SPARE SATELLITE ~~STUDY:~~ Positioning study,

FINAL REPORT

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SPARE SATELLITE STUDY

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GLOSSARY OF ABBREVIATIONS

B	Bandwidth
BER	Bit Error Rate
CCIR	International Radio Consultative Committee
CCITT	International Telegraph & Telephone Consultative Committee
CH	Channel
C/N	Carrier to Noise (Density) Ratio
CRC	Communications Research Centre
CT&R	Command Telemetry and Ranging
dB	Decibel
DBS	Direct Broadcast Satellite
dBW	Decibels above 1 Watt
dc	direct current
DSS	Department of Supply and Services
EIRP	Effective Isotropically Radiated Power
EoL	End of Life (mission)
EQ	Equalizer
E-W	East-West
FET	Field Effect Transistor
FoV	Field of View
ft	foot
G	Gain
H	Horizontal/Hybrid
HP(A)	High Power (Amplifier)
IFRB	International Frequency Registration Board
IR	Infra Red
ITU	International Telecommunication Union
IUS	Interim (Inertial) Upper Stage
k	Boltzmann's constant
LNA	Low Noise Amplifier
LO	Local Oscillator
LP	Low Power
M$C	Millions of Canadian Dollars
MUX	Multiplexer
N	North
No.	Number
NSSK	North-South Stationkeeping
OP	Output
P	Power/Period
POL	Polarization
pWOp	picowatts, psophometrically weighted, at a point of zero relative level
R	Drift Rate (o/day)
Rcvr	Receiver

GLOSSARY OF ABBREVIATIONS (cont'd)

RF	Radio Frequency
RPM	Revolutions per minute
Rx	Receive
S	South
S/C	Spacecraft
Sec	Second
SPDT	Single Pole Double Throw
SSUS-A	Spin Stabilized (spinning solid) upper stage type A
SSUS-D	Spin Stabilized (spinning solid) upper stage type D
T	Noise Temperature
TDF	Track Directional Filter
TM	Telemetry
TTAC	Telemetry Tracking and Command
TWT (A)	Travelling Wave Tube (Amplifier)
Tx	Transmit (ter)
UHF	Ultra High Frequency
U.S.	United States (of America)
V	Vertical
W	Watt

1. INTRODUCTION

This report covers a study carried out by Spar Aerospace Limited on "Satellite Sparing Techniques". The study was carried out for the Department of Communications under Contract 07SU 36100-8-1131 with Mr. R. Milne as the Scientific Authority.

The need for spare in-orbit satellites exists to counter the possibility of catastrophic failure of an operational satellite carrying high priority traffic. Catastrophic failure usually means complete loss of a particular satellite but a spare may also be needed during the outage whilst a non-catastrophic failure event is understood and corrected.

Fifteen years have elapsed since the first INTELSAT communications satellite went into orbit. Since that time many failures have occurred, which can be categorized as either 'Wearout' or 'Unpredictable'.

'Wearout' occurs in all materials, and therefore satellite components, to different extents. When the 'wearout' rate is known, the satellite design can be made to compensate for the associated performance degradation or failure. The satellite lifetime will therefore be defined by the component, which has the shortest compensated wearout time. This type of failure is somewhat 'predictable' and if it was the only failure mechanism would obviate the necessity of the in-orbit spare. The following generation of satellite would be planned to be in-orbit at the predictable wearout time.

Other failures occur, however, without warning. These may be termed 'unpredictable' failures. They often occur early in the life of a new generation of spacecraft and can be caused by either an unrecognized wearout phenomenon or an isolated event. Their effects are mostly minor but some have been catastrophic. Later satellites in the same series can be corrected if the cause of failure can be established.

Table 1.1 clearly shows the predominance of 'unpredictable' failure over the 'predictable' failures experienced by INTELSAT to date.

As demand for communications capacity increases, the satellites filling the few available orbit slots will become more and more complicated, thereby increasing the scope for and likelyhood of 'unpredictable' failures. The need for spare satellites in orbit will therefore perpetuate but because of the ever increasing density of active satellites in some portions of the orbital arc orbit positions available for spare satellites will decrease. Figure 1.1 clearly illustrates that congestion over the major land masses is imminent.

1.0 INTRODUCTION (continued)

To minimize pressure on the geostationary arc, an orbital position should, ideally, be only occupied by a satellite of a certain minimum capacity, which can be determined as a function of orbit meridian, footprint coverage and time. Predictions of capacity requirements can be made for periods well in excess of the present spacecraft design, manufacture and life cycle, thereby permitting each generation of hardware to match the required capacity of that epoch. Often such rigorous requirements are not applied until the pressures on orbit utilization reach crisis proportions. It is more reasonable to expect that frequency reuse through cross-polarization could be made mandatory in the orbital arcs where the crisis level is within the foreseeable future.

There remains, therefore, a need to formulate a strategy for providing (and testing) spare capacity in each of the categories of satellite service, which range from low power UHF maritime communications to high power 12 GHz direct broadcast service.

The study addresses the problem of providing in orbit spare satellites in an increasingly congested orbit without occupying an authorized operational position with a non-operating spare. They may be located either in an interstitial position between two operational satellites or co-located with an operational satellite. In either position the level of radiation they may emit without undue interference with operational spacecraft is strictly limited. The objective of the study is to determine methods for periodically checking the health of satellites for the recommended sparing philosophies.

Spacecraft	Launch Date	No. of Failures (to mid 77)	Failures/S/C–Year
F-2	1/71	15	2.3
F-3	12/71	8	1.5
F-4	1/72	6	1.1
F-5	6/72	8	1.6
F-6	Launch Failure	-	-
F-7	8/73	4	1.1
F-8	11/74	4	1.6
F-1	5/75	1	0.5
		46	1.5

'Wearout' 13 (28%) — 'Unpredicted' 33 (72%)

Unpredicted failures have included:

- Comms Receiver
- Solar Array Bearings
- Thrusters
- Command Decoder
- Earth Sensor
- Propellant
- Propellant Relief Valves
- Initial Erection

Table 1.1 – Intelsat IV : 'Wearout' and 'Unpredictable' Events

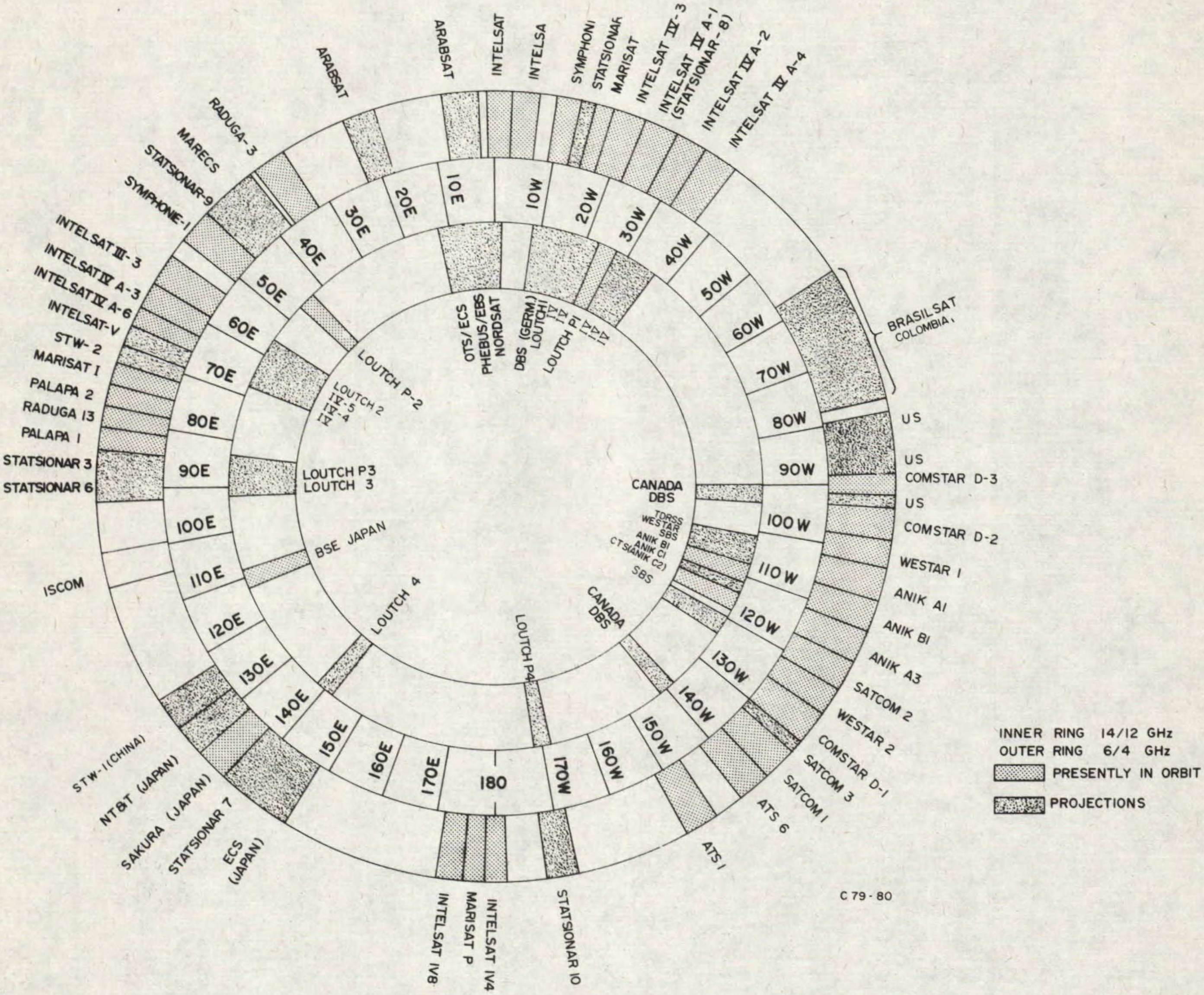

Figure 1.1 - Present and Projected Occupancy of the Orbital Arc in the Major Fixed Service Frequency Bands

2. CATEGORIES OF SERVICE AND SPARING PHILOSOPHIES

2.1 Fixed Satellite Service

Both the International and Regional categories of this service require separate examination for the suitability of the different forms of sparing.

2.1.1 International Fixed Satellite Service

The International Fixed Satellite Service is the domain of the INTELSAT organization, which is a monopolistic supplier of trans-oceanic satellite communications capacity. Such an organization must provide a high grade of service to conform to CCITT Recommendation E542[1], which recommends the allowable reduction of circuits that may be tolerated for short periods of approximately 15 minutes, thereby implying that breakdowns such as a transponder failure should be correctable within this time. (Failure to rapidly correct a breakdown can cause disproportionate degradation of service owing to the additional circuit loading by repeat attempt telephone calls).

Routing diversity, via cables and other satellites, is also recommended to ensure that failure of a satellite can be compensated by excess capacity on alternative paths, but rapid restoration remains highly desirable. A fully operational spare satellite and ground antennas, capable of fast repointing, are fundamental to rapid restoration, although the decision making process and the repointing time will often cause the total restoration time to exceed the recommendation.

INTELSAT is obliged to project and cater for the expansion in type and quantity of the communities' telecommunications requirements, which, from time to time, will result in surplus capacity and/or superfluous satellites that must be configured into the overall sparing philosophy. As an example, the IVA spacecraft will begin to be replaced by the V series at just beyond half the former's original design life. As sole user of the orbital arc in this particular category, optimum use can be made of both the spectrum and the arc by utilization of residual life in predecessor satellites, and the re-location of in-orbit spare satellites in various combinations to provide for system integrity and growth (2) and (3). Table 2.1.1 demonstrates how this evolution of capacity is planned to take place for the Atlantic segment in years to come. This table clearly shows that the in-orbit capacity for the orbital arc between 325.5°E and 340°E contains at all times spare capacity that is capable of immediately backing up the operating satellite. Using knowledge of the circuit capacity of each INTELSAT model (IV - 5,000, IVA - 7,000, V - 12,000 half circuits), the information from this table is plotted in Figure 2.1.1 in terms of total segment capacity, both operating and spare. Against this is plotted the projected requirement including due allowance for path diversity and leased services up to the year end 1990. It can be seen that the total capacity is well matched to the projected requirements. Any shortfall will normally be overcome because of the usual delay in reaching the projection. After the 80's the concepts are numerous (4) but as yet there is no suggestion that the occupancy of the orbital arc will be increased, rather that more use will be made of the ability of INTELSAT space and ground segment to operate with 3° orbit separation as opposed to the present 5°.

This type of service, located in parts of the arc for which there is little demand by other users, could continue to use the "fully operational" spare orbital position. Efficiency of spectrum and orbit useage is high because these spares may be used in an operational mode, to carry traffic that is interruptible, should the spare be required to support a primary satellite failure.

Such a philosophy obviates the need for any special sparing/testing philosophy. Testing will remain as currently practised and will involve the monitoring of live traffic. No further consideration is given to these already well established techniques.

1) CCITT Recommendation E.542 'Acceptable Reduction in the Number of Circuits of a Final Route in the Event of a Breakdown'.

2) 'Operational Planning for the Utilization of INTELSAT V Satellites', J.B. Potts and F.J. Burkitt, paper 78-529 AIAA 7th Satellite Communications Conference, San Diego 1978.

3) 'INTELSAT's Orbital and Spectral Needs in the 1980's', S.B. Bennett, paper 78-531, AIAA 7th Satellite Communications Conference, San Diego 1978.

4) 'Planning for the Post-1985 INTELSAT System', H.L. Von Trees et al, paper 78-532, AIAA 7th Satellite Communications Conference, San Diego 1978

DEGREES EAST	325.5	328.5	330.5	332.5	335.5	340.5	IN ORBIT COMPLEMENT	
MP = MAJOR PATH SP = SPARE P = PRIME	MP1	SP	SP	SP	P	MP2	ON ORBIT	OPERATING
1978	IV-A (F-4)			IV-A (F-2)	IV-A (F-1)	IV (F-3)	IV + 3 IV-A	2IV-A +IV
1979						IV (F-7)	IV + 3 IV-A	2IV-A+IV
1980		IV-A (F-2)	V (F-3)		V (F-1)	IV-A (F-1)	2V +3 IV-A	V+ 2IV-A
1981							2V+3 IV-A	V+ 2IV-A
1982						V (F-7)	3V+ 2IV-A	2V + IV-A
1983	V (F-8)						4V	3V
1984							4V	3V

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Table 2.1.1 - Intelsat Evolution and Usage - Atlantic 1978-84

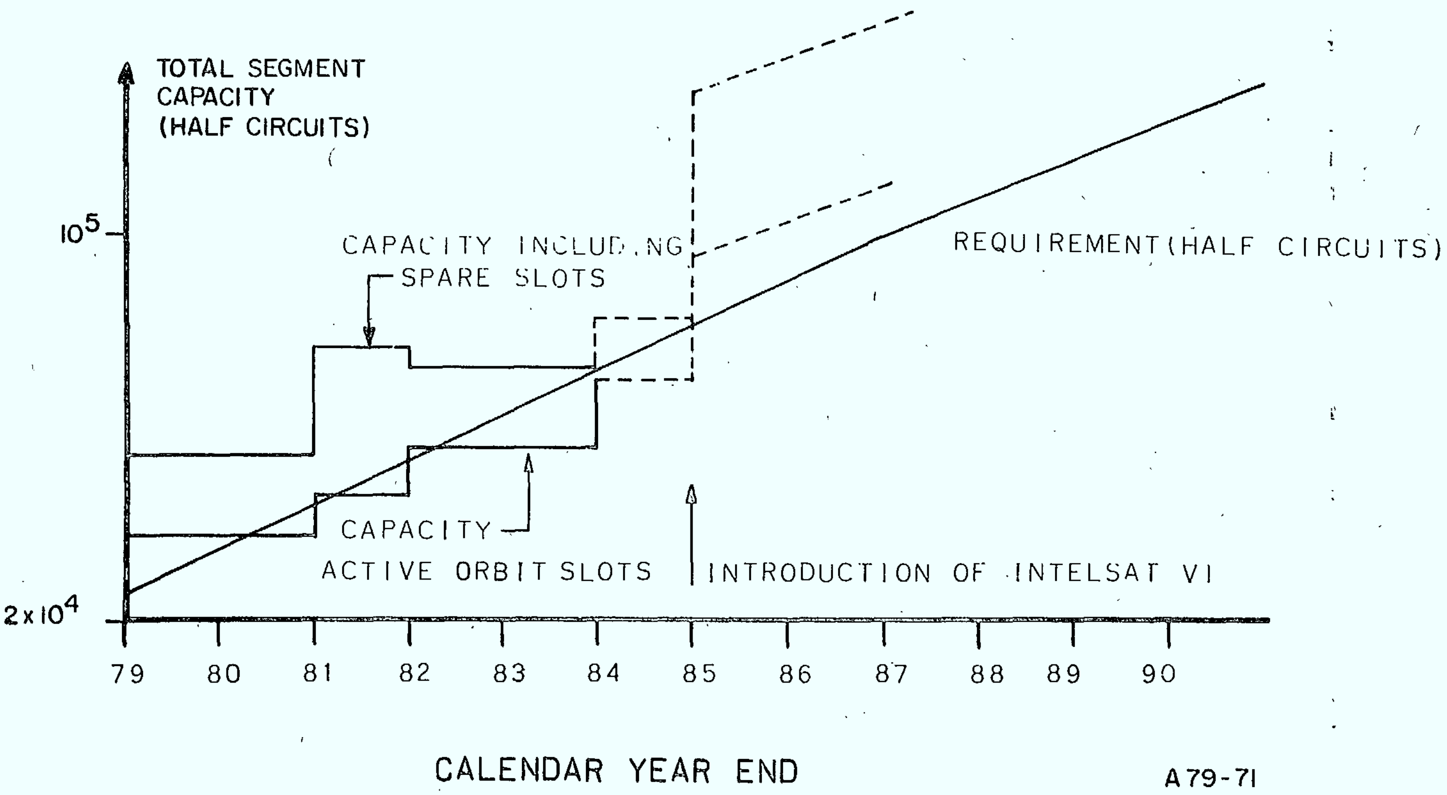

Figure 2.1.1 - Atlantic Telephony Traffic Projection: Requirement/Capability

2.1.2 Regional Fixed Service

The regional fixed satellite service is required to supply a wide range of services, which for the purposes of this study can be categorized as follows:

A) General Coverage-Heavy Route

This form of satellite is typified by a telecommunications common carrier operating long distance high-density traffic in and out of tracking ground stations.

B) Multipoint-Thin Route

This function supplies thin route communications, data or TV distribution to widespread locations where small fixed position ground antennas are installed.

C) Multiple Spot Beam or Scanning Beam

The category is intended to encompass those current and future spacecraft that will service distinct geographical areas by means of very narrow spot beams either fixed or scanning. This distinction is necessary because the satellite sparing philosophy will depend on the on-board interconnectivity of an individual satellite.

Category A has a direct parallel to the International fixed service and so the sparing philosophies will be practically the same: The primary load will have full spare satellite back-up and therefore be capable of rapid restoration. The regional services are concentrated in the heavily congested regions and are difficult to justify. Only a sufficient volume of pre-emptible traffic will justify the use of a fully operational spare satellite in the face of the needs of other regional services. The charges for channels on the spare will reflect the risk of interruption in which case maximum benefit is achieved for all parties[1]: The high priority users have a high grade of service with guaranteed continuity, the low priority users have a very economical transmission path, the operators have a fully laden revenue earning satellite and the orbit resource is working more efficiently than if the spare satellite were dormant.

In Category B systems the fact that many of the ground stations are fixed, unmanned and remote severely limits the in-orbit sparing possibilities. Restoration within hours would obviously require a co-located spare, but in a multiple satellite system this may be uneconomical and so restoration times of the order of a day could be considered when provided by an interstitial spare. The sparing of this category of service is more fully developed in Section 3.

Several factors may cause a change of sparing philosophy for satellites in Category C. The high capacity of such spacecraft is partly due to the customizing of antenna coverage patterns either by the provision of several steerable spot beams or a narrow beam which scans a total coverage area. In either case the custom design and high reliability of a high capacity satellite could well alter the economics of the spare: It may prove unattractively expensive to make a particular antenna design flexible to cover prime and spare orbit positions, there may be insufficient traffic of a pre-emptible nature to fully load an operational spare, inter-connectivity and system reliability trade-offs may result in undesirable bus requirements.

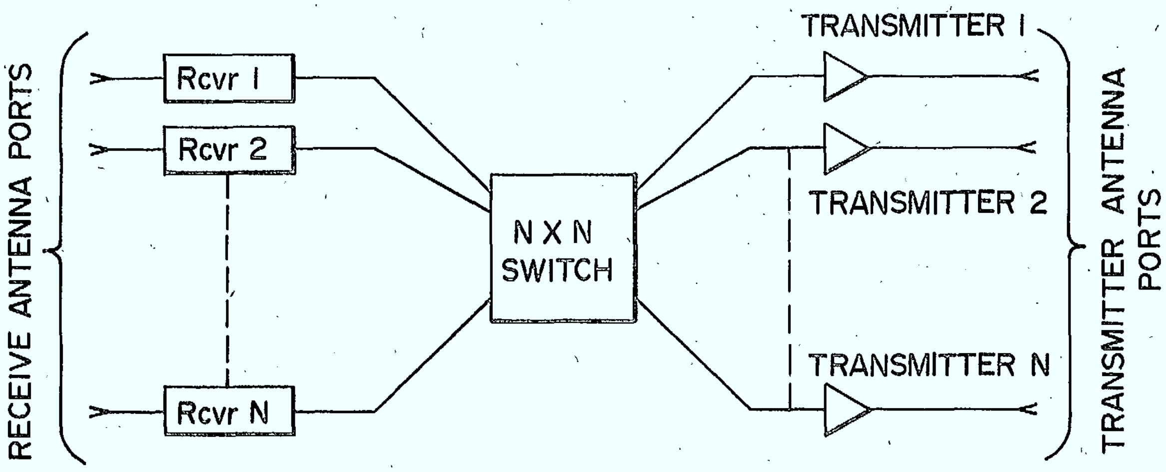

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FIG. 2.1.2. MULTIPLE BEAM SATELLITE: MODEL SYSTEM

The multiple/scanned spot beam system (Figure 2.1.2) is still in the early stages of evolution but there may be scope to use co-located spares in a similar way to that proposed in Section 2.2 for the DBS service: The total system capacity would be provided by several co-located satellites providing a fraction of the total. Each satellite would be capable of providing any of the functions of any other satellite, thus enhancing system reliability but minimizing the cost of the one spare, which would be identical to the operating spacecraft. Interconnectivity would necessitate low power intersatellite links. Preliminary studies show that use of a co-located spare in an active three satellite system, reduces each potential satellite capacity by only 9% compared to 27% for the single satellite system having the same reliability and satellite size [2]. Implementation of improved system reliability by means of an active spare is not as conceptually easy as it is for direct broadcast satellites. Much will depend on the evolution of the hardware to be used, and a very extensive cost, reliability and performance trade-off must be undertaken.

Another possibility exists for the stationing of spare satellites that is the use of a dedicated range of orbit slots for spares, either on a single occupant or shared basis, the latter requiring some test and characteristic co-ordination. A review [3] of the occupied and reserved orbital locations reveals five large, unoccupied, orbital arcs, which, could be tightly packed with non-operational spares in in-orbit storage.

The five arcs are between 10 - 35 E, 90 - 125 E, 145 - 174 E, 180 - 220 E and 300 - 325 E. This method of sparing would be particularly applicable to a multiple satellite operation where failure or unacceptable degradation of an active satellite can be predicted by monitoring performance. If the orbital drift of the replacement into the operating orbit were started to match the time to unacceptable service of the active satellite minimum fuel consumption would result. Because neither back-up for a catastrophic failure nor a revenue earning service is provided this method of sparing is unlikely to find favour until orbital arc overcrowding reaches crisis proportions, by which time, as has been pointed out, it may be totally uneconomic to fly a high capacity dormant spare.

These arcs may however provide a suitable orbit in which to checkout newly launched spacecraft prior to being drifted onto position. This is discussed more fully in Section 3.4.

(1) 'Use of the INTELSAT Space Segment for Domestic Systems', P.H. Schultze et al, Paper 76-305, 6th AIAA Communications Satellite Systems Conference, Montreal, April 1976.

(2) 'Reliability Consideration for Multiple-Spot-Beam Communication Satellites', A.S. Acompara, BSTJ, April '77 pp 575-596.

(3) 'Geosynchronous Satellite Log', W.L. Morgan, Comsat Technical Review, Volume 8, Number 1, Spring 1978.

2.2 Direct Broadcast Service (see Table 2.2.1)

2.2.1 Sparing Philosophy

Direct Broadcast Satellites (DBS) will provide a service to millions of consumers and will be supported in most applications by commercially based enterprises or non-profit organizations. These consumers, whether an existing market used to a reliable service or a new audience, which will grow rapidly to expect the same, will not be sympathetic to a system which could black out their entire TV reception under catastrophic failure conditions. The TV production companies also would find such a system unacceptable because of the impact that loss of service would have on their share of the market, the share of the TV market with respect to other media and, most obviously their daily cash flow. In non-capitalist countries the same arguments would hold true for other, equally valid reasons. Some PTTs already require that a standby system is available for immediate switching.

Not only must complete loss of service be avoided but also channel failures must be recoverable within a very short space of time. Charge rates and/or penalty clauses will no doubt reflect the would-be size of the viewing audience for the period of the loss. Outages during peak viewing times will therefore be more expensive pro-rata than for other spacecraft. Under these circumstances only a co-located spare satellite can adequately back-up the prime satellite(s).

Failure should not be a common occurence making it possible to sell the users a system whereby backup transmission uses another channel [1].

However, this approach makes inefficient use of the spectrum and is an approach which is unlikely to be accepted internationally. A co-located spare satellite having the ability to replace any failed channel is therefore the recommended method of providing backup.
Figure 2.2.1 shows a typical block diagram of a DBS transponder.

2.2.2 Costs

The possibilities for capitalizing on the need for a co-located spare to minimize the total cost of the space segment, including launch costs, was pursued and developed as a function of the actual number of spacecraft making up the total complement required to provide a given service.

High Power TWTAs have a relatively high failure rate, high cost and mass, all of which will have a first order impact on the space segment reliability and cost. The total cost will also be influenced by break points in the size and mass defined by the capabilities of the various upper stages used in conjunction with the shuttle. The physical size of the spacecraft will be dictated by the availability of surface area from which TWTs can radiate dissipated heat to space and by the characteristics of the solar array. Considerable savings are therefore made by reducing the number of TWTs deployed per system and per spacecraft, however, TWT redundancy must remain high at the system level.

Section 6.3 gives a detailed summary of the results of the analysis and the computer's estimate of the primary characteristics and costs, for each type of spacecraft. The conclusion of the analysis is that for the assumed system, the minimum cost space segment would consist of three spacecraft with one acting as a spare. Each would be able to transmit half the total number of channels in the system and would be capable of this in any combination of the allocated channels. This conclusion happens to be consistent with the sparing philosophy presently envisaged for the NORDSAT system.

Previously viewers have shown heavy resistance to increasing their TV reception capability. Commercial broadcasters may therefore decline the possibility of using DBS until the non-commercial users have established an adequate audience. Such a preoperational half-capacity satellite would alleviate the problem of balancing investment and revenue by permitting a phased build up of services and in-orbit experience. Finally the smaller satellite minimizes cost of satellite failure and therefore lowers investment risk.

(1) 'NORDSAT - A DBS System for the Nordic Countries', L. Anderson et al, Paper 78-630 AIAA 7th Satellite Communications Conference

Parameter	Value Assumed
Tx: Frequency EIRP Coverage Zone Channels/Beam TWT OP Power	12 GHz Band 500 MHz Bandwidth 54 dBW approx. Canada 4, 6 or 8 Not restricted to currently available type (150 W type)
Rx: Antenna Beamwidth	Community 1° Domestic 2°

TABLE 2.2.1 - DBS: SYSTEM PARAMETERS

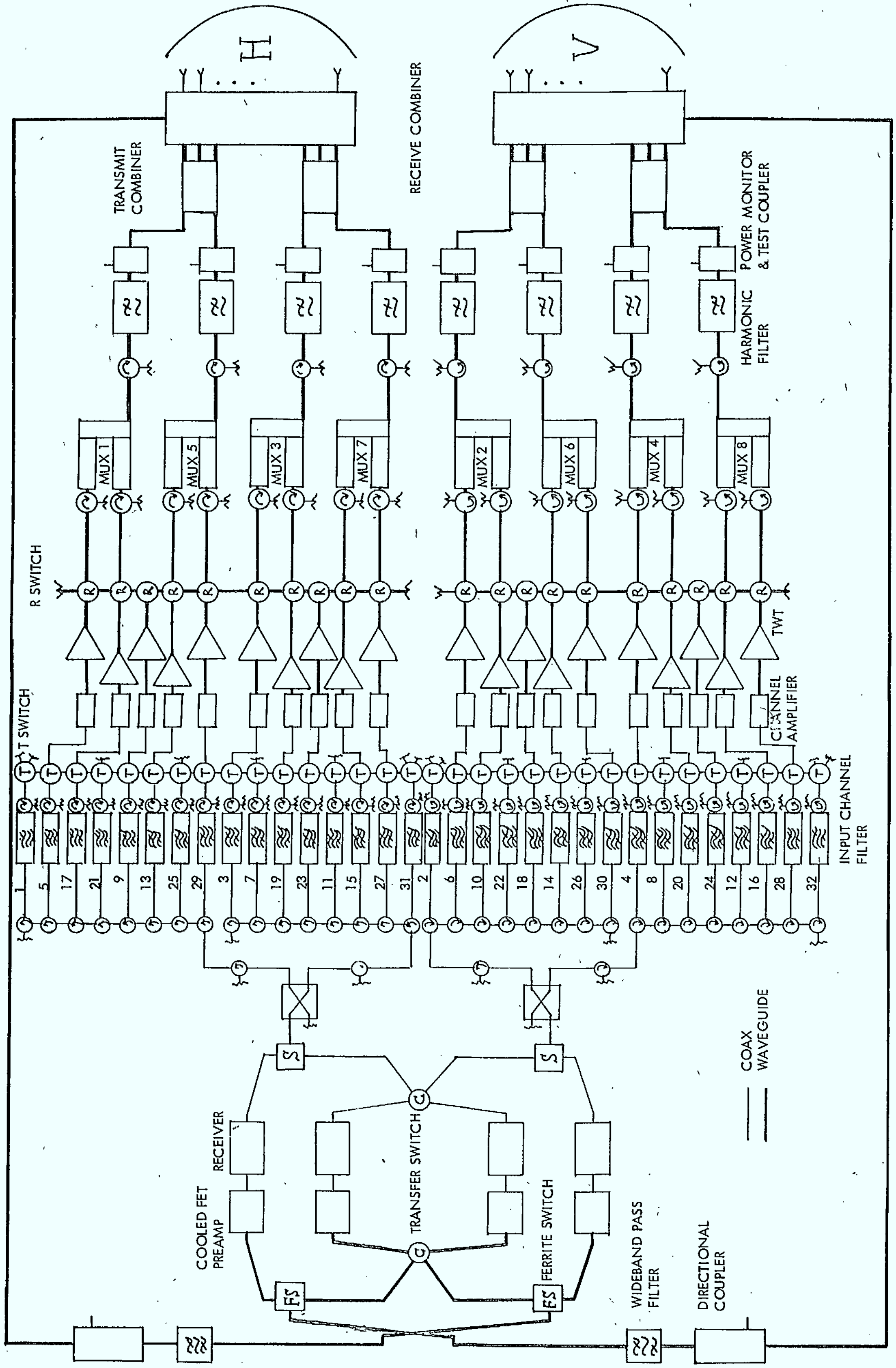


Figure 2.2.1 - Direct Broadcast Satellite: Typical Block Diagram

2.3 Maritime Service

Global ocean coverage is provided between 70° N and 70° S using three satellites in orbit above each of the major oceans, see Figure 2.3.1. Additional coverage including the polar regions remains only a possibility, which is not likely to occur before 1990. Each satellite provides earth coverage in both bands of the 'forward' and 'return' links. The basic system parameters are given in Table 2.3.1.

Parameter	EIRP	Frequency	Antenna Beam Width Tx
Satellite/Mobile	Down 25 dBW Up 37 dBW	1540 - 1542 MHz 1641.5 - 1644.5 MHz	± 9° ±12.5° Max.
Satellite/Shore	Down 3 dBW Up 65 dBW	12 GHz or 4 GHz 14 GHz 6 GHz	±9° 0.2° Approx.
MAIN OBJECTIVES	Safety, rescue, operational data and public correspondence.		

TABLE 2.3.1 - MARITIME SERVICE BASIC PARAMETERS

2.3.1 Sparing Philosophy

The single broadband transponder is more complex than a multichannel communications transponder. Normal precautions can be taken to prevent random single point failures and to provide sufficient reliability, but the system is nevertheless more open to 'unpredictable' failures. Nevertheless development of a world-wide, well patronized maritime telephone/telegraphy service will only be enhanced by the continued availability of service. The revenue, however, from the basic three satellite systems is not anticipated to reach break even for fourteen years, which is twice the design life of the first operational satellite [1]. With costs running at $20M/satellite there can be little justification for a dedicated in-orbit spare, but sparing in some form is obviously essential to guarantee a high grade of service. INTELSAT and INMARSAT have, between them, conducted economic studies of a dedicated system and a hybrid system using INTELSAT trans oceanic satellites. The continued evaluation of investment versus revenue for a

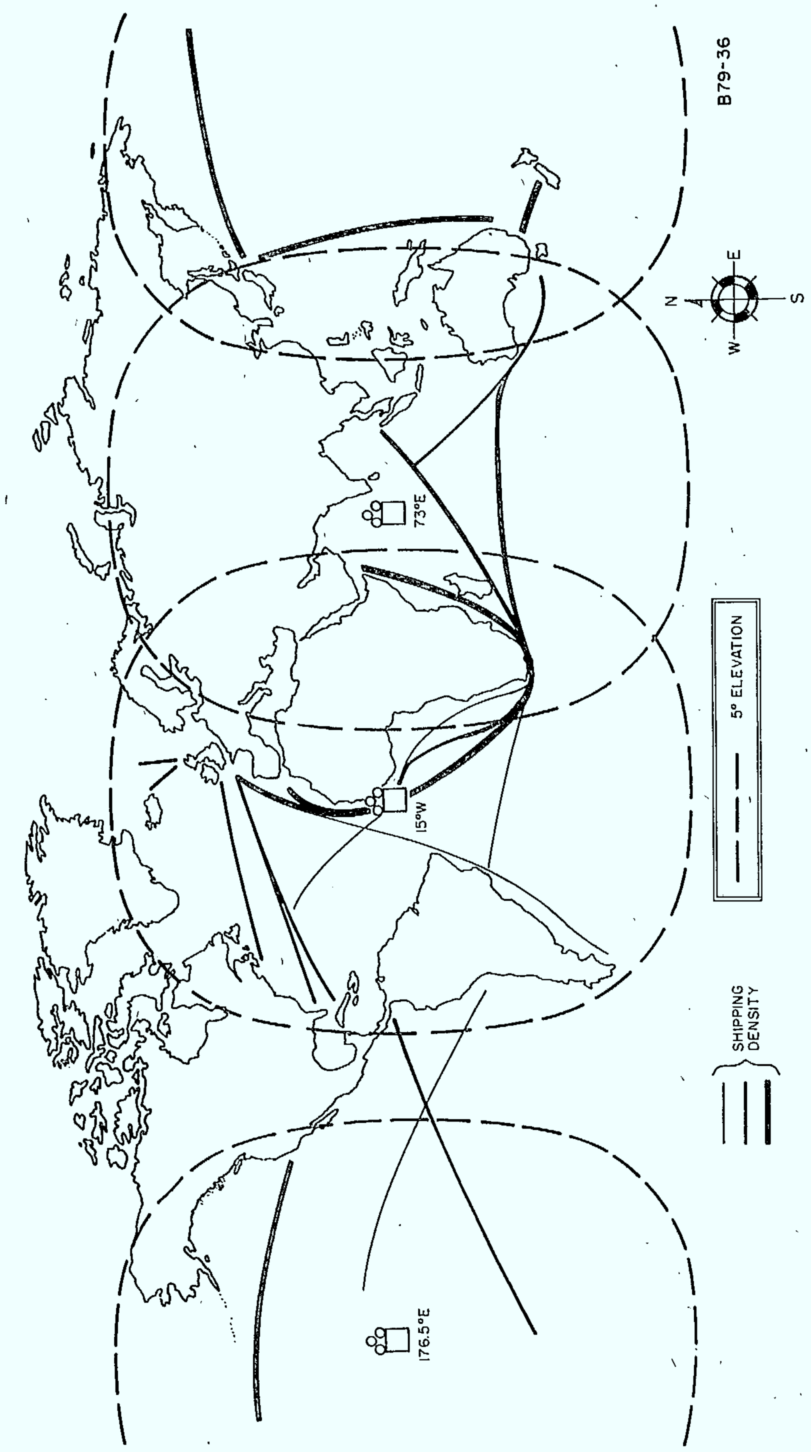


Figure 2.3.1 - Marisat Coverage and Shipping Routes

given quality of service is a vast multi parametered trade-off, which will dictate the eventual sparing philosophy.

Looking at the problem of sparing for the maritime (aeronautical) service more generally, we can say that, a) Co-located spares for all three operational orbit slots would be prohibitively expensive, b) Fewer in-orbit spares would mean inordinate outage times (up to two months) at orbit drift rates of a few degrees per day, but c) The consumers are traditionally autonomous and therefore not dependant on 100% communications coverage for any aspect of their trade, and d) The consumer is mobile and may therefore pass in and out of coverage during the voyage.

Coverage overlap in densely populated areas is a simple cheap way of providing spare capability. The most densely populated shipping route since the advent of super-tankers and the oil crisis has been around the Cape of Good Hope (See Figure 2.3.1) which is, no doubt, good cause for the duplication of coverage from the Atlantic and Indian Ocean MARISAT Satellites in this area. A similar situation arises in and around the South China Sea.

Loss of either the Atlantic or Pacific satellites would therefore have its most major impact on Air and Sea traffic in and out of the U.S. and on Atlantic shipping near the equator. A simple low cost solution exists to this weakness viz: From Figure 2.3.1 it can be seen that there is no coverage of the Central American coast, the Great Lakes, Hudson Bay or the Gulf of Mexico. Coverage of these areas justifies an operational orbit position, but the traffic probably doesn't warrant a satellite of full oceanic capability. A reduced package flown as part of a multi-purpose satellite (capable of operating with 6/4 GHz ground stations) in orbit over the Western Atlantic could well provide the back up needed, especially if the coverage were to abut that of the Indian Ocean Satellite. A suitable 'host' would be the INTELSAT satellite positioned in the 325.5°E slot or BRASILSAT at 300°E, the latter giving better coverage of the US Pacific Coast. Both these spacecraft work with the 4/6 GHz band. The INMARSAT system will probably utilize the 6420-6425 MHz uplink band, and so there will be a need to plan the compatibility of the two systems spectrum usage. The mobile antennas are fully steerable to the new orbit position and have a small enough beamwidth that interference should not be a cause of problems. Figure 2.3.2 shows the orbit locations of the current MARISAT spacecraft and also shows the extent of the mobile antennas first null when pointing at a given satellite. The remaining arc is available for both active or spare spacecraft.

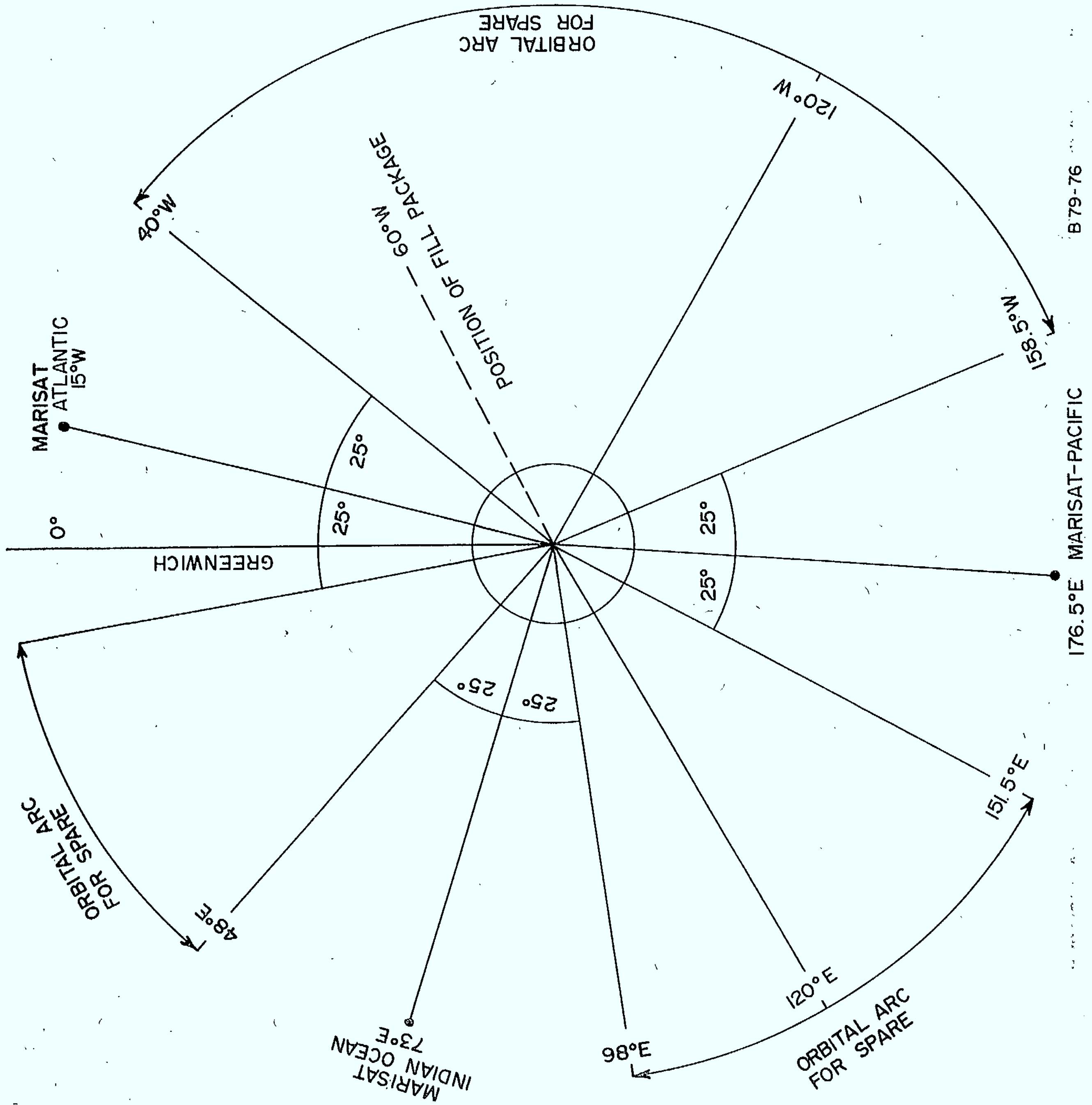


Figure 2.3.2 - MARISAT Orbit Locations

Loss of the Indian Ocean satellite would cause complete loss of service between 60° E and 100° E, since neither the Atlantic nor Pacific spacecraft could be moved without loss of contact with their U.S. ground stations. Even an in-orbit spare would not alleviate the loss since this would logically be no further from the most heavily used satellite than necessary to minimize outage and interference during test. However, the population density between these longitudes is very much less than that of theAtlantic and Pacific (see Table 2.3.2). Of these ships, those that are not close to the Indian sub continent, are on long straight ocean passages. These ships would not be so in need of communication as those operating in confined waters or the major routing decision areas such as the equatorial Atlantic. It is feasable to envisage a very much reduced capacity maritime package being flown on the Indian Ocean INTELSAT.

AREA	NO.
Atlantic	9,500
Pacific	7,500
India	2,500
Misc.	1,500
Total	21,000

TABLE 2.3.2 - SHIP POPULATION: 100 GROSS TONS AND OVER (1969)

1) INMARSAT 'The International Maritime Satellite Organization, Its Genesis, Development and Status William T. Adams, AIAA 7th Satellite Comms Conference Paper 78-552.

2.4 Summary of Sparing Philosophies

The predicted saturation of the goestationary orbital arc with satellites operating in the 6/4 and 14/12 frequency bands will in future cause the sparing philosophies for each category of service to be tailored specifically to the application.

Heavy Route systems employing steerable ground antenna whether regional or international might continue to employ a spare in a fully operational orbital slot. The spare would be loaded with pre-emptible traffic, such as low cost leases. The satellite system would thereby form an integral part of the total system route diversity scheme which would encompass terrestrial and/or cable links. The spare satellite would be essential for rapid restoration of the route in accordance with CCITT recommendations.

Projections up to 1990 for the Atlantic International service indicate that no increase of allocated orbital arc will be necessary although a reduction of satellite spacing to 3° will be necessary, which is compatible with the antenna beamwidths at the higher frequency bands that will then be in use.

Regional (Domestic) system projections must be integrated with other users demanding the same orbital arc. It is therefore more difficult to predict the availability of slots for spare satellites, but it is clear that some constraints on spacecraft design such as making frequency re-use mandatory will become necessary. The sparing philosophy employed will depend on required restoration time, types of ground station, fuel used to relocate and the type of traffic. The possibilities are more fully developed in Section 3. In general, the shared dedicated dormant slot will not be required because, inevitably, there will be fewer spares then operational satellites and therefore each spare should have a dedicated interstitial slot. The high capacity and customized coverage areas of multiple or scanning beam satellites may require a further type of sparing philosophy. The high cost of a sophisticated craft such as this could alter the cost-benefit of having a spare in orbit. An alternative is to co-locate multiple satellites each of which supplies a fraction of the total system capacity. Each would also be equipped with a certain flexibility to provide redundancy within the system, and, of course, intersatellite links for interconnectivity.

The Direct Broadcast Satellite service benefits significantly from having co-located in-orbit spares. Large cost and complexity savings are available whilst the system reliability is augmented. A fuller description can be obtained from Section 6.3.

In view of the marginal funding available for the Maritime satellite system, the recommended method of providing back-up is by means of coverage area overlap for the most densely populated traffic routes, using three dedicated satellites. Reduced capacity transponders forming part of Intelsat payloads would provide the additional coverage which is needed over the Indian Ocean and American waters (East and West). There is little justification for a dedicated in-orbit spare.

These conclusions are tabulated in Table 2.4.1.

Service / Spare Arrangement	Fixed				DBS	Maritime
	Regional			International		
	Cat. A	Cat. B	Cat. C			
— Fully operational spare with preemptable users	X			X		
— Dedicated Dormant Spare Slot		X				
— Shared Dormant Spare Slot						
— Co-located with Operational Satellite		X	X		X	
— Other						X(1)

(1) Sparing using overlap of coverage areas and/or reduced capacity transponder on other satellites

TABLE 2.4.1 - SUMMARY OF APPLICABLE SPARING METHODS FOR DIFFERENT TYPES OF SERVICE

3. SPARING FOR A MODEL REGIONAL FIXED SERVICE SYSTEM

Demands for allocation of orbital arc will, in the foreseeable future cause an overcrowding of crisis proportions. Today these demands for frequency and orbit usage are recorded in the IFRB Master Register on a first come first served basis. The frequency bands are regulated by the ITU but the orbit spacings are determined mainly by the width of ground station antenna beams and the allowable levels of interference with other satellite or ground systems operating at the same frequency. As more nations develop the use of fixed service satellites the pressure on certain orbital arcs may cause reallocation of certain orbital slots. This section therefore examines three of the more likely developments of a model regional fixed service depicted in Figure 3. It is normal at present to have back up satellites in orbit in fully active slots, however, as a result of orbital congestion, it may be necessary to locate a dormant spare satellite in an interstitial position between two operating satellites. Alternatively, the spare may be co-located with an operating spacecraft to provide immediate backup. The advantages and disadvantages of these approaches are discussed in the following sections.

3.1 All Satellites in Active Orbital Positions

In a congested section of arc the use of a fully operational slot for a spare will only be justified provided a) all satellites in the system are heavily loaded and b) that the spare is used to carry peak or pre-emptible traffic, which in case of satellite failure can be routed by other space or ground links. This form of sparing is particularly appropriate when a system consists of the second and subsequent generation satellites. As was discussed in Section 2.1.1, residual life of older generation satellites and the increased capacity of the new generation can be used to provide a very cost-effective backup capability. Because this 'spare' is in revenue earning service, the health of each channel will be apparent from the live traffic performance. No special monitoring procedures or hardware for testing are required.

Unfortunately this method of sparing has a severe disadvantage because of the impracticality of re-pointing numerous fixed ground stations typified by those distributed throughout northern Canada to provide thin route service. The highest grade of service, implying minimum outage time can therefore only be given to fully steerable ground stations.

3.2 The Interstitial Spare

The interstitial position of a spare satellite provides the system with neither traffic capcity nor instant backup. It is an approach which can only be justified when there is overcrowding in the relevant orbital arc.

Figure 3.2.1 depicts the single satellite system or widely separated multi satellite system. The active/spare separation would be as small as possible to minimize outage time and interference to neighbouring systems but would be limited by the uplink beamwidth.

Figure 3.2.2 shows the interstitial spare backing up two operational satellites. Logically the spare would be midway between the operational satellites if drift time to either is to be minimum. All interference is internal to the system. Testing may therefore be coordinated with the operational requirements to minimize the impact.

In a three satellite system (Figure 3.2.3), the spare would obviously be used to support the satellite/s which provide the high priority traffic or service any fixed ground antennas. Interference is again internal and can be coordinated.

The following section shows that the interstitial spare can be fully tested in orbit, subject to the adoption of a modification to the current ITU regulations. No additional on-board hardware is required but there will be restrictions on the test duration.

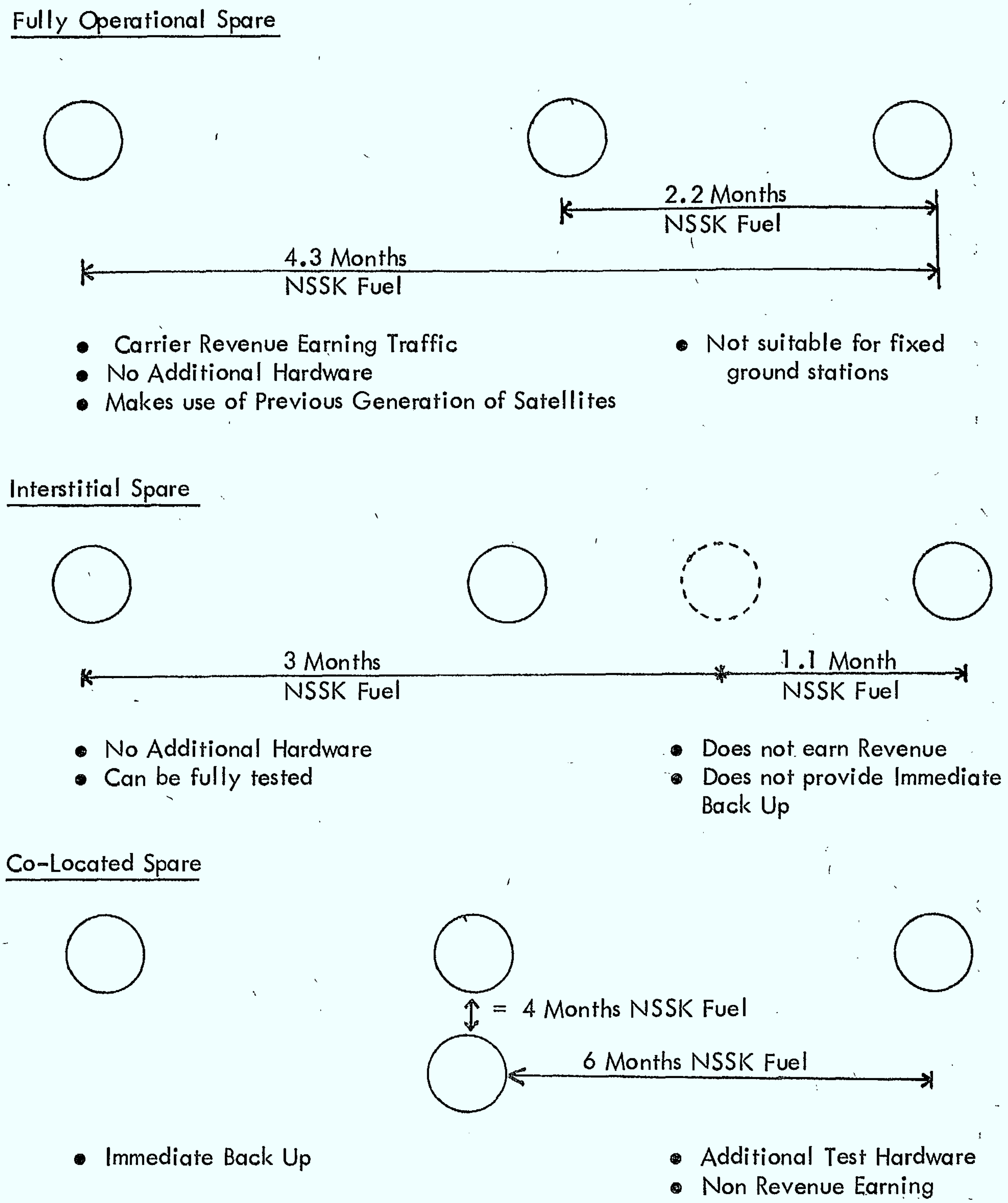
3.2.1 Interference Caused by Testing the Interstitial Spare

It has often been demonstrated that the capacity of the orbital arc is greater if there is homogeneity in the characteristics of adjacent satellites. In practice only segments of the arc will be occupied by similar satellites. The problems of using the interstitial position at segment broundaries can be disregarded because far fewer spares than operational satellites will be in orbit and the disadvantageous interstitial positions can be left empty. Only the interstitial position between two similar satellites is considered further.

CCIR Recommendation 466-2 concerns itself with Interference caused by adjacent systems. It is helpful for the following arguments to quote the following extracts:

"2. that the maximum level of the interference noise power at a point of zero relative level in any telephone channel of the hypothetical reference circuit of a geostationary satellite network in the fixed-satellite service employing frequency modulation, caused by the transmitters of another fixed-satellite network, should not exceed 400 pWOp, psophometrically weighted one minute mean power, for more than 20% of any month;"

SATELLITE SPARING ALTERNATIVES



ADVANTAGES/DISADVANTAGES AND FUEL CONSUMPTION SUMMARY

FIG. 3

Assumptions: 5 days drift time and 5^{o} separation between satellites

X X X O X X

Figure 3.2.1 – Interstitial Spare

: Single Satellite or Widely Separated System

X Operational Slots

X Assigned operating positions

O Interstitial Spare

X X X O X X

Figure 3.2.2 – Interstitial Spare

: Two Adjacent Operational Satellites

X X X O X X

Figure 3.2.3 – Interstitial Spare

: Three Adjacent Operational Satellites

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and that:

"3. that the maximum level of interference noise power caused to that network should be calculated on the basis of the following values for the receiving earth station antenna gain, in a direction at an angle θ (in degrees) referred to the main beam direction:

$$G = 32 - 25 \log\theta \, dB \, (1^{\circ} < \theta < 48^{\circ})$$ "

Other regulations are summarized in Table 3.2.1.

Referring to Figure 3.2.4, extract (3) can also be used to define the increase of interference as a result of activating the interstitial position, i.e. the ground antenna gain in the direction of an adjacent active position is 32 - 25 log θ dB, whilst the same antenna has a gain in the direction of the interstitial spare of $32 - 25 \log \frac{\theta}{2}$ dB.

The increase of antenna gain is therefore:

$$\left(32 - 25 \log \frac{\theta}{2}\right) - (32 - 25 \log \theta) = 25 (\log \theta - \log \theta/2)$$

$$= 25 \log 2$$

$$= 7.5 \text{ dB}$$

provided the orbital separation of the interstitial spare exceeds 1°.

The other system parameter that should be taken into account is the variation of flux to saturate of adjacent operational satellites. For the purpose of this argument a value of 6 dB is deemed more than adequate.

The net change of interference level when the source of interference originates from the testing of an interstitial spare instead of an adjacent operational system is as follows:

Uplink:	Additional Sidelobe Gain	+7.5 dB
	Tx Power	-7.5 dB
	Operating Margin	+6 dB
	Net Interference Level	+6 dB
Downlink:	Additional Sidelobe Gain	+7.5 dB
	= Net Interference Level	

Normally the carrier to noise density of a satellite uplink will be better than or equal to that of the downlink. The worst case increase of interference will therefore be if the whole of the 400 pWOp, allocated to a single source of interference by Recommendation 466-2, is considered to be attributable to the downlink.

Operation of the interstitial spare may therefore cause a total interference level of 2250 pWOp.

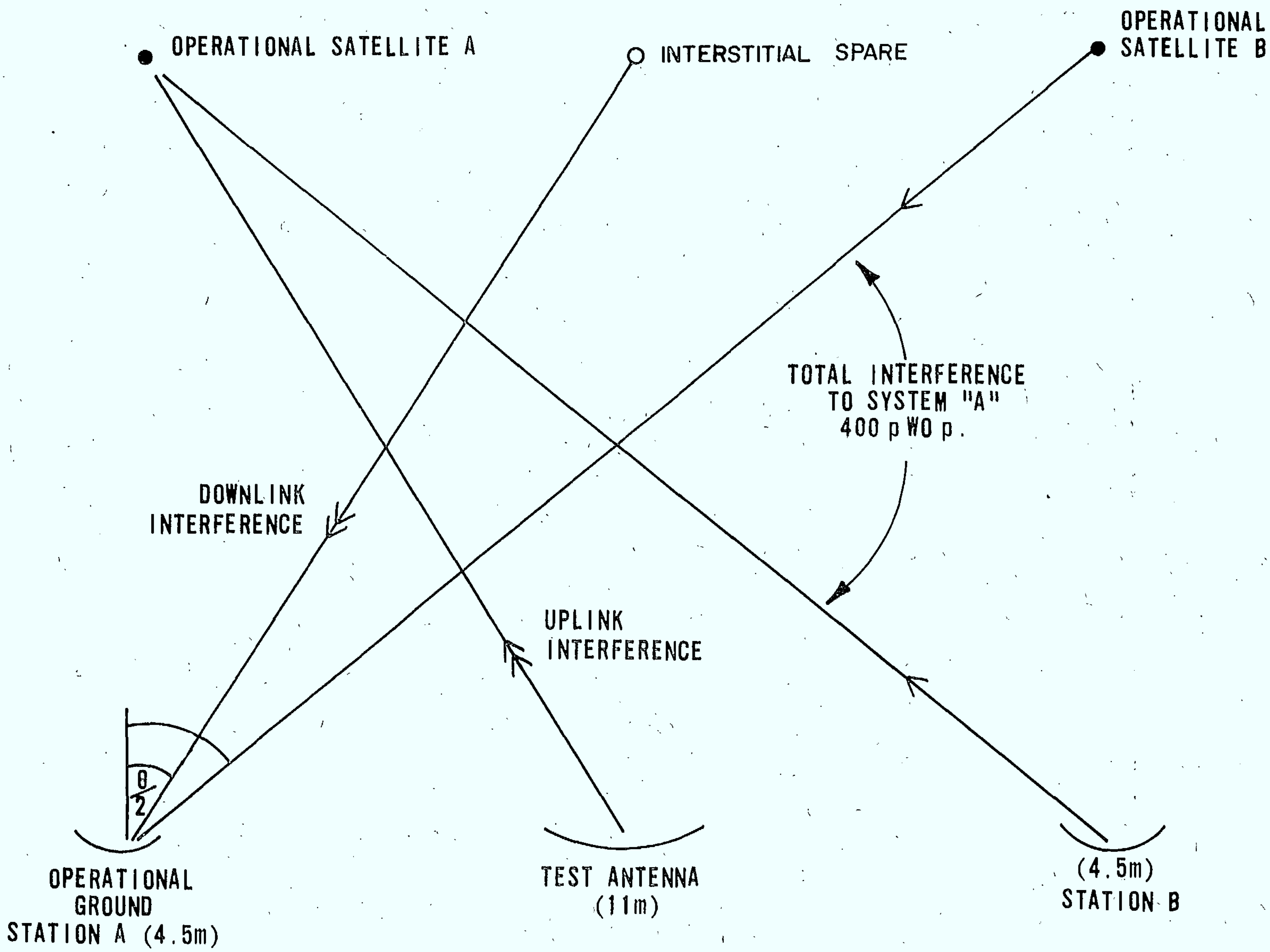

Figure 3.2.4 - Interference from the Interstitial Spare

The first extract puts a limit on the worst case interference from one system into another. The most susceptible system would involve the use of 4.5m dia. operational ground antennas, but we can assume for the purposes of testing an interstitial spare that a much larger antenna would be available requiring less transmit power.

Extract (3), however, does not distinguish between antennas of different sizes when defining sidelobe gain, so the actual interference will be less significant by the ratio of the on-axis gains or 20 log (11/4.5) = 7.5 dB.

This is obviously not compatible with the current wording of Recommendation 466-2, which applies only to satellites operating below 10 GHz. Currently, there exists a proposed modification to the Recommendation to extend the applicable frequency range by catering for increased interference as a result of up link fading caused by rain attenuation at frequencies above 10 GHz. The proposal would add to extract (2):

"and 2000 pW psophometrically-weighted one minute mean power for more than 0.3% of any month".

The additional interference caused by full power testing the interstitial spare is approximately covered by this proposed modification of the wording, provided the testing duration per channel does not exceed a fraction of the 0.3% (2.2 hours).

An automated test set would significantly reduce the test time per channel and a further time saving can result from using low level signals when not required by the nature of the test (EIRP and Flux-to-saturate). Similar Comsat practice permits these tests on each channel to be completed in approximately 5 minutes or 0.01% of a month.

It is felt, therefore, that a revision to CCIR Recommendation 466-2 is also necessary to permit interstitial spare testing, but the content and wording of the proposed modification needs to be reviewed in the context of the new requirement.

Post-launch commissioning tests on Intelsat satellites commence immediately after apogee burn, when the satellite is still drifting. This practice may continue provided the tests take place when drifting through unoccupied orbit segments. These will more likely exist on either side of the oceanic Intelsat locations than Regional satellite locations, where up to ±40 degrees may be occupied by operational satellites. In the event that all the commissioning tests cannot be completed before arriving on station, the remainder may be carried out in the interstitial position provided the time restrictions on interference levels are adhered to.

TABLE 3.2.1 - SUMMARY OF ITU REGULATION FOR FIXED SERVICE SATELLITES

SUBJECT	RADIO REGULATION	SUMMARY	REMARKS
Earth Station Minimum Transmitting Antenna Elevation Angle	470 L	3° above the horizontal plane (lower by arrangement)	In US 6° (down to 3° by arrangement)
Power flux density at earth's surface in shared service bands	470 NL	Units $dBW/m^2/kHz$ -152 $(0 < \theta < 5^\circ)$ $-152 + \frac{\theta - 5}{2}$ $(5^\circ < \theta < 25^\circ)$ -140 $(25^\circ \leq \theta \leq 90^\circ)$	All methods of modulation 3400-7750 MHz
Stationkeeping	470 VC, VD, VE	$\pm 1^\circ$	$\pm 0.5^\circ$ target
Pointing Accuracy	470 VF	10% of half power beamwidth or 0.5° whichever is greater.	If necessary to avoid interference
Interference	466	Existing System:- Aggregate: 1000 pWp for 20% of any month From any one satellite: 0.4 of the total (400 pWp) New System:- 2000 pWp for 20% of any month (system without frequency reuse) 1500 pWp for 20% of any month (system employing frequency reuse)	FDM/FM satellites below 10 GHz

TABLE 3.2.1 - continued

SUBJECT	RADIO REGULATION	SUMMARY	REMARKS
Interference (continued)		The total interference power level averaged over any ten minutes should not exceed for more than 20% of any month X% of the total noise power level at the input to the demodulator that would give use to a bit error rate of 1 in 10^6, where X X = 20% (new systems no frequency reuse) = 40% (new systems frequency reuse) = 10% (existing systems)	PCM Systems
Uplink EIRP		$32-25 \log \phi$ dBW/4 kHz max $2.5^\circ < \phi < 48^\circ$ -7 dBW/4 kHz $48^\circ < \phi < 180^\circ$	
Various	446 356	Carrier Dispersal techniques Interference from terrestrial links to satellite systems	Corollary to 466

3.3 The Co-located Spare

The co-located spare has the distinct advantages that it is immediately available as a backup to the prime satellite in the system, and that command and telemetry links can supply the majority of health tests (see Sections 4 and 5). To replace any other satellite other than the prime is, in general, more time consuming that it would be from an interstitial position. The optimum arrangement would be to position the prime satellite and the spare centrally with respect to other satellites in the system, as indicated in Figure 3, and thus minimize the drift time to secondary positions. Additional flight hardware is necessary to test the co-located spare with anything other than live traffic, which would have to be temporarily routed away from the prime satellite. Section 5.2 details the hardware requirements for various transponder designs. Unfortunately it would be necessary to incorporate this additional hardware into all the spacecraft to preserve commonality and hence reduce costs. This hardware represents a significant mass that could otherwise be used for fuel for NSSK or relocation. No revenue earning service is possible from the co-located position and post launch commissioning tests present a problem. Without heavy expenditure of fuel the simplest method of commissioning the co-located spare would be to first occupy an interstitial position, test the satellite without infringing the interference regulations and then move it into its final position.

3.4 Dedicated Commissioning Slot

It can be seen from plots of the present and predicted use of the geostationary orbital arc Figure 1.1 that large sections of arc will remain unoccupied into the foreseeable future. At each edge of the occupied arc there are therefore orbital slots which could perform a useful task as spare storage areas or provide a dedicated commissioning slot, where newly launched spacecraft could be fully checked out and calibrated before drifting into its allocated orbital position. This is especially useful when the satellite is to occupy an interstitial or co-located spare position. However, a fuel penalty is incurred in bringing the satellite to rest and starting it moving again. The fuel penalty can be calculated from the data in Section 6.1 to be approximately 64 days of NSSK fuel expended to achieve one degree per day drift rate.

The fuel spent to cause drift and the time spent drifting both detract from the satellites useful life remaining. Fuel spent on non-essential manoeuvres will not be available for NSSK, whilst further fuel must be expended to correct inclination build-up during the drift time.

The minimum loss of satellite life caused by relocation is when the drift time equals the loss of NSSK fuel-days (ANIK-D: 12.7 kg = 1 year approx), Figure 3.4.3 demonstrates this. Minimum loss of life is plotted for all possible relocation angles inFigure 3.4.1, whilst Figure 3.4.2 is a plot of fuel used against the drift time, which does not include allowance for attitude corrections, commissioning etc. It shows that moving a satellite from one location to another in one or two days requires a considerable expenditure.

For the utlimate destinations centrally located in densely populated segments of arc the life penalty is unacceptably large, e.g. a satellite wishing to reach 99^{o}W from a position at the extremes of the occupied orbital arc would expend approximately 100 days of life. The commissioning slot cannot be recommended therefore.

Despite this, the commissioning slot may find favour for two particular applications. Firstly, DBS satellites are expected to take station several degrees west of their nominal service area, where local midnight will cause the eclipse period to occur during the normal close down hours for television. These satellites will therefore tend to be at one extreme of the occupied arc over any given land mass and therefore closer to the check out slots. It was proposed in Section 2.2 that the optimum DBS space segment would consist of several co-located satellites each of which would be capable of operating in any of the allocated channels but would operationally do so in only a fraction of these. Pre-operational testing of all the channels of any one satellite on station could therefore only take place, without causing interference, in the television close down hours. The check-out slot is very appropriate therefore for flexing the DBS during its in-orbit commissioning test, without imposing these time restrictions.

Secondly long drift times would not be an influencing factor in the use of check-out slots for spare storage provided wearout of an operational satellite can be predicted many months ahead of the actual date. This negates the significance of loss of life through drift time and allows minimum relocation fuel to be spent (see Figure 3.4.2). Catastrophic satellite failure is not supported by this method of sparing but this may nevertheless be a compromise solution which will be forced on users by the demands for operational orbit slots in the future.

TOTAL LOSS OF DAYS OF USEFUL LIFE = DRIFT DAYS + EQUIV FUEL DAYS

$L = 64R + \theta/R$, WHERE R IS DRIFT RATE (°/DAY), θ IS RELOCATION ANGLE (°). 64 DAYS OF FUEL ARE USED PER DEG/DAY DRIFT.

TO FIND MINIMUM LOSS OF LIFE $\frac{dL}{dR} = 64 - \frac{\theta}{R^2} = 0 \quad \therefore \quad R = \sqrt{\frac{\theta}{64}}$

$$L\ MIN = 2\sqrt{64\theta} = 16\sqrt{\theta}$$

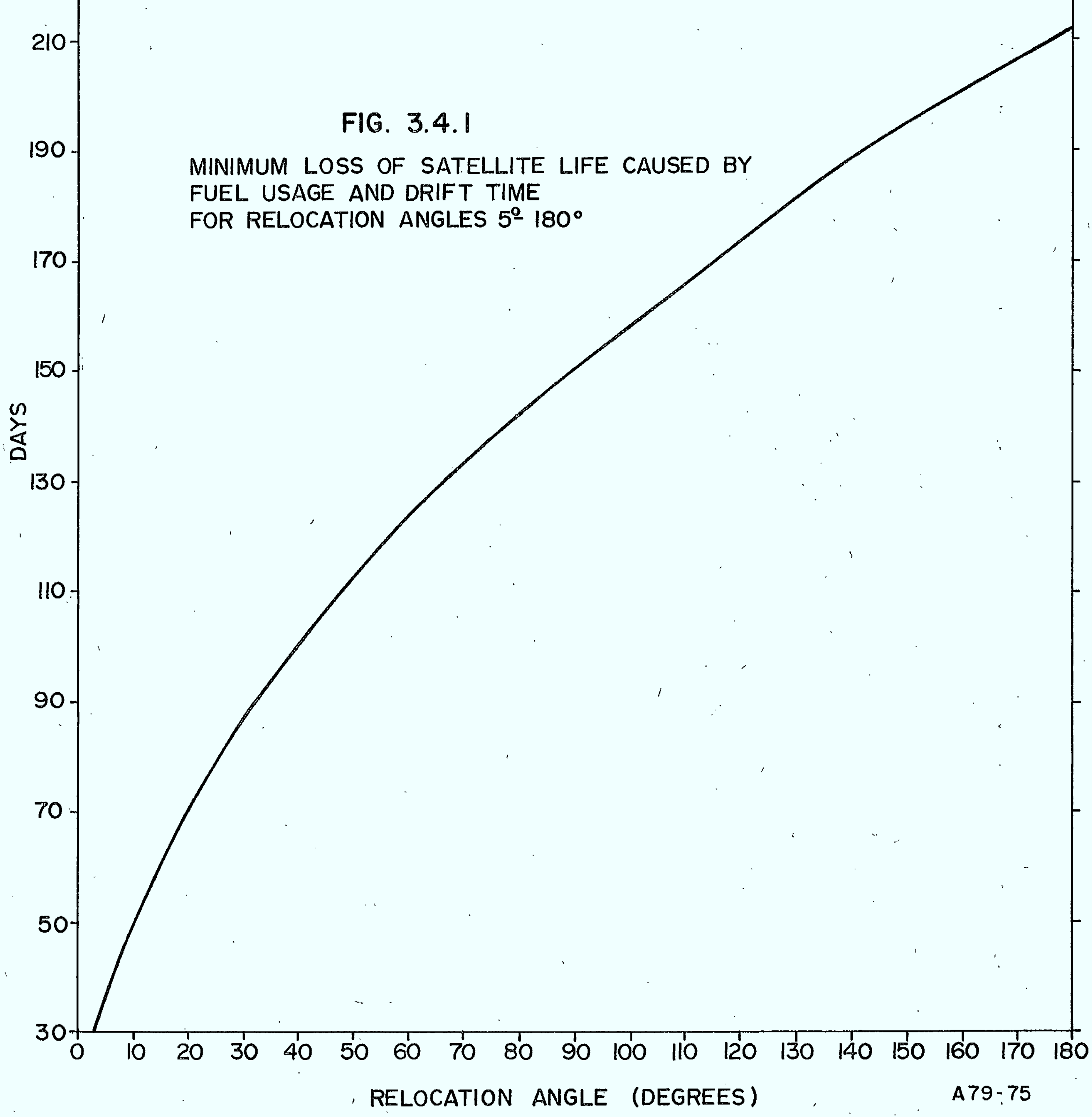

FIG. 3.4.1

MINIMUM LOSS OF SATELLITE LIFE CAUSED BY FUEL USAGE AND DRIFT TIME FOR RELOCATION ANGLES 5° 180°

A79-75

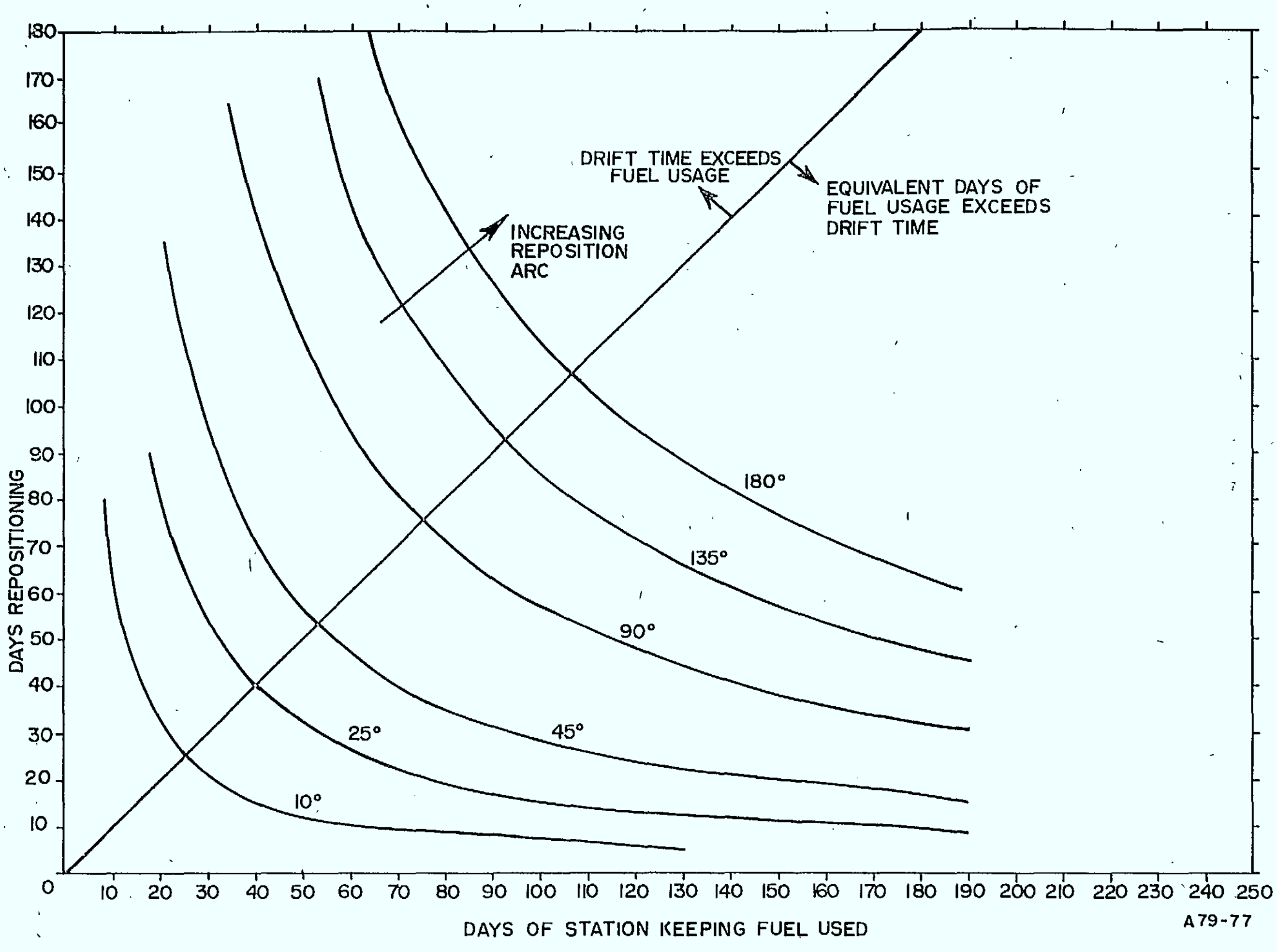

Figure 3.4.2 - Satellite Fuel Consumption for Longitudinal Relocation (large angles)

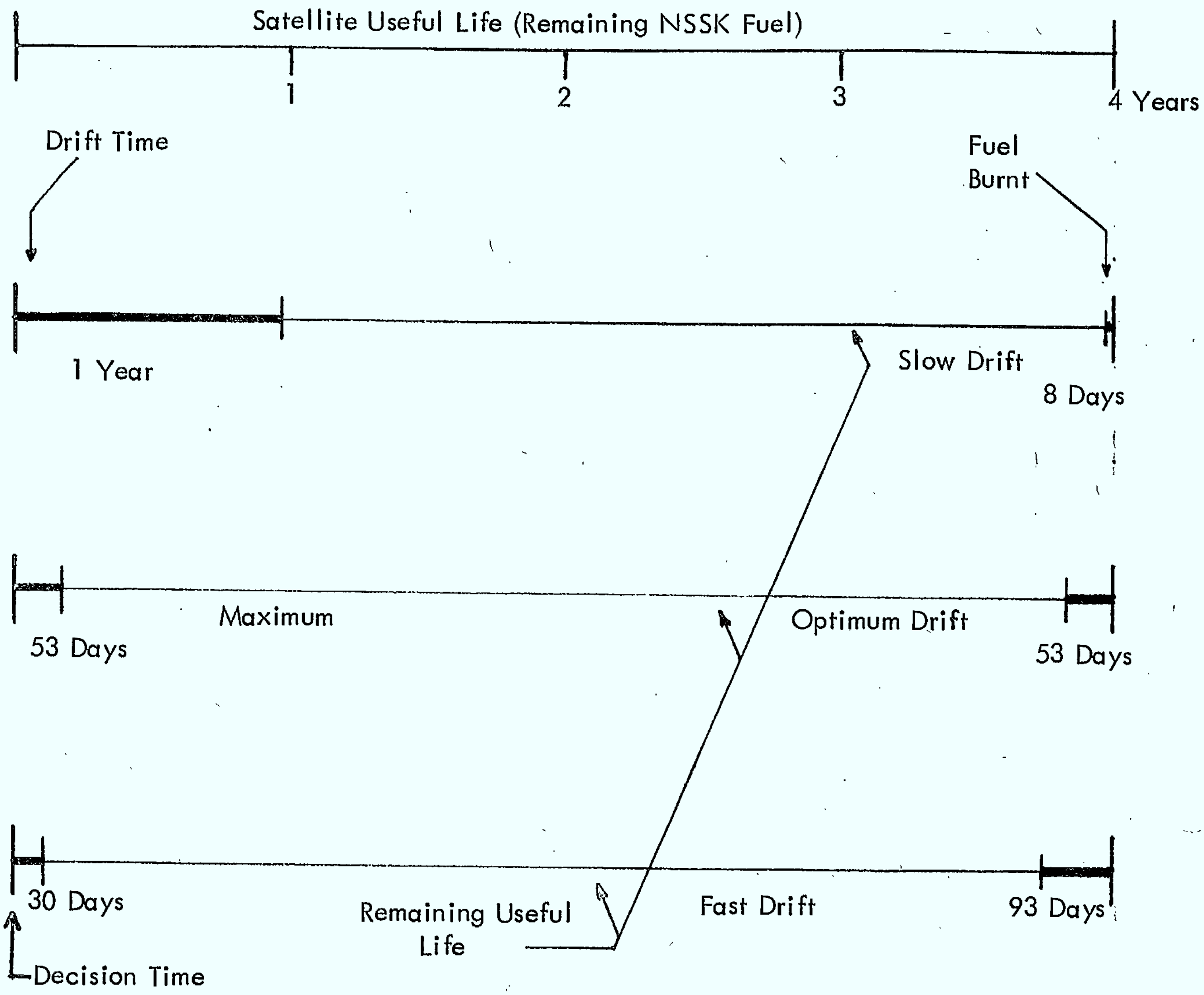

Figure 3.4.3 - Definition of Optimum Drift Time

In this example:

A satellite with four years of NSSK fuel remaining is called upon to replace another located 45° away. The diagram shows three of the options open to the operating agency, when deciding how quickly the replacement is required.

Power Monitor and the TWTA Helix Current Monitor are also checked at this time.

The annual and six monthly tests are performed to check the performance capability of the communications payload of the spacecraft. The spacecraft housekeeping functions e.g. position, altitude, thermal environment, etc. are checked at more frequent intervals.

4. SPARE SATELLITE HEALTH TESTING

4.1 Current Practice in Spare Satellite Testing

After being placed on station in the geostationary orbit, the spare satellite is subjected to the same pre-commissioning test program as the active satellite. The successful completion of the tests constitutes an important in-orbit incentive payment milestone. Thereafter, the spare satellite is subjected only to those tests that will ensure that it can be brought into service, when required, with a minimum of service outage, and that are necessary to provide proof of performance to qualify for in-orbit incentive payments. As a minimum, a typical spare satellite test program would consist of the following tests performed once a year:

— Saturating power flux density

— e.i.r.p.

— Translation frequency stability

— Channel amplitude response

These parameters would be measured for each channel and for all the primary and redundant paths. The switch-over from primary to redundant paths would also be investigated. The measured data and the telemetry outputs are used not only to verify that performance is being maintained but also to check for any trends in their behaviour. Such a test would, for a 24-channel satellite, take from one to two days to complete.

In addition to the annual tests, it is the practice to exercise some subunits at periodic intervals of from 3 to 6 months. The subunits exercised are the TWTAs, the substitute heaters and the redundancy switching. The TWTAs are switched on and then left on for about 50 hours with no RF drive applied. This serves to reduce the residual ion gas pressure build-up, if any, in the TWTAs. Some spacecraft operators perform the "time to knee" measurement of a TWTA to indicate its lifetime degradation but this is not the general practice because of the difficulty in interpreting the results for a particular TWTA. The operation of the substitute heaters is also checked to ensure that the proper thermal environment is maintained in the spacecraft. The switch-over to the redundant paths is also checked for each channel. The continuity of a signal is generally used to indicate satisfactory switch-over. The operation of the RF Output

4.2 Abbreviated Health Monitoring of a Co-Slotted Spare

Co-location restricts the scope of health testing that can be performed on a spare. In theory it would be convenient to swing the coverage area of the spare onto a geographically distant test location, but this imposes design constraints on either the antenna positioning mechanism or the attitude control equipment. Usually mechanisms operate on the antenna reflector because less movement is required and implementation is easier. However, the displacement angle available is unlikely to exceed the operational beamwidth. Beam isolation cannot be obtained therefore.

Satellite inversion as a possibility doesn't survive even a cursory glance at the globe. Canada transposes into Cape Horn and the South Pacific Ocean, where the only ground station plotted on the INTELSAT chart is domestic and at the tip of Chile. Europe transposes to ocean areas south of the Cape of Good Hope. A similar situation exists for the India sub-continent. The only areas where this technique could be readily applied are Africa, South America and the Far East/ Australia. Only the latter would appear to offer the political stability required for such a solution. This method of testing would also mean the possible loss of command security, territorial infringement and loss of direct control over the testing.

Co-location of an active satellite with its spare also precludes the possibility of making tests which involve radiation from the spare in the operational communication channels. These are:

— Saturating Power Flux Density
— E.I.R.P.
— Translation Frequency Stability
— Channel Amplitude Response

Reception is obviously still possible but transponder testing would involve the use of measurements using telemetered helix current and r.f. power monitor data, which would have doubtful accuracy over a long period of time and cannot be recommended.

Using a single test set-up and fixed link frequencies which are unique to a particular satellite, meaningful measurements can be made of each of the above excluding the Channel Amplitude Response. The latter is largely dictated by passive mechanical units such as the input and output multiplexers, which should not experience as rapid 'wearout' as would active devices such as TWTs. It is assumed therefore that monitoring of Channel Amplitude Response need not take place as often as the other tests.

The proposed test method requires modifications to the satellite in orbit and the earth station, which are fully described in Section 5. Because the test frequency is unique to the satellite under test the method has several advantages, not least of which is that the testing can take place without causing interference to the active satellite. A single frequency test also has diagnositc advantages, since the communications receiver amplitude response will not be a variable in deducing the cause of performance change e.g., a change in flux-to-saturate of a fraction of the TWTs, whilst others do not change would immediately suggest degradation of the TWTs because receiver gain is the same for all tubes. The need to correlate data from tests using various combinations of receivers and TWTs is reduced. The telemetry downlink will be required simultaneously so that various payload parameters are available for correlation with test results.

The test uplink frequency may lie anywhere in the command band, (Figures 4.1.1 and 4.1.2)but could quite readily be the actual command frequency, thus minimizing complication in the earth station. The downlink test frequency will have a constant difference from the uplink defined by the spacecraft local oscillator frequency. The frequency plans described by the figures show that this is possible in both examples.

In addition to the above tests, the test set-up is also capable of performing the following tests:

— TWT turn-on transient performance
— Gain transfer characteristic
— Transponder noise temperature

A test procedure for each test is given in more detail in Section 5.1.3.

Before launch it will be necessary to establish a relationship between the test results in band and at the test frequency. Drift may therefore be detected with respect to the test reference at the beginning of life and applied to the corresponding in band performance. When a troublesome amount of drift has been recorded a cross-check should be performed at the channel frequency (at an operationally convenient time). This may also prove an opportune occasion to determine channel amplitude response.

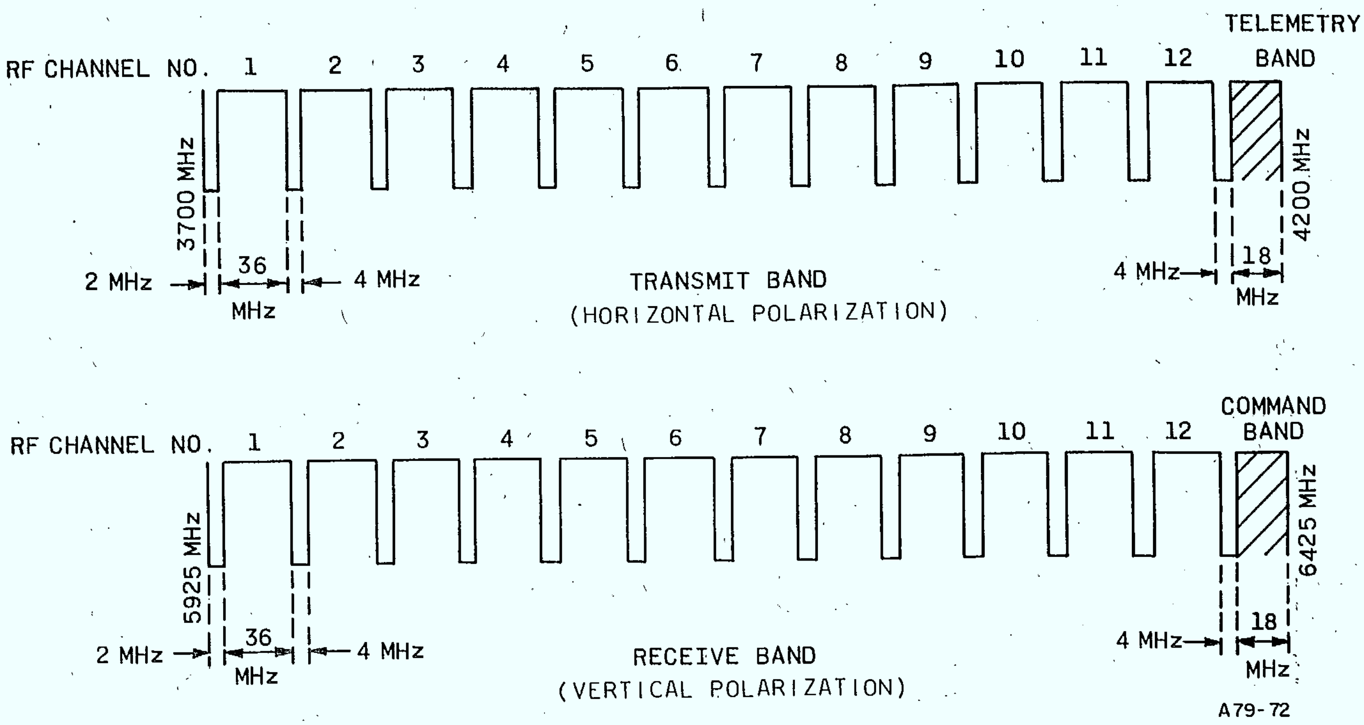

Figure 4.1.1 - ANIK-D: 12 Channel Frequency and Polarization Plan

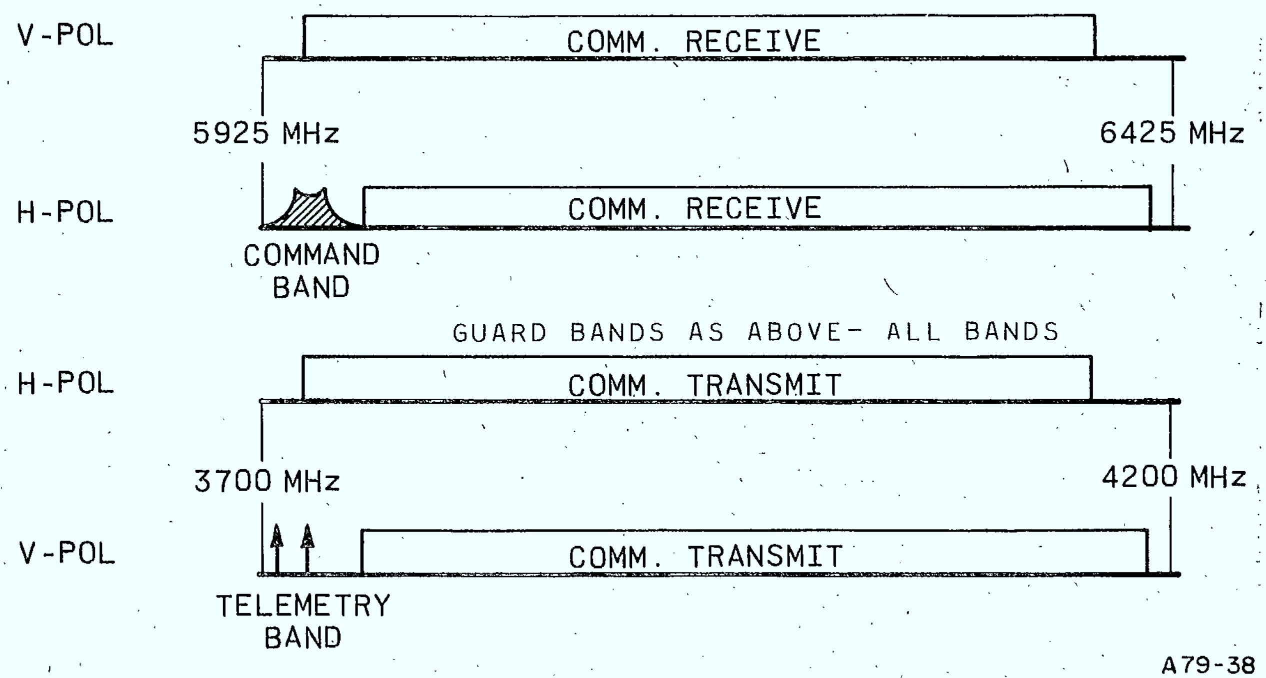

Figure 4.1.2 - ANIK-D: 24 Channel Frequency and Polarization Plan

5. TEST METHODS AND THEIR IMPLICATIONS

5.1 Co-located Spare: Test Plan

The unique up and downlink test frequencies described in Section 4.2 are received, routed, amplified and transmitted by the communications transponder of the satellite under test in the following manner. The additional test equipment required on-board the satellite is described in Section 5.2, whilst its impact on transponder performance is summarized in Section 5.3.

5.1.1 Satellite Test Configuration (Figure 5.1.1)

The test signal, a carrier at the command frequency, enters the transponder via the communications antenna and the feed horn which is normally used to receive the command signal. For greater security, geographical access to the command uplink, in a typical Telesat system, is restricted to a spot beam by filtering the command frequency from the output of this single feed horn. The test signal should therefore emanate from the ground station which has Telecommand control of the satellite.

The test signal could be routed to the Communications Receiver via the normal communications path, but this method has several disadvantages. It requires that the command frequency suck-out filter (TDF) will be designed to have a low residual attenuation in the communications path at the command frequency, such that an adequate signal level (including margin for TWT degradation) will be available to saturate the TWTs. It also requires heavier switches (R-type) at the receiver input and more complicated command control. It also permits operational traffic to the active satellite to reach the transponder.

It is preferable therefore to route the test signal via the command subsystem as shown.

A 3 dB coupler, loaded by an isolator to protect the command system against mismatches on the test port of the coupler, is used to reintroduce the test signal into the communication path via a switch.

The Communications Receiver amplifies and downconverts the test frequency and feeds it to the test frequency filter, which will be appended to the channel dropping Input Mux(es). From the filter output the tone is distributed to/from each of the TWTAs in turn either by multiway switches for conventional transponders or by R-Switches around the ring redundant transponder. The R-Switches

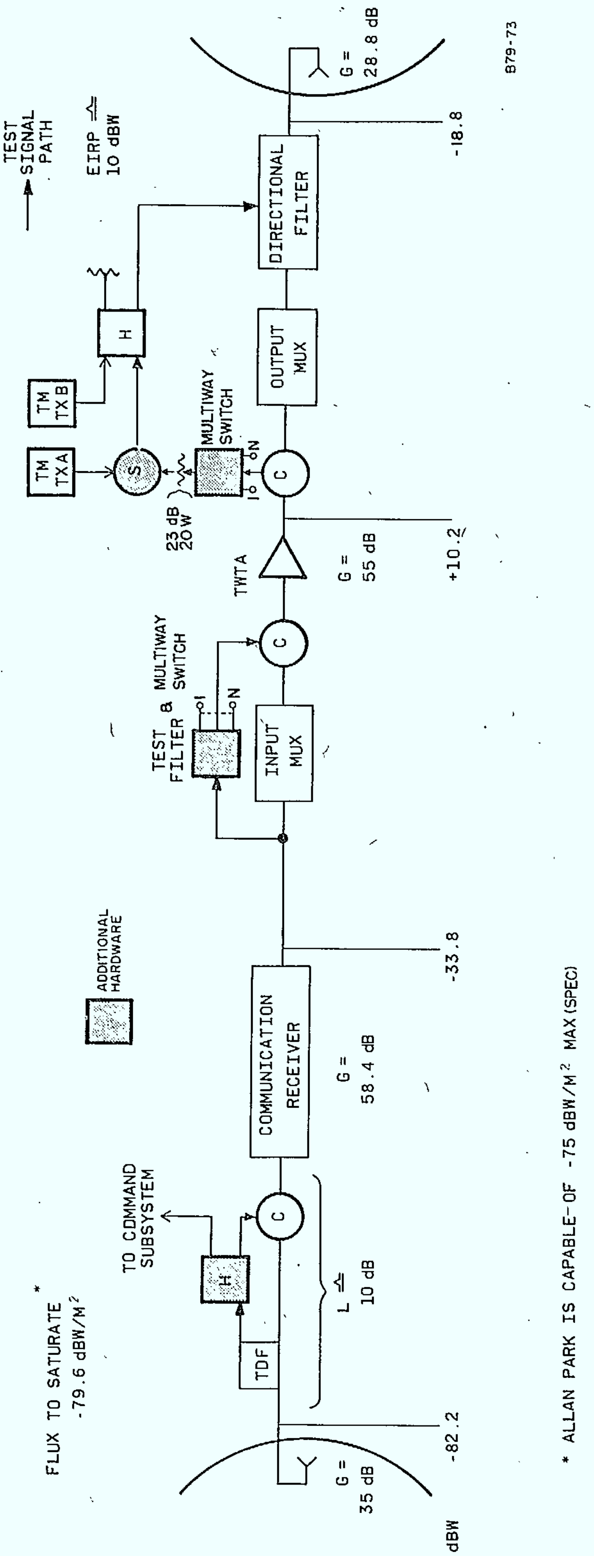

Figure 5.1.1 - Test Signals Routing and Level Diagram

do not represent additional hardware, whereas the multiway switches will probably be a custom designed microstrip structure for the low power application and a flight quality commercial product for the output switching.

After amplification by a TWTA and switching to a common line the test signal is attenuated to a level comparable to that of a telemetry transmitter. It is proposed that the test signal should substitute for one of the telemetry beacons. The test frequency and the telemetry frequency would not necessarily be the same, since the downlink frequency will be tied to the uplink frequency and the on-board local oscillator. A switch is provided for this purpose. Hereafter the test signal follows the conventional telemetry routing to the communications antenna. The second telemetry transmitter will simultaneously transmit telemetry housekeeping data which will be required to support the test results.

5.1.2 Ground Station Test Set-up

Figure 5.1.2 shows how it would be necessary to modify and add to the Telesat TTAC antenna system to permit testing of a co-located spare by the proposed method. Because all the tests are performed using only one pair of up and down link frequencies, automation of the data collection may not be cost effective. It is advantageous to configure the test set-up such that command and telemetry transmission can be interleaved with the tests.

The additional equipment, some of which may already be configured into the ground station, performs as follows:

A PIN modulator before the High Power Amplifier (HPA) is driven by a dc level to control the uplink rf power and a 1 kHz source to amplitude modulate the uplink for the saturation flux density test. The uplink power is monitored by a power head and meter.

The uplink carrier is also mixed with the output of a frequency synthesized source at 2225.1 MHz. The product will therefore be separated from the nominal downlink test frequency by 100 kHz. After amplification and level adjustment an accurate power measurement is made. The signal is then injected into the downlink input to the low noise amplifier (LNA) through a precision attenuator. A separate counter is provided to monitor the reference signal frequency.

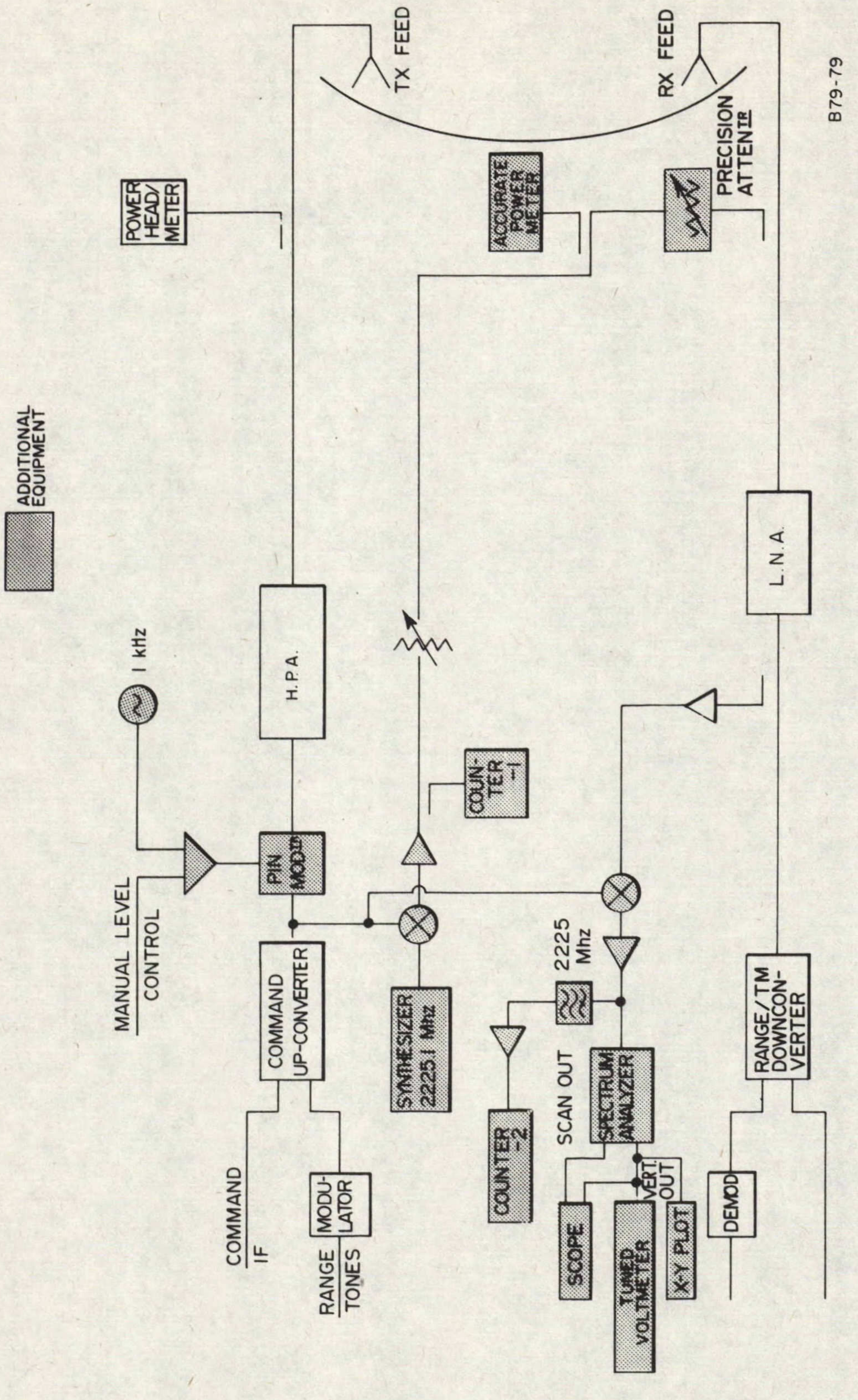

Figure 5.1.2 – Telesat: TTAC Antenna System with Modifications

The downlink test signal, after amplification in the LNA is further amplified and mixed with the uplink carrier to reconstitute the satellite LO frequency. A counter displays this frequency directly. The signal is also displayed on a spectrum analyzer where the test and reference signals can be compared for the measurement of e.i.r.p.

An oscilloscope and tuned voltmeter are slaved to the output of the spectrum analyzer such that when the latter is used in the zero scan mode the tuned voltmeter will detect the amplitude of the 1 kHz modulating signal.

5.1.3 Test Procedures

The following is an outline of the test procedure [1] that could be used to monitor the health of a spare satellite which is in the same orbit position as the operational satellite that it supports. This situation will call for several precautionary safety measures which might include increased surveillance of the operational satellite to guard against possible false commands and also procedural controls to ensure that no channel paths are opened on the spare satellite to cause multi-path interference to the operational link: Some channel paths are not interrupted by the test switches, therefore all receiver input switches should be set up to receive the test tone before any channel path is powered.

a) Turn-On Transient Performance

Ground testing and previous in-orbit testing will have established the flux required to saturate each channel by the test tone. This level, backed off 15 dB to ensure linear TWT operation but still of sufficient power to operate the command link is transmitted to the satellite. The same carrier then transmits the command signal to turn on the first cold TWT. At the same time the substitute heater will be turned off. Using the spectrum analyzer and X-Y plotter (Figure 5.1.2) to monitor the downlink signal over a period of minutes, a plot can be made of the TWTA turn-on transient performance.

b) Flux-to-Saturate

To find flux to saturate of the same TWT, the uplink is amplitude modulated by a 1 kHz sine wave. The spectrum analyzer in zero scan mode is used to receive the downlink signal, the modulation being recovered and its fundamental frequency measured in a tuned voltmeter. As the uplink power is increased the amplitude of the modulation decreases and reaches a minimum at saturation. Further increase in the uplink power past the saturation point causes the modulation to increase again. The flux to saturate can therefore be determined from the uplink power monitor, when the tuned voltmeter detecting modulation on the downlink reads minimum. Normally the flux to saturate each TWT would be measured at channel centre, however the results obtained at the test frequency, when plotted over a period of time should be an adequate indicator to the performance at channel centre since TWT degradation, which is mostly a cathode phenomenon, is independent of frequency. Any contribution from the receiver section can be investigated by exercising the redundant units.

c) <u>EIRP</u>

The saturated output from the TWT allows determination of an equivalent e.i.r.p., the life history of which can also be used to monitor the TWT performance. The downlink power is measured by levelling the received signal with an injected reference to which it is equalized using a precision attenuator. The two signals are compared on the spectrum analyzer display.

d) <u>Gain Transfer Characteristic</u>

Having accurately determined the saturation point, the uplink illumination level can be varied to provide the gain transfer characteristics of the transponder, by plotting uplink power versus received signal power.

Each TWT in turn can thereafter be switched on and the preceding tests made. The sequence will then be followed as necessary by the redundant tubes and receivers. Design precautions should be taken to avoid instabilities which could occur because the TWT will work into several configurations of input and output circuitry at two frequencies, band centre and test frequency.

e) <u>Spacecraft LO Frequency</u>

By amplifying the downlink signal and mixing it with the uplink signal and applying the product to a frequency counter a direct readout of the spacecraft LO is obtained, since the spacecraft LO is the only frequency conversion between the uplink and the downlink frequencies.

f) Transponder Noise Temperature

The spacecraft noise temperature can be determined in the command band from measurement of the received noise power level at the earth station in the absence of any signal in the measurement bandwidth. The receivers input will remain switched to the test signal path to give a defined loss from the antenna. The test path is then powered through to the Communications Antenna. No carriers in the passband of the track directional filter are to illuminate the satellite during the test.

The received noise power can then be expressed in the following terms:

$P_T = P_{s/c} + P_{e/s}$ where P_T = Total Received Power (W)
$P_{s/c}$ = Satellite Noise Level (W)
$P_{e/s}$ = Earth Station Noise Level (W)

$P_{e/s} = k\,T_{e/s}\,B$ where k = Boltzmanns constant
$:1.38\times10^{-23}$
$T_{e/s}$ = Earth Station Noise Temperature

$P_{s/c} = k\,T_{s/c}\,B$ where B = Measurement Bandwidth (On-board Test Channel Filter Bandwidth)
$T_{s/c}$ = Satellite Noise Temperature

$$T_{s/c} = T_{e/s}\left(\frac{P_T}{P_{e/s}} - 1\right)$$

In general the Earth Station Noise Temperature will be well known and the value for P_T can be found by measuring the received power with the Earth Station antenna pointing at the satellite and $P_{e/s}$ with the antenna pointed away such that the satellite does not contribute to the received power. This eliminates the need to know the measurement bandwidth accurately.

Used in conjunction with the Gain transfer measurements and known constants, a figure for satellite G/T can be derived which can be related to the communications path G/T from pre-commissioning test results.

5.2 Additional Flight Hardware

The quantity of additional hardware required to perform the above check-out method will depend greatly on the configuration of the communications package. Three designs were chosen to demonstrate the varying degrees of complexity involved. Two designs are conventional, the 12-channel ANIK-D proposal (Figure 5.2.1) and its 24-channel option (Figure 5.2.2), whilst the third design is of the ring-redundancy type (Figure 5.2.3) [2)]

In each case the unique uplink frequency is assumed to be the command carrier, thus simplifying the operating procedure. This requires that the test signal be extracted from the command signal path and re-inserted into the communications signal path. The use of a coupler is preferred to a switch since the latter would severely reduce the integrity of the command path. For the same reason the coupler should be well matched at all times and hence the output isolator, which will protect the test port of the coupler against mismatch. The test signal is re-introduced into the communication path by replacing the ferrite switches by transfer switches. Up to four additional command lines and switch drivers will be required since the receiver ON/OFF commands can no longer be phased with the appropriate r.f. switch positions.

Where dual polarization frequency re-use is in use, two test frequency filters are added to the 'ODD' channel-dropping input muxes to obviate the need for heavy R-Switches at the receiver outputs. Only one test channel filter need be configured into single polarization systems. This filter requires a few kHz bandwidth but a wider bandwidth may be chosen to avoid the need for custom design in each application.

The conventional transponders will require a multiway switch to distribute the test signal to the individual TWTAs. This is envisaged as a microstrip structure using PIN diodes to perform the switching. Control commands can be phased with the other switches used to route the signal thus minimizing the number of additional commands required.

The 24-channel design will require additional SPDT switches at both the input and output of each TWT but the required routing, in the 12-channel case, can be obtained by simply replacing the present SPDT switches by transfer switches. A great advantage of the ring-redundant system is that the test signal can be routed to any of the TWTAs without any additional hardware when the test signal is connected to the T and R switches associated with the redundant TWTAs.

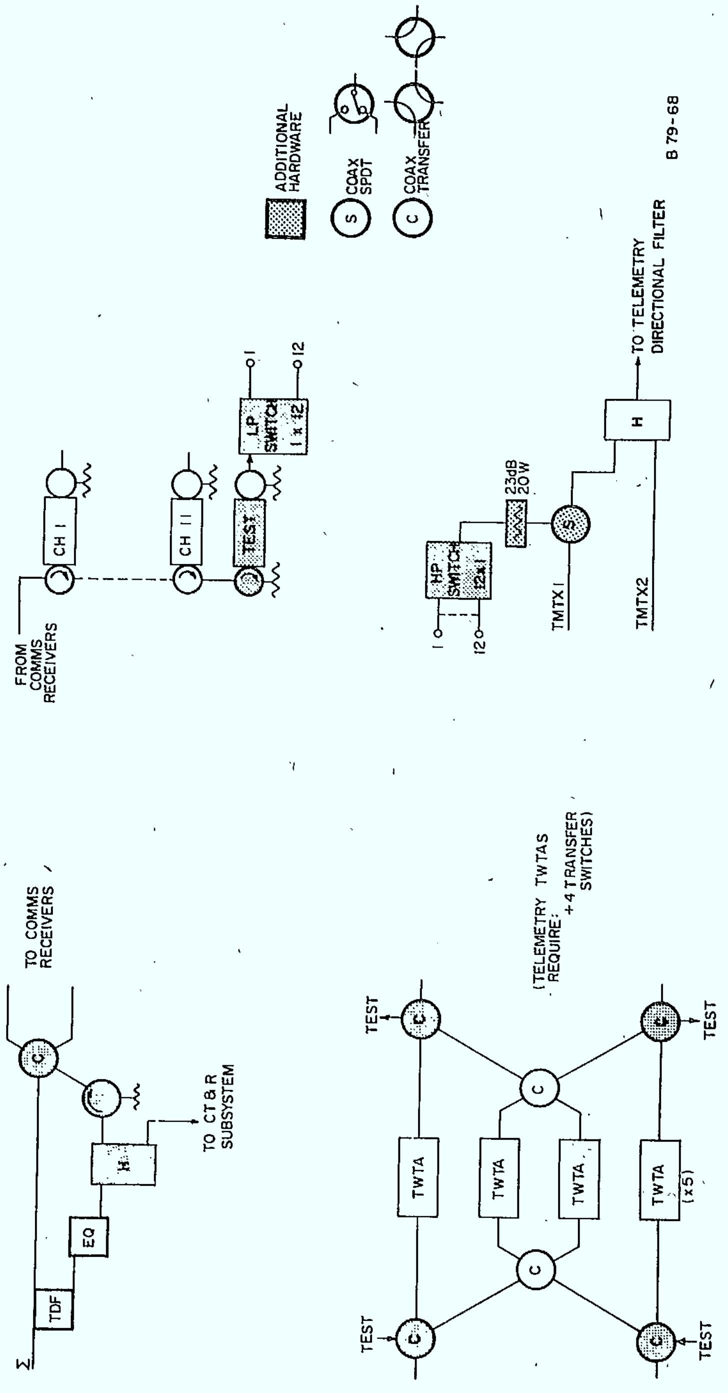

Figure 5.2.1 – Additional Test Hardware: 12 Channel ANIK-D

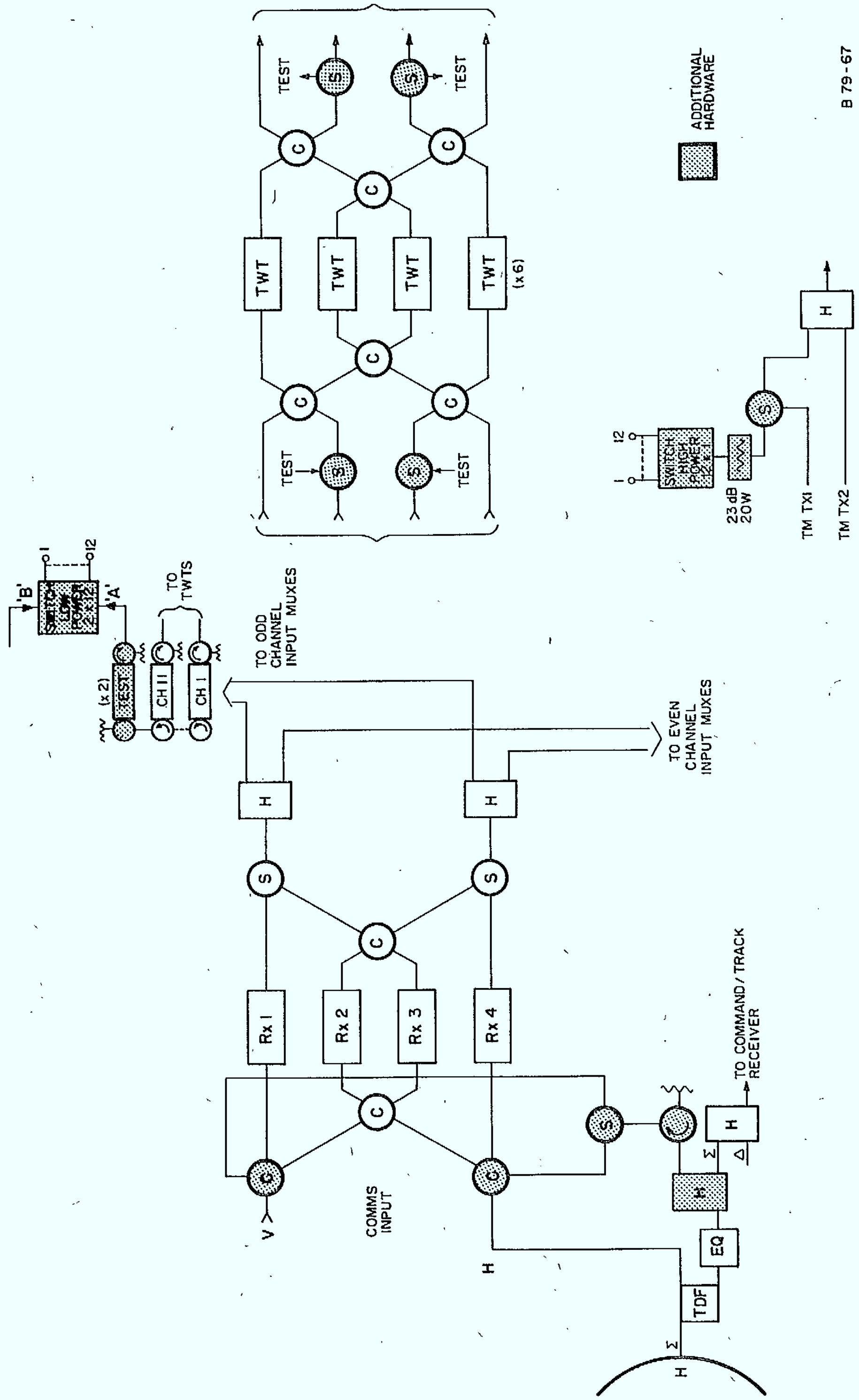

Figure 5.2.2 - Additional Test Hardware:
24 Channel ANIK-D

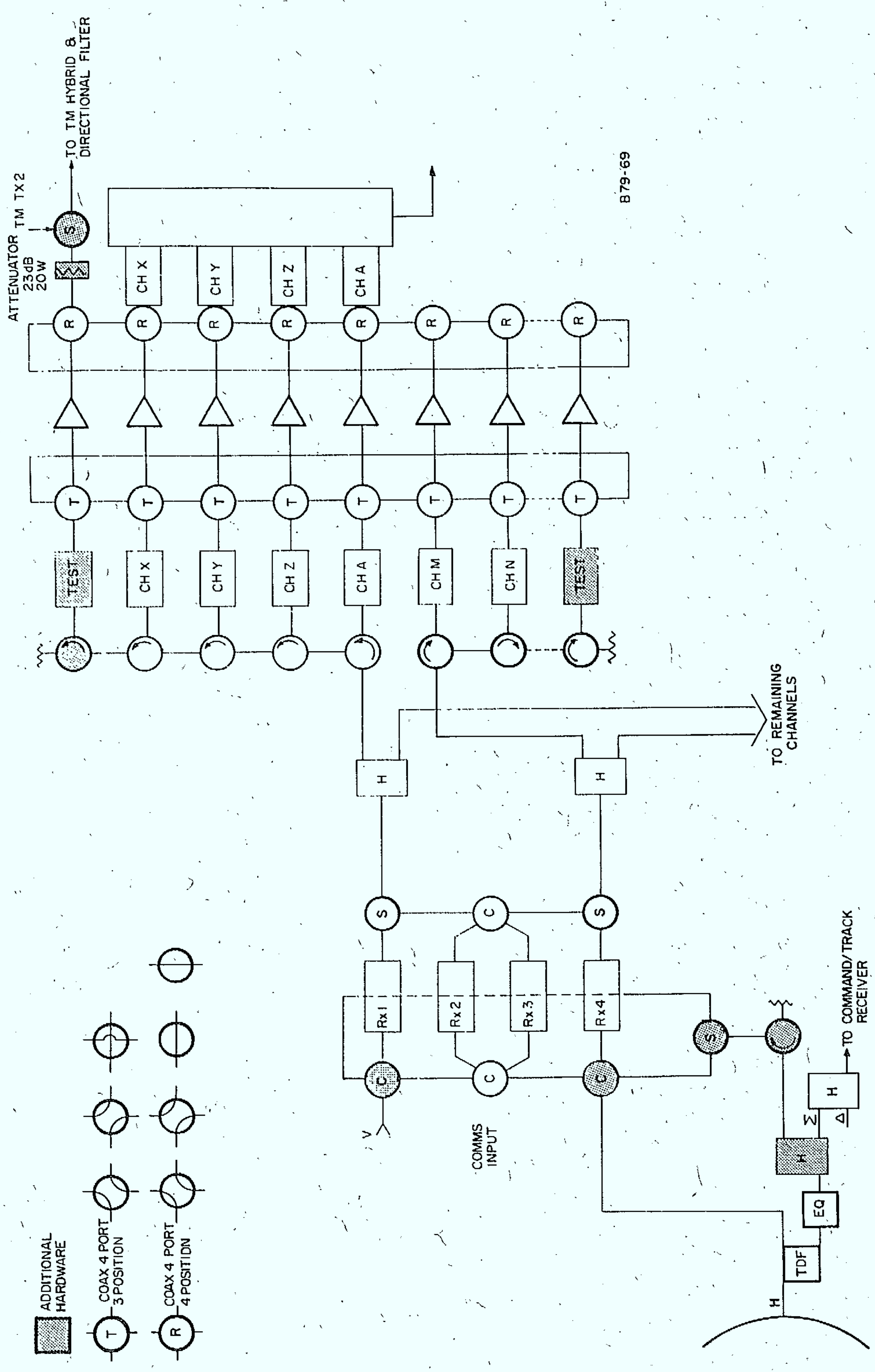

Figure 5.2.3 - Additional Test Hardware:
R-Switch Transponder

The TWTA output routing in the conventional designs will require a 20 W multiway switch and driver, which can be an off-the-shelf commercial design, but which could benefit from some mass reduction. The output R-Switch of a redundant TWTA again provides a simple method of routing through the ring-redundant transponder.

The test signal, now amplified by the TWTAs is attenuated by 23 dB in two 20 W attenuators and routed via the telemetry directional filters to the communications antenna.

It is not proposed to make the testing system redundant since it is not part of the operational payload and serves only to monitor the health of a system that is itself redundant. It also contains no active elements other than the switch drivers.

TABLE 5.2.1 - ADDITIONAL HARDWARE

1) 12 Channel ANIK-D

3, 20 dB 20W Attenuators	0.75 lb
CT & R Hybrid	0.2
Isolator	0.06
† Test Channel Dropper	0.7
† * 1 x 12 switch (low power)	0.2
12 x 1 switching (high power)	2.25
21 C switches to replace SPDT	2.75
5 SPDT (incl. 4 for TM TWTAs) Switches	0.65
Total	7.56 lb
+10%	8.3 lb = 3.75 kg = 3.5 months station keeping fuel

2) Ring Redundant - Typical System

3, 20 dB 20W Attenuators	0.75 lb
CT & R Hybrid	0.2
2 SPDT Switches	0.26
2 C Switches to replace ferrite	0.26
1 Isolator	0.06
† 2 Test channel droppers	1.4
	2.93 lb
+10%	3.25 lb = 1.5 kg

3) 24 Channel ANIK-D

3, 20 dB 20W Attenuators	0.75 lb
CT & R Hybrid	0.2
Isolator	0.06
2 C Switches to replace ferrite	0.26
† * Low Power Multiway (2 x 12)	0.25
High Power Multiway (12 x 1)	2.25
† 2 Test Channel Droppers	1.4
26 SPDT Switches	3.4
	8.57 lb
+ 10%	9.5 lb = 4.3 kg = 4 months Station-keeping Fuel

* Item requires development

† Item requires design

5.3 Performance Implications

The choice of test frequency will not significantly alter the communications performance of the transponder because all the appropriate design compromises in telemetry and command directional filters will have been necessary to accommodate the telemetry and command frequencies themselves. Only the additional hardware will contribute to performance degradations, the most significant of which is the loss of additional 3 dB in the command link. The command link, budget is normally well endowed with margin in which case the additional loss caused by a 3 dB coupler is tolerable, especially as the transfer orbit performance is not degraded.

The second greatest performance implication will be the reduction of reliability as a result of the increased use of in-path routing switches for the conventional transponders. Reliability figures will change imperceptibly, but the failure modes per channel will increase with the possibility of switches remaining in the test position. Only in the case of the 12 channel transponder (Figure 5.2.1) does this constitute an unacceptable system single point failure since a mis-routed switch would interrupt the communications signal path to all communications receivers. To avoid this problem will require that the test signal is routed into each receiver path individually at the cost of an additional switch insertion loss. For the same reason individual commands for each path are preferred to control the routing switches. This eliminates the possibility of failure of the 'ALL ON/OFF' type command. No additional switching is introduced in the communication path of the ring-redundant transponder which therefore undergoes no reliability degradation.

The additional hardware indicated by the shading in Figures 5.2.1 to 5.2.3 does not degrade the hard system parameters (BER/EIRP/C/N) because there is no significant change to the equipment in any one channel path. Mass (and cost) does increase measurably however according to Table 5.2.1. It can be seen from this table that the increased mass of the ANIK-D examples is equivalent to the station-keeping fuel required for a significant period of months (for ANIK-D, 1 year of stationkeeping requires an average 12.7 kg of fuel).

1) 'In-orbit Testing of Communications Satellites', I. Dostis, C. Mahle, V. Rignos and I. Atohoun, COMSAT Technical Review, pages 197-226, Vol. 7, No. 1, Spring 1977.

2) 'Network Topologies to Enhance the Reliability of Communications Satellites', F. Assal et al, COMSAT Technical Review, Vol. 6, No. 2, Fall 1976.

6. SUPPORTING ANALYSES

6.1 Fuel Expenditure and Repositioning Time for Longitudinal Relocation of a Synchronously Orbiting Satellite

The fuel requirement for repositioning a satellite in orbit is a function of orbital velocity, duration of transit, spacecraft mass and propulsion efficiency. The manoeuver is performed by means of applying two velocity increments, $+\Delta V$ and $-\Delta V$ to the satellite separated by the time period of repositioning. The following outlines the calculation of a range of orbit repositioning manoeuvers using classical orbit motion equations and a typical spacecraft configuration viz: ANIK-D.

6.1.1 Orbital Elements

Figure 6.1.1 a) represents a typical situation:

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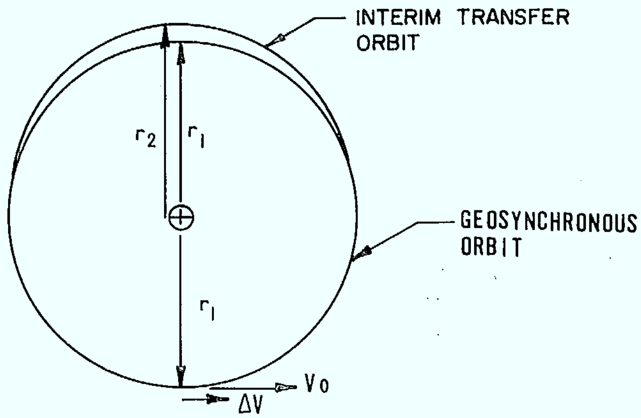

Figure 6.1.1 a) Relocation Definition of Terms

In the figure:

r_1 = radius of geosynchronous orbit
= perigee of transfer oribt

r_2 = apogee of transfer orbit

V_o = velocity in geosynchronous orbit

$V_o + \Delta V$ = velocity at perigee in transfer orbit
$= V_t$

Since the period of an orbit is directly proportional to its size, mainly its semi-major axis A_t, the transit from a synchronous orbit to an interim transfer orbit will change the period from P_o to P_t respectively. The result is that a movement of the satellite position will take place in the longitudinal plane.

It is given that:

$$A_t = \frac{r_1 + r_2}{2} \tag{1}$$

and

$$P_t = 2\pi\sqrt{\frac{A_t^3}{\mu}} \tag{2}$$

where

μ is the gravitational constant of the earth and the interim transfer orbit velocity V_t at perigee is the algebraic sum of:

$$V_t = V_o + \Delta V \tag{3}$$

6.1.2 Longitudinal Repositioning

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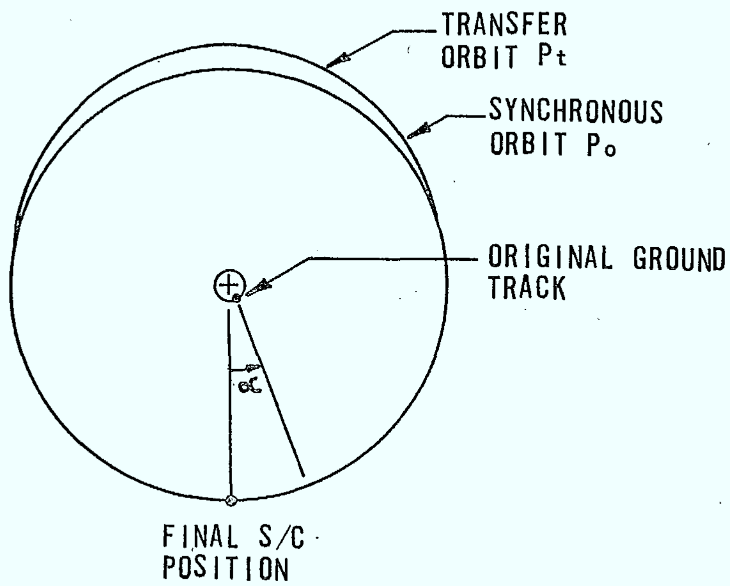

Figure 6.1.1 b) Relocation Definition of Terms

Figure 6.1.1 b) illustrates the longitudinal angular separation α of the satellite from its initial ground track as the result of the longer orbital period P_t of the interim transfer orbit. The angular separation is also a function of the number of revolutions n as demonstrated by the relationship in equation (4).

If it is required to relocate α degrees in n revolutions the angular movement is α^o/n in one revolution.

Relating this to orbital period, we obtain:

$$\frac{\alpha^o}{n} = \frac{P_t - P_o}{P_o} \times 360^o \tag{4}$$

From equations (1) and (2) we get:

$$r_2 = 2A_t - r_1 \tag{5}$$

and

$$A_t^3 = \mu \frac{P_t^2}{(2\pi)^2} \tag{6}$$

Hence for a required value of α^o/n the value of P_t is specified, and from P_t the values of r_2 and A_t may be derived.

From orbital mechanics we obtain the following orbital equations:

$$V_o = \sqrt{\frac{\mu}{r_1}} \tag{7}$$

$$E_t = -\frac{\mu}{(r_1 + r_2)} \tag{8}$$

$$V_t = \sqrt{2\left(\frac{\mu}{r_1} + E_t\right)}$$

Substituting E_t gives:

$$V_t = \sqrt{2\mu\left(\frac{1}{r_1} - \frac{1}{r_1 + r_2}\right)} \tag{9}$$

where,

E_t is the energy of the transfer orbit

V_t is the total velocity at perigee needed to achieve the interim transfer orbit

The constants given are:

$$\mu = 3.986032 \times 10^5 \ \frac{km^3}{sec^2}$$

$$r_1 = 42164 \text{ km}$$

$$g = 9.809 \text{ m/sec}$$

$$V_o = 3.07468 \text{ km/sec}$$

By selecting a range of angular displacements α^o and number of revolutions n, one can solve for V_t and hence the required ΔV can then be obtained.

6.1.3 Solutions for Fuel Consumption Versus Transit Time

The foregoing sections compute the orbital periods as a function of change of velocity.

From the ΔV requirement, the fuel consumption can be obtained by using the rocket equations for mass expulsion and thruster efficiency.

For ANIK-D, the parameters are:

M_o = 608 Kg, and

Thruster I_{sp} = 154.8 sec for radial pulsed mode operation.

Stationkeeping Fuel Life = 12.7 Kg/year, average.

The rocket equation is:

$$M_p = M_o \left(1 - e^{-\frac{\Delta V}{g \cdot I_{sp}}}\right) \qquad (10)$$

where

Mp is RCS propellant mass

Mo is initial spacecraft mass before manoeuver

For small ΔV, equation (10) can be approximated as:

$$M_p = M_o \frac{\Delta V}{g \ I_{sp}} \qquad (11)$$

Since the analysis so far treats only a transit to a new location due to the addition of ΔV, an equal and opposite amount of velocity correction $-\Delta V$ is required to re-synchronize the orbit and to maintain the satellite in its final location. Thus the fuel consumption becomes:

$$Mp = 2 M_o \frac{\Delta V}{g I_{sp}} \qquad (12)$$

Table 6.1 summarizes the calculation of fuel consumption versus time of transit for a range of angles 5^o, 10^o and 15^o in longitude and for $n = 1$ to 10 revolutions. The quantities of fuel indicated can be expended by the thrusters in a period of time which is short compared to the total drift time and has therefore been neglected. A spinning satellite similar to ANIK-D can expend a maximum of approximately 40 kg/hour whilst a 3 axis satellite of similar size can expend approximately 8 kg/hour.

The results are also plotted in graphical form as shown in Figure 6.1.2 'Fuel Consumption' in weeks of NSSK fuel or equivalent mission life is plotted against transit time in days.

6.1.4 Conclusions

It can be seen from the results that fuel consumption goes up at a high rate as the transit time requirements is reduced towards 1 or 2 days. The typical spacecraft (ANIK-D) used is particularly inefficient in E-W propulsion since radial pulsed mode has to be employed.

On the other hand for a 3 axis spacecraft, theE-W continuous thrusting mode is more efficient. Consequently absolute fuel consumption is correspondingly lower. However, the inversely proportional function of fuel versus time is still applicable.

References

1. Fundamentals of Astrodynamics - Bates, Mueller and White Dover, N.Y., 1971

2. Flight Performance Handbook for Orbital Operations, pg. 5-123 - R.W. Wolverton, Editor John Wiley, 1963

TABLE 6.1

SATELLITE FUEL CONSUMPTION FOR LONGITUDINAL RELOCATIONS
(Based on ANIK-D)

DISPLACEMENT	5°		10°		15°	
ORBITS	FUEL (kg)	TIME (hours)	FUEL (kg)	TIME (hours)	FUEL (kg)	TIME (hours)
1	11.24	24.267	22.18	24.599	32.84	24.931
2	5.66	48.201	11.24	48.533	16.75	48.866
3	3.78	72.135	7.53	72.467	11.24	72.800
4	2.84	96.069	5.66	96.402	8.46	96.734
5	2.27	120.003	4.53	120.336	6.78	120.668
6	1.89	143.937	3.78	144.270	5.66	144.602
7	1.62	167.872	3.24	168.204	4.85	168.536
8	1.42	191.806	2.84	192.138	4.25	192.471
9	1.26	215.740	2.52	216.072	3.78	216.405
10	1.13	239.674	2.27	240.007	3.40	240.339

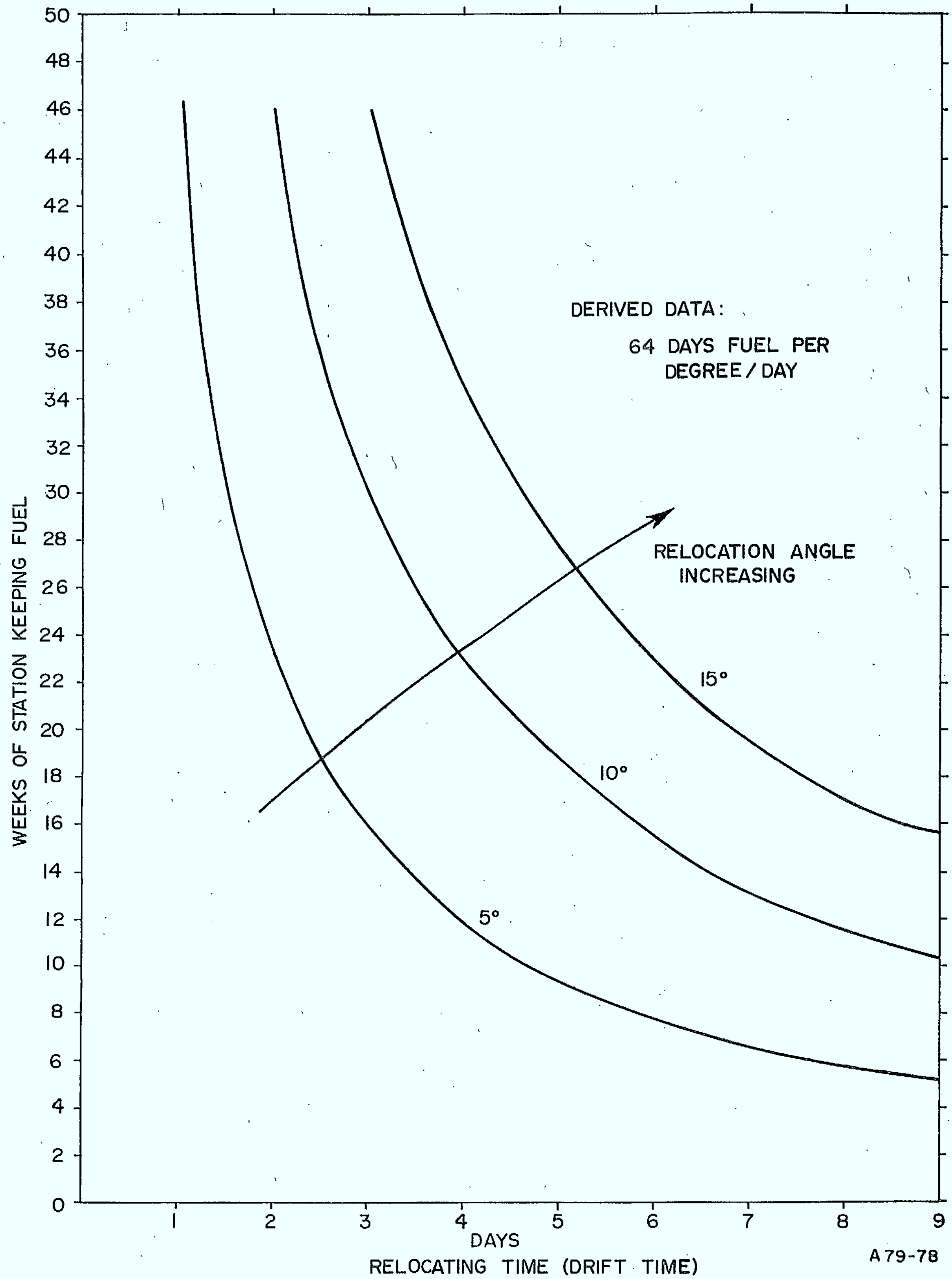

Figure 6.1.2 - Satellite Fuel Consumption for Longitudinal Relocation (small angles)

6.2 Proposed Strategies for Controlling Co-located Satellites to Minimize Probability of Collision

In order to ensure absolute spatial separation between multiple satellites co-located in a narrow angular slot without the use of ground determination, means must be provided on board to detect an allowable minimum distance between each other to effect proper manoeuvering.

A feature of this requirement is that the system needs only to detect, without great accuracies, the presence of a neighbouring spacecraft inside a predetermined distance so as to trigger an alarm or warning signal for station control. It is assumed that spacecraft orbit and attitude history are well known and that attitude data are continuously available. The system must also be simple, rugged, low cost and use available technology.

Four methods have been conceived. One ranges, two sense and another involves orbital parameter control from the ground. They are described in the following sections:

6.2.1 Ranging System

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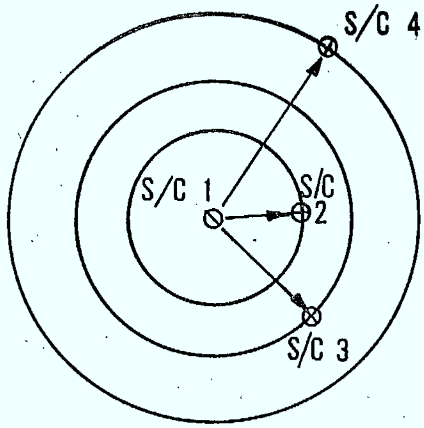

Figure 6.2.1 - Co-located Satellites

The above diagram illustrates four spacecraft co-located in a close proximity sphere. In the case of spinners, the method of implementation is as follows: A primary spacecraft S/C 1 will be carrying an omni-directional antenna and redundant, low power transmitters for the purpose of transmitting range tones originated from the ground.

The secondary spacecraft S/C 2, 3, 4 will be equipped with fan-beam antenna(s) on the spinning drum with 180° polar angular coverage and moderate angular resolution. As the spacecraft rotate, the fan beams will sweep the entire 4π sphere around each spacecraft once per revolution.

Upon receiving the range tones, the secondary spacecraft will re-transmit them to the primary spacecraft and the range information can be processed on board or on the ground. Angular information between spacecraft can be derived from the fan-beam antenna location with respect to the Master Indexing Pulse or sun/earth sensor signal.

Once the range and angular information are obtained, conventional stationkeeping manoeuvers can be carried out to maintain a pre-determined minimum distance between the two spacecraft.

If desired, each spacecraft can carry the omni as well as the directional antennas to act as either the primary or secondary spacecraft and the initiation of interrogation from the primary spacecraft can be commanded from ground.

It may be noted that simultaneous ranging of all spacecraft is not really necessary since relative movement between spacecraft is rather slow - this is to be qualified in Section 6.2.4.

In the case of 3-axis spacecraft, the implementation would be slightly different since no inherent sweeping mechanism is available. Therefore the secondary spacecraft will carry eight circular beam antennas each governing approximately one solid quadrant of the sphere. Angular information can thus be obtained by relating to the receiving antenna quadrant and range is obtained in the same fashion as in the case of spinners. Stationkeeping routine is then carried out according to the directional signal derived. Note that although the angular information is very coarse, it is adequate to ensure that a direction of relative motion may be chosen which will move the spacecraft further apart.

6.2.2 IR Sensing System

An IR Sensing system may be employed for short range (500-1000 meters)* detection of IR emitting bodies in the vicinity by using a modified conventional earth sensor with focal length and sensitivity adapted for this application. This system will be more adaptable to spinners since with a limited FOV, spherical coverage can be easily achieved by mounting 2 sensors on the spinning drum as illustrated in the following diagram:

* Needs further verification

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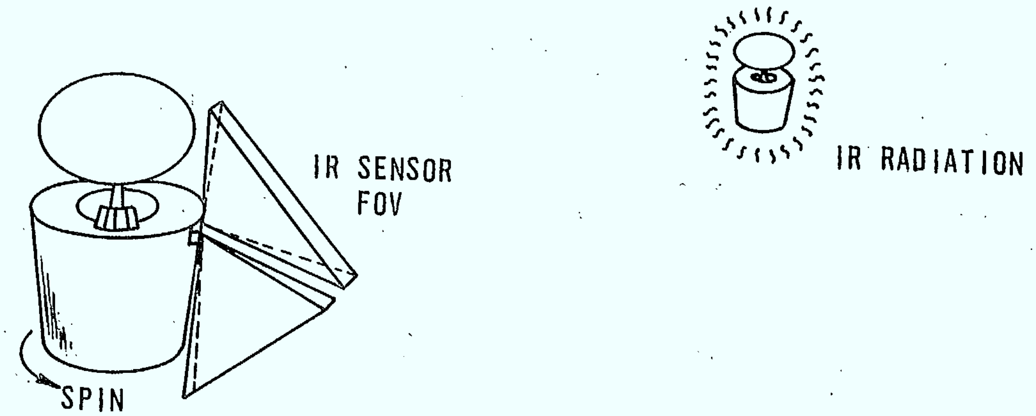

Figure 6.2.2 - Passive IR Sensing System

Each spacecraft will carry its own complete sensing system with internal sun/earth and Master Indexing Pulse reference. Upon detecting the target, the target pulse is compared against the reference pulse and the target bearing can be derived by knowing the spin period and the sensor position. Determining the relative location of more than one spacecraft within the sensor range will be done by triangulation.

The detection method will be a threshold detection of range and conventional stationkeeping routine can be carried out to maintain the desired range. For a two or three spacecraft system, the control logic may easily be implemented on board with ground participation needed only for control initialization and command override.

Sun interference can be eliminated using the sun signal from an existing separate sun sensor and noting that the sun pulse rotates at 0.25 deg/min. with respect to earth referenced objects. However, the IR sensor design must ensure that recovery from sun interference is sufficiently fast to allow proper detection.

Note that in this system, and in the next to be presented, detection by each other of spacecraft located generally N-S with respect to each other is more difficult than for other directions of co-location. This is because the sweeping action of the sensor fields of view diminishes as the look angle approaches the axis of rotation. However, a N-S separation implies that the orbits of the spacecraft are inclined to one another and the orbital planes must intersect. During travel through the parts of the orbit near the line of intersection detection capabilities become nominal. Hence, provided that care is taken when introducing each new spacecraft into the co-location area to minimize the chance of an initial collision with spacecraft already on station, data would shortly be available (in not more than one quarter of a revolution) of any close positioning.

6.2.3 Visible Light Sensing System

This concept is less conventional but is the simplest of all. A primary spacecraft will have a long-life light bulb mounted on an unobstructed location. All secondary spacecraft will have a sun sensor mounted on the spinning drum with detection FOV's tangential to both directions of the spin surface as shown in the following diagram:

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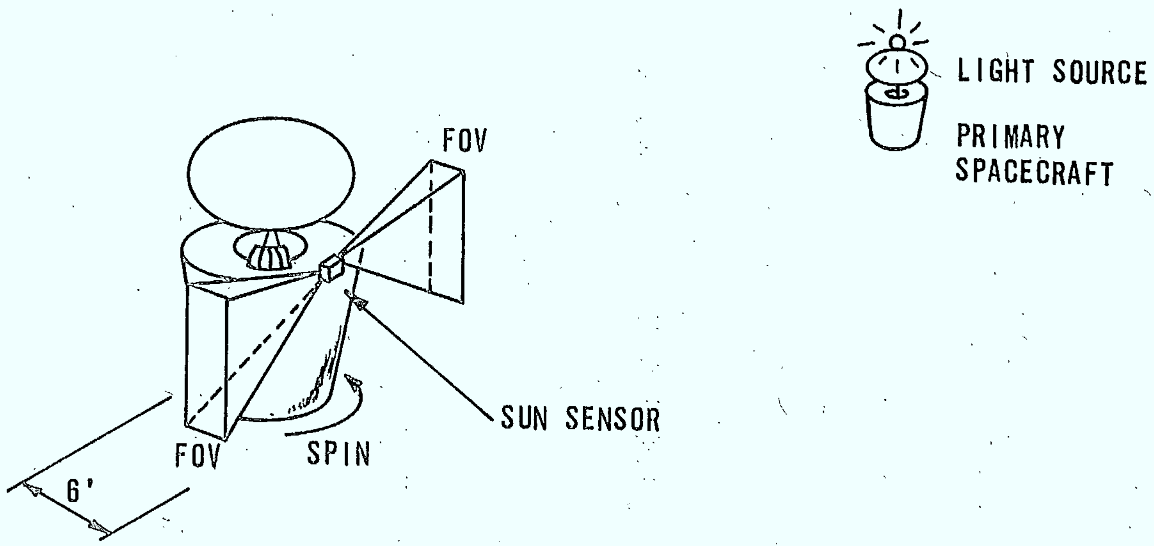


Figure 6.2.3 - Visible Light Sensing System

As the spacecraft rotates, the sensor will detect the light source twice per rotation. With the light source at infinite distance, the angular separation between detection will be 180°. As the light source approaches, the angular separation will be reduced and the detectable range will be a function of a resolvable angular separation. The separation is dependent on the drum width, sensor pulse width, and signal to noise ratio.

For a spacecraft with a 6 ft. dia. drum and spinning at 60 RPM, the differential pulse time versus distance is depicted in the following graph:

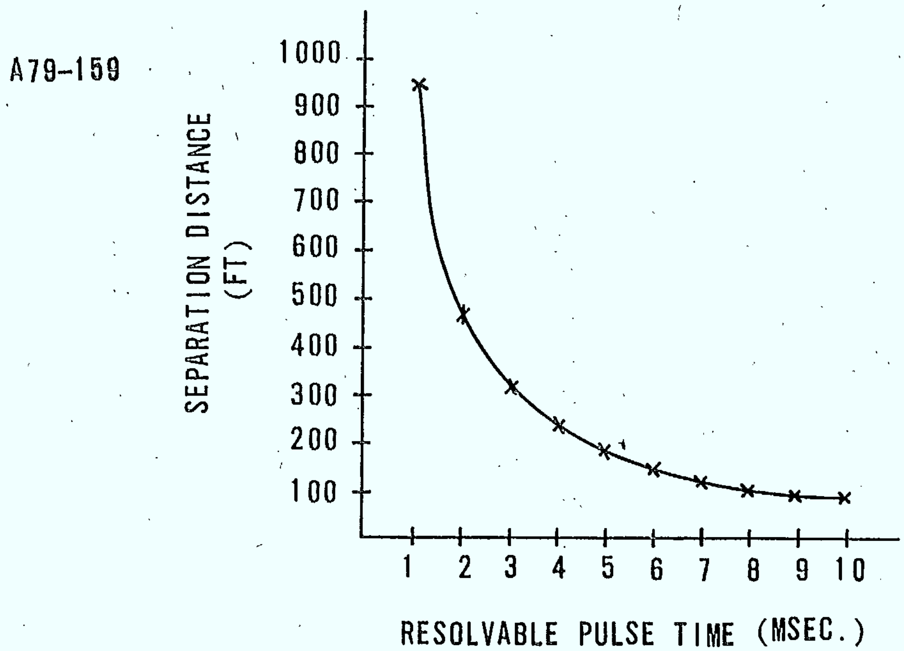


Figure 6.2.4 - Visible Light: Pulse Timing

Note that for a 1 msec differential, the detectable distance is about 500 - 1000 ft. depending on the signal to noise ratio. If the resolvable differential time can be made smaller and the effective drum diameter made larger, then the detectable range can be appreciably increased.

Again, with a multiple spacecraft system, separation between receiving spacecraft can be maintained by triangulation and using sun/earth and MIP angular references. Sun interference can be eliminated using the signal from the existing separate sun sensor and known sun pulse rotation rate.

Note that this method is equally applicable to 3-axis configuration with the primary spacecraft carrying two light souces, one on each tip of its solar panels and the receiving spacecraft carrying scanning light sensors with adequate FOV coverage.

6.2.4 Orbit Control

It can be shown by assuming that satellite motion is Brownian, that the probability of collision of two satellites in the same orbital location is negligibly small. However, satellite motion is far from random. Nevertheless it is possible to control individual satellite orbit parameters such as longitudinal station, inclination, eccentricity and the correction of E-W drift to ensure that the satellites never coincide. This method of collision avoidance will require precise ground orbital measurement, data processing, decision management and command sequencing and execution.

The path of two slightly inclined circular orbits will cross at only two points in the orbit. The possibility of collision at these two points can be reduced by creating a small longitudinal difference between the satellites and possibly augmenting this separation by arranging that the major axes of the inevitable eccentricity in each individual orbit is arranged to give maximum advantage. Having obtained a nominal separation, it will be necessary to correct the E-W drift of each satellite in a defined sequence to avoid the possibility of paths crossing during completion of the manoeuvre. Figure 6.2.5 depicts the situation for three satellites co-located.

This form of orbit control requires a much deeper analysis. Even though extensive software may be required in the control ground station, it would be justified if it obviates the need for hardware in orbit, for active collision avoidance.

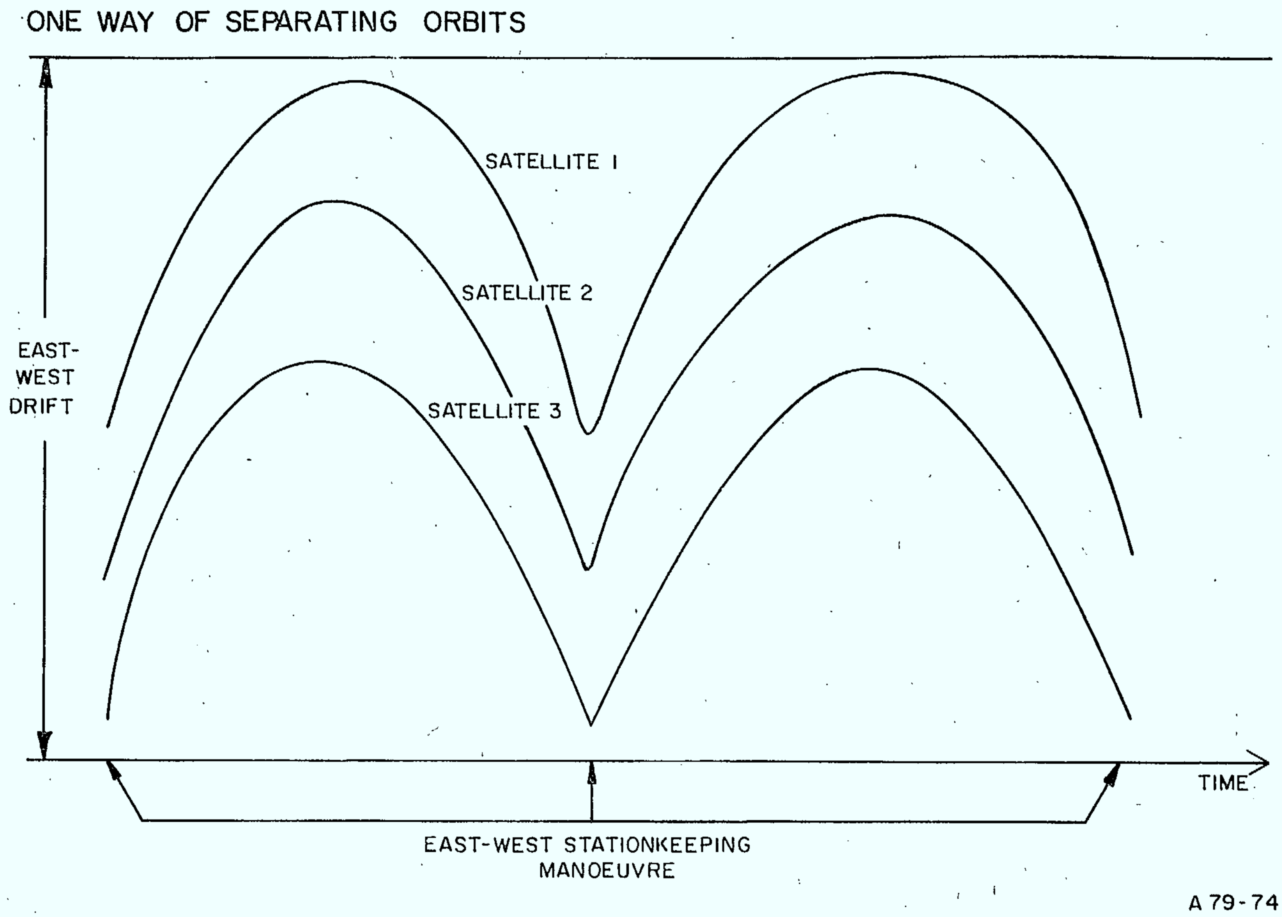


Figure 6.2.5 - Coordination of E-W Station keeping Manoeuvre for Co-located Satellites

6.2.5 Supplementary Information

In order to appreciate the magnitude of minimum space between spacecraft, it is useful to examine the relative movement of the spacecraft in a close area due to perturbations as well as thrusting.

The satellite orbit is constantly perturbed by mainly the sun and moo n and the earth's triaxiality. The sun/moon contribution is more pronounced. The perturbation will cause the satellite to wander away from its normal orbital track.

It is known that a force (acceleration) component in the N-S direction in the amount of 1616 ft/sec in 10 years will be imparted on the spacecraft with the result of a change in orbit inclination.

Using the average value of 161.6 ft/sec/year and multipled by a factor of 4 to account for peaks and 2 spacecraft travelling in opposite directions, the effective acceleration will be about:

646 ft/sec/year

and generously reduced to about:

2 ft/sec/day (worst case)

Therefore the relative acceleration imparted on the two spacecraft is approximately:

2.3×10^{-5} ft/sec^2

As an example, the distance travelled in:

1 hour is: $1/2\,(2.3 \times 10^{-5} \times 3600^2) \sim 150$ ft

12 hours is: $1/2\,(2.3 \times 10^{-5} \times 3600^2 \times 12^2) \sim 20{,}000$ ft.

Complementing the above, it is also worthwhile to look at the spacecraft movement due to firing of the thruster.

Using 5 lb_f thruster and spacecraft mass = 1500 lb_m.

The acceleration is $5 \div \frac{1500}{32.2} \sim \frac{1}{10}$ ft/sec^2

Therefore, thrusting for one minute will cause the spacecraft to move about 180 ft., and leave it with a residual ΔV of about 6 ft/sec (20,000 ft/hour).

It can be concluded from the above analysis that a safe distance between the spacecraft is also affected by the observation (detection) time as well as the thrusting time. Unless a control manoeuver is performed on-board, the reaction time of the ground crew must be factored into the minimum distance calculation. However, it is beyond the scope of this study to provide quantitative descriptions to the aforementioned control methods and their implementation.

It may also be noted that two free bodies (Mass - 1000 kg), having no relative movement, when acted upon solely by their mutual gravitational attraction will take 1 year to collide when initially separated by 500 m and 16,000 years when initially separated by 300 km.

6.3 DBS Space Segment Costs

A high grade of service is assured having an in-orbit back-up capability in the event of catastrophic failure of the prime spacecraft. It is obviously desirable to reduce the size and cost of this normally inactive, noncontributing spare capacity. Further cost savings can be made if all the satellites in a system including the spare are identical. The use of several smaller co-slotted satellites to perform the function of one large satellite is consistent with these observations. The spare, being identical to the in-service units, could then replace any of the individual satellites of the complement. This approach has the additional advantage that the full system capability could be reached in stages thereby permitting investment to be matched to income or availability of funds. For example, an eight channel system could be achieved by putting in orbit a four channel satellite on an 'experimental' basis. This could be followed later by a second identical satellite which would provide the additional capacity to permit the system to become 'fully operational'.

Expansion to a fully operational 8 channel system could take place by placing in orbit a further four channel spare spacecraft.

The computer program 'COMSATMOD', in use at CRC, was used to compute TOTAL SPACE SEGMENT COST, including launch costs for various sizes of individual spacecraft. The system investigated covered Canada with four beams and had 50% eclipse capability. The TWT RF output power per channel and the number of channels/beam per satellite (i.e. the number of satellites in orbit to satisfy the total number of channels per beam) were varied for 4, 6 and 8 channels per beam systems, and the results plotted on Figures 6.3.1 to 6.3.3.

Other characteristics of the spacecraft were specified to the program, but these do not alter the relative relationship of the results. The break points between the shuttle upper stages have been indicated to explain the discontinuties of cost versus TWT RF output power and also, where the mass of a particular configuration is within 10% of the upper stage threshold, the equivalent cost for launch on the smaller upper stage has been indicated by the remark "squeezed".

The plots indicate that each system, when using 125W TWTs, always has minimum space segment costs when the in-orbit complement is two broadcasting satellites backed up by one co-located spare. In the case of the 4 and 8 channel /beam systems, this conclusion depends on whether the spacecraft can be squeezed onto the next smaller upper stage. The fact that in these cases each satellite must carry twice as many channel filters as will be in use to satisfy the spare philosophy will aggravate the squeezing process. Table 6.3.1 summarizes the primary characteristics of each type of satellite and the total system cost for the various system configurations.

Figure 6.3.4 gives a clearer indication of how the mass of the various satellites relates to the capabilities of the three upper-stages to be used in conjunction with the shuttle. Spacecraft mass is plotted as a function of channels/beam per satellite and TWT power. Marginal cases that would require "squeezing" are indicated as X in the figure.

Figure 6.3.5 plots the minimum total system cost as a function of the number of channels/beam and also gives an indication of the cost effectiveness in terms of million dollars per channel.

An alternative and very much reduced cost is indicated for the 6 channel per beam system, but because emphasis has been placed on the 8 channel per beam system, this approach has not been developed. In this case, each full payload consists of 2 + 2 channels of totally independent communications and power subsystems. One half payload would then spare for the remaining one and a half payloads. Catastrophic failure of one satellite would reduce the system to 4 channels/beam capacity.

Conclusion

If the preferred DBS system is a 54 dBW approx. four beams, eight channel per beam system, with 50% eclipse capability, then the cheapest way to implement it would be by putting in orbit three spacecraft carrying 125W TWTs and capable of transmitting in any four of the eight allocated channels. The spacecraft would be launched using an SSUS-A upper stage by the shuttle. The total space segment cost would be 155 M$C at 1978 prices which is equivalent to 19.4 M$C per channel. Partial implementation could take place at a cost of 61 M$C for an unspared 4 channel/beam system or 108 M$C including a full spare.

TABLE 6.3.1 - SUMMARY OF RESULTS FROM SPACE SEGMENT COSTS ANALYSIS (150W TWTA//4 BEAM SYSTEM)

Channels per Beam System Total	Satellites in Orbit Including One Spare	2	3	4	5
4	Approximate Total System Cost m$c (1978)	112	109		145
	System Compliment of TWTAs *	40 (125W)	30 (125W)		25
	Channels Per Beam Per Satellite	4	2		1
	Spacecraft Power EoL	4.6 kW	2.35 kW		1.4 kW
	Launch Upper Stage	SSUS.A	SSUS.D		SSUS.D
6	Approximate Total System Cost m$c (1978)	175	143	138	
	System Complement of TWTAs *	60	45	40 (125W)	
	Channels Per Beam Per Satellite	6	3	2	
	Spacecraft Power EoL	8.2 kW	4.1 kW	2.35 kW	
	Launch Upper Stage	IUS	SSUS.A	SSUS.D	
8	Approximate Total System Cost m$c (1978)	204	155		168
	System Complement of TWTAs*	80	60 (125W)		50 (125W)
	Channels Per Beam	8	4		2
	Spacecraft Power EoL	10.9 kW	4.6 kW		2.35 kW
	Launch Upper Stage	IUS	SSUS.A		SSUS-D

* Assumes 5/4 Redundancy

N.B. Costs and Power derived from COMSATMOD program predictions

4 BEAM DIRECT BROADCAST SYSTEM
4 CHANNELS/BEAM
TOTAL SPACE SEGMENT COSTS INC.LAUNCH

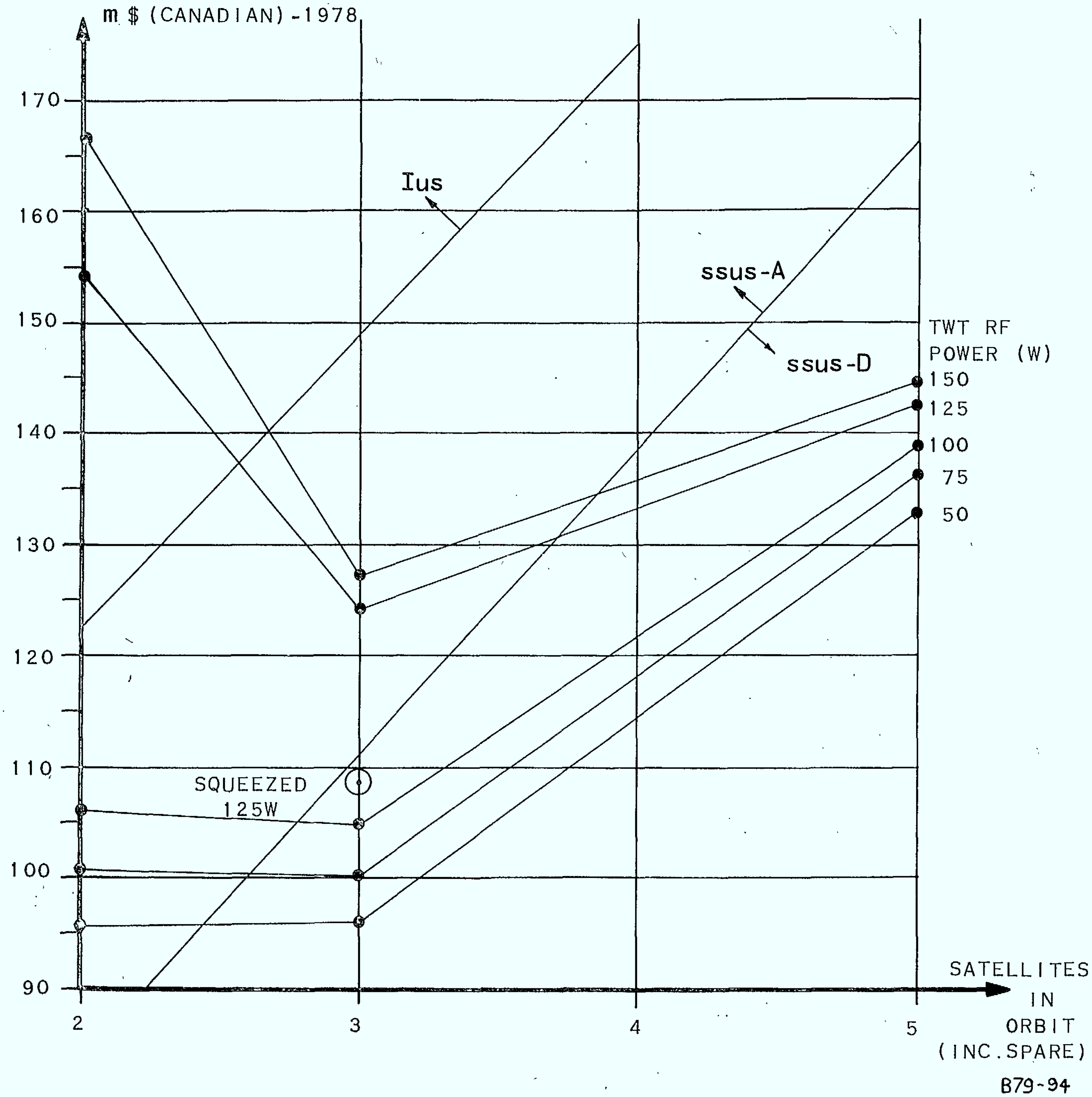


Figure 6.3.1

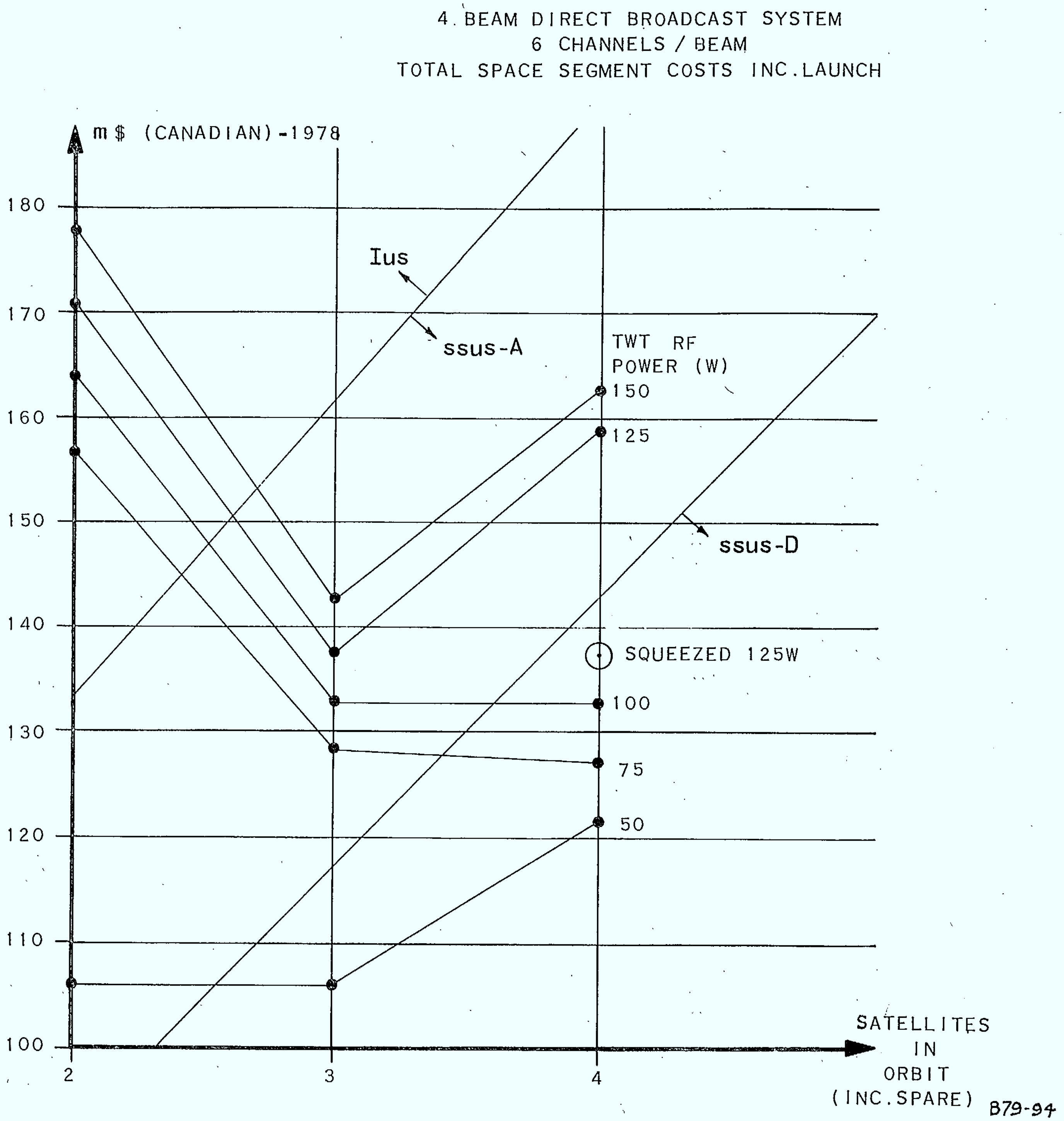

4.BEAM DIRECT BROADCAST SYSTEM
6 CHANNELS / BEAM
TOTAL SPACE SEGMENT COSTS INC.LAUNCH
m$ (CANADIAN)-1978
180
170
160
150
140
130
120
110
100
Ius
ssus-A
ssus-D
TWT RF POWER (W)
150
125
SQUEEZED 125W
100
75
50
2
3
4
SATELLITES IN ORBIT (INC.SPARE)
B79-94

Figure 6.3.2

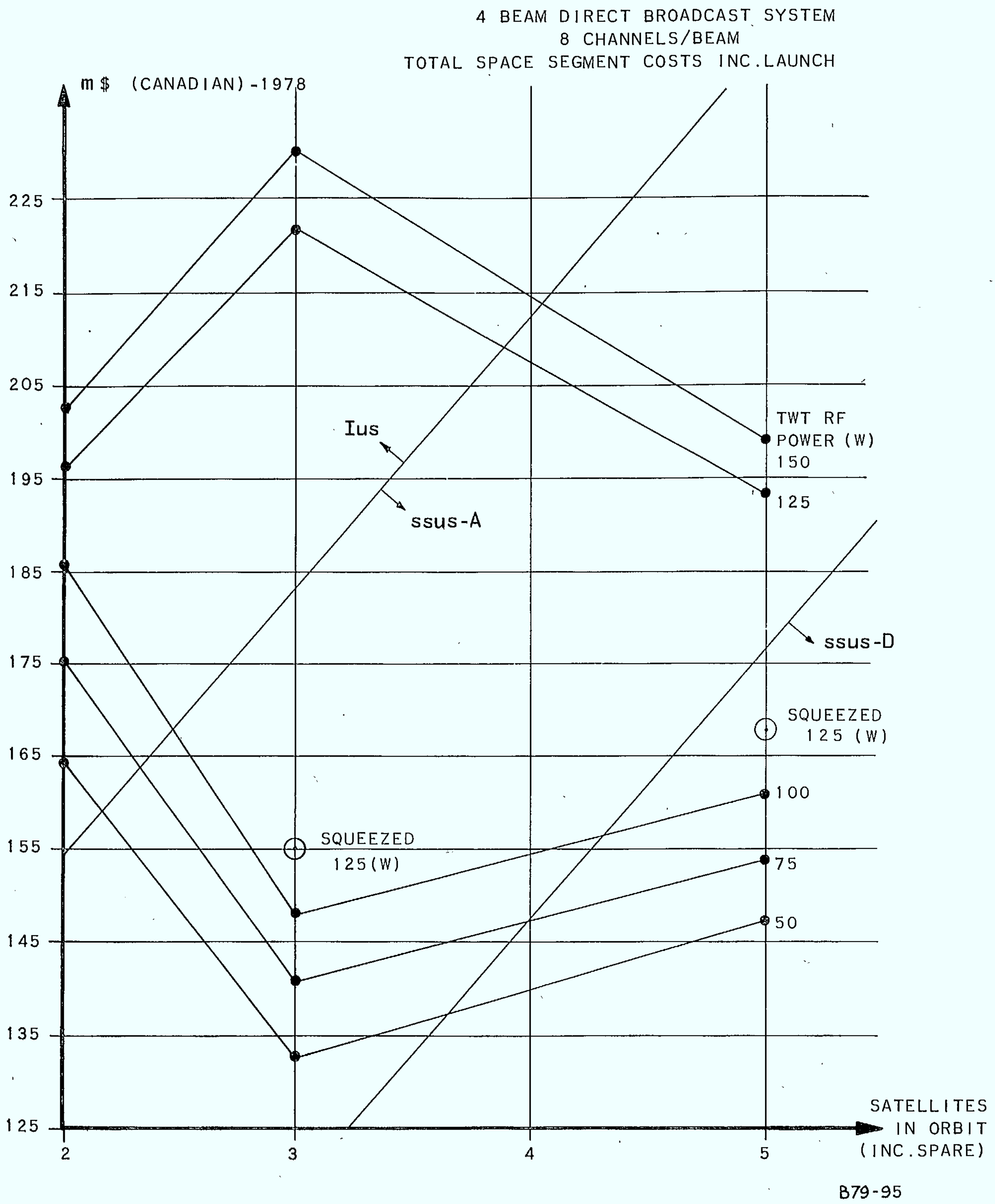
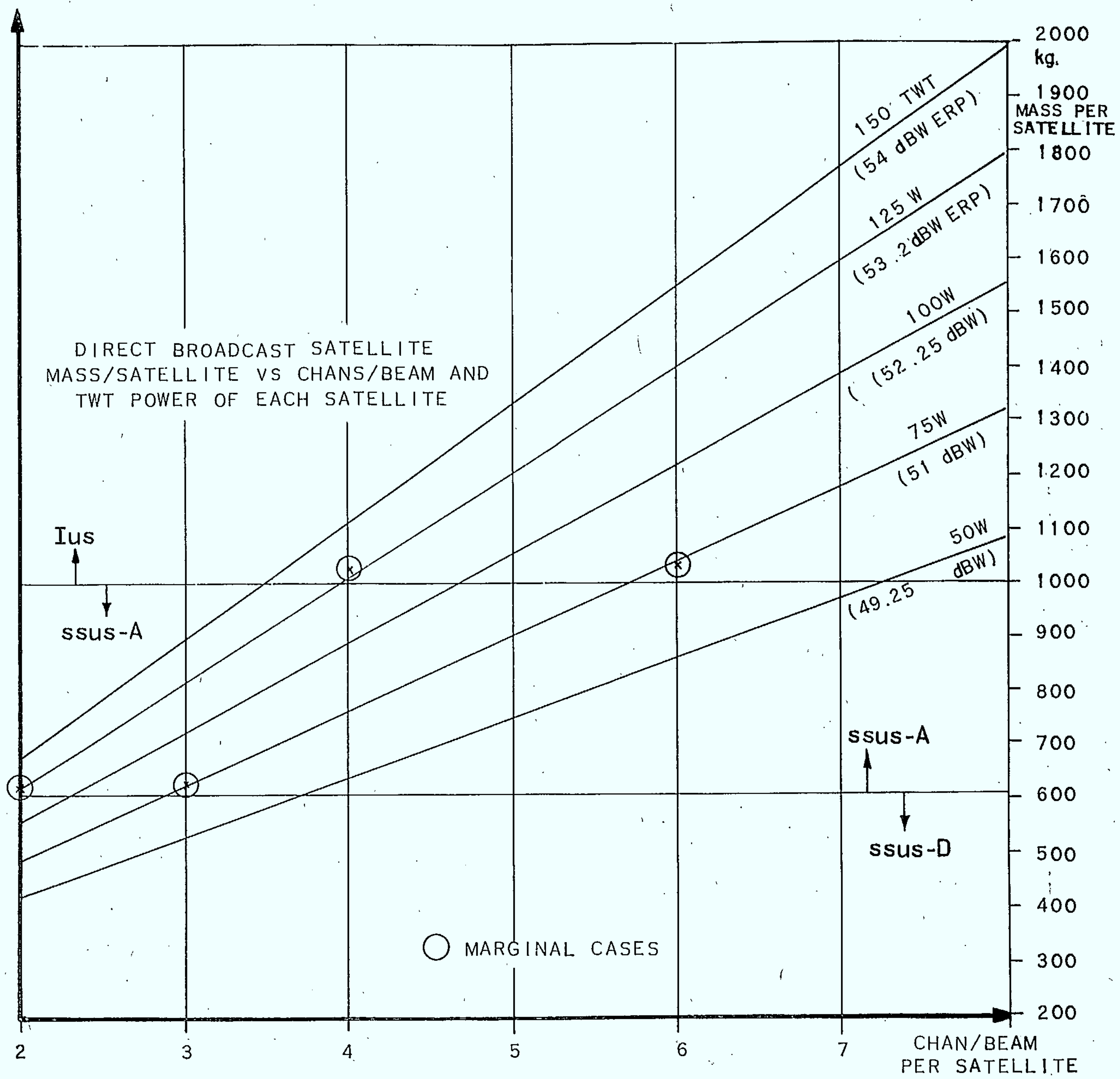
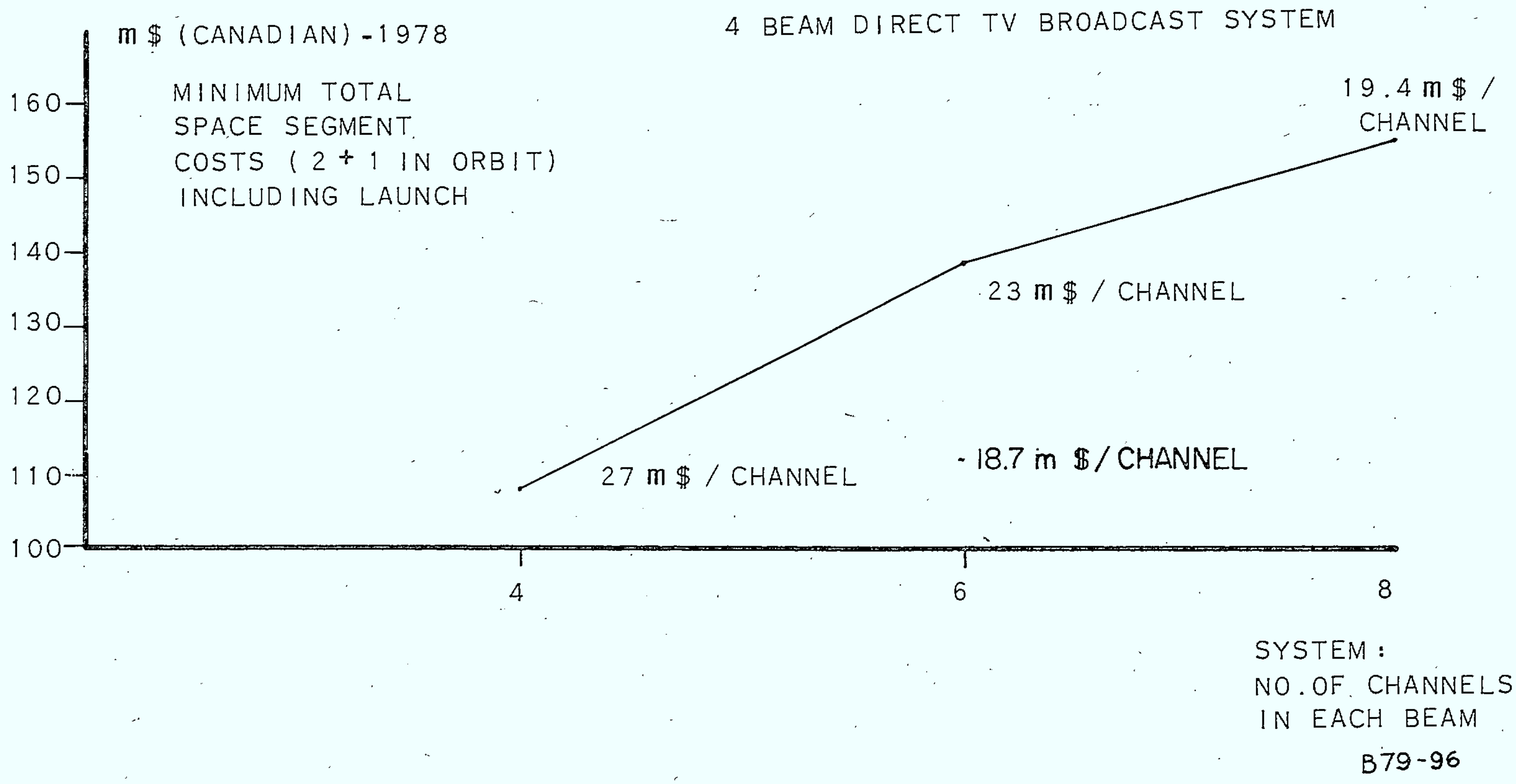

4 BEAM DIRECT BROADCAST SYSTEM
8 CHANNELS/BEAM
TOTAL SPACE SEGMENT COSTS INC.LAUNCH
m $ (CANADIAN) -1978
225
215
205
195
185
175
165
155
145
135
125
Ius
ssus-A
ssus-D
TWT RF
POWER (W)
150
125
SQUEEZED
125 (W)
100
75
50
SQUEEZED
125(W)
SATELLITES
IN ORBIT
(INC.SPARE)
2
3
4
5
B79-95

Figure 6.3.3



B79-95

Figure 6.3.4



4 BEAM DIRECT TV BROADCAST SYSTEM
m $ (CANADIAN) - 1978
MINIMUM TOTAL SPACE SEGMENT COSTS (2 + 1 IN ORBIT) INCLUDING LAUNCH
160
150
140
130
120
110
100
4
6
8
27 m $ / CHANNEL
23 m $ / CHANNEL
19.4 m $ / CHANNEL
- 18.7 m $ / CHANNEL
SYSTEM: NO. OF CHANNELS IN EACH BEAM
B79-96

Figure 6.3.5

7. CONCLUSIONS AND RECOMMENDATIONS

a) Retaining a spare satellite in a fully operational position can be justified for a regional satellite system provided a sufficient volume of pre-emptable traffic or peak traffic loading is available. If such traffic is available, this appears to be the preferred method of sparing.

b) A spare satellite can be retained in an interstitial position between two fully operational satellites. Periodic health monitoring can be carried out in this position provided the time at full power for each channel is much less than two hours. Post launch commissioning test can be carried out in the same manner provided the time restriction is adhered to. This appears to be the second most preferred method of satellite sparing.

c) Positioning the spare satellite in the same orbital slot as the primary satellite is feasible for the fixed regional services but requires some additional hardware on the spacecraft. Tests are carried out using the out of channel command beacon. The difference between in channel and out of channel performance must be calibrated by ground tests or post launch commissioning tests.

d) A dedicated orbital position for post launch commissioning tests for all satellites was considered. However, because of the penalty in station keeping fuel to move the satellite from this position to its final position this procedure is not recommended. The absence of this commissioning slot leaves the method of post launch testing of the co-located spare for fixed regional services unresolved. One possibility is to stop the satellite in the adjacent interstitial location and subsequently move it into the co-located position.

e) Co-located spare satellites are recommended for the direct broadcast service. It is considered that in the foreseeable future there is sufficient system down time in the middle of every night to allow fully testing the spare without additional hardware or other penalty.

f) The fuel expended to move a satellite from one orbit location to another increases as the drift time decreases. The remaining useful lifetime can be maximized by making the drift time equal to the loss of life due to the expenditure of station keeping fuel.

g) Techniques are proposed which should ensure that collisions do not occur between co-located satellites.

h) A more detailed examination of the testing procedures for an interstitial satellite is required. The tests to be performed should be confirmed for both the initial commissioning tests and the periodic health monitoring. The time required to carry out the tests in both instances is critical and needs to be minimized. The optimum spectral characteristics of the test signal need to be established and the ground station test equipment outlined. The content and wording of the proposed modification to CCIR Recommendation 466-2 needs to be reviewed in the context of the new requirement.

8. GENERAL BIBLIOGRAPHY

- 'Communications Satellites: Looking to the 1980's'. H.L. Van Trees et al, pp 43-51, IEEE Spectrum Dec. 77

- 'Cost Competitive Space Transportation System for Geostationary Payloads'. C. Richard Jones. Paper No. 76-278 AIAA 6th Communications Satellite Systems Conference, Montreal, April 1976

- 'Design Factors Affecting Communications Satellite Lifetime' R. Strauss and J.R. Owens. Paper 77-21, International Astronautical Federation (I.A.F.) XXVIIIth Congress, Prague, Sept. 77

- 'Ultimate Communications Capacity of the Geostationary Satellite Orbit', J.K. S Jowett et al, Proc IEEE, Vol. 116, No. 8, August 1969

- 'Satellite Spacing and Frequency Sharing for Communication and Broadcast Services', J.L. Hult and E.E. Reinhart, Proc. IEEE, Vol. 59, No. 2, Feb. 71

- 'International and Domestic Satcom Regulations', R.S. Gould, Telecommunications, Nov. 1978

- 'Advanced Technology for Direct TV Broadcasting Satellites' D.E. Koelle, Paper 78-634, AIAA 7th Communications Satellite Systems Conference, San Diego, April 1978

- 'Future Use of the Geostationary Arc', R. Stammincer, Telecommunications', Nov. 1978

- 'Diversified, Cost-Effective Satellite Service for the 80's', D.S. Bond and J.E. Keigler, Paper 76-221, AIAA 6th Communications Satellite Systems Conference, Montreal, April 1978

- 'Analyzing the Interface of Reliability and Economics of Unmanned Satellites', H.W. Von Guerard, IEEE Transactions on Reliability, Vol. R-23, No. 3, August 1974

- 'The Role of Satellites and Cables in Intercontinental Communications, S. Ramji, Paper 76-308, AIAA 6th Satellite Communications System Conference, Montreal, April 1976

- 'Repairing a Communication Satellite in Orbit', G. Gordon and W. DeRoches, Comsat Technical Review, Spring 1976

SPARE SATELLITE POSI-
TIONING STUDY.