space arms control agreement, access to a body design for that purpose might be attractive. In addition, to the extent that

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

these states support an arms control agreement, and perceive a multilateral verification body as an encouragement to the superpower for the reasons outlined above, third party support for this concept is at least plausible in principle.

However, support for a multilateral treaty administration and verification body need not imply support for a multilaterally controlled verification capability. It could well be the case that states could judge existing capabilities as being adequate for the verification of an outer space arms control agreement, or feel that the costs associated with such a system outweigh the possible benefits in terms of verification. While it is beyond the scope of this analysis to deal with the latter point, it is necessary to examine the plausibility of a scenario where states feel that such a capability is in principle desirable.

As noted earlier, the Superpowers will continue to monitor compliance with an outer space arms control agreement using NTM's, and will probably choose to resolve compliance issues on a bilateral basis. It is possible, however, that in the event of detection of a possible violation and unsatisfactory resolution of the issue through bilateral consultations, the issue would be brought before the treaty administration and verification body for further action. At this point, various options could be pursued ranging from discussion and examination of nationally provided evidence to a full-pledged investigation. Such an investigation, however, if it is to avoid total reliance on information provided by the grieving party (which might be portrayed as questionable by the party whose activities are under examination, or by others) must possess its own technical assets. Such assets, if operated by the treaty administration and verification body, would be perceived as 'objective' in terms of data thereby generated. The possibility of independently derived findings would in turn present the potential violator with the prospect of the sanction of the collective group by signatory states. This might well be viewed as far more serious than the findings of an individual state, findings which could be portrayed as biased or

3.3.3 Multilateral Participation in Treaty Administration and Verification (Continued)

falsified. Given the deterrent effect such a capability might have on potential violators, support for independent multilateral technical means of verification may be forthcoming. Such assets would be the functional equivalent of on-site inspections or investigations of terrestial activities in other arms control contexts.

In addition, investigations conducted by a multilateral verification body using its own assets would avoid perceptions of hostile activity. Given that such activities, as discussed below, there would be due notification of the party under investigation, thereby avoiding surprise. Moreover, the relative level of technological sophistication, combined with knowledge of the precise capabilities of the system should alleviate fears either of intelligence gathering or of actual attack.

Finally, based on the scenario as described thus far, the degree of system sophistication and completeness need not approach that of a stand-alone verification capability. Data provided by the parties to the verification body through the use of their own NTM's would greatly relax the technical requirements of a functional Paxsat system. As described, the role of Paxsat as a multilaterally controlled verification asset at the disposal of a multilateral treaty verification body would be primarily that of an arbitrator or 'court of last appeal'. In this context, an evaluation of Paxsat should be based not so much on the quality of the system in strictly technical terms or on its redundancy in relation to other verification systems, but rather in relation to its ability to perform this political role. Evaluated according to this criterion and given the importance of the function itself, support for Paxsat from the parties is at least plausible.

3.4 Guidelines for Organization and Decision-Making

Although it is beyond the scope of this study to identify the precise organizational structure of a multilateral treaty administration and verification body, or to spell out in detail the decision-making process associated with the use of Paxsat, the following may serve as relevant guidelines in these areas:

3.4 Guidelines for Organization and Decision-Making (Continued)

(a) The organizational structure and decision-making process must recognize the political nature of the arms control agreement and associated verification provisions. Decisions to conduct various verification activities must be subject to political control by the signatory states. This implies an organization governed by official representatives from the signatory states rather than an executive agency empowered to conduct investigation at will.

(b) The special interests and prerogatives of the Superpowers and to a lesser extent other space powers, must be recognized in the decision-making process. This may involve permanent representation by these states in the crucial decision-making body, and a voting procedure which implicitly confers more power on these states.

(c) Voting procedures for the initiation of verification activities, including a Paxsat mission must avoid a veto on the one hand and abuse of the investigatory process on the other through excessive use by individual states or group of states. This implies a voting procedure carefully integrated with the composition of the governing body or bodies which ensures that activities can be neither permanently foreclosed nor persistently launched at will against a particular state.

(d) In relation to the point just made, the governing body must reflect the political complexion of the signatory states. This implies an East-West balance, with LDC participation congruent with their significance in the outer space area.

(e) A technical secretariat will be required to conduct investigations and evaluate data obtained from Paxsat or given by signatory states. This group of experts must be beyond direct political control, but must be representative of the

3.4 Guidelines for Organization and Decision-Making (Continued)

signatory states. In the event of an investigation, experts from both the grieving party and the alleged offender would be excluded from any activities.

(f) Should an investigation be authorized, including those which involve a Paxsat mission, the results of such inquiries should be disseminated immediately to the signatory states. Since a political decision would be required to instigate an investigation in the first place, the results of such an investigation should be circulated to all parties as soon as they are available.

(g) In order to secure agreement on the requirement for a multilateral treaty administration and verification body and to ensure its proper configuration, the essential organizational structure and decision-making process must be outlined in the treaty itself.

4.0 THE PAXSAT CONCEPT - A VERIFICATION ROLE ANTICIPATED AND AN OPERATIONS SCENARIO POSTULATED

4.1 Introduction

The objective of the analysis is to develop a plausible political and operational scenario for Paxsat A within which remote sensing technologies mounted in a space vehicle may be deployed and utilized in the verification of possible violations of an outer space arms control or limitation agreement. This report is limited to the verification of space-to-space and space-to earth weapons systems in which the weapon system is deployed in stable orbit following test and launch phases of its life cycle.

To this end, section 4.0, by taking significant issues from sections 2 and 3, and analyzing these in terms of conventional wisdom strategies, implications and consequence will:

(a) Postulate a scenario of the most likely threat, and the character and magnitude of that threat, relevant to the requirement of embedding a verification capability into an outer space arms control/limitation agreement.

(b) Postulate the most likely plausible political scenario and its implications in prescribing and defining the verification role and mandate.

(c) From the analysis described in (a) and (b) above, develop the conceptual description and operational profile of a plausible Paxsat system which conforms to the political scenario and establishes technical parameters for the systems operational capability and deployment.

4.2 Space and Space Weaponry (Character and Magnitude of the Verification Requirement)

Section 2.0 of this report has provided a broad description of the present inventory of spacecraft by generic types, the most widely used orbits and has within the parameters of unclassified information, discussed the possible space weapons that might be deployed. These spaceborne weapons are grouped into two major families:

4.2 Space and Space Weaponry (Character and Magnitude of the Verification Requirement)(Continued)

(a) Space-to-space weapons

(b) Space-to-earth weapons

Additional reference and comment has been made with respect to operational factors and physical features, which make such weapon systems either more or less difficult to verify.

With respect to the present situation in space, it is significant in the political context to note that of the approximately 5,000 spacecraft deployed (comprising active spacecraft, spacecraft presumed dead, and space junk) that 90% of these are owned by the US and USSR with the remaining number owned by assorted non-superpower nations.

The Superpower assets are divided, approximately as follows:

(a) US - Military - 50%
Civilian/Commercial - 50%

(b) USSR - Military - 80%
Non-Military - 20%

(c) 70% of all satellites are military.

None of the inventory of national space assets is known or suspected to be a weapon system of the space-to-space or space-to-earth categories.

The foregoing spacecraft population is deployed in orbits in three magnitudes of distance from earth:

(a) 200 km or less

This does not provide stable orbit and such spacecraft have relatively short lives.

4.2 Space and Space Weaponry (Character and Magnitude of the Verification Requirement)(Continued)

(b) 200 km to 50,000 km

This is a stable orbit range and spacecraft in this area are candidates to be targets for anti-satellite weapon systems if they are also rewarding targets.

(c) 50,000 km and beyond

These are stable orbits but are at extreme range for most antisatellite weapon systems. Such systems, if deployed in optimal positions relative to those targets would be quite distinguishable.

Of the space weapons systems referred in section 2.0, each represents somewhat different levels of technological sophistication not only in terms of the weapon systems capabilities but also in the ways in which they are deployed. All are expensive to deploy and it must be assumed that they would not be deployed unless such deployment can be optimal in terms of target of substance. Targets of opportunity do not seem to be a cost effective role for space weapon systems. This requirement of course eliminates most of the space assets inventory as target candidates for antisatellite systems. Although colliding and explosive weapon spacecraft are probably well within state-of-the-art, deployment and range factors are limiting to the scope of these weapons as antisatellite systems. In the form of space-to-earth weapons, only nuclear explosive spacecraft appear to have any plausible role (either single or multiple warheads), but precise targeting presents difficulties and other systems may prove to be substantially more cost effective. The same limiting factors may not apply so stringently to space-to-earth chemical weapons, however, although meteorological factors will be important as an additional dimension which could make their effectiveness uncertain at critical points in time. The point of the argument is that spaceborne weapon systems whether space-to-space or space-to-earth dedicated to be practicable, will require high levels of optimization and will be focussed on targets which are of such a nature as to justify the complexity and cost of the weapon systems envisaged.

4.2 Space and Space Weaponry (Character and Magnitude of the Verification Requirement)(Continued)

While the foregoing assessment is considered valid for state-of-the-art systems currently feasible, a different scenario may be justifiable for the employment of X-ray lasers and particle beam weapons that may become feasible by the turn of the century providing the population and character of targets should justify the development of these technologies. (It is important to note here that the Paxsat A context does not include the ground based components of ballistic missile defense.)

Essentially therefore, in assessing space and space weapon factors applicable for postulating a first generation space weapon verification system in the Paxsat context (this is to say in the timeframe of the next 15 years), a primary focus is on possible space weapon systems which do not involve Twenty-First Century technology. This is not to suggest, however, that a Paxsat would be ineffective against more sophisticated weapon systems. In reviewing the probable character and configuration of such potentially feasible future systems, they seem to present less challenge to the verification requirement than current more primitive feasible systems.

As pointed out earlier, not all spacecraft at present deployed (or at any given time for that matter) constitute logical and rewarding military strategic or tactical targets for space-to-space weapon systems. Table 2-2 of section 2.2 has listed sixteen types of ASAT targets. The highest risk space assets are suggested to comprise:

(a) Dedicated targeting systems which direct earthbased weapon systems

(b) Spacebased weapon systems

(c) Surveillance and reconnaissance systems

(d) Navigation systems

4.2 Space and Space Weaponry (Character and Magnitude of the Verification Requirement)(Continued)

In this context, it should be noted that the first two categories of space assets are not currently deployed. It is anticipated that targeting spacecraft will probably be deployed at distances (100,000 km or more) which would make them relatively safe from the shorter range more primitive space weapons currently feasible. The present space population does not include any operational spacebased weapon systems. Advanced space weapon systems such as sophisticated laser systems and particle beam weapons capable of operating with precise discrimination and at extreme ranges are unlikely to be feasible within 15 years. Consequently, spacebased counter ASAT weapons are unlikely to appear prior to the turn of the century.

Taking that military surveillance and reconnaissance satelites and military navigation satellites, with the prospect of the introduction of deep space targeting satellites in the relatively near future represent the highest risk assets, the volume of space in inclinations from 60° through 105° appear to be the most sensitive. Such spacecraft are mostly deployed in distances ranging from 200 km to 1500 km (excluding more distant targeting spacecraft). Such areas are easily accessible by currently feasible spacebased weapons of geosynchronous orbit (36,000 km) were an additional 800 items reside.

Therefore, in the context of what might be classified as the midterm (up to 2,000 to 2,005), the highest risk space assets appear to be the categories of surveillance and reconnaissance systems and navigation systems. The most sensitive of these groups are those which embody real-time fast data processing features. Military systems of these categories are exclusively the property of the US and USSR. This point is of particular significance in contemplating the substance of any possible outer space arms control agreement (including one which embodies a verification feature), since vested interest will be a major factor or force of the political process which may lead to such an agreement.

4.2 Space and Space Weaponry (Character and Magnitude of the Verification Requirement)(Continued)

Section 2.0 has made reference to the extent of the current thinking with respect to the USSR space weapon threat and also to statements by US authorities on the ways in which they would counter that threat. The currently perceived threat to US spacecraft seems to consist of Soviet colliding or explosive weapons in the midterm but no doubt including more sophisticated technological challenges in years to come. Current US focus (argued from the popular defensive/protective philosophical base rather than from first strike/pre-emptive principles) is to counter Soviet weapons that might be put into place.

Irrespective of the differences in philosophical positions from which the arguments and statements come, it would seem clear that taking into consideration the relative positions of the two Superpowers as the predominant owners of military operated space assets, no unplanned verification capabilities will be tolerated in a Paxsat type system. On the other hand, predetermined mandates agreed between the major owners can result in an arms control agreement defining parameters which act as a deterrent to the exploitation of outer space beyond those bounds.

4.3 Paxsat Verification Role and Mandate in the Political Context

Section 3.0 of this report has provided a comprehensive view of the political process and the relationships among and between nations and ideological 'blocs' which play their part in the achievement of an acceptable form of outer space arms limitation agreement. Within the limits and constraints discussed in section 3.0, this section postulates what is believed to be the most plausible political scenario and its implications both for the verification process and for Paxsat.

Recognizing the dominant positions of the US and USSR in the world milieu, the special relationship between these two Superpowers, and their ownership of the

4.3 Paxsat Verification Role and Mandate in the Political Context (Continued)

preponderence of space assets, it is difficult to envisage any arms control agreement and associated verification function which may seem to intrude upon the exclusive rights of either. Consequently, no independent third party notion of such a treaty or agreement seems plausible or achievable. It is doubtful that such an initiative would have any support of substance from either East or West bloc aligned nations. The cost to a third party group of nations taking such an initiative would be extremely high and incentive to undertake investment of such magnitude relative to the scale of third party ownership of spacecraft at risk would not be likely to result in a highly effective verification capability. The technical effectiveness of the third party verification capability would always lag the technology of the Superpowers vis-a-vis each other and efforts to replicate such technology by the third party of nations could well be destabilizing and counter-productive.

Working from the position that an effective verification capability must be a part of any outer space arms control agreement or treaty, the following points suggest certain parameters that are important politically in the achievement of an agreement and which play a significant role in development a plausible Paxsat operational concept.

(a) The concept of the bilateral imperative must continue to be respected. This is to say that the Superpowers will continue to insist upon bilateral exclusive negotiation whenever their central security interests are involved.

(b) The exclusive proprietary technologies of the Superpowers vis-a-vis each other will continue to be restricted and unavailable for purposes of space weapon system verification.

(c) The verification methods and the verification process must be discriminatory in the sense of limiting its function and activity to areas outside the limits of the bilateral imperative and

4.3 Paxsat Verification Role and Mandate in the Political Context (Continued)

proprietary technologies. (Failure to achieve this would mean that data and intelligence collected by non-superpower operated verification system would be subject to superpower filtering prior to release to any verification authority established within an outer space arms control agreement. Failure in this respect could present insurmountable obstacles in achieving an agreement.)

(d) The verification system must function multilaterally and not seem to be focussed exclusively on the Superpowers. Consequently, multilateral participation would be extremely important.

In consideration of these factors, it appears to be most plausible that an outer space arms control agreement would be most likely achievable by following what section 3.0 has described as the 2-track negotiating process. This is to say that the two superpowers negotiate initially the crucial elements of a treaty for the control of outer space weapon systems which is tolerable in terms of their individual central security concerns and which defines the limits of verification acceptable in that context. The second stage of the 2-track process would be the multilateralization of that agreement followed by the establishment of the defined verification capability. Assuming the multilateralization of the agreement, and that it is executed as a multilateral agreement not focussed exclusively upon the Superpowers and the degradation of the bilateral imperative, it would follow that the verification capability also would be multilateral. Viewed in this way, the verification system becomes in essence a joint bilateral-multilateral verification function in two parts as follows:

(a) Bilateral exclusive areas of verification denied to the multilateral capability, or Paxsat operational mission, which have been defined bilaterally (and accepted on a multilateral scale).

4.3 Paxsat Verification Role and Mandate in the Political Context (Continued)

(b) A multilateral verification capability which is created through contribution of national technical means of the multilateral participants augmented by a multilaterally sponsored Paxsat. In this context, the Paxsat augmentation represents a space-to-space verification capability not in the inventory of non-superpower space assets.

It would be argued that such a formula for achieving an outer space arms control agreement preserves the bilateral imperative, creates a formula for verification which does not pose a challenge to the exclusive technology preserves of the Superpowers, and assures a situation in which the multilaterally sponsored capability does not aspire to achieve a level of technological sophistication which acts as an external technology escalation stimulus on the Superpowers.

In considering the validity of a joint agreement and verification role of this kind, there must be sufficient benefits to constitute an incentive for both parties to the joint undertaking. Realistically, it must be conceded that there is no conceivable outer space arms control agreement that will not be abrogated by either of the superpowers should there be considered to be justifiable cause to do so irrespective of the thrust of world opinion. On the other hand, there are steps that might be taken to reduce the risk of such actions.

Although the verification mandate of the multilateral based verification function is operationally limited by the bilateral imperative and is technically limited by the proprietary technological preserves of the superpowers, the following mutually beneficial plausible roles and missions of the multilateral function (in the context of two spacebased weapon families) can be seen to be of considerable substance and potentially a key element in reducing the risk of treaty violation or abrogation:

(a) The multilateral element of the verification function and capability provides the opportunity for a court of last appeal for the superpowers in bilateral disagreement or suspicion of violation.

4.3 Paxsat Verification Role and Mandate in the Political Context (Continued)

In this context, the existince of a Paxsat verification system could be a crucial instrument if it embodies a truly effective verification capability and does not duplicate available contributed technical means which might otherwise be mobilized.

(b) The multilateral element of the agreement verification capability would also provide a watchdog function against third party proliferation and would eliminate the need for either superpower to become directly involved. The third party non-proliferation verification role would also inhibit the transfer of space weapon technology from the superpowers to their bloc members.

(c) The multilateral verification element of the agreement would be the most logical body to deal with allegations of violations arising out of the multilateral group with respect to another member or members of the multilateral group.

(d) The existence of a multilateral verification capability provides a level of confidence for those non-superpowers which have assets deployed in space.

It could be argued that the most crucial roles of the multilateral verification capability are those which relate to resolving differences in perceptions which may arise within the bilateral imperative and those which concern third party proliferation in space weapon technolgies. It is not difficult to see that such a verification capability would have to be highly effective and relevant to the verification tasks which are implicit in these roles. Such a level of effectiveness is unlikely to be possible without significant augmentation of national technical means of the multilateral nations group. The operational requirements of these two roles will be the crucial determinants of the form and substance of multilateral verification capability.

4.4 Paxsat System - Operational Concept

Key points emerging out of sections 4.1 and 4.2 of this section, which have significance for the character and capability of a viable multilateral verification system, include the following:

(a) The number of potential spacebased weapons systems that might be justified in terms of cost effectiveness either in space-to-space or space-to-earth roles is not great.

(b) The technology requirement of the verification function is not extreme taking into account:

i) The limitation of the verification role in terms of bilateral imperative.

ii) The distinguishable characteristics of the more advanced and spectacular potential spacebased weapon system

(c) The act of verification is limited to situations in which the political process has been unable to achieve an acceptable consensus level of confidence and the final step of physical verification is considered mandatory.

(d) That each act (or mission) of verification requires specific political authorization.

An additional consideration is the matter of the extent of which the aggregate of contributed national technical means on the part of the non-superpower nations constitutes a verification system which is able to fulfil the role and missions postulated above.

It must be assumed that the multilateral verification system and capability to be envisaged is not confined to any single phase of the development cycle of a spacebased weapon system but that it applies, within the limits of political consent to any or all of the six detection sensitive phases as follows:

(a) Design and build

(b) Test at full scale/power

4.4 Paxsat System - Operational Concept (Continued)

(c) Deployment

(d) System test

(e) Deploy the stable orbit/position following test

(f) Change to alert status

Envisaging the role and conceivable missions of a Paxsat system as a space-to-space verification instrument augmenting contributed national technical means (consisting almost exclusively of long range earthbased surveillance and remote sensing facilities), the Paxsat role is relevant to (d), (e) and (f) above, but primarily to phase (e). Phases (a) and (b) verification or surveillance is more relevant to downward looking spacecraft or earthbased methods. Phases (c) and (d) are likely to occur more quickly than the political process can react and utilize Paxsat effectively. Phase (f) would involve Paxsat speculate deployment before the fact, but might also be detected by earthbased systems utilizing intelligence and data previously provided by Paxsat operating in a phase (e) surveillance and verification mission. While the Paxsat role is almost exclusively focussed on phase (e) of the space weapon development cycle when deployed and configured optimally for that role, it provides a unique high performance verification capability superior to earth-based systems.

The space and space weaponry scenarios combined with the political factors affecting multilateral verification and practical applications of space and remote sensing technologies offer options:

(a) Stand-off and/or close-in verification missions.

(b) Prior on-station deployment and/or event triggered deployment

(c) Dual mode optimized spacecraft performing both stand-off and close-in inspections or single mode optimized spacecraft.

4.4 Paxsat System - Operational Concept (Continued)

The choices to be made with respect to these options will be determined by the operational/tactical verification scenario which is compatible with the technical and political parameters which have been developed in this report.

There are certain features or characteristics of the foregoing options in system configuration which should be noted since they have some significance for defining a plausible tactical scenario.

(a) Stand-off verification implies relatively long ranges of 1000 km or more. In this mode, the spacecraft payload would be heavy and somewhat awkward if optical remote sensing was utilized. If imaging radar was utilized, the power demands would be extreme. Pre-event deployment in a space patrol mode is implicit. If the spacecraft was also to perform close-in inspection missions, a maximum bus capability would be mandatory and trade-off problems between payload and fuel load would be severe (possibly unacceptably limiting on spacecraft endurance and ability to execute its assigned mission).

(b) Single mode spacecraft optimized either for stand-off long range surveillance or for close-in inspection are likely to be more effective than dual mode spacecraft. In choosing between the two modes, the optimized close-in inspection mode would be superior to the stand-off long range mode.

(c) A relatively short range 100 km stand-off capability could be incorporated into an optimized close-in inspection spacecraft and would be highly effective especially if the close-in inspection preceded the stand-off surveillance.

(d) Close-in inspection spacecraft may be maneuvered from space patrol positions into relatively close fly-by contact (1 km to 5 km depending upon the relativity of the two initial orbits) with the verification target. Similarly, optimized

4.4 Paxsat System - Operational Concept (Continued)

spacecraft may be launched on command from earth directly to the target to close ranges (2 km to 5 km). The exposure of the spacecraft's remote sensing equipment to the target may be extremely short in situations where the verification spacecraft is in a space patrol mode (fly-by) and the verification target is in a different orbit.

(e) Maximum time to target rate for a space patrol deployed spacecraft would approximate ninety days although statistically, the period would be less. Time to target rate for an earthbased Paxsat would approximate two months assuming maximum pre-programming and readiness state. The less attractive average response time of the earth launched Paxsat is offset by higher levels of endurance in the mission and the ability to co-orbit with the target.

It is suggested that effective verification conforming to the implicit requirements of the two major roles of Paxsat, and performing within the political parameters discussed earlier in the report can involve a more complex tactical procedure than a single look at the verification target. Follow-on extended surveillance of the activity and performance of the target may be crucial in gaining sufficient data and intelligence to arrive at sound conclusions. Second order targets may be found to be or suspected to be associated with the primary target and second order intercpets may be required. The character and duration of the verification mission must be sufficient to provide a credible verification. The Paxsat must be configured and deployed so that the capabilities for flexiblity, maneuverability and endurances are maximized.

On this basis, the tactical scenario for Paxsat which might be favored would be the following sequence of events:

STEP 1 Paxsat is launched either from earth or an initially dormant parking orbit when there is agreement and authorization by the political authority.

4.4 Paxsat System - Operational Concept (Continued)

STEP 2 Paxsat is launched initially on an intercept and verify mission. Intercept takes pace at 2 km to 5 km range.

STEP 3 Following its initial intercept and reporting mission, Paxsat takes up stand-off surveillance role at distance 90 km to 100 km from target and utilizing optical and other remote sensing methods, continues to report data on target behavior and performance.

STEP 4 Paxsat stands by for additional intercepts should these be required.

STEP 5 Depending upon the level of residual endurance following the mission, Paxsat is retrieved or parked, a political authority decision.

The attributes of the foregoing scenario may be seen to include the following:

(a) Full control of Paxsat and its mission activity resides within the political process.

(b) A Paxsat optimized for close-in inspection followed by short range stand-off surveillance and possible further intercept is feasible within the constraints of the Bilateral Imperative.

(c) Paxsat optimization for event triggering response is totally discriminating and prevents unauthorized and provocative surveillance and data collection.

(d) Maximum mission capability and endurance.

(e) Provides high level of verification without extreme technological challenge.

Subsequent chapters of this report will examine in detail the technical aspects of the configuration, remote sensing and performance of a Paxsat operating in the second phase (second order intercept) of the event triggering mode and optimized for close-in inspection missions.

5.0 MISSION ANALYSIS APPROACHES FOR PAXSAT SPACECRAFT

5.1 Introduction

The objective of the mission analysis activities was to select a Paxsat investigation philosophy and to determine the requirements that this would place on a Paxsat spacecraft design. The key parameters to be specified were the amount of maneuvering (and hence the amount of fuel) required, the mission sequence, the stand-off distance during the investigation and the spacecraft platform hardware requirements.

The following mission scenarios were investigated:

(a) Paxsat is launched from the ground to co-orbit with the target, investigate it and then on-board fuel is used to effect further investigations (launched on demand scenario).

(b) Paxsat is launched into space, parked and then long range sensors are used to investigate targets, while Paxsat maneuvering is kept to a minimum (fly-by scenario).

(c) Paxsat is launched into space, parked and then maneuvered to within several kilometers of the target under investigation (rendezvous scenario).

An important factor in selecting the mission profile is the amount of time available for the investigation. From political considerations, it seems that after a decision to investigate is reached, the investigation should happen as quickly as possible.

Analysis of the rendezvous scenario show that a period of 90 to 120 days must be allowed in the worst-case for Paxsat orbit to drift to the target orbit plane. It appears that although long, this might be acceptable.

Since it was considered that the time required to launch the satellite directly into the target orbit plane is too long or alternatively, that the cost of keeping a launch vehicle ready to launch within a few months is too high in order to make scenario (a) the accepted solution, scenario (c) was preferred to scenario (a). All of the data to make a final decision have not been considered as yet.

5.1 Introduction (Continued)

What was concluded, however, was that a rendezvous scenario, whether involving a direct-to-target launch or prior stationing in space was preferable as a baseline method of investigation to a fly-by investigation. Essentially, a rendezvous mission allows the best opportunity for a complete investigation of the target in many of its operating modes, while requiring relatively unsophisticated technology as compared to a fly-by investigation (which would require high performance control and/or sophisticated optical processing and extremely powerful optics and long range sensors).

It was determined that a rendezvous mission could be performed by a spacecraft whose net mass is made up approximately three parts fuel to one part hardware. The tankage is not unlike that available in recently produced liquid upper stages for use by communications satellites launched by the shuttle.

In the rest of section 5.0, the arguments used to determine the baseline mission as well as the resultant spacecraft requirements are detailed. Section 5.2 presents and discusses the merits/demerits of the launch on demand scenario. Section 5.3 defines the fly-by scenario and discusses operations of Paxsat in this mode. Section 5.4 presents trade-off considerations for a fly-by versus a rendezvous operations. Finally, section 5.5 defines the baseline rendezvous scenario.

5.2 Launch on Demand Scenario

5.2.1 Definition and General Implications

Simply speaking, the launch on demand scenario is a mission events sequence whereby the Paxsat spacecraft is launched from the ground to co-orbit with the target, investigate it and then loiter in an alternative orbit until called upon to perform other investigations using the on-board fuel capability of the spacecraft. In more detail the scenario may, but not necessarily consist of the following events. (The scenario is presented as one of a multitude of possible alternatives to illustrate the main characteristics of the scenario and to provide a basis upon which its merits can be ascertained.)

5.2.1 Definition and General Implications (Continued)

The first event to happen in the scenario is the mission 'go' command issued by the governing body of the treaty. The governing body of the treaty controlling the operation of the Paxsat spacecraft will order the Paxsat spacecraft to investigate a suspected felon only after exhausting all political avenues at its disposal to defuse the event using the contributed national technical means of its member states. Until this go ahead is given, the Paxsat spacecraft may have been in some general state of readiness in its ground storage location and upon receiving the go directive the spacecraft would be flown to a suitable selected launch site to be launched on a vehicle that also needs be at some general state of readiness. From this point on, the launch pad becomes a beehive of activity with detailed mission sequences plans being created, Tracking, Telemetry and Control links with numerous ground control stations around the world being established, last minute systems checks being undertaken on the spacecraft, and mission sequence rehearsals being enacted, all culminating with the launch of the spacecraft into its orbit. Control of the spacecraft is then effected to maneuver the spacecraft into close proximity of the target using some predetermined mission strategy to begin the investigation of the alleged offender. After the investigation is completed, the Paxsat spacecraft moves from the vicinity of the target spacecraft to another orbit, sitting there to await its next calling.

The major advantages gained by a launch on demand scenario over the prelaunch scenarios is the reserving of valuable fuel on the spacecraft to enable the spacecraft to perform additional missions. By launching directly into the desired orbit instead of an initial parking orbit, the spacecraft need not burn as much fuel to rendezvous with its target. Thus, the spacecraft may be available to perform more investigations as a reult of this fuel savings and thereby the cost effectiveness of the spacecraft life cycle cost is increased. However, considering the operations sequences above, the launch on demand scenario is a complex logistics and operations problem requiring time to plan for and to execute effectively.

5.2.1 Definition and General Implications (Continued)

Commercial launch campaigns are similar to that presented above. From a programmatics point of view, companies generally like to allow between 3 to 4 months to plan for and conduct a launch campaign. However, for a higher state of readiness, time on the order of two months may be required to conduct the campaign assuming that a launch vehicle is also in a high state of readiness. To have a launch vehicle in such a state of readiness may require a fully integrated launch vehicle to stand upon its own dedicated launch pad as the timing of a Paxsat mission may not entirely phase in with the launch activities of individual nations. Sharing existing launch facilities may introduce potential launch campaign conflicts with regularly scheduled military and civilian launches. However, as the Paxsat compatibility with the number of launch vehicles increases from different contributing nations, the probability of interferring with regularly scheduled launch campaigns decreases. Such a state of readiness however, would still entail the requirement for a fully built and integrated launch vehicle since launch vehicles of themselves require about 30 months to be manufactured and assembled.

Political factors are also at play in a launch on demand scenario. The more evident factor is that a decision to launch and subsequently, the excitement aroused by an investigative launch can be seen to exasperate the crisis at hand. During such international crises, it is often wise to have actions done on the quiet, since they do not contribute to the fever and excitement of the general populace. Launches are exciting events in themselves and cannot in any event be done without arousing the curiosity of the media.

A somewhat more latent political factor arises from the fact that launch vehicles are not always successful in placing their payloads into orbit. Because launch vehicles are not 100% reliable, a further operational constraint is introduced to the launch on demand scenario as a result of political psychology. If the suspected felon were to be an American satellite, would the remaining states to the treaty entrust the Paxsat spacecraft to be launched by an American launch vehicle from an American launch pad when a deliberate act could foil the investigative mission under the guise of a

5.2.1 Definition and General Implications (Continued)

statistical launch vehicle failure? Or conversely, if the situation was Soviet? This is not to suggest that these nations would commit such actions, but only to illustrate that the question of trust is a real concern within the international community. This dilemma is resolved if Paxsat is launched by a nation different to that under question and ideally one who can be neutral to the constestants involved. This requirement on operations combined with the need for a fully integrated launch vehicle soon raises the system cost for the launch on demand scenario by requiring several integrated launch vehicles, and may well exceed the spacecraft life cycle cost savings which was asserted in the initial scenario.

Additionally, technical factors involved in the launch on demand scenario also questions its relative effectiveness when compared to its alternatives. These factors regarding launch windows are present in the next subsection.

The results of the analysis on the launch on demand scenario is a function of the assumed operations scenario. That scenario presented here points to an alternative mission scenario for the Paxsat spacecraft. Further study on this subject would be required to prove conclusively that the launch on demand scenario is not an ideal mission concept for the Paxsat spacecraft.

5.2.2 The Rendezvous Problem

The possibility of launching on demand requires the ability to launch from earth and closely approach an orbiting satellite. The technical problems involved are identical to those of the rendezvous problem, excluding the final approach and docking. The procedure for rendezvous was therefore explored and is described in detail in Appendix B. The major elements of this procedure are sketched here.

When a target is identified and its orbital parameters are determined, a procedure for rendezvous may be established. The objective of this procedure is to bring the chasing satellite to the same place at the same time and at the same speed as the target. The method consists of precise timing of all events from launch to rendezvous.

5.2.2 The Rendezvous Problem (Continued)

The near approach of one satellite to another will require careful monitoring and the best possible visibility from earth stations. The point where rendezvous will occur will be selected with consideration of the available tracking sites and control centers. The choice of an interception point will be most limited for the targets at the lowest altitudes, as these will have the highest speeds and the shortest visibility times from any ground station. The first step in the procedure is the selection of this rendezvous point.

A transfer orbit is designed to bring the chasing satellite to this position and a phasing orbit is used to assure that it arrives at the same time as the target. The design of these orbits is a process of trial and error to find the best combination of time and fuel costs which will meet the rendezvous objectives. The altitude of the phasing orbit is open to choice and a variety of values will be tried in finding a suitable combination. The process begins with an arbitrary selection of an altitude for the phasing orbit. This is used together with the target orbit to predict the transfer orbit parameters.

The time of launch must be coordinated with the position of the target satellite in its orbit at the launch time. This is obtained by working backwards from the rendezvous point and equating the times for both vehicles to reach this point. The time for the homing satellite to reach the rendezvous point is the sum of the time in the transfer orbit, the time in the phasing orbit and the time of ascent to the waiting orbit. The point of reference for these times is the perigee of the target orbit.

Launch from earth is planned to occur when the launch site crosses the plane of the target orbit. This scheme is necessary to avoid the prohibitively high costs in fuel to make changes in the plane of any satellite orbit. With the inclination of the target orbit known and the latitude of the point of interception selected, the distance along the equator from the longitude of the launch site to the equator crossing of the target may be

5.2.2 The Rendezvous Problem (Continued)

calculated. This leads to an expression for the time of which the launch site crosses the orbital plane referenced in the vernal equinox direction.

The launch time, measured in the orbital plane, and the time the launch site crosses this plane, measured along the equator, must then be related to a common base so they may be equated.

Matching these times shows when the target must cross the equator for a successful rendezvous. A different equator crossing point will result for each different number of revolutions in the phasing orbit. The one which matches one of the actual equator crossings of the target will allow a launch which will lead to rendezvous.

A new set of solutions will arise from a new altitude of the phasing orbit. The procedure is repeated until an appropriate launch time is found corresponding to a sufficiently short time to intercept. In a simplified but representative sample case, the time required to rendezvous with a satellite orbiting at 1000 km above the earth was 15 hours using nine revolutions in the phasing orbit (see Appendix B).

A very specific launch time results from these calculations. Deviations from this time of launch impose penalties in fuel which rapidly become prohibitive. For example, the launch window for the Solar Max repair mission was reported to be 3 to 13 minutes per day over a particular one week period. A missed launch would generally require the selection of a new set of orbits for a new launch time.

Launch from earth to intercept an orbiting target is feasible in terms of orbit design. Ground station controlled maneuvers may be used to bring a satellite into acquisition range for rendezvous. By launching into the plane of motion of the target, the fuel required is limited to that needed for altitude raising and corrections. Penalties would be incurred should the stringent timing requirements be relaxed.

5.3 Fly-by Scenario

5.3.1 Geometry

Figure 5-1 illustrates two orbit planes of different inclinations (i1, i2) and right ascension (1, 2) relative to the equatorial plane. Any two such planes may be characterized by a relative inclination and their line of intersection. In the analysis which follows, the orientation of the planes relative to earth's equator will be ignored and only the relevant parameter, the relative inclination of the orbit planes, considered.

The relative inclination of the planes can be obtained from a specification of i_1, Ω_1, $i2, \Omega 2$ by the equation:

$$\cos(\Delta\beta) = \cos(\Omega_1 - \Omega_2)\sin i_1 \sin i_2 + \cos i_1 \cos i_2$$

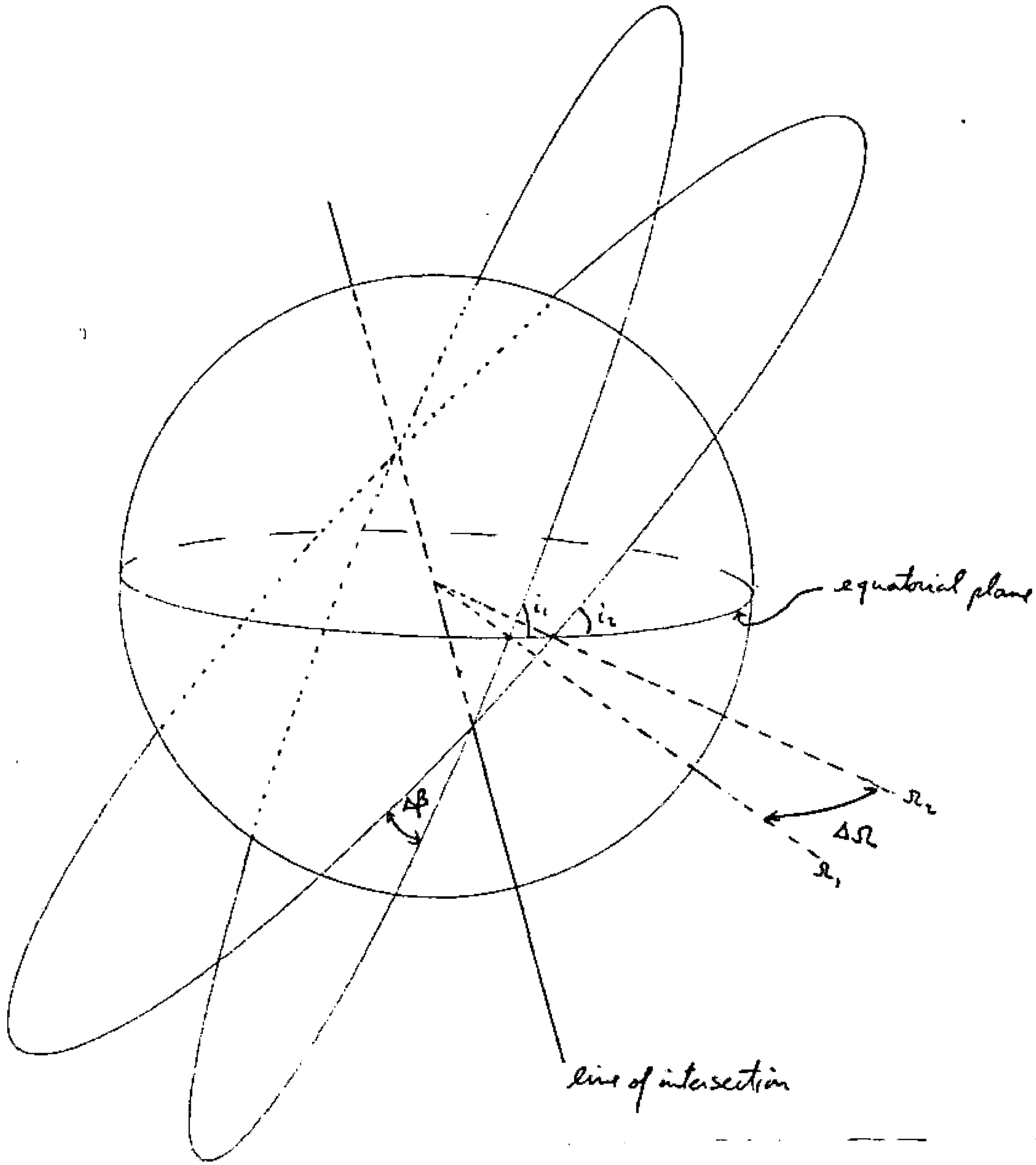
This can be derived by taking the inner product of the normals of the two planes expressed in the equatorial reference frame.

As shown in Figure 5-2, the relative satellite phasing angle (ϕ) is defined as the target true anomaly referenced to the line of intersection at the time that Paxsat crosses the line of intersection.

For the purpose of determining parameters such as range, range rate, azimuth and elevation angles and other parameters of interest, it will be assumed that both Paxsat and the target are in circular orbits, and are operating at the same altitudes. These assumptions simplify considerably the analysis. The assumption of circular orbits is justifiable on the grounds that many orbits of interest are circular. Further, the insight gained in an analysis of circular orbits can be applied to elliptical orbits with allowances for variations in satellite phasing and altitude.

5.3.1.1 Sensor Slewing

In order to maintain the target satellite within the field of view (FOV) of the Paxsat sensors, it will be

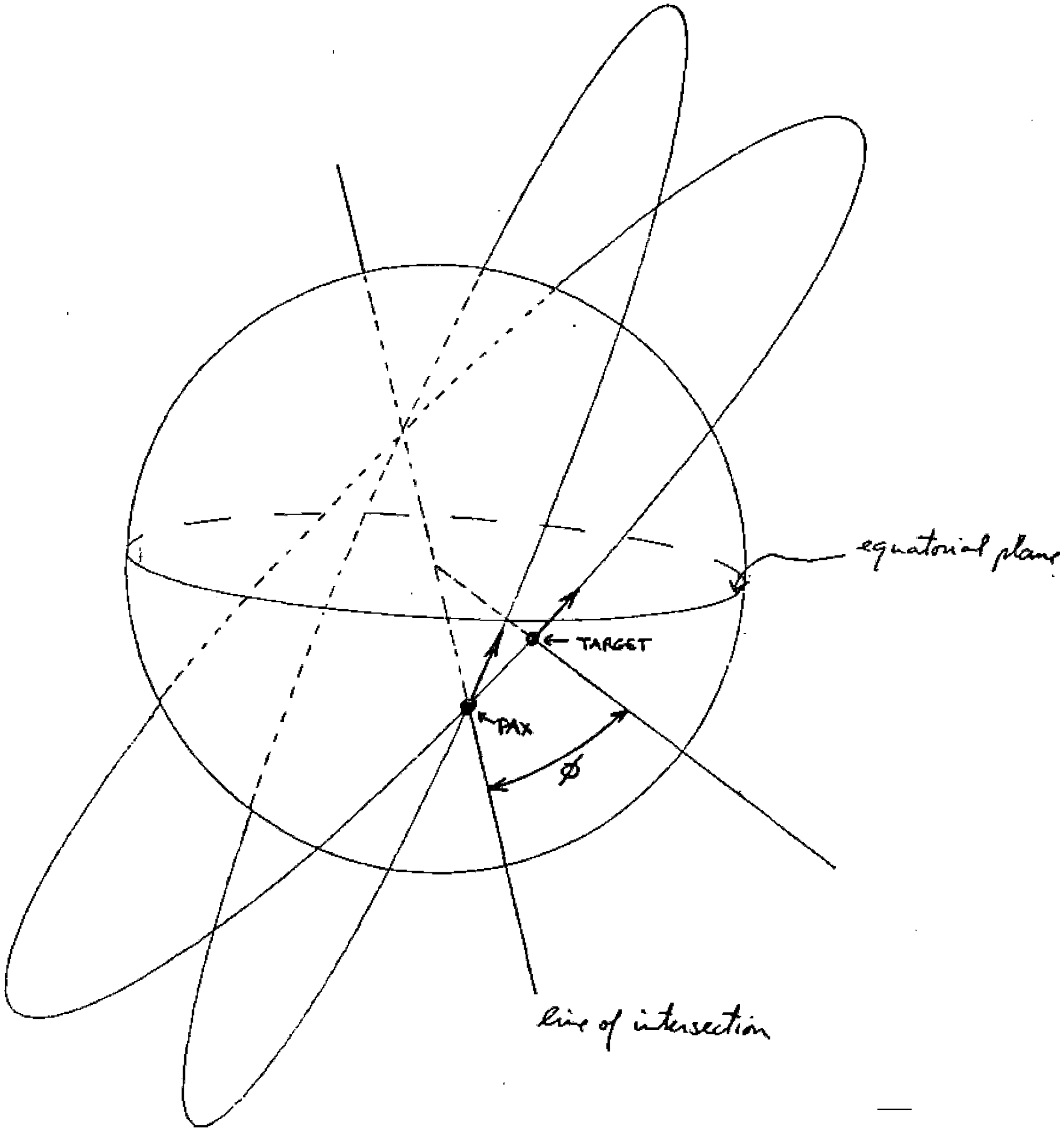


i_1, i_2 INCLINATION RELATIVE TO EQUATORIAL PLANE

Ω_1, Ω_2 LONGITUDE OF THE ORBIT OF THE ASCENDING MODE

$\Delta\beta$ RELATIVE INCLINATION

FIGURE 5-1 DEFINITION OF RELATIVE INCLINATION



Ø RELATIVE PHASE
(OR PHASING ANGLE)

FIGURE 5-2 DEFINITION OF RELATIVE PHASE

5.3.1.1 Sensor Slewing (Continued)

necessary to slew the FOV's relative to the orbital reference frame. This may be accomplished in any one of the following ways:

(a) The sensors are fixed to the satellite which maintains an unchanging attitude relative to either the earth or inertial space and the sensor head scans mechanically or electronically to follow the target.

(b) The sensors are mounted on a platform which slews to follow the target. The platform is coupled to the satellite body through a gimbal system. The satellite body remains pointed to some reference frame.

(c) The sensors are fixed to the satellite body which slews to follow the target.

A combination of these methods can also be used of course.

The following are the equations of motion of the target in a Paxsat centered reference frame which rotates so that the X-axis remains pointed away from the center of the earth and Y is the direction of flight. This reference frame represents the attitude reference for an earth centering control system with yaw control such as might be typical for a LEO spacecraft.

$$\underline{D} = \begin{bmatrix} R_T(\cos\theta_p \cos\theta_T + \sin\theta_p \sin\theta_T \cos\Delta\beta) - R_p \\ R_T(\sin\theta_p \cos\theta_T + \cos\theta_p \sin\theta_T \cos\Delta\beta) \\ R_T \sin\theta_T \sin\Delta\beta \end{bmatrix}$$

$$\theta_p = \omega_p t$$
$$\theta_T = \omega_t t + \phi$$

5.3.1.1 Sensor Slewing (Continued)

From these equations, one may determine the requirements for slew rates, slew angles, angular accelerations, target range and target range rate which make up the constraints on the sensor and spacecraft design for a fly-by mission. Figure 5-3 illustrates the definition of main and secondary bearings.

Figure 5-4(a-f) illustrates these quantities in the case where Paxsat and target heights are 1000 km, relative inclination is 90° and phase angle is 1.0°. This gives a distance of closest approach of just under 100 km.

From the point of view of the sensor design, the range and the observation time are of most interest. The slewing rates effect the spacecraft momentum management philosophy and so are important to the spacecraft design. The angular accelerations, along with the intertia properties of the items slewed determine the torque actuation requirements.

For exampled, a maximum yaw slew rate of 6.5° per second (see Figure 5-4(c)) imposed on a satellite whose yaw inertia is 100 kgm^2 gives an angular momentum L of

$$L = \left(\frac{6.5 \cdot \pi}{180}\right) 100$$

$$L \doteq 11.3 \text{ Nms}$$

which corresponds to the reaction wheel capacity required to maintain the satellite body stable.

The corresponding maximum angular acceleration (Figure 5-4(d)) is approximately $0.48^{\circ}/S^2$ and translates to a required torque T of

$$T = (100 \text{ Kg m}^2)\left(\frac{0.48 \cdot \pi}{180}\right)$$

$$T \doteq 0.84 \text{ Nm}$$

5.3.1.1 Sensor Slewing (Continued)

which, in conjunction with the angular momentum requirement, effectively specifies the reaction wheel capacity required on the spacecraft to perform the yaw maneuver.

Figure 5-4(d) therefore also shows the torque profile which would be commanded by the main bearing slew controller during a pass of the target. It should be noted that the torque profile is highly non-linear, and so would require sophisticated controls to implement.

From the plots in Figure 5-4, it is clear that most parameters reach a critical value at the time when the target is closest to Paxsat. It is of interest, therefore, to see the relationship between these critical parameters and the orbit configuration parameters (relative inclination, phasing angle).

5.3.1.2 Values of the Parameters of Interest at the Time of Closest Approach

Taking the derivative of target range with time and setting it equal to zero gives the time of closest approach t_{CA} as

$$t_{CA} = \frac{1}{\omega}\left(-\frac{\phi}{2} + n\pi\right) \qquad n = 0,1,2,3,\ldots$$

$$\omega \equiv \text{orbit rotation rate}$$

Evaluating the expression for target range at this time gives the range at closest approach to be:

$$R_{CA} = R\sqrt{2(1-\cos^2(\phi/2) + \sin^2(\phi/2)\cos\Delta\beta}$$

The angular rate on the secondary (elevation) bearing is zero at closest approach. However, the main bearing angular rate reaches its maximum which is evaluated to be:

$$\left.\frac{d(\theta_{At})}{dt}\right|_{ca} = \frac{\omega\sin\Delta\beta\sin\phi}{(1-\cos\phi)(1+\cos\Delta\beta)}$$

5.3.1.2 Values of the Parameters of Interest at the Time of Closest Approach (Continued)

The table below shows the values of time of closest approach, range and azimuth rate for the case of 1000 km altitude orbits inclined 90° relative to each other with Paxsat and the target phased 1° apart.

t_{CA}	R_{CA}	$\left.\frac{d\theta_{AZ}}{dt}\right\|_{CA}$
-8.8 s	91 Km	6.53°

These values may be compared to those illustrated in Figure 5-4.

5.3.1.3 Range from Paxsat to the Target

The magnitude of the displacement, i.e. the distance between the satellites is then given by D, where

$$D = \sqrt{2}R \cdot \sqrt{1 - \cos(\omega t+\phi)\cos(\omega t) - \sin(\omega t+\phi)\sin(\omega t)\cos\Delta\beta}$$

With this information, one can determine the extent of visibility of the target orbit from the Paxsat orbit. However, for low earth orbits, the effect of the earth in hiding parts of the target orbit will be pronounced, therefore, an algorithm for determining whether the target is eclipsed by the earth must also be introduced.

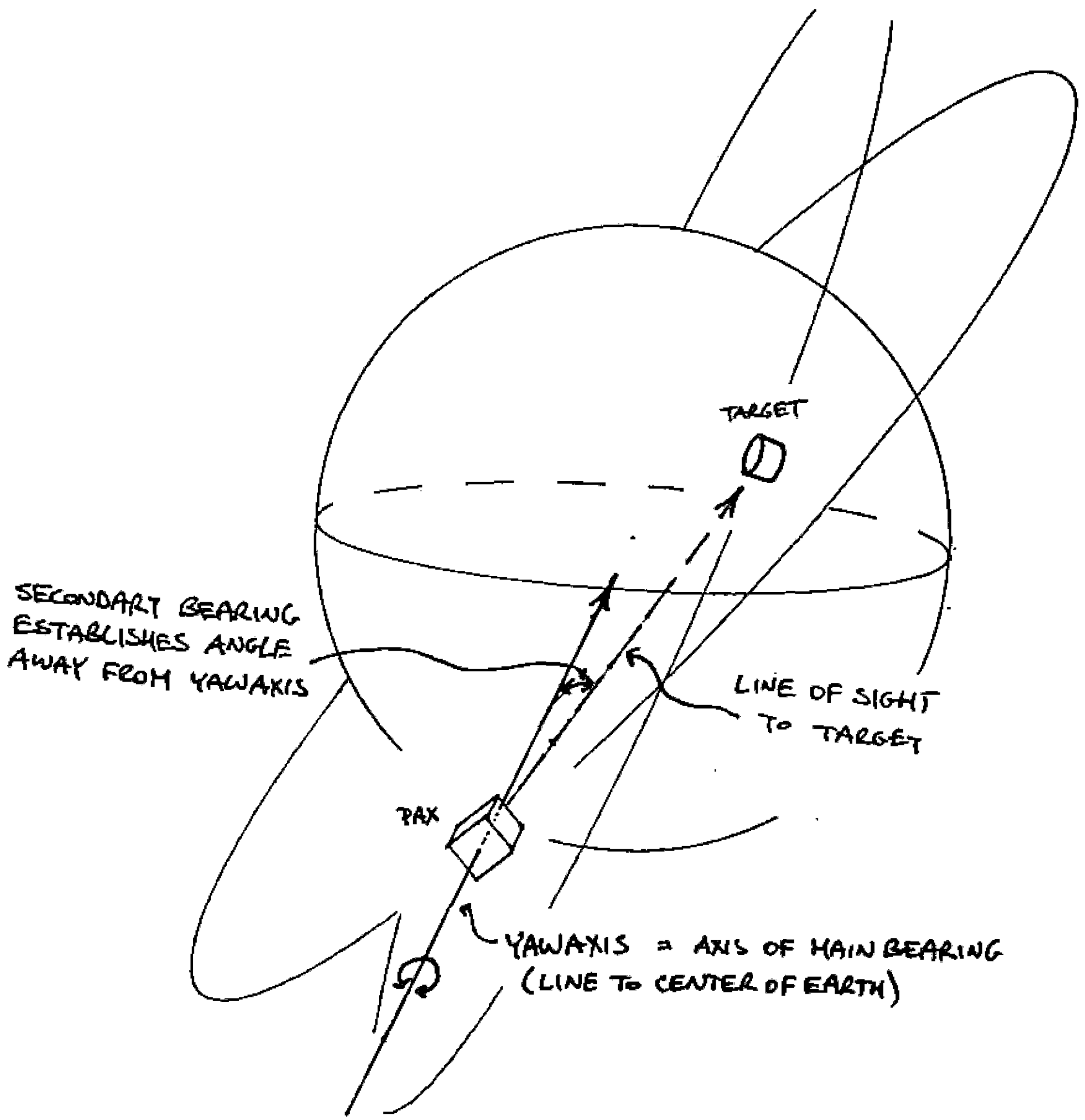


FIGURE 5-3 DEFINITION OF MAIN AND SECONDARY BEARINGS

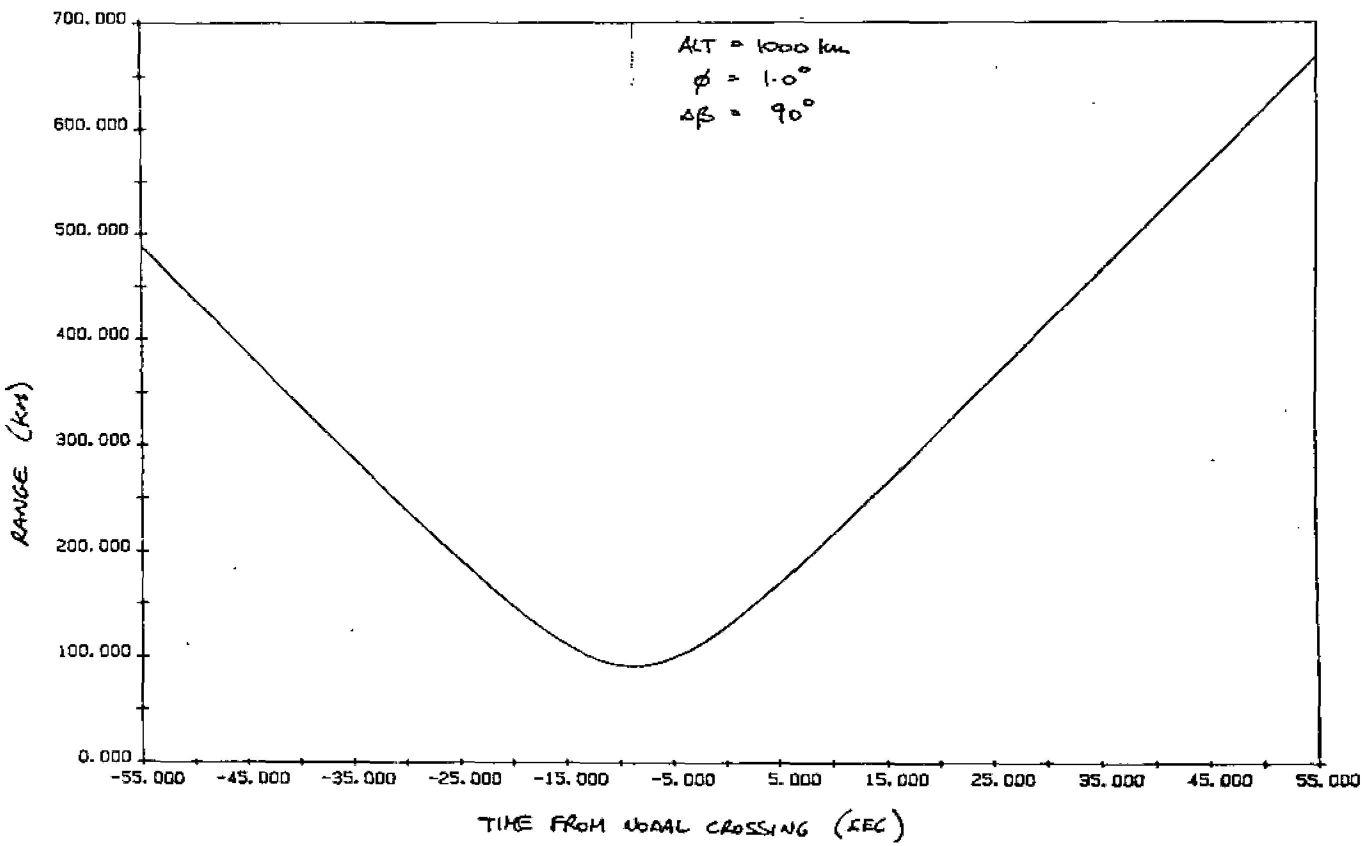


FIGURE 5-4(a) RANGE TO TARGET VERSUS TIME FROM NODAL CROSSING

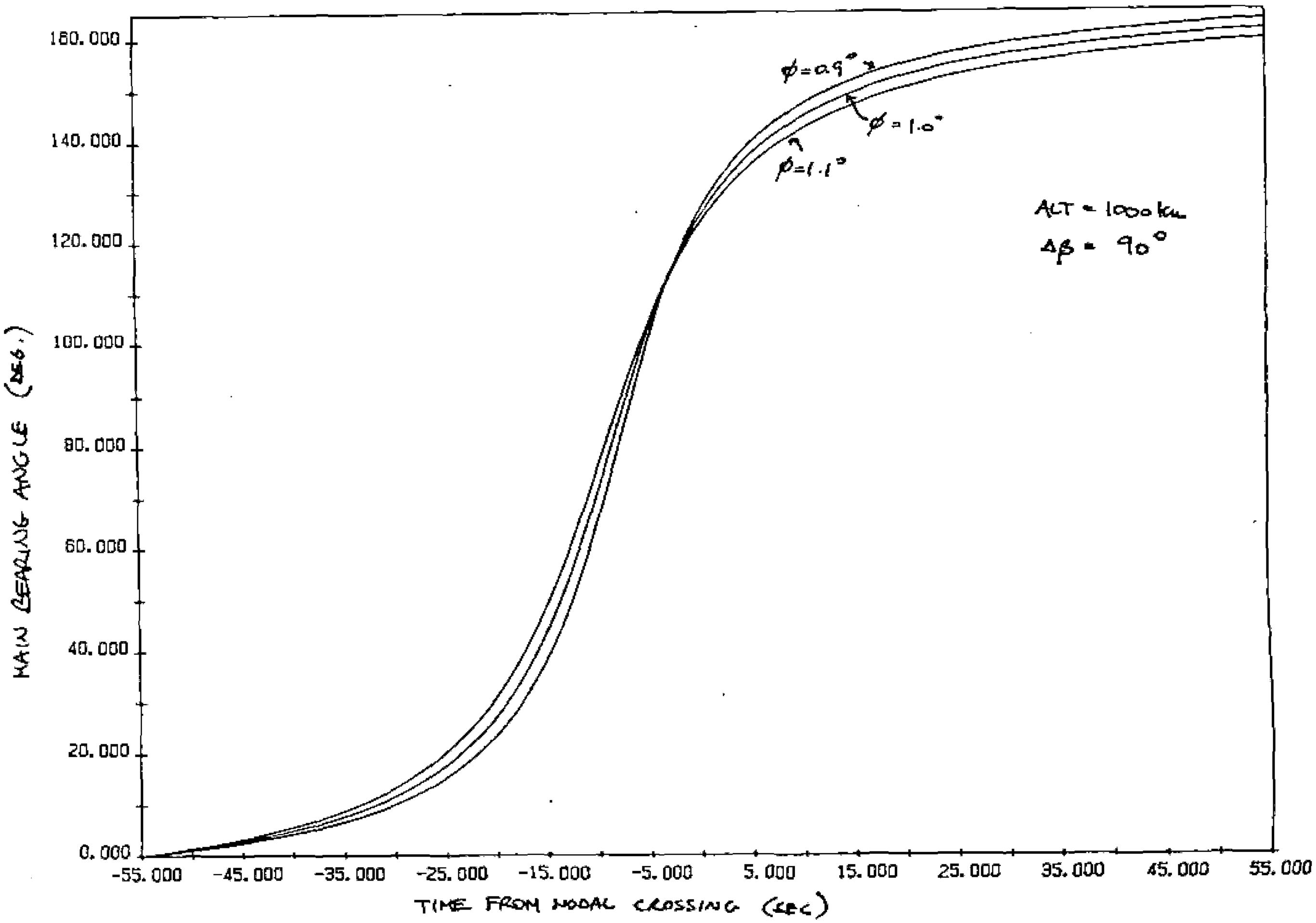


FIGURE 5-4(b): MAIN BEARING ANGLE VS TIME FROM NODAL CROSSING AND PHASING ANGLE

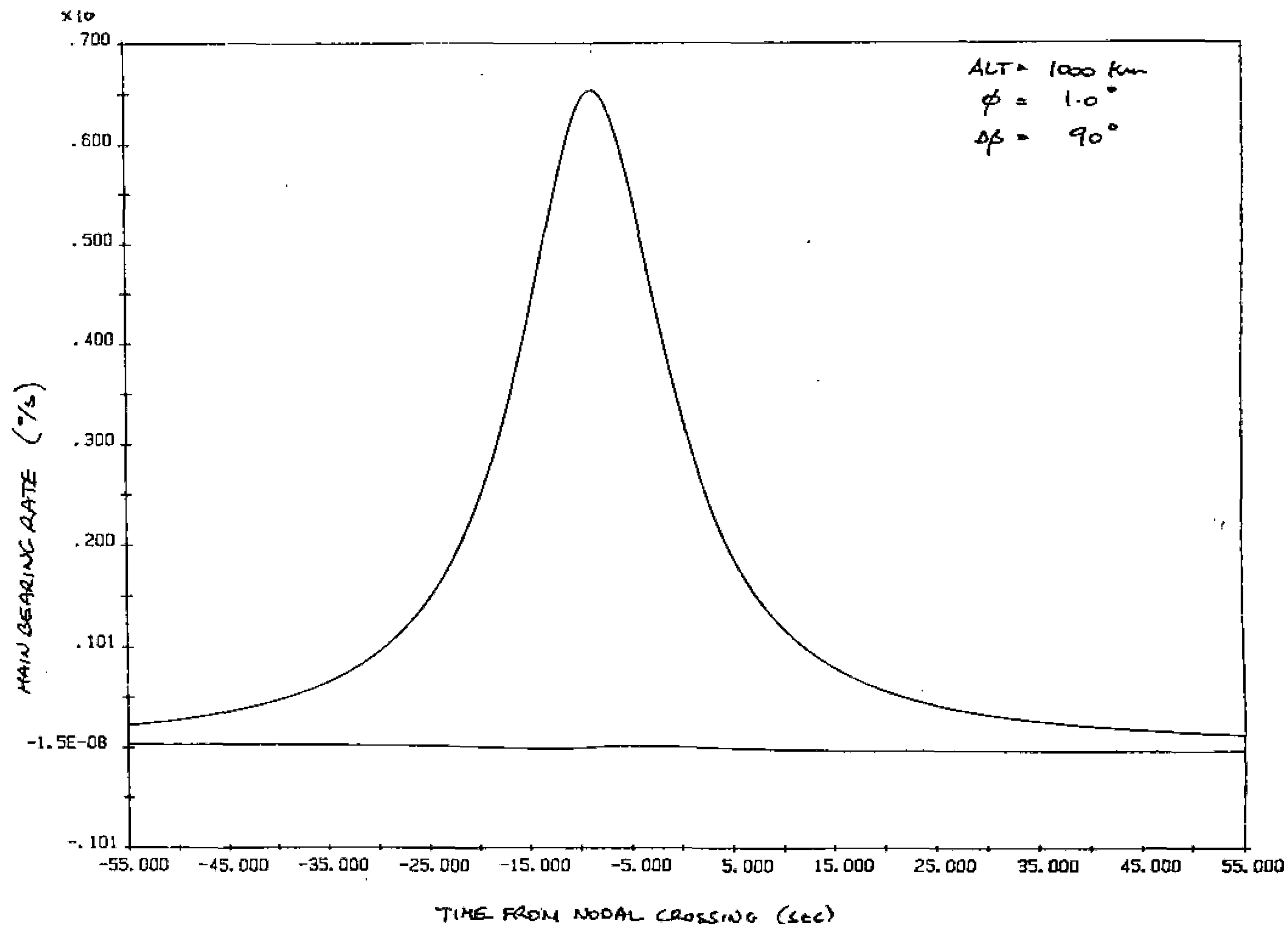


FIGURE 5-4(c) : ANGULAR RATE ON MAIN BEARING VS TIME FROM NODAL CROSSING

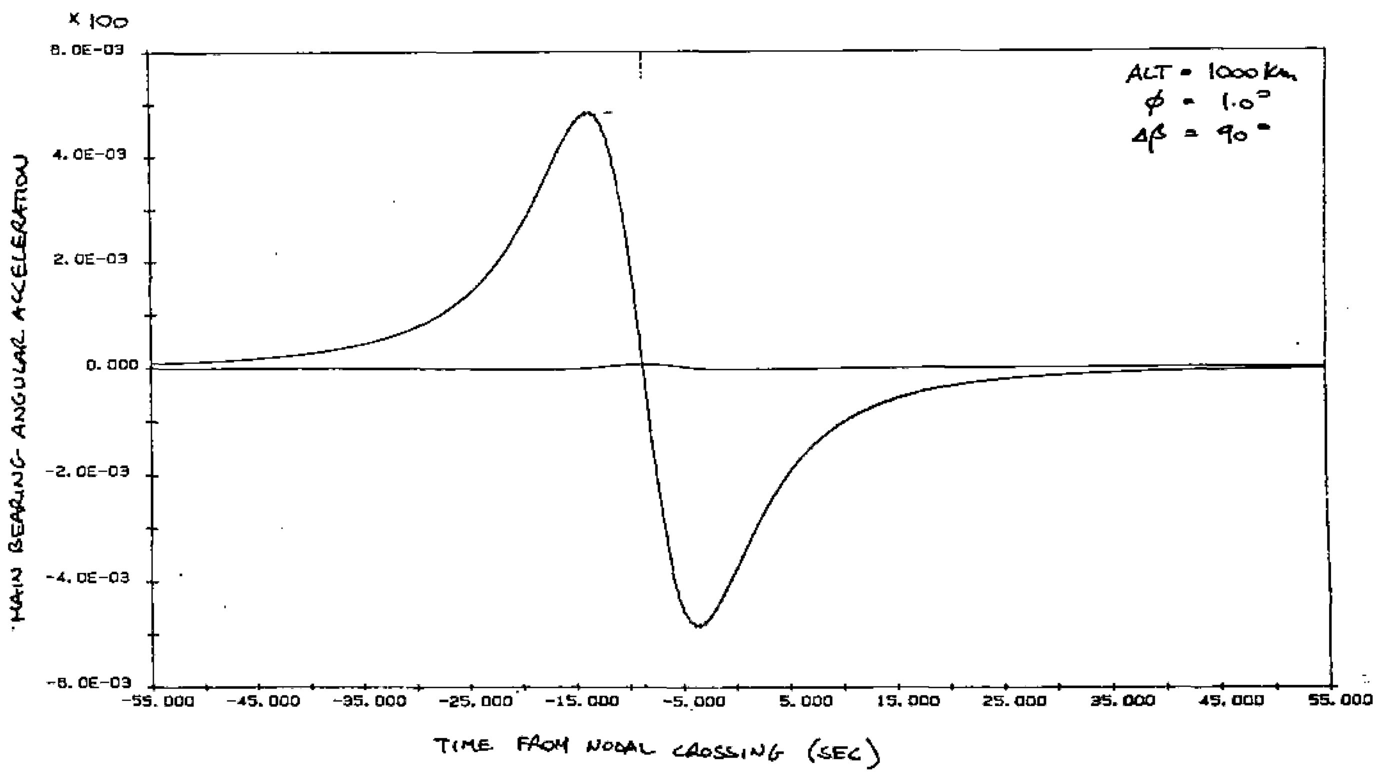


FIGURE 5-4(d): MAIN BEARING ANGULAR ACCELERATION VERSUS TIME FROM NODAL CROSSING

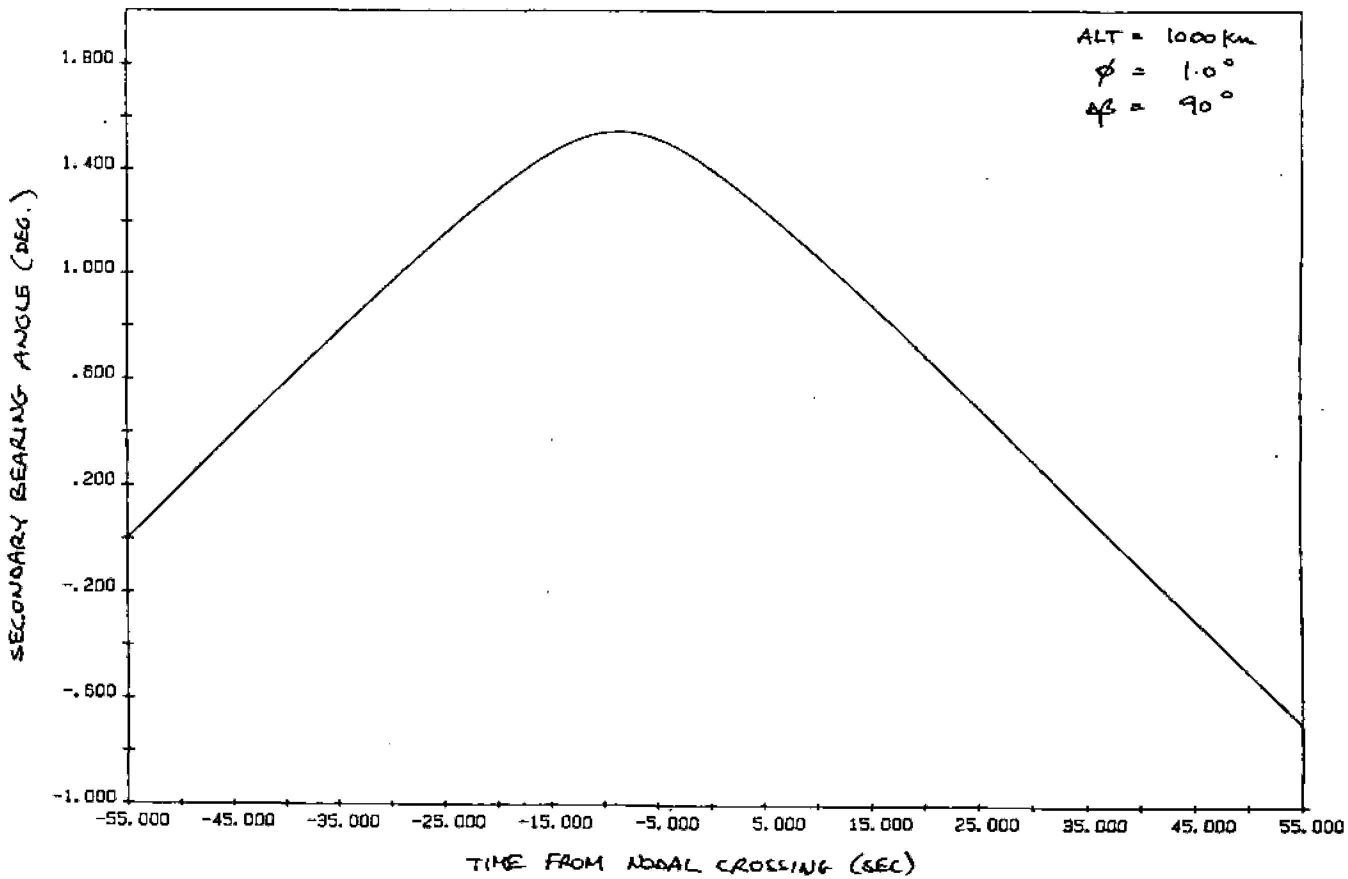


FIGURE 5-4(e): SECONDARY BEARING ANGLE VS TIME FROM NODAL CROSSING

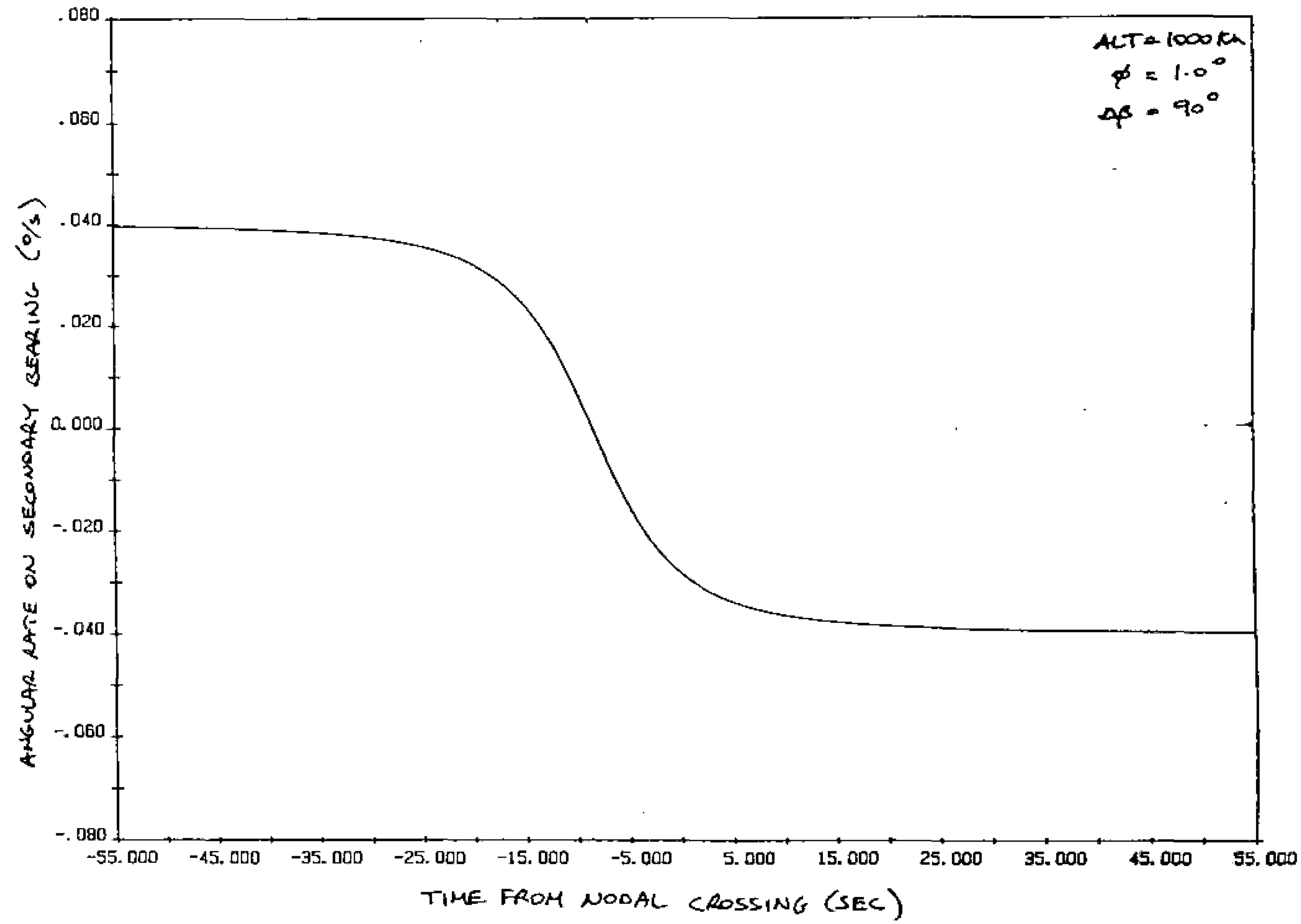


FIGURE 5-4(f): ANGULAR RATE ON SECONDARY BEARING VS TIME FROM NODAL CROSSING

5.3.1.4 Conditions for Earth Interference in the Line of Sight

From Figure 5-5, it can be seen that the distance L may be obtained from P, the vector from earth's center to Paxsat, and D a unit vector from Paxsat to the target by equation:

$$L = \left| \underset{\sim}{P} \times \hat{D} \right| = \left| P \sin \measuredangle DPD \right|$$

replacing $\underset{\sim}{P}$ and $\hat{D}$ vectors from above gives

$$L = \frac{R^2}{|D|} \cdot \left| \sin\theta_T \cos(\theta_P - \Delta\beta) + \cos\theta_P \sin\theta_T \sin\Delta\beta - \cos\theta_T \sin\theta_P \right|$$

where $\theta_P = \omega t$
$\theta_t = \omega t + \phi$

if $L < R_e + \epsilon$

then, the earth interferes

Re = earth radius
ϵ = thickness of atmosphere through which Paxsat sensors cannot operate

Figure 5-6 shows the maximum possible satellite phasing which will allow at least 1 minute of viewing per orbit of the target by Paxsat for a selection of orbit altitude.

One may interpret the graph to say, for example, that if the target is randomly phased relative to Paxsat, if both satellites are at an altitude of 200 km, and if the relative inclination of the orbit planes is 45^o, then the probability of viewing the target is only 17% even with an infinite range camera (31° out of 180°) due to earth interference in the line of sight.

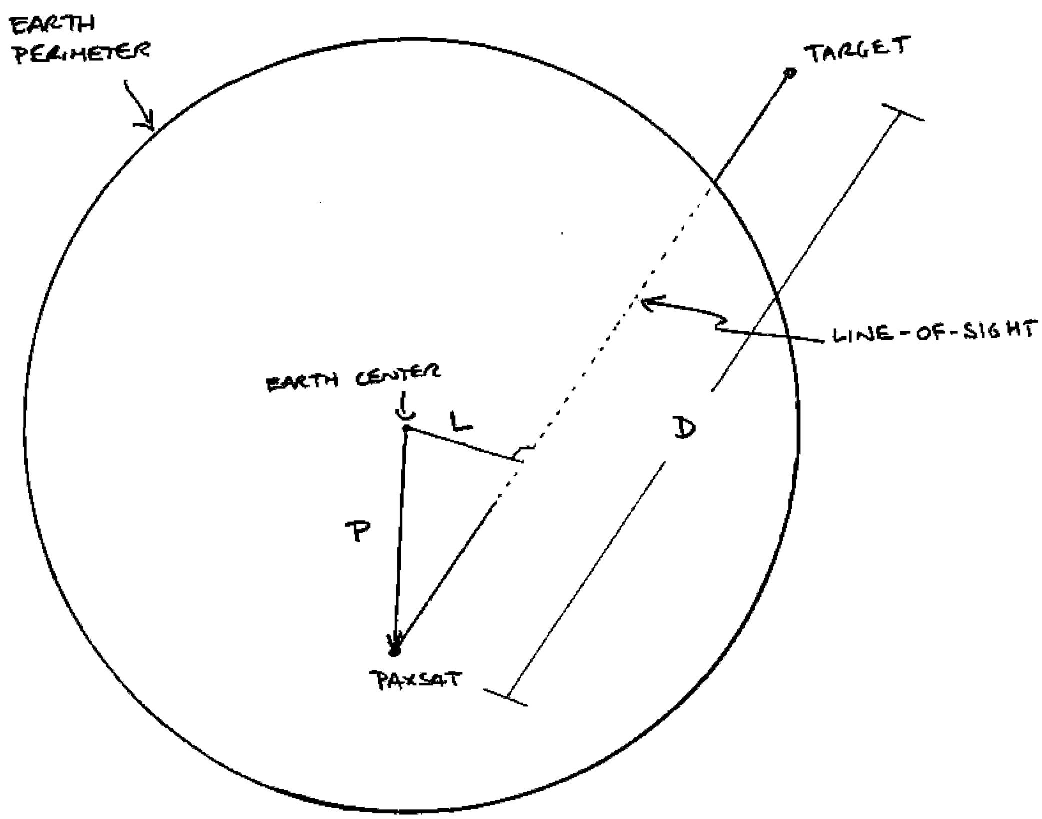


FIGURE 5-5: GEOMETRY FOR CALCULATION OF EARTH INTERFERENCE

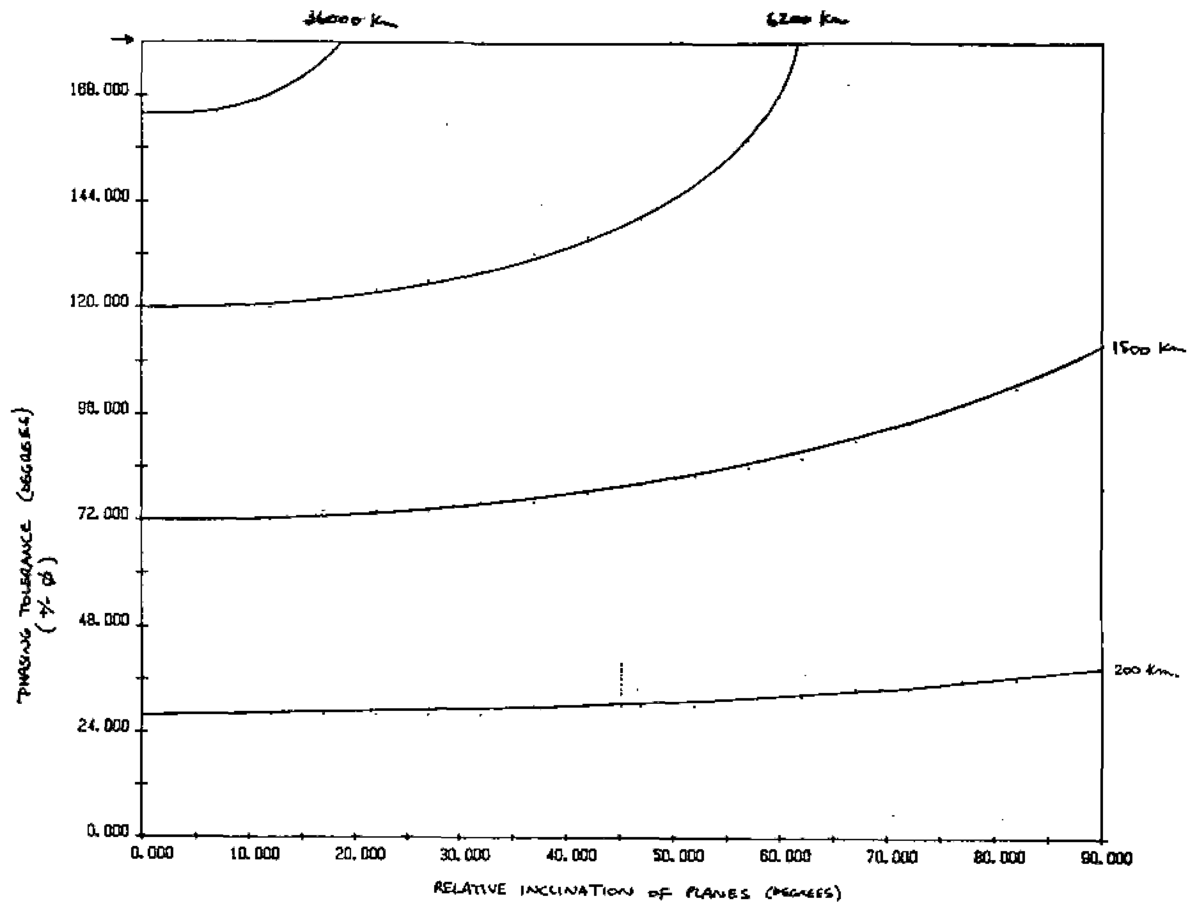


FIGURE: 5-6 MAXIMUM POSSIBLE PHASING TOLERANCE VS RELATIVE INCLINATION

5.3.2 Target Visibility versus Relative Inclination and Phasing Angle

An expression can be derived which gives the allowable set of inclinations and phases for which the satellite is visible a set period of time. As Appendix B shows,

if, $\theta + = \Delta\beta/2 + \omega T/2$

$\theta - = \Delta\beta/2 - \omega T/2$

$$K = \frac{\text{camera range}}{\text{orbit radius}}$$

$\Delta\beta$ = relative inclination
ω = orbital rate of rotation
T = time that the target is in view for each encounter (there are 2 encounters per orbit)

Then

$$\cos(\Delta\beta) = \frac{K^2/2 + \cos(\theta+)\cos(\theta-) - 1}{\sin(\theta+)\sin(\theta-)}$$

Using this expression, one may plot a graph, Figure 5-7, of how long the target is visible each orbit (expressed as a percentage of the orbit period) as a function of the relative inclination of the Paxsat and target orbit planes and the phasing of the satellites in their respective orbits.

Plotting these graphs for the case of Low Earth Orbit (at altitudes of 200 km and 1,500 km) and camera ranges of 100 km and 1,000 km (see Figure 5-7), one can see that continuous coverage may be obtained only for orbits with a small relative inclination. This in itself does not preclude the use of the fly-by as a means of target interrogation. It does show, however, that a trade-off exists between camera range and fuel (which can be used to get near the target). This trade-off favours long camera ranges only if they can be obtained relatively cheaply. This is discussed in the following sections.

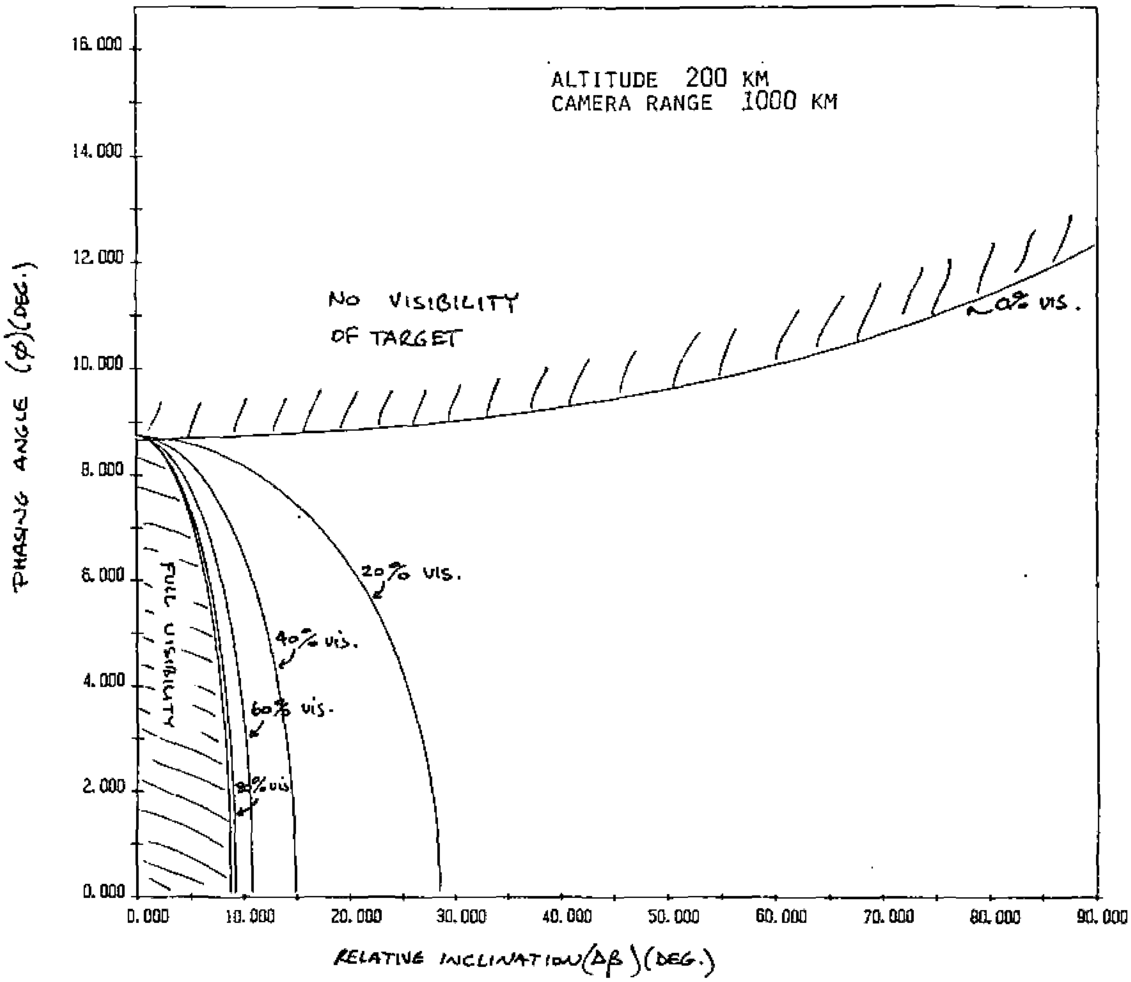


FIGURE 5-7(a) TARGET VISIBILITY VS TARGET RELATIVE INCLINATION AND PHASING ANGLE; ALTITUDE 200 KM, CAMERA RANGE 1,000 KM

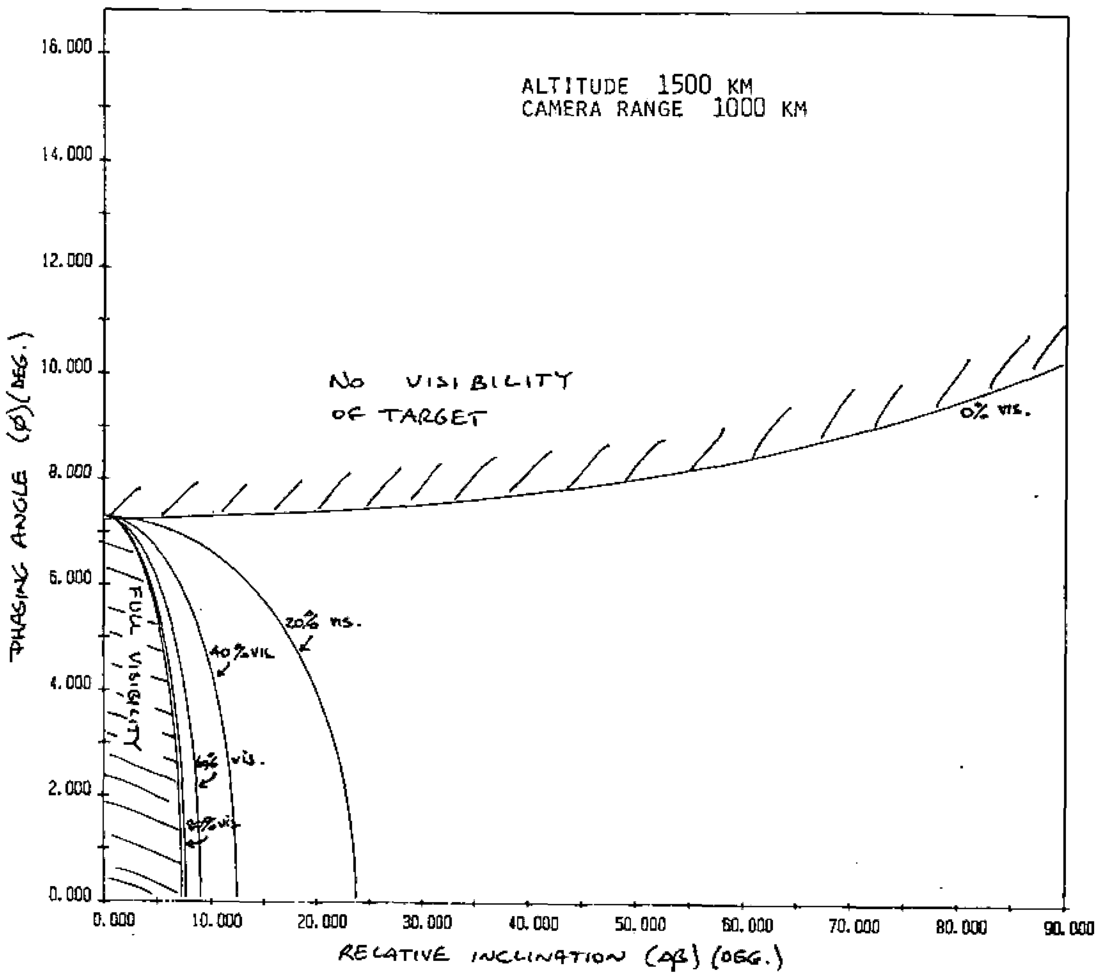


FIGURE 5-7(b) TARGET VISIBILITY VS TARGET RELATIVE INCLINATION AND PHASING ANGLE : ALTITUDE 1,500 KM, CAMERA RANGE 1,000 KM

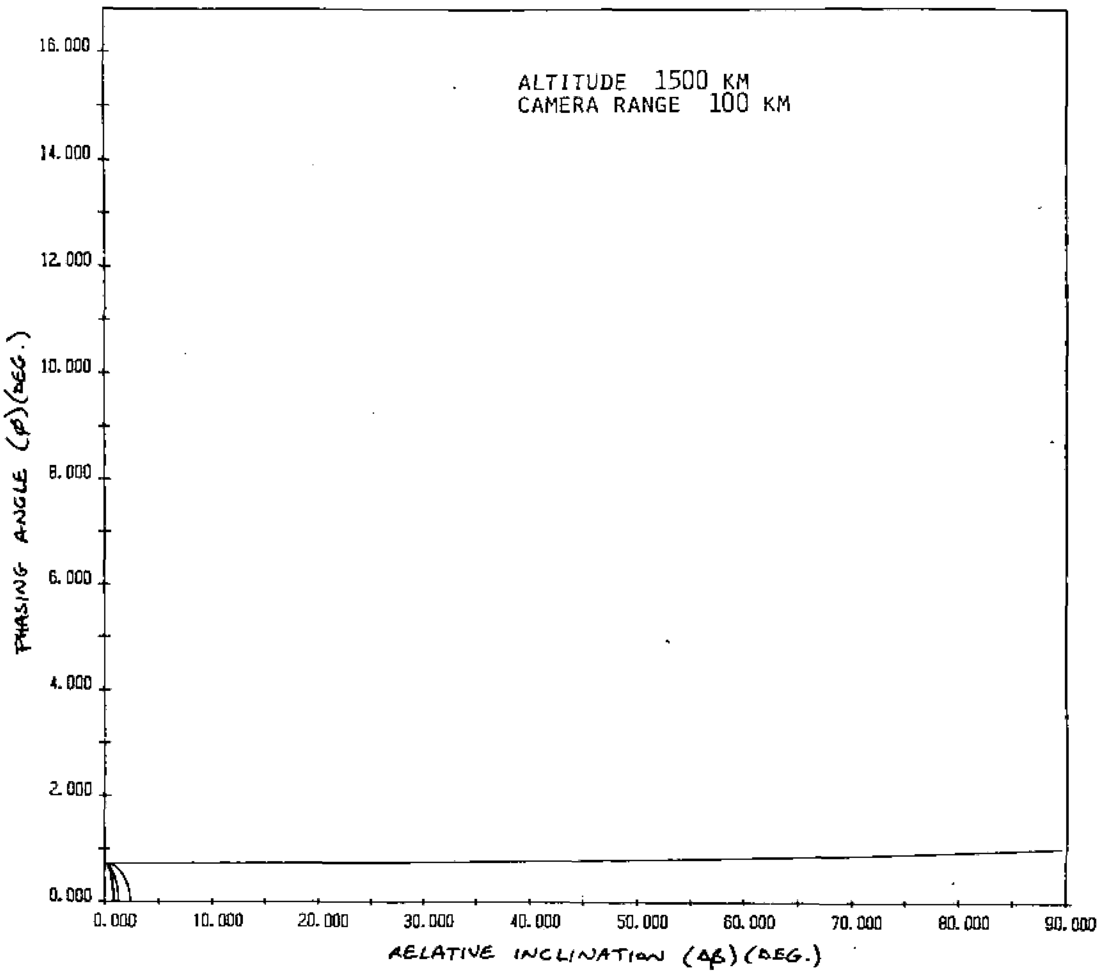


FIGURE 5-7(c) TARGET VISIBILITY VS TARGET RELATIVE INCLINATION AND PHASING ANGLE : ALTITUDE 1,500 KM, CAMERA RANGE 100 KM

5.4 Fly-By Versus Rendezvous

Since it is possible to observe another spacecraft event though it is not co-orbital with the investigating satellites, the question to be answered is, To what extent should Paxsat be maneuvered towards the target?

The options are as follows:

(a) Perform a fly-by making no effort to align the orbital planes, concentrating solely on satellite phasing and orbit altitude.

(b) Perform a rendezvous by expending fuel to rotate the Paxsat orbital plane to make it coincide with that of the target and then maneuver to get near it.

If it is considered that continuous coverage of the target is required during the investigation period, then a trade-off between coverage at a distance versus co-orbital coverage favors the co-orbital scenario. The fuel required to co-orbit is cheaper to carry than a camera which could maintain continuous coverage even in the presence of a relative inclination of the satellite planes.

If, however, it is considered that periodic coverage giving an access time of approximately one minute per encounter is sufficient to investigate the target, then a fly-by investigation would be favored because the sophistication and expense of the on-board hardware would be offset by the speed with which the investigation could yield results, and by the endurance of the investigating satellite, neither of which could be matched by a rendezvous investigation.

For the purpose of this study, it was assumed that continuous coverage was a requirement due to the number of observations and measurements deemed necessary to identify the function of the target. Subsequent analysis (see section 5.5.3) show that the fuel requirement does not render a rendezvous unfeasible provided that sufficient time is allowed for the transfer. However, a large number of investigations can probably not be mounted with the same investigating satellite unless the targets to be investigated are distributed favorably.

5.4 Fly-By Versus Rendezvous (Continued)

Drawbacks of the fly-by scenario are:

(a) High angular velocities of the target with respect to the Paxsat create a requirement for a high performance control system and in any case, limit the range of relative inclinations across which an observation could be made.

(b) Due to earth interference and also due to the fact that only a limited relative inclination of the orbit planes will permit observations to be made, it is clear that a fly-by configured Paxsat would have to carry a propulsion system and a reasonably large fuel supply in any case.

(c) Because of the high angular velocities and long camera ranges, the quality of the information gathered will, in principle, not be better than that obtainable through observation from the ground, where a larger and more sophisticated and flexible investigative capability could be constructed.

(d) The quantity of information gathered by a fly-by satellite would not be larger than obtainable from the ground, especially considering that several ground based installations could be used.

The benefits of the rendezvous scenario are the following:

(a) Relatively unsophisticated, freely available hardware can be used in the spacecraft design.

(b) The number of measurements and observations taken of the target is large and the target can be observed in a large number of operating modes.

(c) The fact that the number of investigations are limited puts more emphasis on and highlights the political process which makes the decision of whether to investigate or not.

(d) The rendezvous optimized satellite is relatively myopic and so poses no threat in any sense to the satellites not under investigation.

5.4 Fly-By Versus Rendezvous (Continued)

(e) Because it operates at the closest practical range to the target, it affords the best opportunity available to investigate a satellite short of retrieving the satellite from orbit and examining it on the ground.

5.5 Rendezvous Scenario

5.5.1 Geometry

Figure 5-8 illustrates the classical orbital elements which are used to define the position of a satellite in orbit about the earth. Since these elements describe a perfectly elliptical trajectory, they pre-suppose an orbit about a point mass and ignore effects of earth triaxiality, solar and lunar gravitational perturbations, aerodynamic forces and solar pressure on the orbit dynamics.

The first assumption made is that all of these perturbation effects except those due to earth oblateness are negligible. The problem addressed is that of the transfer of Paxsat from one defined orbit to another. In essence, it is assumed that both the initial and final Paxsat orbits can be defined prior to a transfer being made.

The second assumption is that the initial Paxsat orbit is circular.

The orbit transfer is then divided into a gross maneuver which places Paxsat into near proximity of the target, a mid-course phase in which the target is acquired and station-keeping phase which locks Paxsat on the target and maintains a desired station relative to it.

The gross maneuver phase itself is divided into separate maneuvers each of which is dedicated to adjusting one or more of the six orbital elements as follows:

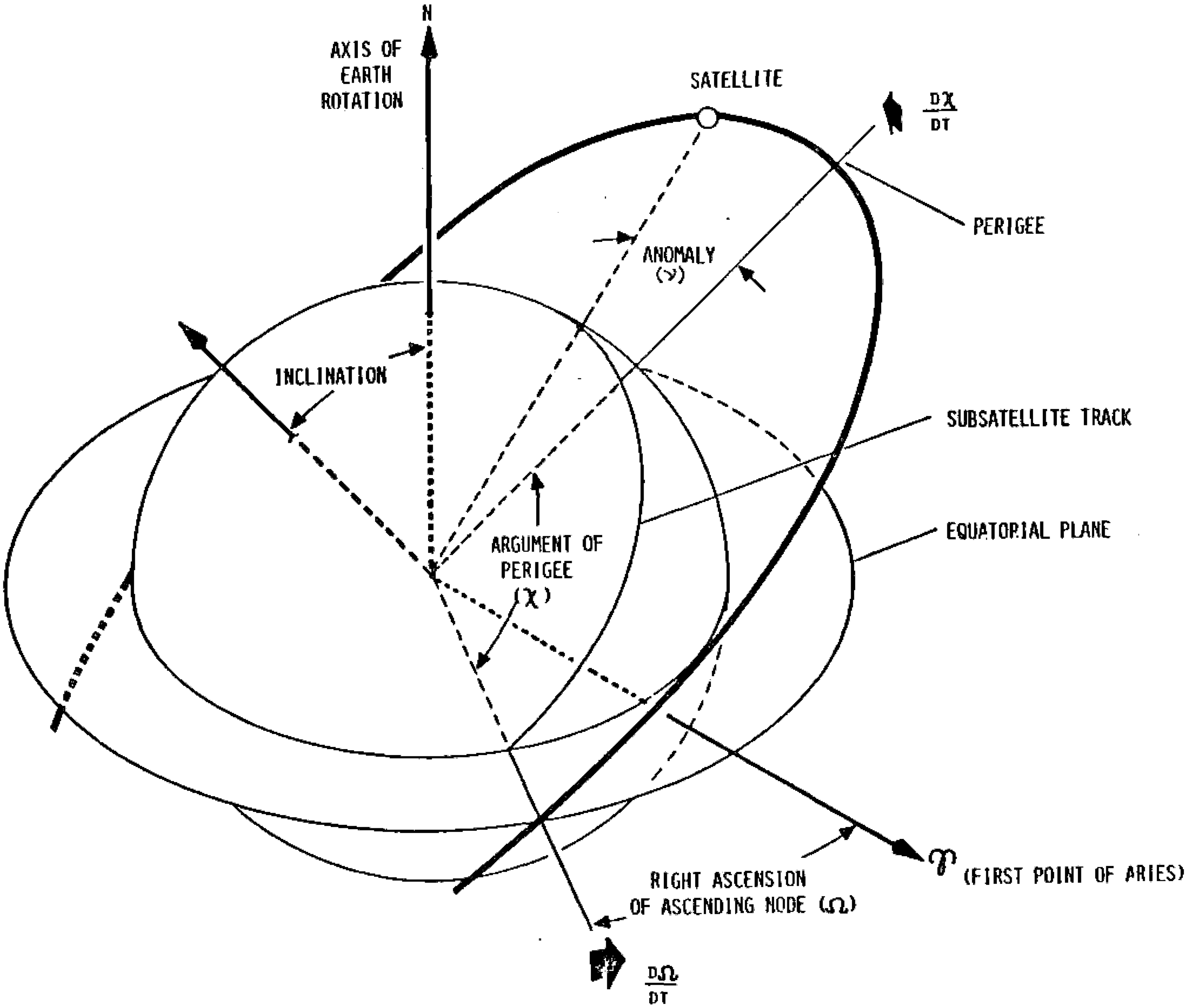


FIGURE 5-8: ILLUSTRATION OF ORBITAL ELEMENTS

5.5.1 Geometry (Continued)

MANEUVER	ORBITAL ELEMENT ADJUSTED
1	Right ascension (Ω)
2	Semi-Major Axis, Inclination (a,i)
3	Eccentricity, Argument of Perigee (e,w)
4	True Anomaly (ν)

As shown in Figure 5-8, right ascention (Ω) and inclination (i) orient the plane of the orbit relative to the earth; eccentricity (e) and semi-major axis (a) define the ellipse; argument of perigee (χ) defines the orientation of the ellipse in the orbit plane and true anomaly (ν) locates the position of the target in its orbit.

The parameters requiring the largest proportion of fuel to adjust are right ascension, inclination and semi-major axis.

Eccentricity, argument of perigee, and true anomaly will, for most orbits be more costly in time-to-transfer due to synchronization requirements rather than fuel. The exceptions will be those highly eccentric orbits of the Molniya type for which the fuel required to adjust eccentricity will obviously be important. For virtually all other target orbits, the fuel expenditures will be as indicated above.

The mid-course phase begins when Paxsat is at a lower or higher altitude, in a nearly coplanar orbit, and closing on the target vehicle and when the distance between the two satellites is such that the Paxsat homing sensor (nominally radar) can acquire relative position and velocity data.

Two types of homing laws were considered:

(a) The proportional navigation laws developed for early homing missiles

(b) The state estimation/optimal filtering techniques based on Kalman filtering developed more recently.

5.5.1 Geometry (Continued)

These two techniques were chosen as the most reliable and the most efficient respectively, and so to give an indication of the effectiveness and efficiency characteristics of these maneuvers.

Finally, the proportional navigation laws were modified slightly to be used as station-keeping algorithms which can be used to maintain Paxsat at a desired station relative to the target.

5.5.2 Determining a General Transfer Strategy

5.5.2.1 Direct Injection Transfers

As was shown in section 5.3.1, the angle of intersection ($\Delta\beta$) of two planes is given by:

$$\cos(\Delta\beta) = \cos(\Omega_1 - \Omega_2)\sin i_1 \sin i_2 + \cos i_1 \cos i_2$$

This means that for a minimum time transfer, maneuvers 1 and 2 of the gross maneuver phase may be combined into a 2-burn sequence which injects Paxsat into the target orbit directly.

If the Paxsat orbit maneuvering engine can be assumed to provide an impulsive burn, then the optimal 2-impulse transfer between inclined circular orbits of different altitudes requires the amount of velocity change shown in Figure 5-9. These curves present velocity change requirements (ΔV) assuming that some of the relative inclination is taken out at perigee and some at apogee so that the net ΔV is minimized.

It may be seen from these curves that the maneuvering ΔV required for all but very modest changes in relative inclination are too high to make direct injection practical for most orbit transfers.

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9
ΔV VALUE (m/s)	1000.	2000.	3000.	4000.	5000.	6000.	7000.	8000.	9000.	10000.

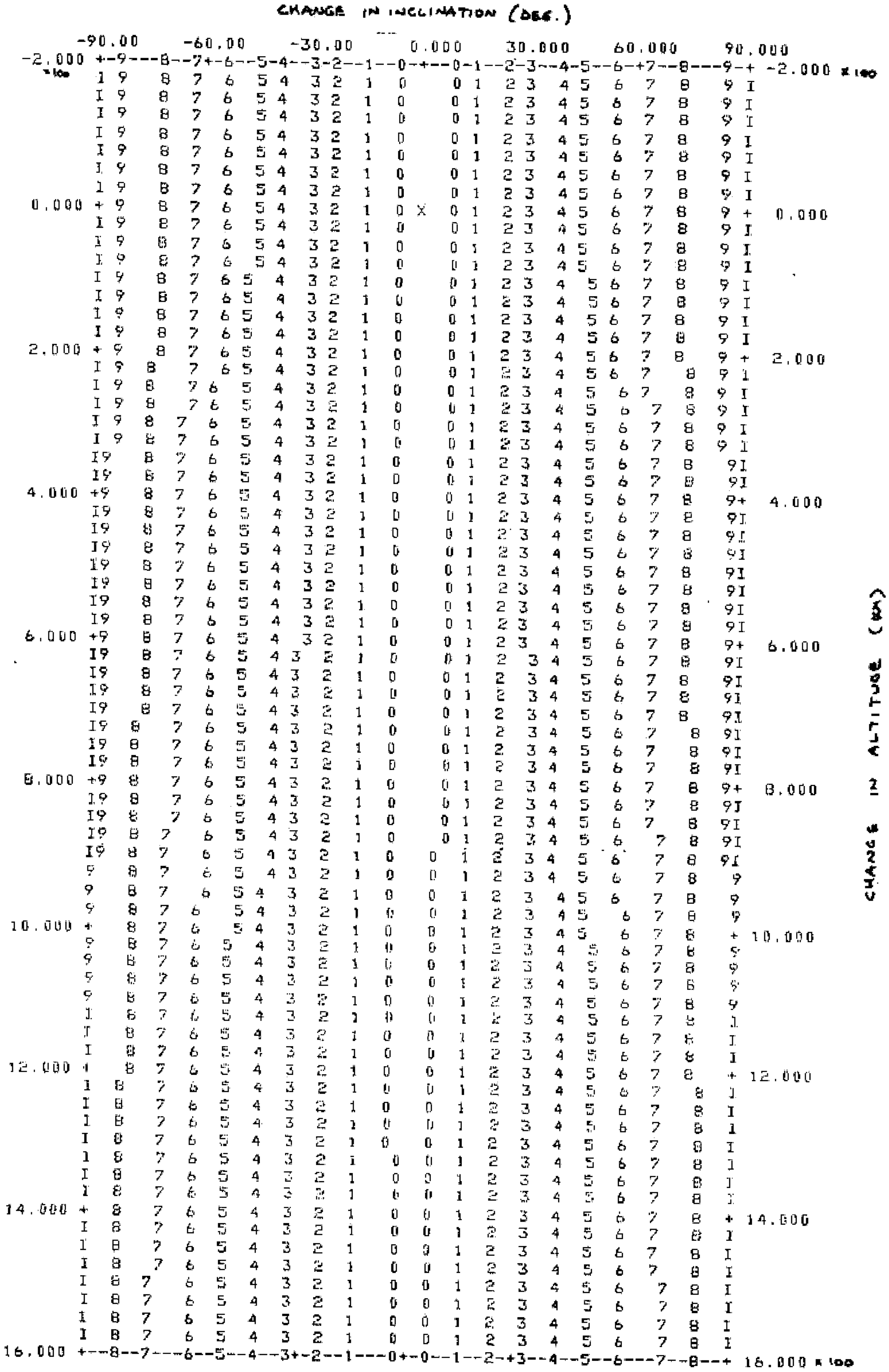


FIGURE 5-9: ΔV VERSUS INCLINATION AND ALTITUDE (NO CHANGE IN ASCENDING NODE)

5.5.2.2 Transfers Using an Intermediate Drift Orbit

Due to the nonspherical nature of the earth, there exist gravitational perturbations which change the orbital elements of any orbit.

Using Vinti's potential as a representation of the gravitational potential for an oblate body having axial symmetry, truncating all higher order terms and solving for the first order secular perturbations gives:

$$\Delta a = 0$$

$$\Delta e = 0$$

$$\Delta \omega = 3\pi J_2 \left(\frac{R}{p}\right)^2 \left(2 - \tfrac{5}{2}\sin^2 i\right) \text{ rad/rev}$$

$$\frac{\Delta M}{nt} = \frac{3}{2} J_2 \left(\frac{R}{p}\right)^2 \sqrt{1-e^2}\left(1 - \frac{3}{2}\sin^2 i\right) \text{ rad/rev}$$

$$\Delta i = 0$$

$$\Delta \Omega = -3\pi J_2 \left(\frac{R}{p}\right)^2 \cos i \text{ rad/rev}$$

[Ref.18]

The perturbations of primary interest are to altitude, inclination and right ascension. Since a and i are both zero (to first order), the only usable perturbation is to right ascension. Converting the formula to units of degrees per day gives:

$$\Delta \Omega = \frac{-\tfrac{3}{2} J_2 \sqrt{\mu}\, R_e^2 \cos i}{a^{3.5}\left(1-e^2\right)^2} \cdot \frac{(180)(3600)(24)}{\pi} \text{ degrees/day}$$

and this is plotted in Figure 5-10 on a modified altitude/inclination plot.

There are other perturbations to the orbit of a satellite due to solar and lunar gravity, solar radiation pressure and aerodynamic drag. However, these do not produce siginificant effects in the elements of

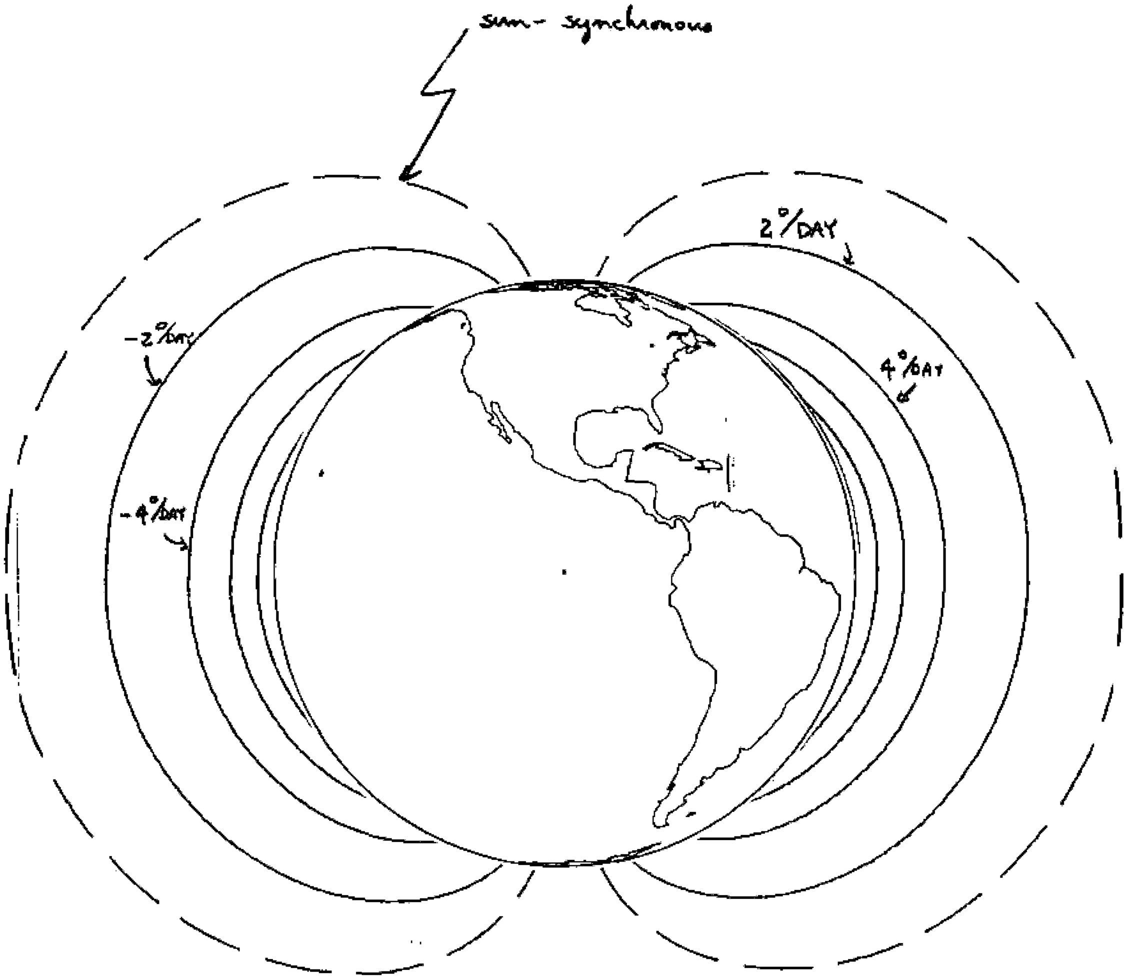


FIGURE 5-10: AN ILLUSTRATION OF LINES OF CONSTANT DRIFT (TO FIRST ORDER)

5.5.2.2 Transfers Using an Intermediate Drift Orbit (Continued)

interest (a, i, Ω) for nearly circular orbits. Therefore, for the purpose of this preliminary analysis, they are ignored.

Earth oblateness may be used then to eliminate without using fuel the component of relative inclination ($\Delta\beta$) due to right ascension. Assuming that an infinite amount of time is available to perform the orbit transfer, only semi-major axis (a) and inclination (i) changes need to consume fuel.

5.5.2.3 Determining the Optimum Drift Orbit

Using natural perturbations to precess the Paxsat orbit implies the selection of a drift orbit in which Paxsat would wait while the perturbation acts.

The algorithm chosen to determine the optimum drift orbit is constrained by the following requirements:

(a) It must be quick enough to allow parametric investigations (i.e. to calculate ΔV to a range of target orbits) without using an excessive amount of computer time.

(b) It must be accurate enough to give a realistic estimate of ΔV required.

(c) It must be clear enough to implement (write, test, debug) on a digital computer in a short time.

(d) It must handle non-coplanar transfers and bi-directional transfer (orbit raising and lowering).

(e) It should account for aerodynamic drag in very low orbits.

(f) It should find the drift orbit which optimizes net ΔV for the transfer (parking to drift to target).

The problem was subdivided into three parts:

(a) Determining the optimum eccentricity (e) of the drift orbit

5.5.2.3 Determining the Optimum Drift Orbit

(b) Determining the optimum semi-major axis (a)

(c) Determining the optimum inclination (i)

This was done because these orbital elements (a,e,i) control the first order expression for drift rate.

The search for an optimum drift orbit could then be done in the following way:

(a) Allocate a desired drift period in which the Paxsat orbit phase must precess towards that of the target.

(b) Determine the initial offset of the Paxsat ascending mode with respect to that of the target.

(c) Determine the required drift rate of the ascending mode (this reduces the degree of freedom of the problem from three to two).

(d) Perform a 2-dimensional parameter scan in altitude and eccentricity to determine which drift orbit minimizes net ΔV.

This algorithm would require the calculation of between 10^3 to 10^4 scenarios to optimize just one parking orbit-to-target orbit transfer.

The calculation of ΔV required to burn from one orbit to another was based on the following assumptions:

(a) Hohmman type 2-impulse transfer

(b) Optimum inclination change split between perigee and apogee.

(c) Combined correction of inclination and semi-major axis.

(d) No consideration of burn duration effects on transfer efficiency.

5.5.2.3 Determining the Optimum Drift Orbit (Continued)

Impulse burns rather than finite duration burns were assumed due to the difficulty in determining the effect of non-impulsive burns on out of plane transfers. An in-plane correction could have been applied but it was felt that a uniform set of assumptions was preferable.

Also, though impulsive burns do not give a conservative estimate for transfer ΔV, it is possible in theory to apply operational constraints on the way the transfer is accomplished so as to minimize any additional fuel impact. Transfer efficiency can be traded-off against transfer time.

It was assumed then that the fully optimized 2-impulse transfer would be representative of an achievable transfer ΔV requirement.

In time trials, the algorithm for one orbit transfer calculation was executed in approximately 5 ms on the available computer system. To perform a parameter scan of the kind suggested above for a set of 100 target orbits would require:

$(10^3)(100)(2)(0.005) = 1000$ S (17 min) at least

and

$(10^4)(100)(2)(0.005) = 1000$ S (2-3/4 Hrs) at most

This time was considered excessive in view of the fact that it would take this time to determine the volume of space accessible by Paxsat given just one initial Paxsat parking orbit and one transfer time specification.

Therefore some analytical means to reduce the number of options for a drift orbit was required. Further, since semi-major axis and inclination are tied intimately with the properties of the target orbits (most targets in low earth orbit and geosynchronous orbits are in near-circular orbits), therefore some means of fixing drift orbit eccentricity to a given value would probably be

5.5.2.3 Determining the Optimum Drift Orbit (Continued)

the best way to reduce the number of degrees of freedom. It was therefore to be determined whether and under what conditions it is more efficient to change semi-major axis than eccentricity in order to change the drift rate.

5.5.2.4 Eccentricity of the Optimum Drift Orbit

In transfers where transfer time is critical, it is necessary to precess at the highest possible rate relative to a target plane. This is achievable either through a high absolute precession rate or a very low (or negative) precession rate. The focus is on finding any drift orbit that will give a fast enough drift to meet the requirement. In this case, finding the best orbit is relatively straightforward.

For transfers in which time is not a factor of great importance, the problem is to find the most efficient drift orbit as well.

In setting the eccentricity for an orbit whose drift rate is as great as possible, the first problem can be rephrased as follows:

> "If the greatest drift rate is desired, is it better to have a circular orbit of the smallest semi-major axis possible, or to have a slightly larger semi-major axis and some eccentricity as well?"

Choice 1

$$a = R_p = R$$

Choice 2

$$R_p = R$$
$$R_A = R_p \left(\frac{1+e}{1-e}\right)$$
$$a = R_p/(1-e)$$

where e takes any value.

5.5.2.4 Eccentricity of the Optimum Drift Orbit (Continued)

The drift rates are compared as follows:

Choice 1

$$\dot{\Omega}_1 = \frac{-K \cos i}{R_p^{3.5}}$$

Choice 2

$$\dot{\Omega}_2 = \frac{-K \cos i}{\left(\frac{R_p}{1-e}\right)^{3.5} (1-e^2)^2}$$

Choice 1 is preferred if:

$$\frac{(1-e^2)^2}{(1-e)^{3.5}} > 1$$

Since this condition is always true except when e=0, therefore it is true that the fastest drift orbit is circular. The drift orbit should therefore be at the lowest feasible altitude and be circular if the highest drift rate is desired.

The investigation then turns to how a small absolute drift rate could be achieved.

In order to lower the absolute drift rate, three choices are possible:

(a) A change in inclination towards the pole.

(b) A change in semi-major axis and/or eccentricity (this will lower the drift rate but not allow an absolute drift rate of opposite sign, only going to the other side of the pole can change the sign of the drift rate).

5.5.2.4 Eccentricity of the Optimum Drift Orbit (Continued)

(c) A combination of the above.

In order to clarify the choice given in (b), one might ask, "Is it more efficient in fuel to change semi-major axis rather than eccentricity to lower the drift rate?"

It is assumed that the initial (parking) orbit and final (target) orbit are both circular, then one must determine the optimum eccentricity and semi-major axis of the drift orbit whose drift rate is known.

$$\dot{\Omega} = \frac{-\frac{3}{2}\sqrt{\mu}\, R_e^2 \cos i}{a^{3.5}\,(1-e^2)^2}$$

where

$\dot{\Omega}$ is the drift rate
R_e is the radius of the earth
μ is a constant (38600 km^3/S^2)
a is the drift orbit semi-major axis
e is the drift orbit eccentricity

First, the effect of changing eccentricity on the drift rate is given by:

$$\frac{\partial\dot{\Omega}}{\partial e} = \frac{-6\sqrt{\mu}\, R_e^2 \cos i}{a^{3.5}\,(1-e^2)^2} \cdot \frac{e}{(1-e)}$$

It may be noted that

$$\frac{\partial\dot{\Omega}}{\partial e} = 0$$

5.5.2.4 Eccentricity of the Optimum Drift Orbit (Continued)

for an initially circular orbit. The effect of changing semi-major axis is given by:

$$\frac{\partial \dot{\Omega}}{\partial a} = \frac{\frac{10.5}{2}\sqrt{\mu}\, R_e^2 \cos i}{a^{4.5}(1-e^2)^2}$$

In order to guage the fuel effectiveness of changing eccentricity or changing semi-major axis in order to change the drift rate, the following derivatives are desired:

$$\frac{\partial \Delta V_a}{\partial \dot{\Omega}} \quad \text{and} \quad \frac{\partial \Delta V_e}{\partial \dot{\Omega}}$$

but $$\frac{\partial \Delta V_a}{\partial \dot{\Omega}} = \frac{\partial \Delta V_a}{\partial a} \cdot \frac{\partial a}{\partial \dot{\Omega}}$$

and $$\frac{\partial \Delta V_e}{\partial \dot{\Omega}} = \frac{\partial \Delta V_e}{\partial e} \cdot \frac{\partial e}{\partial \dot{\Omega}}$$

As is shown in Appendix B, the following expressions are true

$$\frac{\partial \Delta V_a}{\partial a} = \frac{1}{2}\sqrt{\frac{\mu}{a^3}} \quad \text{and} \quad \frac{\partial \Delta V_e}{\partial e} = \frac{1}{2}\sqrt{\frac{\mu}{a}}$$

For an initially circular orbit of radius a.

Therefore $$\frac{\partial \Delta V_a}{\partial \dot{\Omega}} = \left(\frac{1}{2}\sqrt{\frac{\mu}{a^3}}\right)\left(\frac{a^{4.5}}{\frac{10.5}{2}\sqrt{\mu}\, R_e^2 \cos i}\right)$$

5.5.2.4 Eccentricity of the Optimum Drift Orbit (Continued)

and

$$\frac{\partial \Delta V_e}{\partial \Omega} = 0$$

This may be interpreted to mean that although changing eccentricity alone is not more expensive in ΔV, it is relatively ineffective for an initially circular orbit.

Since it appears that the optimum adjustment of a circular orbit to produce a relative drift appears to imply adjustment of semi-major axis rather than eccentricity, the question remains whether this is also true for an initially eccentric orbit.

Again from Appendix B, the following expressions are true for eccentric orbits:

$$\frac{\partial(\Delta V_{AE})}{\partial a} = +\frac{1}{2a}\sqrt{\frac{\mu(1+e)(1-e)}{a}}$$

and

$$\frac{\partial(\Delta V_{ee})}{\partial e} = \frac{1}{2}\sqrt{\frac{\mu}{a(1+e)(1-e)}}$$

Forming the expressions for

$$\frac{\partial \Delta V_a}{\partial \Omega}$$

and

$$\frac{\partial \Delta V_e}{\partial \Omega}$$

as before gives

$$\frac{\partial \Delta V_a}{\partial \Omega} = \frac{1}{2a}\sqrt{\frac{\mu(1+e)(1-e)}{a}}\left(\frac{a^{4.5}}{5.25\sqrt{\mu}\,R_e^{2}\cos i}\right)$$

5.5.2.4 Eccentricity of the Optimum Drift Orbit (Continued)

$$\frac{\partial \Delta V_e}{\partial \dot{\Omega}} = \frac{1}{2}\sqrt{\frac{\mu}{a(1+e)(1-e)}}\left(\frac{a^{3.5}(1-e^2)^3}{6\sqrt{\mu}\, k_e^2 \cos i \; e}\right)$$

To determine which element (semi-major axis a or eccentricity e) gives a more favorable decrease in the satellite drift rate , a ratio can be formed

$$\frac{\dfrac{\partial \Delta V_a}{\partial \dot{\Omega}}}{\dfrac{\partial \Delta V_e}{\partial \dot{\Omega}}} = \frac{6e(1+e)(1-e)}{5.25(1-e^2)^3}$$

This expression is evaluated in Table 5-1 for a range of eccentricities.

This may be interpreted to mean that for orbits which have an initial eccentricity of less than approximately 0.5, semi-major axis adjustment produces the same change in drift rate as eccentricity change for less fuel.

For orbits of the Molniya type, whose eccentricity is greater than 0.5, changing eccentricity produces drift rate changes more efficiently.

It may be concluded then that for LEO satellites in near-circular orbits, one may select circular drift orbits as a reasonable approximation to the optimal eccentricity. For Molniya orbits, a different strategy must be used.

It must be noted that this analysis does not take into account inclination changes or air drag effects. In the final analysis for detail design purposes, the full blown parameter scan will need to be performed to optimize both semi-major axis and eccentricity and to account for finite burn times.

Summarizing, the following are the features of the algorithm:

(a) Spends approximately 1 S of computer time to optimize a single transfer thus allowing parameter studies.

SPAR

TABLE 5-1 RELATIVE EFFICIENCY OF CHANGING SEMI-MAJOR AXIS, AS A FUNCTION OF INITIAL ORBIT ECCENTRICITY

e	$\frac{\partial \Delta V_a}{\partial \Omega} \Big/ \frac{\partial \Delta V_e}{\partial \Omega}$
0	0
0.05	0.057
0.1	0.117
0.2	0.248
0.4	0.648
0.5	1.016
0.6	1.674
0.8	7.055
0.0	

5.5.2.4 Eccentricity of the Optimum Drift Orbit (Continued)

(b) The accuracy is difficult to estimate because for any given transfer, a more optimal strategy may exist. The algorithm computes fully-optimized 2 impulse transfers making no allowance for finite duration burns, and assuming circular drift orbit.

(c) Algorithm is quite easy to understand, there are no special cases or difficult logic sequences.

(d) Handles non-coplanar and bi-directional transfers.

(e) Uses an analytical version of the Jacchia 77 model atmosphere, making no allowance for day/night variations. A first order estimate of altitude loss due to drag is used which allows for atmospheric rotation.

(f) Finds the optimum drift orbit in terms of minimum ΔV.

For the purpose of parameter studies which produce a contour plot of ΔV against target orbit altitude and inclination, the contour plotting algorithm uses a quadratic Lagrange interpolation formula.

5.5.3 Delta-V Requirements for LEO Operations

The transfer strategy developed above requires as input the following parameters:

(a) Parameter 1 - Initial Paxsat position

(b) Parameter 2 - Transfer (drift) time

(c) Parameter 3 - Range of target altitudes and inclinations

(d) Parameter 4 - Required change in right ascension of ascending node.

Parameter 2 was varied in relatively coarse steps (60, 90, 120 days). Parameter 3 was determined from the sample satellite data base which indicated a typical spread of operational military satellites. Parameter 1

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D
ΔV VALUE	600.0	300.0	1000.	1200.	1400	1600.	1800.	2000.	2200	2400	2600.	2800.	3000.	4000. m/s

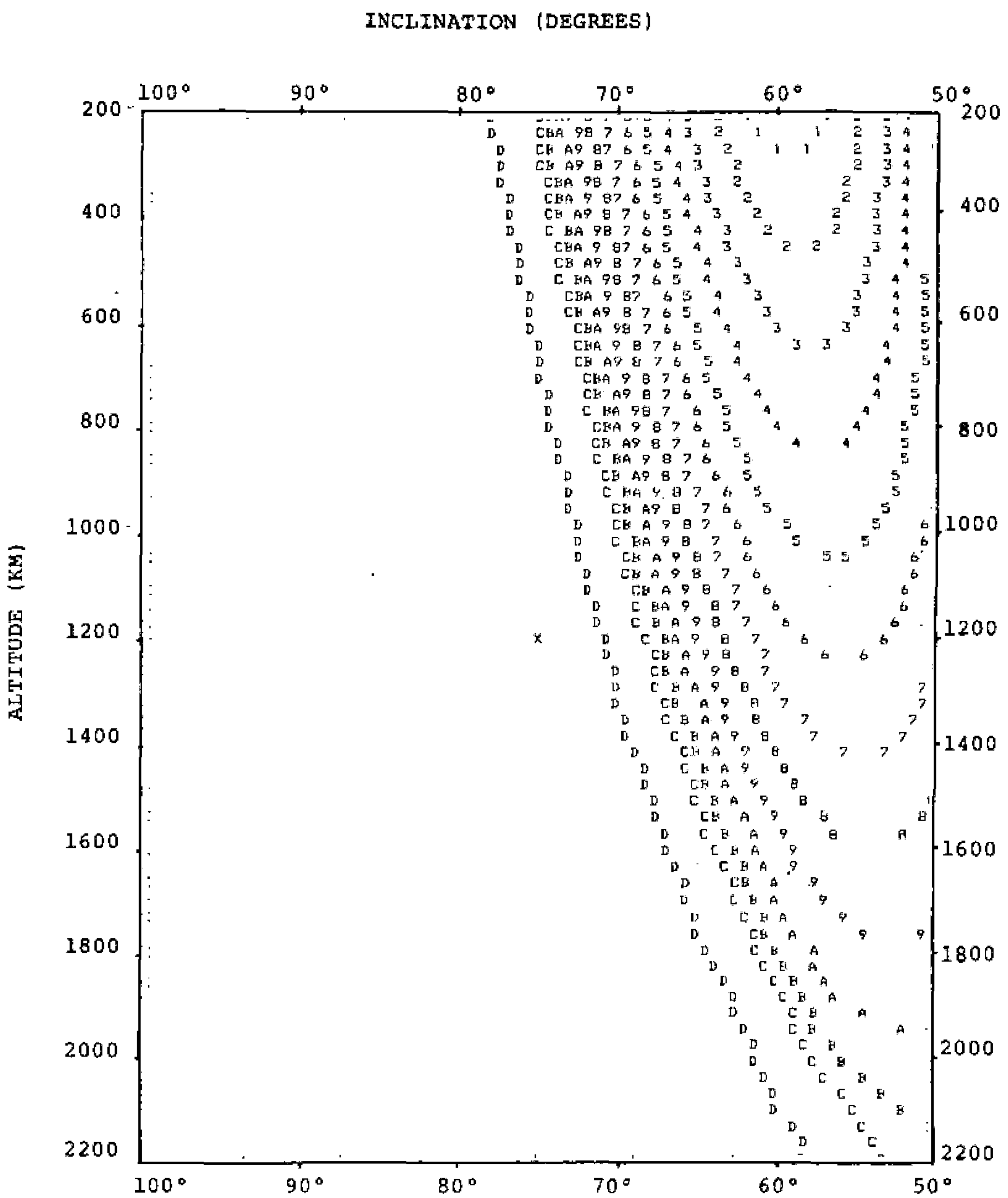


FIGURE 5-11(a): 60-DAY DELTA-V CONTOURS (FOR 90 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING AV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D
AV VALUE	600.0	800.0	1000.	1200.	1400.	1600.	1800	2000.	2200	2400.	2600.	2800.	3000.	4000. m/s

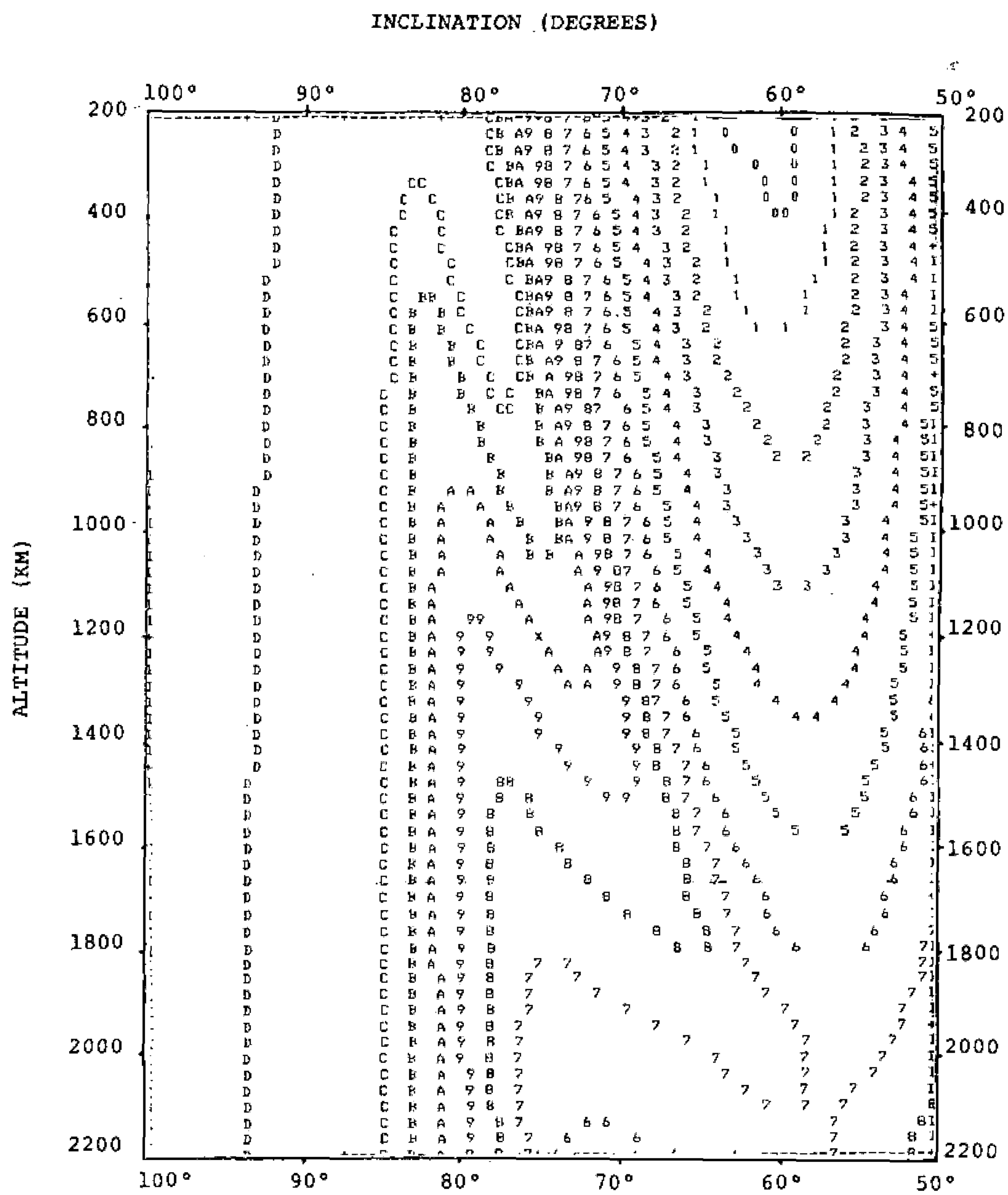


FIGURE 5-11(b): 90-DAY DELTA-V CONTOURS (FOR 90 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D
ΔV VALUE	600.0.	900.0	1000.	1200.	1400.	1600.	1800.	2000.	2200.	2400.	2600.	2800.	3000.	4000.0 m/s

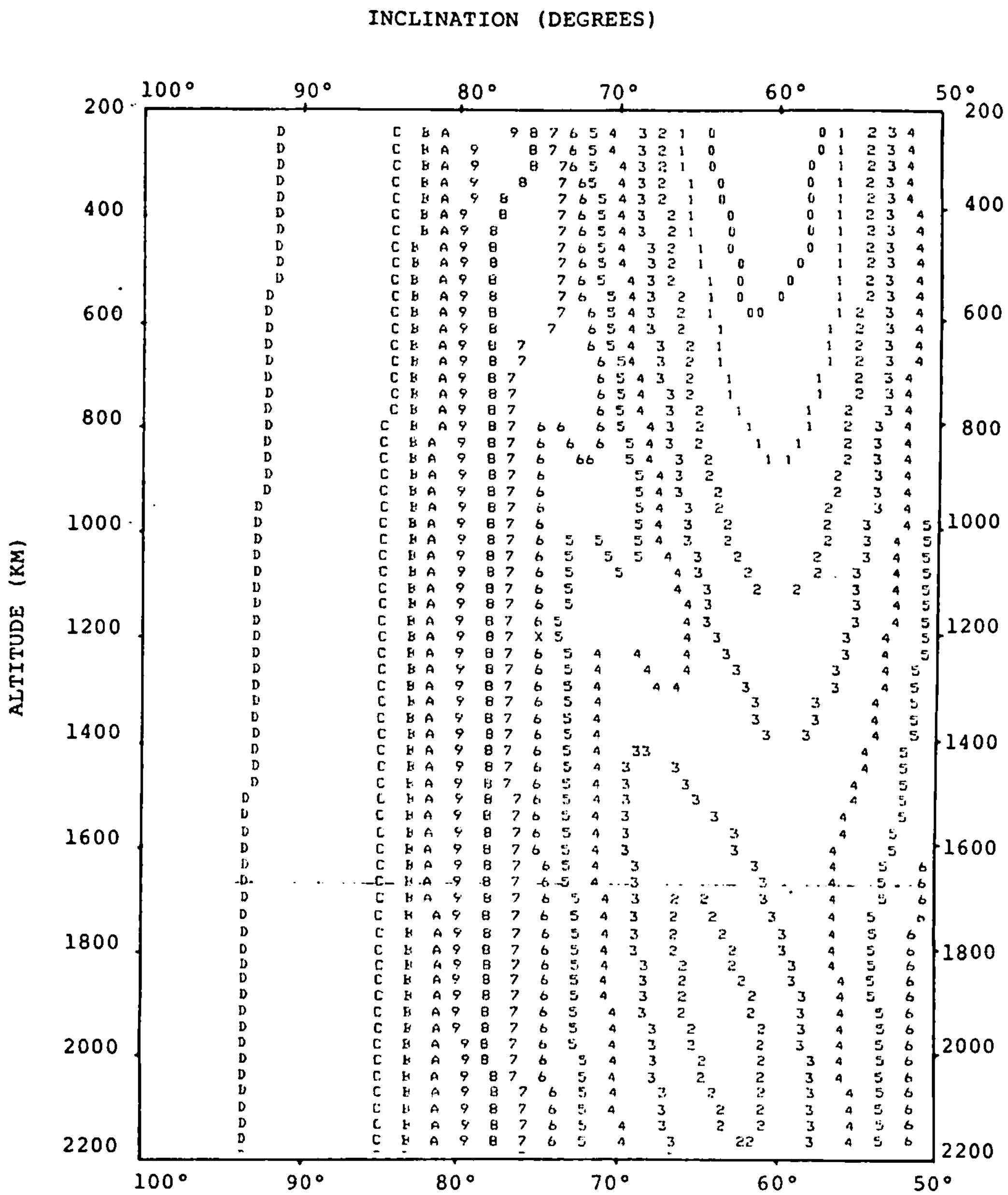


FIGURE 5-11(c): 120-DAY DELTA-V CONTOURS (FOR 90 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D
ΔV VALUE	600.0	800.0	1000.	1200.	1400	1600.	1800.	2000	2200.	2400.	2600.	2800.	3000.	4000. m/s

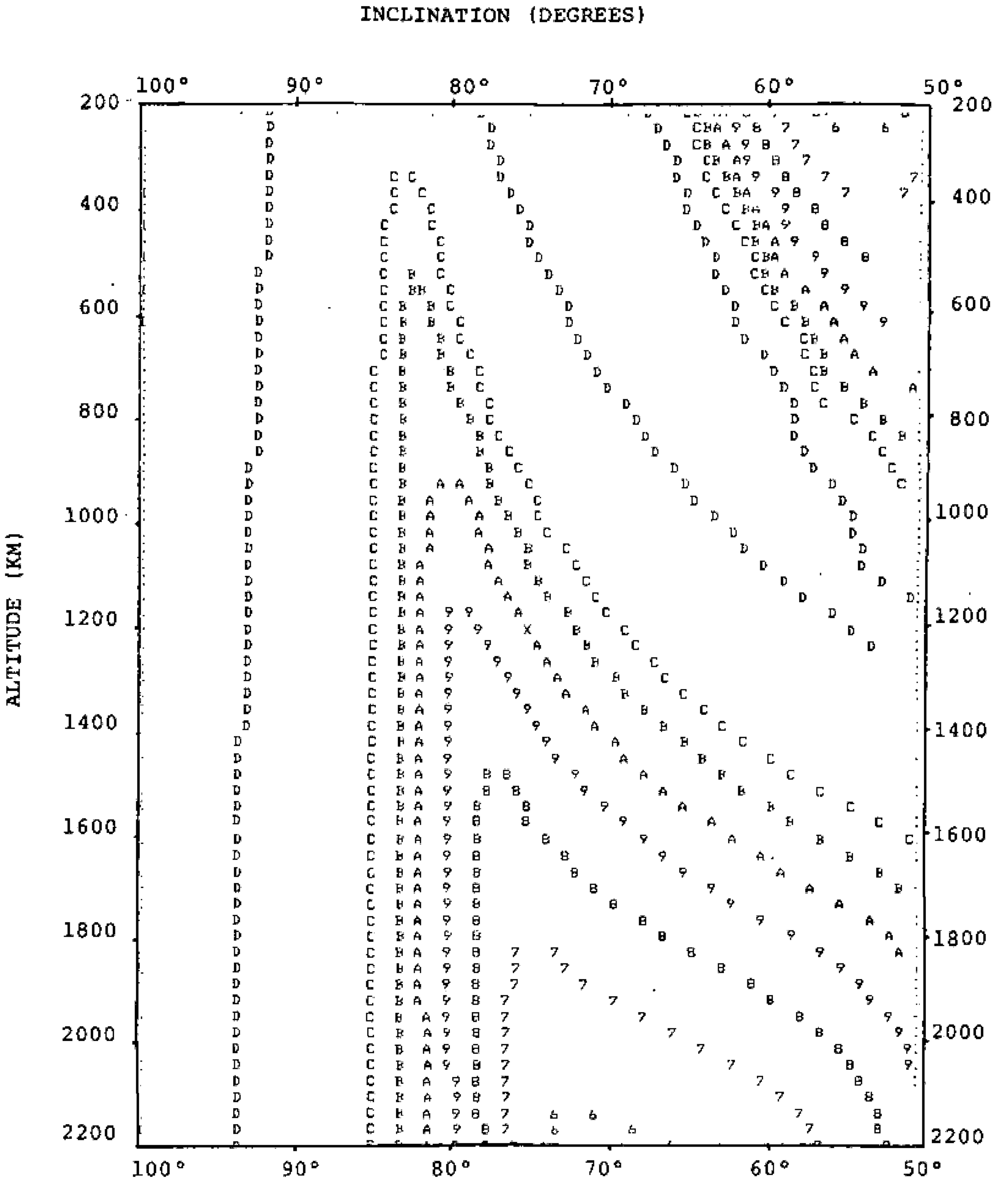


FIGURE 5-11(d): 60-DAY DELTA-V CONTOURS (FOR 180 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D
ΔV VALUE	600.0	800.0	1000.	1200.	1400.	1600.	1800.	2000.	2200.	2400.	2600.	2800.	3000.	4000. m/s

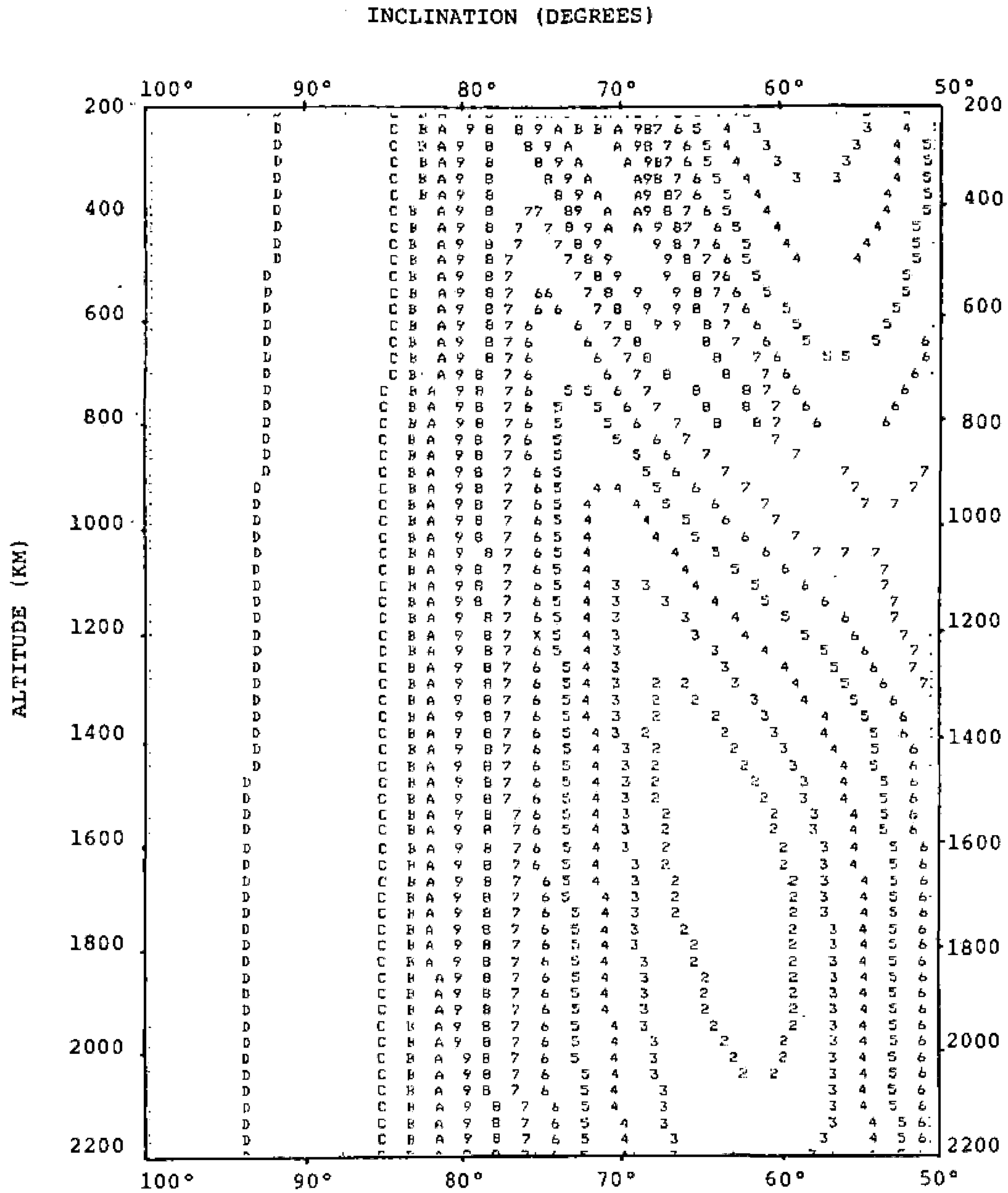


FIGURE 5-11(e): 90-DAY DELTA-V CONTOURS (FOR 180 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D	
ΔV VALUE	600.0	800.0	1000.	1200.	1400.	1600	1800	2000.	2200.	2400.	2600.	2800.	3000.	4000.	m/s

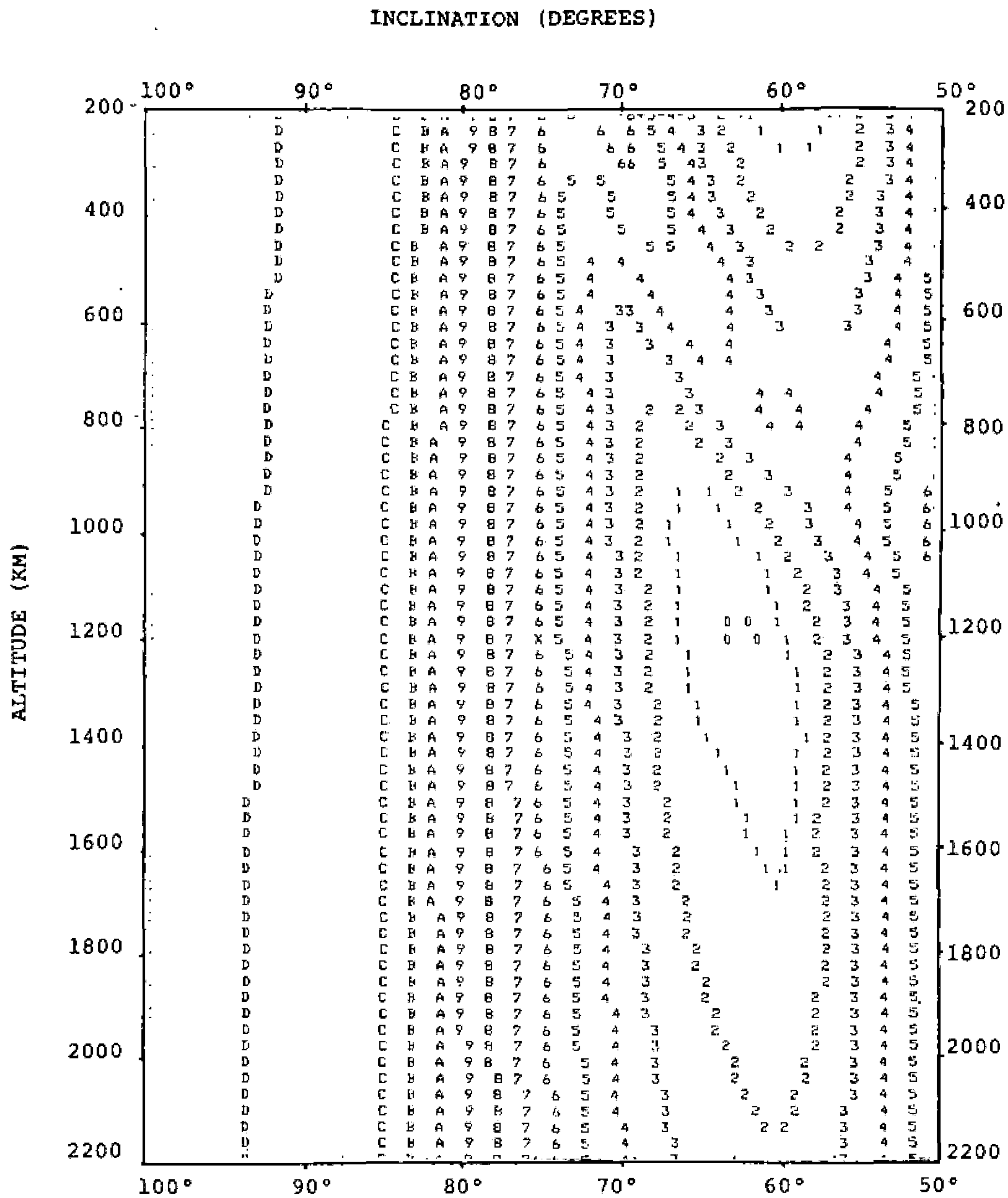


FIGURE 5-11(f): 120-DAY DELTA-V CONTOURS (FOR 180 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D	
ΔV VALUE	600.0	800.0	1000.	1200.	1400.	1600.	1800.	2000.	2200.	2400.	2600.	2800.	3000.	4000.	m/s

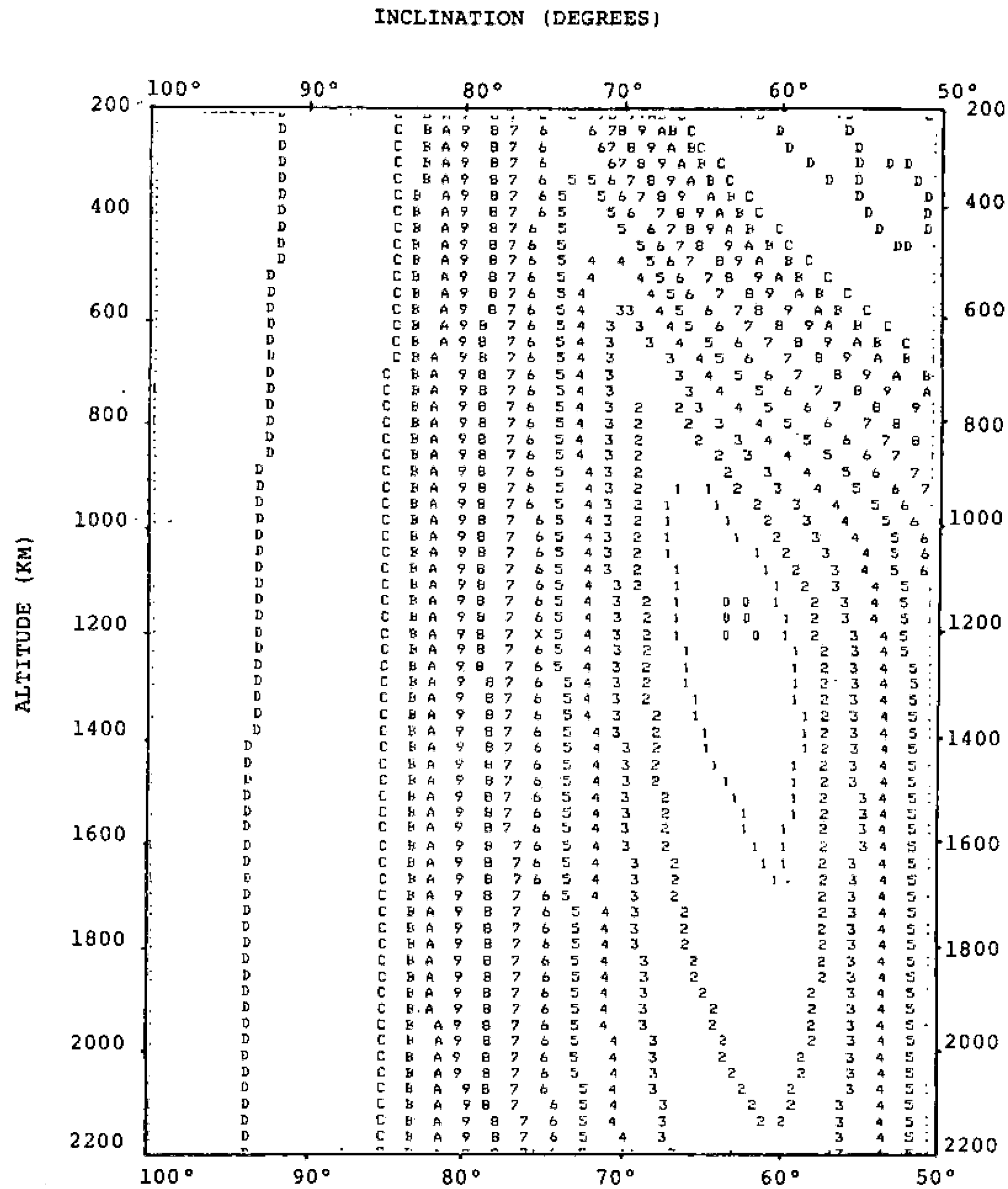


FIGURE 5-11(g): 60-DAY DELTA-V CONTOURS (FOR 270 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D	
ΔV VALUE	600.0	800.0	1000.	1200.	1400.	1600.	1800.	2000.	2200.	2400.	2600.	2800.	3000.	4000.	m/s

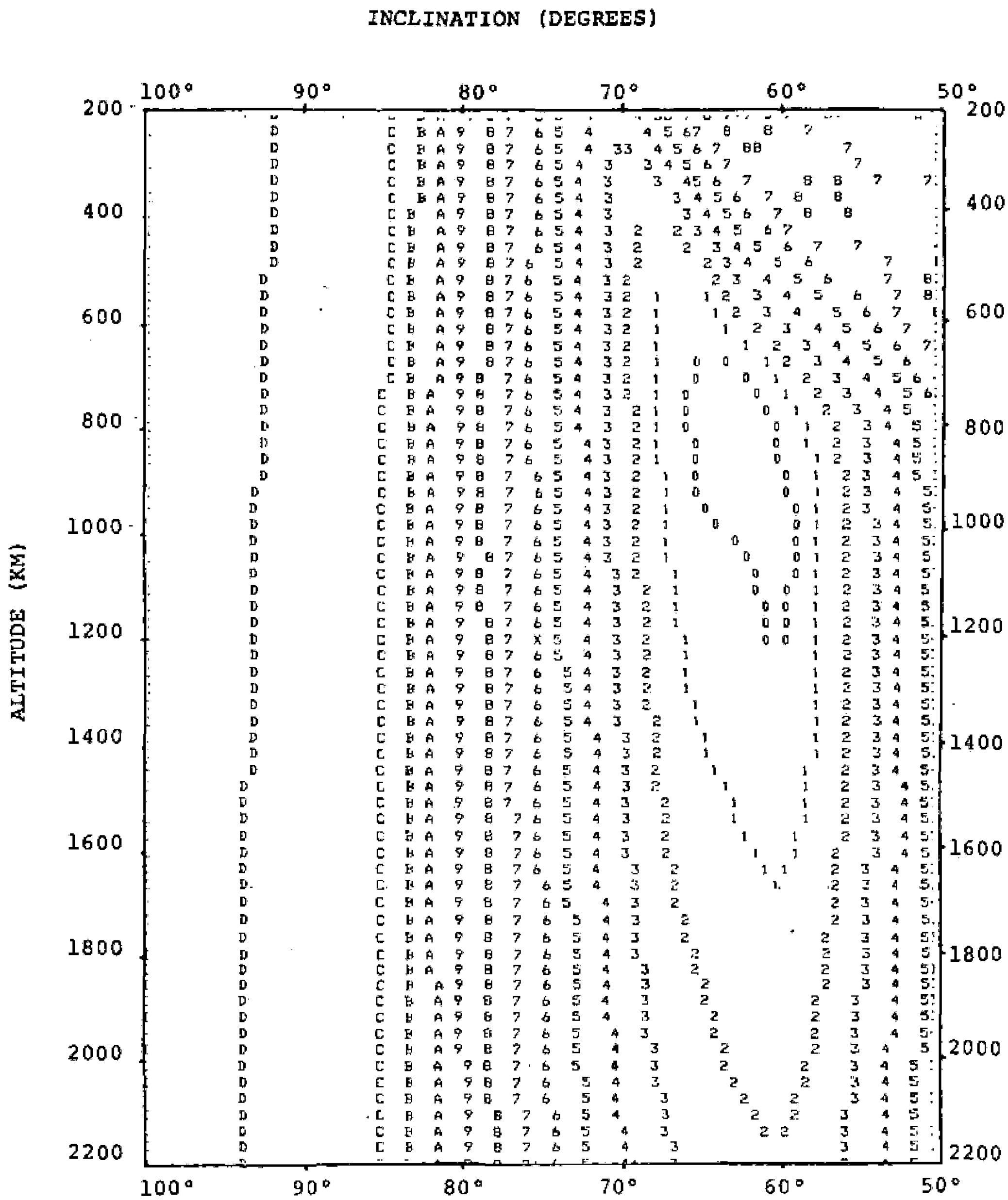


FIGURE 5-11(h): 90-DAY DELTA-V CONTOURS
(FOR 270 DEG. DRIFT)

CONTOUR SYMBOLS AND CORRESPONDING ΔV VALUES

SYMBOL	0	1	2	3	4	5	6	7	8	9	A	B	C	D
ΔV VALUE	600.0	800.0	1000.	1200.	1400.	1600.	1800.	2000.	2200.	2400.	2600.	2800.	3000.	4000. m/s

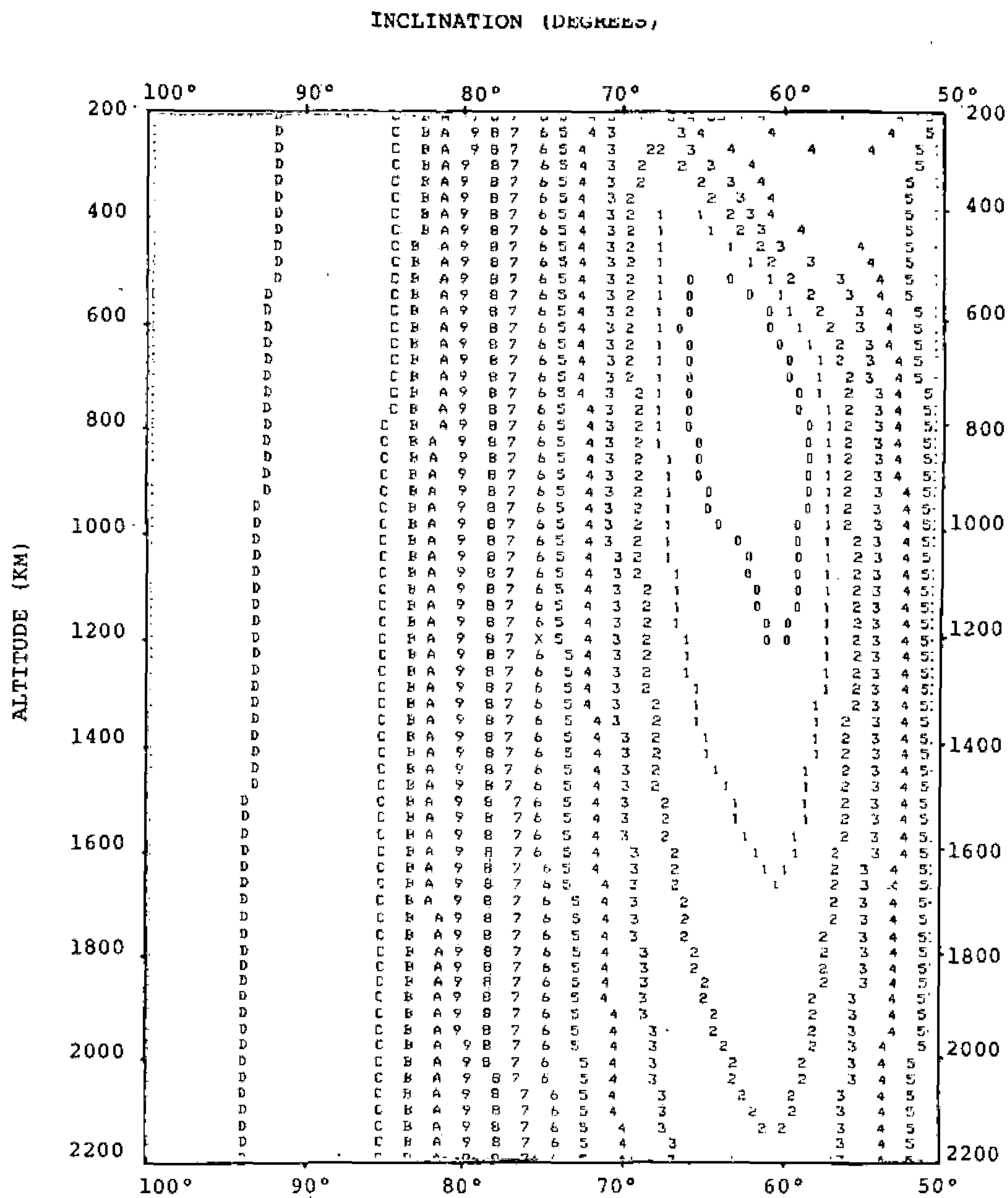


FIGURE 5-11(i): 120-DAY DELTA-V CONTOURS (FOR 270 DEG. DRIFT)

5.5.3 Delta-V Requirements for LEO Operations (Continued)

was an optimized variable. The initial Paxsat position was chosen which maximized the Paxsat range of operations.

The parameter most difficult to guage was parameter 4, the required change in ascending node. Ideally, the sample data base would have held a typical distribution of ascending nodes, and the V curves could have corresponded to a certain percentage of satellites investigated. Since that information was not available, curves were developed for 90°, 180° and 270° of node rotation. Figure 5-11 shows the ΔV requirements for LEO in the form of ΔV contours on altitude/inclination plots.

The Paxsat spacecraft was located at an altitude of 400 km and an inclination of 62°. The velocity increment (ΔV) required to transfer to any other altitude and inclination is given by the ΔV contour at that altitude and inclination.

The figures show ΔV versus altitude and inclination for a range of ascending node differences and drift times.

Some generalization may be made of the behavior of the ΔV contours with different drift times and drift distances. These are the following:

(a) The direction of drift matters. That is, drifting +90° of node is not the same as drifting -90° of node.

(b) There are basically two regions of accessibility distinguished by whether a drift orbit other than the parking orbit (initial Paxsat orbit) is required or not. The character of the ΔV contours is different in these two regions.

(c) Changing node becomes less significant as the target inclination goes further away from the initial Paxsat inclination. This is because the relative drift between the Paxsat and Target planes increases with the difference in their inclinations.

5.5.3 Delta-V Requirements for LEO Operations (Continued)

(d) When looking at the full range of operation of Paxsat, net V required for the orbit transfer is a stronger function of time allowed for drifting than of node required to drift in that time for nodal drift changes of 90°.

More specifically, the results indicate that for large node changes, a drift time of less than 60 days is extremely expensive in fuel. A drift of 90 days provides better fuel performance and a 120 days drift provide close to the best achievable performance for the large majority of transfers.

(e) Because orbits relatively close in inclination experience similar perturbations, they are the orbits where the selection of a good drift orbit is most critical. For these orbits, the majority of the fuel is spent getting to and from the drift orbit rather than in eliminating the (relatively small) difference in inclination.

(f) Because of the way that the lines of constant natural perturbation are distributed, it is more difficult to find a good drift orbit near the pole than away from the pole.

It is interesting to note that in order to cover all of the LEO inclinations (58° to 104°) with a minimum number of satellites, the fuel requirements per satellite is driven by the inclination change per satellite.

Assuming no node change requirement at all, the fuel requirement to cover 23° (i.e. in a system with 2 LEO Paxsats, each Paxsat would cover 23° of inclination) would be given simply by:

$$\Delta V = 2V_c \sin \Delta i/2$$

where

Vc is the characteristic satellite velocity (≑7.5 km/S)
Δi is the required inclination change (23°)
or V = 3000 m/S

5.5.3 Delta-V Requirements for LEO Operations (Continued)

Looking at the ΔV contours, one can see that the node change impacts this fuel requirement very little, provided that enought drift time is allowed so that no drift orbit must be selected which would cause an expenditure of more than 3,000 m/s for the transfer.

To recapitulate, using the reversed approach, the logic goes as follows:

(a) 3,000 m/s of ΔV is required as a minimum in order just to meet the inclination change requirement (assuming 2 LEO Paxsats).

(b) Therefore, 3,000 m/S is the limit on any parking orbit to Target orbit transfer.

(c) Therefore, the available drift orbits are limited and therefore also, the maximum relative drift is limited.

(d) Therefore, the drift time must be selected so as to allow the desired node drift with the desired maximum fuel expenditure. A drift time of between 90 and 120 days, it turns out, must be allowed for most transfers.

5.5.4 Delta-V Requirements for Semi-Synchronous Operations

There are two types of semi-synchronous orbits in use, circular and eccentric (Molniya).

Time constraints did not permit a full analysis of the semi-synchronous case, but because the characteristic velocity for node changes are 3,800 m/S and 2,500 m/S for the circular and Molniya orbits respectively, as opposed to a 7,600 m/S characteristic velocity for a 400 km altitude orbit, it is felt that the semi-synchronous case is not a worst case. Also, semi-synchronous satellites, especially Molniya satellites, tend to have a common inclination rather than a wide range of inclination as with LEO satellites. The analysis of semi-synchronous satellites was therefore placed second in priority behind analysis of LEO operations.

5.5.4 Delta-V Requirements for Semi-Synchronous Operations (Continued)

Additionally, most of the primary targets for weapons and therefore presumably the weapon themselves, would be situated in LEO, also making it more important to investigate.

It appears that the main uses for fuel in semi-synchronous operation would be involved with getting to station in the first place. Below are some order-of-magnitude calculations showing how much velocity increment would be required to go from LEO to semi-synchronous for an investigation.

5.5.4.1 Molniya Orbits

The drift rate of a Molniya orbit is:

$$\dot{\Omega} = \frac{-\frac{3}{2} J_2 R_e^2 \sqrt{\mu} \cos i}{a^{3.5} (1-e^2)^2}$$

$$J_2 \doteq 1082.6 \times 10^{-6}$$
$$R_e \doteq 6378 \text{ Km}$$
$$\mu \doteq 398600 \text{ Km}^3/\text{s}^2$$
$$a \doteq 26600 \text{ Km}$$
$$e \doteq 0.743$$
$$i \doteq 63.0^\circ$$

$$\dot{\Omega} \doteq 3 \times 10^{-8} \text{ rad/sec}$$

$$\dot{\Omega} \doteq 0.154 \text{ deg/day}$$

A satellite in a 63° inclination and 400 km altitude circular orbit has a drift rate of:

$$\dot{\Omega} = 3.65^\circ/\text{day}$$

and this provides a drift of 360° in less than 100 days.

5.5.4.1 Molniya Orbits (Continued)

The change in velocity required to go from a circular 400 km altitude orbit to a Molniya orbit is 2,500 m/S. This means that a Paxsat stationed in LEO at 63° could make one investigation of any Molniya-type satellite within 100 days (worst case, 50 days on average) from a LEO position.

Lengthening the encounter to 180 days (worst-case, 90 days in average) allows a transfer to any Molniya satellite to be accomplished with a ΔV of 1,900 m/S, if the Paxsat is parked in a 2,950 km by 400 km orbit inclined 63° from a LEO position.

As was determined in the drift orbit eccentricity analysis of section 5.5.2.4, an optimal drift strategy for the highly eccentric Molniya orbits is not intuitively evident, although it appears that a strategy should be possible which would give accessibility to Molniya satellites for similar penalties as seen in the LEO case.

5.5.4.2 Circular Orbits

The drift rate of a 63° inclined semi-synchronous circular orbit is approximately 0.031°/day. This means that a Paxsat could start in LEO in a 400 km by 2,950 km orbit and with 2,850 m/S of velocity increment would intercept any semi-synchronous satellite within 180 days (worst-case, 90 days average).

The selection of the appropriate drift orbit for semi-synchronous operations depends heavily on the performance of the launch vehicle, which must be factored into the net fuel usage equations.

5.5.5 Delta-V Requirements for Synchronous (GEO) Operations

Whereas for LEO operations the requirement for maneuvering is driven by the need to match Paxsat and target planes, for GEO operation, the planes of most of the target satellites are coincident. However, getting to GEO requires far more energy from the launch vehicle than getting to LEO, so that a large proportion of the Paxsat fuel supply will have to supplement the launcher.

5.5.5 Delta-V Requirements for Synchronous (GEO) Operations (Continued)

For an Ariane IV launch, a further 1,500 m/s or so are required to get from the transfer orbit (GTO) into which the launcher puts Paxsat to the synchronous orbit in which the targets are.

In addition, most satellites in GEO carry 300 m/S to 400 m/S of maneuvering capability to counteract natural luni-solar gravitational perturbations which tend to change the inclination and, if the inclination is not zero, the right ascension of the orbit plane.

As for fuel required to perform a rendezvous, this would be negligible if Paxsat is already in the target plane and no more than 200 m/S or so if the target plane is slightly inclined.

5.5.6 Homing Strategies

The subject of autonomous rendezvous received intense study in the late 1950's and early 1960's. As the Apollo program trailed off and as the civilian space program declined in general, advances in homing and rendezvous strategy were made largely in the area of homing missiles and other targeted weapons.

Whereas, the previous homing strategies were based largely on proportional navigation, that is on the feedback of range, range rate, azimuth and azimuth rate, to null out the range and range rate and so to effect rendezvous, the modern approach has been to develop and apply the optimal control theory of Kalman to use that same information (range, range rate, etc.) to drive state estimation algorithms which allow a more fuel optimal rendezvous. The relatively large amount of computation this requires, has been made possible by advances in compact digital computers.

More recently, interest has been revived in automatic rendezvous and docking with programs such as the Teleoperator Retrieval System (TRS) which was to have boosted Skylab to a higher orbit, assembly of large space structures, manned and unmanned space platforms, STS, and retrieval of used satellites from orbit. Interestingly, although previous work has not attached importance to a station-keeping phase in which the chase

5.5.6 Homing Strategies (Continued)

vehicle observes the target from a distance, recent work has tended to include such a phase for the purpose of identifying the condition of the target (its spin rate, axis, etc.) and so is more directly applicable to the Paxsat mission scenario.

For the Paxsat mission scenario, it may not be immediately clear that any homing scheme at all need be employed. For example, if Paxsat is to take up station 50 or 60 km away from the target and perform all observations from there, then it is conceivable that all requisite maneuvers can be commanded by ground control. In this case, the satellite need not carry heavy and power-consuming radar.

There are several reasons why an on-board radar is desirable, they are as follows:

(a) The object being investigated may perform a routine maneuver which creates a relative velocity large enough to cause a collision with Paxsat before the satellites are within reach of ground control. This depends on the ground station network keeping track of the encounter.

(b) Natural perturbations along with some target maneuvering may make it very difficult to acquire the target in the fields of view of the Paxsat sensors.

(c) The radar itself can provide data on the motion of the target, and so can enhance the total amount of data gathered.

(d) With radar or some other proximity sensor, Paxsat may autonomously maneuver very close to the target (to within 1 or 2 km or less) and so enable the most thorough investigation practical in space using an unmanned vehicle.

If these arguments are powerful enough to indicate that a radar should be carried, then the problem must be addressed as to how it would be best used. The details of such a trade-off were not performed, but the following scenario was baselined.

5.5.6 Homing Strategies (Continued)

(a) The Paxsat is maneuvered by ground control to a region 50 km to 100 km from the target, when the on-board radar is activated and commanded to acquire the target.

(b) The radar is used to perform two functions:

i) to navigate relative to the target
ii) to point the on-board optical sensor towards the target.

(c) The radar is optimized to provide range and range rate data as a first priority, and then angular information as a second priority. Above all, the radar must be able to provide the target acceleration information required to both characterize the target mass and to guard against collisions due to target maneuvering.

(d) The radar must provide enough angular information to steer the optical sensor to acquire the target in its field of view. Then, higher accuracy information can be derived from the optical sensor, if the radar performance is not adequate in this respect.

No selection of homing laws was made. Instead, recognizing the desirability of efficiency and fail-safe operation, a system was suggested in which the primary navigation calculations were performed in an on-board computer using algorithms based on modern optimal control theory and in which back-up navigation is provided through slightly modified proportional navigation laws implemented in relatively more simple hardwired logic.

This system allows Paxsat to remain in an operating condition if the main computer develops an error and ensures a measure of graceful degradation in performance for failures in either the radar or the optical sensor.

6.0 PAYLOAD ELEMENTS OF THE PAXSAT SPACECRAFT

6.1 Introduction

The ability of Paxsat A to determine the exact function of an unknown spacecraft injected into space is based on the fact that a high degree of individual characterization and optimization is inherent in the design of all spacecraft in their orbital parameters and in the nature of signals to and from the spacecraft.

Clearly, to the extent that form follows function, visual images of the spacecraft are highly determinate of its function and its purpose in space. From high resolution data, particularly in respect to appendages and apertures, a skilled interpreter can provide data regarding the presence and magnitude of on-board propulsion capabilities, the presence and approximate capabilities for optical and/or infrared imaging as well as data regarding the generation of electrical power. Next generation weapon systems employing plasma or laser beams are likely to be even more distinctive in configuration. In the present conceptual payload for the Paxsat mission, emphasis has been given to only optical sensing capabilities, although images in the thermal infrared region would also be very useful for deriving data regarding the energy balance and energy utilization of the unknown spacecraft.

The operation of almost any type of spacecraft involves substantial communications to and generally from the spacecraft. The nature of these transmissions, particularly the frequency bands of operation, radiated power and operational cycles, are of high diagnostic value. These data, when combined with the visual image data in respect to antenna apertures and power available are deterministic of communication and remote sensing missions. Thus the second Paxsat payload is given to electromagnetic wave analysis with the ability to determine the basic parameters of all radiated emissions from the unknown spacecraft.

6.2 Optical Payload

6.2.1 Background, Optical Observation of Satellites from Earth

With the launch of the first artificial earth satellite, the Soviet Sputnik, routine observation of satellites from earth began. Within days of the launch, the USAF obtained photographs of Sputnik using 24 inch aperture telescopes with 5 foot detail resolution. (The aperture size of a telescope limits its resolving power. A basic treatment of resolving power and Modulation Transfer Function (MTF) of a telescope is given in Appendix C.) The size and orientation changes (tumbling rate) of the satellite were quickly deduced with reasonable accuracy. The 24-inch cameras were located at the Air Force Missile Test Center at Cape Canaveral, Florida, a region of only average visibility. To overcome atmospheric effects, a new 48 inch telescope was built at Cloudcroft, New Mexico. A second and similar unit was built for the Advanced Research Projects Agency, ARPA, at the Mount Haleakala facility in Hawaii. The 48-inch instruments were planned to obtain images to detect size and shape, and to record dynamic photometric properties and tracks of satellites. At the time of the Sputnik launch, in a companion non-military development, the Smithsonian Astrophysical Observatory was undertaking a satellite tracking program using a new telescope design by J.C. Baker and Joseph Nunn. This undertaking was to be part of the 1957 International Geophysical year (IGY). Tracking by the Baker-Nunn telescope/camera was to be done on the first satellite scheduled to be launched by the United States in 1957. In the course of time, a large number of Baker-Nunn tracking telescope cameras were deployed around the globe for use by civilian and military agencies.

The quality of the images in the late 50's and 60's was limited by atmospheric turbulence, film speed and residual hardware aberrations in the optical components of the telescope. In the middle 60's with the availability of fast computers with large memories and new techniques in digital signal processing, reports of experiments to correct for atmospheric turbulence appeared in the literature. The work of Harris [19] and McGlamery [20] is noteworthy in this regard. Both workers report impressive results for restoring images blurred by motion or corrupted by noise or atmospheric

6.2.1 Background, Optical Observation of Satellites From Earth (Continued)

turbulence. In a milestone paper in 1970, Labeyrie [21] described the technique which has become commonly known as speckle interferometry. Labeyrie suggested that multiple short-time exposures of a bright object (m=7) from a large telescope (a few meters) could be processed to yield resolution approaching 2/100 arc/S if the signal-to-noise ratio was high - a 100X improvement over conventional 'seeing' from earth. The technique would only apply to objects with a center of symmetry, for example a double star. Labeyrie's work was followed by that of Knox and Thompson [22] and others for example Sherman and Abdelmalek [23 and 24], in which additional phase information was recovered from the images. New numerical techniques were employed resulting in the technique being extended to irregularly shaped objects, and restoration of images from instruments with non-ideal spatial resolution. In simulations, a 'restored' image of the original object, comparable in resolution to the diffraction-limited performance of a large telescope has been demonstrated. Successive efforts during the past decade have led to the assumption that given sufficient motivation, most problems in conventional space object imagery can be overcome, if it is accepted that noise can always prevent a complete restoration [25].

Recent work by astronomers [26] indicates that with proper alignment, clusters of telescopes can be deployed to emulate the diffraction, limited performance of an instrument compatible in size to the effective aperture of the entire cluster.

The present outlook for successfully observing a satellite in an earth telescope is, therefore, to approach the theoretical limitations of the instrument itself if the optical signal is very strong with respect to the instrument noise. The effects of atmospheric turbulence can be processed out of the image. With very sophisticated image processing, image restoration beyond the diffraction limit now appears feasible with quantitative improvements in the order of two to ten times being possible.

6.2.1 Background, Optical Observation of Satellites From Earth (Continued)

In addition to the degradation of satellite images by atmospheric turbulence, three other limiting factors apply to successful imaging of satellites in earth telescopes:

(a) Satellites in low earth orbit often move into the earth's shadow during the best viewing time, i.e. after dark so they are no longer visible.

(b) Thin clouds or fog will block completely a passive optical system such as a telescope.

(c) Some satellites, because of their orbits, will never come into the viewing cone of a particular terrestial telescope.

Experiments to track satellites during the day have been successful [27], but imaging is more difficult because of the light scattered from the intervening air mass. This scattered light is a source of noise in the imagery causing degraded performance.

As a general statement, the availability of a ground based telescope to track a particular satellite is less than 30% of the time the satellite is in view, because of the presence of the atmosphere (clouds) and the need for darkness for high quality image tracking.

6.2.2 Earth-based Telescopes in a Paxsat System

In a space weapons verification system, earth based telescopes could serve as a powerful complement to a remote sensing Paxsat satellite which would function as the principal information gathering component of the system. For the case of a satellite in a very low altitude earth orbit (below 300 km) images of such a satellite taken by an earth telescope under nearly ideal seeing conditions with subsequent image restoration processing, could compare with images taken by Paxsat, although Paxsat images would always be superior simply because they are not degraded by the atmosphere in the first place.

6.2.2 Earth-based Telescopes in a Paxsat System (Continued)

Satellite characteristics being sought from the earth based component of an optical remote sensing system would be similar, basically to what is being sought from the space based component. In both cases, the wanted characteristics are obtained by optical imaging, optical tracing and by obtaining a satellite's photometric parameters. To be more precise, information obtained optically pertains to size, shape, surface features such as shutters and windows, appendages, surface texture, status and dynamic photometric characteristics and point dynamics for operations and temperature control of a satellite. The information gathered through imaging must be available in a high resolution format with wide dynamic and spectral range. If high resolution imaging is performed using an electro-optical sensor accurate real-time tracking is also obtained, as is the static and dynamic photometric data.

Knowledge of the maneuverability of a satellite can be a key input to an analysis of a satellite's mission. To assess the maneuverability of a satellite, knowledge of its on-board thruster system is necessary. This implies that the general size, number and location of these thrusters must be available to an interpreter. Obtaining this information would normally be representative of the most demanding requirement of an optical remote sensor, insofar as high resolution is concerned. It is convenient therefore to use the case of thrusters to establish the ideal resolution performance for an earth based (or space based) telescope.

Thrusters range in size from about 10 cm upward, frequently occurring in clusters of three or more at several locations in the satellite. Accordingly, 10 cm has been taken as the smallest detail required for optical imaging of a satellite.

Using Figure C-1 of Appendix C, the minimum size of telescope aperture required to observe 10 cm features of a target at long range can be calculated as follows:

(a) Calculate the viewing angle, , subtended by a 10 cm object at range R by noting that

6.2.2 Earth-based Telescopes in a Paxsat System (Continued)

$$\frac{10\ (cm)}{R}$$

radians where R is expressed in centimeters also. Radians can be converted to the more familiar seconds of arc by noting that 1 radian is approximately equal to 200,000 arc seconds or 57.3°.

(b) For high resolution of the 10 cm object, must equal the diffraction limited beamwidth of the telescope.

(c) From Figure C-1, Appendix C read off the aperture corresponding to .

As an example, for a 10 cm object at 300 km, =0.33 microradians and the corresponding telescope aperture is 1.5 m.

In the light of the foregoing, the scenario for the earth based telescope (or telescopes) component supporting the Paxsat component of a verification system could be as follows:

(a) Telescopes of up to 2 m aperture, single or in clusters of smaller individual apertures 'optically' aligned to emulate larger apertures are required.

(b) Telescopes would employ electro-optical imaging arrays in the image plane to capture 'fast' images for subsequent digital processing.

(c) Sophisticated but conventional image processing techniques are used to restore atmospherically degraded images.

(d) The practical limit on the minimum size of a satellite to be observed in detail is in the order of 1 m, in order that a sufficient number of 'resolution elements' can be projected onto the satellite to obtain a useful data set.

6.2.2 Earth-based Telescopes in a Paxsat System (Continued)

(e) Telescope slew rates up to 1.5° per second, depending on the orbital elements of the satellite are required.

(f) 'Daylight' imaging of satellites is conducted where degraded detail is acceptable.

(g) An effectiveness figure of 30% can be expected for any particular installation.

For the application being considered, the technology required is essentially in place. Over the next decade, further developments in image processing can be expected permitting image restoration beyond the diffraction limit.

6.2.3 Space-to-Space Optical Sensing

In a previous report [1], remote sensing of a satellite by optical imaging was rated as the most informative single source of information about the nature and purpose of a satellite. The effectiveness of an optical image lies in its ability to capture detail on a satellite's overall size, shape and color. High resolution optical images of small hardware items such as attitude and tracking sensors and thrusters provide information on a satellite's ability to maneuver and change orientation. Information on the size, shape, color and texture of appendages leads to knowledge of what these appendages are and in turn, why they are there. When coupled with orbital information and information obtained by an electronic intelligence (ELINT) sensor of the type planned for Paxsat, optical sensed data can be the basis of a reliable estimate of the mission and purpose of a particular satellite.

6.2.3.1 Telescope Primary

High resolution images of small hardware items such as attitude and tracking sensors, thrusters and (macro) texture implies resolution of a few centimeters. When optical wavelength and the maximum stand-off distance for remote sensing are specified the theoretical lower limit on a telescope's primary aperture is set. In short, the larger the aperture the better the resolution

6.2.3.1 Telescope Primary (Continued)

at long range. For an application in space, however, severe limits are placed on the size, weight, shape, power requirements and reliability of an optical instrument. A compromise is therefore necessary between payload performance and payload physical characteristics. The performance compromise agreed for this conceptual study is to have the telescope produce a well resolved image of a white/black checkerboard test pattern at 100 km in yellow light, the dimensions of a white or black pixel in the test pattern being 10 cm on a side, that is, 20 cm in either direction represents a complete 'spatial cycle'. Ten centimeters at 100 km subtends 1 microradian. Taking the wavelength of yellow light to be 500 nm (5000 A^o), and specifying the full width half power diffracted beamwidth to be 10 cm at 100 km, i.e. 1 microradian, the telescope aperture turns out to be 0.5 m. The Modulation Transfer Function (MTF) of the square wave black/white object pattern under these conditions is 0.6 since a white/black cycle of the patterns subtends two optical beamwidths. The MTF falls to zero when the spatial frequency of the pattern doubles. According to the Rayleigh criterion, the system will resolve two point sources 1.22 microradians apart. At 1000 nm, the diffraction-limited performance will be twice as coarse as for the 500 nm case, so the MTF for the specified test pattern is zero.

The specification on the telescope is therefore near-diffraction-limited performance at 500 nm. A primary mirror of this quality is within the state-of-the-art, including the requirement for ruggedness in a spacecraft.

6.2.3.2 Focal Plane Sensor

The image of a 5 m by 5 m satellite would be subtended 50 by 50 microradians in the focal plane of the telescope. The Field of View (FOV) should be at least twice as large.

6.2.3.2 Focal Plane Sensor (Continued)

The image plane sensor of a Paxsat telescope would be an electro-optic array. Charge Coupled Devices (CCD's) are indicated in view of the excellent sensitivity, wide spectral response, wide dynamic range, electrical stability and physical robustness of CCD arrays. Current technology can provide a reliable 200 by 200 pixel array on 27 micron centers, with negligible optical cross coupling. 96 by 2049 arrays have been built on 13 micron centers and are in regular use [28 and 29] but for maximum resolution, a larger pixel size in the order of 30 microns is preferred to assure minimum cross coupling between pixels.

The diffraction-limited optical beam should be subtended in the focal plane by two CCD pixels. Thirty (30) micron pixels dictate an equivalent telescope focal length of approximately 60 m, since a single CCD subtends 0.5 microradians. The optical design is therefore an f/120 system. An f number of 120 is unusually high in terms of conventional photography, but the high sensitivity and low noise characteristic of CCD's can be expected to provide high signal-to-noise, given that satellites normally have an equivalent stellar magnitude of six to ten depending on the sun angle. It has been suggested that in space, the telescope described here might even see another satellite in the earth's light given the proper dark sky viewing conditions.

For optical tracking, it is desirable to increase the field of view of the telescope to several minutes of arc, but with greatly reduced resolution. A relatively coarse annular tracking array surrounding the imaging array and subtending 30 arc minutes would permit handover of radar tracking to optical tracking at the 0.5° field of view point.

6.2.3.3 Telescope Volume

An effective focal length of 60 m can be readily accomplished in a physically short length using a hybrid Cassegrain-catadioptic design approach. The relatively narrow field of view subtended by the central high resolution portion of the electro-optical array eases the design problem considerably. An overall length and diameter of 1.5 m and 1.0 m respectively are suggested as allowances for volume.

6.2.3.4 Telescope Weight

The weight of the Paxsat A telescope could vary from 100 kg to 200 kg depending upon its construction. A paper [30] describes an ultralight weight telescope similar in concept to what Paxsat would carry. The primary reflector for the referenced telescope is 0.5 m in diameter and weighs approximately 10 kg. The entire telescope weighs less than 30 kg. This telescope was designed by long range stand-off photography in military aircraft and is therefore presumed to be of rugged construction.

A weight allowance of 100 kg for optical shielding, heaters, a mounting trunnion, image processing electronics and signal conditioning puts the probable overall weight of the Paxsat instrument in the 100 kg to 200 kg range.

6.2.3.5 Telescope Power Requirements

Power requirements for a Paxsat telescope include power for thermal control heating, image processing electronics and for telescope slewing. A peak power requirement of 200 W is envisaged.

6.2.3.6 Telescope Telemetry Requirements

High (electrical) resolution digital readout of the imaging array sensor is required to assure a high signal to quantization noise ratio for subsequent image processing on earth. A 14-bit digitalization is recommended. Overall telemetry rates are dictated by the repetition rate of CCD array readout and by the size of the target. (It is assumed that only pixels with signal sensibly above the noise level will be read out during imaging, although all pixels would be read out from time to time to monitor pixel dark current and pixel aging effects.) Taking the previous example of a 5 m by 5 m satellite at 100 km, the image will cover 100 by 100 pixels. Assuming 20 bits/pixel to include digital error correction and other incidental digital overhead, a readout of the 100 by 100 pixels represents 200 kbits of data. A modern telemetry system can readily handle a bit rate of several tens of megabits/S so the telemetry system is not limited at the high end, i.e. maximum repetition rate by the time required to integrate charge on the pixels and at the low end by the

6.2.3.6 Telescope Telemetry Requirements (Continued)

maximum permissible blur during charge integration. From current practice, a minimum charge time of a few milliseconds is required to obtain a high signal-to-noise ratio. Maximum allowable integration time with acceptable blur limits is controlled by basic tracking accuracy and the slewing rates required for a particular imaging session. Tracking 'lock' for periods in the order of a second may be practical in some cases. In others, blur may be unavoidable even with a millisecond-order exposure times.

It is assumed that all image processing will be done following ground reception of the raw pixel data.

6.2.3.7 Telescope Image Processing Requirements

Noise-free image data can be processed to remove the effects of stable geometric and electrical aberrations in the optical elements and the electro-optical sensor. Assuming that several images will be taken in rapid succession, and that the successive images will move and/or rotate slightly in focal plane vis-a-vis the particular CCD pixels covered, image restoration to, and in some cases beyond the diffraction limit, will be possible. The series of slightly displaced or slightly rotated images is a highly correlated data set with very high redundancy which can be used to advantage in digital processing.

6.2.3.8 Telescope Summary

The design and manufacture of a telescope for Paxsat (an instrument meeting the functional requirement of wide spectral response and very high resolution in a narrow field of view and within the constraints on power, weight, size and reliability characteristic of a satellite payload) is within the state-of-the-art. Image processing following reception of telemetered data by earth terminals will remove degradation of the image due to residual inaccuracies in the telescope optics and the focal plane array. With very high signal- to-noise images processing, to remove more fundamental diffraction effects may be possible, especially in the case of an imaging session in which a series of fast exposures 'freezing' the image over several points in the image plane are obtained.

6.2.3.8 Telescope Summary (Continued)

The principle characteristics of an optical telescope for Paxsat are summarized in Table 6-1 and 6-2.

6.3 ESM Payload

This section summarizes the results of a study made on available Electronic Support Measures (ESM) technology to be used in a wideband antenna/receiver system which meets Paxsat mission requirements for the detection and measurement of a wide range of electromagnetic (EM) signals.

An antenna/receiver system is described with an estimated DC power consumption of 450 W maximum, and weight of 125 kg including redundancy. It is proposed as a baseline against which trade-offs of flexibility and versatility versus circuit complexity, weight and DC power can be made if required. The rationale for choosing the proposed configuration is given and the expected performance characteristics are summarized.

The conclusions of the study are based on rather sketchy and incomplete data available in the literature because of the military nature of the subject matter. Therefore appropriate caution must be exercised in using the results. References 31 to 47 were consulted for this analysis.

6.3.1 Requirements of the Paxsat EM Antenna/Receiver System

Using military terminology, the types of EM signals of interest can be divided into two categories.

(a) ELINT (Electronic Intelligence) of radar pulsed signals, where pulse widths may vary from 100 nS to 25 uS, and where chip bandwidths up to 20 MHz may be in use.

(b) COMINT (Communications Intelligence) of communications signals, where AM, FM or PM, analog (CW) or digital pulsed types of carrier modulation may be in use, with channel bandwidths down to 25 kHz.

TABLE 6-1 FUNCTIONAL SPECIFICATIONS FOR A TELESCOPE ON PAXSAT A

1. High resolution at 100 km range of a test pattern of alternating black and white squares having a spatial wavelength (black-to-black) of 20 cm.

2. High speed framing, in the order of ten frames per second.

3. Spectral response to cover the visible and near infrared spectrum to 1000 nm. Spectral band selection is desirable.

4. Useable as the optical sensor in a target tracking loop.

5. Compatibility with the physical and electrical constraints of Paxsat A.

TABLE 6-2 PHYSICAL AND ELECTRICAL SPECIFICATIONS FOR A TELESCOPE ON PAXSAT A

Primary Reflector Diameter	0.5 m
Diffraction Limited Performance	500 nm
Equivalent Focal Length	60 m
200 by 200 CCD array subtending 100 microradians for high resolution imaging, surrounded by a low resolution array with a 0.5° FOV for tracking	
Instrument Shape	Generally Cylindricalr
Overall Cylindrical Dimensions	1.0 m Diameter x 1.5 m Length
Weight	200 kg maximum
Primary Power Requirement	200 W maximum
Image Telemetry Requirements	2 Mbits/S typical

6.3.1 Requirements of the Paxsat EM Antenna/Receiver System (Continued)

The EM environment may consist of multiple, but not too many, simultaneous or time-overlapping signals, with either interlocked or random timing and falling within an assumed frequency range from 0.35 GHz to 40 GHz. The receiver must therefore be capable of operating with multiple simultaneous signals within this frequency range and must have good dynamic range, say greater than 50 dB, achieved with low noise figure and high linearity/low intermodulation performance.

The signals may be continuous or pulsed with a repetition rate ranging from very high to a very low rate approaching a monopulse; they may be single-frequency, or spread spectrum using chirp, or frequency hopping techniques, etc. The receiver must have a high probability of intercept (POI) for all of these signals. High speed acquisition and measurement, while not as critical as it would be in a hostile military environment, are important so that the Paxsat surveillance mission can be completed in a timely manner to conserve fuel and DC power.

The receiver should be able to analyze the signals to measure or identify power levels, operating frequencies and bandwidths, types and characteristics of modulation being used, and Time of Arrival (TOA).

6.3.2 Available Receiver Approaches

A number of possible receiver techniques will be described and evaluated for size and weight, DC power, technical maturity for satellite applications, hardware complexity and for the following performance characteristics:

(a) Types of signals that can be processed; Probability of Intercept (POI)

(b) Operation with multiple simultaneous signals.

(c) Speed of signal acquisition

(d) Sensitivity

(e) Dynamic range

6.3.2 Available Receiver Approaches (Continued)

(f) Upper operating frequency limit, compared to the 40 GHz required

(g) Instantaneous bandwidth

(h) Frequency resolution

(i) Time of Arrival (TOA) resolution

The limitations of each technique on a stand-alone basis will be noted and hybrid configurations will be described that overcome some of these limitations by using different combinations of techniques.

6.3.2.1 Crystal Video Receiver (CVR)

A block diagram of a CVR using conventional RF to video detection is shown in Figure 6-1. This is the simplest and technically most mature approach and one that potentially gives instantaneous frequency coverage up to 40 GHz and beyond.

This technique is normally used only in a low duty cycle pulsed signal environment. In its simplest form without input filtering, the CVR provides no frequency information. With multiple simultaneous or time-overlapping pulse signals, the amplitude data provided by the CVR becomes distorted. Under such circumstances, the CVR is then used only to provide a signal presence indication or warning.

6.3.2.2 Instantaneous Frequency Measurement (IFM)

The IFM is essentially an instantaneous frequency measuring discriminator circuit. The most common implementation is the delay line discrimination shown in Figure 6-2(a). The signal is split into two paths, one having a known delay with respect to the other. Signals in the delayed path are shifted in phase with respect to the undelayed path as a function of the signal frequency. A phase detector at the output of the two lines yields a voltage proportional to frequency.

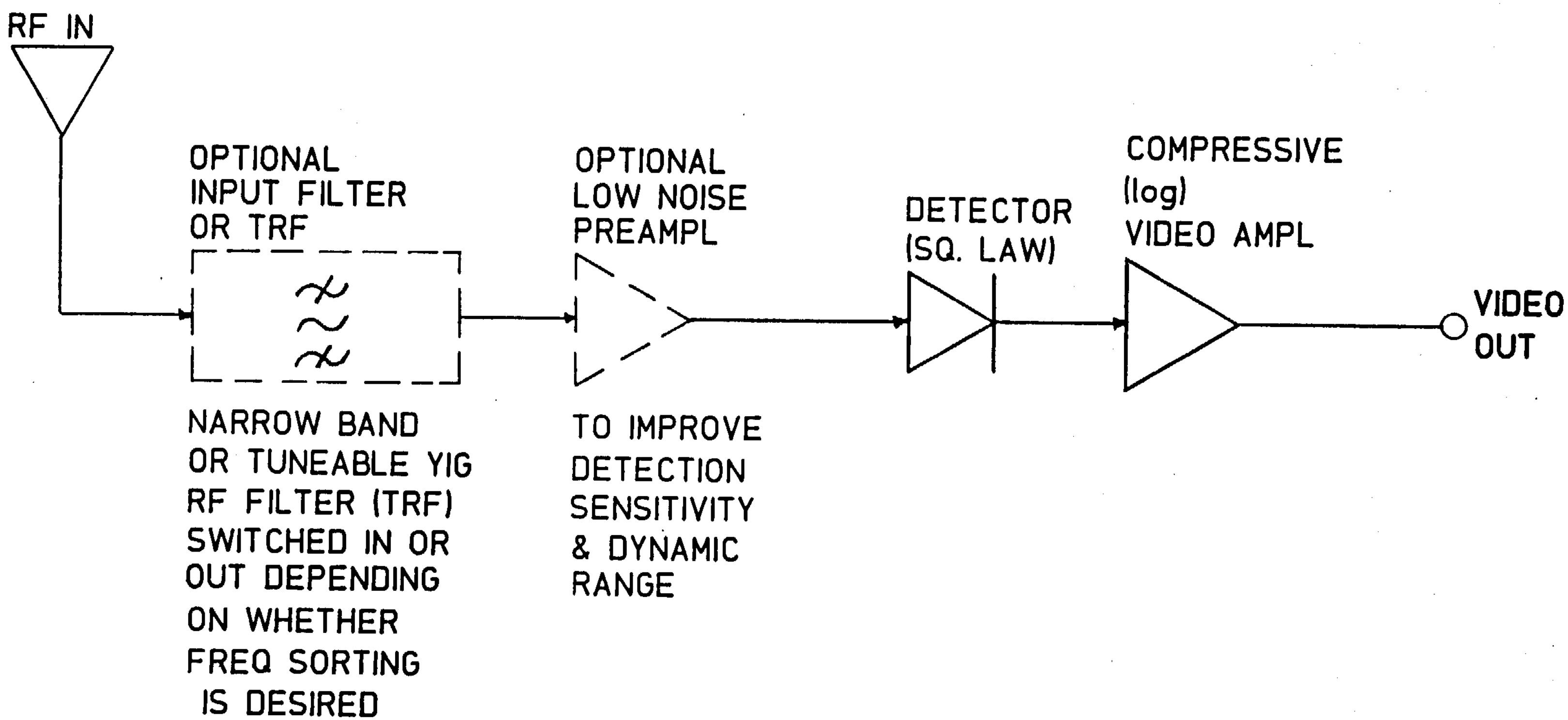


FIGURE 6-1: CRYSTAL VIDEO RECEIVER (CVR) SIMPLIFIED BLOCK DIAGRAM

6.3.2.2 Instantaneous Frequency Measurement (IFM) (Continued)

The delay lines are implemented as microwave delay lines or as meander lines which have wide open frequency response, or at lower frequencies, by using non-dispensive SAW delay line filters that have controlled bandwidths, and are normally placed in each arm of the discriminator, as shown in the insert of Figure 6-2(a).

Another IFM realization is shown in Figure 6-2(b). By adding circuitry at the delay line outputs, it is possible to generate two orthogonal signals: A sin and A cos , where the phase angle is a function of the input signal frequency, and the data is in a form that can be conveniently digitized and processed. Alternatively, the sum of the signals in the two paths can be taken at IF, followed by video detection with a single output or two orthogonal outputs as shown in Figures 6-2(c) and 6-2(d) respectively. These configurations provide a coarser frequency measurement but increase the dynamic range.

For all the circuits in Figure 6-2, the delay line IFM has an unambiguous frequency range, BW $\approx$1/2T, and the frequency resolution is equal to a fraction of 1/2T. The inverse relationship to T places a limit on the achievable resolution because there are upper limits on the value of T that can be used. First, T must be small compared to the period of any message modulations on the signal and secondly, the delay must be less than one-half the pulse width for a pulsed signal, to ensure that the delayed and undelayed pulses can be correctly compared. For example, with a signal pulsewidth tp of 100 nS, the maximum differential delay T, that can be used is 50 nS. 1/T equals 20 MHz and the unambiguous frequency range is 10 MHz. The frequency resolution with a 4-bit code dividing the coverage range into 16 parts, would be 10/16 or 0.625 MHz, and with a 7-bit code would be approximately 80 kHz. As the delay is reduced, the bandwidth increases up to the gigahertz region subject to bandwidth limitations of the delay lines being used and to the bandwidth/phase stability limitations of the summing circuits, etc.

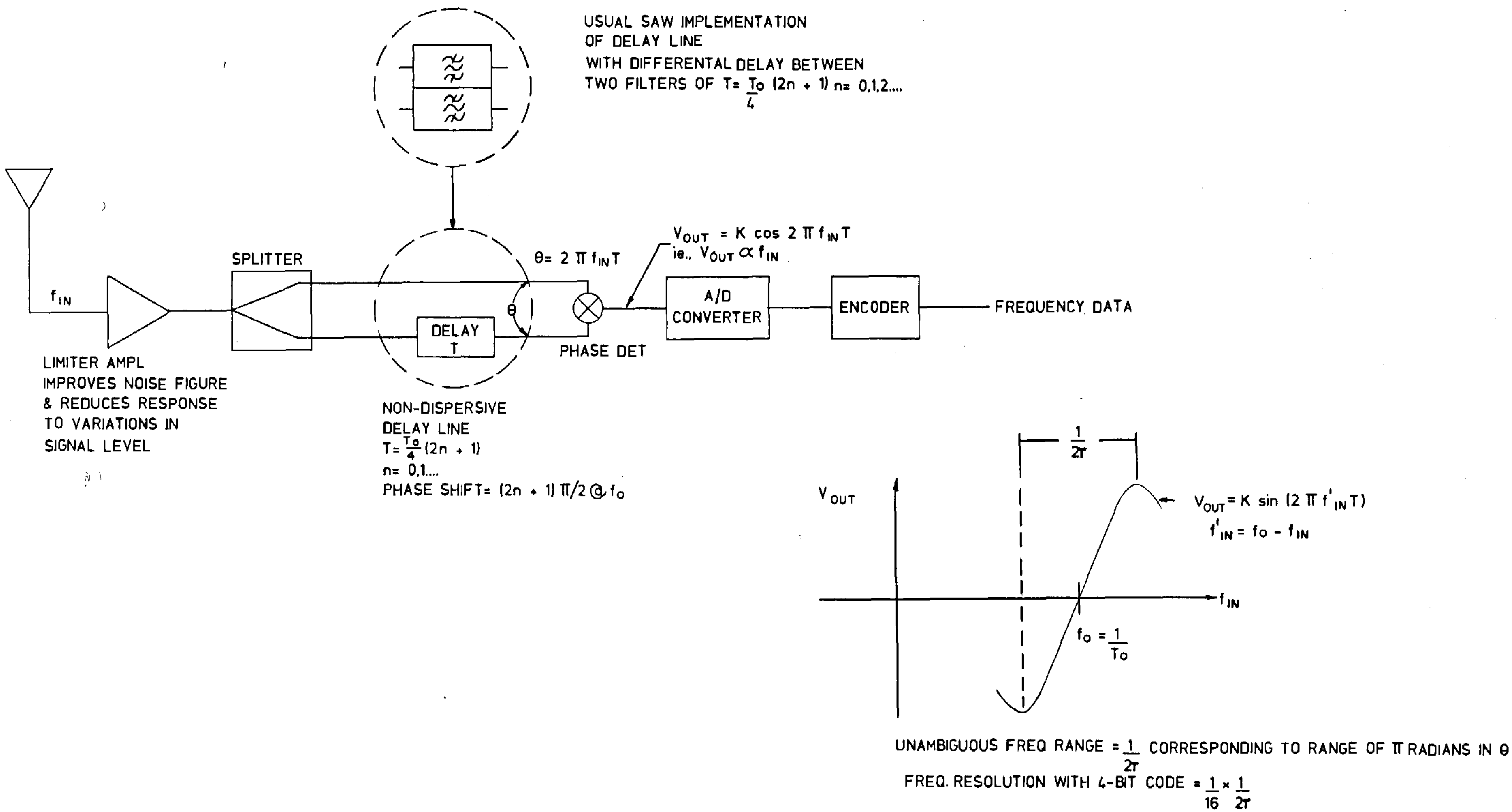


FIGURE 6-2(a): IFM WITH DELAY LINE DISCRIMINATOR USING PHASE DETECTOR, SIMPLIFIED BLOCK DIAGRAM.

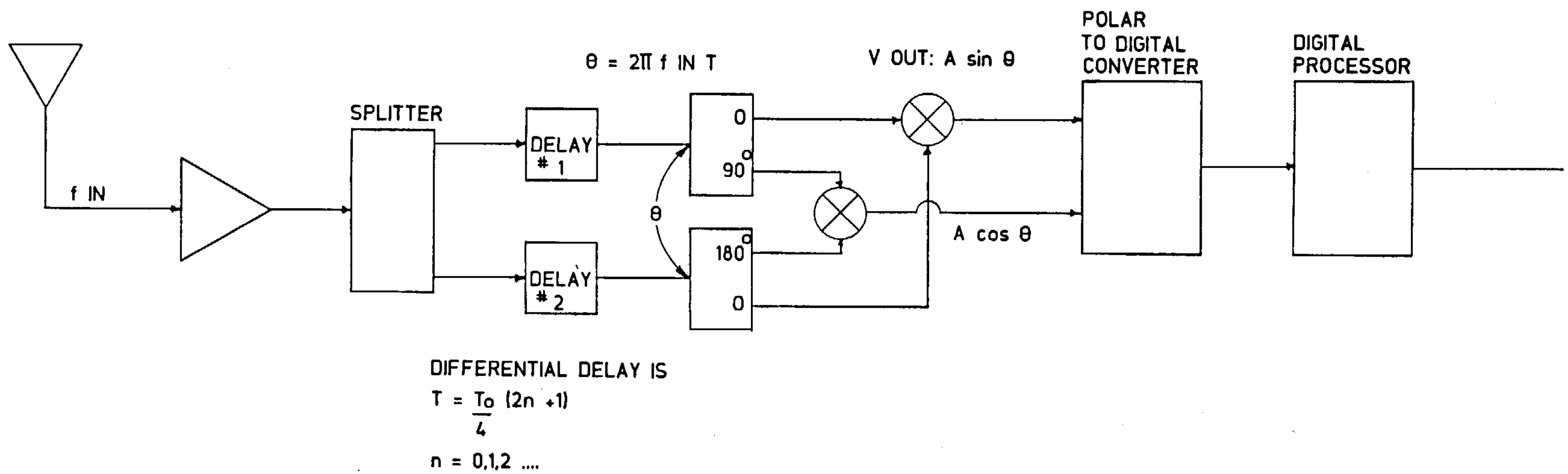


FIGURE 6-2(b): IFM WITH DELAY LINE DISCRIMINATOR-PHASE DETECTOR AND ORTHOGONAL OUTPUTS, SIMPLIFIED BLOCK DIAGRAM.

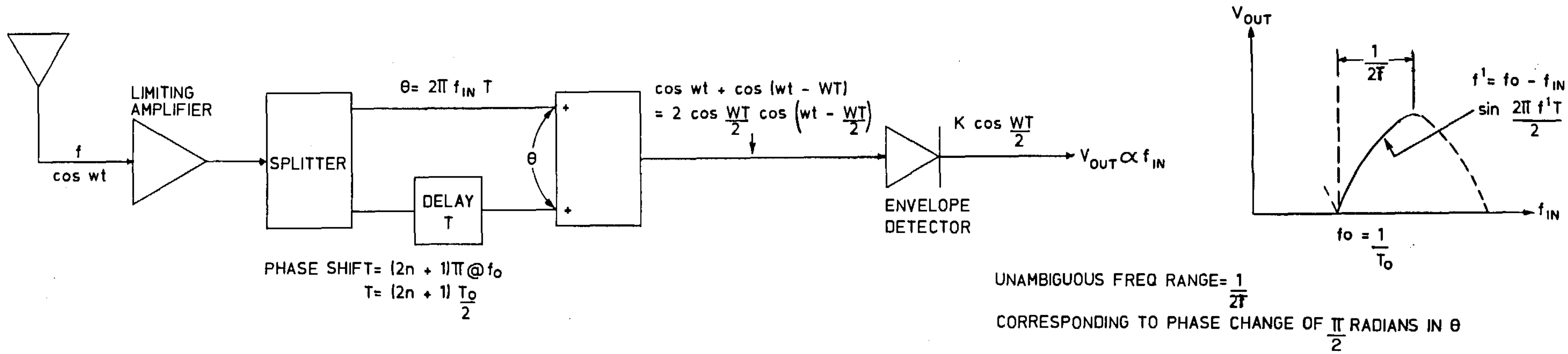


FIGURE 6-2(c) IFM WITH DELAY LINE DISCRIMINATOR USING ENVELOPE DETECTION

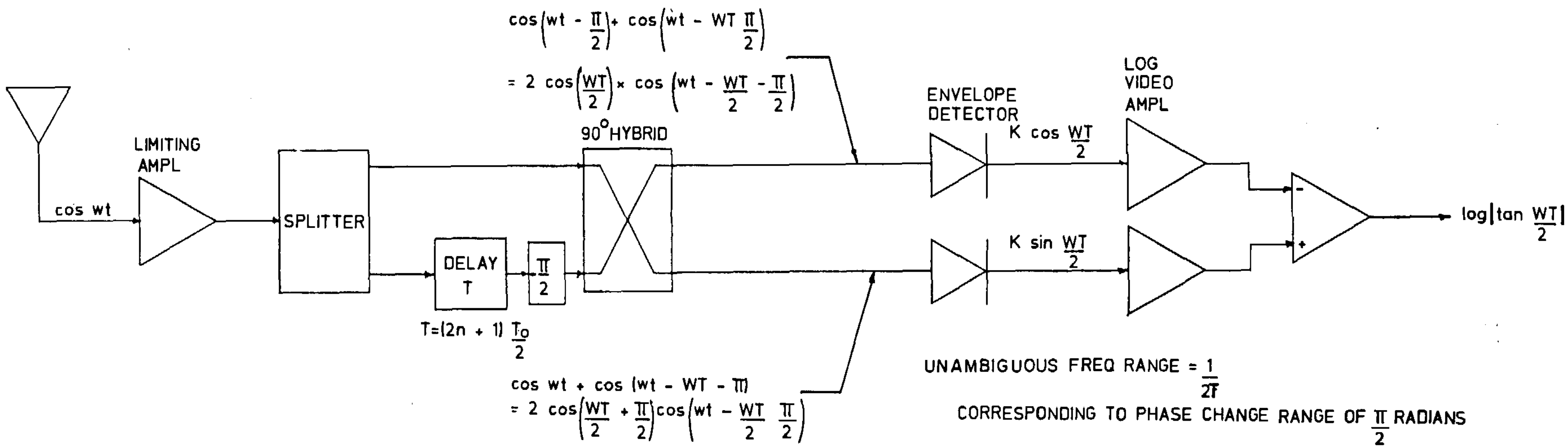


FIGURE 6-2(d): IFM WITH DELAY LINE DISCRIMINATOR USING ENVELOPE DETECTION AND GENERATING ORTHOGONAL OUTPUTS.

6.3.2.2 Instantaneous Frequency Measurement (IFM)(Continued)

For broadband applications, it is possible to increase the bandwidth and achieve a high degree of frequency resolution and accuracy (0.1% or better) by overcoming the periodicity ambiguity of the discriminator through the use of multiple parallel delay line discrimination, each with a different delay. This is illustrated in Figure 6-2(e). With a total of say, 4 parallel delay lines with delays of 50, 25, 12.5 and 6.25 nS in the previous example, the bandwidth would be increased by a factor of 8 to 80 MHz. With a 4-bit code, the frequency resolution would be 5 MHz, with a 7-bit code, it would be 0.625 MHz and with a 10-bit code, it would be approximately 80 kHz or 0.1%. Unfortunately, it is often not practical to use this paralleled approach because it demands very tight control on the delay lengths over all environmental conditions.

Another realization of the discriminator characteristic that provides fine frequency resolution even for very narrow-pulses, and provides faster reaction time by eliminating the delay lines per se, is shown in Figure 6-2(f). It consists of two filters, one tuned above and one tuned below the operating band center frequency, followed by subtraction of the two detected signals.

A serious disadvantage shared by all the different IFM configurations is their poor response to multiple simultaneous signals since there can be only one discriminator output. Hence, only the strongest signal is measured, or there are erroneous results in the case of equal or nearly equal level (less than 4 dB difference) signals. The situation may be improved to some degree if only one of the signal is to be measured, by narrowbanding the IFM. In practice, bandwidths down to 0.5% are realizable with SAW delay lines. Another approach to the problem is to use a simultaneous signal detector. It indicates the presence of simultaneous signals and by appropriately setting the detector threshold, it can be made to inhibit the IFM output when there is less than 4 dB difference between signals and false frequency readouts can occur. Finally in some cases, IFM frequency identification of multiple signals can in fact be implemented with a tuneable CW reject filter commanded to sweep the frequency range of interest. At the instant the filter rejects one of the signals, the simultaneous signal indicator turns off and

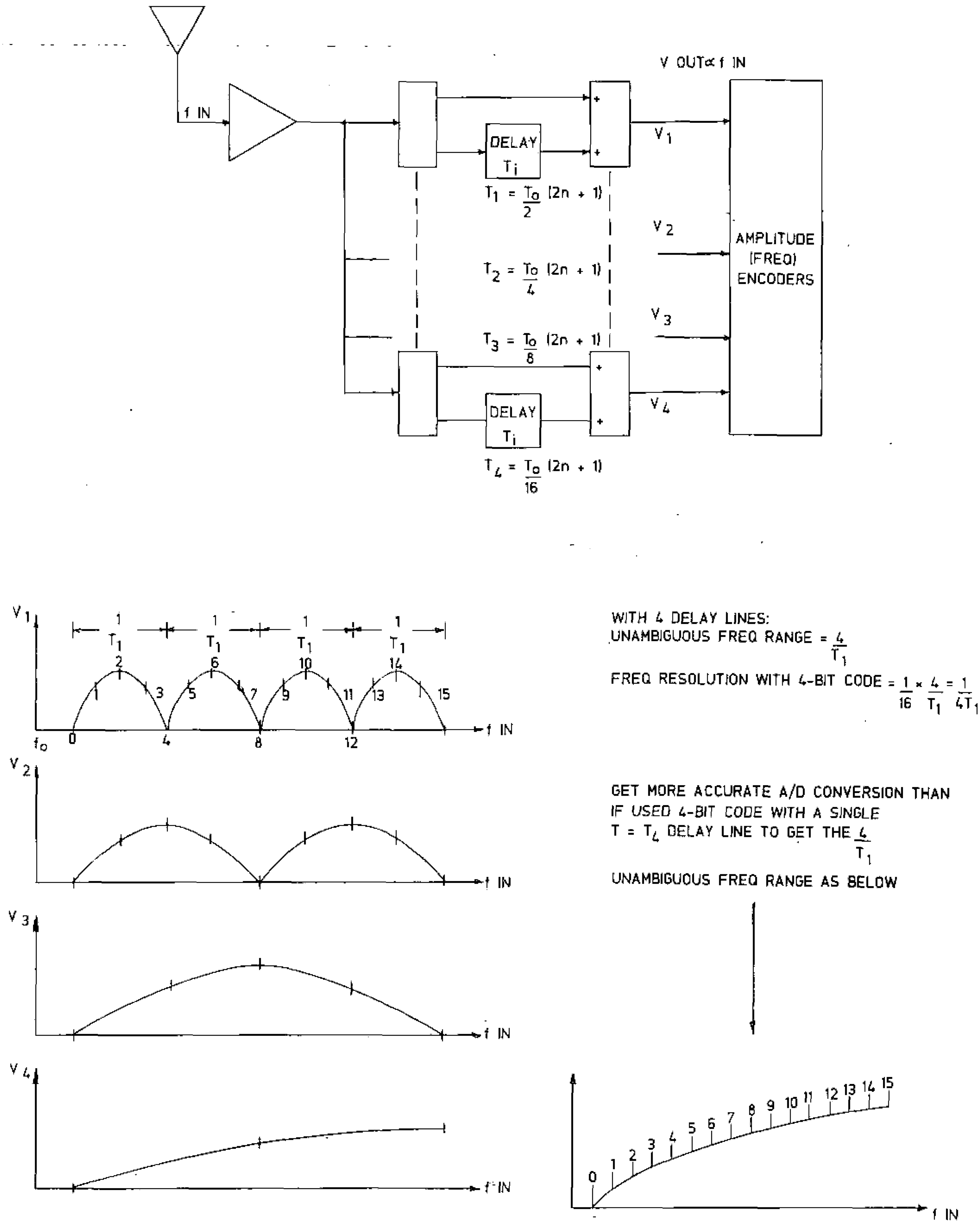


FIGURE 6-2(e): IFM, EXAMPLE OF FOUR PARALLELED DELAY LINES TO RESOLVE FREQUENCY AMBIGUITY.

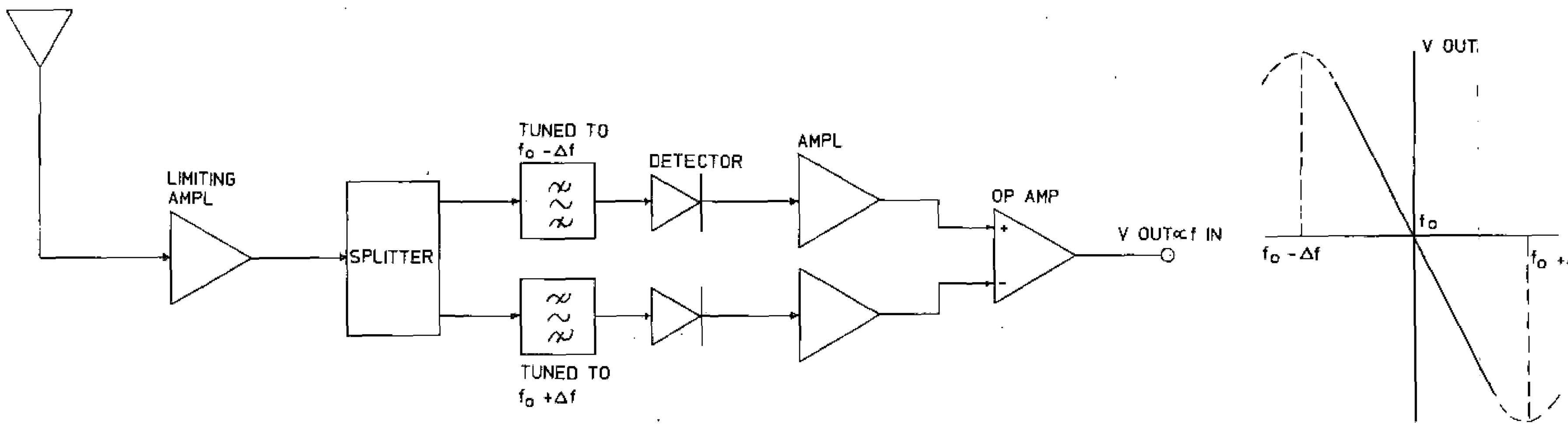


FIGURE 6-2(f): IFM USING TWO FILTER DISCRIMINATOR, SIMPLIFIED BLOCK DIAGRAM.

6.3.2.2 Instantaneous Frequency Measurement (IFM) (Continued)

the IFM gives a true reading of the unrejected signal. At the same time, knowledge of the frequency of the some degree if only one of the signals is to be measured, by narrowbanding the IFM. In practice, bandwidths down to 0.5% are realizable with SAW delay lines. Another approach to the problem is to use a simultaneous signal detector. It indicates the presence of simultaneous signals and by appropriately setting the detector threshold, it can be made to inhibit the IFM output when there is less than 4 dB difference between signals and false frequency readouts can occur. Finally in some cases, IFM frequency identification of multiple signals can in fact be implemented with a tuneable CW reject filter commanded to sweep the frequency range of interest. At the instant the filter rejects one of the signals, the simultaneous signal indicator turns off and the IFM gives a true reading of the unrejected signal. At the same time, knowledge of the frequency of the tuneable reject filter indicates the frequency of the other signal. The CW reject filter can also be used to detect and automatically notch out any undesirable signal.

6.3.2.3 Superheterodyne Receivers

The superhet is a mature and well-developed technology. Four variations are shown in Figure 6-3: the fixed wideband superhet, the scanning superhet, the scanning superhet with Tuneable RF (TRF) preselection and the set-on narrowband superhet.

The fixed superhet shown in Figure 6-3(a) provides instantaneous translation of all input signals that, after translation, fall within the IF passband B_{IF}. Generally, with wideband mixers available up to 40 GHz, it is the IF amplifier-filter that determines the instantaneous frequency coverage, and large bandwidths up to 10 GHz are possible. However, the wideband superhet does not provide much frequency information and is therefore used mainly as an input to some other receiver configuration to translate a high RF band to a common lower IF baseband and so increase the frequency coverage.

The scanning superhets shown in Figure 6-3(b) without preselection, and in Figure 6-3(c) with a tuneable

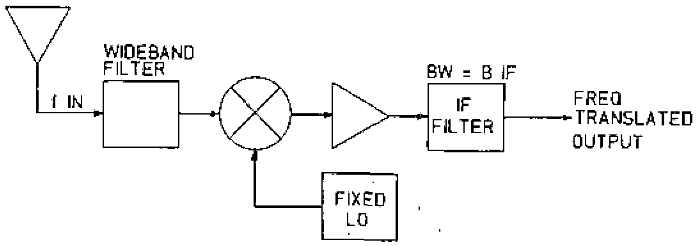


FIGURE 6.3(a): FIXED TUNED WIDEBAND SUPERHET

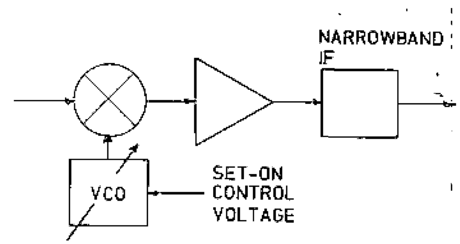


FIGURE 6.3(e): NARROWBAND SET-ON SUPERHET

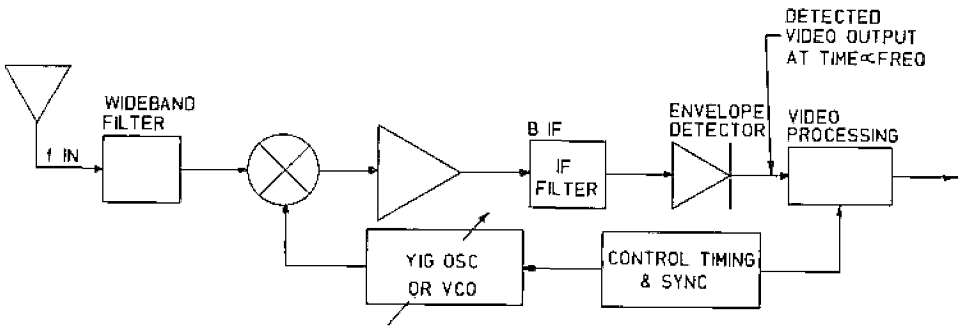


FIGURE 6.3(b): SCANNING SUPERHET

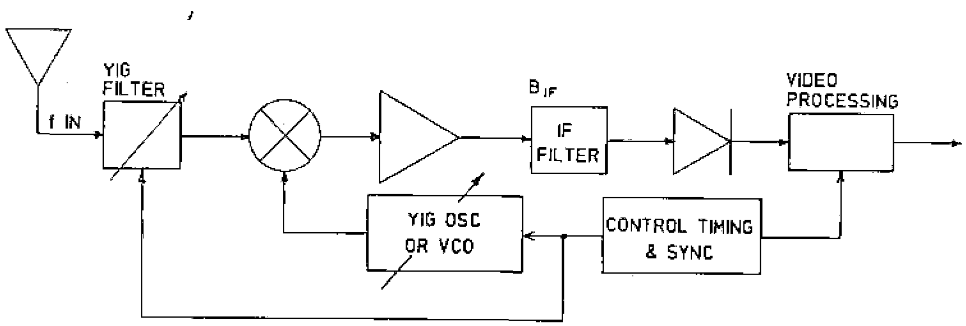


FIGURE 6.3(c): SCANNING SUPERHET WITH TRF PRESELECTOR

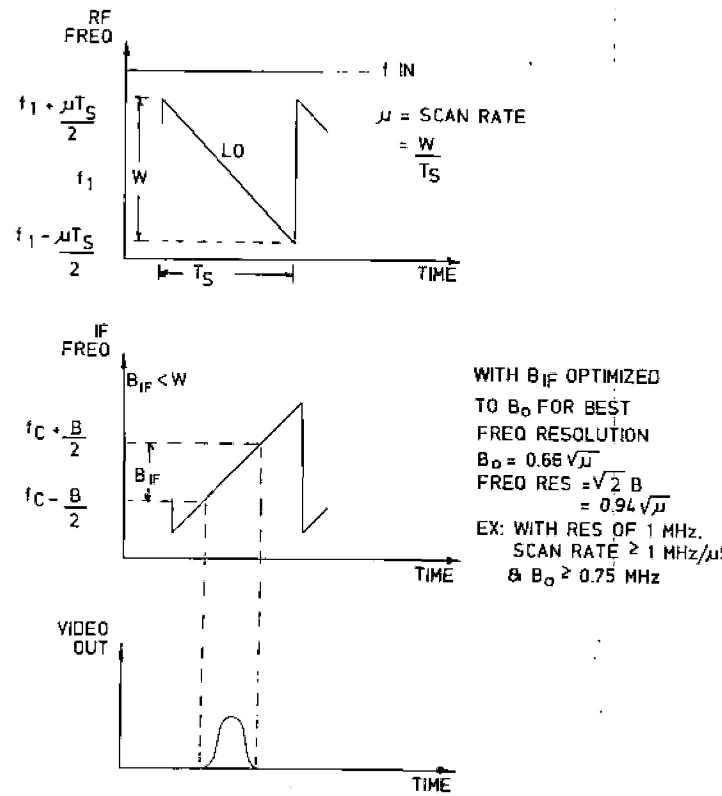


FIGURE 6.3(d): SUPERHET SCANNING TIME DIAGRAMS

6.3.2.3 Superheterodyne Receivers (Continued)

preselector, do provide frequency information as a function of time after the start of the scan. Time frequency diagrams are shown in Figure 6-3(d). The optimized frequency resolution is $\sqrt{2}\ B_{IF}$. However, the scanning process may lead to a low probability of intercept (POI) for low duty cycle or monopulse signals or for frequency hopping signals because of the time interval between re-visits to any specific frequency in the scan. The POI can be improved by using a faster scan rate - up to 100 MHz/mS for YIG preselectors and 100 MHz/0.1 uS for wideband varactor VCO's are typical - but this may require a wider IF bandwidth since for proper detection the required B_{IF} must be proportionally increased as the square root of the scan rate. If B_{IF} is increased, there will be a degradation of frequency resolution.

With any superhet, there is a potential problem with multiple simultaneous input signals, particularly high level ones, due to the danger of spurious mixer products. This danger may be reduced by using a synchronously tuned RF preselector, normally a YIG filter, ahead of the mixer as shown in Figure 6-3(c). YIG filters are tuneable over an octave bandwidth or over a full waveguide band at higher frequencies. They have an instantaneous bandwidth of up to 50 MHz. This narrowbanding is desireable for reducing spurious outputs in a multiple signal environment, but becomes a limiting factor for wideband signal applications.

One important derivative of the scanning superhet that eliminates the sometimes harsh trade-off between scan rate versus frequency resolution versus POI, is the narrowband superhet shown in Figure 6-3(e). It contains a VCO and a narrowband IF filter and may or may not use a tuneable YIG preselector. The VCO (and YIG if used) is fix-tuned by some external control to set-on the superhet to receive a specific active part of the frequency coverage band.

6.3.2.4 Compressive (Microscan) Receiver

The microscan receiver and its operation are described by the diagrams in Figure 6-4. The microscan is essentially a scanning superhet except that it looks at the entire frequency coverage range during all of its

6.3.2.4 Compressive (Microscan) Receiver (Continued)

operating aperture time T. It is implemented by replacing the superhet IF filter with a chirp dispersivedelay line filter, referred to as the convolver, that has a time delay versus frequency characteristics, i.e. time delay dispersion, that equals the aperture time and that is matched to the local oscillator or multiplier scan rate $u = B/T$. This causes a CW or wide pulse (i.e., $tp > T$) signal that falls within the passband of the delay line for a duration of T seconds, to be time compressed by a factor N called the compression ratio where N=BT and is typically equal to 500 to 1000. For a fixed frequency input, the signal at the delay line output is a narrow sin x/x - shaped
pulse, with a mainlobe duration of

$$\frac{T}{BT} = \frac{1}{B} = \frac{1}{uT}$$

and with time sidelobes ,that can be calculated by performing an inverse Fourier transform on the truncated rectangular-like frequency spectrum of the output signal.

The time at which the delayed output pulse occurs after the end of the local oscillator sweep, depends on frequency, and the time scale at the output can be converted to frequency by multiplying B/T. The resulting frequency spectrum after conversion, has a sin x/x envelope that would result from taking the Fourier transform of a pulse width T. The frequency resolution is

$$\frac{1}{T} = \frac{u}{B} = \frac{B}{N}$$

This represents an improvement by a factor of $\sqrt{2}$ N compared to the scanning superhet with an IF bandwidth of B. In practice, some form of weighting is used with the delay line to suppress the sidelobe associated with the sin x/x output. This causes some spreading of the output pulse width, which tends to degrade the achieved frequency resolution by a factor of approximately 1.5. Nevertheless, good resolution is achieved even with high scan rates. In summary, the compressive receiver looks

6.3.2.4 Compressive (Microscan) Receiver (Continued)

at its complete coverage band B all during its aperture period T, and has a resolution 1/T and sensitivity that are equivalent to a superhet with an IF bandwidth of B/N.

Another degrading factor on the output pulse sidelobes, are non-linearities in the local oscillator and delay chirp characteristics and results in degraded receiver dynamic range for signals closely spaced in frequency.

Normally, for COMINT applications with narrowband requirements, the delay line chirp time is made shorter than the local oscillator sweep time as assumed in Figure 6-4. This gives superior frequency resolution and makes it easier to implement sin x/x sidelobe suppression weighting.

The chirp lines are normally realized using surface acoustic wave (SAW) technology or IMCON (reflection mode delay lines fabricated on steel acoustic media) technology. The difficulty of producing these devices with suitable sidelobe suppression weighting is one limitation on the instantaneous frequency band coverage. Another, often more severe limitation, can arise from high speed and power demands on A/D converters and other digital interface circuitry required for serial readout of frequency data. For example, a readout of five 8-bit encoded frequencies in a 10 uS aperture time requires a speed of at least 40/10 = 4 MHz.

Because of the effective 50% duty cycle operation of the receiver, the aperture time must be less than one-half the pulsewidth of the narrowest pulse of interest, and/or less than one-half the period between frequency changes of a frequency-hopping or MSFK signal, to guarantee 100% probability of intercept and undegraded sensitivity. Otherwise, two inter-leaved receivers must be used such that one receiver is sampling the RF input while the other is outputting the spectrum. Furthermore, any time the aperture time exceeds the input pulse duration, the output sin x/x spectrum is dictated by the actual input pulse width, rather than by T, and the peak amplitude is reduced at a rate of 20 dB/decade of input pulsewidth reduction, from that obtained from a CW (or large pulsewidth) input at the

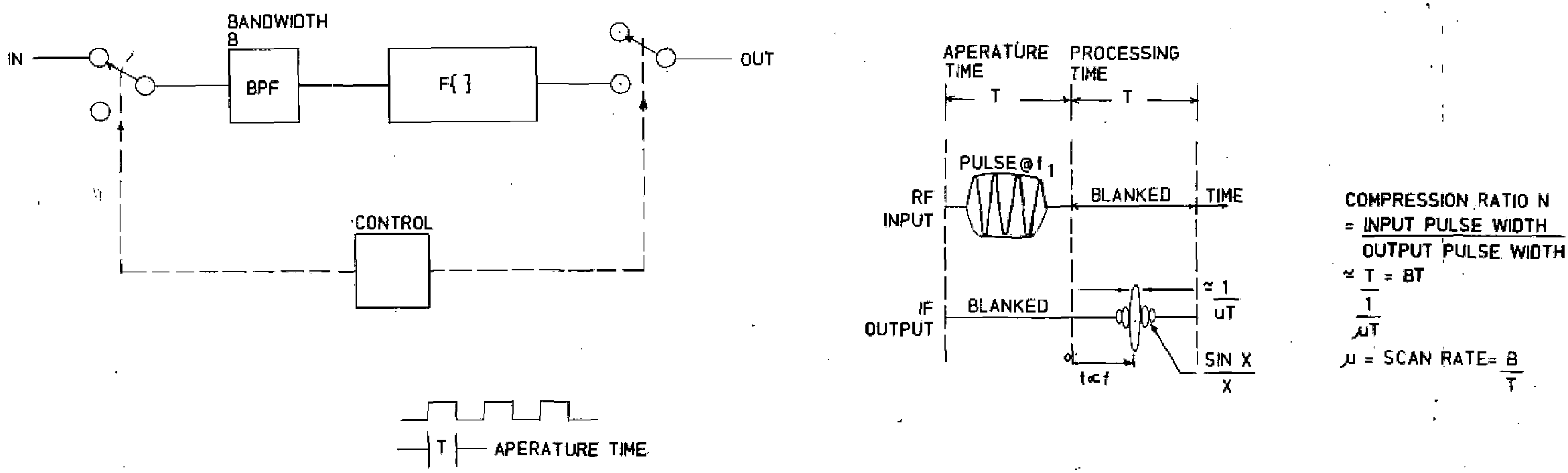


FIGURE 6-4(a): MICROSCAN RECEIVER, SIMPLIFIED EQUIVALENT CIRCUIT

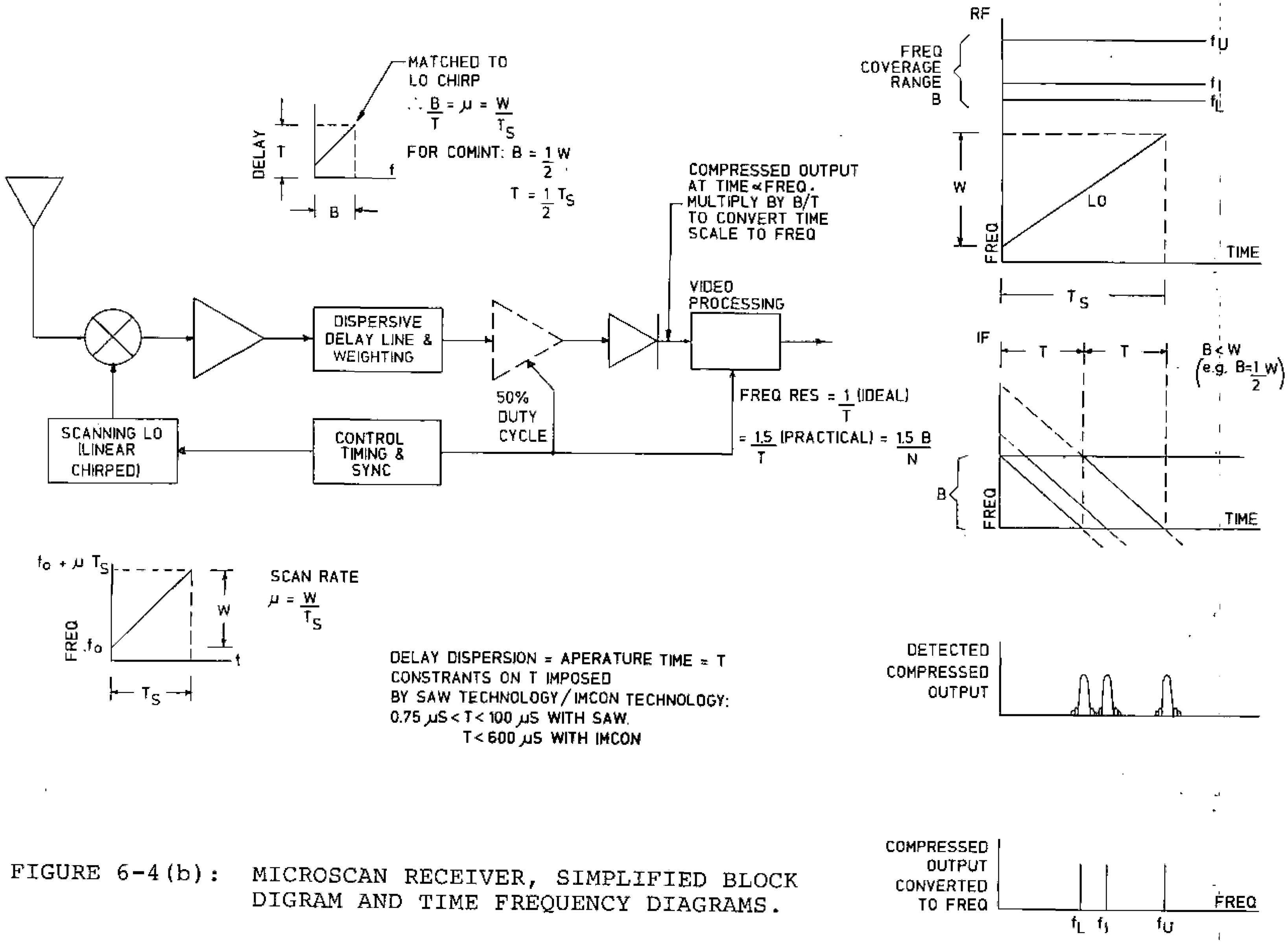


FIGURE 6-4(b): MICROSCAN RECEIVER, SIMPLIFIED BLOCK DIGRAM AND TIME FREQUENCY DIAGRAMS.

6.3.2.4 Compressive (Microscan) Receiver (Continued)

same peak power level. Ten decibels is due to a reduction in the compression ratio and the other 10 dB is due to the frequency spectrum spreading that occurs as a pulse is narrowed. This can significantly degrade the achievable sensitivity for pulsewidths must less than 1 uS, because there is a lower limit, typically 0.75 uS, on the achievable receiver aperture time, imposed by the minimum possible dispersion delay achievable in SAW delay lines and/or by the practical 100 MHz/0.1 uS upper limit on VCO sweep speeds relevant to very wideband applications.

6.3.2.5 Channelized Receiver

A receiver with three stages of channelization is shown in Figure 6-5. The signal is inputted to banks of contiguous filters with each filter designed to operate at a different center frequency. Sampling the filter outputs provides a direct measurement of the input frequency.

Band folding or time sharing between stages has been assumed in Figure 6-5 to conserve volume, weight and power consumption which are usually major problem areas associated with channelization. In a pure channelized system, the total frequency range to be covered, W would be divided into N bands, each band would be divided into a set of M sub-bands, and then each sub-band would be divided into Ku-channels of bandwidth B. The total number of filters would be (N) x (M) x (K), the frequency resolution would be approximately

$$B = \frac{W}{N \times M \times K}$$

and the frequency accuracy would be approximately B/2. In the configuration of Figure 6-5, there are N+M+K filters, giving the same frequency resolution and accuracy. However, the size, weight, power improvement does not come free. With band folding using combiners, there is a penalty paid in noise performance and therefore in detection sensitivity and there is ambiguity as to which band contains a detected signal. The ambiguity must be resolved by auxiliary means, such as individual detection in each band and in each sub-band channel. On the other hand, with switching between

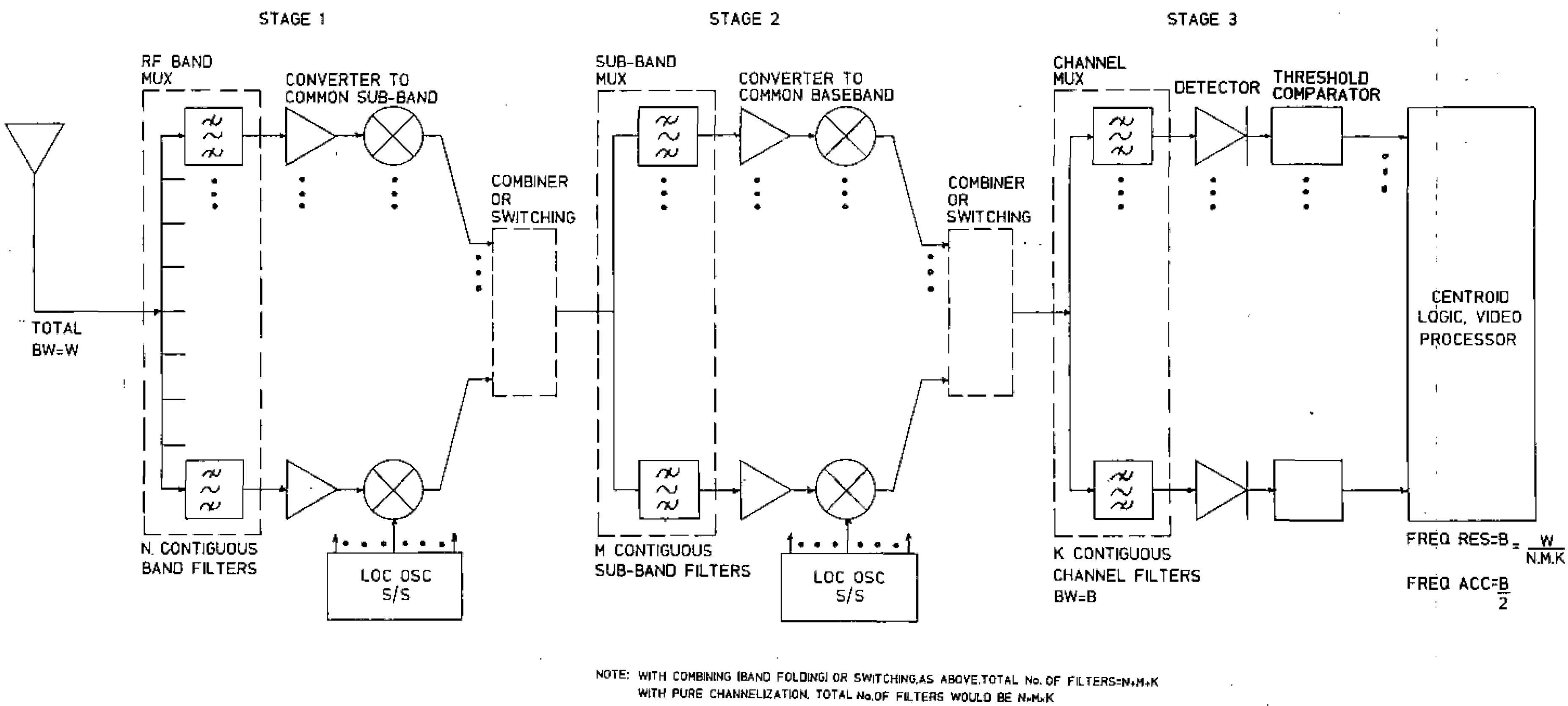


FIGURE 6-5: THREE STAGE CHANNELIZED RECEIVER, SIMPLIFIED BLOCK DIAGRAM

6.3.2.5 Channelized Receiver (Continued)

stages, the penalty is a reduction in the probability of intercept for non-CW signals and a degradation in acquisition time for all signals, when the switching is pre-programed to step through all bands and sub-bands. This can be improved if the switching is pre-programmed or controlled by external command, but with some a priori knowledge of the signals, or if the switching can be controlled by activity detectors in each band or sub-band channel, or by some combination of the above.

The frequency resolution equal to B and the bandwidth equal to B times the number of filters are determined in the final stage of channelization, where SAW technology is normally used in banks of typically 8 to 16 filters. Individual filter bandwidths of 0.5% to 40%, or typically 1 to 50 MHz, and maximum operating frequencies approaching 1 GHz, are realizable. At very low frequencies and very narrow bandwidths, physical size becomes a major constraint. Magnetostatic wave (MSW) channelizers up to X-band and with 10 to 50 MHz wide contiguous filters have also been reported, but it is unlikely that this technology has been qualified for satellite application.

The implementation of a channelized receiver is complicated for very narrow pulse signals if the sin x/x frequency spectrum extends over more than one filter bandwidth. Some method is required to identify the true power centroid of the signal. As reported in the literature, this can be done in one of the following ways:

(a) With contiguous filters, sample and hold the filter outputs and compare the detected outputs to determine the one larger than either of the contiguous adjacent ones. This method has the disadvantage of delaying the original frequency readout and adding to the processing complexity.

(b) Associate a wideband filter with each narrowband contiguous filter and analyze the two outputs in a comparator circuit to determine if the signal is in the narrowband filter. This approach works provided there is not more than one simultaneous signal within the wideband filter passband. It has advantage of instantaneous readout, but

6.3.2.5 Channelized Receiver (Continued)

doubles the number of filters and associated hardware required. It may also impose close tolerances on sampling time to avoid errors due to different responses in filters of different bandwidths.

(c) Use adjacent overlapping filters and make instantaneous adjacent filter comparisons. This eliminates the accurate sampling time requirements but still requires extra filters and associated hardware.

(d) Use autocorrelation techniques based on frequency and time domain responses of the filters so that the output of each filter provides sufficient data for determining the centroid.

To conserve hardware, some of the video processing may be time shared by the final filter (or groups of filters) outputs. At least two signal pulses are then required, one pulse to determine activity and to steer the processing circuitry and one pulse to get the measured data.

6.3.2.6 Acousto-Optic Bragg Cell Receiver

The Bragg Cell receiver is shown in Figure 6-6. There are three basic components:

(a) A solid-state laser light source,

(b) An acousto-optic device or Bragg cell with its associated optical system,

(c) An output photodetector array using either photodiodes or charge-coupled-devices (CCD).

The Bragg cell converts a received electrical (RF) signal into a travelling accoustic wave and interacts it with an optical beam to cause the beam to diffract with a diffraction angle proportional to the RF input frequency and with a diffracted light intensity linearly proportional to the RF power level. Frequency of the RF signal is determined by the location of the active detector in the detector array.

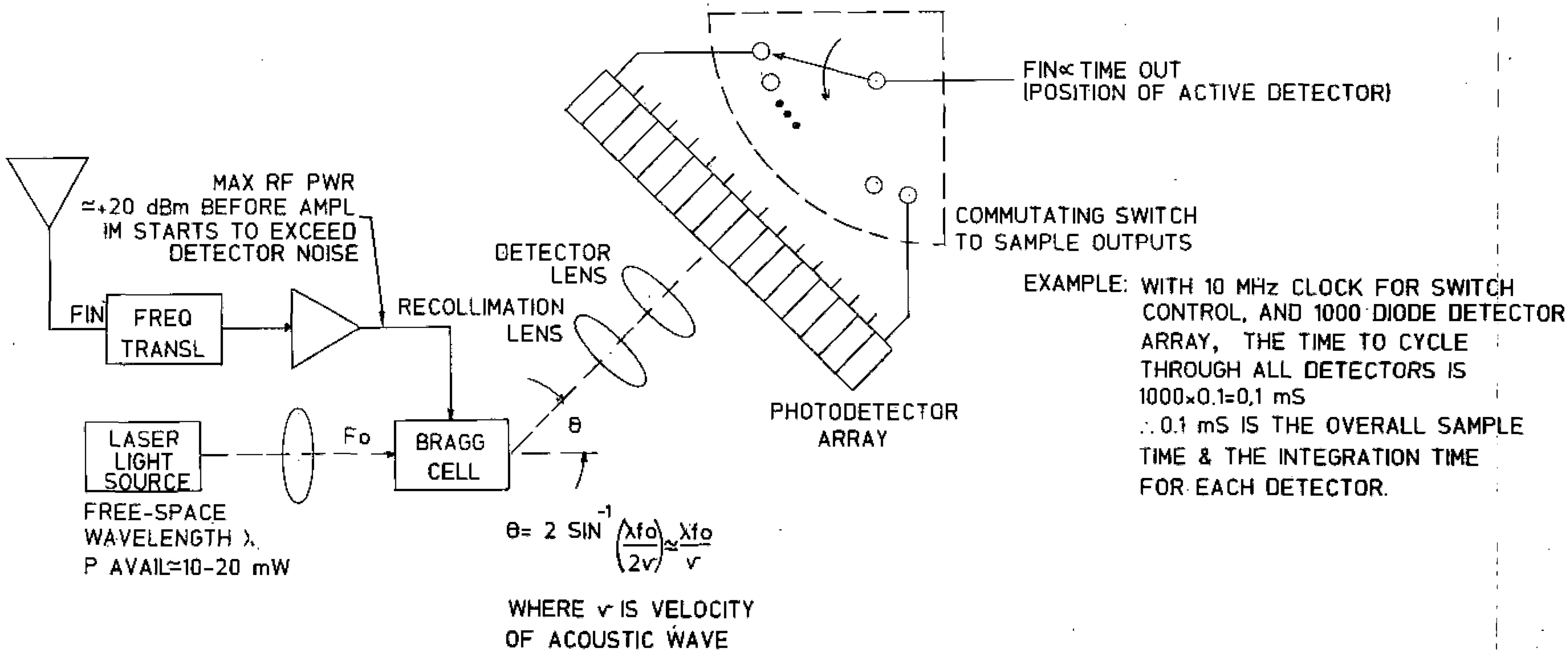


FIGURE 6.6(a): BRAGG CELL RECEIVER, SIMPLIFIED BLOCK DIAGRAM

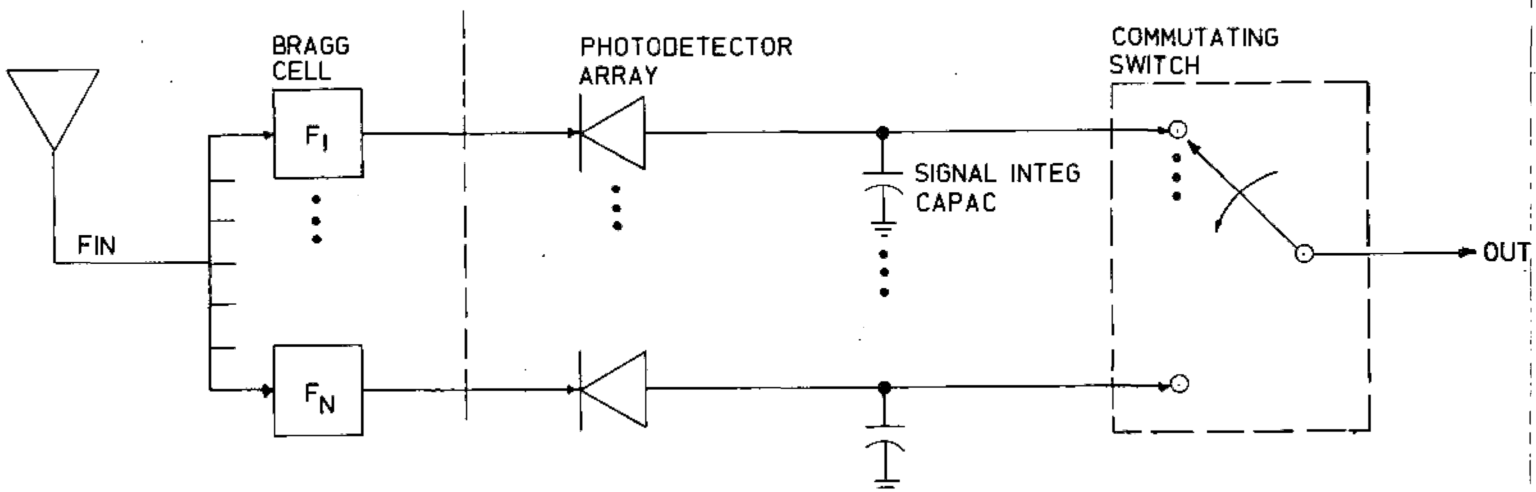


FIGURE 6.6(b): BRAGG CELL RECEIVER, EQUIVALENT CIRCUIT

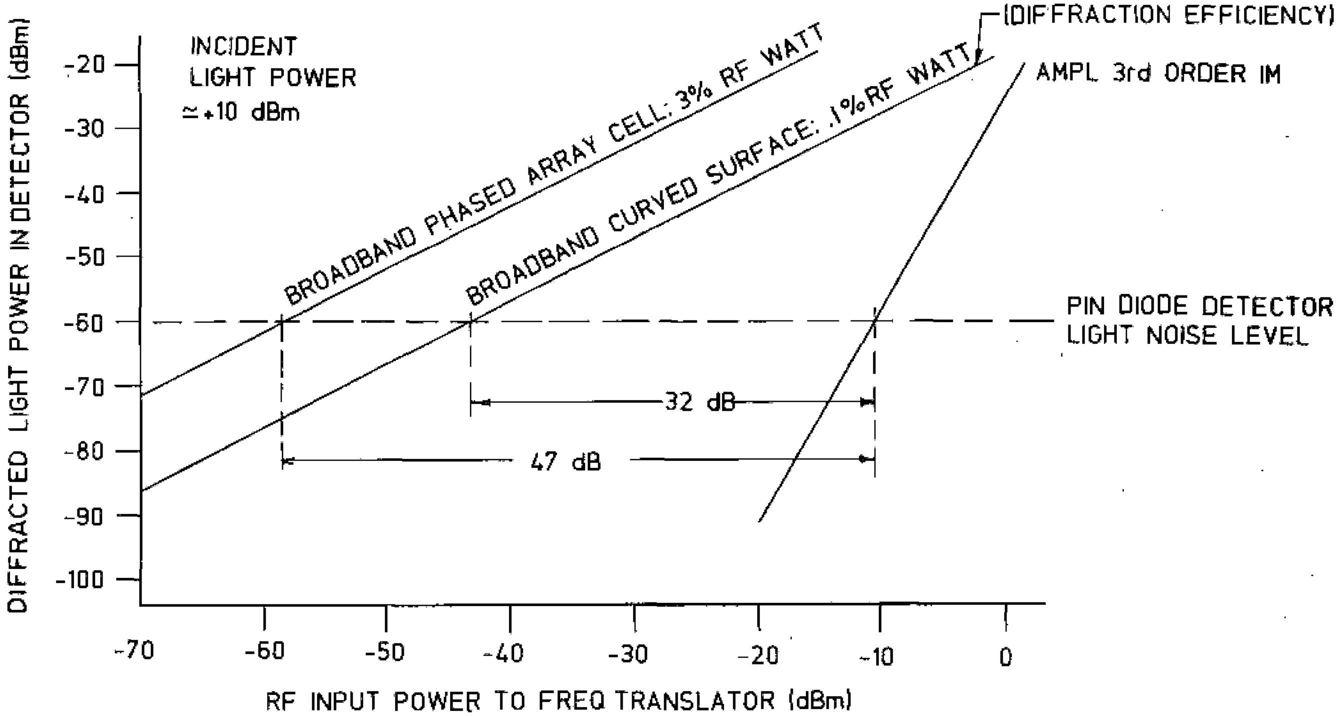


FIGURE 6-6(c): BRAGG CELL RECEIVER, DYNAMIC RANGE

6.3.2.6 Acoustic-Optio Bragg Cell Receiver (Continued)

The receiver essentially provides an Instantaneous Fourier Transform (IFT) of the signal and in terms of operating parameters, it is broadly similar to filter channelization and compressive receivers. The receiver is capable of receiving a wide frequency spectrum while providing good frequency resolution. The performance is characterized by a BW/Resolution of approximately 1000, for example up to 1000 MHz bandwidth with 1 MHz resolution and a maximum operating frequency of 2 GHz, or up to 30 MHz bandwidth with 30 kHz resolution and a maximum operating frequency of 75 MHz. Up to 2000 MHz bandwdith with 10 MHz resolution and a maximum operating frequency of 3 GHz has also been reported. The resolution is inversely related to the Bragg cell effective aperture (size) and is limited by the technologies of growing long crystals and then effectively spreading the light beam over a wide aperture.

The main performance limitations of the Bragg cell receiver is its relatively small dynamic range, see Figure 6-6(c) and its poor sensitivity to narrow pulses or to low duty cycle signals because the output photodetection is an energy detection process requiring and adequate integration time, typically greater than 100 nS. It also tends to be difficult to provide weighting in the cells to achieve spectral sidelobe suppression equivalent to that achieved with SAW devices. This may degrade resolution (deciding where the power centroid is in the detector array) for pulsed sin x/x spectral signals or it may further degrade dynamic range in a multiple pulse sin x/x spectral environment. Finally, although Bragg cell receivers are commercially available, their qualification status for satellite applications is unknown.

6.3.2.7 Digital Fast Fourier Transform (FFT)

With this technique, samples of the input signal are taken, digitally encoded using analog-to-digital conversion, and then processed by digital means. Usually down-conversion is required at the input to get down within the operating frequency range of available digital circuitry. The technique is not applicable to wideband applications because even after downconversion, high operating frequencies would still be required. For

6.3.2.7 Digital Fast Fourier Transform (FFT) (Continued)

narrowband applications, say up to 10 MHz, the performance can be comparable and competitive with SAW compressive receivers, but the digital FFT processor tends to be expensive and power-hungry and highly dependent on developments in very high speed integrated circuit (VHSIC) technology.

6.3.2.8 Available Receiver Approaches - Summary of Characteristics

The characteristics of each of the previously described receiver types are summarized in Table 6-3. Generally, no one receiver technique, on a stand-alone basis, can meet all the requirements of a given system and it is then necessary to use a combination of techniques that best exploits the advantages and minimizes the disadvantages of the different techniques.

SAW and IMCON dispersive and non-dispersive delay lines are fundamental to many of the receivers. Limits on frequency, bandwidth and time delay for these devices that impact their use in these receivers are shown in Figure 6-7.

6.3.2.9 Some Possible Receiver Combinations

(a) Use of Wideband Superhet at the Input

One simple and commonly-used combination technique, is the use of an input wideband superhet or bank of superhets with a common lower IF baseband output frequency, followed by one of the other receiver types, such as a SAW IFM, a TFR scanning superhet, a SAW compressive receiver, or a Bragg cell receiver, to increase the coverage of these receivers to higher frequencies and/or over wider bandwidths.

(b) Channelized Receiver With Internal IFM

SAW IFM techniques can be incorporated within the filterbank of a coarse SAW channelized receiver by proper choice and design of adjacent channel filters to achieve a discriminate characteristic capable of measuring narrowband pulsed and CW

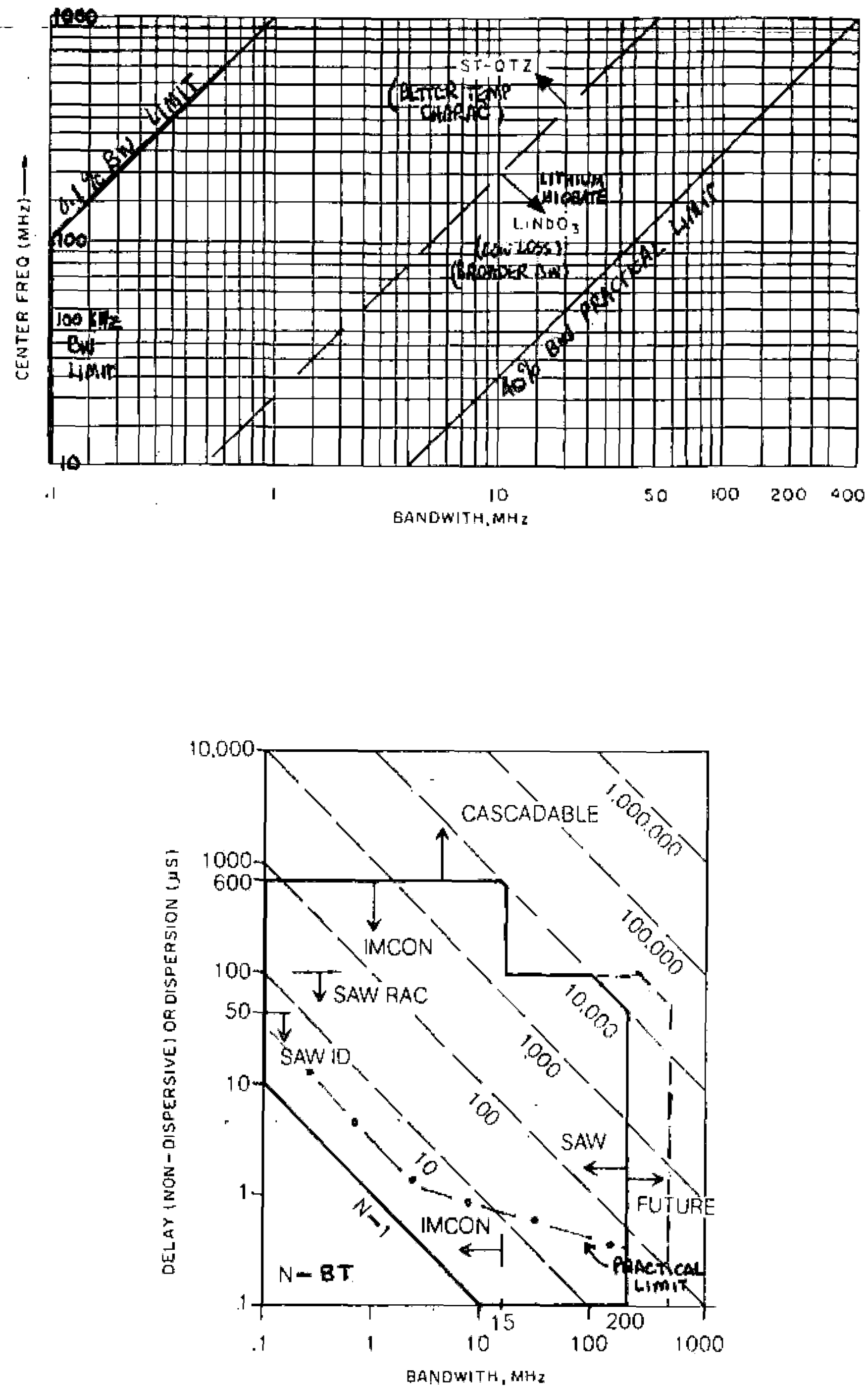


FIGURE 6-7: FREQUENCY, BANDWIDTH AND TIME DELAY LIMITS FOR DISPERSIVE AND NON-DISPERSIVE SAW AND IMCON DELAY LINES.

PERFORMANCE CHARACTERISTIC	CVR	SCANNING SUPERHET	IFM	COMPRESSIVE RECEIVER	CHANNELIZED RECEIVER	ACOUSTO-OPTICAL RECEIVER	DIGITAL
POI (%) for:							
CW	100	100	100	100	100	100	
Pulsed	100	Depends on signal duty cycle & on scan speed	100 (if tp>T)	100 ⎤ If use 2 in parallel	100	100	X
Chirped	100	Poor	100	100	100	100	
Frequency Agile (or MFSK)	100	Poor	100	100 ⎦	100	100	
Operation with Multiple Simultaneous Signals	Poor. Get Ampl Distortion	Good, but possible IM and spurious; and except for CW, POI degrades	Poor. Measures only highest power signal; with equal power signals, frequency data is erroneous	Excellent	Potentially excellent, but limited if time sharing used	Excellent	X
Speed of Signal Acquisition	Instantaneous	Slow; depends on scan BW and speed	Instantaneous	Near Instantaneous	Potentially instantaneous unless time sharing used	Potentially instantaneous except for detector integ time	X
Sensitivity, Typical (dBm)	-45 without preampl	-70 to -105 dependent on coverage band and resol BW	-55 to -70 with input preampl and dependent on coverage band	Comparable to superhet with equiv resol BW. Degrades with very narrow pulses (tp<T)	Comparable to equiv BW superhet but slightly worse because of channelization losses; degrades for tp<1/B	-60, but degrades with very narrow pulses (tp<100 nS)	X

TABLE 6-3: SUMMARY OF RECEIVER CHARACTERISTICS ON STAND-ALONE BASIS

PERFORMANCE CHARACTERISTIC	CVR	SCANNING SUPERHET	IFM	COMPRESSIVE RECEIVER	CHANNELIZED RECEIVER	ACOUSTO-OPTICAL RECEIVER	DIGITAL
Dynamic Range (dB)	50 without preampl	70 with TRF to eliminate spurious if necessary	70 for single signal	40 to 45 for ELINT 50 for COMINT; limited by compressed pulse sidelobes; degrades for narrow pulses	70; degrades due to spectrum spreading for narrow pulses	40; limited principally by detector array	70
Upper Operating Freq Limit (up to 40 GHz reqd)	>40 GHz	>40 GHz	1 GHz with SAW; tens of gigahertz with microw disc	1 GHz using SAW	>40 GHz	4 GHz for ELINT 75 MHz for COMINT	60 MHz
Instantaneous Freq Coverage	Full 40 GHz possible	With YIG 50 MHz maximum; without YIG, 1 MHz to 10 GHz due to IF; also scan coverage is up to 1 octave or 10 GHz whichever is less	1/2T; 1.5 octaves possible with WB D/L disc; 40% possible with SAW down to 0.5%, except for tp<100 nS, then minimum is >10 MHz	B; 250 MHz maximum with SAW, 15 MHz maximum with IMCON	Full 40 GHz possible but weight, size, DC power very high	2 GHz maximum for ELINT, 30 MHz maximum for COMINT	60 MHZ
Frequency Resolution	Frequency not measured	$\sqrt{2}\,B_{IF}$ ≈ scan rate e.g. 1 MHz possible with scan rate of 1 MHz/uS	Fraction of 1/2T; down to 0.1% accuracy or 50 kHz possible with one D/L and BW=10 MHz or with multiple D/L's and wider BW	1.5/T=1.5 B/N; down to 20 kHz with SAW and 2 kHz with IMCON possible	B; down to 1 MHz with SAW before weight and size become constraint	2 MHz for ELINT 30 kHz for COMINT	Trade-off vs cot complexity size and weight

TABLE 6-3: SUMMARY OF RECEIVER CHARACTERISTICS ON STAND ALONE BASIS cont'd....

PERFORMANCE CHARACTERISTIC	CVR	SCANNING SUPERHET	IFM	COMPRESSIVE RECEIVER	CHANNELIZED RECEIVER	ACOUSTO-OPTICAL RECEIVER	DIGITAL
TOA Resolution	< 50 nS; determined by time resolution of digital logic	Ts; down to 0.2 uS possible, with W=10 B_{IF} and max scan rate (100 MHz/0.1 uS)	< 50 nS as for CVR	T; down to 1 uS possible, limited by dispersive delay filter	50 nS, as for CVR, plus allowance for switching if time sharing involved	100 nS for detector integ plus allowance for output readout scan	X
Hardware Complexity	Low	Moderate	Moderate	High	High particularly if fully channelized	Moderate	X
Size/Weight	Moderate	Moderate	Low	Moderate	High	Low	X
DC Power	Low	Moderate	Moderate	Moderate to high depending on processing speed	High	Moderate	High due to high speed logic
Technical Maturity for Satellite Application (NOTE 1)	Good Old technology	Good Old technology	Good	Good for narrowband applications	Good	Questionable although commercially available	Question-able
Definition of Special Symbols used in Table	None	B_{IF} = IF BW T_S = scan period W = scan BW	tp = input pulse period T = delay used in disc D/L= Delay line	tp = input pulse period T = aperature time = dispersion delay B = IF BW N = compression ratio	tp = Input pulse period B = channel filter BW	None	None

NOTE 1: Does not include military satellite applications for which there is little or no data.

TABLE 6-3: SUMMARY OF RECEIVER CHARACTERISTICS ON STAND-ALONE BASIS ..../Final

6.3.2.9 Some Possible Receiver Combinations (Continued)

signals. The bandwidths of the coarse channelized filters can be selected for wideband signals so that the weight/size penalty associated with very narrowband SAW filters is avoided.

(c) Coarse Channelization Followed by Fine Resolution IFM with Switching to IFM Controlled by Channelizers

This configuration shown in Figure 6-8 can be used to overcome frequency resolution limits and harsh weight tradeoffs associated with channelized receivers but with the preselection in the channelizer used to improve the performance of the IFM in a dense multisignal environment. Signal activity detectors at the output of the coarse SAW channelizer are used to control the connectivity of the delayed output to a smaller number of SAW IFM's that can make finer frequency measurements. Compressive receivers or narrowband superhets might be used instead of the IFM's.

(d) Coarse Channelizer Used to Set-On a Parallel Narrowband Superhet and/or Compressive Receivers

The set-up shown in Figure 6-9(a) is another way to work around the frequency resolution/weight limitations of a channelized receiver by using a superhet in a manner that minimizes any degradation of probability of intercept associated with the scanning process in the superhet.

The outputs of a coarse frequency channelizer are used to control the frequency of the local oscillator in a narrowband superhet (or in a small number of them, for multiple simultaneous signals) to set on the superhet to receive a delayed replica of the active part of the IF input frequency band.

Alternatively, by adding a stage of downconversion, the input signal can be translated down to a lower frequency appropriate for even finer frequency resolution. This is shown in Figure 6-9(b), where the coarse frequency SAW channelizer outputs are used to steer the

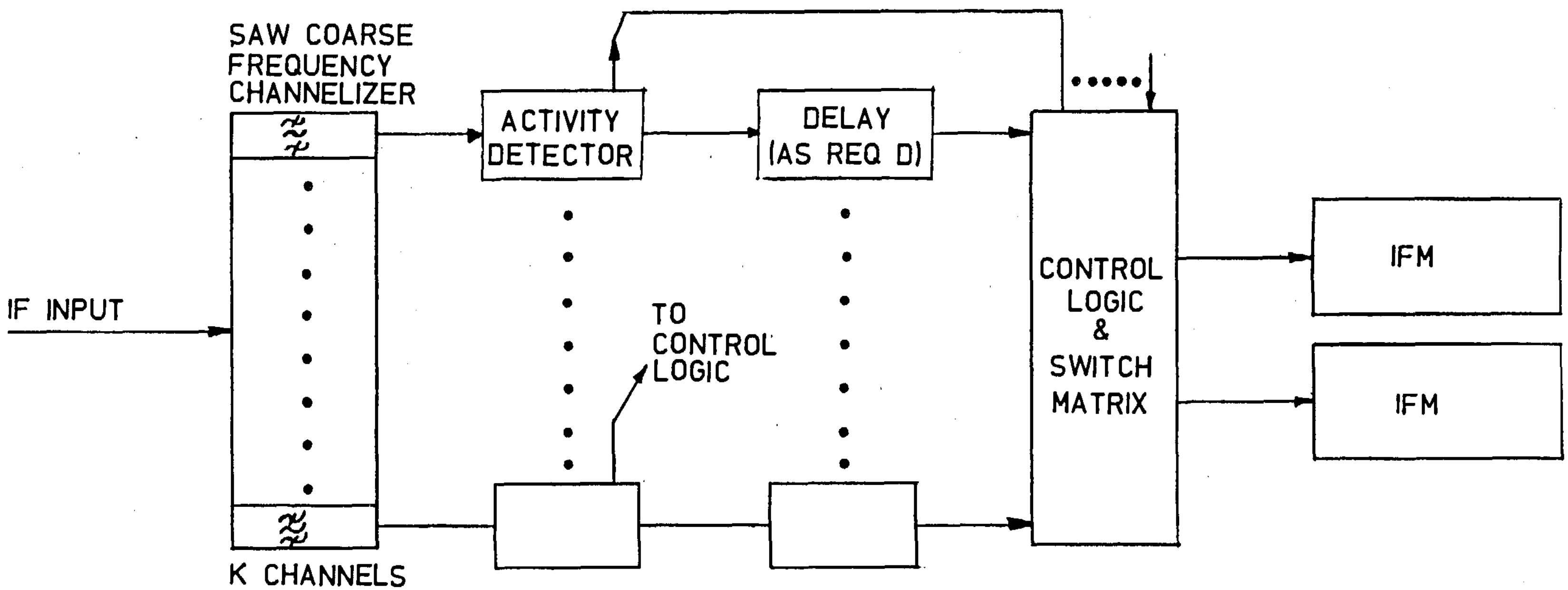


FIGURE 6-8: COARSE CHANNELIZER FOLLOWED BY IFM

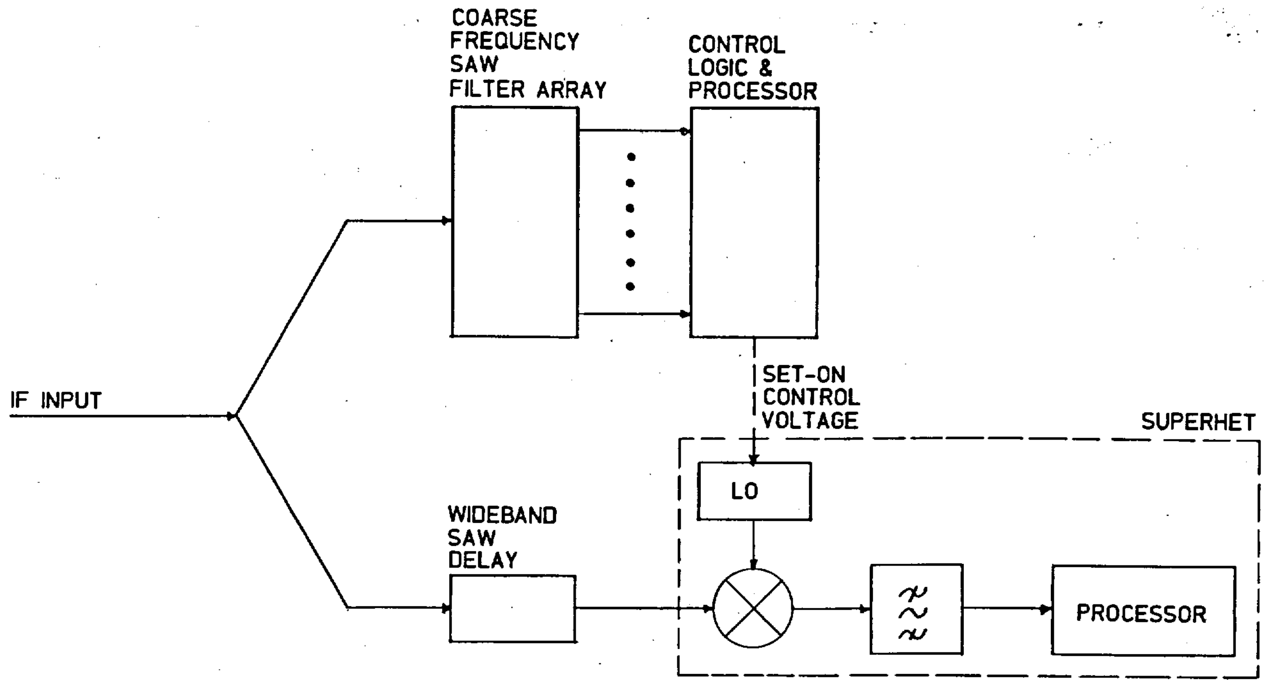


FIGURE 6-9(a) COARSE CHANNELIZED USED TO STEER SUPERHET

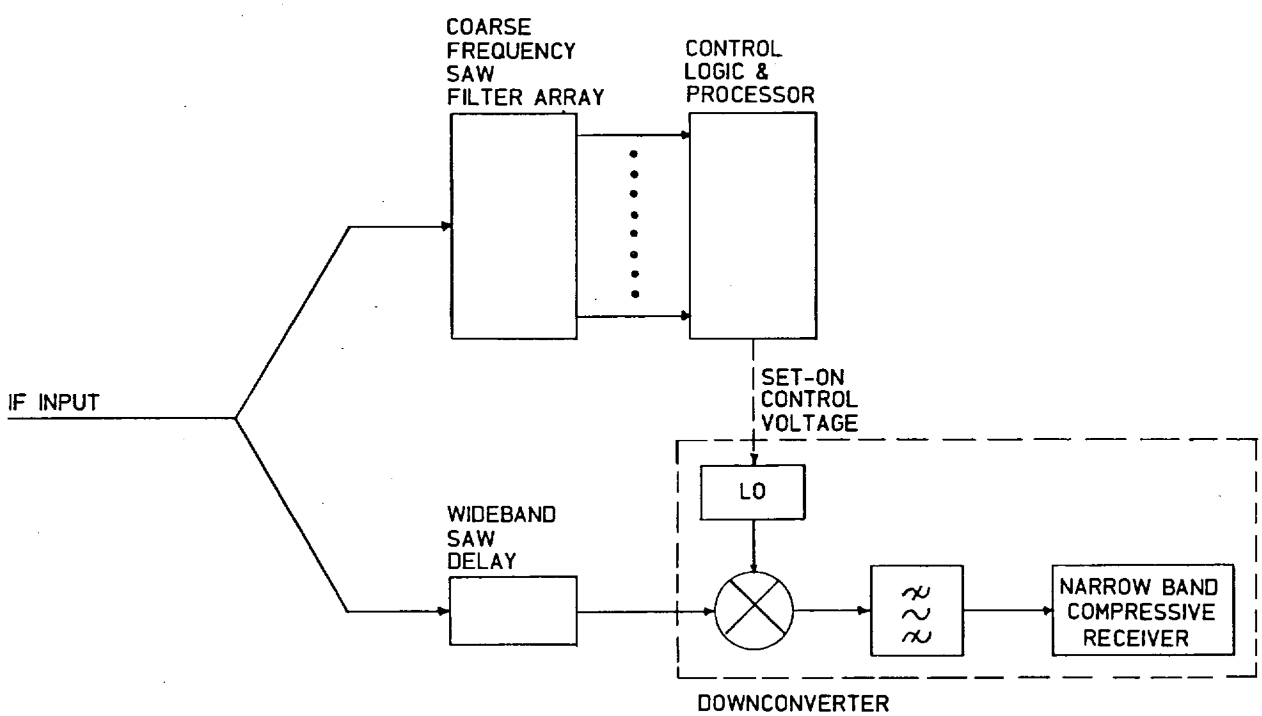


FIGURE 6-9(b): COARSE CHANNELIZER USED TO STEER DOWNCONVERTER/ COMPRESSIVE RECEIVER

6.3.2.9 Some Possible Receiver Combinations (Continued)

downconverter local oscillator so that a delayed replica of the active part of the IF input band falls at the correct input frequency for a narrowband compressive receiver.

(e) Compressive Receiver Used to Set-On Narrowband Superhet

This configuration is shown in Figure 6-10 and might be used to obtain better time of arrival (TOA) resolution, than can be provided by the compressive receiver, which has a limit of about 1 uS set by its aperture time. In a non-scanning set-on superhet, the time resolution is determined by the speed and hence time resolution of the receiver processor clock and other digital logic circuitry, which even with conventional low-to-medium power technology should be less than 100 nS. Another application for this approach would be for interferometry, where the set-on superhet can supply accurate phase difference data using conventional narrowband phase detection circuitry in preference to the more complex phase detection circuitry that would be required in the compressive receiver. By using the compressive receiver to set-on the narrowband superhet to the active part of the input frequency coverage band, the need for scanning in the superhet and the associated scan rate/frequency resolution/POI limitations are eliminated. In a narrowband fixed superhet, the frequency accuracy and dynamic range are also improved.

An alternative arrangement would be to use an IFM receiver in place of the compressive receiver. This can provide wider frequency coverage, particularly in a configuration such as shown in Figure 6-11, but performs poorly when there are multiple simultaneous input signals.

6.3.3 Receiving Subsystem Proposed for Paxsat

A receiver configuration with the flexibility and technological maturity to meet the PAXSAT mission requirements is shown in Figure 6-12. It consists of three stages of channelization with the final stage used

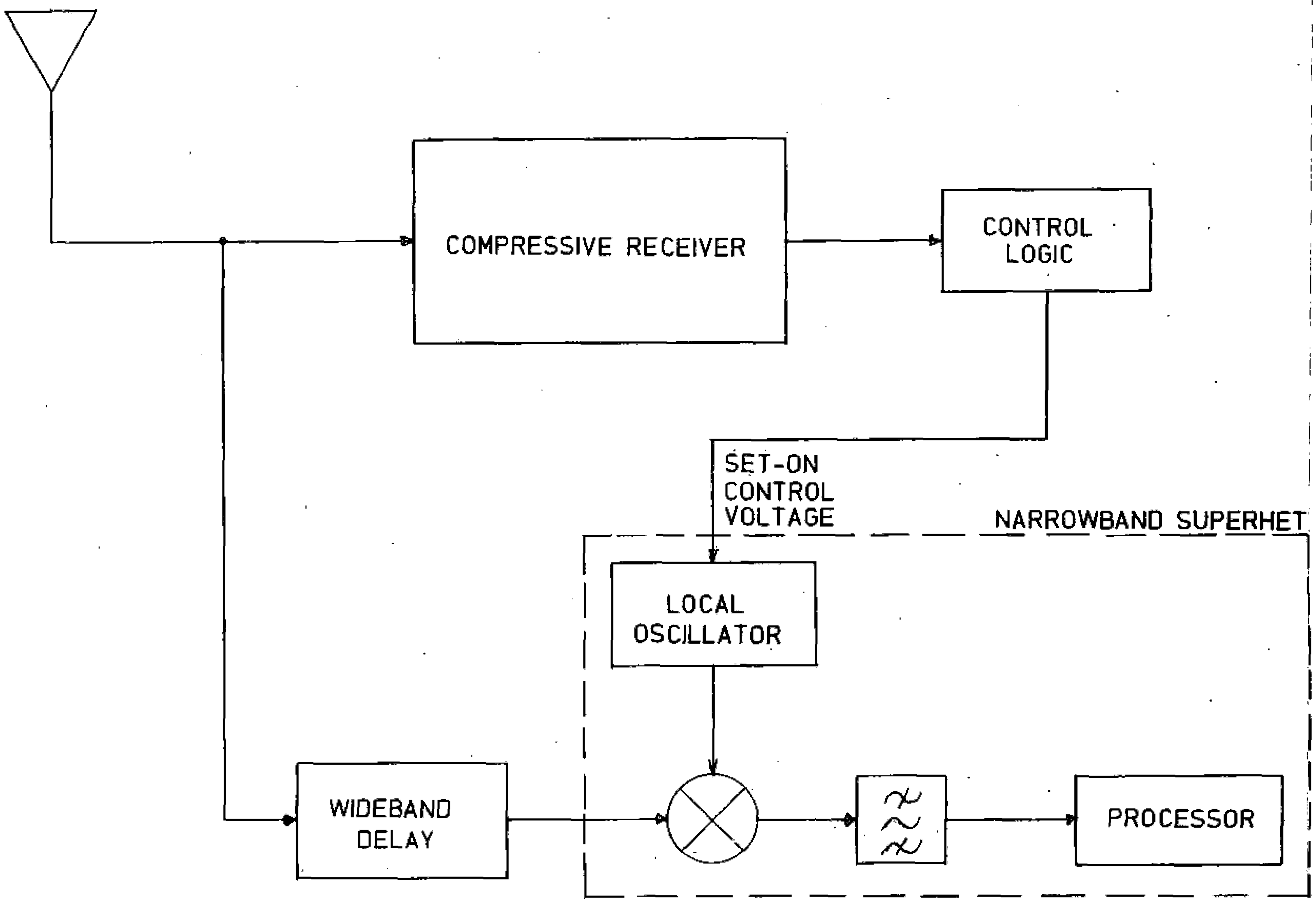


FIGURE 6-10: NARROWBAND SUPERHET SET-ON BY COMPRESSIVE RECEIVER

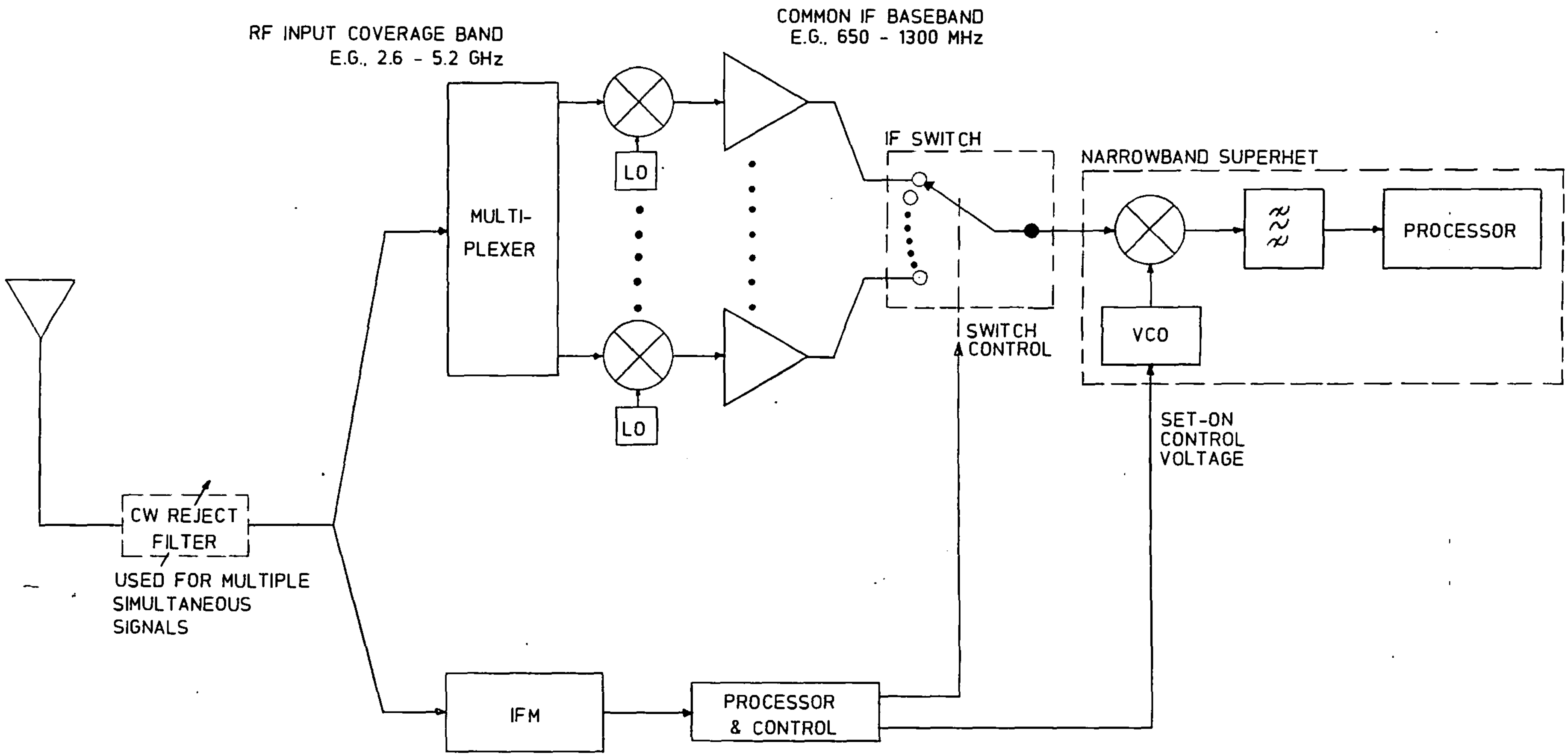


FIGURE 6-11: NARROWBAND SUPERHET SET-ON BY WIDEBAND IFM

6.3.3 Receiving Subsystem Proposed for Paxsat (Continued)

to set-on the input to a narrowband compressive receiver. Switching matrices, normally controlled by activity detectors in each of the inputting channels, are used between the channelization stages. These provide circuit time sharing so as to reduce circuit complexity, size, weight and dc power compared to a fully channelized system. Solid-state PIN diode multi-throw switches are available for this application.

The first channelization stage divides the very broad 0.35 to 40 GHz input frequency coverage range into nineteen contiguous slices, each 2 GHz wide. Each of these is converted to a common 2 to 4 GHz sub-band. There are also 2 bands/sub-bands below 2 GHz. 2-4 GHz is a commonly used sub-band for this type of application, and there is a lot of mature technology available in this band. Waveguide and coupled-line suspended stripline in air dielectric are candidate filter technologies for the input multiplexer. The input multiplexing has been divided into three somewhat arbitrary blocks which should be compatible with available antenna technology.

The second channelization stage divides the 2 to 4 GHz and the 0.35 to 2 GHz sub-bands into fourteen contiguous slices, each 150 MHz wide. Each of these is converted into a common 0.35 to 0.5 GHz IF baseband which is within the range of conventional SAW technology. Coupled-line or possibly magnetostatic wave (MSW) filter technology are candidates for the de-multiplexing in this stage.

The final channelization stage divides the common 0.35-0.5 GHz IF baseband into twelve contiguous slots, each 12 MHz wide, to match typical radar transmitter bandwidths. The outputs provide coarse frequency resolution of approximately 12 MHz and other data depending on the processing that follows. They are also used to set-on the input to a paralleled 70 MHz, 12 MHz wide compressive receiver which can provide fine frequency resolution to 25 kHz and other processed data for narrowband communications type signals, that cannot be efficiently measured with a channelized receiver approach.

The input to the compressive receiver is set-on by a downconverter controlled by the coarse channelizer outputs so that the active part and only the active part

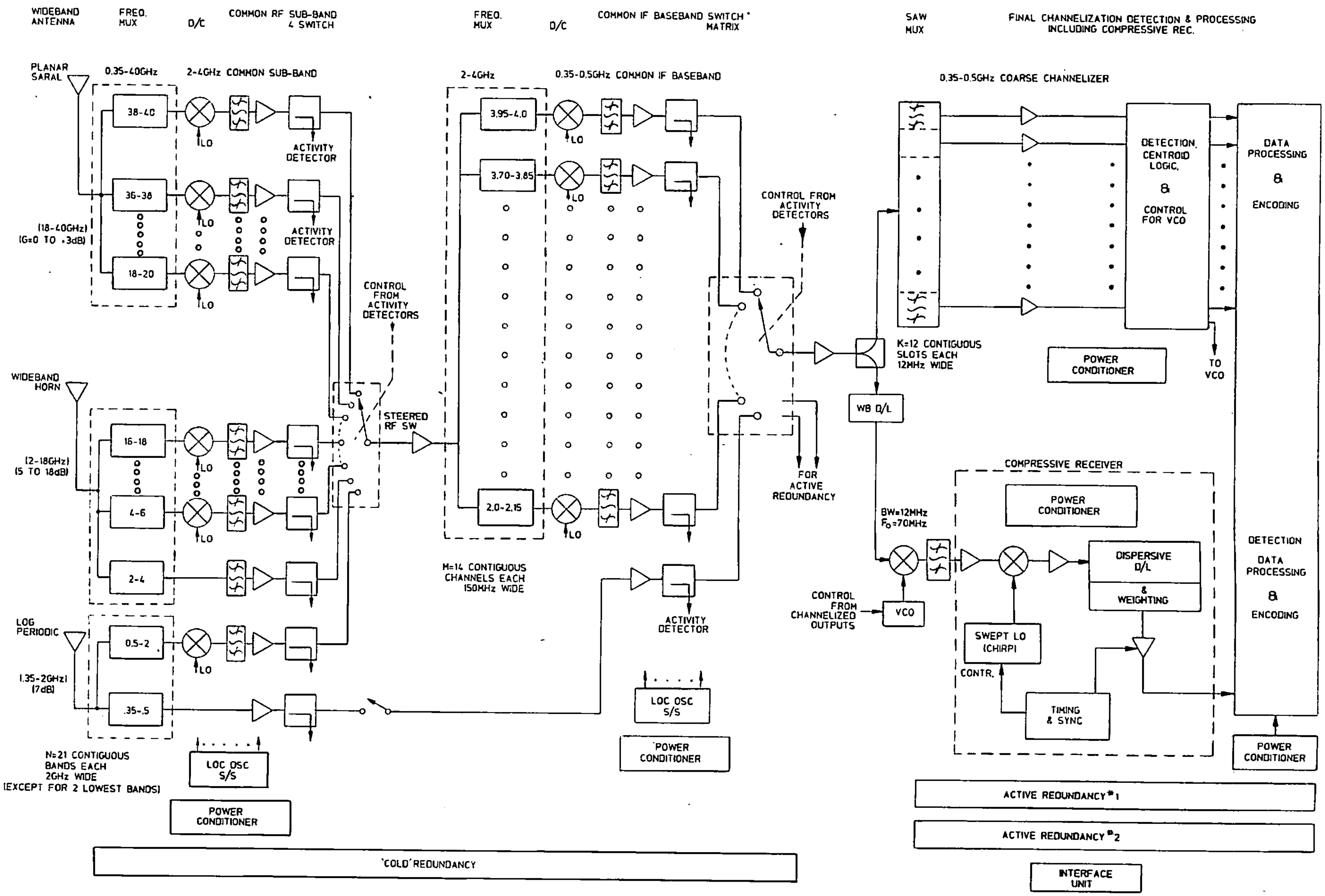


FIGURE 6-12: PROPOSED ELECTROMAGNETIC RECEIVING SYSTEM FOR PAXSAT.

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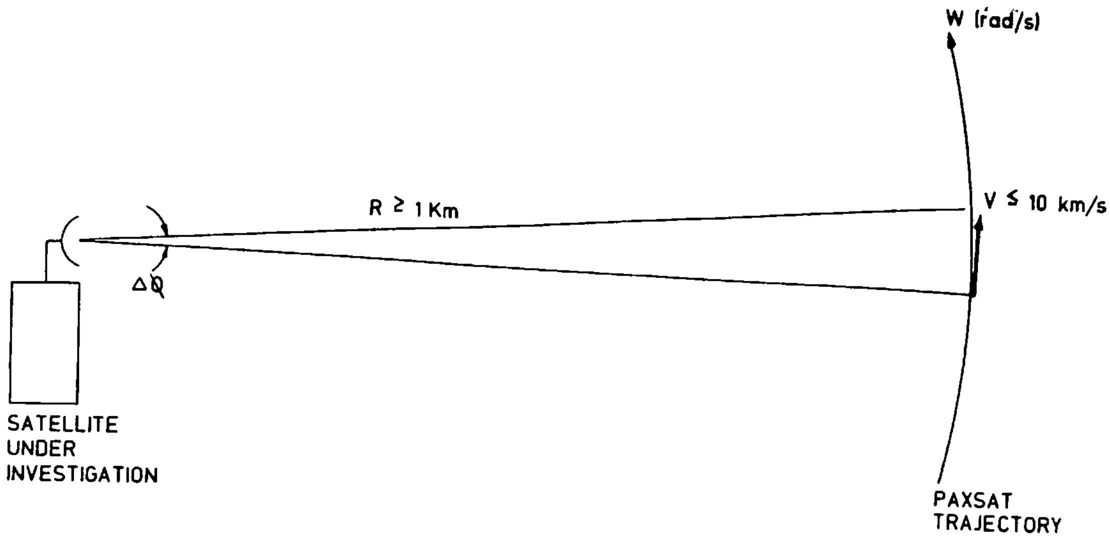
6.3.3 Receiving Subsystem Proposed for Paxsat (Continued)

of the 150 MHz wide common IF baseband is converted to the correct input frequency for the compressive receiver. The compressive receiver is able to achieve a resolution of 25 kHz while looking at a complete 12 MHz band, which is a useful feature say, for a frequency-hopping signal.

In order to improve flexibility for analyzing multiple simultaneous signals, or wideband signals, or frequency-hopping signals that cover more than one 150 MHz channel in the coarse channelizer or more than one 12 MHz slot in the compressive receiver, a configuration with three final stages in 'active' redundancy is proposed. Indeed, for the compressive receiver in the final stage, at least two interleaved receivers are required to guarantee 100% probability of intercept for all signals of interest. With the proposed active redundancy, there would be a graceful performance degradation, i.e., reduced capability, as redundant sections failed. To conserve power, the number of active sections could be reduced to two.

In the proposed receiving system, antenna pattern measurements of the satellite under surveillance would be made by measuring received power level vs time in a fly-by scenario as shown in Figure 6-13. It is assumed that the Paxsat location, orientation, range and velocity, vis-a-vis the satellite under investigation are known through other means, e.g., through a radar subsystem. The time of arrival measurement resolution of the compressive receiver and of the coarse channelizer receiver are adequate for making the antenna measurements to better than a 0.05 degree (10^{-3} radian) angular resolution, provided the relative velocity and range are known with sufficient accuracy, and are not larger than 10 Km/s and not less than 1 km respectively, as shown in Figure 6-13.

The estimated weight/size and DC power budgets for the proposed receiver are shown in Tables 6-4, 6-5 and 6-6. The total weight including antennae, is 275 lb or 125 kg and the total primary DC power required is 450 W. The peak DC power requirement could be reduced by say 25%, if it is arranged to sequentially power on certain parts of the receiver on an "as required" basis, with allowances for warm-up times that would probably exclude some subsystems such as the local oscillators, from this



For V_{MAX} = 10 km/S and R_{MIN} = 1 km, $W_{MAX} = V_{MAX}/R_{MIN}$ = 10 rad/S.

To achieve a 0.05° (10^{-3} rad) antenna pattern angular resolution, the required accuracy for Time of Arrival (TOA) measurement is $10^{-3}/W_{MAX} = 10^{-4}$ S.

For a compressive receiver, TOA resolution = aperture time. Therefore for antenna pattern measurement mission, receiver aperture time must not exceed 10^{-4} S = 100 uS.

In practice, receiver aperture time = $\dfrac{1.5}{\text{Frequency Resolution}}$

Therefore, for a receiver with 25 kHz frequency resolution, the aperture time is 1.5 x 40 = 60 uS.

This is compatible with requirements for antenna measurement.

For the channelized receiver with a frequency resolution of 12 MHz, the TOA resolution/accuracy is better than 1/B or better than 0.1 uS, say 0.05 uS due to time resolution of digital logic.

FIGURE 6-13 ANTENNA PATTERN MEASUREMENT (POWER LEVEL VERSUS TIME)

TABLE 6-4 RECEIVER CHAIN WEIGHT, SIZE AND DC POWER BUDGET ESTIMATES

UNITS	WEIGHT		SIZE		DC POWER
	lbs	kg	(cu in)	(cu cm)	(watts)
RF Input Multiplexer	7.5	3.4	125	2000	0
RF/Sub-Band D/C	32.	14.6	500	8200	42
RF Switching	3.5	1.6	60	1000	4
Sub-Band Multiplexer	5.	3.3	85	1400	0
Sub-Band/IF D/C	21.	9.5	350	5700	28
IF Switching	5.5	2.5	90	1500	6
SAW Mux Channelizer	6	2.7	100	1600	12
IF/70 MHz D/C	0.5	0.2	8	130	1
Compressive Receiver	4	1.8	65	1000	10
Output Data Processing	15.	6.8	250	4000	45
Heaters for LO's and for Final Stage SAW's	-	-	-	-	20
TOTALS	102	46.4	1633 (Approx. 1 cu ft)	26530 (27 litres)	168

TABLE 6-5 OVERALL EM ANTENNA/RECEIVER SUBSYSTEM WEIGHT

UNITS	FULL CONFIGURATION FIGURE 6-12		REDUCED VERSION FIGURE 6-14		FULL CONFIGURATION FIGURE 6-12		REDUCED VERSION FIGURE 6-14	
	PER UNIT WEIGHT				TOTAL WEIGHT INCL.			
	lb	kg	lb	kg	lb	kg	lb	kg
RF Input to IF Switching Output	74.5	33.9	46.2	21.0	149.0	67.8	92.4	42.0
Final Stage & Processing	25.5	11.5	25.5	11.5	76.5	34.6	51.0	23.0
Redundancy Switching	-	-	-	-	4.0	1.9	4.0	1.9
Interface Unit	10.0	4.6	10.0	4.6	10.0	4.6	10.0	4.6
Cabling & Harness	10.0	4.5	10.0	4.5	20.0	9.0	20.0	9.0
TOTAL RECEIVER	120.0	54.5	91.7	41.6	259.5	117.9	132.4	80.5
TOTAL ANTENNA	15.0	6.8	15.0	6.8	15.0	6.8	15.0	6.8
OVERALL TOTAL	135.0	61.3	106.7	48.4	274.5	124.7	152.4	87.3

TABLE 6-6 OVERALL EM RECEIVER SUBSYSTEM DC POWER REQUIREMENTS

UNITS	FULL CONFIGURATION FIGURE 6-12	REDUCED VERSION FIGURE 6-14	FULL CONFIGURATION FIGURE 6-12	REDUCED VERSION FIGURE 6-14
	PER UNIT POWER (WATTS DC)		TOTAL POWER (WATTS DC)	
RF Input to IF Switching Output	80	50	80	50
Final Stage & Processing	68	68	204	136
Heaters	10	10	20	20
Redundancy Switching	-	-	1	1
Interface Switching	10	10	10	10
SUBTOTAL	168	138	315	217
Power Supplies at 70% Efficiency			135	93
TOTAL			450	310

6.3.3 Receiving Subsystem Proposed for Paxsat (Continued)

approach. Also, there could be a power saving if the active redundancy in the final stage of the receiver was reduced.

Another measure to reduce size, weight and DC power would be to use two sets of stepped superhets in place of the first two stages of channelization. This is shown in Figure 6-14, where the number of superhets used is rather arbitrary but does illustrate the concept. The actual number of superhets required would be subject to a careful analysis of potential spurious products that arise because of the wider input frequency and local oscillator frequency bands. Another disadvantage of this configuration is slower signal detection because the superhets must be slowly stepped through their frequency coverage bands before they can be set-on, if required, using the activity detectors. The improvements in weight and DC power, however, are significant as indicated in Tables 6-5 and 6-6, which include the effects of also reducing the output stage redundancy from three to two. There is an estimated weight saving of 37 kg and DC power saving of 140 W after allowing for some extra spurious related filtering and processing, etc.

6.3.4 Receiver Antenna Configuration for Paxsat

There are three main considerations for the selection of the PAXSAT receive antenna subsystem.

(a) The selected antennae should have large frequency bandwidths to minimize the number and hence complexity of the antenna subsystem required for the total frequency coverage band from 0.35 to 40 GHz. They should be small and lightweight.

(b) The selected antennae should have moderately wide beamwidths to provide reasonable instantaneous spatial coverage so that accurate antenna pointing is not necessary. At the same time, the antenna gain should be as large as possible, although it will necessarily tend to be small because of the beamwidth requirement.

(c) The selected antennae should respond to multiple types of polarization to insure reception of all signals of interest.

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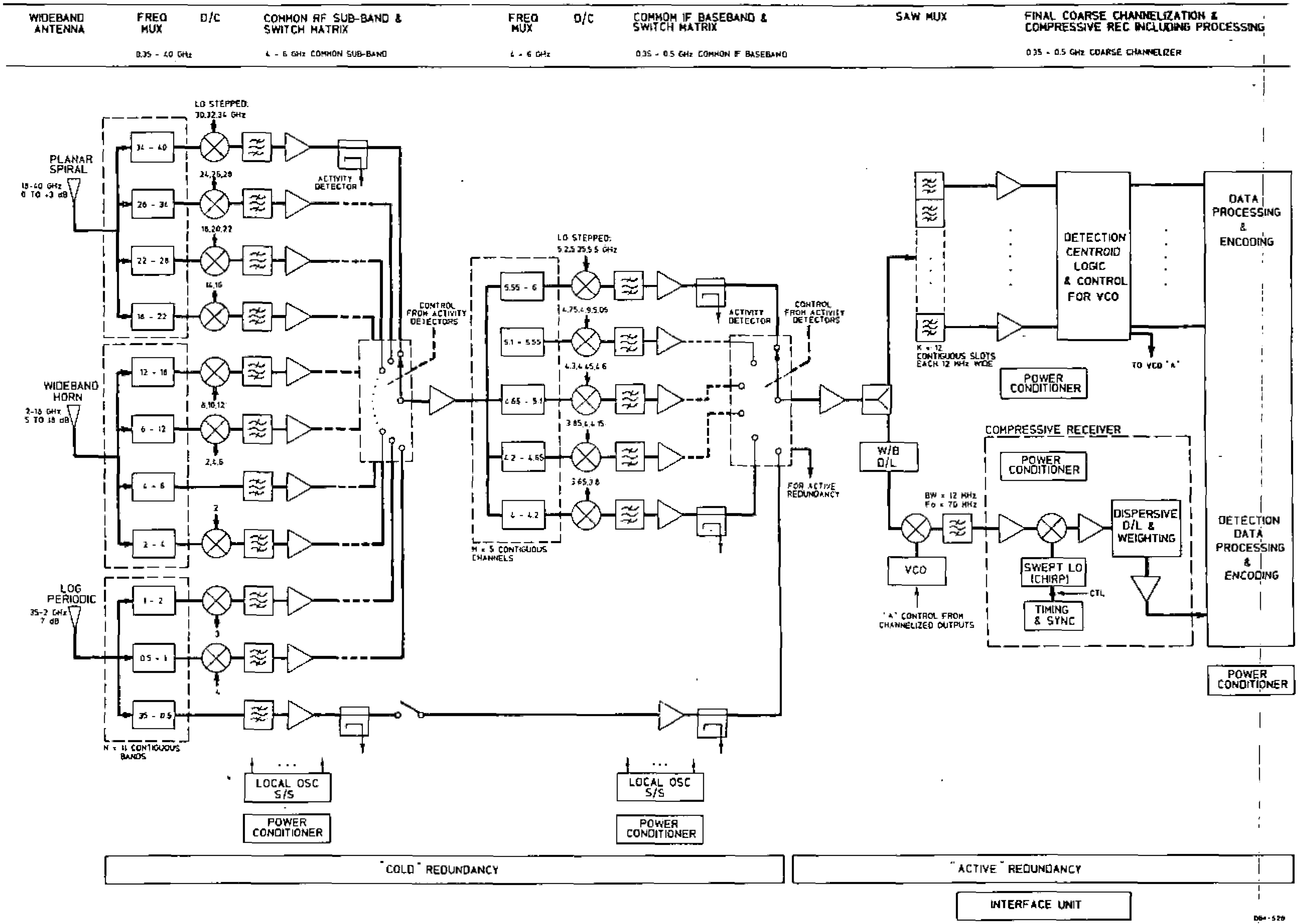


FIGURE 6-14: REDUCED VERSION OF EM RECEIVING SYSTEM FOR PAXSAT

6.3.4 Receiver Antenna Configuration for Paxsat (Continued)

Antenna types commonly used for electronic support measures (ESM) are the log periodic, the conical spiral, the planar spiral and the wideband horn. Some of their characteristics are summarized in Table 6-7.

As shown in Figures 6-12 and 6-14, the 0.35 GHz to 40 GHz range can be covered using three antennae, for example, a log periodic from 0.35 to 2.0 GHz, a wideband horn from 2 GHz to 18 GHz and a planar spiral from 18 to 40 GHz. With that arrangement, the gain will vary from 7 dB up to 14 dB and then down to 0 dB, with beamwidths from 65°, down to 35° and then back up to 60°, from low to high frequency. The estimated weight for this antenna farm, including auxiliary hybrids, etc, is 15 lbs.

Typical sizes for the antenna are as follows:

(a) Dual Polarized LPDA

Cone, with length = 150 cm (60 in)
and aperture = 86 cm (34 in)

(b) Dual Polarized Wideband Horn:

Cone, with length = 33 cm (13 in)
and aperture = 16 cm (6 in)

(c) Planar Spiral:

Cylinder, with length = 2.5 cm (1 in)
and aperture = 5 cm (2 in)

6.3.5 Summary of Proposed EM Receiving Subsystem Performance Capabilities

The performance capabilities of the proposed receiving system are summarized in Table 6-8.

In principle, the channelized receiver and/or the two paralleled compressive receivers have a 100% probability of intercept (POI) for all signals of interest. The system is able to operate in a multiple simultaneous signal environment, although simultaneous processing of the signals is limited by the time-sharing used between receiver stages and in the output processing and by the sequential set-ons that would be required at the compressive receiver inputs.

ANTENNA TYPE	FREQUENCY BW RATIO	OPERATING FREQUENCY RANGE & TYPICAL COVERAGE RANGES	POLARIZATION	TYPICAL VALUES FOR (*) ANTENNA	
				GAIN (dB)	BEAMWIDTH (degree)
Log Periodic, Planar Dipole Array or Dual - Polarized Array	10:1 to 50:1	20 MHz to 18 GHz; e.g. 20 to 1000 MHz 100 to 2000 MHz 1.0 to 18 GHz (*)	Linear; or circular, selectable by appropriate summing at O/T	7 to 8 dBli	E-plane: 65° H-plane: 100° (Fairly independent of frequency)
Conical Spiral Normal Mode and Axial Mode	2:1 to 10:1 (for normal mode)	0.1 to 12.4 GHz (to 18 GHz with degraded performance); e.g. 1 to 12 GHz (*)	Circular	0	Azimuth (in-plane of base of cone): 360 Elevation: 55 for normal mode; 180 for axial mode
Planar Spiral	9:1 (Cavity loaded) to	0.5 to 40 GHz e.g. 2 to 18 GHz (*) 8 to 40 GHz	Circular	-6 to +2 dBli from low frequency to high.	110 to 60 from low frequency to high
	single octave (unloaded	1 to 2 GHz 8 to 18 GHz		3 dB higher than cavity loaded	
Wideband Horn, Single or Dual Linear Orthogonal Polarized	2:1 to 9:1	1 to 26 GHz e.g. 8 to 18 GHz (*) or 2 to 18 GHz (*)	Linear; or circular, selectable by appropriate summing at O/T	6 to 14 dBi 5 to 18	80 to 35 from low frequency to high 60 to 10 from low frequency to high

TABLE 6-7: WIDEBAND ANTENNA FOR ESM APPLICATIONS

6.3.5 Summary of Proposed EM Receiving Subsystem Performance Capabilities (Continued)

The sensitivity through the 12 MHz wide channelized output stages that are normally used for ELINT, and through the compressive receiver outputs, that have an effective bandwidth of 25 kHz and are normally used for COMINT, should typically be better than -85 dBm and -100 dBm respectively.

The sensitivity degrades from these values for narrow input pulse signals if the pulse widths are less than the channel filter reciprocal bandwidth, or less than the compressive receiver aperture time. For pulse widths down to 100 nS, this will not be a significant factor for the 12 MHz wide channelized outputs, but would be very significant for the compressive receivers with an aperture time of 60 uS, except that such narrow pulses would never be used in a narrowband COMINT application.

The dynamic range for the channelized outputs and for the compressive receiver outputs typically should be 70 dB or better, and 50 dB, respectively, with the risk of degradation for narrow pulsewidth input signals as noted in the previous paragraph.

The receiver is capable of operating from 0.35 GHz to 40 GHz. The total frequency coverage band is divided into 2 GHz wide sub-bands and then into 150 MHz wide IF channels, which are accessed by means of switch matrices on a time-sharing basis. The frequency resolution is 12 MHz through the channelized outputs and 25 kHz through the compressive receivers. The time of arrival (TOA) resolution is 0.1 uS and 60 uS, respectively, which are adequate for making antenna pattern measurements with an angular resolution of 0.05°, or better.

The estimated total receiver/antenna weight, including redundancy, is 125 kg (275 lbs) and the receiver volume is 71 litres (2.5 cu ft) for the full configuration, and 88 kg (194 lb) and 50 liters (1.8 cu ft) for the reduced version. This does not include allowances for data storage or downlink formatting/transmission. The DC power required with the complete full system operating is estimated at 450 W. By selectively powering on some sections only as required, this can be reduced to 325 W, and by reducing the final stage active redundancy from 3 to 2, this can be reduced to, say 275 W. The estimated maximum DC power for the reduced configuration is 310 W.

TABLE 6-8 CHARACTERISTICS OF PROPOSED PAXSAT EM ANTENNA/RECEIVER SUBSYSTEM

POI	CW	100%
	Pulsed	100%
	Chirped	100%
	Frequency Agile (or MFSK)	100%
Operation With Multiple Simultaneous Inputs		Yes
Sensitivity,	Channelized O/T (ELINT):	-85 dBm. Degrades if $t_p < \frac{1}{B}$ (i.e. 80 nS)
	Compressive Receiver (COMINT):	-100 dBm Degrades if $t_p < T$ (i.e. 60 uS)
Dynamic Range, (typical) above	Channelized O/T:	70 dB. Degrades for narrow pulse signal, as
	Compressive Receiver:	50 dB. Degrades for narrow pulse signal, as above
Frequency Coverage		40 GHz, time-shared in 2 GHz sub-bands/150 MHz channels
Frequency Resolution,	Channelized O/T:	12 MHz
	Compressive Receiver	25 kHz
Time of Arrival (TOA), Resolution	Channelized O/T:	0.1 uS
	Compressive Receiver	60 uS
Weight, Including Redundancies	Full Configuration:	125 kg (275 lbs)
	Reduced Version:	88 kg (194 lbs)
Size, Receiver only	Full Configuration:	71 litres (2.5 cu ft)
	Reduced Version:	50 litres (1.8 cu ft)
DC Power	Full Configuration:	450 W, max.
	Reduced Version:	310 W, max.

7.0 A CONCEPTUAL PAXSAT SPACECRAFT DESIGN

7.1 Introduction

The spacecraft design approach taken in this study has not been so much the design of a spacecraft, but rather the assessment of subsystems from past and current satellite programs and the assembly of these building blocks to determine if the system as a whole can perform the Paxsat mission. Arising from this approach is a concept design as opposed to a design proper whose feasibility can be judged in relation to current and future programs. If the total system appears to be reasonable in terms of contemplated near term future missions, then the Paxsat concept is itself deemed feasible.

The purpose of the Paxsat platform is to carry the sensing devices required to identify the function of a target satellite, to approach the target within the specific stand-off distance, and to provide thermal control, power, and data handling facilities by which data may be collected and transmitted to the ground.

To achieve this, Paxsat must be able to fly in orbits of arbitrary inclination, hour angle and altitude, operate at least to some degree autonomously for avoiding collisions with the target and to downlink the sensed data to interpreters on the ground. If possible, the configuration should not preclude the ability to perform the fly-by investigation. Subsections 7.2, 7.3 and 7.4 address these design issues with presentation of the spacecraft configuration, on-board radar and the Command and Data Handling (C&DH) concept designs.

The design of Paxsat should not rely on militarily sensitive technology and should therefore be built-up from common, commercially available systems. To this end, it was considered that the initial Paxsat should be made modular, modules being taken from other existing spacecraft and modified to suit the Paxsat requirements. A modular design brings the concomitant benefits of ease of spacecraft integration and testing,and the possibility of in-orbit repair, servicing and/or refuelling to extend spacecraft life.

7.1 Introduction (Continued)

Each of the major spacecraft bus subsystem modules on Paxsat are considered in turn. Section 7.5 presents the attitude and orbit control system required for Paxsat. The propulsion module for the Paxsat spacecraft baselining the rendezvous mission operation is considered in section 7.6. Section 7.7 presents a concept design for the power subsystem while section 7.8 discusses thermal control on the spacecraft. Finally. Section 7.9 presents a structure concept and an overall mass budget for the spacecraft.

A summary of the Paxsat spacecraft is presented in section 7.10 where the feasibility of the concept design is concluded.

7.2 Configuration

Figures 7-1 through 7-3 show an exploded view, an on-orbit view and a stowed view of a Paxsat conceptual configuration.

Four fuel tanks holding a total of 3,000 kg of fuel are placed in the center of a cruciform support structure.

The Attitude and Orbit Control Subsystem (AOCS) and Command and Data Handling Subsystem (C&DH) each occupy one of the sides of the cube formed by the exterior of the main support structure.

On the remaining two sides are mounted the solar arrays. The bottom of the cube contains the rest of the power subsystem as well as the interface ring to the launch vehicle.

The top face of the cube is left free for the sensor payload.

In the initial concept, the payload face is attached directly to the main support structure, with only enough clearance allowed to fit the propellant tanks, lines and valves underneath. However, should more payload mounting area be desired, the spacecraft could be stretched to allow equipment to be mounted on the underside of the payload face as well. This would effectively double the payload mounting area.

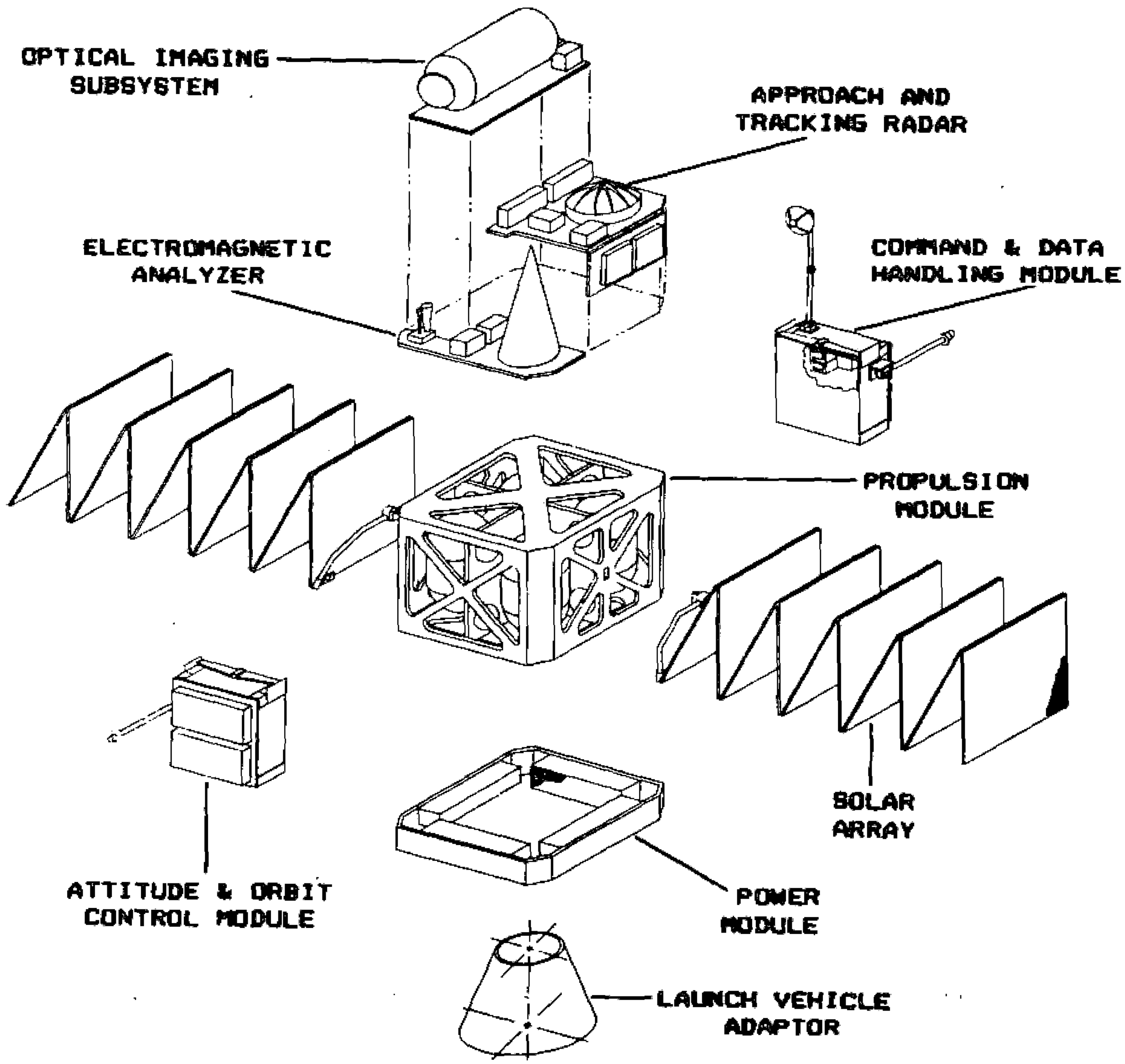


FIGURE 7-1: PAXSAT CONFIGURATION EXPLODED VIEW

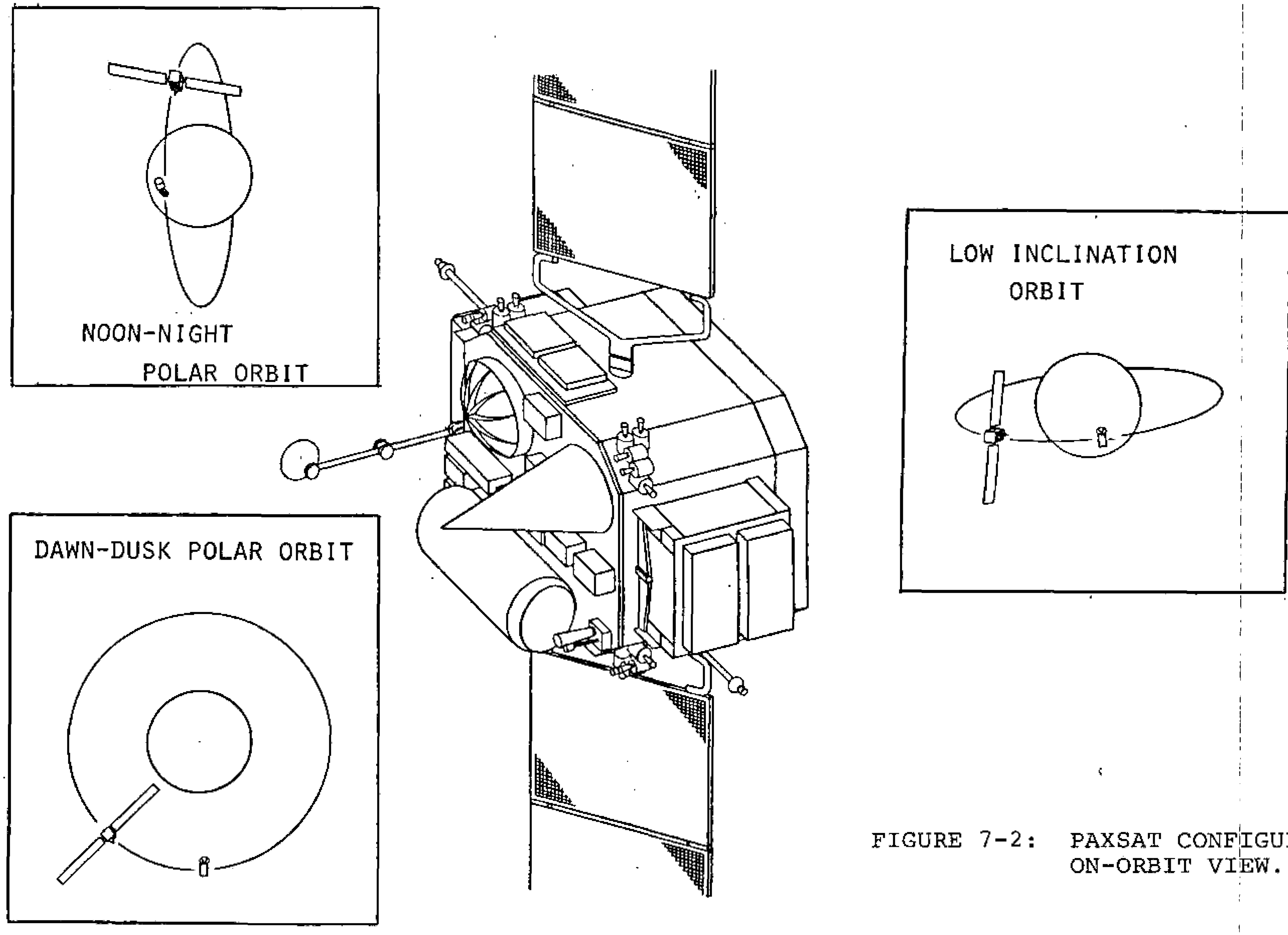


FIGURE 7-2: PAXSAT CONFIGURATION
ON-ORBIT VIEW.

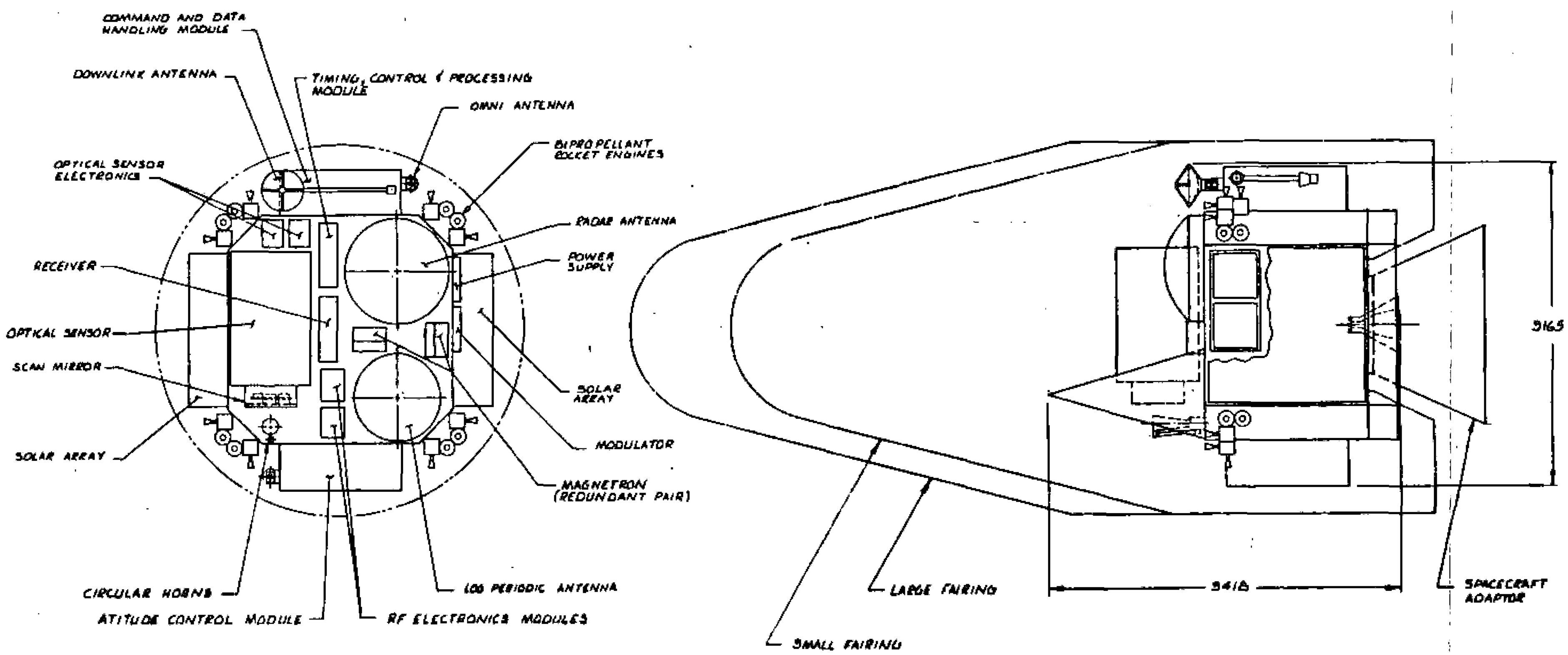


FIGURE 7-3: PAXSAT CONFIGURATION STOWED VIEW

7.2 Configuration (Continued)

Modularization occurs at the subsystem level. Each subsystem (except the structure and thermal subsystems) is housed in its own module which can be removed from the spacecraft with a minimum of effort and replaced by an new module (as would be done, for example, during an in-orbit repair). Figure 7-4 illustrates the Paxsat concept block diagram from which the major subsystems are identified.

7.2.1 Orientation in Flight During Stand-Off Observation

There are three distinct orbits based on the hour angle orbital parameter in which Paxsat must be capable of operating (see Figure 7-2). They are:

(a) Equatorial

(b) Dawn-Dusk

(c) Noon-Midnight

In each case, the following functions must be carried out:

(a) The payload face of Paxsat must be kept pointed at the target.

(b) The solar arrays must be kept as closely as possible to being perpendicular to the sun vector.

(c) The high gain antenna (used for transmitting the data gathered) must be kept pointing at the earth.

(d) Thermal control must be maintained.

(e) The attitude control system must be able to gain sufficient sensed information to operate.

To achieve this, two flying attitudes are seen to be necessary. The first, to be used for equatorial and noon-midnight orbits, orients the solar arrays perpendicular to the orbit plane, and so is referred to as the out-of-plane orientation. The second, called the zenith orientation, is to be used in dawn-dusk orbits

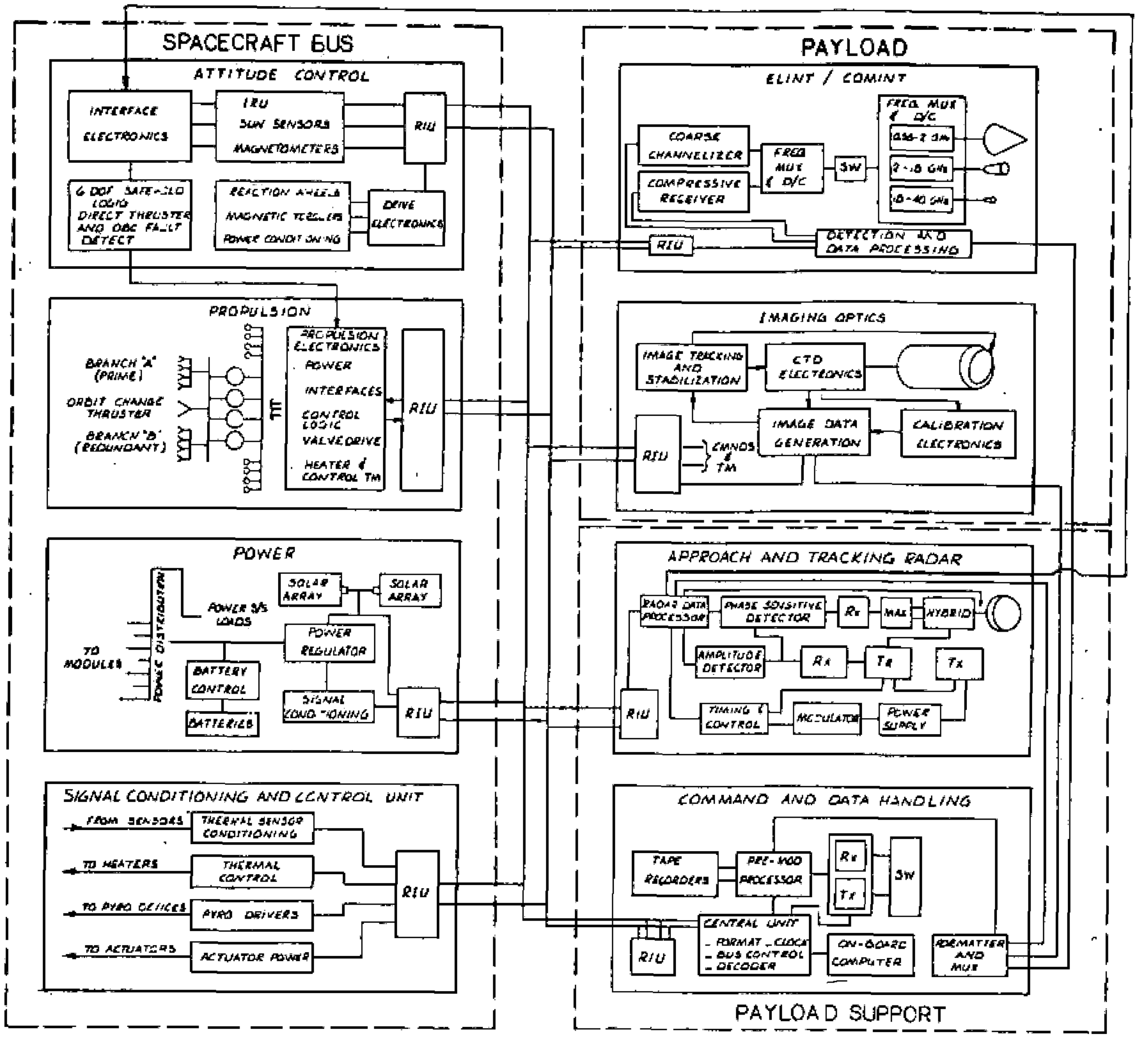


FIGURE 7-4: PAXSAT SUBSYSTEM BLOCK DIAGRAM

7.2.1 Orientation in Flight During Stand-Off Observation (Continued)

and calls for the line formed by the solar array axis to be directed at all times to the center of the earth. A third orientation, which is preferred by many earth observation satellites in dawn-dusk orbits aligns the solar arrays along the direction of flight, was not acceptable because it would not allow the Paxsat payload face to be directed along the flight vector at the target.

7.2.2 Orientation in Flight During Maneuvers

In order to observe the target from all sides, Paxsat requires the capability to circumnavigate the target.

For the purpose of determining Paxsat orientation during such maneuvers, two distinct types of maneuvers were considered.

7.2.2.1 Out-of-Plane Maneuver

For the out-of-plane maneuver, the Paxsat orbit is perturbed slightly so that it is no longer coplanar with that of the target. Paxsat then appears to drift from side to side relative to the target, while staying behind (or in front) of the target and maintaining the same attitude.

This maneuver poses no difficulty for the zenith orientation because the Paxsat body can rotate about the solar array axis to maintain the target in view at the same time as keeping the solar arrays pointed towards the sun.

For the out-of-plane orientation (i.e. with the solar arrays perpendicular to the orbit plane), the solar arrays must also be rotated along with the body if the target is to be kept in view. This implies that solar power input is redduced. Because the sensor heads are conceived as being able to slew to some extent (approximately $\pm 10^{\circ}$), a maneuver in which the aspect angle of the target changed by 35°, would experience a power loss of only 10% while 45° would be available with a 20% power loss, and 55° with a 30% power loss.

7.2.2.1 Out-of-Plane Maneuver (Continued)

Since the maneuver would in any case be cyclic (with the period of the orbit), the average power loss would only be half of the worst-case power loss quoted above.

7.2.2.2 In-Plane Manuever

For the in-plane maneuver, the Paxsat orbit is perturbed to be slightly different in eccentricity (though not in period) than the target orbit. This causes Paxsat to alternately assume a lower altitude and higher velocity than the target so passing underneath it, and a higher attitude and lower velocity than the target so passing over top of it, relative to the earth (i.e. if earth is considered down).

This maneuver is quite different from the out-of-plane maneuver in that Paxsat flies revolutions about the target rather than just swinging from side to side behind or in front of the target. A combination of these maneuvers can, of course, also be performed.

For the in-plane maneuver, the out-of-plane orientation allows the Paxsat body to rotate about the solar array axis to follow the target while maintaining the solar arrays themselves sun pointing.

The zenith orientation demands that the entire Paxsat, solar arrays included, revolve about the orbit normal. In a perfect dawn-dusk orbit, this would involve no reduction in power whatsoever. In nearly dawn-dusk orbits, however, some reductions would take place although they should not exceed 10% to 20%.

7.3 Paxsat Radar Systems

7.3.1. Introduction

This section of the final report is concerned with the role, performance and end-to-end system impacts of radars upon the Paxsat satellite-to-satellite reconnaissance requirements.

The prime role of the radars are limited to acquisition and track of the target satellite, whether from the ground or space segments. The tracking data is

7.3.1. Introduction (Continued)

converted to relative target position and rate for the purpose of vectoring the Paxsat onto the target while maintaining a safe distance between them.

Secondary roles include slaving the Paxsat optical sensor(s) onto the target and measuring target acceleration, which in conjunction with other data may be used to estimate the mass of the target.

Three distinct classes of radar have been examined to determine their usefulness for the Paxsat mission.

(a) Existing ground based systems which are required to estimate the orbital parameters of the target.

(b) Existing space-borne 'docking' radars as used by the USA in Gemini, Apollo and STS (shuttle) programs.

(c) Special purpose space-borne radars for the Paxsat mission.

Radar operation is commonly divided into three distinct phases:

(a) Search

The search phase consists of searching a given volume of space and indicating the presence of targets. Important radar system characteristics during this phase include the volume of space to be searched, the elapsed time between successive searches, the acceptable time between false alarms and the required probability of detection of the target. Some of these parameters may be determined by other system impacts while others are subject to engineering judgements and trade-off. It is usual during this phase to obtain course estimates of the target position.

7.3.1. Introduction (Continued)

(b) Acquisition

The acquisition phase is a transitory phase between search and track. The radar performs a restricted search for designated target(s) detected during the search phase. Tracking loops are initialized. Upon confirmation of the targets presence within the field-of-view, the radar enters a full track mode. Important characteristics during this phase include time allowable for acquisition, target position knowledge, target dynamics and strategy in the event of failure to acquire the target. Because of the transient nature of the acquisition phase, it is difficult to apply analytic techniques to its performance prediction.

(c) Track

The tracking phase is the ultimate result of a successful acquisition. Successive target position and rate measurements are used to obtain accurate estimates of the true target position and dynamics. Within this document, tracking will refer solely to closed loop (amplitude comparison monopulse) tracking whereby the radar boresight is slaved to the target position and internal tracking loops are slaved to the targets range and/or range rate. This ensures high quality tracking by maintaining a high data rate from the target. Alternatives to closed loop tracking such as Track-While-Scan (TWS) are applicable to situations where several targets need to be tracked simultaneously or when the fact that attention is being paid to some particular target is to be disguised. Neither situation is likely in the Paxsat scenario.

During the tracking phase, radars typically may measure any or all of the following target parameters:

(a) Elevation angle

(b) Azimuth angle

(c) Range

7.3.1. Introduction (Continued)

(d) Elevation Angular rate

(e) Azimuth angular rate

(f) Range rate

(g) Acceleration

The accuracy and rate at which these parameters are estimated is subject to engineering trade-offs dependent upon the requirements of other parts of the Paxsat system.

Radar stages may be considered as passive or active. Passive targets are indicated by their skin return only. They may or may not be specifically designed to either enhance or suppress their skin echo. Active targets contain transponders or Secondary Surveillance Radars (SSR's) which react when illuminated by an interrogator. The transponders retransmit a signal, usually on a different frequency which may simply be an echo of the received signal or a more complex waveform identifying the target. Active transponders have found great application in both civil and military aviation, as well as space-space tracking and docking. Long ranges and great accuracy may be achieved with relatively modest equipment. However, for the purpose of this study, it has been assumed that the target does not contain an active transponder. The reasons for this assumption are two-fold, namely:

(a) Problems/objections might be encountered in negotiating a treaty requiring all future satellites to carry such a device. Commercial satellites are highly optimized and it is unlikely that requiring extra 'black boxes' with all their implied spacecraft impacts would be welcome.

(b) In the event of a malfunctioning transponder in the target, then the Paxsat must still be capable of a rendezvous or else the treaty verification process is open to abuse through the deliberate sabotage of the SSR.

7.3.2 Ground Based Radar Systems

This section of the report summarizes the findings on the applicability of current ground based systems to the Paxsat mission.

A number of possible candidates were identified and are listed below:

(a) Millstone Hill

(b) FPQ-6

(c) Cobra-Dane

Table 7-1 lists some of their principle characteristics [48 through 50]. As can be seen from Table 7-1, the technology base spans over two decades of radar development.

The MIT Millstone Hill radar was built during the 1950's. It is a civilian radar specifically designed for the track of extraterrestial objects. The maximum quoted range is 2000 nm against a target with a radar cross section of 1 m^2. Table 7-2 is a more detailed list of the Millstone Hill radar parameters, including tracking accuracy.

The RCA AN/FPQ-6 radar was built during the 1960's and is an upgrade of a previous radar, the AN/FPS-16. The original purpose of the AN/FPQ-6 was to track guided missiles for instrumentation purposes. The quoted maximum rage is 1000 km against a 1 m^2 target. Table 7-3 is a more detailed list of the AN/FPQ-6 radar parameters. This radar was successfully used in the GEOS program [51]. However, this was basically a transponder/SSR experiment. Skin tracking was only possible at the point of closest approach despite the fact the GEOS-II satellites had enhanced skin returns by virtue of carrying a Van Atta array (passive C-band retro-reflector). The AN/FPQ-6 was also fitted with an integrated laser rangefinder, slaved to the radar boresight [52]. Bias errors between the radar and laser of only about 1 m were reported.

TABLE 7-1 PRINCIPLE CHARACTERISTICS OF GROUND BASED RADARS

RADAR	SUPPLIER	ORIGINAL PURPOSE	TECHNOLOGY DATING FROM	MAXIMUM DETECTION RANGE
Millstone Hill	MIT Lincoln Labs	Track of extra terrestial objects	1950's	3700 km (2000 nm) 1 m^2 target
AN/FPQ-6	RCA	Instrumentation Track of guided missiles. Updated version of AN/FPS-16	1960's	1000 km 1 m^2 target
Cobra Dane	Raytheon	Search & Track of missiles + satellites	1970's	1850 km (1000 nm)

TABLE 7-2 MILLSTONE HILL (MIT, LINCOLN) RADAR PARAMETERS

PARAMETER	VALUE
Antenna Size	25.6 m (84')
Antenna Type	Parabolic reflector
Frequency	UHF (440 Mc)
Peak Power	2.5 mW
Average Power	150 kW
Power Source	2 high power Klystrons
Maximum Range	3700 km (1 m^2)
Range Tracking accuracy	8 km
Angle Tracking Accuracy	0.2° (3.5 m rad)

TABLE 7-3 AN/FPQ-6 (RCA) RADAR PARAMETERS

PARAMETER	VALUE
Antenna Size	8.8 m (29')
Antenna Type	Cassegrain or parabolic
Frequency	C-band
Peak Power	3 mW
Average Power	3 kW (?)
Power Source	Magnetron (?)
Maximum Range	1000 km (1 m^2)
Range Tracking accuracy (Maximum)	±2 m
Angle Tracking Accuracy (Maximum)	±15 S of ARC (0.073 m rad)

7.3.2 Ground Based Radar Systems

The Raytheon Cobra Dane (AN/FP) was implemented as part of the USA ballistic early warning system during the 1970'S. Its secondary purpose was the search and track of satellites. Table 7-4 is a more detailed list of its radar parameters. The tracking accuracy quoted in Table 7-4 was estimated by the author. Another (unattributable) source quoted 0.4 km, 50 cm/S and ±0.1° as the typical tracking accuracy obtainable using the NORAD (Northern Radar Air Defense) network.

7.3.2.1 Summary or Ground Based Radars

Skin tracking of passive low earth orbiting satellites has been performed by a number of US systems using technology dating back to the 1950's. Skin tracking of GEO satellites is another proposition with ranges 10 times that quoted for the Millstone Hill radar. (Due to the '4th power' law governing radar range, a factor of 10 in range is equivalent to a factor of 10,000 in power, all other parameters being equal.)

Converting ground based radar accuracy to predicted orbital element accuracy is a complex problem, but it has been addressed in [53], which describes the mathematical basis for a software program known as SEEM (Satellite Ephemeris Error Model) published by Analytic Services Inc, Arlington, Virginia. The model reportedly accommodates drag forces for satellite altitudes above about 180 km, encompassing both sensor measurements and prediction times of up to at least nine hours. The model validity accommodates non-central forces gravitational fields for low altitude satellite passes across as many as three earthbased radars, over somewhat longer measurement and prediction time intervals. Assumptions here are:

(a) Current capability for predicting drag forces

(b) Current understanding of geoid and other gravitational perturbations

(c) No radical radar accuracy improvements beyond current state-of-the-art.

TABLE 7-4 COBRA DANE (RAYTHEON) RADAR PARAMETERS

PARAMETER	VALUE
Antenna Size	29 m
Antenna Type	Phased Array
Frequency	L-band
Peak Power	15.4 mW
Average Power	920 kW
Power Source	96 TWT's
Maximum Range	1850 km (1000 nm)
Tacking Ability	Simultaneous track of >100 targets
Tracking Accuracy (Estimated)	Range < 300 m Angle < 0.1°

7.3.2.1 Summary or Ground Based Radars (Continued)

Example calculations demonstrating SEEM results used a satellite altitude of 400 km and radar accuracies as quoted to in Table 7-5. The resulting ephemeris prediction errors are shown in Figure 7-6, for the case of a single horizon-to-horizon observation of the satellite with unknown bias errors as shown in Table 7-4.

As can be seen from Figure 7-5, (reproduced from [53]), the dominant error is the along track prediction, which reaches a value of some 100 km after 9 hours. This may be substantially reduced by calibrating out bias errors and making multiple observations.

7.3.3 Existing Spaceborne Radar Systems

Three existing spaceborne radars have been identified which perform similar functions to that required of the Paxsat space segment. These are:

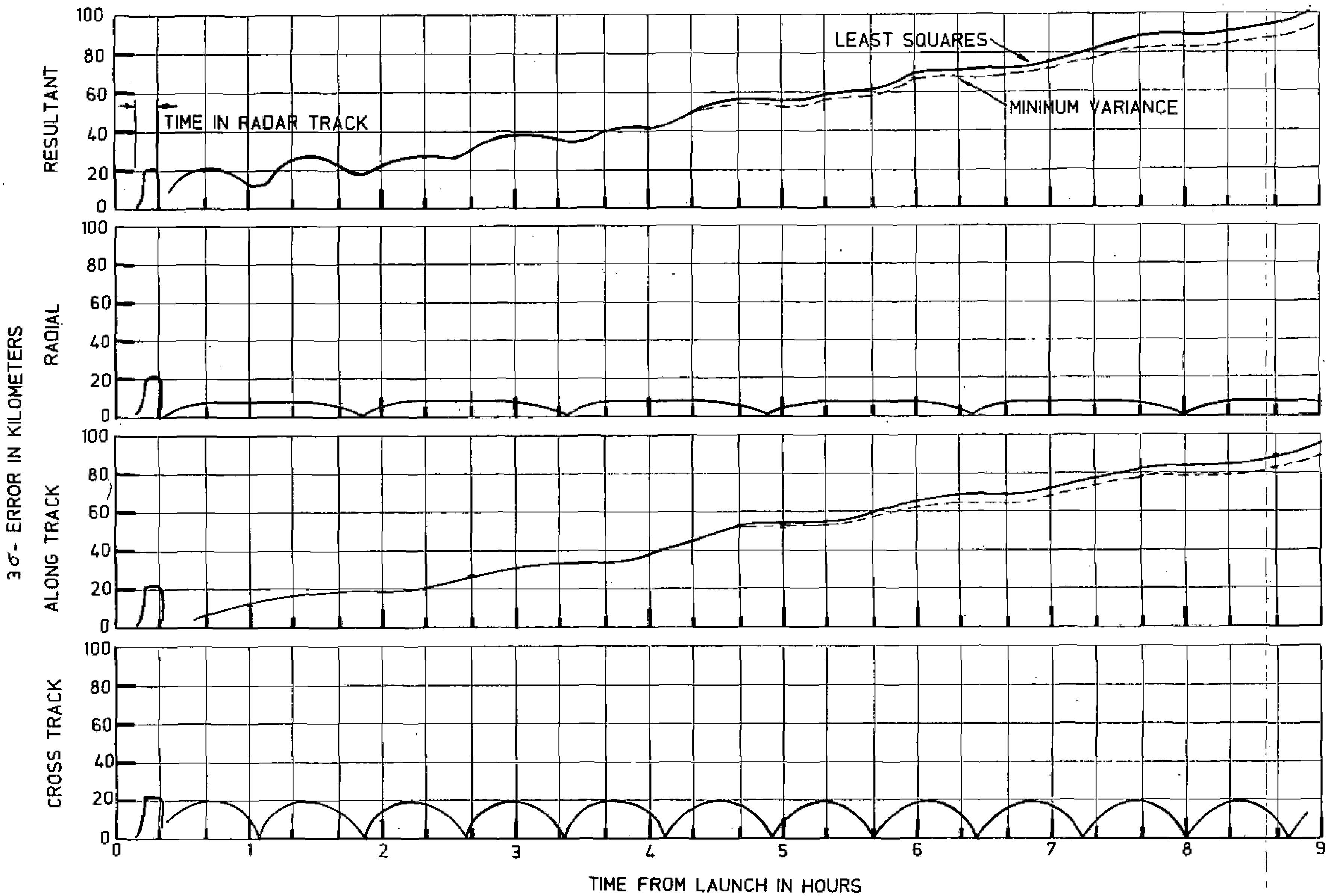
(a) Gemini Docking Radar

(b) Apollo Docking Radar

(c) STS (shuttle) Acquisition and Tracking Radar

Table 7-6 lists the radars with frequency and function. As can be seen from the table, Gemini and Apollo operated in a transponder (SSR) mode only. Table 7-7 gives a more detailed comparative assessment of these two systems [54].

The STS Ku-band system combines both communication and radar system [55,56,57]. The radar system can operate in either a transponder (SSR) or a skin return mode. However, the skin return mode maximum range is 19 nm against a 6.3 m^2 target. This is probably insufficient for the Paxsat mission. Figure 7-6 is a graph of predicted range measurement accuracy for the STS Ku-band radar [56]. It shows predictions by two different companies (Aximatic and Hughes) as well as the specified requirement on the same graph.



[Ref. 53]

FIGURE 7-5: EPHEMERIS ERROR PREDICTION VERSUS TIME

TABLE 7-5 NOMINAL STANDARD DEVIATIONS AND BIAS ERRORS OF A GROUND BASED RADAR AS USED TO CALCULATE ERRORS IN PREDICTED EPHEMERIS DATA [53]

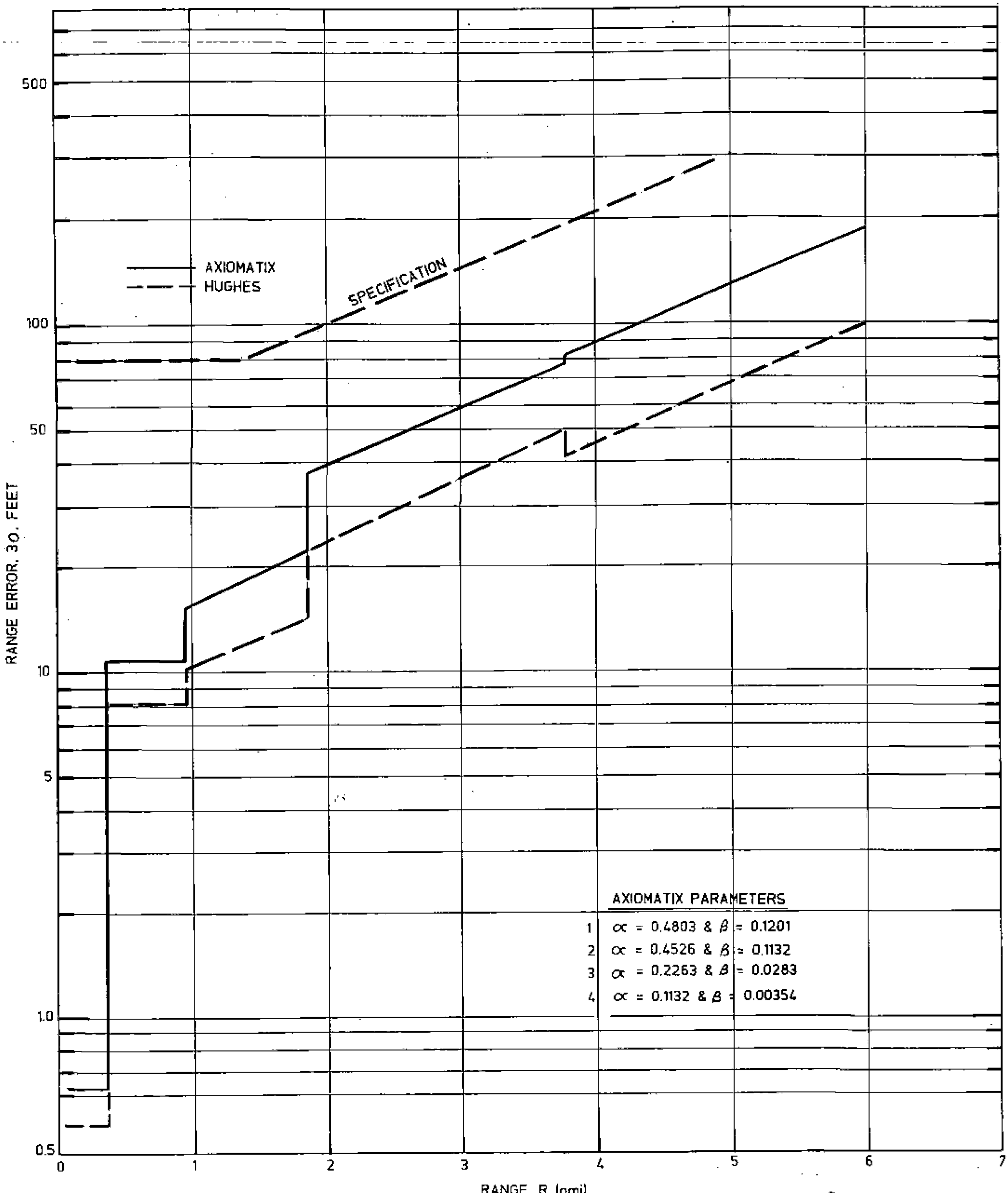
RADAR COORDINATE	NOISE	BIAS
Azimuth	0.05°	0.05°
Elevation	0.05°	0.05°
Range	50 m	50 m

TABLE 7-6 PRINCIPLE CHARACTERISTICS OF CURRENT SPACEBORNE RADARS

RADAR	FREQUENCY	FUNCTION	MODE
Gemini	L-band	Rendezvous	Transponder
Appollo	X-band	Rendezvous	Transponder
STS	Ku-band	Rendezvous	Transponder + Passive

(1) Rely on transponder for long-medium range detection and tracking

(2) STS Limited to 19 nm against 6.3 m^2 passive target.



[Ref. 56]

FIGURE 7-6: STS KU-BAND RADAR TRACKING ACCURACY VERSUS TIME

TABLE 7-7 RADAR PARAMETERS OF GEMINI AND APOLLO TRACKING RADARS (DOCKING)

CHARACTERISTIC	GEMINI	APOLLO
Frequency	L-band	X-band
Range	250 nmi	400 nmi
Range Accuracy	0.1% or 75'	1% or ±80' random
Range Rate	±500 FPS	±4900 FPS
Range Rate ACC	±5% or 1 FPS	1% OR 1 FPS random
Angular Coverage	±25° Pitch & Yaw	±55° Yaw, 225° pitch
Angular Accuracy	8.5 m RAD to 17 m RAD	2 m RAD Random + 8 m RAD Bias

7.3.3.1 Summary on Existing Spaceborne Systems

All past and current spaceborne rendezvous radars have relied on a cooperative target for ranges of greater than a few tens of nautical miles. With transponders ranges of several hundred nautical miles have been achieved with quite modest power and aperture resource demands.

7.3.4 Special Purpose Paxsat Space Segment Radar

Within this section, a number of themes are followed. First, an introduction to some of the radar system trade-offs possible, with a typical trade-off against frequency. Secondly, an example design at C-band is explored, its spacecraft impacts enumerated and possible alternatives discussed. Thirdly, a similar example design at Ka-band is presented, alternative technologies examined and a baseline set chosen for Paxsat against which the mass and power budgets were calculated.

The objectives are to determine the feasibility of a special purpose Paxsat space segment radar bracket performance limitations and assess any particular technological difficulties.

7.3.4.1 System Trade-offs

It is rare that sufficient information and/or constraints are imposed such that a unique radar solution for a given mission presents itself. The Paxsat is no exception and within this subsection an outline of some of the trade-offs is presented. In later sections, performance is always quoted against certain constraints. These constraints may not always be rigid and this section outlines the effects of these constraints upon the radar performance and resource demands.

A table of various radar system parameters, which may determine either the performance or the design, is included as Table 7-8. The parameters have been divided into three columns; target parameters, system requirements and radar parameters.

TABLE 7-8 GENERIC RADAR SYSTEM PARAMETERS/TRADE-OFFS

TARGET PARAMETERS	SYSTEM REQUIREMENTS	RADAR PARAMETERS
Relative Position	Search Volume	Frequency
Size	Detection rapidity and probability	Aperture
Dynamics	Tracking parameters	RF power
Scintillation	Tracking accuracy	Pulse repetition frequency
Uncertainty	Time between false alarms	Pulse length
	Resource demands	Losses/Noise
		Technology

7.3.4.1 System Trade-offs (Continued)

Target parameters include radiometric as well as spatial parameters. The targets radiometric size and its scintillation properties are beyond the control of the Paxsat designer. Reference [58] lists some typical early communication satellite radar cross sections, calculated by RCA from a project known as TRADEX. No data has been discovered on satellite scintillation or glint. In any case, radiometric properties can be a sensitive function of frequency. For the purpose of comparison in this study, targets were assumed to have a constant (swirling type '0') 5 m^2 cross section. Spatial target characteristics include the targets true relative position and dynamics as well as the uncertainty in target position. These parameters will drive system parameters such as search volume, etc.

System requirements include the data required from the radar and the envelope of resources available to the radar. In any final design, these will be subject to a number of trade-offs viz-a-viz optimizing the entire satellite sensor platform with all its various demands and facilities.

Radar parameters characterize and are chosen so as to meet the system requirements for all targets of interest. Included under radar parameter is the choice of technology for implementing the radar. In practice, availability, cost and reliability of technology also steer the radar design towards preferred configurations and limit acheivable system performance.

One of the most fundamental choices of radar parameter is the operating frequency.

Two different constraints can affect the results of a frequency trade-off, constraints on beamwidth and constraints on aperture.

If a particular beamwidth is required, then as frequency increases so does the required power. If a particular aperture is available, then as frequency increases, power decreases.

TABLE 7-9 EXAMPLE RADAR FREQUENCY TRADE-OFFS FOR FIXED BEAMWIDTH

CM	GHZ	BAND	ANTENNA SIZE (M)	ANTENNA MASS[2] (kg)	PEAK POWER (kW)	AVERAGE POWER (W)	S/C POWER[1] (W)
10	3	S	14.3	2200	0.10	2.0	7
5	6	C	7.16	550	0.39	7.8	26
2	15	Ku	2.86	88	2.44	48.8	163
1	30	Ka	1.43	22	10.0	200	700
0.5	60	MM	0.716	5.5	39.0	781	2.6 K
0.2	150		0.286	0.88	244	4.88 K	16.3 K

(1) DC to RF efficiency = 30%
(2) Antenna Density = 10.7 kg x m^{-2}

7.3.4.1 System Trade-offs (Continued)

Table 7-9 shows the result of a typical frequency trade-off for constant beamwidth and a particular set of requirements on maximum range, tracking accuracy, etc.

7.3.4.2 Example C-Band Designs

One of the most critical technologies for any radar is the high power amplifier HPA (RF) generator.

C-band was chosen for the initial calculations because of the accessibility of space qualified technology in this frequency band. The US, Canada and the Europeans are engaged in programs to develop space qualified C-band HPA's.

The Canadian program calls for a 5.3 GHz (C-band) amplifier with mean power capability of 500 W and a peak power capability of 10 kW. The HPA is being developed for the Radarsat program which is scheduled for a 1990 launch. The Europeans are developing a slightly lower power (300 W) device for the ERS-1 (Earth Resources Satellite) program. At least one American company is working on solid-state C-band amplifiers for the NASA SIR (Shuttle imaging radar) program.

Operating the proposed Canadian HPA at its peak and mean power limits (and assuming they could be achieved simultaneously with a rather diffirent pulse length than the that proposed for Radarsat), it was found that an antenna diameter of about 2.5 m was required to obtain 200 km range. The relevant radar parameters used for this example option are listed in Table 7-10.

This example design had a number of major drawbacks in its resource deamnds upon the satellite.

(a) The power drain upon the satellite was very high.

(b) The HPA itself is very heavy (approximately 80 kg).

(c) The aperture of 2.5 m by 2.5 m was considered too large.

The following paragraphs take each of these objections in turn and indicate what may be done to overcome them.

7.3.4.2 Example C-Band Designs (Continued)

(a) High Power Drain

The high powers are required to obtain the fairly long range of 200 km. It is probable that this figure is too high. Even if it becomes desirable to range out to 200 km, it is unlikely that continuous ranging would be required, i.e. the radar need only be on for some fraction of the time at furthest range.

At shorter ranges, the power required drops rapidly until at about 60 km only one one-hunredth of the mean power is necessary, that is, about 5 W RF. In principle this can be achieved by transmitting short pulses only, and maintaining the peak power.

(b) Mass of HPA

Restricting the maximum rage of the radar to about 60 km reduces the RF output power to about 5 W. In this case, solid-state power amplifiers (SSPA) become feasible. A single 5 W SSPA would weigh only in the region of 5 kg. However, due to their peak power limitations, they would only work with relatively long pulses, implying the need for pulse compression.

(c) Aperture Size

For reasons of spacecraft impacts, it was decided that an aperture of about 1 m diameter was the maximum. This implies an increase by a factor of over 30 in the required mean power. Some trade-off is allowable between trading off increases in the peak power and increases in the pulse length. Increasing the peak power will push the system towards the multipaction zone. Increasing the pulse length degrades range resolution and hence tracking accuracy. An alternative approach is to install pulse compression. At some increase in mass and cost, pulse compression can give the range resolution of a short pulse system with the detection capability of a long pulse system.

TABLE 7-10 EXAMPLE C-BAND DESIGN RADAR PARAMETERS

PARAMETER	VALUE
Frequency	5.3 GHz
Aperture	2.5 m diameter
RF Power	500 W
DC Power	1.7 kW
Mass	178 kg
Range	200 km
Range Tracking Accuracy (Thermal)	400 m (max. range 1 Hz rate)
Angle Tracking Accuracy (Thermal)	0.19° (max. range 1 Hz rate)

7.3.4.2 Example C-Band Designs (Continued)

A major drawback is the degradation in angle tracking ability with the decreased aperture. It was calculated that only 0.5° rms angle tracking accuracy could be achieved (at a 1 S update rate) with the example design.

Combining SSPAs with a 1 m aperture and realistic pulse compression capabilities probably would not give good enough range tracking ability either, so a heavier HPA technology would be required.

For these reasons, higher frequency designs were also examined.

7.3.4.3 Example Ka-Band Designs

Holding the aperture constant, the peak power required from a radar falls with increasing frequency, everything else being equal.

A number of different HPA technologies are available at millimeter wavelengths [Ref. 59, 60, 61]. For these example system designs, coaxial magnetrons [Ref. 62] were chosen. Magnetrons are self oscillators and hence do not require an input from a low power transmitter chain. Operating in that mode, magnetrons are completely incoherent from pulse-to-pulse, i.e. there is no deterministic relationship between the phases of successive pulses. In general, magnetrons are chosen where small size and portability are more important than stability and high mean power [Ref.48].

Such devices have been identified in the catalogues [63 & 64] with peak powers varying from 20 kW to 135 kW at Ka-band.

The standard waveguide for this band of frequencies is WR28. Calculations indicate that in vacuum, WR28 will experience multipaction at about 6 kW peak power at 35 GHz. Hence, to realistically use such devices implies pressurizing the entire high power RF circuitry or filling with dielectric which would probably induce too high a loss.

7.3.4.3 Example Ka-Band Designs (Continued)

The alternatives are:

(a) To use a pulse compression system which means disregarding the magnetron in favor of a more massive and complicated transmitter such as Klystron and TWTA based subsystems with a low power transmit chain and a SAW expansion/ compression system incorporated.

(b) Reduce range and range accuracy requirements.

Table 7-11 contains a list of radar parameters and tracking accuracies for an example 35 GHz design with a magnetron transmitter and pressurized high power microwave circuitry.

Antennas may be divided into three broad categories:

(a) arrays

(b) Reflectors

(c) Hybrids

Arrays have not received much attention in this study. The antenna is relatively large, measured in wavelengths and microstrip is very 'lossy' at these frequencies.

Millimeter reflector antenna technology is discussed in References [65] and [66]. The flat plate cassegrain is a mechanically steerable antenna with low inertia moving parts, wide fields of view and no moving waveguide joints. These qualities led to it being selected for the example design.

A possible disadvantage results from its polarization sensitivity if square waveguide was to be used for multipaction purposes.

An interesting millimeter hybrid concept is described in Ref. [67]. Hybrids offer the relatively simple feeding system of the reflector with the electrical scanning properties of a plannar array. Such advanced concepts would need considerably more research before definitive decisions could be made on their applicability to Paxsat.

TABLE 7-11 RADAR PARAMETERS FOR KA-BAND 60 KM RANGE RADAR

PARAMETER	VALUE
Frequency	35 GHz
Aperture	1 m diameter
RF Power	20 W
DC Power	280 W
Mass	80.5 kg
Range (5 m^2 target)	60 km
Range Tracking accuracy (Thermal)	4 m RMS
Angle Tracking Accuracy (Thermal)	0.05° RMS

7.3.4.4 Technological Choices/Alternatives

Table 7-12 summarizes possible alternative technologies used in radars. It is divided into three columns representing the antenna, the transmitter and the receiver/signal processor. Technology chosen for the example design is set in heavier type and underlined. thus we have a system with a 1 m diameter mechanically scanned reflector (flat plate Cassegrain) with a crossed field amplifier of the coaxial magnetron type. The radar processing is incoherent and angular information is derived from monopulse sum and difference signals.

The configuration of the radar upon the sensor platform is illustrated in Figure 7-7, with all the various assemblies constituting the radar indicated separately. The two orthogonal diffference channels are multiplexed pulse-pulse (or burst-burst) down a single channel.

A single redundant receiver was assumed, which may be used for either the sum of difference channel in case of failure. A single redundant tube has also been assumed.

The radar has an estimated mass of 83 kg and a DC power requirement of 280 W.

7.3.4.5 Summary on Paxsat Space Segment Radar

All existing space-space radar systems depend upon cooperative target transponders for ranges greater than a few tens of a kilometer.

Ranges of several hundres of kilometers will require powerful high powered amplifiers and large antennas. The radar technology is feasible but the impacts upon the size, mass and maneuverability of the satellite are highly undesirable.

The final configuration chosen was designed to give several tens of a kilometer range with an aperture limited to 1 m diameter. To keep the mass down, magnetrons were chosen as the RF source. The cost is a high peak power and the need to pressurize much of the high power circuitry.

7-36

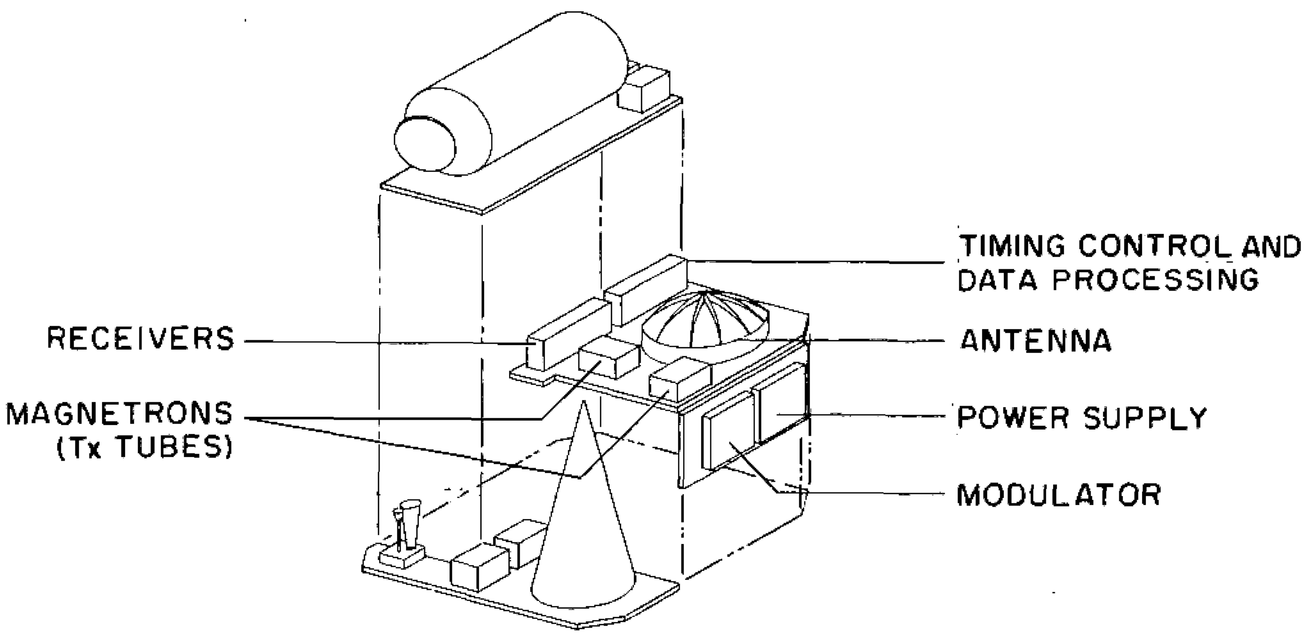


FIGURE 7-7: EXPLODED VIEW OF KU-BAND RADAR ACCOMMODATION

TABLE 7-12 TECHNOLOGY ALTERNATIVES/CHOICES

ANTENNAS	TRANSMITTERS	RECEIVERS/PROCESSORS
Electrically Scanned	Crossed Field	Coherent
Mechanically Scanned	Linear Beam	Incoherent
Arrays	Solid-State	LNA
Reflectors	Coaxial Magnetron	Pulse Compression
Hybrids		Monopulse
Flat Plate Cassegrain		Conical
		Track While Scan

7.4 Command and Data Handling (C&DH)

The concept for command and data handling is based on the Communications and Data Handling module of the Fairchild Multi-Mission Spacecraft (MMS) design.

It contains the spacecraft master computer which implements the attitude control and homing laws, handles all telemetry and telecommand functions and routes the data generated by the payload sensors either to the tape recorders or directly to the ground stations.

The C&DH also includes a communications transponder and its associated antenna which provide the link between the spacecraft and the ground station. A half meter aperture high gain antenna mounted on a 2-axis gimbal provides the primary high rate data link through which information passes to and from the satellite. The gimbal system allows antenna repositioning so that the link is maintained during Paxsat maneuvers and in different flight attitudes.

A second low rate data link is provided through two omni-directional antennas which give 4 Str (whole hemisphere) coverage, so that contact with the satellite can be maintained even if the high gain antenna is inactive.

Two tape recorders are provided for data storage. Initially, it was considered that a record rate of 2 Mb/S and a total storage of 10^9 bits would be sufficient. However, the record/playback frequency and the data storage requirements will need further analysis in order that a storage system with the required capacity and reliability can be configured.

Alternatives to on-board storage exists. One is the provision of a link through other satellites (as in TDRSS, the NASA tracking and data relay system) in which Paxsat would relay information continuously to stations not visible directly via another satellite or satellites to which both Paxsat and the station are visible. Another possibility is to provide data relay stations spread throughout a large number of countries which would record downlinked Paxsat data and then relay it to the main processing center again via a communications satellite. Of course, a combination of on-board storage

7.4 Command and Data Handling (C&DH)(Continued)

and data relay may be optimal from the point of view of reliability and the capability to provide a graceful degradation of performance in the face of equipment failures.

Finally, the C&DH will be required to provide a measure of satellite autonomy through the on-board computer. For example, the satellite will need to operate in a safe manner while unsupervised by ground control. Also, some data reduction will probably need to be made on-board the satellite to reduce the volume of data needed to transmit the desired information.

A preliminary assessment of the computer hardware requirements was performed for the purpose of mass and power estimation, and is included in Appendix D.

7.5 Attitude and Orbit Control

7.5.1 Requirements

The Attitude and Orbit Control System (AOCS) maintains the spacecraft in the commanded orientation. During periods of time when the spacecraft is not investigating a target, this orientation is not critical, the only requirement being that sufficient sunlight falls on the solar arrays to provide needed electrical power and that a low data rate link to the ground be available. In this respect, Paxsat is more typical of some early science satellites than of modern earth resources, communications or scientific missions.

During an investigation though, the Paxsat attitude control system must also maintain its payload face pointed roughly at the target (in the current conception, to within 10^{o}). This means that the AOCS must take information from a sensor which identify the angular displacement of the target from the payload face boresight. This sensor is nominally the radar with back-up being provided by the optical sensor.

Whereas the EM Receiver system is not very sensitive to angular rates in the satellite body, the optical imaging system is due to possible image blurring during lengthy exposure times. There exists therefore a trade-off between image stabilization within the optical sensor

7.5.1 Requirements (Continued)

itself and satellite body stabilization. In addition, the optical sensor stability requirement itself depends on the sophistication of the image data processing. The present conception assumes an image motion compensation system within the camera itself providing a tracking capability for use when the radar is inoperative.

7.5.2 Implementation of Control Laws

Measurements of range to the target, range rate and azimuth and elevation angles and rates are used by the attitude and orbit control algorithms to maintain Paxsat at a desired distance from the target and in the correct attitude.

The calculations are carried out by the spacecraft computer. As a back-up, when the computer is inoperative, hardwired logic within the attitude and orbit control system will maintain the Paxsat at a safe distance from the target and will execute avoidance maneuvers should the target maneuver towards Paxsat. To this end, radar data is provided directly to the attitude and orbit control subsystem (AOCS) independently of the computer-driven digital network.

The hardwired logic contains the circuitry required to detect faults in the on-board computer not diagnosed by the computer itself and to switch over to the back-up system.

7.5.3 Attitude Sensors

During an interrogation, Paxsat attitude is driven by the radar, the optical sensor and sun sensors. The information from the sun sensor is used to ensure that the solar arrays are aligned to the sun. As such, they do not need to be extremely accurate.

The tightest attitude control requirement comes during the period of time while Paxsat is scanning to acquire the target from a distance of 50 km to 100 km but before the target is found. The smaller the angular extent of the search region, the easier the search. Information will likely be available which pinpoints the target to

7.5.3 Attitude Sensors (Continued)

virtually any arbitrary accuracy. The largest angular error terms will come from errors in orbit determination, timing, and the angular pointing error of Paxsat itself.

In order to minimize this, it was considered that Paxsat should be equipped with a high accuracy inertial measurement unit (IMU), which would measure Paxsat attitude with respect to the stars. Not only would this system give the highest available performance, it is also virtually independent of the satellite orbit.

Such an attitude measurement system is similar to one used by the MMS platform. Whether or not additional sensors are required for back-up control depends on the reliability and the up-time achievable for the computer. At most, some earth sensors might be required for back-up when Paxsat is not investigating a target. For the purpose of the current study, an inertial measurement unit combined with sun sensors would provide adequate, reliable performance, in that a sun facing attitude serves as back-up in case of IMU inoperability during loitering (i.e. during a period in which no investigation is being performed).

7.5.4 Actuators

Attitude is controlled primarily through the use of reaction wheels. When these saturate, external torques are generated by magnetic torquing coils which react against the earth's magnetic field. Magnetometers are used to measure the orientation of that field so that the magnetic torquers can be used to full advantage.

One advantage of this system is that it allows the spacecraft platform to be kept very stable as opposed to the performance of a system which uses thrusters to de-saturate the reaction wheels.

The logic required to drive the magnetic torquers could be either hardwired or implemented through software. For the present concept, a software implementation was used.

7.5.4 Actuators (Continued)

When the back-up attitude and orbit control system is activated, the reaction wheel/magnetic torquer system is bypassed and control is performed through the thruster system.

Control during major orbit change burns is also performed through the thrusters, though the logic is implemented in the on-board computer.

7.6 Propulsion

The propulsion subsystem carries 3,000 kg of bi-propellant fuel (Monomethyl Hydrazine, Nitrogen Tetroxide) in four tanks. These tanks feed one high thrust (100 lbf) high efficiency (310 Isp) thruster and twenty low thrust (5 lbf) lower efficiency (285 Isp) thrusters.

Propellant expulsion is performed by Helium pressurant in eight tanks. The system is pressure regulated throughout the life of the spacecraft.

The thrusters are positioned to allow all required attitude control, homing and evasive maneuvers. Their configuration is such that only firing the main (high thrust) engine causes Paxsat to accelerate directly towards the target, and so the likelihood of accidental collision is minimized.

The entire propulsion subsystem is conceived at present to be integrated into the main support structure of the satellite. Access room is allowed to permit refuelling and repressurization in space should this be desired. A concept that has not been pursued, but which could be developed, would allow the propulsion subsystem to be easily detachable as a unit from the rest of the spacecraft, making it replacable as a module.

7.7 Power Subsystem

7.7.1 Requirements

The power subsystem on-board a spacecraft generates and distributes the power necessary for the spacecraft to function throughout its entire life from launch until the end of its mission life (EOL). The power subsystem

7.7.1 Requirements (Continued)

provides the power required by the payload instruments and sufficient power for housekeeping services required by the spacecraft bus subsystems to run the spacecraft, and to charge the batteries during sunlight operations for subsequent employment during eclipse operations.

The power subsystem required by a Paxsat spacecraft unlike modern earth resources, communications or scientific satellites needs to be able to operate in a variety of orbits ranging from LEO to GEO. Not only must the power subsystem maintain a daily energy balance between eclipse and sunlight periods in low earth orbits, it must also supply a sufficient power margin for the spacecraft to operate at EOL after experiencing its mission years in the harsh radiation environments of the higher GEO and Molniya type orbits.

Additionally, the power subsystem aboard Paxsat must be able to generate sufficient power to operate in orbits whose hour angles do not permit maximum power output for its configured solar arrays. A combination of unfavorable hour angles, high orbit inclinations and seasons of the year can rapidly reduce the power output from a configured solar array. This factor is of great concern in any power subsystem design and its effects are aided by defining the angle between the normal of the solar array's area to the sun itself, as the solar aspect angle. Since a solar array produces its maximum power when the sunlight is perpendicular to the surface of the array, the design philosophy attached to the design of solar array configurations is to minimize the solar aspect angle.

A power subsystem concept for Paxsat is presented in the following sections.

7.7.2 Generation

Power for spaceborne applications can be generated in a variety of fashions ranging from Radio-isotope Thermal Generators (RTG's) through solar cells to exotic nuclear reactors. Most satellites in the past have employed the solar cell technology to generate electricity from incident light energy. The result of this activity is a mature and reliable technology upon which to base future missions. Each power generating

7.7.2 Generation (Continued)

technology has a characteristic power level capability with solar cells supplying the demand for most applications. Thus, it is not surprising that the Paxsat spacecraft which requires 2.0 kW of EOL power to utilize solar cell technology for its power generation requirements.

The power budget for the Paxsat concept spacecraft is presented in Table 7-13. Of the 2,000 W EOL requirement, 410 W is to be supplied to the payload as determined in section 6.0 of this report. 1,590 W of power is required for the bus and payload support subsystems including the power necessary to charge the batteries for LEO operation. This power budget represents the end of life operations requirement for the spacecraft. The solar array will need to generate more than this requirement to counteract the radiation and the solar aspect angle losses.

The characteristics of a solar cell to be used on a future low earth orbit satellite is given in Table 7-14. This cell is typical of those used on satellites in low earth orbits. Table 7-15 illustrates the radiation factors which will affect the performance of a solar cell after flying a 5 year mission in the radiation environment at an altitude of 1,000 km. The total term implies that a solar cell's output power will diminish to 69% of its initial capability after a 5 year exposure to the radiation at a 1,000 km altitude including other cell losses. In the Paxsat concept, a similar efficiency was assumed to account for the radiation environment. This factor underestimates the radiation degradation for GEO and Molniya orbits and consequently overestimates the performance of the solar array for these orbits. However, in the case of the GEO orbit, the maximum solar aspect angle will be less than that required for LEO operation and thus Paxsat will not experience any degradation in its performance. In the Molniya orbit, however, Paxsat will experience some limitations as it nears end of life. However, time-sharing optical and ESM receiver payload operations should offset this limitation. Increasing the size of the solar array to account for the increased

TABLE 7-13 PAXSAT END-OF-LIFE POWER BUDGET

PAYLOAD		
Optics	100	
EM Analyzer	310	
		410 W
PAYLOAD SUPPORT		
Radar	280	
Command & Data Handling	235	
		515 W
SPACECRAFT BUS		
Attitude & Orbit Control	142	
Thermal Control	100	
Power Electronics, Battery Charging & Losses	833	
		1,075 W
TOTAL (EOL)		2,000 W

TABLE 7-14 TYPICAL SOLAR CELL CHARACTERISTICS

Cell Type	1 ohm-cm BSR
Dimensions	20 mm x 40 mm
Thickness	180 microseconds
Current per cell	0.308 A at the operating point
Voltage per cell	0.34 V at the operating point

TABLE 7-15 SOLAR ARRAY LOSS FACTORS

LOSS FACTORS	YEARS 0	1	2	3	4	5
Calibration Error	0.98	0.98	0.98	0.98	0.98	0.98
Cell Mismatch	1.0	0.998	0.996	0.994	0.992	0.99
Micrometeors & Ultra-violet	1.0	0.995	0.990	0.985	0.980	0.975
Wiring Loss	0.94	0.94	0.94	0.94	0.94	0.94
Radiation Damage	1.0	0.896	0.855	0.82	0.798	0.778
TOTAL	0.921	0.820	0.777	0.739	0.715	0.692

Data from reference [69]

7.7.2 Generation (Continued)

radiation degradation is a factor that can be undertaken in further study. Suffice it to say that the solar radiation degradation factor of 69% is a reasonable concept estimate.

A second factor which the solar array design must include is the solar aspect angle. An analysis performed using the techniques available in Reference [68] demonstrate that the worst case solar aspect angle would be no greater than 50°. A solar aspect angle of 50° requires an increased solar array area by a factor of 1/(cos 50) or 1.5625. The worst case solar aspect angle is maintained at 50° by employing rotating solar arrays in two alternate orientations defined in section 7.2 as the out-of-plane orientation and the zenith flight orientation. A third solar array orientation, the in-plane velocity orientation was prohibited by a need to view the target spacecraft without obstruction. By assuming the zenith and out-of-plane orientations, the solar aspect factor can be limited to a 64% rated of power. When the Paxsat orbit inclination is high the hour angle such that the orbit normal points in the direction of the sun (as in a polar dawn/dusk orbit) the zenith flight orientation is preferred. Conversely, if the inclination is low or high and the hour angle is such that the orbit normal is perpendicular to the direction of the sun (noon/midnight), the out-of-plane flight orientation is preferred.

Since the radiation degradation and solar aspect angle factors are multiplicative in nature, the solar array must be oversized by 2.25 times that of the EOL requirements. Thus, the Paxsat solar array is configured to provide a maximum of 4,500 W at Beginning of Life (BOL).

To provide 4,500 W of power and using the solar cell defined above, 45 square meters of array area is required. The Paxsat concept array is divided into two rotating wings each having the dimensions of 1.5 m in width and 15 m in length. The array is of a rigid panel construction employing composite honeycomb materials to increase the structural stiffness over that of a similarly powered flexible array, and thereby avoid potential interactions with the attitude control subsystem.

7.7.2 Generation (Continued)

Each wing is divided into ten panels to enable stowage on the sides of the Paxsat configuration during launch. With such a solar array configuration, weighing but 167 kg, the Paxsat is able to provide power for all envisioned missions of the Paxsat spacecraft and allow compatibility with the Ariane IV and STS launch vehicles.

7.7.3 Storage

Batteries are typically employed on spacecraft to enable operation when the earth eclipses the sun. Batteries enable energy to be stored when the spacecraft is in the sun, and released for operations during eclipse periods.

There are two major kinds of batteries employed in the spacecraft industry today. The first type is Nickel Cadmium (NiCd) and the second type is Nickel Hydrogen (NiH_2). The NiH_2 batteries are relatively new developments having a much higher energy density by mass than NiCd batteries. In addition, NiH_2 batteries enable higher depth of discharge and charging rates than NiCd batteries, but because they are new developments, the lifetime of these batteries is unknown. Future applications however are expected to utilize NiH_2 batteries exclusively. In the Paxsat concept, the more robust NiH_2 batteries are employed.

The battery concept for the Paxsat spacecraft is driven by the low earth orbit regime. This domain places the most demands on a battery system design. In particular, a 90 minute period orbit was selected as being the worst orbit in which Paxsat would need to operate. In a 90 minute orbit, 36 minutes are spent in eclipse leaving only 54 minutes in each orbit to charge the batteries sufficiently for the next eclispe period. This cycle is repeated sixteen times a day.

Battery and solar array designs for spacecraft are highly coupled based on the need to maintain a daily energy balance. If a balance is not at least maintained during each day, the mission will soon come to an end as the batteries continue to be charged to a lower and lower state, until the spacecraft cannot provide enough power to maintain itself during eclipse. Conversely, if

7.7.3 Storage (Continued)

more energy is generated than required, the spacecraft is overdesigned which may equate to an associated mass penalty. The power subsystem is modelled using efficiencies to define characteristics of the system. Figure 7-8 illustrates the Paxsat power subsystem model.

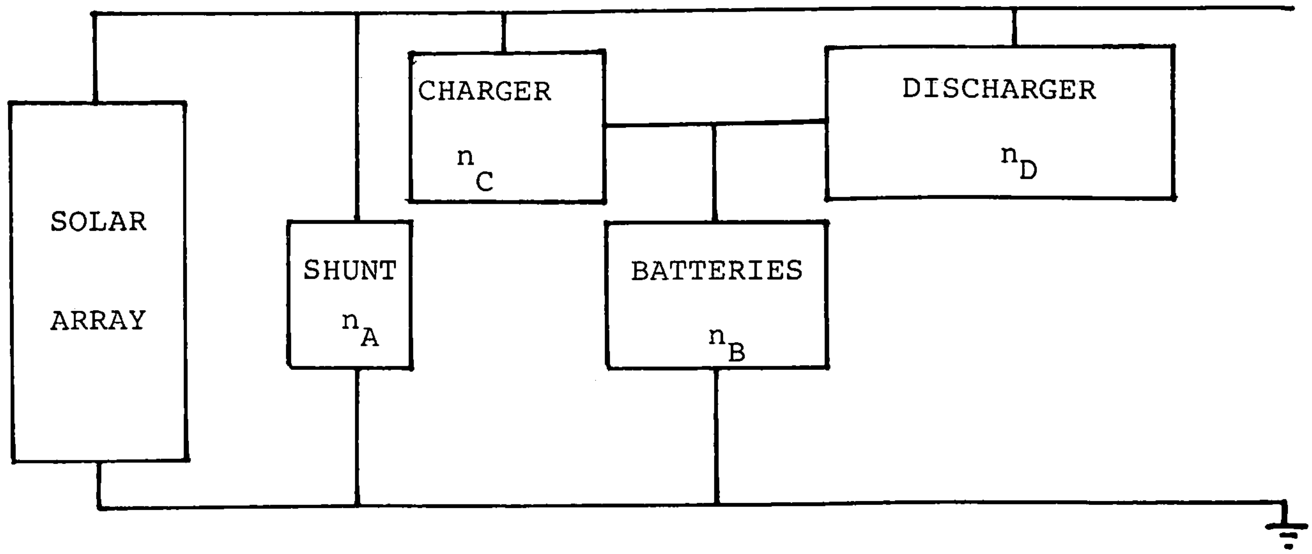
In the Paxsat concept, an end-of-life power generating capability of 2,000 W provides sufficient power to charge two 22 cell, 50 AHr NiH_2 batteries and provide three 12 minute operations of the payload during eclipse per day and full operations during sunlight. The battery DOD never exceeds a recommended 40% for lifetime considerations. The charge rate for the batteries never exceeds C/1. NiH_2 batteries are now being tested at C/1 and C/2 rates for 90 minute orbits and current thinking does not foresee any difficulties with such rates.

Thus, two 22 cell 50 AHr NiH_2 batteries weighing a total of 33 kg enables the Paxsat spacecraft to monitor a target spacecraft for one complete eclipse period a day and all sunlight periods in a 90 minute low earth orbit. At geosynchronous where eclipses last a maximum of 72 minutes, the Paxsat batteries are able to give full eclipse operation without exceeding an 80% DOD.

7.7.4 Distribution

The distribution and the power electronics on-board the Paxsat spacecraft are based on the power module of the MMS spacecraft. At a weight of 52 kg, this power subsystem offers a maximum power tracking capability over a 28 V nominal unregulated power bus. This system requires power conditioners specific to the needs of each spacecraft subsystem and payload element. This is the usual design feature of a modular spacecraft and since Paxsat is of a modular design, the feature is employed on the Paxsat spacecraft.

The MMS power module can accept a maximum input power of 3,000 W. Since the Paxsat spacecraft can generate



GENERALIZED POWER SUBSYSTEM

$$n_A \quad \frac{\text{POWER FROM ARRAY}}{\text{POWER TO BUS}} = 0.94$$

$$n_B \quad \frac{\text{BATTERY CHARGE ENERGY}}{\text{BATTERY DISCHARGE ENERGY}} = 0.72$$

$$n_C \quad \frac{\text{POWER INTO BATTERIES}}{\text{POWER INTO CHARGER}} = 0.78$$

$$n_D \quad \frac{\text{POWER FROM BATTERIES}}{\text{POWER TO BUS}} = 0.96$$

FIGURE 7-8 POWER SUBSYSTEM MODEL

7.7.4 Distribution (Continued)

4,500 W of power at BOL, the power electronics will need to be uprated from this current design. Such a change should not be exceedingly difficult to accomplish with allowances made for mass increases.

7.8 Thermal Control

The Paxsat spacecraft must operate in an arbitrary orbit, and so the thermal control problem is somewhat greater than that for a spacecraft in any one particular orbit.

Spacecraft have been designed though with such a constraint. One example is the MMS bus mentioned earlier.

The basic approach is to thermally decouple the individual modules and provide each with the capability of controlling the internal temperature. For the current Paxsat concept, this philosophy has been largely retained.

The payload face of Paxsat will never see continuous sunlight and neither will the opposite face which houses the power subsystem. The side faces to which the solar arrays are attached may receive continuous sun input, but not beyond a sun angle of 50° or so, due to the change in flight orientation from out-of-plane to zenith. For this reason, the radar transmit chain is spread over the payload face and partway down one of the solar panel faces.

The side faces containing the attitude control and C&DH subsystems can see sunlight in a dawn-dusk orbit. For that reason, units dissipating a relatively small amount of heat are mounted there. The modules also are equipped with louvres to aid in thermal control.

Batteries normally require a tightly controlled thermal environment and are therefore mounted on the aft side of the spacecraft towards the side of one of the solar arrays. In addition, a thermal transport ring is included which can transport heat to whichever face of the spacecraft is cold. This heat transport ring is mounted to a radiator skirt in which the batteries are also attached at the base of the spacecraft.

7.8 Thermal Control (Continued)

This thermal concept would probably ensure that the required thermal control can be accomplished.

7.9 Structure and Mass Properties

The structure of the Paxsat spacecraft must be capable of sustaining the launch vibration environments of the Shuttle and Ariane IV launch vehicles and provide a rigid platform upon which the subsystems may be attached. The structure is therefore cubelike employing cruciform bracing to contribute to the rigidity of the spacecraft. Strict structural dimensions enables the propulsion tanks to also contribute to the structural integrity of the spacecraft as that which appears to being considered in the design of near term future upper stages. Mounts are provided for attaching the Power, Command & Data Handling and the Attitude and Orbital Control Subsystem modules. The payload elements are also provided with modular interface units specifically designed for the payload element. A launch vehicle adapter interface on the bottom of the spacecraft permits mating to the selected launch vehicle airborne support system.

Since subsystems other than the payload elements have not been designed to specific details, subsystem powers and weights are necessarily estimates of the typical current resource allocations required of a spacecraft to perform the Paxsat mission. More detailed estimates are performed in the latter phases of a satellite development culminating with an almost 100% certainty when the elements have been measured.

Various typical spacecraft designs have been consulted to estimate Paxsat resource demands. The most notable programs were the MMS spacecraft, Radarsat of Spar/BAe and a study conducted by RCA for NASA on the National Oceanic Satellite System (NOSS). From these spacecraft, programs mass and power estimates were made.

Table 7-16 details the mass estimates on a subsystem basis for a Paxsat concept spacecraft. The total spacecraft weights 1,466 kg dry at end-of-life with a 20% margin. 3,000 kg of fuel enables the spacecraft to operate over the required regions of space. The spacecraft weight at beginning of life lies in the

TABLE 7-16 PAXSAT MASS BUDGET

PAYLOAD		
Optics	138	
ELINT/COMINT	90	
	------	228 kg
PAYLOAD SUPPORT		
Radar	83	
Command & Data Handling	137	
	------	220 kg
SPACECRAFT BUS		
Attitude Control	79	
Reaction Control	244	
Thermal Control	30	
Power Subsystem	52	
Batteries	33	
Solar Arrays	167	
Structure	159	
Balance Mass	10	
	------	774 kg

TOTAL SPACECRAFT (EOL)		1,222 kg
MARGIN @ 20%		244 kg
TOTAL SPACECRAFT (DRY)		1,466 kg
		=========
Fuel & Pressurant		3,000 kg
TOTAL SPACECRAFT (WET, BEGINNING OF LIFE)		4,466 kg
		=========

7.9 Structure and Mass Properties (Continued)

vicinity of 4,500 kg. At this weight, the Paxsat spacecraft is easily accommodated by the STS launch vehicle of NASA offering the advantageous opportunity of acquiring a shared launch manifest. Paxsat could also be launched on an Ariane IV launch vehicle to be available in the late 1980's. Since the Soviets offered commerical launch services for the next generation of Inmarsat spacecraft, it may be possible to launch the Paxsat spacecraft utilizing a sufficiently rated Soviet vehicle. Design information on Soveit launch vehicles was unavailable for incorporation into the current Paxsat concept configuration and thus the spacecraft was only configured for Shuttle and Ariane IV launch vehicles.

The resource demands for the Paxsat spacecraft on the basis of weight are not prohibitive in terms of launch vehicles currently available or those to be made available in the near future. In this respect, Paxsat is a feasible mission.

7.10 Summary

A brief review of the salient features of the Paxsat spacecraft is presented forthwith.

The spacecraft configuration reflects the high fuel capacity requirement of the rendezvous mission scenario. A roughly cubic propulsion module carries 3,000 kg of bi-propellant fuel and also serves as the primary load carrying structure. The other support subsystems are in modules on five sides of the cube, leaving the sixth side open for the payload.

Large orbit maneuvers are performed using a high efficiency motor. A further 20 thrusters are used for fine maneuvers, and are positioned to minimize the possibility of accidental firing towards the target. Another safety feature is the independent back-up electronics system. It monitors the performance of the attitude and orbit control system which uses computerbased algorithms to guide Paxsat to the desired separation from the object under investigation. Should that computer fail for any reason, this back-up system takes over ensuring continuity of control.

7.10 Summary (Continued)

All spacecraft software is executed by a central computer located in the Command and Data Handling subsystem. This subsystem also provides a link between Paxsat and the ground through a Ku-band TT&C subsystem. A high rate data link (2 Mb/s) allows downlinking of the acquired sensor data. A tape recorded facility records up to 15 minutes of high rate data for those periods when ground station are not visible.

The most visibly conspicuous elements of the spacecraft are the two 15 m solar power gathering arrays of the power module. They, in conjunection with two 22 cell, 50 AHr NiH_2 batteries, provide power to allow full sunlight operations and an equivalent daily surveillance of a single eclipse period for most of the mission life. Some reductions of operations during eclipse near the end-of-life of the spacecraft may be experienced in the highly elliptic Molniya orbit.

All of the spacecraft modules are well within the scope of technology of civilian organizations of non-superpower countries that have a space industry, with some modules readily available without further development. Paxsat is within the launch capabilities of the French Ariane IV launch vehicle. Thus Paxsat is judged to be a feasible spacecraft to fulfill the designated mission role.

8.0 A PROGRAM PLAN FOR THE PAXSAT SYSTEM

8.1 Introduction

This section of the report presents a phased implementation plan for all of the necessary elements in a Paxsat system. Until now, discussion has focussed on the space segment of the Paxsat system by discussing the legal and political considerations of a Paxsat space-to-space verification role, by addressing the operational aspects of politically controlling the spacecraft and by constructing a technically feasible spacecraft concept design. In section 8.2, an entire Paxsat system is defined according to broad functional considerations. Implicit within the functional description lies an assumed system configuration. Other system configurations may be possible and the optimum may not be addressed here, but the system presented is characteristic of the type required to operate the Paxsat spacecraft.

Having defined a Paxsat system into its critical subsystem elements, a schedule of these elements is presented in section 8.3. Discussion centers on the various phases of a system development and encompasses all further R&D, design test and implementation periods prior to the spacecraft launch. Operational lifetimes of the spacecraft are also postulated.

8.2 System Elements

The Paxsat concept system is comprised of six major segments. For a Paxsat mission, these elements are:

(a) Spacecraft

(b) Mission Control Facility (MCF)

(c) Ground Receiving Center (GRC)

(d) Communications Network (CN)

(e) Intelligence Interpretation Center (IIC)

(f) Treaty Governing Body Office (TGBO)

The system concept is illustrated in Figure 8-1.

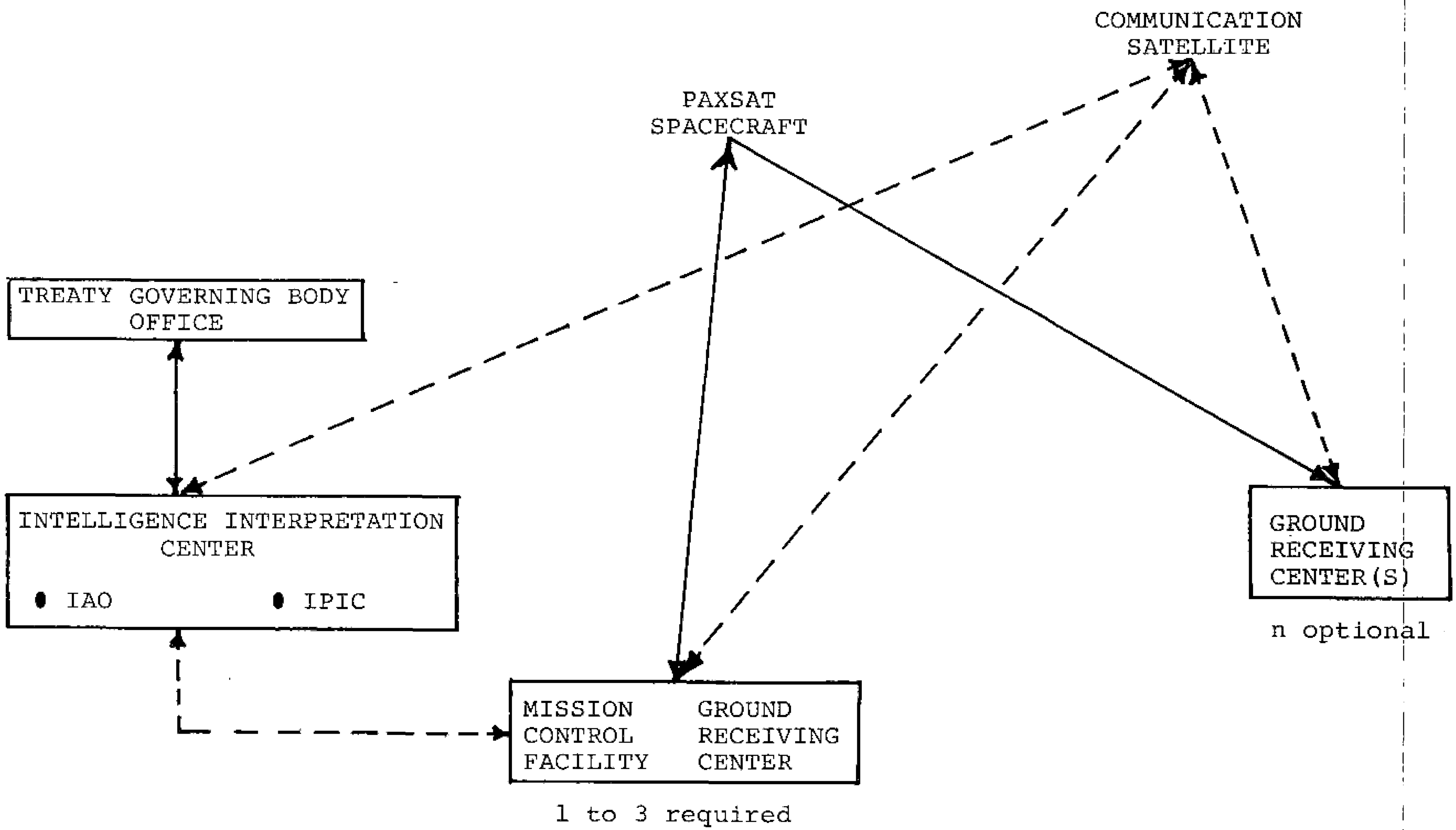


FIGURE 8-1 PAXSAT A SYSTEM CONCEPT

8.2 System Elements (Continued)

The space segment would comprise of a number of identical spacecraft as defined in sections 6.0 and 7.0 of this report orbiting earth in the four regions of space as identified in section 2.0. Considering the utilization of space discussed in section 2.0 and the mission analysis concerns of section 5.0, four operational satellites are required to survey all of the utilized regions of space. Two spacecraft deployed into two distinct low earth orbits are able to survey the necessary regions of the low earth orbit domain between them. Another satellite initially launched in the Molniya semi-synchronous orbit would also be capable of reaching the circular semi-synchronous orbits should the requirement arise. A fourth satellite placed into the geosynchronous orbit would enable the surveillance of this region of space. A fifth satellite would be retained on the ground to act as a spare should one of the other satellites fail to achieve orbit or function for the full duration of its life due to mechanical or electrical failures. Additionally, the fifth reserve satellite could investigate an unidentified satellite launched into a new orbital regime which is unattainable from the four in-situ spacecraft should this event occur.

A full five spacecraft complement with four in space and one on-ground spare is probably the maximum spacecraft investment required to cover all the utilized regions of space to a high degree of effectiveness. Alternative lower spacecraft investment schemes can be envisaged at the penalty of reduced system effectiveness or at increased launch vehicle state of readiness investments. For example, a three spacecraft compliment with two satellites launched into the low earth orbit domain could survey the region of space where there exists the largest threat for potential spacebased weapons, and a third satellite in reserve for launch on demand situations into higher orbit domains. This scenario decreases the space segment investment by two satellites and launch vehicles at the cost of being unable to investigate more than one incident in the semi-synchronous and geosynchronous orbits. Postulating

8.2 System Elements (Continued)

alternative satellite system employment configurations on the basis of cost effectiveness requires knowledge of two elements:

(a) Knowledge of the number of investigations a Paxsat spacecraft is likely to have to make during its lifetime.

(b) Detailed estimates of satellite, launch vehicle and launch pad facility costs.

In this study, it is believed that the former question is answered by stating that full scale politically driven Paxsat investigations will be greater than one in a lifetime and that Paxsat is optimized for numerous encounters. However, further study into the frequency of past alleged treaty violations could bound this estimate. Regarding the second question, this study has not attempted to price the Paxsat system. The ability to price satellite systems and produce meaningful estimates requires knowledge of the spacecraft elements to a greater degree of accuracy than that generated in this concept study. Price tags are first introduced after a phase A study of any given concept.

The Mission Control Facility could be a dual purpose facility containing a single Mission Management Office (MMO) and three Mission Control Centers (MCC). The MMO would be responsible for all the administrative and managerial functions while the MCC would perform the routine system monitoring and control functions of the spacecraft during the dormant loitering operations, and command the satellites during investigative operations under direction from the MMO. A small technical wing of the MMO would provide technical support to management decisions, and plan the mission event sequences for the Paxsat spacecraft encounters with suspect satellites. From this central office, the MCC could be directed, although redundant mission planning facilities should also be available at the MCC locations.

8.2 System Elements (Continued)

Three MCC facilities are required, equally spaced about the globe to effect control over the spacecraft operating in the geosynchronous orbit. Three such locations could be France, Canada and Australia. If, however, operations were confined to the low earth orbit domain, a single MCC would be sufficient under the assumed degree of automation postulated for the Paxsat spacecraft. Each of the MCC facilities would contain the necessary elements to command the Paxsat satellites including command and tracking antennas with ranging capabilities and sufficient computer resource facilities to effect control of the satellites. The MMO could also be co-located with one of the MCC facilities.

Ground Receiving Centers (GRC) are also required to receive the payload imaging data from the Paxsat spacecraft during investigations of suspect satellites. The image data will be received by the tracking antennas, demodulated and then may or may not be initially processed at the stations depending upon the security required on the images. Archiving facilities would be maintained at each station for temporary storage before distribution to the Intelligence Interpretation Center. Received data would be then sent either by computer compatible tapes or over the communications network to the Intelligence Interpretation Center. Security of data may be insured by introducing an encryption step before transmitting data over the communications network. As with the MCF's, three GRC's are required for satellite operations in the geosynchronous orbit but only one need be built if operations were confined to low earth orbit.

More than three GRC's may be employed around the world to collect and distribute data to the IIC though this is not entirely needed. The advantage of numerous GRC's strategically located around the globe is the lessened requirement for on-board data storage for later playback on the Paxsat spacecraft. The current Paxsat spacecraft concept includes tape recorders to store data when the satellite is not within view of a GRC.

8.2 System Elements (Continued)

The Communications Network may simply be the employment of communications links between the numerous GRC's and MRF's to the central IIC and MCF on current Intelsat or Intersputnik satellites. Communications equipment for both uplink and downlink communications would be needed at each site. Additionally, if only one IIC colocated with MCF and a GRC is required to fulfil the data and control requirements in a LEO operation only situation, the need for a communications network is negated.

The Intelligence Interpretation Center may be a dual function entity with two wings under a central management office. The first office would be an Intelligence Acquisition Office (IAO) with the function to gather readily available information of national space activities and in particular, information pertinent to the development of technologies for spacebased weapons. The office would function much like an Institute for Strategic Studies, gathering, recording, archiving and analyzing activities related to space. One such important piece of data the IAO would maintain is an up to date space objects log containing the orbital elements of all objects in space at any given time. This data may be made available through contributions of signatory nations with the National Technical Means to gather such information. The second office would be an Image Processing and Interpretation Center (IPIC) with the function to process the image data generated by Paxsat spacecraft and interpret the function of the subject spacecraft for the determination of whether the satellite carries on-board, or is in itself, a weapon in space. This office would contain the spacecraft engineering, optical, computer communications, chemical and physical expertise to determine the presence of weapons in space from the remotely gathered information by Paxsat. The office would also be responsible for the engineering evolution of Paxsat designs in subsequent generations of spacecraft procurements to perform its investigations to a higher degree of efficiency. Within the Paxsat system only one Intelligence Interpretation Center is required.

8.2 System Elements (Continued)

Finally, a Treaty Governing Body is required to administer the entire Paxsat system. Political representatives from treaty signatories may sit in council as discussed in section 3.0 and exercise control over the launching of investigations against suspect spacecraft. A single building housing this body is sufficient for the Paxsat system and may be co-located with the Intelligence Interpretation Center and the master Mission Control facility.

8.3 Paxsat Program Plan

This section presents a phased implementation plan for all the Paxsat system elements. It encompasses all further R&D, design, test and implementation phases prior to the spacecraft launch. The schedule is shown in Figure 8-2.

The different program phases have been defined as follows:

(a) Phase A - Concept Definition

This phase encompasses the user studies technology studies and overall system concept definition study (Phase A Study) required to define a set of mission requirements and confirm the feasibility of meeting them. It also provides the first detailed cost estimate and definition of critical technologies.

(b) Phase B - Program Definition Phase

This phase delegates the system design effort to the level required to define a unique configuration which is the preferred method of meeting the requirements. It carries the element trades to a lower level of detail than does the Phase A study, and results in an overall hardware definition, identification of the major units, preliminary subsystem and major element specifications, and a detailed implementation plan and cost for the remaining phases.

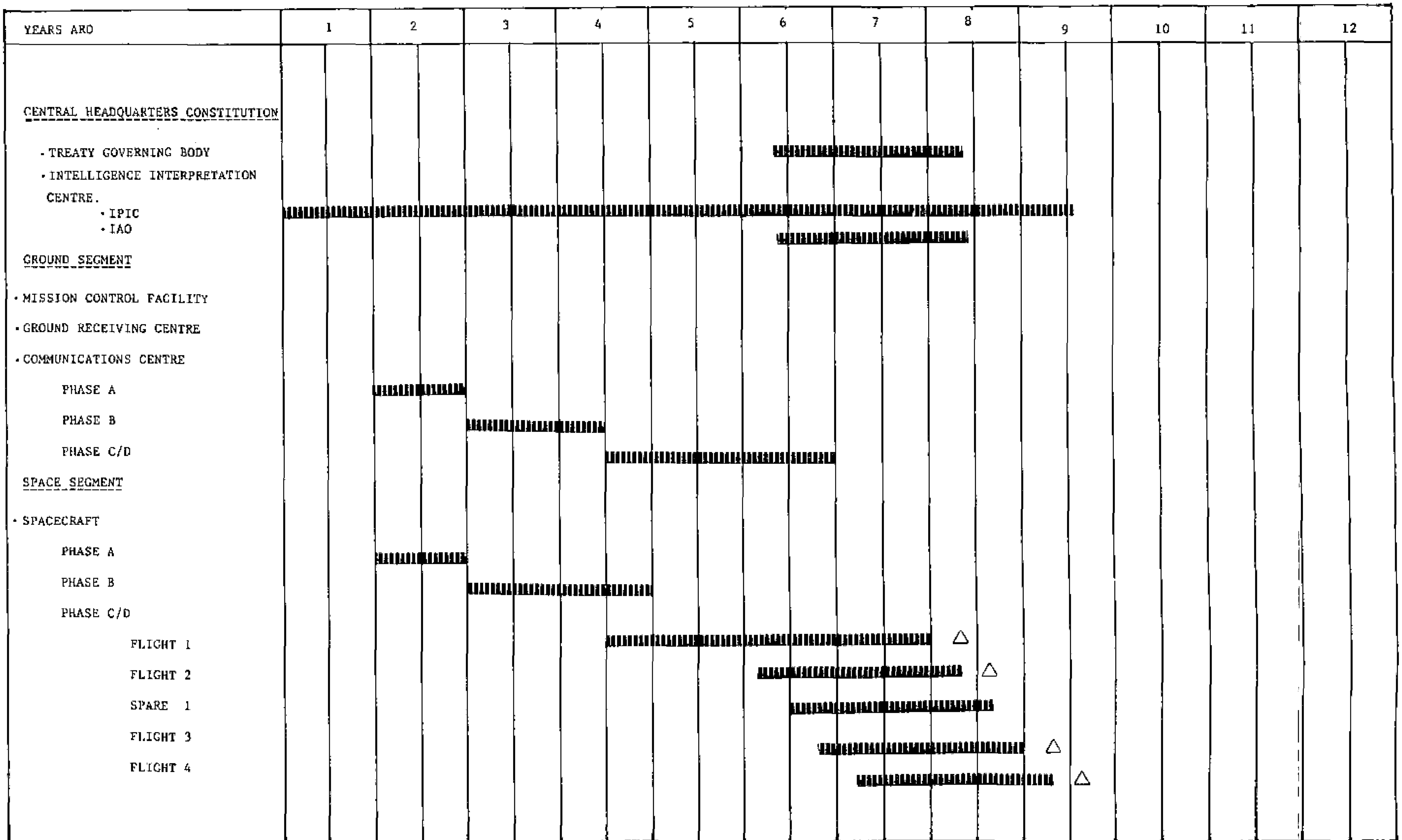


FIGURE 8-2: PAXSAT A PROGRAM PLAN

8.3 Paxsat Program Plan (Continued)

(c) Phase C - Detailed Design and Development

During this phase, all system elements are designed and preliminarily tested. In the case of the space segment, Phase C is usually considered to include all the necessary and/or subsystem spacecraft qualification testing. For the ground system components, Phase C encompasses system integration and testing at the manufacturers's sites, prior to shipment to the operational site.

(d) Phase D - Production

This is the final integration and test phase. The flight spacecraft is assembled, tested, shipped to the launch pad, and mated to the booster and the ground station hardware and software are integrated and tested on site.

The boundary between phases C and D is not well defined in general, since they must often overlap to meet program schedule constraints. While this does increase risk somewhat, it is seldom a real difficulty if sufficient attention is paid to the programmatics from the start. For this reason, phases C and D are often combined into one joint phase. This has been done for some of the Paxsat elements.

Figure 8-2 defines a possible program schedule for the Paxsat system. The schedule is divided into three main sections:

(a) Central Headquarters

The Central Headquarters implementation plan schedules the development of the Treaty Governing Body Office, and the Intelligence Interpretation Center. The relationship of the Central Headquarters to the other project elements is based upon the following proposition, that a core team of capable people would be assembled to serve as the project management office for the duration of Paxsat system development. This core of individuals would form the initial Managing Office

8.3 Paxsat Program Plan (Continued)

and the initial Image Processing and Interpretation Center of the Intelligence Interpretation Center to which staff would be added as the Paxsat spacecraft come into operation. Consequently, the formation of this entity is seen to proceed all other elements in the schedule. Approximately 24 months prior to the launch of the first Paxsat spacecraft, the second wing of the IIC, the Intelligence Acquisition Office, would be formed and brought up to speed before operation of the Paxsat satellites.

The Treaty Governing Body Office come into being prior to the launch of the first Paxsat spacecraft. Here, it is explicitly assumed that a treaty has been created for which Paxsat is to be an integrated member. Implicit within this scheduling is the fact that the Paxsat concept is developed in parallel to the creation of a treaty.

Development of the Paxsat concept could be undertaken in parallel to the negotiation of a treaty with little financial risk. Most of the financial cost of any mature spacecraft program is in the phase C/D portion of the program phase. Although the Paxsat system is a unique mission, the technologies are sufficiently mature for the non-recurring development costs prior to phase C/D to be of manageable levels. In addition, section 3.0 of the report suggested benefits that a Paxsat spacecraft could bear on the negotiations of an outer space weapons ban. A period of two years was deemed to be sufficient for the creation of the Treaty Governing Body organization.

(b) Ground Segment

The Ground Segment schedule contains the three elements of the Mission Control Facility, Ground Receiving Center and the Communications Network. Each of these centers are postulated to be developed in parallel with one another with the schedule allowance shown. Additionally, as three

8.3 Paxsat Program Plan (Continued)

of each such elements would be required for operations within the geosynchronous orbit, three such complete systems would also proceed in parallel. This parallel development, despite first impressions, is not so severe. Primarily, the three elements of a Missions Control Facility, Ground Receiving Center and the Communications network is slightly more complicated than building a Command, Tracking and Ranging Facility and a Communications Ground Station for commercial fixed satellite services. Such a system is typically manufactured (phase C/D) and installed on site within 24 months. Secondly, the three parallel networks need be installed in three separate nations around the world thereby suggesting three contributors to the financial load of the Paxsat system. The program schedule is typical of a similar ground control network proposed for a future earth resource mission requiring 12 months of phase A activities, 18 months of phase B activities and 30 months of phase C/D activities.

(c) Space Segment

The Space Segment schedule illustrated in Figure 8-2 is characteristic of a typical satellite program. One year of Phase A activities followed by approximately two years of Phase B activities is typical of a spacecraft program. A six-month prestart on Phase C/D activities is characteristic of the jump start commercial spacecraft contractors employ to meet schedule constraints imposed by the spacecraft buyer community. A well-defined spacecraft system can be manufactured, integrated, and tested within a 36 month period. In the schedule illustrated for the Paxsat spacecraft, an additional 6 month period is allowed for the building of the first spacecraft to account for unforseeable events that may delay normal program integration and test procedures. The learning curve phenomenon indicates that subsequent spacecraft could be built, integrated, and tested in the more usual length of time. Spacecraft deliveries can typically be staggered on 4 month centers depending upon the integration facility at the selected contractor. Finally, a 4 month launch campaign is assumed for each of the flight spacecraft.

8.3 Paxsat Program Plan (Continued)

Launch vehicle manufacturing and integration times based on the payment schedule for such procurements will take 30 to 36 months. Thus, launch vehicles are usually procured 30 to 36 months in advance of the launch date with notification of projected launch demands given prior to this period.

A 5 year operational lifetime for the low earth orbit and semi-synchronous orbit satellites and 10 year lifetime for geosynchronous orbit satellites can be expected. Prior to the expiration of these satellites, replacement satellites need to be launched to insure continuity of coverage. Technological updates to the satellite designs may be made on 10 year centers corresponding to the lifetime of the geosynchronous satellites. Refuelling capability coupled with the modular design of the Paxsat spacecraft may increase the operational lifetime of the low earth orbit satellites with in-orbit types manned repaired missions like that demonstrated with the recent repair of the Solar Maximum Mission by a shuttle crew.

9.0 CONCLUSION

The purpose of this study for the Canadian Government Department of External Affairs, was to determine the feasibility of a spacebased remote sensing system designed to determine the presence of weapons in space.

The Paxsat System Concept was based on the supposition that a properly configured set of observations in space could determine the function of an unknown satellite to an acceptable high degree of confidence such that it can contribute to the determination and control of weapons in space.

The feasibility of the Paxsat System Concept in the performance of the study was addressed by three principal questions:

(a) Can observations of an object in space determine the function of the object, particularly in reference to a weapon system?

(b) Are there one or more political/international agreements or treaty contexts in which these observations could be made?

(c) Would the observational requirements and the political constraints of a governing treaty permit a viable mission and spacecraft design?

The results of this study taken in context of its predecessors [1,2] conclude that all three questions are answered in the affirmative. The Paxsat System Concept was judged to be a feasible vehicle in which to effect the determination and control of weapons in outer space within the context of specific scenarios developed in the study. Highlights of the study as they pertained to the conclusion of the Paxsat System Concept effectiveness are presented forthwith.

Prior to the answering of the first question, a review of future putative spacebased weapons systems was conducted to determine the characteristics of these weapons, and the possible regions of space where these weapons would be deployed. Of the four categories of weapons systems identified, a weapon threat analysis

9.0 CONCLUSION (Continued)

concluded that the greatest threat for a spacebased weapon system is to be directed against space assets in an antisatellite role. Questions of versatility and cost effectiveness of other basing schemes relegated spacebased weapons directed towards terrestial targets a lower threat rating. Ballistic missile defense systems were judged to be at a slightly lower threat rating than antisatellite systems on the basis of technological maturity. It being possible that a BMD system would be implemented as a followon to the first generation of a spacebased ASAT weapon. The technological kinship between BMD and ASAT weapon systems however, makes such assessments difficult. The fourth weapon system directed against space weapons themselves would only surface after the initial deployment of a spacebased weapons system.

Of the weapons systems examined, each represents somewhat different levels of technological sophistication, not only in terms of the weapon capabilities but also in the ways in which they are deployed. The destructive mechanisms which may be employed very from simple chemical explosives through nuclear bombs, to exotic laser and particle beam weapon technology. Each technological means has a characteristic merit of effectiveness and is at a different state of technological maturity, thereby implying a cost effectiveness measure. It was thereby concluded that spaceborne weapon systems would require high levels of optimization, and would be focussed on targets which are of such a nature as to justify the complexity and cost of the weapon systems envisaged.

These considerations suggested that the verification of space for the presence of weapons should be oriented towards particular spacecraft configurations and specific orbits. Not all of the spacecraft presently deployed were found to constitute logical and rewarding military strategic or tactical targets for space-to-

9.0 CONCLUSION (Continued)

space weapon systems. The highest risk satellites were determined to be military satellites foremost. Within the military satellite classification, the satellites of highest risk were determined to be:

(a) Dedicated targeting systems

(b) Spacebased weapon systems

(c) Surveillance and reconnaissance systems

(d) Navigation systems

(e) Communication systems

where the first two categories having yet to be deployed in space. Of the satellite systems that have been deployed, four orbital quantizations were found to be employed. These orbital domains were classified as:

(a) Low earth orbit

(b) Semi-synchronous

(c) Highly elliptic

(d) Geosynchronous

and all fall within an altitude of 50,000 km above the surface of the earth. In addition, the most utilized orbit was determined to be the low earth orbit and it was found to contain most of the higher risk military satellites.

In answering the first question:

> "Can space observations determine the role or function of an object in space?"

Reference [1] went a long way towards defining the observations required to detect the presence of weapons in space. This earlier report focussed its attention to a particular class of future spacebased weapons, the antisatellite weapon. The present study expanded the

9.0 CONCLUSION (Continued)

discussion of weapons to include space-to-ground weapons and ballistic missile defenses and concluded from a space-to-space remote surveillance point of view the range of technological alternatives for an antisatellite system encompasses those for the added weapon systems under consideration.

The relatively primitive technologies like chemical explosives and nuclear weapons, covertly deployed within otherwise normal looking spacecraft to serve antisatellite functions, pose the most difficult verification role for a Paxsat System Concept. As weapon systems develop more technical finesse like employing laser or particle beam technologies, the characteristics of the system become more conducive to the determination of function. Thus, it was concluded that not only must the spacebased, space-to-space remote sensing system detect the presence of exotic weapon systems, but also possess sufficient faculties to determine the function of legitimate spacecraft missions.

The high degree of optimization inherent in the design of all spacecraft and in their orbital parameters, together with the nature of signals to and from the spacecraft, provide highly significant data as to function. Clearly to the extent that form follows function, visual images of the spacecraft were deemed to be highly determinate of its function and its purpose in space. If the images can also be acquired in the thermal-infrared region, then important data can be derived regarding the enemy balance and utilization of the unknown spacecraft. The operation of almost any type of spacecraft involves substantial communications to and generally from the spacecraft. The nature of these transmissions, particularly the data rate, frequency band of operations, radiated power and the operational cycle are of extremely high diagnostic value. Other sensors including gas analyzers to detect materials of chemically powered lasers, or of chemical explosives, and radiation detectors to infer the materials associated with power sources or weapons extend the faculties of spacebased, remote sensing system for the determination of weapons in space.

9.0 CONCLUSION (Continued)

Within this present study, emphasis has been given to the optical and electronic support measures payload for the Paxsat System Concept spacecraft. Priority of thought had been given to these elements considering the highly diagnostic value of the data sensed by these payload elements. Additionally, it was of the opinion that these systems would drive the resource allocations in the conceptual design of the spacecraft, since it was postulated that the thermal imaging of the unknown spacecraft could be accomplished with a reduced resolution using the optics of the visible imaging system. Further study is recommended on the designing of the thermal imaging system and its implementation impacts on the current spacecraft conceptual design. Additionally, the tertiary sensors comprising of gas analyzers and radiation detectors were not investigated as a priori knowledge of the spacecraft mission operation was not available for the inclusion of these close proximity instruments. It is also recommended that further study on these sensors be conducted to increase the sensing faculties of Paxsat, now that the rendezvous mission scenario has been judged to be feasible.

This study, concentrating on the optical and electronic support measures payload, had determined a payload concept design that is capable of providing the necessary high resolution data within acceptable spacecraft resource demands for the Paxsat System Concept to determine the function of any space object to a high degree of certainty. Furthermore, the technology required to design and manufacture these instruments is concluded to be within the current day state-of-the-art.

Having concluded that observations of an object in space can determine the function of the object for the detection and control of weapons in space, the political/legal implications enabling such observations were analyzed as a response to the second question:

> "Are there one or more political/international agreements or treaty contexts in which these observations could be made?"

9.0 CONCLUSION (Continued)

Analysis of this question on the plausibility of an outer space arms control regime conducive to a Paxsat System Concept concluded the requirement for three necessary conditions:

(a) A multilateral arms control agreement

(b) A multilateral administrative and verification organization

(c) A recognized requirement for multilateral technical means of verification

The first requirement requires a treaty for the verification role of a Paxsat System Concept to be sanctioned. Operation of the Paxsat system outside a treaty context was concluded to be politically unacceptable as the treaty signatories would not be legally bound to the findings of a third party. The political context considered most appropriate for the Paxsat System Concept was an element of a treaty negotiated primarily between the Superpowers but extended to the multilateral participation to avoid potential space weapon proliferation. This study concluded that the inherent vulnerability of space weapon systems and the extensive use of space for commercial and national purposes may make a multilateral treaty more attractive than might otherwise be the case.

A multilateral administration and verification organization was concluded to be a necessary condition as the requirement for a political decision making process is compulsary in the multilateral operation of the verification system. The administration forum would act as a political control mechanism which could protect the bilateral imperatives of the two Superpowers. Additionally, considering the sensitivity of violations, the administration organization could be constructed where the existing bilateral practices of the Superpowers would not be prejudiced.

The study concludes as a third necessary condition the recognized requirement for a multilateral technical means of verification. It postulated that all members of the treaty would contribute data from their National

9.0 CONCLUSION (Continued)

Technical Means, generally the spacecraft tracking data derived by ground based radars or optical installations. This would exhibit the advantage of providing a more economical overall system with more countries participating in its operation.

With the observational, legal/political questions of the study answered in the affirmative, attention turned to the answering of the third question:

> "Would the observational requirements and the political constraints of a governing treaty permit a viable mission and spacecraft design?"

The question was answered in two parts:

(a) The mission analysis operations

(b) The spacecraft concept design

Of the three mission operation alternatives examined in the course of the study, the rendezvous scenario was concluded as the most feasible alternative over the launch on demand scenario and the fly-by scenario. The rendezvous scenario presented the least demands from the payload sensor performances and provided the greatest operational flexibility to perform the most powerful analysis of the target spacecraft's function through co-orbiting over an extended period of time. Such benefits did not come without associated penalties. Just as a significant amount of energy is required to place a mass into an orbit in space, the Paxsat spacecraft would require a considerable, though not excessive, consumption of fuel to change its orbital inclination and altitude in order to carry out a particular intercept requirement. In addition, the placement of the Paxsat spacecraft into initial loiter or parking orbits may require up to a maximum of 90 days before the Paxsat spacecraft is on-station beside the target spacecraft. Such a period of time was judged as a politically acceptable and was employed to minimize the expenditure of fuel to enable the Paxsat spacecraft to conduct multiple missions under ideal conditions. The quantization of military satellites into four orbital domains enables a Paxsat system of four satellites to

9.0 CONCLUSION (Continued)

survey all of military space. Two Paxsat spacecraft in distinct low-earth orbits, provides coverage of this orbit domain. A single Paxsat spacecraft enables observations for both the highly elliptic and the circular semi-synchronous orbits and another single spacecraft in the geosynchronous orbit permits verification in this orbital region.

The conceptual spacecraft design baselining the rendezvous mission scenario developed in the study was determined to be feasible within the scope of the technology of civilian organizations of a non-superpower countries that have a space industry. In fact, some of the modules proposed in the concept design were readily available without further development. The Paxsat spacecraft concept design was shown to be within the launch capabilities of the French Ariane IV launch vehicle and the American Space Shuttle. The spacecraft bus resources supplying sufficient power and mass carrying capabilities enabled operation of the spacecraft in all of the required orbital regions of space for a lifetime between five and ten years. Thus the developed Paxsat spacecraft was concluded to be a feasible spacecraft to fulfill its designated mission role.

In conclusion, the Paxsat Concept System was judged to be a feasible spacecraft based system to determine the presence of weapons in space and contribute to the effective verification of a treaty banning the deployment of weapons in space.

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