Library

(1)

UHF COMMUNICATIONS SATELLITE SYSTEM

FINAL REPORT / VOLUME I

RCA

Space Systems

RCA LIMITED
1001 Lenoir St., Montreal 207, Canada

checked 11/83

UHF COMMUNICATIONS SATELLITE SYSTEM

FINAL REPORT VOLUME I

The work reported here was done under DDS Contract OPL1-0005 "Consulting Services for Cost Studies of UHF Satellite Communications Systems" and under the design authority of the Communications Research Center of the Department of Communications.

Prepared by:

Government and Commercial Systems Division,
RCA Limited, Ste. Anne de Bellevue, Quebec

Project: 2217-0

December 9, 1971

TABLE OF CONTENTS

1.1 System Requirements and Constraints

The system requirements, and the constraints under which it was to be designed was given in the baseline definition at the beginning of the study.

The outline of the system and basic requirement is well stated in the baseline definition as follows:

"The UHF satellite communications system to be studied under this contract basically is intended for low capacity voice telephony service to remote areas of Canada. The system is envisaged to comprise a large number of terminals accessing a satellite transponder in a single channel per carrier frequency division multiple access mode. Terminals to be used in the system will range from two-way voice telephony fixed terminals to air and sea mobile terminals as well as radio program channel terminals and telemetry terminals for remote sensing platforms.

One purpose for carrying out this study is to develop cost figures which can be used for comparison with other systems which can meet some of the requirements. Therefore a primary requirement of the study is that it concentrates on a portion of the overall system requirement that can be used as a baseline for the comparison. This portion is designated to be that part needed to meet the two-way voice telephony service requirements to fixed and transportable stations. The other applications of the system, namely the provision of mobile, radio program distribution, and telemetry services, are to be considered as of secondary importance for the purposes of this study and should be treated only to the depth that the scale of contracted effort permits. It should be borne in mind that these secondary services will nevertheless be required and the implications of their

1.1 Cont'd.......

inclusion in an eventual system must be taken into account in the transponder design.

Several constraints are placed on the two-way terminal design by the intended application in remote areas with severe environmental conditions. These are:

i. the terminals should be small, easy to transport and place into operation

ii. they should operate fully automatically and those intended for commercial service should be capable of interfacing to local telephone exchanges.

The earth segment requires large numbers of terminals. Therefore cost trade-offs should be made between the space and earth segment portions of the system so that within the constraints placed on the configuration of the ground terminals the net result is the most economic system meeting Canada's unique requirements.

Users of the system will be both military and civilian. Military uses will be of a tactical nature and will be mainly mobile and transportable. Civilian uses will be mainly to fixed and transportable stations. Some overlap between the types of uses by both kinds of user can be expected. Of prime interest to both types of users is flexibility and ease of deployment to meet rapidly developing and constantly changing situations.

These notes include the likely range of terminal types which may be used in a system. However detail design should be undertaken only for the fixed and portable terminals required by military and civilian users."

1.2 Traffic Model

Two systems were to be considered, with identical traffic models. The greatest emphasis was placed on a 1500 MHz system, as that appeared most likely for potential implementations, prior to the 1971 WARC meeting. After the WARC meeting the allocation turned out to be 2.5 GHz, however the analysis at 1.5 GHz is generally valid and necessary modifications are indicated in the body of the text. The other system was in the 225 - 400 MHz.

Table 1.2.1 summarizes the traffic requirements for telephony and radio program distribution. Mobile terminal requirements are not included.

TABLE 1.2.1

	Minimum Model	Maximum Model
Fixed Terminals Civilian Users	100	400
Transportable Terminals Civilian Users	20	60
Radio Channel Terminals	20	80
Fixed Terminals Military Users	20	50
Transportable Terminals Military Users	20	50
Transponder Channels Commercial Quality	20	60
Transponder Channels Military Quality	20	40
Transponder Channel Program Quality	1	5

The channel capacities are duplex channels. The system is predicated on demand assignment and each station is under the central control of a Network control station.

1.3 Launch Vehicle

The launch vehicle to be used for the space segment for purposes of this study program is designated in the baseline definition as the Thor-Delta. In the same document it is stated that the program is to be considered for potential implementation in the 1975 - 1978 time frame, this permitting a reasonable extrapolation of the launch capability. The Thor-Delta has shown substantial capability growth over the past few years from about 950 lbs (excluding the interstage adapter) in early 1968 to the 1500 lbs presently allocated to the CTS program. There appears to be some possibility of a 1700 lb capability in the 1975 time frame, and a high probability by 1978. It may be noted that the nature of a total (twelve year) system would under some circumstances require additional launches at the program mid point, i.e. about 1980 - 1985, at which point the Thor-Delta may have more capability or even may have been replaced.

For purposes of this study a transfer orbit capability of 1500 lbs has been assumed for several reasons:-

(a) If increased capability were not required in some other program by the time of launch, substantial non-recurrent costs could be assessed against this program and would invalidate the costing.

(b) Only a small portion of the 200 lbs difference becomes available to the spacecraft payload. For example, the apogee motor absorbs over 100 lbs of this. For weight margins, 5% or 10 lbs should be allocated. Secondary propulsion fuel absorbs about a further 10 lbs, leaving about 80 lbs. which must include all other changes, i.e. the strengthening of the structure, any interstage effects and, of course, assumes that there is no additional demand on the power or othe subsystems.

(c) The detailed influence of the proposed fairing has not yet been derived.

Thus the 1500 lbs has been selected as a conservative launch capability or alternatively it may be considered that an additional weight margin of about 3% (of transfer orbit weight) probably would be available at the time of implementation.

1.3 Launch Vehicle Cont'd

The fairing selected for use in this mission is that of the so-called "straight-eight" as shown in Figure 1.1, which is presently under discussion for the NATO 111 mission. It appears likely to be a standard fairing in the time frame of interest. The eight foot diameter fairing offers the opportunity of designing a stable duel spin configuration, and allows a lower center of gravity than the Telesat or smaller diameter fairing s, thus easing the interstage adaptor loads and the problem of launch vehicle dynamics. Further, if this program were to be implemented under circumstances where the program had to absorb the fairing development, these costs are not prohibitive.

1.4 Spacecraft Design Alternatives and Trade-Offs

1.4.1 Introduction

The final design of a spacecraft evolves from considerations of two primary inputs - the traffic and performance model and the operational philosophy of the system.

The traffic model represents the predicted performance demands of the various types of users and the traffic patterns and density if the system is available (and usually assumes an unlimited system capability). In terms of the spacecraft designer the traffic model translates almost directly into performance demands on the satellite or satellites designated operational in the sense of carrying that traffic.

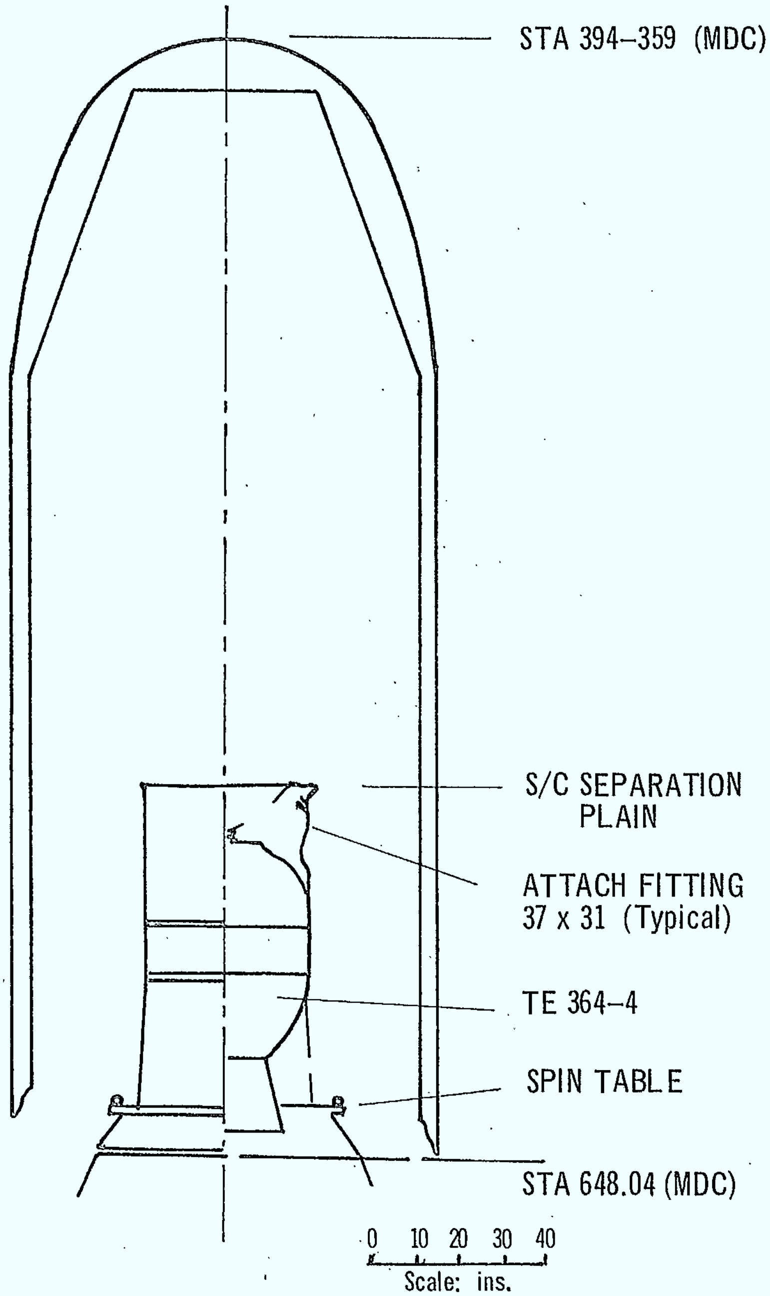


Figure 1.1 – Launch Configuration – MDC 8 ft. Fairing

1.4.1 Cont'd

The system operating philosophy ranges over a wide range of topics almost all of which tend to constrain the designer. This philosophy may specify the launch vehicle, the reliability of service, allowable outage periods, tracking and station-keeping intervals and procedures, launch back-up requirements, replenishment cycles, system overload characteristics, the investment period of interest and so on. The philosophy may be stated in detail in which case , if not compatible with the traffic model and the general class of spacecraft capabilities, it may be a severe design constraint. Alternatively, as in this program, the designer may be required to develop the system philosophy within loose limits and the design tends to emphasize system optimization.

Consideration of the traffic model and philosophy leads to a design concept of the satellite, particularly of the communications subsystem, which then is traded off against possible configurations. Within potential configurations, the various subsystems are examined and the implications of each related back to the performance requirements and the process repeated. In the present study the terms of reference provide the traffic model and general system requirements. With the exception of general guidance provided by the requirements, document and by the contracting officer in the course of the study, the contractor has had to develop a system operating philosophy appropriate to the user's requirements and the economic objectives of the program.

1.4.1 Cont'd

Because the operating philosophy is normally determined by the operating entity and is subject to many alternatives and because it is a strong element in the program cost, it was deemed desirable to discuss the philosophy in some detail. Thus, the next section discusses the system philosophy itself and as it relates to the spacecraft configuration and design.

1.4.2 Systems Analysis of a Commercial Operational System

In considerations of the trade-offs available in performance, reliability and availability of a commercial space communications system, it is found that there are many factors involved outside the direct concern of the spacecraft engineer. It is the intent of this section to delineate these factors, the terminology used and the nature of the trades so as to permit an appropriate choice of system philosophy. Unfortunately, there is no true optimization of the system, only a best compromise against any set of acceptable and largely arbitrary criteria.

To illustrate the type of problem one may examine briefly three types of overall programs. In the purely scientific space program, one is generally concerned with a specific mission in the sense of obtaining information on one or more aspects of the universe. Thus, usually one is concerned with two spacecraft - the second protecting against launch failure and not intended ot fly. In terms of program optimization, a criterion might be dollars per scientific bit gained, which at least is more readily established than the ratio of dollars spent to understanding gained. However, "Understanding gained" provides a weighting factor towards unexpected observations on early

1.4.2 Cont'd

data, with subsequently less improtance towards later perhaps redundant data. Thus, in a sense mission success in a scientific program tends to obsolete the spacecraft, the payload, the particular mission orbit, etc. If the mission requires repetitive synoptic data then one is dealing with what might be termed a scientific operations satellite.

In a pilot program one is setting a limited mission with which to demonstrate an operational system but is willing to accept limitations which would normally not be acceptable. Thus, the mission might be to demonstrate feasibility and then generally to develop the procedures to be used in the final system. It is also obsoleted by success in that the operational system is likely to be implemented, although on occasion the pilot may merge with the operational system.

The operational system is generally of a commercial or sociological nature fulfilling a need, and thus the mission is of continuing importance and becomes obsoleted only by inability to provide service - either quality or quantity. Typically, if the service is useful, its value or apparent necessity becomes greater the longer it is available, particularly if it obsoletes any existing alternatives. Further, the general nature of the market is such that the demands increase in time, i.e. the system is not self-obsoleting but rather obsoleted by the demands on it, implying replenishment or expansion of capabilities.

1.4.2.1 Accessability

From the standpoint of the user, he is really only interested in being able,

1.4.2.1 Cont'd

when he so desires, to make use of the service provided to him, i.e. the reception of audio services of an acceptable quality, or the ability to contact another subscriber with a connection of acceptable quality. Any time the subscriber cannot complete an acceptable connection by reason of any factor other than lack of availability of the other party, i.e. equipment failure, excessive demands on the satellite, sun transit, spacecraft manoevers, etc., the accessability is decreased and the system has not fulfilled the "social" requirement.

The specifier of the traffic model in effect takes this problem out of the domain of the sedigner except in regard to graceful degradation as discussed later.

1.4.2.2 Availability

The specifier of the total system, normally the operator, from the statistics of the traffic, i.e. the accessability demands, develops projections of demands for availability of the system. As a matter of economics, this usually entails reduction of accessability by the user at peak demand periods or alternatively the system performance may be relaxed at such times.

The specified capacity and performance may be required at all times or reduced performance may be allowed during certain periods (usually defined), such as eclipse of the satellite or sun transit of the ground station. Random factors which may be specified are outages during manoevers, changeover of units to cope with failures, or major outages associated with replacement operations.

1.4.2.3 Reliability

Reliability is in essence the probability at a given time that the unit, subsystem or system meets its specification. In practice at the higher levels one neglects for purposes of calculation the warm up period, change over time or replacement period, but these periods must be examined in relation to accessability as they profoundly influence the system philosophy and thus the economics. For example, a spacecraft coming out of eclipse is in terms of specified (required) performance out of eclipse yet will require a period of time to come to thermal equilibrium. If such a period is not allowable, a major penalty may have to be paid in eclipse power capability to maintain thermal operating limits. Similarly, an in orbit standby spacecraft may require relocation to permit traffic changeover.

1.4.2.4 Graceful Degradation

It is notable that accessability and availability as defined here are largely independent in as much as one does not guarantee the other. For example, high user demands may develop a lack of accessibility while the spacecraft is within specification and nominally available. Conversely, a partially failed spacecraft having say reduced eclipse capability is not available in the sense of meeting specification, but may be accessible to a user possibly giving full performance. Somewhat similarly, the top level reliability associated with complex changeovers in the system.

This lack of correlation between system characteristics suggests another important design parameter or at least design goal - graceful degradation of performance quality and acpacity. For example, a spacecraft nominally

1.4.2.4 Cont'd

outside the specified inclination will result in a loss in received power at the ground. Under some operating conditions this could result in its being unavailable yet able to handle full traffic of lower quality or by various manipulations of the signals, less traffic but of specified quality. Of course, apparent full operation may be obtained if the traffice model has overstated the demand. There are various other graceful degradation techniques available to the designer such as multiple batteries which involve weight and complexity penalties, as well as alternative operational procedures such as sacrificing battery charging and thus eclipse operation in the case of a severely reduced solar array output. In all cases, the attempt is to maximize the accessibility despite the nominal failure of the system.

1.4.2.5 System Design Concepts

The specification document for this program provides a traffic model as has been discussed earlier. The question of accessibility has been initially treated by taking the model maxima as the design targets. Only in the case where these produce severe design constraints which showed little possibility of being surmounted has consideration been given to backing down. In fact, the maxima have been retained although the eclipse capabilities have been reduced at the lower frequency assignment. The argument is that on a spacecraft having a relatively small number of channels, the economics of the system are extremely sensitive to that number when stated as dollars per channel. The implicit assumption is that the demand is such as to produce a high load factor, i.e. the traffic model is realistic and representative of channel "sales". Thus, dollars per

1.4.2.5 Cont'd

channel year is commonly used as a system figure of merit.

Availability has been considered with respect to both graceful degradation and reliability. For example, the anticipated reliability figures for a single spacecraft are such as to generally meet the requirements document, but if only one spacecraft were in orbit, its failure would produce a system outage of three to six months minimum. An outgae in excess of a few days in an operational system was considered unacceptable and therefore, the system was based on orbit standby. Even then there is a major choice as between on station or off station spare as is discussed in a leter section.

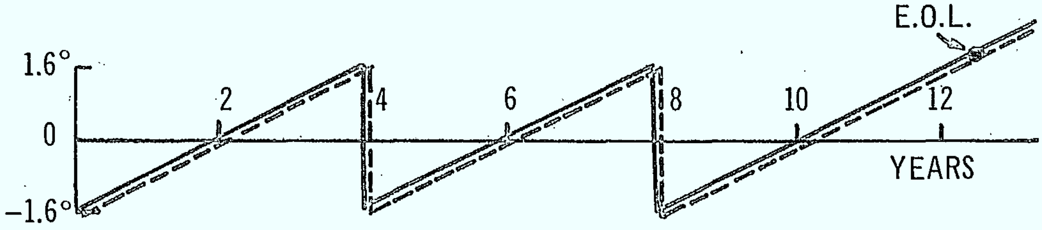
1.4.3 S/C Operations Trade-Offs

There is a distinct class of trade offs based on the operational considerations and total system aspects. Trade offs in this area are of such factors as lifetime, station keeping ability, eclipse capability and spacecraft reliability. Of these the lifetime is a particularly strong influence on the total costs of implementation, affecting the number of spacecraft, launch intervals, program duration and risk; as well as the economic figure of merit-dollars per channel year. All these trade-offs are, of course, interrelated to spacecraft configuration and design but at a first iteration can be treated separately.

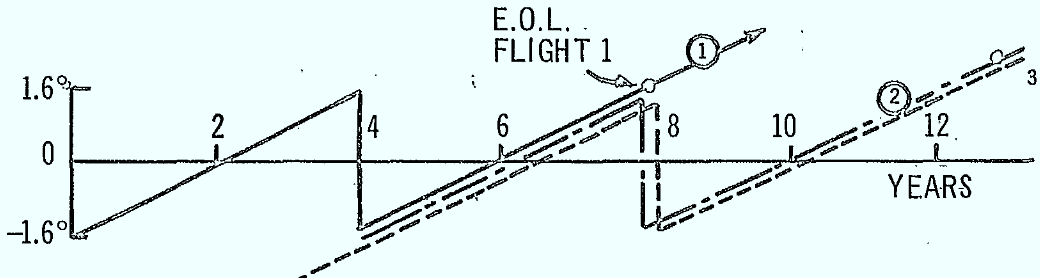
1.4.3.1 Lifetime

The requirements document calls for a twelve year system, with the assumption of zero value after this time. Assuming unlimited spacecraft resources, the simplest and cheapest program consists of a three spacecraft/two launch program, i.e. one operational spacecraft, one on orbit standby, and a third to protect against launch failure as shown in Figure 1.2(a). Indeed, in a minimum program, the third might be the prototype. Assuming the fuel is available and the reliability figures themselves are acceptable, one is troubled by the validity over a twelve year period of the basic failure rate input data, and particularly, which devices on the spacecraft exhibit shelf life for wearout modes not incorporated in the random failure rates. In terms of the spacecraft procurement, costs are a minimum not only because of the minimum number of spacecraft involved, but also, because the program duration is a minimum avoiding restart costs and inflation in the spacecraft and launch costs. The other question is in regard to the validity of the traffic model over such a long period. In the present study, it became apparent that such a minimum program was probably not realistic, although with the reservations as above, Figure 1.3, curve a shows the system reliability is adequate. What is not revealed is the weight penalty associated with such a design as is discussed later.

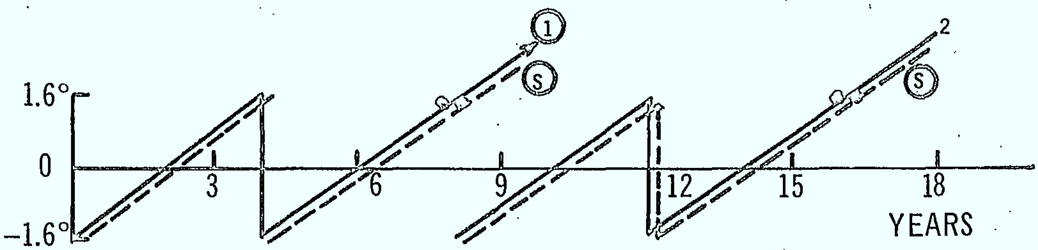
An alternative philosophy was developed based on a four spacecraft procurement as shown in Figure 1.2(b). It was found that the spacecraft resources could be allocated on the basis of a four year inclination fuel supply and with an appropriate orbit bias eight years of "in the slot" operation obtained. The fuel load was thus sized for one inclination



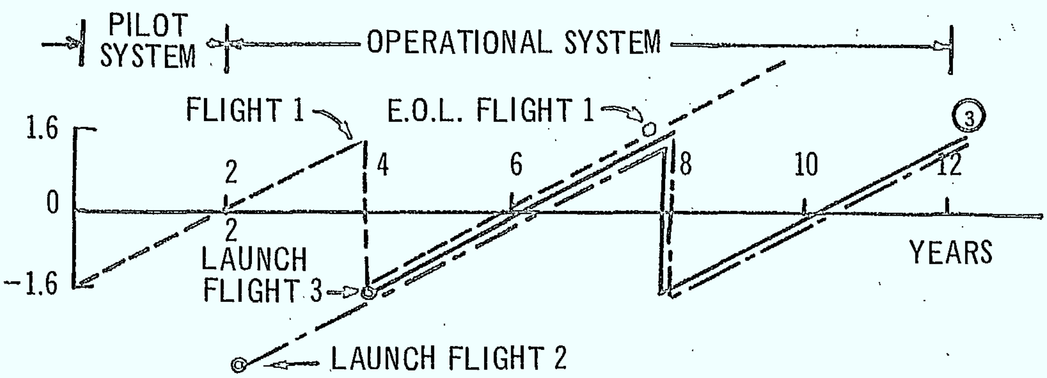
(a) 3-Spacecraft Mission – 1 Operational
1 In-Slot Spare] 8 Years Inclination Fueling
1 Ground Backup



(b) 4 Spacecraft Mission – 2 Operational [4 Years Inclination
1 In Orbit Spare [4 Years Inclination plus 12 Years Altitude & Longitude
1 Ground Backup



(c) 5-Spacecraft Mission – 2 Operational
2 In-Slot Spares
1 Ground Standby



(d) Combined System Strategy – 2 Operational [4 Years Inclination
1 In Orbit Spare [4 Years Inclination
1 Ground Backup

Figure 1.2 – Allocation of Spacecraft for Various Missions

1.4.3.1 Cont'd

correction equal to four years of inclination buildup plus attitude and longitudinal drift compensation fuel for eight years. If one accepted an outage period of 24 hours maximum on failure of the operational spacecraft, it is feasible to have an out of slot on orbit standby in which case the standby spacecraft would be nominally available for about 12 years even though fueled for four in inclination and eight in attitude and longitude. Although the reliability numbers (as shown in Figure 1.3, curve b) are appropriate to the requirements, their validity starts to be in question.

Further, the third launch is now a minimum of four years and a maximum of eight after that of the first. With such a long period one is unlikely to build the third spacecraft until required unless very extensive refurnishing is contemplated. Thus, in either case, the procurement activity is lengthened with attendant increase in costs and a high probability of further inflation. The choice between a four and eight year launch is determined by balancing the factors of maximizing reliability over the period, maximizing reliability near the end of the period (presumably the time of maximum traffic), having the system value truly zero at twelve years, and minimizing the program cost. It will be noted that this system involves four spacecraft with one standing by and three launched.

It may be noted that additional fuel on flight one would increase the system reliability significantly and, in particular, that if the fuel could compensate for an eight year buildup of inclination and twelve years of attitude, either fllght one or two could act as backup on flight three. It is also noteworthy that the fuel carried to remove booster dispersions is

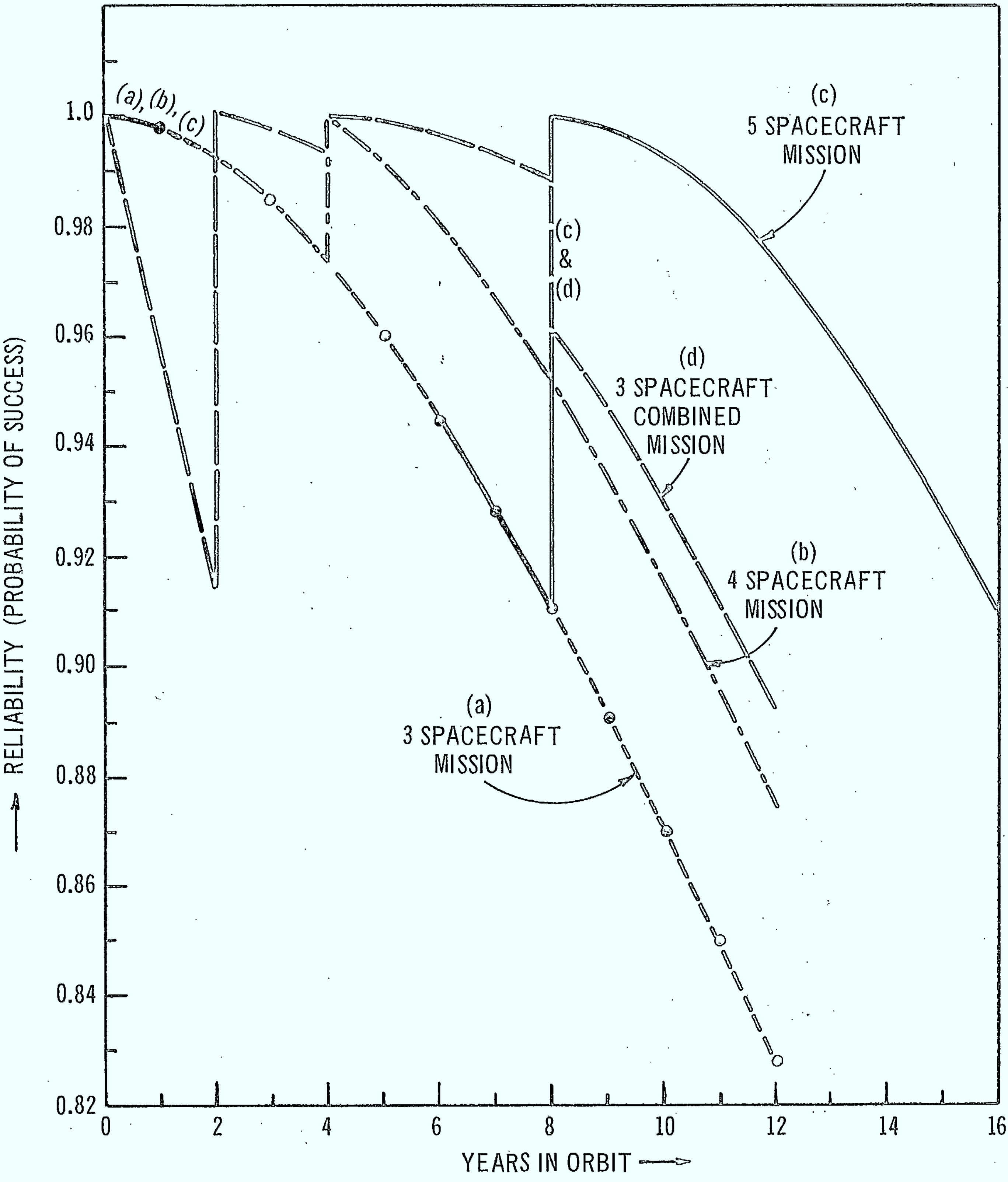


Figure 1.3 – Reliability for Various Missions

1.4.3.1 Cont'd

adequate for a further lifetime in excess of one year and that for some time a degraded backup is available even on drifting from the slot.

If the concept of a twelve year lifetime satellite is unacceptable, the addition of one more spacecraft provides a conservative system as shown in Figure 1.2, curve c, and Figure 1.3 (c). The choice is then whether to maintain the nominal eight year lifetime and have the system go to zero value at sixteen years, or to maximize the reliability, regain some weight by carrying less fuel and have the system zero out at twelve years. Alternatively, for the shorter system life some EIRP could be regained by tightening the dead band in station keeping which is discussed next.

1.4.3.2. Station Keeping

The trade-offs based on orbit inclination removal and attitude and longitude control are strongly related to the trades in lifetime. Because the secondary propulsion fuel is such a major portion of the total initial on station weight, trades here are a strong factor as to whether a design can be lifted off. Inasmuch as inclination removal is the major portion of the fuel, the relevant trades must be examined with great care.

In synchronous satellites such as the Intelsat series which operate with high directivity ground stations, to a first approcimation the fuel consumption is almost independent of the station keeping dead band or limits. There is a small loss of thruster efficiency associated with brief firings but this is a few percent. The smallest realistic dead band is set by the orbit measurement and spacecraft attitude accuracies as well as thruster predictability. Ignoring these, the lowest

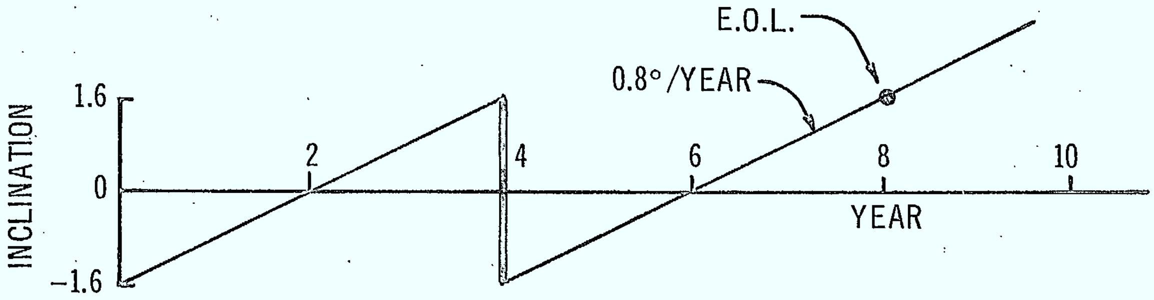
1.4.3.2 Cont'd

limit is when the dead band could be kept so small that diurnal disturbances required additional fuel consumption. However, the dead band being independent of the fuel consumption is only valid where the number of corrections required to maintain station is large over the satellite lifetime. Where such is not the case, substantial fuel and thus weight savings are possible by opening up the "station". Figure 1.4(a) shows a typical approach in that the inclination buildup over four years of life is removed. Figure 1.4(b) shows how substantially the same fuel could be utilized in a series of shorter thruster firings.

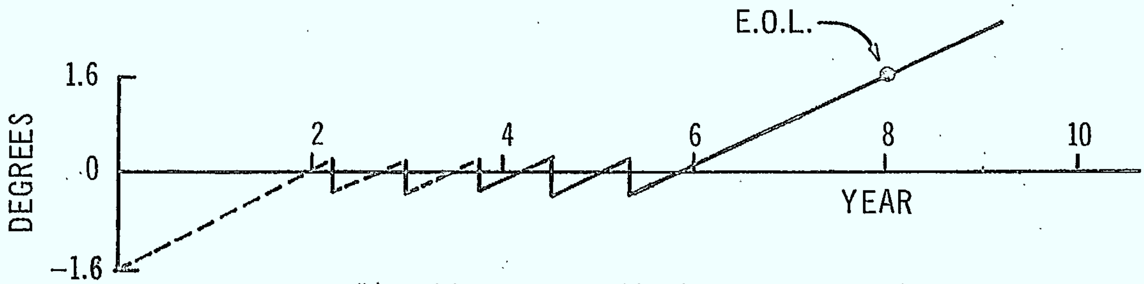
Even more substantial advantages accrue when one is satisfied with an "off-station" standby, with the spare gradually drifting into position at the time of anticipated loss of the operational satellite (usually by moving outside the dead band). The off-station spare years are not totally free in that some fuel will be required to maintain longitude and attitude, and there is a probability of fuel expenditure to come into the slot. The outage of such a movement, in general, will be a maximum of 24 hours. In the off-station spares it is assumed that only inclination is off, which saves fuel and gives a minimum changeover outage as well as simplifying satellite control. Figure 1.4 (c) shows a typical launch sequence.

In the present case, the desired ground station diameters coupled with an appropriate loss of EIRP (0.3 dB) suggested a dead band of $\pm$ 1.6^{o} as a practical compromise, i.e. removal of four years inclination buildup over an on-station lifetime of eight years.

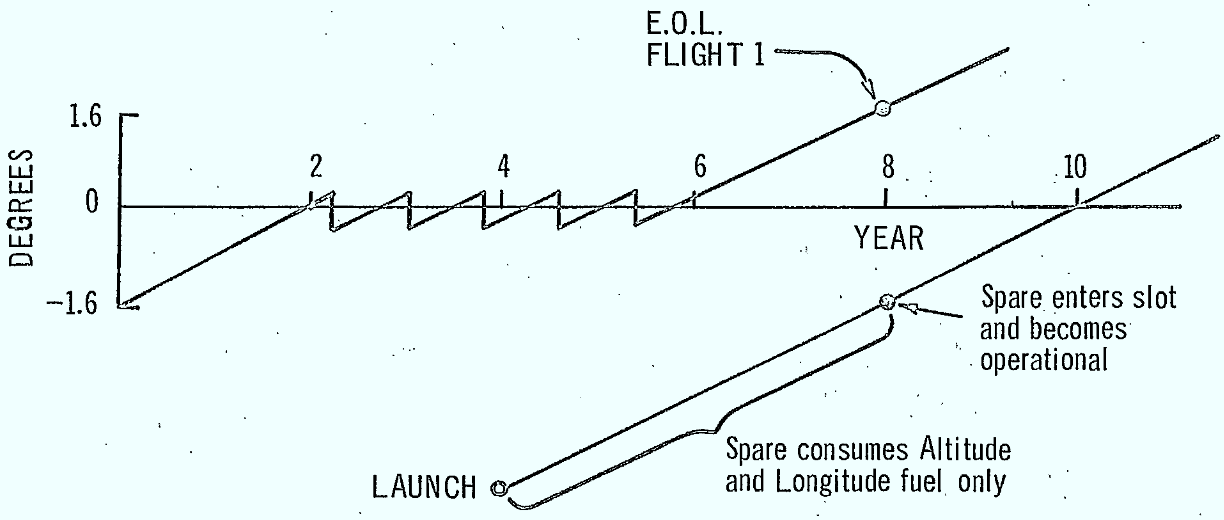
It may be noted that such on/off-station trade-ffs are not desirable under all circumstances. For example, with a traffic model exhibiting high peaks beyond



(a) Single operation removal of inclination



(b) Multiple removal of inclination with same fuel load as in (a). Note period of minimum deadband over center four years.



(c) Off-station spare providing backup with minimum fuel consumption.

Figure 1.4 – Station Keeping Strategies

1.4.3.2. Cont'd

a single satellite capacity, the spare might be kept in an allocated slot and used for a separate traffic net at peak periods. The system degrades relatively gracefully for a single satellite failure but such may entail an outage for relocation of the spare to the operational slot or time for the repointing of the ground stations. Although such a scheme of in slot spares is feasible with a five spacecraft - 12 year system and indeed is worthy of greater analysis it has not been detailed here because of another factor. Becuase of the low ground station spatial discrimination slot to slot spacings are likely to be large in which case fuel consumption for slot changeover in a reasonable time becomes a real factor in that fuel has to be used for both acceleration and deceleration. The off-slot standby, as noted also suffers somewhat in that there is a probability that fuel will have to be expended at the time of an operating satellite failure.

There is another trade-off factor which must be considered, and that is in regard to hot standby. For present purposes, it is assumed and it is virtually a certainty that thermal considerations and housekeeping operations will necessitate operation of most of the normal complement of subsystems, and particularly, virtually the entire communications system. Thus, the standby satellite will be characterized by normal operating failure rates and not standby or non-operating failure rates. The standby rates, of course, would apply to all non-operating redundancy carried.

Finally, it is possible to combine strategies, using on-orbit/off-station sparing early in system implementation with a shift to a second slot operating spare as the traffic develops. Such a program is illustrated in

1.4.3.2. Cont'd

Figure 1.2(d), where a 12 year mission is obtained in the following way. The first 2-4 years is a pilot operation (no in orbit spare) followed by an operational period of 8-10 years with coslot backup for rapid changeover or separate slots for a peak traffic dual network. Three flight spacecraft with 4 years inclination fuel are required.

One other factor is of concern and that is in regard to orbit inclination biasing. While such procedures narrow the launch windows and restrict the available launch times, which is a consideration where a failure at launch must be rapidly compensated, it is felt that the broad dead band does give some latitude in the launch time.

1.4.3.3. Reliability

The reliability of a spacecraft is subject to trade-offs particularly against weight. With the exception of critical subsystems such as command most redundancy is non-operating and thus the demand on the spacecraft resource is in weight rather than power. The other class of techniques applied to improve reliability is to allocate contingencies, particularly in respect to solar array power, battery capacity and secondary propulsion fuels.

In the design of an operational spacecraft such as for communications, the lifetimes involved are generally such that a high degree of reliability is demanded. Thus, usually as a matter of course, one for one redundancy of basic electronic units is required and the spacecraft resources allocated. This is the case for the telemetry and control subsystem, earth sensors, valve drivers in the reaction control system, power converters and the like. For

1. 1.4.3.3. Cont'd

other units such as the solar array and battery, margins for single component failure are incorporated as a part of the design phase. Finally, there are a large number of units such as the apogee motor, despin motor, antennas, structure and the like, where conservative design is the only recourse.

Usually the procedure is to design and to cope with most single unit or component failures and then to attempt to cope with second failures through redundancy of function or alternative modes of operation. While many backup modes are designed in during the initial stages and others developed in the course of failure mode analysis of the actual designs, it seems to be a fact that many backup alternatives are invented as a consequence of failure in test or in orbit.

The communications transponder usually incorporates as much redundancy as possible. The receiver is usually fully duplicated and may include extensive cross strapping. Power stages are a problem in that not only are the active devices of relatively low reliability but the introduction of extensive switching of units introduces the probelm of swithc failure itself and often, more significantly, increases the r.f. power loss. Additional redundancy can also produce problems in feedback and system matching and flattening to the point of diminishing returns.

The reaction control subsystem is also a severe design problem in terms of reliability. While the fuel load invariably carries reasonable margins, the spacecraft resource of weight invariably prohibits a redundant fuel load. The problem is to allocate component redundancy in such a manner as to allow a single failure to occur without loss of fuel yet retain access to the total fuel load.

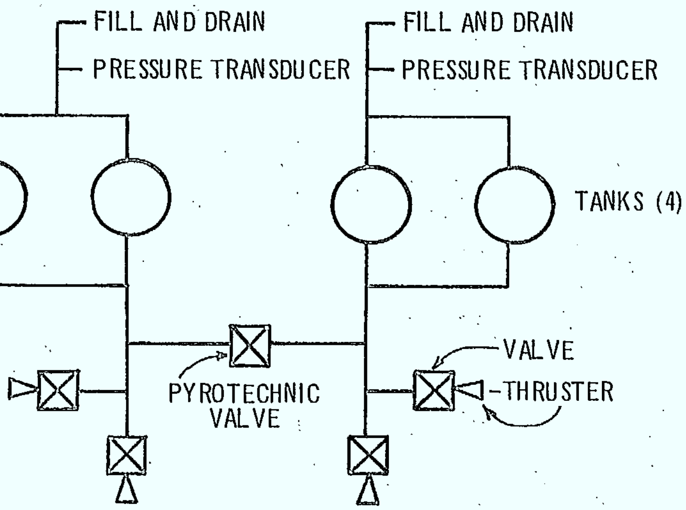
1.4.3.3. Cont'd

Three typical approaches to such systems are shown in Figure 1.5. Any of these systems in implementation would incorporate redundant valve drivers and possibly other features such as double seated valves, etc. The main point is that beyond basic redundancy, additional backup may reduce reliability at least numerically even after an extensive failure mode analysis. For eample, (a) is numerically less reliable than (b), but the hazards are different, relating to whether the fuel is available after a single failure of a given type. The final choice is often based on a gut feel of the subsystem designer. A similar choice commonly develops in the transponder subsystem where additional redundancy or cross strapping complicates the commands and test program, and introduces r.f. penalties as well as increases cost.

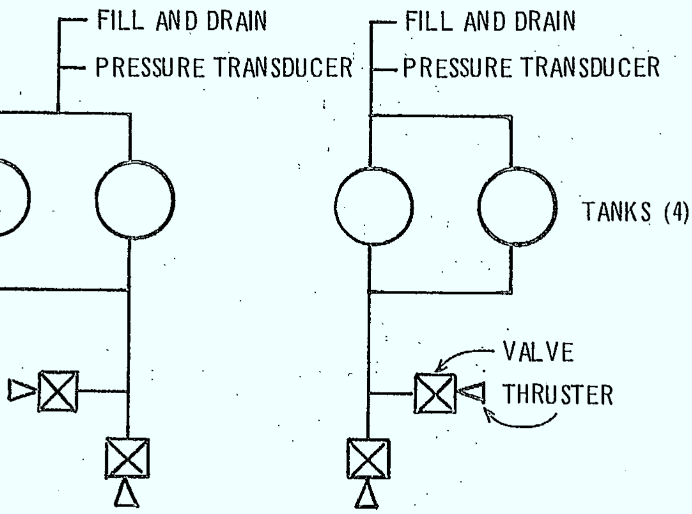
1.4.4 S/C Configuration Alternatives

After a relatively few iterations against the traffic model and system concepts with particular emphasis on the communications, a spacecraft objective is usually broadly defined in terms of weight, prime power, antenna size, pointing accuracy requirements and lifetime. From these initial data, a tentative spacecraft configuration is selected.

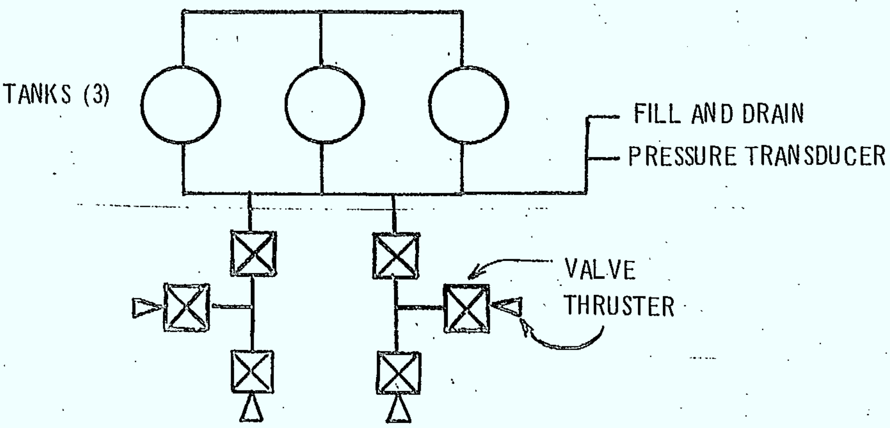
The simplest possible configuration is usually a spinning satellite without a despun antenna. Typically, the peak antenna gain is 0 dB for isotropic pattern, 8 dB for earth coverage and 14 dB for Canadian coverage. The isotropic case is essentially that of the early non-synchronous satellites such as Telstar and Relay or the telemetry system of more advanced configurations.



(a) Two half capacity systems with normally closed pyrotechnic valve



(b) Two half capacity systems – no cross connect



(c) Valve isolated manifold system. Note: 1. odd number of tanks
2. single Fill and Drain, Pressure Transducer
3. single seat thruster valves may be used

Figure 1.5 – Typical RCS System Configurations

1.4.4 Cont'd

The next configuration is to despin the antenna through electrical circuitry (the original Intelsat III concept,) despin of the reflector as used for ATS - which incidently could jetisan the reflector and degrade gracefully, or despin a portion of the natenna and a reflector - the Intelsat III approach. The duel spin system which results, gives <u>peak</u> gains about 19.5 dB on earth coverage or about 29.3 dB on Canadian coverage but in some cases, limits the polarization plans available. With greater complexity, more of the feed is despun giving the same gain and one achieves the Anik approach. With a greater portion of the feed despun, one has the possibility of changing the polarization. Neglecting other factors, this configuration is nominally applicable up to transfer orbit weights of perhaps 1500 lbs. The exact point at which the configuration is changed is determined by the prime power requirements, the complexity of the antenna field and, particularly that of the interface to the transponder. As a rough guideline, prime powers below 500-750 watts are candidates for duel spin, with something like 400 watts as the Thor Delta limit leaving a reasonable antenna volume. If the antenna configuration is much more than a relatively simple transmit/receive of single beams, the rotary coupler becomes complex and consideration must be given to a despun platform wherein the transponder and antenna are despun together.

The despun platform becomes of interest when the transfer orbit weight (generally indicative of complexity) exceeds 1500 lbs or the antenna field is complex. Typically, such spacecraft are conditionally stable by virtue of their interia ratios. The complexity of the r.f. rotary joint has been traded for that of transfer of power and control across the rotary interface. There is

1.4.4 Cont'd

a significant weight penalty associated with this configuration as a result not only of the slip rings but also additional duplication of subsystem units to reduce the reliability hazards. The despun platfom is considered to be of special interest up to transfer orbit weights of about 2500 lbs. Intelsat IV is an example of the configuration and it may be noted that the program was initially proposed at the 2200 lbs level and grew across the weight boundary suggested into the 3000 lbs. class.

Above the 2500-3000 lb range the three axis configuration is usually considered to be optimal on the basis of present technology.

As indicated, the exact weights at which configurations are changed are dependent on several technical factors, primarily power and complexity. In addition, there are several other factors primarily associated with the operator's or user's perferences, particularly in the operationsl aspects. There is a notable trend to stay with a stable configuration if at all possible, even at the cost of a larger launch vehicle. Similarily, the despun platform configuration is being pushed to the higher weight ranges. Further, to date, the three axis configurations have been avoided for operational systems. The trend has been reinforced by the growth in launch vehicle capabilitites such as the Thor Delta, the improvements in communications capability per pound of payload, the military emphasis on reliability (and security), and the fact that most designs are evolutionary from earlier spacecraft.

1.4.4 Cont'd.....

In the present study, the first cut trade-offs at 1.5 GHz lead to a prime power of 300-400 watts according to the power devices used, and an antenna about 7 feet high. At 1500 lbs, a simple stable dual - spin configuration appeared feasible on the Thor Delta. (In fact, in the study a 3-axis design at 1.5 GHz also was carried for some time beyond the first trade-offs.)

At 300 MHz, the required EIRP is just a few dB above that of the 1.5 GHz system, the increase being associated with higher path margins. However, to acheive the maximum gain for Canadian coverage would require an antenna aperture of about 66 foot diameter. This was clearly incompatible with a dual spin by virtue of stability complexity, and solar pressure effects. Reducing the antenna to a more appropriate size meant that the transponder output and, thus, prime power had to increase to compensate. The principal difficulty is that the power requirement rose so high as to to prohibit a dual spin system. One alternative was considered as a potential interest; the solar array was made as large as possible within the Thor-Delta shroud and the antenna dispersed within the array surface with appropriate provision for electrical despinning of the pattern. This option was examined to the point where it was apparent that the area of the elements of the antenna array reduced considerably the solar array area, and that the antenna and its feed system involved a significant weight penalty. It was felt that the narrow band telephony signals would permit electrical despin without encountering the original Intelsat III problems at video bandwidths.

1.4.4 Cont'd.....

Thus, in selection of an appropriate base configuration at 300 MHz, it was found necessary to go to a three axis configuration on the basis of prime power while in a weight regime where it is not normally considered optimal. With the antenna back near the center of gravity in a three axis configuration, the antenna can be increased in size, thus reducing the prime power requirement but requiring the solar array structure (and weight) to still be large. The solar array itself must be moved away from the body to aviod shadowing by the antenna structure. Thus, only a minimal solar array weight was regained although there was a more substantial weight reduction from the reduced battery requirements.

Thus the design at 300 MHz was basically a series of trade-offs to obtain payload weight and lifetime where most of the three axis configuration advantages were unuseable. For example, CTS operating at higher frequencies can take advantage of the potentially higher pointing accuracy of a three axis configuration, whereas in the 300 MHz design, the small aperture antenna will never have a beam so narrow as to benefit from it.

1.4.5 Space craft Design Trade-offs

The trade-offs in spacecraft design (as opposed to configuration) fall into two broad and overlapping classifications. Within a given subsystem performance specification, the trades usually deveop between potential suppliers with respect to cost, schedule, risk (program or technological), and sometimes additional features and weight. These trades are normally only available after detailed specifications of the subsystem are developed and usually only finalized at the point of program subcontracting. The

1.4.5 Cont'd......

spacecraft designer must attempt to keep these options open through allocation of adequate spacecraft resources of weight, power and volume, and realistic performance and reliability demands.

The second class of trade-offs involve two or more subsystems and effectively the distribution of overall performance across the subsystem requirements. In the present study, for a given EIRP, the antenna size and gain directly trades against transponder and prime powers. Similarly, the control system of a three axis stabilized spacecraft may be traded in weight and complexity against antenna pointing accuracy which, in turn, affects the transponder requirements.

The trade-offs, particularly as they pertain to communications, are discussed in detail in Volume 2 of this report.

2.0 SPACE SEGMENT SELECTED DESIGNS AND MISSION

This section of the report deals with the operational concept of the systems recommended at 1.5 GHz and 300 MHz and the top level performance requirements of the spacecraft designs developed in this study. Volume II details the spacecraft design to the subsystem performance level.

2.1 Space Segment - 1.5 GHz

The spacecraft design developed for a system operating at 1.5 GHz (and potentially adaptable to 2.5 GHz) lends itself to a five spacecraft program with four scheduled launches as shown in Figure 1.2 (c). Such a space segment would have a life to zero value of sixtenn years and a system reliability (two spacecraft in orbit at a time) well in excess of the 0~~1~~7 requirement. With such a program there are two spacecraft in orbit at any given time - one operational and one standby. The standby may be held "in slot" beside the operational satellite in order to have minimum changeover thime in the case of failure or may be operational in a second slot to provide peak or "second net" capability with, of course, an outage for changeover time in the event of failure. It may be noted that this strategy does not require the use of "out of slot" standby or drifting into position in order to save furel in as much as the fueling allocated is adequate for the full eight year lifetime. It may also be noted that there is a fairly high probability that the individual spacecraft will exceed the design eight year lifetime in that the fuel allocated for launch dispersions may be available and the design margin remaining after implementation is usually allocated to fuel, assuming the tanks are oversize. A high inclination spare drifting into position is not particularly attractive for this conservative system concept although it could be applied to further extend the lifetime to zero value

2.1 Cont'd.....

as shown in Figure 1.2(b).

Alternatively, the extra on-station capability could be given over to improving the reliability near the end of the system life, but the gain is marginal and more or less contrary to the conservatism of this system concept.

This spacecraft design also could be applied to a four spacecraft procurement in which case the system is only applicable to a zero value at twelve years. As discussed earlier, the reliability is numerically acceptable but is basec on extrapolation of data into a regime of potentially poor validity. It could be brought into a viable regime by accepting a possibility of major system outages early in the system lifetime (e.g. no standby for the first two or four years), or by using an adaptive strategy based on the launch dispersions, etc. of flight one. This type of less conservative program is most readily handled through contracting for the first, say, two launches plus on-ground standby and options for additional spacecraft. At greater risk, the on-ground standby, e.g. the refurnished prototype, could be scheduled for flight, with an option for a further spacecraft if an early failure occurred.

2.2 Spacecraft - 1.5 GHz

As discussed earlier, this study rapidly norrowed down to a dual spin stable spacecraft as most applicable to the performance requirement at 1.5 GHz. The crux of the trade-offs became the efficiency of the power stages of the transponder and, particularly, the devices to be used, solid state or TWT's. The prime power is

2.2 Cont'd........

adequate for TWT's but potentially introduces an eclipse capability problem. Solid state devices introduced a problem in the combining of large numbers of lower power devices. In regard to the latter, the combination did ease redundancy plans relieving the weight problem, and further is in an area where technological advances may be anticipated to further ease the problem.

The chief problem in the antenna was the necessity of going to a space erected scheme, although there is adequate fairing space to allow a rigid mast thus alleviating much of the problem.

Table 2.1 gives the basic spacecraft performance specifications while Figure 2.1 shows a general view of the spacecraft.

2.3 Space Segment - 300 MHz

The spacecraft design developed for a system operating at 300 MHz, requires a minimum procurement of five spacecraft with four scheduled launches as shown in Figure 1.2(c) in order to approach the twelve year system operational life. However, this assumes that the inclination dead band at 300 MHz is the same as that for the 1.5 GHz design. At the lower frequency the earth station beam width is such that for the same pointing loss a higher inclination is possible. Thus, at 300 MHz there should not be any problem in achieving the required lifetime even though there is no inclination correction fuel aboard. Additional attirude control fuel would, of course, be required for the extra year of operations. This area has not been fully developed because the basic spacecraft reliability is of concern, and

TABLE 2.1 - Spacecraft System Characteristics

Spin Stabilized Configuration	
Size	Height 55", Diameter 86"
Weight	1500 lbs (including 118.5 lbs margin)
Communications	1.5 GHz Transponder - Single Channel Multiple Access 100 duplex, 5 simplex. EIRP: 37.5 dBW G/T : -2.4 dB
Coverage	All of Canada
Stabilization	Dual Spin - Favourable Moment of Inertia Ratio
Attitude Control Sensor	Spinning Earth Sensor
North South Inclination Control	Yes
Power	Body Array with 2 Batteries giving full Eclipse Capability
Telemetry	PM/PCM
Command	PCM/FSK/AM
Lifetime	8 years
Reliability	.660

2.3 Cont'd.....

because the use of very wide station keeping slots is of some international concern. In regard to the latter, it is assumed that the proposed technique of wide inclination limits but narrow limits of longitude is much less objectionable. It is considered that at the time of implementation the technology and international agreements would permit a six-year lifetime for this general design.

2.4 Spacecraft - 300 MHz

The crux of the design problem at this frequency was the weight. The lack of weight margin prohibited major increases in reliability and, as discussed previously, lead to a design having no inclination control except for that associated with launch dispersions.

The antenna design is relatively straight forward and has a relatively small aperture for the required coverage. The trades of increasing antenna aperture in order to decrease the down-link power and prime array power were not effective in that the solar array outside the antenna shadow. Thus, the return for increase of antenna weight and complexity was in the fewer power devices in the transponder and a decrease in battery capacity.

Table 2.2 gives the spacecraft performance specifications while Figure 2.2 shows a general view of the spacecraft.

TABLE 2.2 - Spacecraft System Characteristics

Three Axis Stabilized Spacecraft	
Size	Height 46", Diameter 71" x 55"
Weight	1500 lbs (including 76.6 lbs margin)
Communications	300-400 MHz Transponder - Single Channel Multiple Access: 100 duplex, 5 simplex, EIRP: 41.2 dBW G/T : -11.0 dB
Coverage	All of Canada
Stabilization	Double Gimballed Reaction Wheel
Attitude Control Sensor	Static Earth Sensor
North South Inclination Control	None
Power	Flexible Solar Sails with 2 Batteries giving 14% Eclipse capability
Telemetry	PM/PCM
Command	PCM/FSK/AM
Lifetime	5 years
Reliability	.695

2.5 Ground Segment Parameters

A preliminary examination of the overall system showed that 44 dBW seemed to be the maximum available satellite EIRP (equivalent saturated power) and thus if the required traffic model was to be met, earth station figures of merit on the order of -2 dB/^{o}K were required. The selected values are shown on Table 2.3-1.

To achieve acceptable circuit performance, companding has been assumed, and thus the performance objective is subjectively equivalent to 10,000 pWp clear weather, and it is only by the use of companding that the system is feasible. The use of companding imposes constraints on the overall system design and a subtle constraint on the spacecraft transponder. The difficulty arises from the fading performance of a companded system, which resembles threshold performance of an FM demodulator, and as a result the uplink contribution to fading must be removed by an active EIRP control at each Earth Station. To avoid unstable interaction between the controllers, the satellite transponder must be operated at a point where the gain compression is less than a few tenths of a dB. If a beacon is used for reference (this is necessary at 300 MHz, there may be some problems in using the beacon, such as aging and the possibility of failure. While this is consistant with the present system (i.e. the system must be backed off from the IM considerations) it may not be the case if the system is operated with different modulation techniques or different noise allocations which allows the transponder to operate close to saturation.

TABLE 2.3

Earth Station Parameters

Service	1.5 GHz G/T	1.5 GHz Antenna Diameter	1.5 GHz Receiver	300MHz G/T	300MHz Antenna Diameter	300MHz Receiver
Civilian (Commercial quality)	-1dB	4.5'	Paramp	-17dB	5.5'	Transistor Amplifier
Military (Transportable)	-5dB	3.5'	Paramp	-19dB	5.5'	Transistor Amplifier
Program	+1dB	5.5'	Paramp	-16dB	6.5'	Transistor Amplifier

1. G/T at 10^{o} elevation angle
2. Parametric amplifier has 75^{o}K noise temperature
3. Transistor amplifier has 2dB noise figure
4. Ground reflection loss included in transportable stations.

Performance Objectives

Commercial Quality

(Exact from Recommendation 353-1, CCIR Documents of XIth Plenary Assembly, OSLO, 1966)

"A target performance objective for the commercial quality service to be provided to fixed civilian station in to meet the standards laid down by the CCIR".

The CCIR unanimously recommends:

1. that the noise power, at a point of zero relative level in any telephone channel in the basic hypothetical reference circuit as defined in Recommendation 352, should not exceed the provisional values given below:

1.1 10,000 pW psophometrically-weighted mean power in any hour;

1.2 10,000 pW psophometrically-weighted one-minuted mean power for more than 20% of any month;

1.3 50,000 pW psophometrically-weighted one-minute mean power for more than 0.3% of any month;

1.4 1,000,000 pW unweighted (with an integrating time of 5 ms), for more than 0.03% of any month;

2. that the following Notes should be regarded as part of the Recommendation:

NOTE 1 Noise in the multiplex equipment is excluded from the above.

NOTE 2 It is assumed, that noise surges and clicks from power supply systems and from switching apparatus (including switching from satellite to satellite), are reduced to negligible proportions and therefore will not be taken into account when calculating the noise power.

Note 3

In applying the basic hypothetical reference circuit and the allowable circuit noise to the design of satellite and earth station equipment for a given overall signal-to-noise performance, the system characteristics preferred by the C.C.I.R., as found in its Recommendations, should be used where appropriate; where more than one value is recommended, the designer should indicate the value chosen; in the absence of preferred values, the designer should indicate the assumptions used.

Note 4

Not applicable to single carrier per channel systems

Note 5

It is not yet possible to make firm recommendations regarding requirements to be met, if VF telegraphy and data transmission are required over telephone channels in a communications-satellite system.

Note 6

The system should be designed to operate under the noise conditions specified, including noise due to interference within the limits defined in Recommendation 356-1 for line-of-sight radio-relay systems sharing the same frequency bands and noise during periods of adverse propagation conditions such as those resulting from atmospheric absorption and increased noise temperature due to rain. In certain cases, however, additional noise may cause the limits fixed in the general objectives to be slightly exceeded. This should not cause serious concern, provided that the provisions of CCITT Recommendation G.222, para. 6, are met.

Military Quality (Appendix I)

The question of what transmission standards will be applied to the voice circuits of the system seems unresolvable if present government standards are to be the basis. MIL-STD-188B treats tactical communications systems at some length but does not include standards that are directly applicable to the system under consideration. One can, however, extract values for various parameters from MIL-STD-188B that might be applicable. Such an extraction is attemped below for the following pertinent transmission parameters:

1. Channel Passband

300-3500 Hz between 3-dB points.

2. Total Harmonic Distortion

-30dBmO for any test signal in the band at a power level of 0 dBmO. This standard seems to be applied to what are termed "high performance systems." "Low performance systems" are allowed 10% harmonic distortion (i.e., -20dBmO).

3. Channel Overload Characteristics

Overload is specified in terms of dB compression for 100-Hz signals inserted at a point of zero relative level as follows:

Input Power	Compression
-10 dBm	None
0 dBm	Less than 0.4 dB
15 dBm	At least 7 dB

4. Channel Gain Variation

- $\pm$ 2dB

5. Channel Noise

The tactical reference circuit of the Standard comprises seven trunks in tandem, each having noise of -46 dBmOp. The noise for these seven tandem trunks is -38 dBmOp. The Standard considers the use of radio subscriber loops each having noise of -32 dBmOp, as well as wire subscriber loops having noise of -57dBmOp. Thus, the noise for a circuit comprising seven trunks in tandem with radio loops at each end would be as high as -28dBmOp. This would seem a reasonable maximum channel noise level to adopt for his system if it is to be used for direct subscriber-to-subscriber service, or if it is to be used with relatively quiet and zero-loss subscriber loops. If it is to be used with radio or other subscriber loops as noisy as those of the Standard, its channel noise would have to be reduced to -38dBmOp. The Standard considers that, for the average talker the subscriber set has an output of -6vu and the output for 1% of the talkers is below -15vu. If compandors or other means of volume adjustment are used to raise the level of weak telkers, as appropriate increase in channel noise should be allowable.

6. Time Variation of Performance

The Standard allows trunks carried in tropospheric-scatter systems to have noise performance worse than the specified value for 5% of the worst month. Perhaps a similar approach can be applied to this system.

Program

53dB S/N RMS 100 Hz - 8KHz, CCITT emphasis and CBC weighting.

No specific percentages allocated.

3.0 Costs and Conclusions

It is concluded that the maximum traffic model of the baseline requirement can be met at 1.5 GHz, with no major deviations from the other requirements of baseline definition.

With an assigned frequency of 1500MHz, a dual spin configuration spacecraft can provide the required communications capability based on a Thor Delta launch. Some spare weight margin exists which could be used to increase channel capacity or otherwise improve system performance. For example, with a transistor amplifier at 1.5 GHz or at 2.5 GHz some weight margin remains on the spin stabilized spacecraft while carrying the specified maximum channel capacity. It would be uneconomic to launch a spacecraft at less than the maximum launch weight and during the design phase this weight margin would be utilized and accomplish some system objective. There are a number of possible uses for which this weight margin may be utilized, some of which are listed below.

1. Increase channel capacity.
2. Increase the expected life of the satellite by improving reliability and the increasing fuel load.
3. Reduce the cost of the ground station by some means that absorbs the extra capacity of the satellite.

For the purposes of this study, the weight margin is being kept in reserve in case predicted efficiency figures are not realized in the development phase. It is concluded that a five spacecraft, eight year spacecraft lifetime mission would permit a conservative overall system going to zero value at 16 years. At 12 years for zero value, a four spacecraft program is conceivable but somewhat risky.

3.0 Cont'd

On the basis of a somewhat less detailed analysis it appears that a similar program would be feasible at 2500 MHz based on the technology now available in the laboratory and likely to be available in the time frame of interest.

At the possible 300 MHz assignment, the traffic model makes it virtually mandatory that a three axis stabilized design be utilized. In such a design there is a very significant weight problem which, in the present study, has necessitated a reduced eclipse capability and a reduction of lifetime. A five spacecraft procurement at this time is short of providing a system end of service at 12 years. It is considered that the probable launch capabilities in the 1975-1978 or those at the time of the second launch series (1980 - 83), coupled with advances of technology would permit a five spacecraft system going to zero value at twelve years.

It is concluded that the maximum traffic model can be met, but only at the expense of implementation margin. The system has been designed with adequate fading margins, however the maximum achievable satellite EIRP is equal to that required for the maximum traffic model. Traffic capacity is the only area amenable to trade offs, as the antenna size (and hence the (G/T) of the ground segment) is about the largest acceptable.

3.1 Costs

The costs are discussed in the body of the report and are summarized below:

Space Segment 1.5 GHz

	costs in millions
Non Recurring	12.7
Prototype Satellite	7.25
Flight Spacecraft	5.60
Cost to refurbish to prototype to Flight Status	1.0
Program Management & G.S.E.	7.0

Earth Segment

	cost in thousands
Tracking Network Control Station	1,500
Non tracking Network Control Station	850
Telephony Station, 100 of	33.5
Telephony Station, 500 of	21.1
Telephony Station, 1000 of	18.9

Space Segment 300 MHz

	cost in millions
Non Recurring	14.90
Prototype Satellite	8.25
Flight Satellite	6.20
Cost to Refurbish the prototype to Flight Status	1.00
Program Management and G.S.E.	7.00

3.1 Cont'd.....

Earth Segment Costs

	costs in thousands
Tracking Network Control Station	1.5
Non Tracking Network Control Station	750.
Telephony Station, 100 of	26.3
Telephony Station, 500 of	15.3
Telephony Station, 1000 of	13.4

The total system implementation costs are of course directly related to the scale of the implemented system and in particular to the required system availability, as it relates to the number of required launches and spacecraft to be procurred.

Similarily for the earth segment, although the production cost of a large number of stations (greater than 200) can be estimated with some confidence, the estimate becomes less firm when small quatities are to be procured. It is felt that a pilot system, at least for civilian service, would be acceptable for the first few years of operation and this would consist of single satellite in orbit and a small number of stations (about 50). The cost of each station becomes very high if all development costs are prorated on quantity 50, and the tendency could be to design a station which would not be suitable for larger production runs. The result would be that a fully implemented system would cost substantially more if implemented in a piecemeal fashion. The cost is thus sensitive to the type of operating entity and the implementation philosophy adopted.

3.1 Cont'd........

For large quantities, the earth station cost is estimated as $21K FOB factory.(1.5 GHz system) This includes all development costs prorated over a production run of approx. 500 and over a 4-year period. The installation costs are highly variable and are traditionally underestimated. The availability of these stations should be such that no outage will ever occur due to factors such as high wind, and in view of their cost, they should be designed to survive the 20 year return period wind. A wind of 120 mph is generally considered an adequate survival design target, and such a wind exerts a thrust of 1,500 pounds on a 5 foot dish. The fixed commercial earth station will thus not be a tripod mounting scheme and some foundations will be required. As an indication, assume 2 cubic yards of concrete, and the price poured per yard in Resolute is $300. From this and other considerations, an installed cost of $25K seems reasonable.

3.2 Areas of Sensitivity and Study Limitations

For the 1.5 GHz system, the greatest area of sensitivity is the design of the transistor transponder. This is also the greatest limitation of the study, in that the intermodulation behaviour of the transponder is imperfectly known. This area of technology however is one area which is advancing rapidly and it appears certain that devices which can increase the reliability and perhaps efficiency of the transponder will be available in the envisaged time frame. Recent acts by the WARC indicate that 2.5 GHz is a more likely frequency for this type of service and if we consider only 2.5 GHz then the major limitation is removed, as a TWT type transponder can be used, and its behaviour is well known.

The comments about a transistor amplifier also apply to a 300 MHz system and additionally there is another area of sensitivity to the weight of a three-axis spacecraft. The 300 MHz satellite is feasible (though with very little margin) based on what is known about the Communications Technology Satellite, now in a study phase. Changes in CTS as it progresses to a flight spacecraft may have severe implications on the design of a 300 MHz satellite system.

RCA