INTEGRATED ORBITAL SERVICING AND PAYLOADS STUDY

Final Report

Volume II

Technical and Cost Analysis

Contract NAS 8-30849
Data Procurement Document No. 463
Data Document No. MA-04

Submitted to

Marshall Space Flight Center
Alabama 35812

September 1975

by

Communications Satellite Corporation
COMSAT Laboratories
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FOREWORD

This study was performed under Contract NAS8-30849 for the George C. Marshall Space Flight Center of the National Aeronautics and Space Administration under the direction of James R. Turner, the Contracting Officer's Representative. The final report consists of two volumes:

Volume I - Executive Summary
Volume II - Technical and Cost Analysis

Additional documentation in the form of working papers and drawings have been provided to Mr. Turner. Inquiries regarding this material may be addressed to the following individuals:

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Supporting information was prepared under a parallel study, Integrated Orbital Servicing Study for Low-Cost Payload Programs, Contract NAS8-30820. Inquiries regarding this material may be addressed to Mr. Turner or to:

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I. INTRODUCTION

The main conclusions and results of this study are presented in Volume 1, Executive Summary. Included in this second volume are some of the details and background used in the study. The organization of this volume is based on the tasks given in the Final Study Plan; that is, each of the major sections in this volume corresponds to a separate task.

This study was done in parallel with a study[1] performed by Martin Marietta. Close coordination was maintained between the two studies by phone, mail, and monthly coordination meetings. The results of the two studies complement each other, and this report should not be studied without the corresponding report written by Martin Marietta.

The opinions and conclusions in this report were generated in the course of this study. They should not be construed as official COMSAT policy. COMSAT has made no commitments about on-orbit servicing.

II. REVIEW OF PREVIOUS MODELS AND STUDIES

NASA MISSION MODEL

The Shuttle Systems Payload Data (SSPD),[2] prepared in 1974, has been used throughout most of the studies. The spacecraft descriptions contained therein represent a large collection of data and have been quite valuable to the study, particularly in the evaluation of servicing. Within the limitations of the time available, some evaluation of the adequacy of the model for the purposes of this study was made. The broad picture given by the model is valid and useful, but some of the details are outdated or inadequate. Where necessary, minor changes have been made. The next few paragraphs summarize some comments on the details of international communications satellites, the use of a rigid flight schedule in evaluating servicing, and the predicted number of operating satellites.

The section on international communications satellites has been read critically, since COMSAT is the manager of the INTELSAT system of satellites. According to the references, information on the international communications satellite has been obtained from Lockheed and from an MCI application, As a potential bidder, Lockheed has done extensive work on a future INTELSAT satellite, yet it is hardly a prime source of information. The number of operating satellites is listed as greater than or equal to one; actually, the required number today is four. It is likely to be larger in the 1980's, and certainly will never be less than three. INTELSAT satellites should be positioned in orbit over the three ocean areas, while the U.S. domestic satellites will be positioned over North America. Ten deployed antennas are listed. While there may be as many as 10 antennas, the number that require deployment will be minimized, and probably will be no more than five. All passive thermal control is listed; actually, heat pipes or louvers may be required.

For the propulsion system, both hydrazine and cesium ions are listed. Although electric propulsion is being seriously considered, it is by no means certain. (Servicing is more attractive for an all hydrazine system since large amounts of fuel are required for N-S stationkeeping.) A 4400-W electric power system is listed. This value is too high, and a value between 1000 and 2000 W is far more likely. In the 12/13-GHz band, twenty-four 36-MHz channels are listed; although this may be a good guess, no one really knows what will be done in the 1979 to 1991 era. While a single TV link is listed for each 36-MHz channel, by that time it is likely that two TV signals can be put through one channel.

One difficult question to resolve in evaluating servicing is whether the number of flights will be held constant or allowed to vary. If it is held constant, then at some particular time the satellite will either be replaced (expendable mode) or fixed (servicing mode). For most of the analysis in these studies, the number of flights has been held constant. Although this is not necessarily the best way of evaluating servicing, it was the only way to perform the evaluation with the time and data available. The problem in the study of allowing flexibility in the number of flights is that it requires going back in each program to the basic philosophy that sets the required number of flights, which is quite a task.

In some of the studies the advantages of more frequent servicing have been investigated, particularly those associated with exchanging modules while the satellite is still able to perform its mission. At the moment this increases the cost (relative to the cost of not performing a service mission at that time), but there are some benefits in terms of reducing the cost per servicing and improving spacecraft reliability.

NUMBER OF COMMUNICATIONS SATELLITES MAINTAINED IN ORBIT

A key parameter in the study of on-orbit servicing is the number of satellites in orbit. For communications satellites, an independent estimate has been made and compared with the SSPD data. The total is about the same, but details differ; the comparison is given in Vol. I, Executive Summary. The details and reasons are given in the following pages.

The following is an estimate of the average number of communications satellites (operational or in-orbit spare) maintained in geostationary orbit from 1982 to 1991. The Russian satellites and the U.S. military satellites are excluded, but NATO and SKYNET satellites are included. The basic categories used here are from the SSPD document. A total of 45 satellites has been estimated as follows:

International Communications Satellites	9
Domestic Satellites	10
Disaster Warning	2
Traffic Management	
Civil Transoceanic Aviation	3
Maritime Carriers	4
Foreign Communications	12
DOMSAT "C" - TDRSS	3
Communications R&D	2

The above includes most of the non-DOD satellites in geostationary orbit. In addition to the communications satellites, the SSPD includes half a dozen earth observation geostationary satellites.

International Communications Satellites

The present configuration for active INTELSAT IV international communications satellites is 2, 1, 1; that is, two over the Atlantic, one over the Pacific, and one over the Indian Ocean basin. The policy is to have one spare in each ocean; the total now required is seven satellites.

Most of the present studies are taken only to 1985. These require a series of INTELSAT IV-As to be launched starting in late 1975, followed by INTELSAT Vs in the late 1970s. The minimum configuration, a 2, 1, 1 configuration, would require larger capacity satellites to handle the expected growth in traffic. The maximum configuration is 5, 2, 3. This configuration would require the least growth in individual satellite capacity, but more complexity in traffic assignment and earth station capabilities. The median approach, and the one adopted for this estimate, is a 3, 1, 2 configuration, which has been widely studied for the period up to 1985. For this configuration with three additional in-orbit spares, the estimated number of satellites is 9.

Domestic Satellites

This category includes all communications between earth stations in the United States (a combination of the DOMSAT "A" and DOMSAT "B" categories in the SSPD document). The DOMSAT "A", based on a filing by the American Satellite Corporation to the FCC (1/23/73), was a small 576-lb (261-kg) satellite to be launched in the 1979 to 1983 period. The DOMSAT "B", based on a filing by MCI and Lockheed, was a large 3200-lb (1472-kg) satellite to be launched in the 1984 to 1991 period. Both programs have large unknowns, and it is difficult to distinguish between an "A" and a "B".

There are three domestic satellite programs that provide some information on future figures. Western Union has filed for three locations, and already has two WESTARs in operation. RCA has filed for four locations, and is presently building the satellites that will soon be launched by a Delta 3914 rocket. The COMSTAR satellites, soon to be launched by COMSAT for use by AT&T and GT&E, represent the set of heavier communications satellites. COMSAT has filed for three locations, and growth in this area is to be expected. While there are numerous other plans for future programs, some individuals doubt that there will be enough traffic growth to support them all. An estimate of 10 comestic satellites appears reasonable.

Disaster Warning Satellites

The mission objective is to provide NOAA with an independent mass communications system for warning the public of impending disasters and issuing bulletins for corrective action to protect lives and property. The SSPD estimate of two satellites in geostationary orbit is left unchanged. Recent information indicates that the satellite may be dropped and a chain of terrestrial VHF radio stations (at 162.55 MHz) may be used for the warning network.

Traffic Management Satellites

Plans to use a satellite for providing tracking, control, and weather information to civil transoceanic aviation are presently being made by the European Space Research Organization (ESRO) and by COMSAT. Initially, this service would cover the Atlantic and Pacific Oceans, which would require a minimum of two satellites. A spare satellite, or an Indian Ocean satellite, would increase the required number to three. If the program is successful, the number may easily grow to four by the end of the next decade. For the present, an estimate of three is taken for AEROSAT.

There are two programs for satellite communications to maritime carriers. COMSAT will soon be launching a MARISAT that will be used initially by both the U.S. Navy and maritime carriers. Service is planned for the Atlantic and Pacific, with two satellites initially. This number will probably grow to at least three, with a spare or with one over the Indian Ocean. In addition, Europe is developing its own satellite through the MAROTS program, which will have at least one satellite. Whether the American and European programs later combine or continue separately, an estimate of four satellites for maritime carrier communications appears reasonable.

Foreign Communications

While it is difficult to predict the future of foreign communications satellites, there is no doubt that this category will continue to grow. Some countries are planning to have their own satellites; these are summarized in Table 1. Under the TELESAT program, Canada has two ANIKS which are quite successful. Japan will soon have at least two more, one for broadcast and one for communications. Germany plans to have a TV broadcast satellite, and Indonesia will buy two satellites of the ANIK type. In addition, foreign military satellites include the NATO and SKYNET programs.

Table 1. Purchase of Communications Satellites (launched or under contract)

Country	Program	Number
Canada	TELESAT	2+
Japan		2
Indonesia	ANIK +	2
NATO	NATO III	2
UK	SKYNET	2

Other countries are now renting transponders from INTELSAT or have filed applications for rental (see Table 2). Several of these countries will have their own satellites in 10 years or the number of INTELSAT satellites will grow to handle this service. Brazil already has four earth stations and plans for more. Other countries, not listed in Table 2, such as Australia, Iran, Pakistan, have some interest in satellites. While it would be technically and economically feasible to provide this service with a small number of satellites, for reasons of politics and prestige many of these countries may choose to have their own satellites. An estimate of 12 satellites is made, which includes both those countries desiring to buy and those desiring to rent.

Table 2. Rental of Communications Satellites
(renting from INTELSAT or filed application)

Country	Cost per Transponder* (M$)	No. of Transponders	Annual Cost (M$)
Brazil	3.24	1	3.24
Algeria	1	2	2
Zaire	1	1	1
Spain/Mexico	1	1	1
U.S. (COMSAT)	1	1	1
Philippines	1	1	1
Malaysia	1	1	1
Norway	1	1/2	0.5

*3.24 M$/yr is for "non-interruptable service"; 1 M$/yr uses spare capacity for "interruptable service."

DOMSAT "C" Satellites

This category was based on the tracking data relay satellite system (TDRSS), which assumes two operational satellites and one on-orbit spare. NASA is continuing with this project, although plans for funding will require congressional approval. The program estimate of 3 satellites is retained here.

Communications R&D Satellites

At present there are no firm plans for a U.S. advanced technology communications satellite. Nevertheless, Canada has the Canadian technology satellite program (CTS), France and Germany have launched SYMPHONIE, Europe has another advanced satellite (OTS), and Italy is moving ahead with the SIRIO program. It appears reasonable that similar programs will continue in the next decade, and with the possibility of regenerating a U.S. program, an estimate of 2 satellites appears reasonable.

DEMONSTRATIONS OF MODULE EXCHANGE

A few observations can be made concerning the EOS demonstration[3] at Goddard and the Bell Aerospace demonstration[4] at Buffalo last year. In both cases the work was well done in view of the budget limitations.

a. Hardware demonstrations are useful for evaluating on-orbit servicing.
b. The automation by Bell Aerospace is preferable to the manual operation of EOS.
c. The benefits obtained from TV, as used, were not worth the cost in dollars, complexity, or bandwidth. (Bell Aerospace did not use TV for module exchange but for rendezvous.)
d. The reliability of the module exchange was poor in both cases. While this was understandable in terms of budget, demonstration of reliability will be necessary before servicing is really sold to many projects.
e. Both demonstrations used horizontal withdrawal of modules. For testing on earth, vertical withdrawal of modules might be better. Requirements of testing may influence the choice of module exchanger.

SERVICING THE DSCS-II WITH THE STS

A detailed study of the servicing of a communications satellite in geostationary orbit was done by TRW for the Air Force. This study was of special interest because it dealt with the advantages of servicing for a specific ongoing program, and because the mission of this program was communications satellites, in line with COMSAT's expertise. Several reports were received, including the final report, and COMSAT attended both a presentation by Abe Fiul on the East Coast and the final presentation at SAMSO on March 11. After the final presentation, the attendees were invited to stay for informal discussion with TRW participants. Both Martin Marietta and COMSAT took maximum advantage of the invitation; points were clarified on various parts of the study.

Costs and availabilities had been calculated in this study. The cost comparison was influenced greatly by the treatment of availability. If the effect on availability was ignored, costs of the serviceable mode were comparable to or higher than those of the expendable mode. On the other hand, if systems of equivalent availability were compared, especially for high values of availability, servicing became more attractive in the cost comparison.

A Monte Carlo simulation had been done using a reliability model for the spacecraft and specified criteria indicating when the spacecraft was considered operable. The computer simulation provided information on the required frequency of servicing; thus, it was useful for calculating both the costs of operating the system and the system performance in terms of availability.

The study was impressive in terms of both performance and depth, yet it apparently resulted in no definitive conclusion. It certainly did not state that servicing was not cost effective; rather it tended to favor servicing. On the other hand, it did not prove conclusively to those involved in military communications that on-orbit servicing should be implemented. This is indicative of many space programs: although on-orbit servicing is cost effective, there is no urgent need for implementation, and its use may be delayed for many years.

SUGGESTED METHODS OF SERVICING

The suggested methods of utilizing the STS for servicing automated spacecraft have been reviewed. Most studies consider fairly standardized methods. Spacecraft are designed to be serviceable by modularized design. In low orbit servicing is performed by the orbiter; in high orbit it is performed by the tug. Costs of these methods have been estimated, and the totals compared. The studies conclude that these methods are technically feasible.

In terms of cost effectiveness, the variety of methods studied is limited. While a low-cost method can be chosen from those studied, the most cost-effective method may not be found. The following are some alternative methods of servicing; more discussion is given in other parts of this report:

a. free flying servicer for multiple servicing,
b. servicing of operating satellites that have not yet failed,
c. reduction of initial satellite redundancy,
d. low-cost refurbishing without complete subsystem inspection and testing, and
e. use of servicing to improve system performance (e.g., availability).

III. APPLICATION OF SERVICING TO COMMUNICATIONS SATELLITES

Due to COMSAT's experience and the unique position of communications satellites in the NASA mission model, a special analysis of the servicing of these satellites was made. Particular emphasis was given to the benefits of servicing for international communications satellites because this system has been in existence for a number of years. Requirements for other communications satellites are similar, and conclusions are applicable to the approximately 40 communications satellites expected to be in geostationary orbit in the 1980s. In addition, a few other payloads, especially earth observations satellites at geostationary orbit, have similar requirements.

Much of the analysis of communications satellite servicing has been given in the Executive Summary, Volume 1 of this report. That summary also described the design of a serviceable communications satellite. Some of the analysis that led to that design is in this section. Since waveguide connectors are essential to a modularized serviceable communications satellite, an analysis was made to ensure that such connectors are feasible. The design presented here is not intended to be the only possible design, nor necessarily the best, but simply one possible way of building waveguide connectors into a module.

The availability of a communications satellite is an important measure of the satellite system performance and the benefits of servicing. The last sections of this chapter include the requirements for satellite availability, the present methods used to achieve such an availability, and some proposed new methods for achieving this availability. While servicing is one method of achieving a desired performance, it is not the only tool available. Other ways of achieving the desired results must be compared with servicing for an overall evaluation of on-orbit servicing.

THE ON-ORBIT SERVICER/SATELLITE INTERFACE

The means by which the on-orbit servicer can dock with a satellite and their effect on the satellite design are described (Figure 1). The advantages and disadvantages of the docking face location are also reviewed. Table 3 and Figures 2-12 review the impact of the location of the docking face. There does not seem to be an easy way to dock, exchange modules, and undock from a spinning satellite (see Figures 10-12).

An attempt has also been made to indicate the impact of the docking location upon the satellite layout. In most cases the structure would be quite different from today's designs. It seems that docking in the region of the center of mass is preferable in terms of attitude control; however, this requires balanced fuel consumption to eliminate imbalances and causes loss of pointing accuracy during servicing. If the center of mass argument is abandoned, docking can be done on corners or other places, thus giving access to the north and south faces (Figure 7).

A possible new satellite design is the split satellite approach (Figures 8 and 9). Once on-station, the satellite divides into north and south boxes. Each has its own north and south viewing faces for thermal radiators, thus doubling this valuable area. The servicer docks between the halves and services either half or both halves, as required. In the center post version (Figure 9) the docking device is on a threaded shaft. The device may be rotated through 180°, thus permitting the servicer to approach in the b, c, or d plane. It is also possible to dock in one plane (e.g., c) and undock in another (e.g., d).

The energy requirements of the various approach planes have not been analyzed. Tug retrieval information may be found in NASA/MSFC 68M00039-3. Further thought reveals that the east, west, earth, and anti-earth face dockings may use a common approach plane if the satellite is rotated about the pitch axis (also called orbit normal or N/S line). This pitch rotation would also permit the use

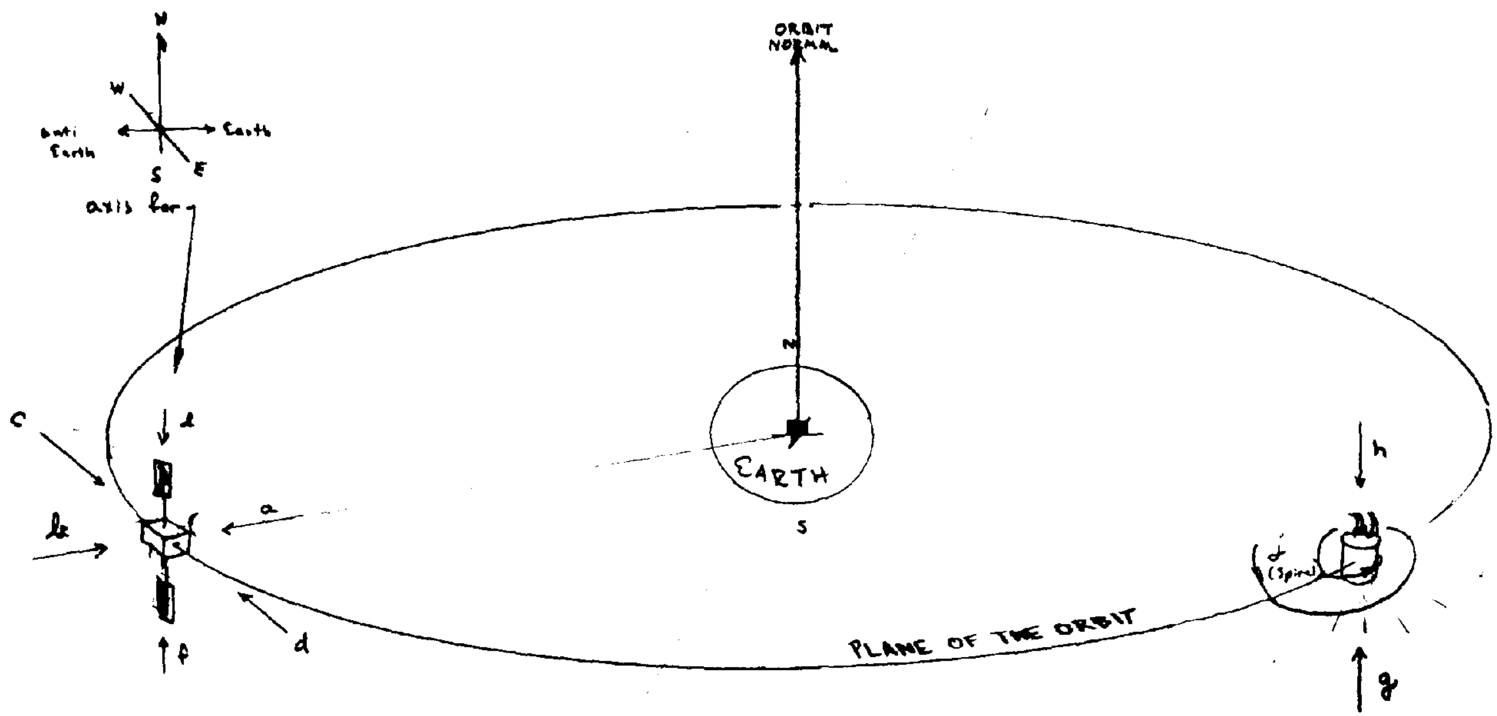


Figure 1. Planes of Servicer Approach

Table 3. Docking Face Location

Docking Face	Approach Plane*	Faces	Existing Equipment or Requirements on the Face	New Limitations Imposed by Docking	Advantages	Disadvantages	Conclusion	See Fig. No.
				BODY-STABILIZED SATELLITES				
Earth Viewing	a	1	Antennas, Sensors	RF/IR blockage, etc.	--	Significant	Avoid	--
Anti-Earth	b	1	Primarily thermal	May lose center of spacecraft as far as mounting equipment	Surface is generally inactive (previously used for AKM)	Docking may be in region of solar array thru-shaft (if used). Only one layer of modules.	Acceptable	2
Anti-Earth (one end or one edge)	b	1	(same)	Long length of servicer arm	(same)	Docking may be in region of RCS	Acceptable	3
Anti-Earth (Multiple Corners)	b	>1	(same)		Shorter servicer arm; greater access	Weight of added docking; more docking maneuvers	Acceptable	4
East and West	c and d	2	(same)	Satellite becomes long along the local vertical axis. Stowing the antennas in the shroud may be a problem.	Surface is generally inactive; modules may be two deep	Two sides for easier access (E and W)	Acceptable	5
East or West (not both)	c or d	1	(same)	Satellite may be more planar (flat square)	Surface is generally inactive	Access to only one face is allowed. Limited N/S area for radiators	Less Acceptable	6
North and/or South	e and/ or f	1-2	Solar Panels and radiators	Solar panels must be offset, thus creating solar torque problems. Dumping heat is a problem.	---	Significant	Avoid	7
Split Satellite Anti-Earth	b	1	Thermal	Scissors jack needed to separate satellite halves. Some radiators see one another.	Doubles the N-S radiator area. Doubles the module area. Separates the antenna farm (better for phased arrays and interferometers)	Docking ring is "soft." Added complexity and structure	?	8
Split Satellite Center post	b	1	(same)	Telescoping rods needed to separate satellite halves. Center post is threaded in docking area. Some radiators see one another.	(same) Servicer can pivot around post (about 180°) and thus have greater access means.	(same)	?	9
				SPIN-STABILIZED SATELLITES				
Anti-Antenna end	g	1	(same)	Need to transfer spin from satellite body to its antenna farm which may fly apart.	Module in a ring	Docking with a rotating body and momentum transfer. Cannot be used with dual spin satellites.	Poor	10
Antenna end	h	1	Lots of antenna (and thermal)	Antennas must be offset and counter-balanced	Service the despun part of a dual-spin satellite.	Access. Cannot service spun equipment.	Poorer	11
Side(s) of drum (spiral)	j		Solar Array	Equipment must be clustered near access doors	---	(same) Difficult approach; unbalanced when docked. How to undock?	Poorest	12

*See Figure 1

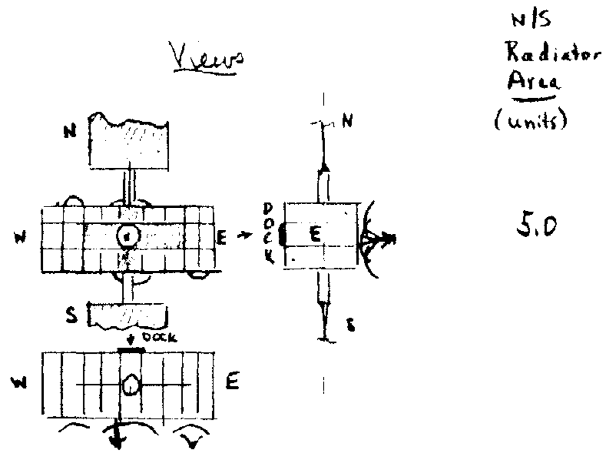


Figure 2. Docking Face
Anti-Earth (Center)

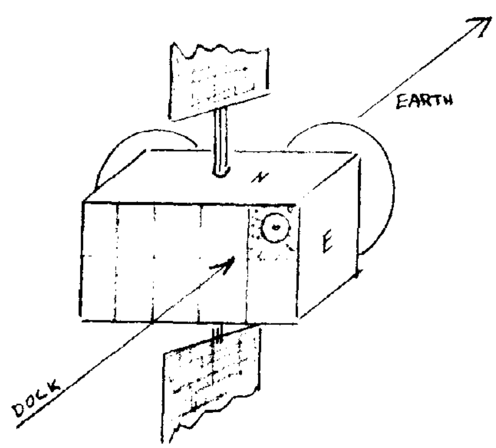


Figure 3. Docking Face, Anti-Earth
(One End or One Edge)

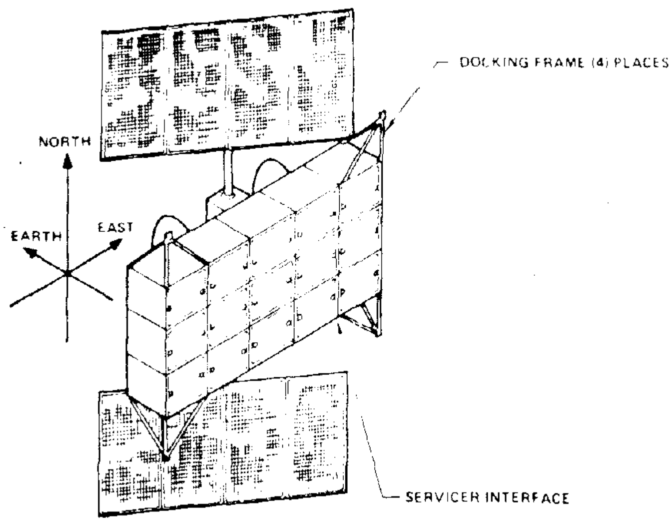


Figure 4. Docking Face, Anti-Earth
(Multiple Corners)*

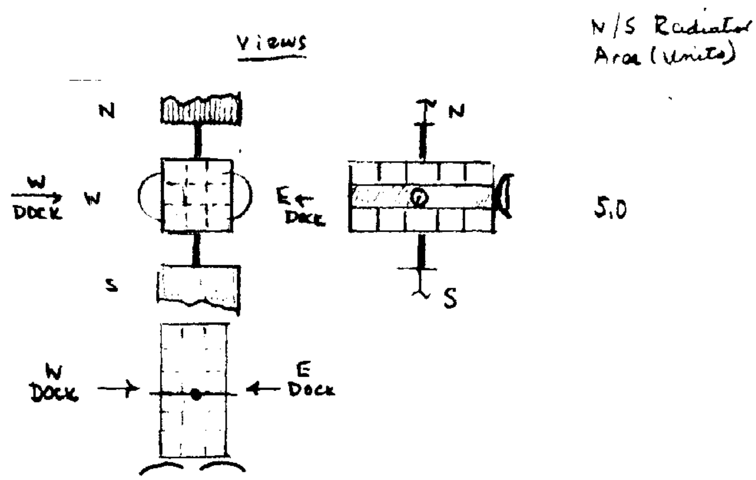


Figure 5. Docking Face
East and West

*Reference 9

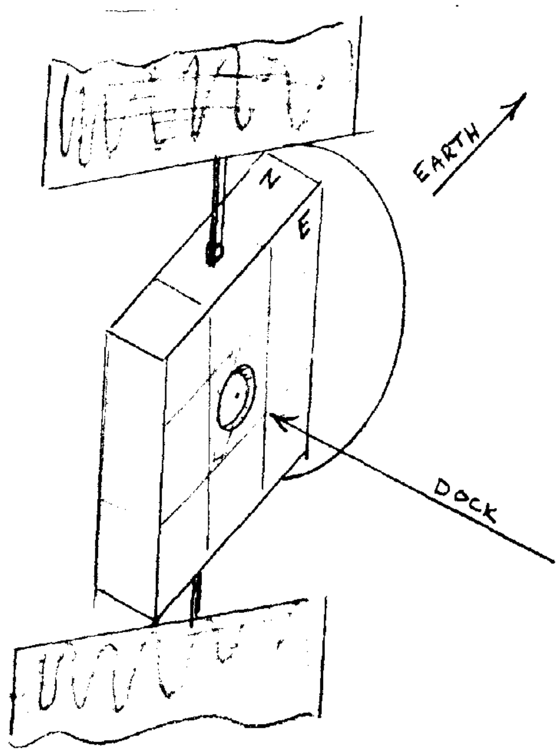
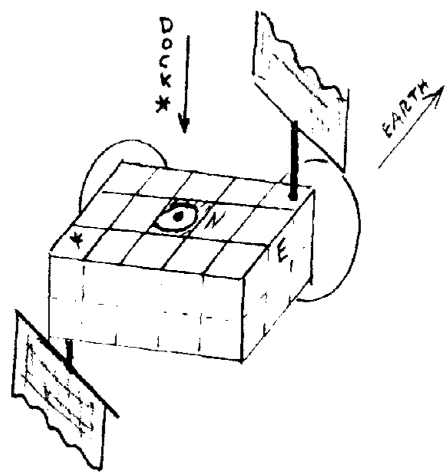


Figure 6. Docking Face,
East <u>or</u> West



*Docking device could be relocated to the corner opposite to solar array drive.

Figure 7. Docking Face,
North and/or South

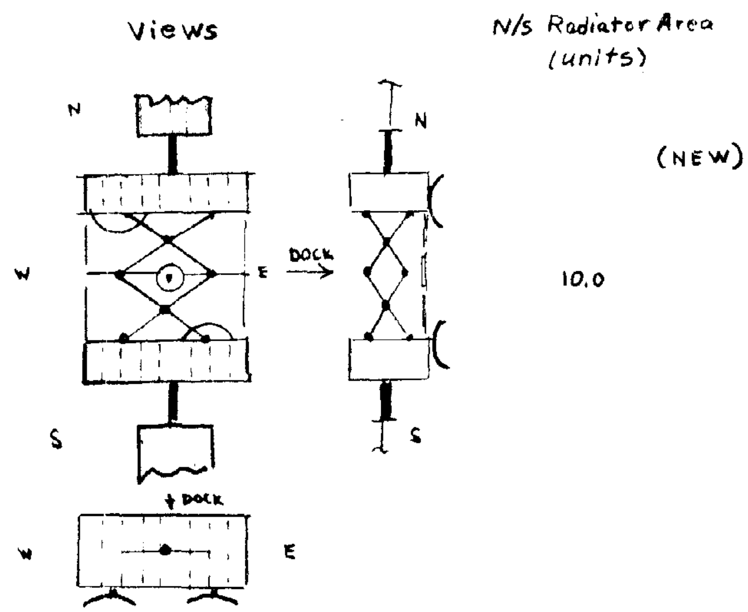


Figure 8. Split Satellite
Anti-Earth

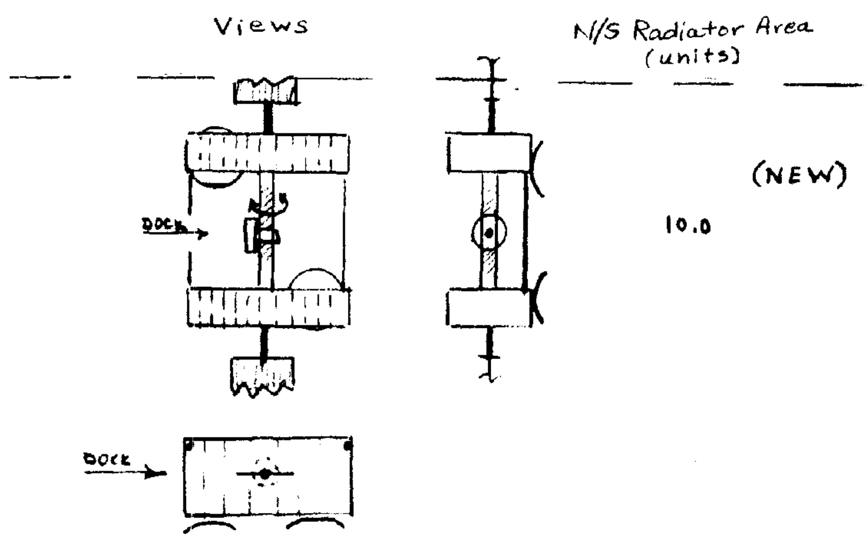


Figure 9. Split Satellite
Center Post

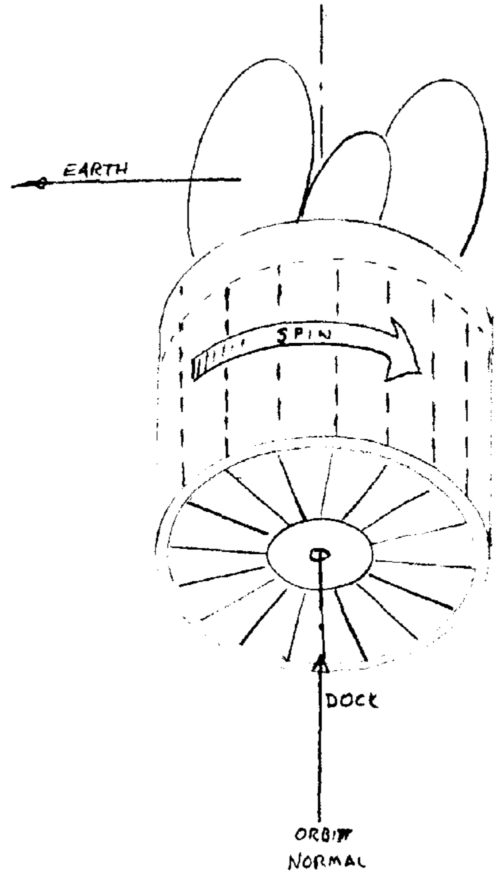


Figure 10. Spin-Stabilized Satellites
Docking Face, Anti-Antenna End

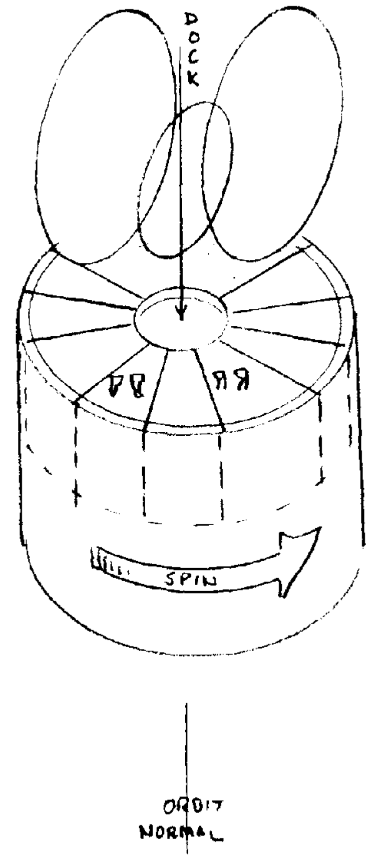
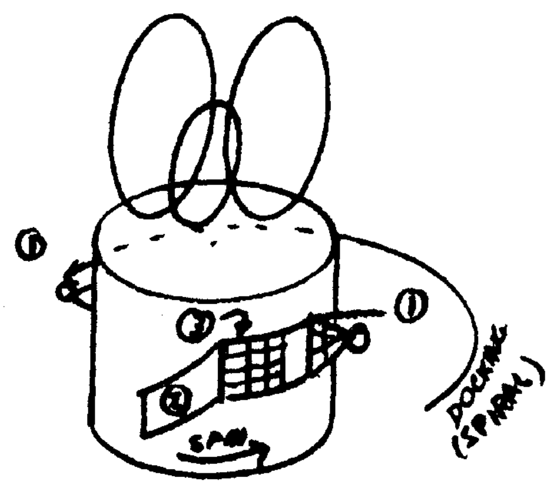


Figure 11. Spin-Stabilized Satellites
Docking Face, Antenna End



(1) Docking Devices (2)

(2) Doors (2)

(3) Modules

Figure 12. Spin-Stabilized Satellites
Docking Face, Side(s) of Drum (Spiral)

of different approach and departure directions. For a preliminary configuration, docking at the center of the anti-earth side was chosen (Figure 2).

CONFIGURATION

A body-stabilized communications satellite has one side (with antennas) always facing the earth; solar panels are contained on the north and south sides. Most modules should have areas on the north or south faces for thermal considerations. The prime candidates for docking faces are the east, anti-earth, and west faces.

A preliminary configuration for a serviceable communications satellite is shown in Figure 13. The docking would be done on the anti-earth side, and the modules would be extracted in that direction. All the modules would have an area on the north or south faces which would be used for thermal control. The main thermal design considerations are as follows:

a. north/south radiators for each module, 20 to 30 W/ft using second surface mirrors;
b. east/west surface insulation integral with the spacecraft structure;
c. earth/anti-earth insulation integral with the module;
d. module removal from the anti-earth side;
e. heat transfer preferably through some means other than thermal contact conductance;
f. radiation coupling from high- to low-power modules;
g. heat pipes permitted within a module, but probably not between modules;
h. electric heaters to maintain spacecraft temperature between module failure and servicer arrival; and

i. favorable sun angle maintained by the solar array drive during servicing, or by the servicer if the array drive is being serviced.

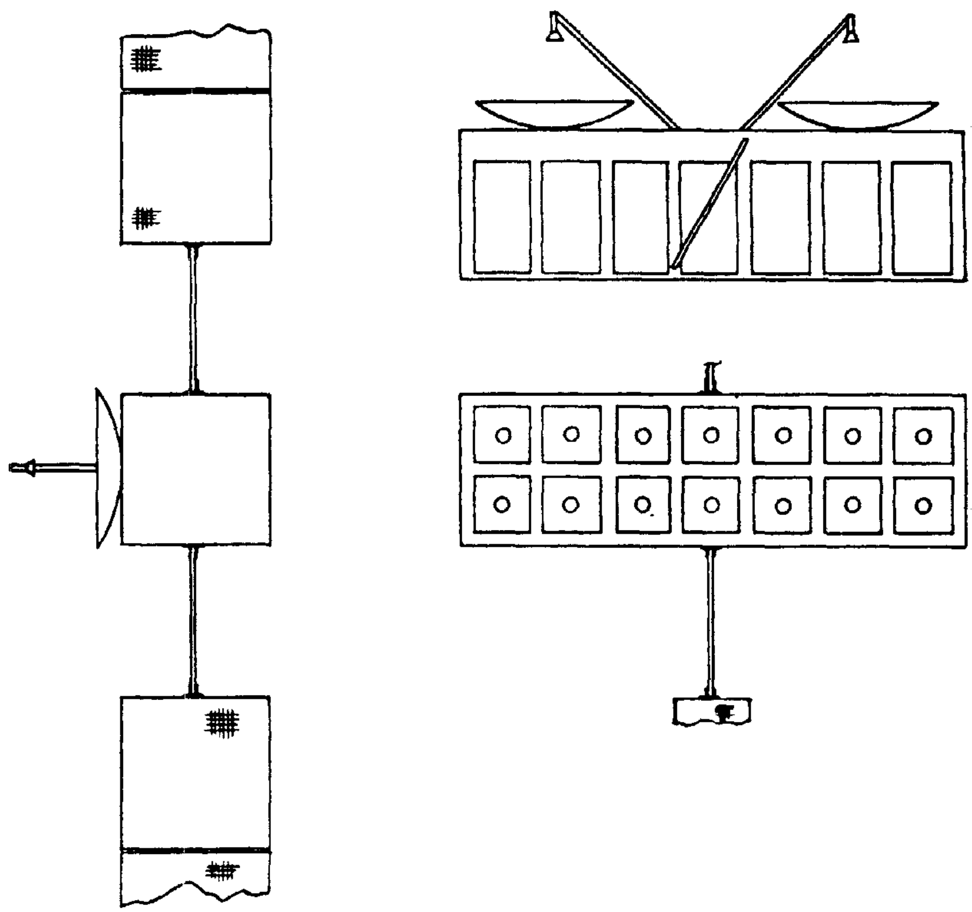


Figure 13. Preliminary Configuration for a Serviceable Communications Satellite

The impact of spacecraft thermal design on the servicer is as follows:

a. the servicer must maintain replacement module temperature before transfer,
b. the servicer must dock before significant spacecraft cooling, and
c. the servicer may have to supply power to spacecraft heaters for temperature control.

PROBLEMS OF THERMAL DESIGN AND ATTITUDE CONTROL

Redesign of the spacecraft subsystems to permit satellite servicing will present some major problems. Thermal design seems to be a prime concern. In the usual (expendable) design, satellites of this type are designed with the north and south faces as the heat-rejecting radiators and the other four faces more or less adiabatic. Thus, all dissipative components are in thermal communication with the radiators. In a satellite with replaceable modules, this may not be possible. For a module to be easily changeable, it must somehow be removable. Thus, there must be contact surfaces between the module and the satellite. This type of joint offers contact resistance to conductive heat flow. In the usual application, this type of joint is kept under high pressure with many mounting bolts and often has grease or other soft material in the interface. Some thought will have to be given to thermally self-contained modules with the radiator surface on the face of the satellite looking away from the earth.

Problems concerning the attitude stabilization and control system will also have to be solved. This may be a good argument for the use of zero momentum or reaction wheel stabilization systems rather than biased momentum systems. The docking itself may deactivate the reaction wheels and desaturation thrusters. In a biased momentum system, a high-speed wheel (4000-5000 rpm) takes two to three hours to spin down upon removal of motor power.

WAVEGUIDE CONNECTORS FOR MODULES

The design of the serviceable communications satellite shows nine modules requiring waveguide connectors. The receiver is contained in one module, which would require up to eight connectors. This may cause problems and require some system changes. The eight transmitter modules require up to four connectors: two for input and output at the standard 4/6 GHz, and in some cases two more for the 11/14-GHz input and output. These connectors would have to make and break a number of times.

Typical requirements for the waveguide connectors would be power loss (dissipative), as low as 0.01 dB, low radiofrequency interference (leakage), and low VSWR loss (0.01 dB). The requirements would depend on the actual application and relative location of the modules. Leakage from the transmitter output connectors could cause radio frequency interference (RFI) at the receiver input connectors. A research and development program to build and operate connectors at various power levels and measure leakage would be useful.

A configuration using a flexible waveguide in the module is shown in Figure 14. The main point in this design is that the moveable parts are in the module, which can be replaced, and the rigid parts are in the spacecraft. The flexible waveguide introduces increased losses, but allows the use of alignment pins for precise alignment of the two waveguide sections. The various waveguide parameters of interest for an advanced communications satellite of the early 1980s are listed in Table 4.

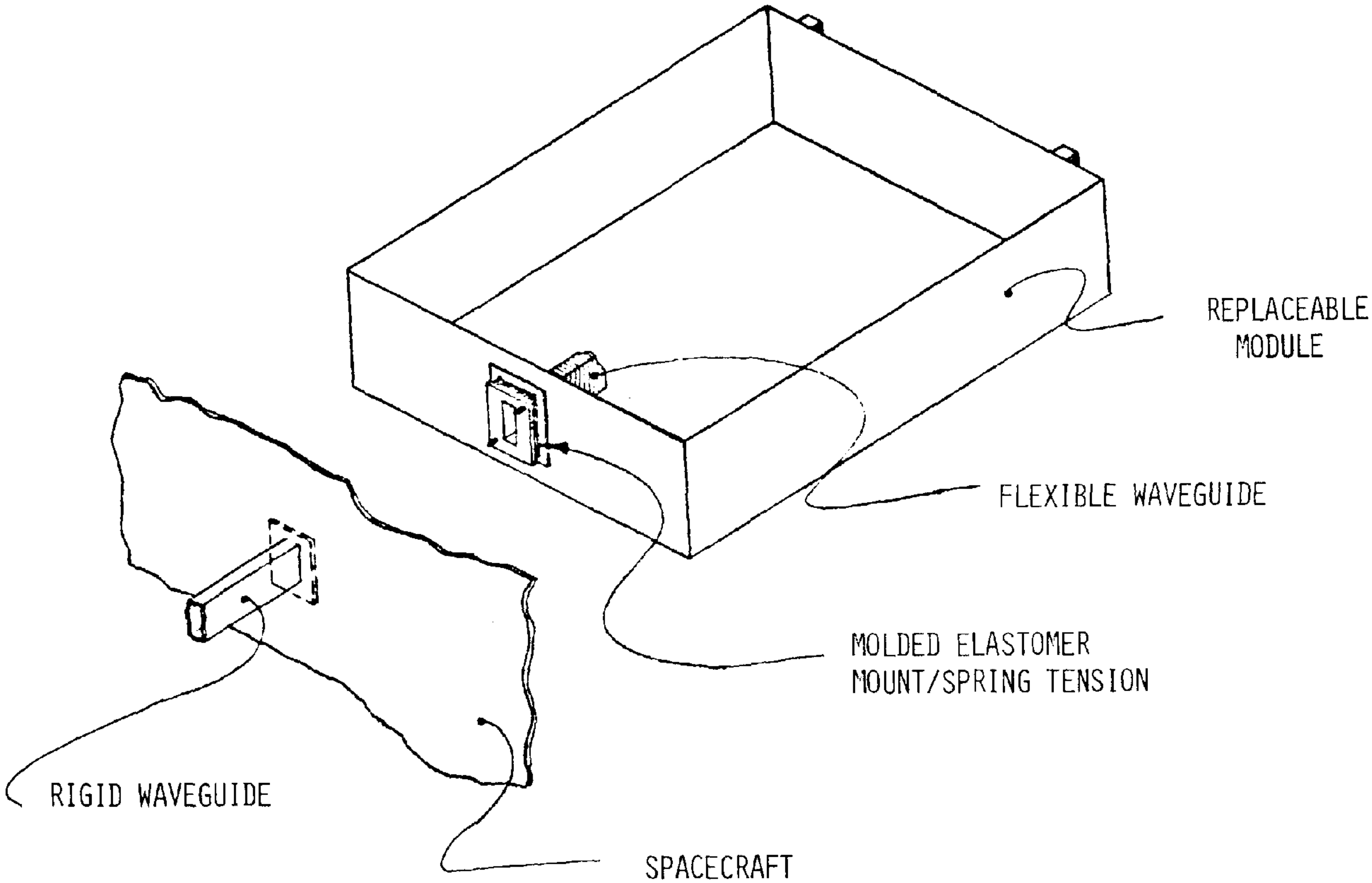


Figure 14. Waveguide Alignment Compensation Device

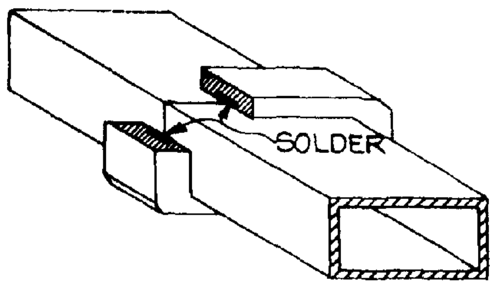
Table 4. Waveguide Sizes

Frequency (GHz)	Wave Length (in.)	Waveguide Size, Internal (in.)	Wall Thickness (in.)	Alignment Tolerance (in.)
4	2.95	2.29 × 1.45	0.064	±0.005
6	1.96	1.59 × 0.795	0.064	±0.004
11	1.02	1.020 × 0.510	0.064	±0.003
14	0.91	0.750 × 0.375	0.050	±0.003

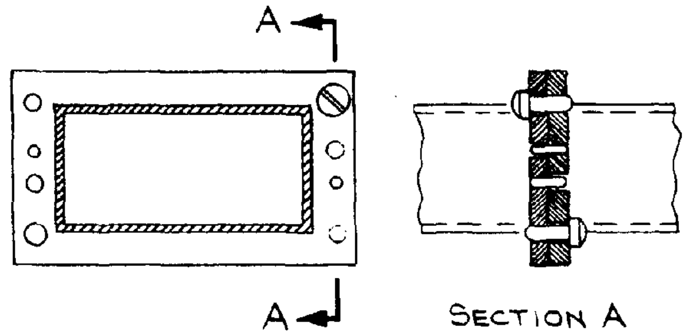
The conventional type of waveguide connections are shown in Figure 15. The soldered-sleeve type of waveguide joint is fixed and cannot be disconnected. The flange type of waveguide joint has a number of screws or bolts to hold the waveguides together. While these can be very low-loss components, they are not suitable for replaceable modules.

For a connector suitable for a module, with easy connect/disconnect capability, some gap is to be expected between the two parts. For static and rotating RF joints, a choke coupling, such as that shown in Figure 16, is often used. Most of the RF energy that leaks through the gap into the cavity is reflected, due to the dimensions of the cavity, and not absorbed. For rotating RF joints the total insertion loss has been less than 0.2 dB.

Based on the principles discussed above, a waveguide connector using a flexible elastomer to provide the force to push the two surfaces together was designed. One design is presented in Volume 1, the Executive Summary. A similar design using a spring tension plate is shown in Figure 17. While these designs provide some assurance that waveguide connectors for serviceable satellites can be built, the actual building and testing of such connectors will provide additional confidence in their technical feasibility.



A SOLDERED-SLEEVE TYPE OF
WAVEGUIDE JOINT



A FLANGE TYPE OF WAVEGUIDE JOINT

Figure 15. Waveguide Couplings

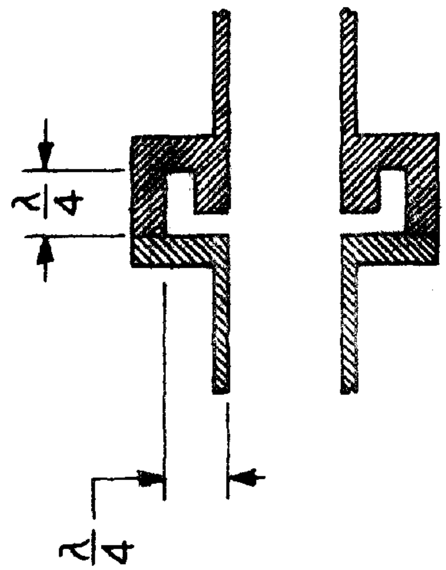


Figure 16. Choke Coupling

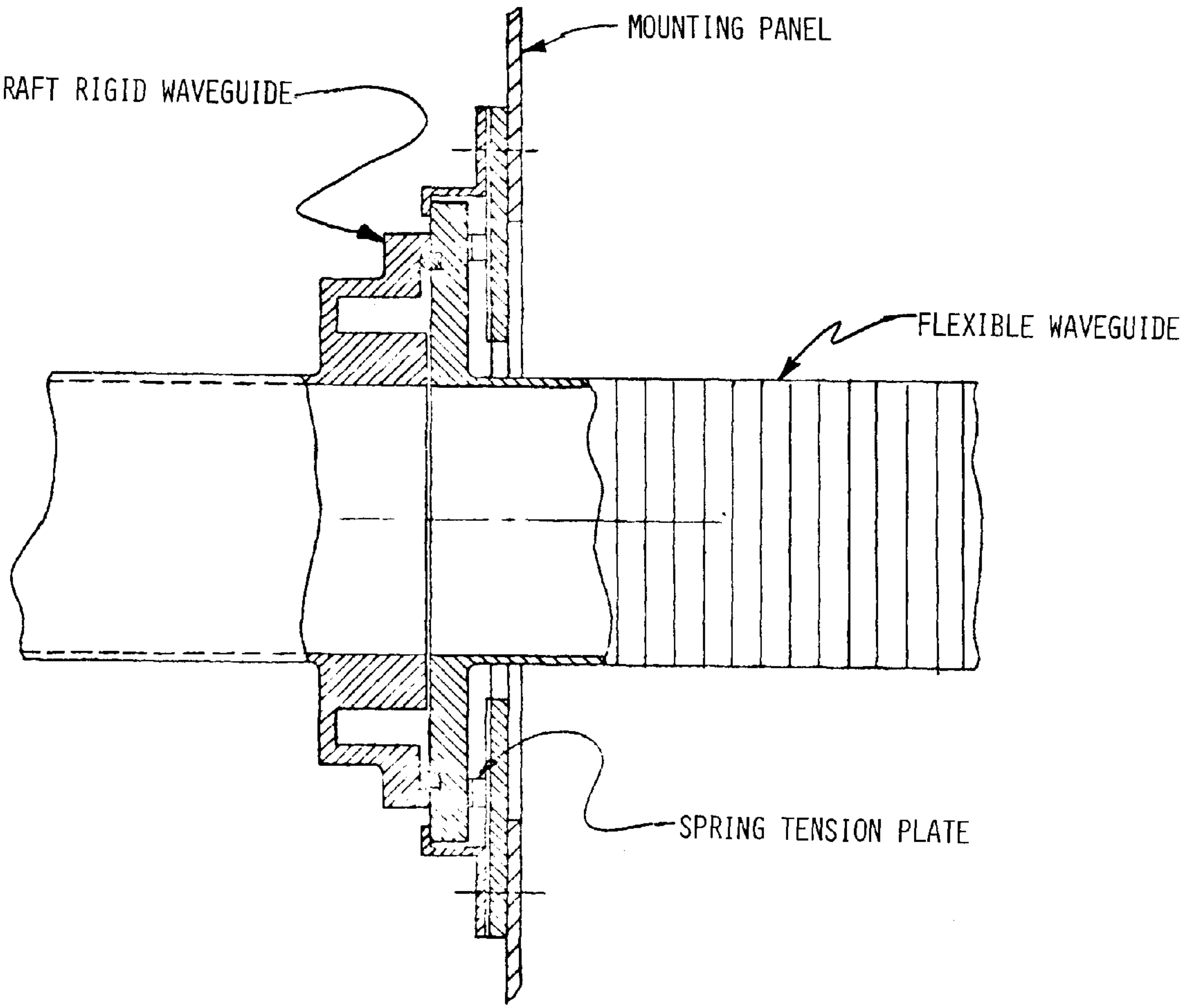


Figure 17. Alternate Waveguide Alignment
Compensation Device

ON-ORBIT SERVICING AND SATELLITE DOWNTIME

The use of the revenues lost as a measure of the "cost" of a satellite outage has been examined. Too frequent outages result in the loss of customers.

The use of route diversity (e.g., via a terrestrial competitor) may be limited for some systems and impractical for certain classes of service (e.g., wideband).

The impact of outages is most severe on the services requiring wider bandwidths. These same services, due to economies of scale, return a lower revenue per unit of bandwidth.

It is unrealistic to assign the entire down-time budget to a satellite. Customer-to-customer links include local loops and earth stations as well as a satellite. A proposed outage budget is included.

Satellite failures may be catastrophic (e.g., loss of attitude control or burned-out traveling wave tube) or gradual (e.g., degradation of solar cells or thermal control surfaces). In either case some or all of the telecommunications circuits being carried via the satellite may incur an outage (service failure) unless remedial action is taken. These circuits may be restored by switching in redundant onboard equipment, by "pointing over" all the earth station antennas to a spare satellite, or by diversion to another facility.

The "costs" of outage are often assumed to consist simply of the loss of revenue during that time. This section explores other effects that result from the outage and shows that their impact may be more severe.

Telecommunications services may take various forms:

a. telephony,
b. telegraphy (including Mailgram),
c. narrowband data (e.g., 4800-9600 bps),

d. wideband data,

e. television,

f. facsimile (including telemail),

g. leased lines (with any of the above services), or

h. restoration for terrestrial facility outages.

Grade of Service

The public has grown accustomed to high-grade terrestrial service. The Bell system claims a 99.98-percent reliability (long haul). The recent AT&T data-under-voice tariff has reduced this figure to 99.9 percent. The time allowed for outages is shown in Table 5.

Table 5. Outages and Reliability of Service

Reliability (continuity of service)	Average Maximum Outage (rounded to the nearest minute)	
	per month	per year
99.9%	43 min	8 hr, 44 min
99.95%	22 min	4 hr, 22 min
99.98%	9 min	1 hr, 45 min
99.99%	4 min	52 min

A domestic satellite must compete directly with services having these reliabilities. A failure that causes an outage in excess of that shown in Table 5 may result not only in the loss of revenue during that period, but in a loss of confidence by the customer. If the customer becomes sufficiently dissatisfied, he will go to another common carrier; thus, the long-term revenue losses are many times that of the outage.

Outage Effects

In the case of voice and low-speed data (e.g., 2.4 kbps), a short outage may cause the customer to hang up and try later. If that fails, an alternative route (e.g., dial-up via a terrestrial connection) may be possible. In most instances, the public telephone network (AT&T) and private line services must be kept isolated for tariff reasons; therefore this approach may be limited.

Higher speed and wideband data (e.g., 50, 128, 240, and 1544 kbps), represent special problems. A user with four 9.6-kbps data streams may use four individual circuits. If one circuit fails he still has three data streams. If, however, he multiplexes the four streams into one 50-kbps circuit (for economy or flexibility) and loses that circuit, he has no remaining capacity. This problem is shared by all forms of communication. On the other hand, if he is using a wideband data link to tie two or more computers together, he must provide storage, forward error control, and retransmission time (if required).

The tolerable bit error rate (BER) for voice (typically 10^{-4}) is not useful for data, which requires a bit rate of 10^{-6} or more. Fortunately, the BER of satellite services may be made very high, since only one in-orbit repeater is required (as opposed to one repeater per 30 miles for terrestrial microwave links).

In television services the timing and duration of the outage are important. An outage occurring just as a sporting team is about to kick a tie-breaking goal has a different impact than an outage occurring at the start of the game.

Although it is impossible to relate circuit bandwidth, outage, and lost customers, it appears that the more demanding services may require a disproportionately higher reliability. These services are often supplied at a lower price per unit of bandwidth due to the inherent economy of scale of wideband operations. Thus it is misleading to equate the "cost" of an outage to the lost revenue.

Outage Allocation

Experience indicates that about one third of the minutes of earth station to earth station outage is attributable to the two earth stations; the remainder is attributable to the satellites and the links between the satellite and the stations. An additional allocation should be made for local distribution links (from the earth stations to customer's premises). A proposed budget is given in Table 6.

Table 6. Outage Budget

Outage Source	Service Reliability (%)			
	99.9	99.95	99.98	99.99
One Satellite	4 hr, 37 min	2 hr, 19 min	55 min	27 min
Earth Stations (2)	2 hr, 20 min	1 hr, 10 min	28 min	14 min
Local Loops	1 hr, 47 min	53 min	22 min	11 min
Maximum Time per Year	8 hr, 44 min	4 hr, 22 min	1 hr, 45 min	52 min

Reliability

The performance of COMSAT and INTELSAT systems is evaluated on the basis of a figure of merit known as continuity of service, or the percentage of time during which circuits are operating satisfactorily. It is computed using the following formula:

$$\frac{\text{Operating Circuit Hours} - \text{Circuit Hours of Outage}}{\text{Operating Circuits Hours}} \times 100$$

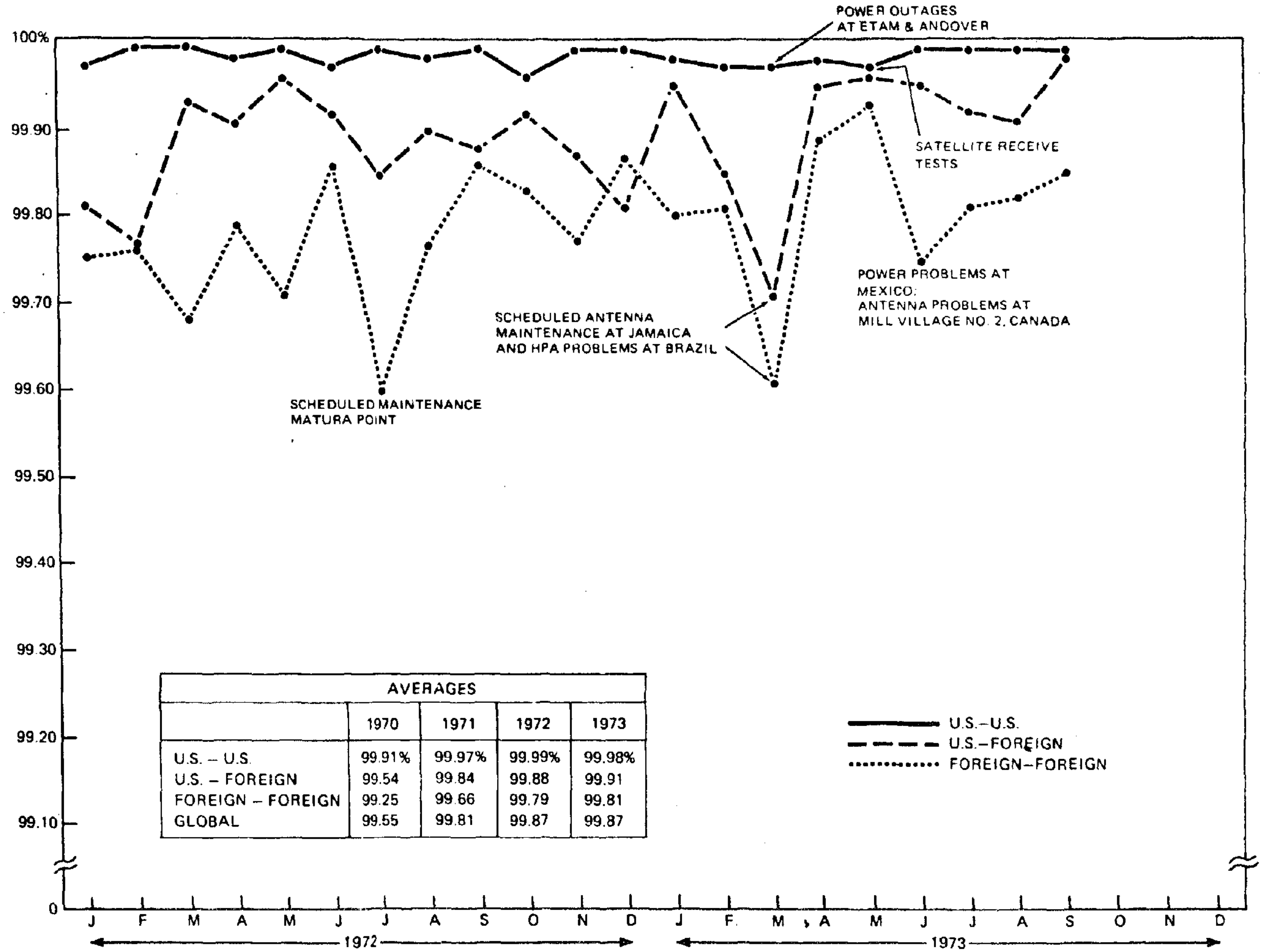
This is availability from the customer's viewpoint; notice that outages of excess capacity are not included.

This can be computed for the earth station or the satellites. For overall performance the most meaningful computation is based on the earth station-to-earth station link, which includes the satellite. COMSAT, as Management Services contractor for INTELSAT, reviews this type of performance data continuously and publishes a quarterly report of statistics concerning the global (INTELSAT) communications system. For the third quarter of 1973 and the first quarter of 1974 the continuity of service is shown in Figures 18 and 19, respectively. These figures show that the 5-year average was 99.96 for U.S. to U.S. service and 99.87 for the global system. It should be emphasized that this includes both earth station and satellite outages so that the satellites alone are even better. It should also be realized that this is required and achieved performance over five years.

Figure 19, which shows some 1974 data, also shows the dramatic effect of a satellite outage. On March 21 all traffic carried through the major path satellite in the Atlantic region, INTELSAT IV F-7, was interrupted because of an apparent malfunction in the electronic despin control system in the satellite antenna. All services were restored via the Atlantic region spare satellite, INTELSAT IV F-2, 2 hours and 13 minutes after the interruption occurred.

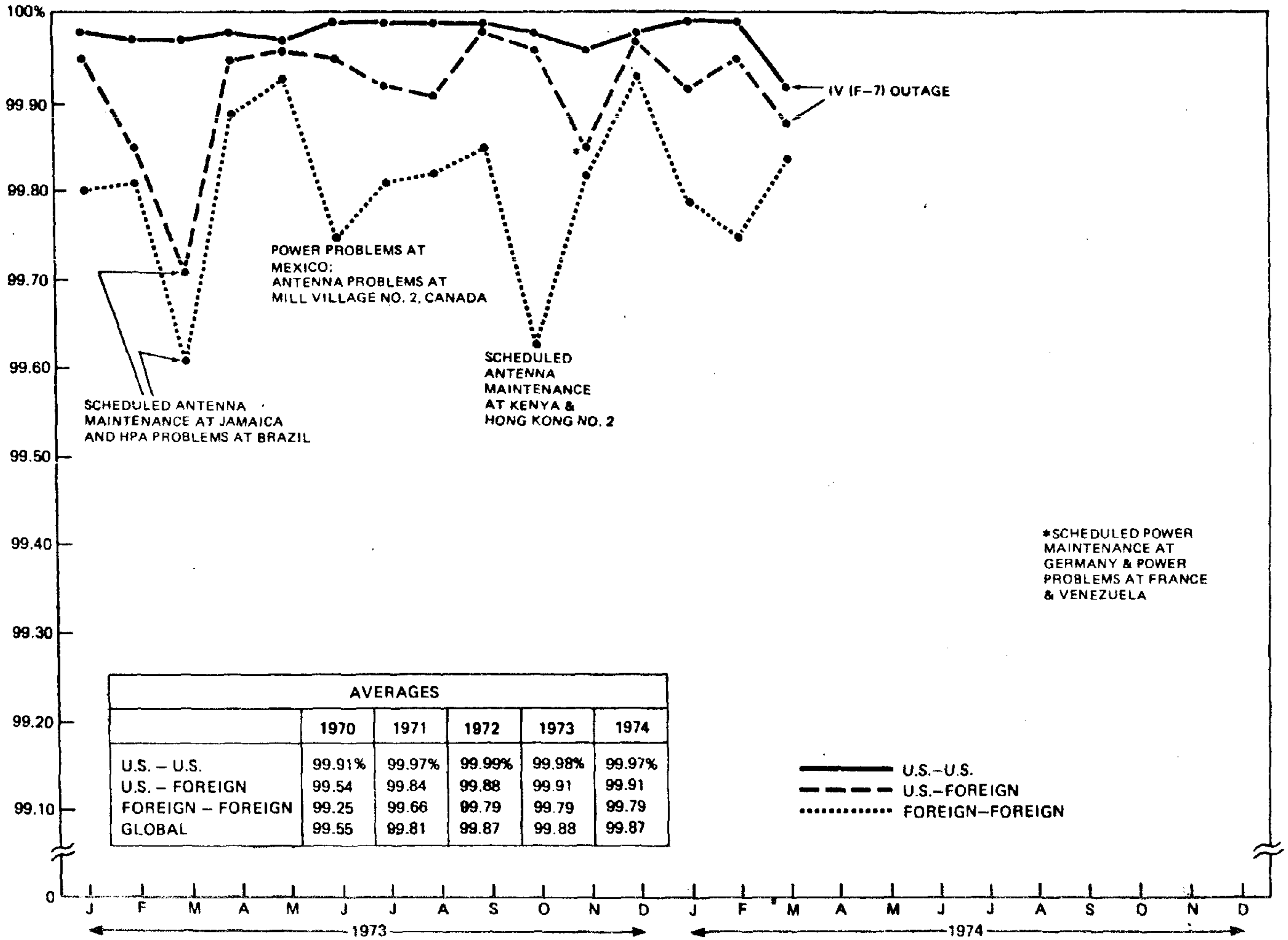
AVAILABILITY REQUIREMENTS AND FEASIBILITY

If increased availability were available at a modest cost, a higher value would always be sought. Although there are no clear-cut requirements for availability, the availabilities actually achieved in the past may be used as a guide. The previous section indicated the availability from one U.S. earth station to another U. S. earth station, including not only the outages due to the satellite, but also those due to failures of



AVERAGES				
	1970	1971	1972	1973
U.S. – U.S.	99.91%	99.97%	99.99%	99.98%
U.S. – FOREIGN	99.54	99.84	99.88	99.91
FOREIGN – FOREIGN	99.25	99.66	99.79	99.81
GLOBAL	99.55	99.81	99.87	99.87

Figure 18. Continuity of Service, 1972-1973
Earth Station-to-Earth Station



AVERAGES					
	1970	1971	1972	1973	1974
U.S. – U.S.	99.91%	99.97%	99.99%	99.98%	99.97%
U.S. – FOREIGN	99.54	99.84	99.88	99.91	99.91
FOREIGN – FOREIGN	99.25	99.66	99.79	99.79	99.79
GLOBAL	99.55	99.81	99.87	99.88	99.87

Figure 19. Continuity of Service, 1973-1974
Earth Station-to-Earth Station

the earth station equipment. These availabilities ranged from a low of 99.91 percent in 1970 to a high of 99.99 percent in 1972.

An availability of 99.99 percent is equivalent to a 1-hour outage during the course of a year. While there will be variations in the method of calculation and the particular communications satellite system which is chosen, an availability of 99.99 percent is a reasonable goal for many systems. Such an availability can be achieved with a satellite with 99.99 percent availability. However, it is usually achieved with a less reliable satellite (availability of the order of 99 percent), an in-orbit spare, and a means of switching from one to the other.

For tradeoff studies, it would be useful to know the financial value of increased availability. This cannot be defined precisely, but an order of mangitude estimate is $10 M/yr for an increase of availability from 99 percent to 99.99 percent. The value is far more than just the loss of revenue due to the outage. If a satellite revenue of $10 M/yr is assumed, an availability of 99 percent implies only a 1-percent outage, or an annual loss of revenue of $100,000.

A better estimate is obtained by considering that one country is willing to pay $3.5 M for the "non-interruptible" service of one transponder instead of the $1M paid by other countries. This is roughly equivalent to stating that a high-availability satellite might generate revenues of $35M as opposed to the $10M a year generated by a low-availability satellite. Thus the high availability is "worth" $25M a year. Another estimate can be obtained from the fact that many systems assume an in-orbit spare. Thus, instead of obtaining revenues from the in-orbit spare of the order of $10M a year, a premium of $10M a year is being paid to increase the availability of the operating satellite. That is, the improvement in availability is worth $10M a year. Similar arguments concerning the desirability of route diversity, through both multiple satellites and through cables, lead to estimates that high availability is worth of the order of $10M per satellite per year.

Present Methods of Achieving Availability

When the satellite ultimately fails, the system is maintained by replacing satellites. Hence, many steps have been taken to ensure high availability in the present communications satellite system. These include quality control, testing, redundancy, and failure warning. Satellites are built of components that have been carefully built, selected, and tested. The basic foundation of any satellite is the inherent reliability of its parts. A testing program is maintained before, during, and after manufacture to pinpoint failure mechanisms as soon as possible. The final test of a satellite includes thermal vacuum testing and vibration testing that often uncover several weak points.

Redundancy, which is a basic tool in the design of satellites, starts with the basic elements, such as a diode. At a higher level, a subsystem, such as an earth sensor or traveling wave tube, may be redundant. Frequently the entire satellite is also made redundant by including an in-orbit spare. The basic equation for the resulting redundandy, R_s, is

$$R_s = 1 - (1 - R)^n$$

where R is the component reliability and n the number of redundant components. At present satellites probably have subsystems whose availability ranges from 0.95 to 0.99. Introducing 2-for-1 redundancy and then combining all the subsystems in a reliability model makes it possible to achieve satellite availabilities of 0.99. If it is desired to increase the availability of a single satellite to 0.9999, it will also be necessary to increase the availability of components by two orders of magnitude.

While the derivation of this equation is simple, it rests on a basic assumption that the probability of failure in one component is independent of failure in a parallel component. This assumption does not hold for design failures for which the correlation may be high. Putting two components in parallel has proved useful, but additional components should not be added without further analysis.

Other Ways of Achieving Availability

New techniques are probably necessary to increase the availability of a single satellite to 99.99 percent. Four methods are suggested here. While these methods are not necessarily new, they have not yet been implemented to full advantage.

In spite of years of development, there are still a few weak areas in communications satellites. Notable examples are nickel-cadmium batteries, mechanical bearings, and traveling wave tubes. One method to improve availability is to focus on those problem areas. Hence, research has been done on alternative techniques. Nickel-hydrogen batteries are more lightweight, and in addition, tests so far indicate that they have higher reliabilities and longer life. Magnetic bearings still need development, but may be the ultimate answer to bearing problems in space. Solid-state amplifiers have been worked on for years, although it is still doubtful that they can replace all traveling wave tubes. While the advantages of these new developments have been publicized, not enough weight has been given to their effect on satellite reliability.

Testing on the ground uncovers many defects, but the ultimate test of a satellite occurs when it is launched. Another method of improving availability is to launch the first satellite two or three years before the system is really needed. Then any design defects can be detected and corrected in the remainder of the program. At this point the temptation to add other improvements in the satellites must be resisted. While such a program appears difficult to implement in practice, a number of programs have been rather expensive because the schedule does not allow for design defects to be eliminated from the second, third, or fourth launch. Figure 20 depicts a schedule in which one satellite is launched two years early. The other four satellites can then take full advantage of the investigation of any design problems that showed up on the first satellite.

To achieve the maximum improvement from redundancy, failures must be independent. A third way of improving availability is to procure redundant subsystems from different manufacturers. When subsystems are procured together, they are based on the same design, and usually have components from the same lot. The effect of a 10-percent correlation in failures for fourfold redundancy is

$$R_S = 1 - 0.1(1 - R) - 0.9(1 - R)^4$$

as illustrated in Figure 21. The curve starts off with the fourfold redundant values, but the resultant reliability is never more than one order of magnitude greater than that of no redundancy at all. For components with a 0.99 or 0.999 reliability, fourfold redundancy with 10-percent correlation is not even as good as twofold redundancy with 0-percent correlation. In general, twofold redundancy from the same manufacturer is often justified on the basis of cost savings. However, for additional reliability, a second manufacturer should be used.

A fourth method of increasing availability is to use unmanned module exchange at geostationary orbit. Studies have shown that satellite availabilities of the order of 99.99 percent can be achieved. The values depend on the delays inherent in servicing a satellite, and shorter delays are costly to implement. One possibility is to have a free-flying servicer in geostationary orbit. The costs can be minimized by servicing a number of different satellites. The satellite reliability can be maximized by

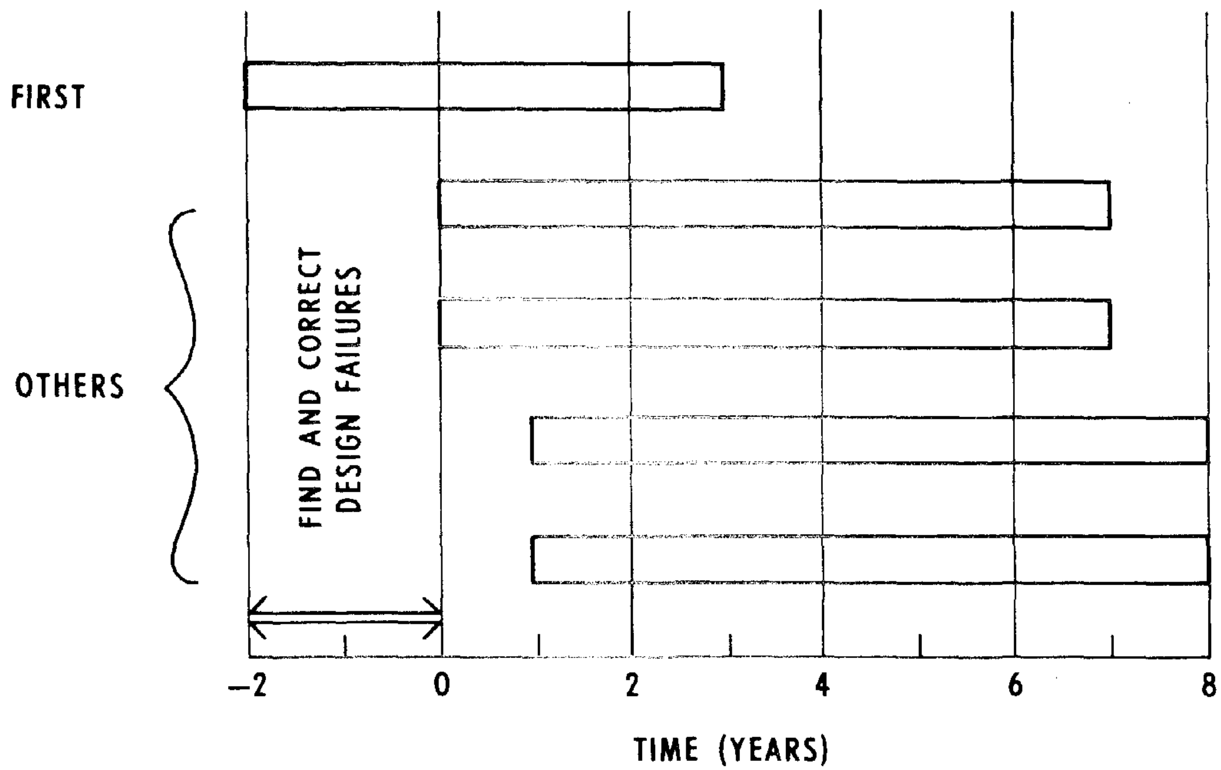


Figure 20. Schedule in which One Satellite is Launched 2 Years Before the Others

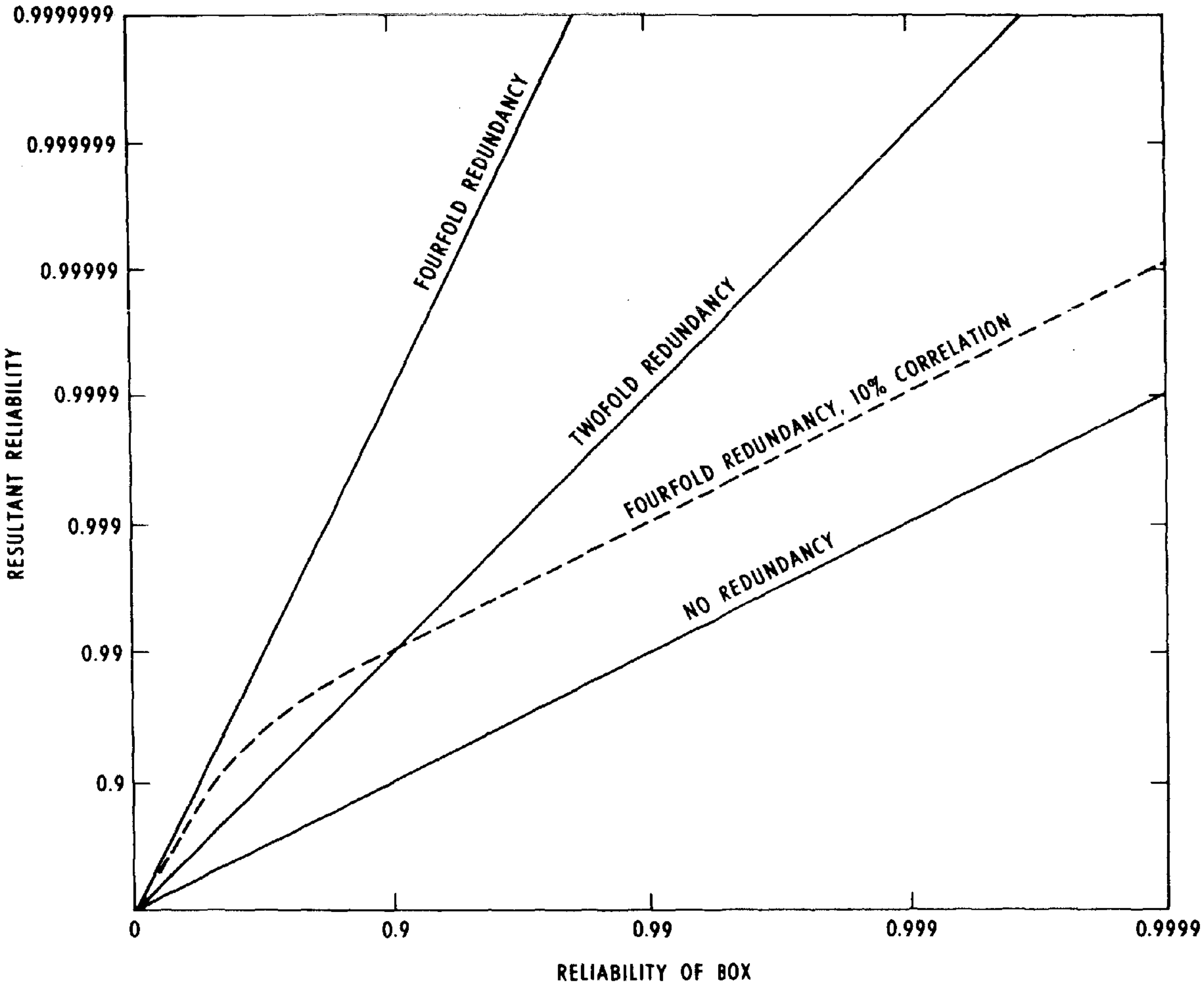


Figure 21. Effect of a 10-percent Correlation in Failures on Reliability

replacing design failures and failed redundant components, and if a module is available in geostationary orbit (either on the servicer or in another satellite) a replacement can be made within a few days.

As the history of past failures shows, a variety of failures can be expected. There are several indications that satellite availabilities of over 99.99 percent are required, and that achieving these values is worth at least $10M a year. To do this, new techniques will be needed. Some possibilities include developing new subsystems, flying a prototype two or three years before launching a system, improving the effectiveness of redundancy by diversity of manufacturers, and fixing satellites by unmanned module exchange in geostationary orbit. A comparison study of the cost effectiveness of these different methods has not yet been made. For a good appraisal, servicing must be compared against other new innovative techniques for accomplishing a desired mission.

IV. ASSESSMENT OF SPACECRAFT SERVICING

The results and conclusions of the assessment of servicing are given in Section V of Volume I. This section gives some of the background and data that were used in the evaluation. The use of one type of servicer to service all satellites or various types of servicers built to service a limited class of satellites is considered. In addition, criteria used to determine the need for a servicing operation and the benefits that accrue from servicing are discussed. Additional details of the past failures of communications satellites are included, since the occurrence of servicing is critically dependent on the occurrence of failures. Finally, the possibility of using proximity sensors for rendezvous and docking is discussed. While there appears to be no insurmountable problem, a more comprehensive survey and analysis of rendezvous and docking as it applies to unmanned servicing is needed.

SERVICER TYPES

Servicer types may be categorized according to criteria other than the type of hardware used to accomplish the required manipulations. A possibly useful alternate means of describing broad categories of servicers is suggested below (see Table 7). Boundaries between types are rather vague and a mix of types will probably develop.

General Purpose Servicer

A general purpose servicer consists of a single system designed to service all spacecraft types at most altitudes. The variety of missions to be serviced implies that the rendezvous, servicer, and module require maximum versatility to be responsive

Table 7. Servicer Types

Servicer Attributes	General Purpose	Class Exclusive	Manufac-turer Exclusive	Customer Exclusive	Spacecraft Exclusive
Spacecraft Design Limitation	Highest	Moderate	Low	Low	Lowest
Absolute Development Cost*	Highest	High	Moderate	Moderate	Lowest
Service Cost,* Non-recurring	Lowest	Low	Low	Low	Moderate
Versatility Required of Servicer, i.e., Complexity	Highest	High	Moderate	Low	Lowest
Responsible Agency Development	Govern-ment	Govern-ment	Spacecraft Contractor	Customer	Spacecraft Contractor
Servicing	Govern-ment	Govern-ment	Spacecraft Contractor or Customer	Customer	Customer or Spacecraft Contractor

*While there will be some differences in cost, these differences are small compared to the total cost.

to the needs of a variety of possible users. However, in absolute terms, a general purpose servicer is impossible; hence, it is inevitable that some users will find the compromises made to achieve generality in the module or servicer prohibitive in terms of spacecraft design. Additionally, many other users may suffer substantial inefficiencies.

It appears that only NASA can coordinate the variety of needs and develop and operate such a service system. The development costs will be amortized over a large population, but it does not necessarily follow that the costs associated with the service function will be the lowest among the alternative types. If the general purpose servicer is too sophisticated it may tend to drive recurring costs upward. An aspect which encourages servicer generality is the possibility of cost sharing of a service mission among two or more users. This has little effect upon the total scheduled service cost to a large user, but it does reduce the cash flow required. In addition, it can markedly improve service schedule flexibility, and for the small or limited user, make a service mission economically practical.

Class Exclusive

A class exclusive system consists of a servicer system designed for a class of spacecraft, such as all geosynchronous satellites built by all manufacturers. The primary motivation is to reduce the spacecraft design limitations and the servicer complexity experienced with the general servicer. Meeting the requirements for a class of spacecraft should involve fewer design compromises for both servicer and spacecraft. Government development and operation is required. The population should still be large enough to amortize developments, while a less ambitious servicer will reduce recurring costs.

Manufacturer Exclusive

A manufacturer exclusive system is one in which servicer and modules are designed to service satellites built by one manufacturer, for example, the ANIK series, including MARISAT and WESTAR. The spacecraft manufacturer is in the best position to determine the modularization strategy and the best way of implementing it with minimum penalty in terms of spacecraft design. Of course, only a manufacturer expecting to participate in a variety of programs could implement such a system, and then he would do so only if he perceived a competitive advantage. This type may represent the natural evolution of the spacecraft exclusive type. In this, as well as the following types, it is assumed that government has developed and demonstrated the fundamental principles of servicing. Future service operations could be done by the contractor under contract to the user, or the servicer could be delivered to the user for his operation.

Customer Exclusive

A procuring entity has several spacecraft programs of a sufficiently narrow class to "design" a service system that is optimum for and exclusive to his needs. This approach requires the customer to assume the primary responsibility, but presumably this responsibility is offset by improved competition among contractors during procurement. The customer may be required to perform most of the standard spacecraft design himself, hence reducing the contractor's role to that of a fabricator. The large commercial customer may elect this approach if servicing can improve his competitive position in terms of mission costs or flexibility. The needs and capabilities of several spacecraft manufacturers must be considered in designing the service system; therefore, it will be less efficient than the spacecraft exclusive type. It is implicit in this approach that the customer will perform the service function.

Spacecraft Exclusive

A spacecraft exclusive service system is one that has been optimized for a single spacecraft design. The modules and manipulator do not require versatility; therefore they (particularly the modules) can be simple and responsive to the spacecraft designer's needs. The probably low spacecraft population will make nonrecurring costs a significant element of a service function, but the absolute costs of service development can be relatively low because of simplicity through exclusiveness.

SERVICER COMMUNICATIONS REQUIREMENTS

Since the service cycle will probably be automated, a TV link is not mandatory. However, TV is beneficial in terms of providing confidence in the cycle and would have real value for spacecraft inspection for partially deployed arrays, degraded coatings, etc. The more versatile the manipulator, the more useful the TV link becomes.

Telemetry and command must be fairly complex so that the entire service cycle can be monitored and commanded. Such remote manipulations may be ground-commanded, or several service cycles can be stored onboard the servicer and initiated from the ground. A hybrid providing both capabilities appears desirable. The routine service cycles can be stored onboard, while more sophisticated functions can be handled from the ground such as fault detection and diagnosis.

It appears that the servicer and servicer function may require special telemetry capability points on the spacecraft that need to be monitored only during servicing, e.g., latch indicators, proximity sensors, and others that may overload a conventional spacecraft telemetry system. The servicer may not only have to mechanically dock, but also to provide an electrical interface with the spacecraft for these additional telemetry channels and possibly support power as well. The spacecraft telemetry system may occasionally not provide sufficient data for fault diagnosis from the ground. It is possible that the servicer could carry a diagnostic module to be inserted in the spacecraft. Such a module would permit a more refined checkout than would be possible by using the telemetry link; at the very least, it might reduce the amount of telemetry to be transmitted. In any case, it appears that the servicer would require some communications capacity.

TIMES OF SERVICING AND BENEFITS

The decision as to when to service a communications satellite will depend on the state of health of the satellite, the benefits to be obtained from servicing, the cost of servicing, and perhaps on when the servicing can be done. The state of health of a communications satellite can be classified as follows:

a. not serviceable,
b. not functioning,
c. reduced performance,
d. reduced reliability,
e. limited lifetime, or
f. perfect health.

Some of these classifications overlap, and yet each class is fairly distinct. At present, most studies concentrate on servicing satellites in the "non-functioning" category (class b), but there are a number of satellites in classes c, d and e. In fact, a majority of recently launched communications satellites are probably in these categories.

It may be useful to describe these states and to give a few examples. A "non-serviceable" satellite cannot be serviced; it may be in a spin, it may have a non-operational docking or rendezvous target, or it may have a failed non-replaceable unit. A "non-functioning" satellite can no longer perform its main mission; it is of no benefit unless it is serviced, and if availability is important, the servicing is urgently needed. A "reduced performance" satellite may have lost some transponders, some gain in its RF output, a telemetry channel, or eclipse capability, for example. While it may no longer meet specifications, it is still a useful satellite, and may be meeting all the present performance requirements. A "reduced reliability" satellite (class d) still provides full performance, but for some reason the probability that something will go wrong has increased; examples are a design failure on another similar satellite, a loss of redundancy, or a warning that some component is not quite normal (quite possibly on batteries or bearings). A "limited lifetime" satellite is similar to the above, but its reliability over the near future has not changed; perhaps the only clear-cut example is fuel depletion.

The possibilities of servicing satellites in classes c, d, and e should be investigated for the following reasons:

a. availability can be vastly increased,
b. the number of servicing operations can be increased,
c. urgency of servicing can be reduced, and
d. costs per servicing operation can be reduced.

Servicing is not a question of "either/or"; instead, there are various gradations between servicing only complete failures and servicing all satellites with anomalies.

The usual study has taken a reliability curve that shows the probability of failure (class b) and assumed that servicing is performed when the satellite has failed. Theoretically it would be possible to use a reliability model of the satellite to predict when servicing would be desirable for these other modes. However, even if this were possible, it would probably not be desirable because servicing on demand is expensive and probably not necessary. A better procedure would be to assume that servicing will be done at a certain frequency (for example, once a year). This could be in addition to servicing on demand when a total failure occurs, but the frequency of this service on demand would be reduced because of the maintenance cycle. The reliability model could then be used to find the satellite availability, and the frequency of module exchange could be based on this maintenance concept.

HISTORY OF FAILURES

A previous COMSAT report[4] to NASA on Contract NAS 8-30285, "Assessment of On-Orbit Servicing of Synchronous Orbit Spacecraft," included a table entitled "Typical Communications Satellite Failures." This table, with an added column of comments, is shown as Table 8.

As noted in the table, apogee motor failures caused an INTELSAT II to fail to achieve orbit. An INTELSAT III satellite, F-8, also experienced failure during apogee motor firing and did not achieve synchronous orbit. In addition, of the eight launches in the INTELSAT III program, two satellites failed to achieve orbit due to launch vehicle failures. Launch vehicle failures are not noted on the table.

Table 8. Typical Communications Satellite Failures

Satellite	Component Failure	Type	Reparable	Comments
SYNCOM, INTELSAT II, III	Apogee Motor	Design	No	INTELSAT II motors had a cold nozzle
INTELSAT II	Fuel Lines	Design	Probably	Hydrogen peroxide system not completely passivated.
INTELSAT IV	Thruster	Design	Yes	Not properly designed for thermal soak back.
INTELSAT III, TACSAT	Structural Bearings	Design	Difficult	Exact trouble not fully known. Complex design.
NIMBUS	Solar Array Bearings	Design	Difficult	
INTELSAT II	Solar Array Degradation	Design	Probably	
INTELSAT II, III, TELSTAR	Battery	Random	Yes	Low battery voltage during eclipse
INTELSAT III INTELSAT IV	Earth Sensor	Design Random	Yes	Detector noise level on some sensors gradually increased causing beam-pointing errors on INTELSAT IV INTELSAT III had an off-axis response.
INTELSAT III	Receiver	Design	Yes	Tunnel Diode Amplifier
INTELSAT III	Transponder	Random	Yes	
SYNCOM	Telemetry	Random	Yes	
TELSTAR, COURIER	Decoder	Design	Yes	
EARLY BIRD	Fuel Depletion	Wearout	Yes	
RELAY, TELESAT	Power Conditioning	Random	Yes	Poor manufacturing quality control on transistors
DSCS-2	Deployable Structures	Design	No	Telemetry Antenna
INTELSAT IV	Telemetry Beacon	Random	Yes	Failed during launch
ATS-5	Attitude Control	Design	No	
INTELSAT III	Low Orbit	Random	Yes	Underperformance of launch vehicle that was compensated for with onboard propulsion
INTELSAT IV	Receiver	Design	Yes	Intensive investigations point to cathode degradation in the low level traveling wave tubes
WESTAR-1, SMS-1	Low Orbit	Design	Yes	Component failure in Thor/Delta DIGS

It should be noted that, with the exception of the despin bearing, structure, and apogee motor subsystems, INTELSAT satellites have no single point failure modes which will terminate the mission. Thus the loss of an earth sensor on INTELSAT IV is not catastrophic since there are three earth sensors, two sun sensors, an onboard clock, and a ground despin control mode. The loss of one receiver is not catastrophic, since there are four receivers, only one of which must operate. On INTELSAT IV F-2 one axial thruster was lost, but there was a redundant one. In fact, there were two independent hydrazine systems onboard, and the later models have a cross-connect valve between the two systems.

As a representative group of commercial communications satellites, the INTELSAT satellites have a very high degree of redundancy. This has been a prime factor in their remarkable success. It has also caused weight penalties. Full redundancy and, in some cases, 3- and 4-for-1 redundancy has been used. This reliable design has given good performance.

Over the history of satellite launches, including one INTELSAT I, three INTELSAT IIs, eight INTELSAT IIIs, and five INTELSAT IVs, four out of the 17 failed to achieve orbit, two due to launch vehicle problems and two due to apogee motor problems. It can be presumed that these types of failures will not apply to future launches with a successful shuttle and tug system. Even though it still seems technically impossible to have a redundant spacecraft structure, most structural failures occur during the launch phase due to high g loading. The shuttle should have a much more benign loading situation; thus a nonredundant structure may cause no problems. From this point of view, deployment failures of deployable members and devices are classified as mechanism failures rather than structural failures, but this is semantics.

The despin bearing on TACSAT and INTELSAT III caused in-orbit problems. This joint is the heart of a 2-body or dual spinner design. It is buried deep in the spacecraft body (a good location for thermal design) and is impossible to replace in orbit. In addition, there is as yet no good way of making that subsystem redundant. On the other hand, many of the next generation of communications satellites, such as FLEETSATCOM, the RCA GLOBECOM, the Japanese broadcast satellite, and the global positioning satellite, are body-stabilized satellites. The satellite body is stabilized toward the earth and the solar arrays toward the sun. The stabilization device consists of momentum or reaction wheels in concert with attitude sensors and onboard thrusters. These devices can be made redundant (with a weight penalty) and may be replaceable in orbit, at least much more easily than the despin bearings of a 2-body spinner design. Of course the replacement concept must be factored into the design from the start, which has not been done for the aforementioned next generation communications satellites.

As mentioned previously, the successful commercial communications satellites are designed with a high degree of redundancy. They are also designed for what is known as "graceful degradation." A good example is the INTELSAT III battery problem noted in Table 8. INTELSAT III F-6 was launched in February 1970. More than three years later, in the spring eclipse season of 1973, the battery voltage dropped below the required level with both transponders turned on. In subsequent eclipse seasons, one transponder could be turned off, thus reducing communications capacity to result in a graceful degradation rather than a catastrophic failure.

The final point is that this table is representative of satellite problems of the 1960's and early 1970's. Even Relay, Telstar, and Early Bird are included. On-orbit servicing will be applied to communications satellites in synchronous equatorial orbit in the mid to late 1980's. Device R&D is underway in all areas to solve the current problems. This study must attempt to define future satellite subsystems rather than those of the past or current time frame. Nickel-cadmium batteries which have cycle life problems may be replaced with nickel-hydrogen or hydrogen-oxygen cells by that time. Conventional bearings may be replaced with magnetic bearings; structural bearings disappear on body-stabilized satellites. Hydrazine tanks, thrusters, and lines may be replaced with electric thrusters, and traveling wave tubes with solid-state devices.

Despite all this, the one type of problem that will always recur is the design failure due to human error. Thus it may be economically justifiable to design these new, very technically advanced subsystems in replaceable modules for on-orbit servicing of modular-type satellites.

DESIGN FAILURES

Data presented in the last sections show that almost half of the failures that have occurred in communications satellites can be classified as design failures. For convenience in classification, these are failures for which subsequent analysis has shown that the actual reliability was lower than the planned reliability. This may be due to the initial design or to methods used in implementing the design (quality control). A more detailed survey of the design failures in three series of satellites has been made so that conclusions can be drawn for servicing studies.

The main subsystem failures or anomalies are listed in Table 9. Note that, in spite of these difficulties, the satellites were successful; in some cases the problem was not serious enough to affect the satellite mission and in other cases redundancy or alternate operating modes were available to remedy the problem.

Each line in Table 9 notes the number of satellites in which the design failure was observed. The next column shows the number of satellites in which a replacement module was needed in an attempt to balance the severity of the problem with the estimated cost of a servicing mission to the satellite. In a number

Table 9. Design Failures or Anomalies in Communications Satellite Subsystems

Satellite	Component	Number Failures Observed	Number Needing Replacement	Total Satellites Injected	Total Satellites Launched
INTELSAT II	Propellant Feed	3	3		
	Relief Valves	3	3	3	4
	Solar Array	1	0		
INTELSAT III	Structural Bearings	5	5		
	Receiver	1	1	5	8
	Earth Sensor	5	0		
INTELSAT IV	Receiver	4	4		
	Thruster	1	1	7	8
	Structural Bearings	2	0		
	Earth Sensor	1	0		
	Totals	26	17	15	20

of cases, the severity was not sufficient to justify a servicing mission; yet if the satellite were to be serviced for another reason, that module would be replaced.

The following subsections briefly describe the design failures.

INTELSAT II

Propellant Supply and Feed System. Improper cleaning of the fuel system prior to filling left some impurities. This produced foreign particles that eventually led to thruster system failure in the closed position.

Relief Valves. The valves that were designed to relieve the pressure buildup in the tanks (due to decomposition) failed prematurely.

Solar Array. The solar cell cover failed to cover completely. The fraction of power lost was far greater than the fraction of the cell exposed because radiation damage tended to short circuit the entire cell.

INTELSAT III

MDA. Improper design of the mechanical drive assembly (MDA) caused an intermittent seizure of bearings that was highly sensitive to temperature. Heaters installed on later satellites partially alleviated but did not cure the problem.

Receiver. One receiver failed.

Earth Sensor. Internal reflections produced a response to the sun at certain angles far from the axis. There was no testing for this problem prior to first launch. It was operationally solved by switching to alternate modes.

INTELSAT IV

Receiver. There was a gradual loss of amplification due to a defect in a cathode of the traveling wave tube. Defects were present in satellites launched in first year and a half of the program, but it is believed that they were corrected in satellites launched after three and a half years.

Thruster. One thruster failed due to heat soak-back after the thruster was turned off and then refired. Operational procedures were changed so that the firing sequence would not be used. A valve was added on later satellites so that two fuel systems could be interconnected by ground command.

Earth Sensor. Discrimination circuits against the moon failed to work properly when sun, earth, satellite, and moon were in a direct line. The problem was operationally solved by switching to alternate earth sensors.

On the basis of these statistics, the following average prediction can be made for future programs: Each new program can expect three design failures, one of which will be sufficiently serious to warrant a module replacement. The time at which a failure has been detected has varied from a few hours to four years. On the average, a design failure appears about a year after injection of the first satellite in the program. An additional year or two is required to identify the cause and procure replacement modules without the defect.

PROXIMITY SENSORS

In an earlier study[6] for NASA, COMSAT wrote in October 1973:

> "It appears that little thought has been given to developing an ideal target on the spacecraft. A study is needed that would develop a variety of targets and docking techniques so that a better choice can be made.
>
> "As a possible example of a target for a spacecraft, consider the system shown on Figure 22. For simplicity, only 2-D motion is considered, although extension to 3-D is straightforward. On the spacecraft there are two light bulbs, with a shield in between, so that one is visible from one side and the other, from the other side. One bulb has its intensity modulated at one frequency, e.g., 1 kHz, and the other at a different frequency, e.g., 2 kHz. The service unit has two solar cells mounted at appropriate angles. With suitable filters four signals are obtained: A_1, A_2, B_1, and B_2, where A_1 is the 1 kHz signal from solar cell A, and A_2 is the 2 kHz signal from solar cell A (similar notation for B). The attitude error signal for the service unit is obtained from $A_1 + A_2 - B_1 - B_2$, and the translational error signal would be obtained from $A_1 + B_1 - A_2 - B_2$. This example has the additional advantage that sensitivity would increase when the light is near grazing angle to both solar cells, and the geometry can be chosen so that this occurs at the most critical point of the maneuver, that is, just before docking.
>
> "With two more light bulbs at two additional frequencies and two more solar cells, a 3-D system can be built. Information on range and yaw error is also present although it may take more sophisticated circuitry to derive the needed signals."

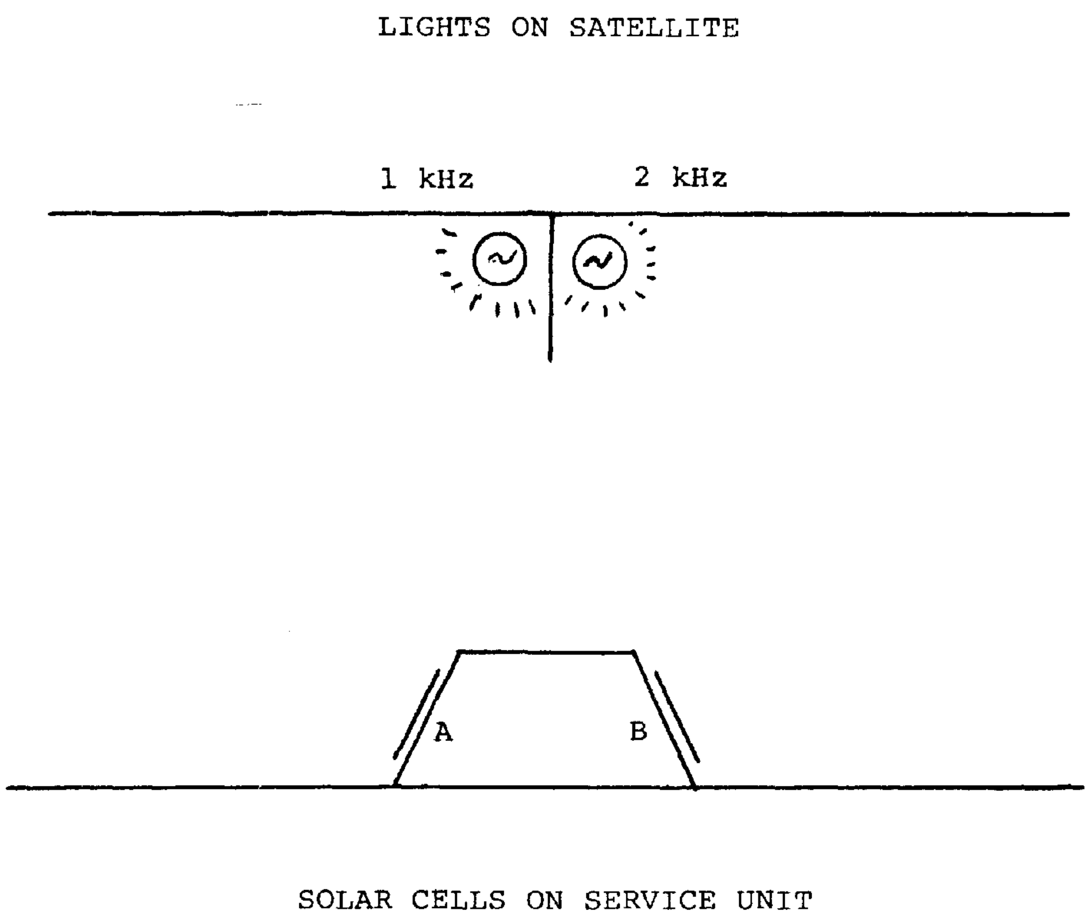


Figure 22. Example of Target for Spacecraft

Unknown to the author, a few months earlier Alan R. Johnston of Jet Propulsion Laboratory had conceived and recorded a similar, but better, cooperative 6-axis sensor. The target, which would be mounted on the spacecraft, was completely passive. The following paragraphs describe the concept; more details are provided in the New Technology Report.[7]

> "To bring together the position and direction references, a composite mirror is proposed, as shown in Figure 23. The composite mirror contains a retroreflector and two plane mirrors, which together define an object-fixed coordinate system. Three LED light sources would be mounted as shown, one operating in conjunction with each reflector surface. LED 1 operates with the corner reflector as described above, while LED 2 and LED 3 are reflected toward the detector module by the two plane mirrors. Light sources LED 2 and LED 3 are mounted far enough from the sensor axis that the retroreflector return from them does not enter the detector. Figure 23 ignores the complications caused by limited physical size of the plane mirrors, and by the small misalignment of the return from LED 2 and LED 3 at null. Appropriate shaping of M1 and M2 will be necessary to avoid these difficulties [this probably refers to M2 and M3--GDG].
>
> "The two remaining coordinates are azimuthal rotation about the sensor axis ϕ, and linear distance from sensor to mirror r. Both ϕ and r depend on appropriate combinations of the signals already described. For example, the detector unit will indicate the apparent direction R2 of the ray from LED 2 reflected by M2, in terms of a signal V_{x2}. This in turn will depend both on the orientation of the composite mirror as a unit, shown as β_x, and the distance r.
>
> "The signals associated with the three separate light sources would be obtained from the same detector assembly. Each LED would be pulsed in turn. The associated signal would be separated by means of specialized phase detection circuitry which would in essence

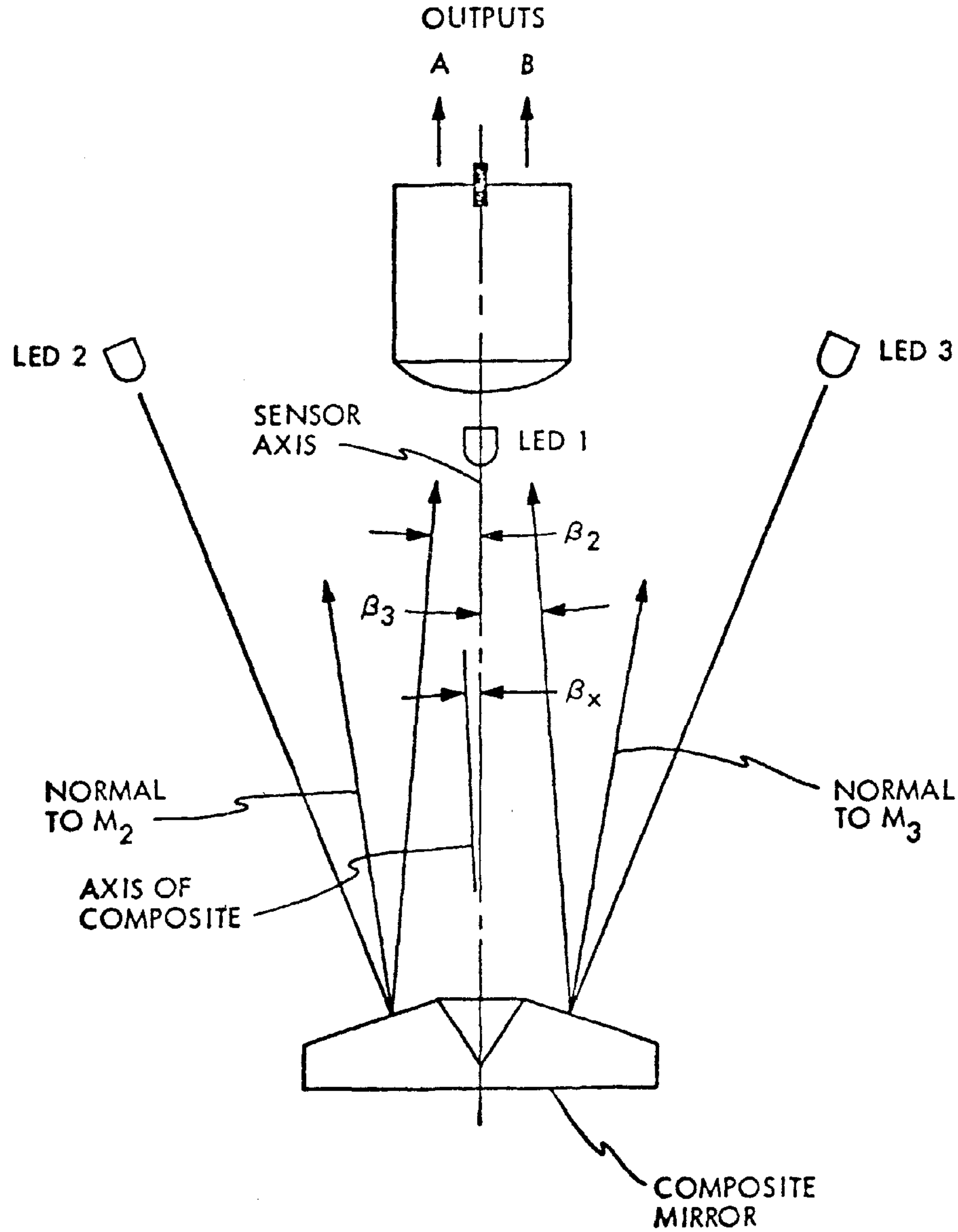


Figure 23. Cooperative Six-Axis Sensor

gate the output signal observed in response to a specific LED pulse into its own output channel. Considerable simplification of the detector package results because the sensitive low level detector electronics are in effect multiplexed. The range of position over which this type of sensor will operate is an important question, as it determines the volume over which it will acquire its target reflector. The retrochannel, LED 1, is expected to have the largest range, and therefore one expects control to be initiated using only the α_x, α_y channels. Control strategies required in order to reliably bring the manipulator to its sensor null from any position within its acquisition volume would have to be developed in parallel with the sensing device itself.

"The sensing head contains a light collecting telescope lens, and a detector, tentatively a silicon photovoltaic cell divided into four electrically independent quadrants by etching through the junction layer. Overall size of the detector package could be of the order of 2 cm diameter × 3 cm long. An output signal would be brought out from each quadrant, but for simplicity only two, labeled A and B, are shown in the figure. An IR filter would prevent room light from entering the detector.

"The lens would be positioned to bring a bundle of light from a distant small area source to focus in a small spot on the detector. The position of this spot on the detector would vary with the direction to the light source. Therefore, the proportioning of the total light flux between detector A and B would depend on the angular position of the source with respect to the axis of the detector package."

It would be desirable to have a sensor that can operate from initial acquisition to within a few feet, with accuracies in the latter phase of the order of an inch. At present, orbit determinations can be used to bring the servicer within a kilometer or so of the satellite without modification of present tracking methods. The target on the satellite is completely passive and fairly simple. The accuracy during the final phases of rendezvous should provide the capability for closing velocities smaller than those used in most docking maneuvers.

V. COSTS OF SERVICING

Different approaches can be used to determine the costs of a program. One approach is to estimate the cost for each part of the program and add the results. Another approach is to examine the totals, determine the driving assumptions, and compare the totals with other similar cost figures.

During the course of this study, cost figures were generated by COMSAT, discussed at monthly coordination meetings, and reported in quarterly reports. Cost figures were also generated by Martin Marietta, where a very detailed and thorough cost analysis was performed.[1] Close coordination was maintained between the two studies, and some suggestions by COMSAT were incorporated into the material presented in the Martin study report.

Instead of presenting different cost estimates, Table 10 shows estimates based on the Martin data, but from a user's viewpoint. The costs per unit are given rather than program totals. The DDT&E costs ($690M, $2456M, and $6894M for each column) have been excluded. For the expendable case the costs are the totals divided by the number of spacecraft. For the maintenance modes, the totals for the initial placement of satellites have been subtracted, and the remainder divided by the number of servicing operations.

The costs given in Table 10 are for 7 communications satellite programs, for all medium- and high-orbit programs (MEO/HEO), and for all mission models that can be serviced. Thus, for the first column there are 41 initial launches, and 42 (83 - 41) additional operations: replacements, refurbishings, or services. For the initial satellites, the cost per spacecraft is $22M plus an additional $5M for delivery to geostationary orbit. The cost of refurbishing the spacecraft is $9M for transportation and another $8M for the rest of the operation. Finally, for on-orbit servicing, the cost of transportation is $4M and that of the replacement

Table 10. Costs Per Unit
(excluding DDT&E Costs)

	Communications Satellites	Medium and High Orbit	Complete Set
Initial Spacecraft	41	61	93
Total Number Expendable	83	160	340
Expendable (per spacecraft)			
Transportation (M$)	5	7	6
Spacecraft (M$)	22	31	47
Total (M$)	27	38	53
Ground Refurbishment (per replacement)			
Transportation (M$)	9	11	11
Spacecraft (M$)	8	19	23
Total (M$)	17	30	34
On-Orbit Maintenance (per service)			
Transportation (M$)	4	5	3
Spacecraft (M$)	11	12	13
Total (M$)	15	17	16

modules and operations is $11M per servicing. This includes any additional cost for building the spacecraft to be serviced.

The figures for all the medium- and high-orbit programs are slightly higher, but fairly comparable. (Transportation costs for medium orbits include the full charge for a tug.) When the low-orbit programs are included, the transportation costs are comparable, but the spacecraft costs are higher since some large spacecraft are included.

It appears to be more cost effective to service a communications satellite for $15M instead of replacing it for $27M. However, this satellite is seven years old, and changes may be desirable for a new satellite. From a project manager's viewpoint, it may be necessary to lower the cost of a servicing operation considerably below 50 percent of the cost of a new satellite before in-orbit servicing becomes attractive. Further studies should be made to optimize servicing operations and bring the cost per service below $10M.

VI. LAUNCH VEHICLE EFFECTS ON SPACECRAFT

This section discusses the possibility of using excess cargo bay volume for satellites. Several of the present concepts of satellite layout must change to exploit the available space.

SPACE FOR SATELLITES

The clear portion of the shuttle cargo bay is 60 ft long and 15 ft in diameter. The tug and tug/shuttle interface equipment shall not exceed 30 ft. The length required for the tug/spacecraft interface adapter can be charged to the spacecraft length[8] which is determined as follows:

overall cargo bay length:	60 ft
maximum tug:	-30
	30
tug/spacecraft adapter:	-1
remainder for spacecraft:	29 ft

The total volume (15 ft diameter by 29 ft long), as shown in Figure 24a, is about 5000 cubic feet or 1.4×10^8 cm^3.

SHARING THE SPACE AMONG SEVERAL SIMILAR SATELLITES

The cylindrical area may be divided among two or more satellites. These satellites need not all be intended for the same service, but it is anticipated that the present-day grouping of satellites into similar weight and size classes will continue. Figure 24b shows two cylindrical sections, while Figure 24c shows two half-cylinders. An INTELSAT IV is shown in each model.

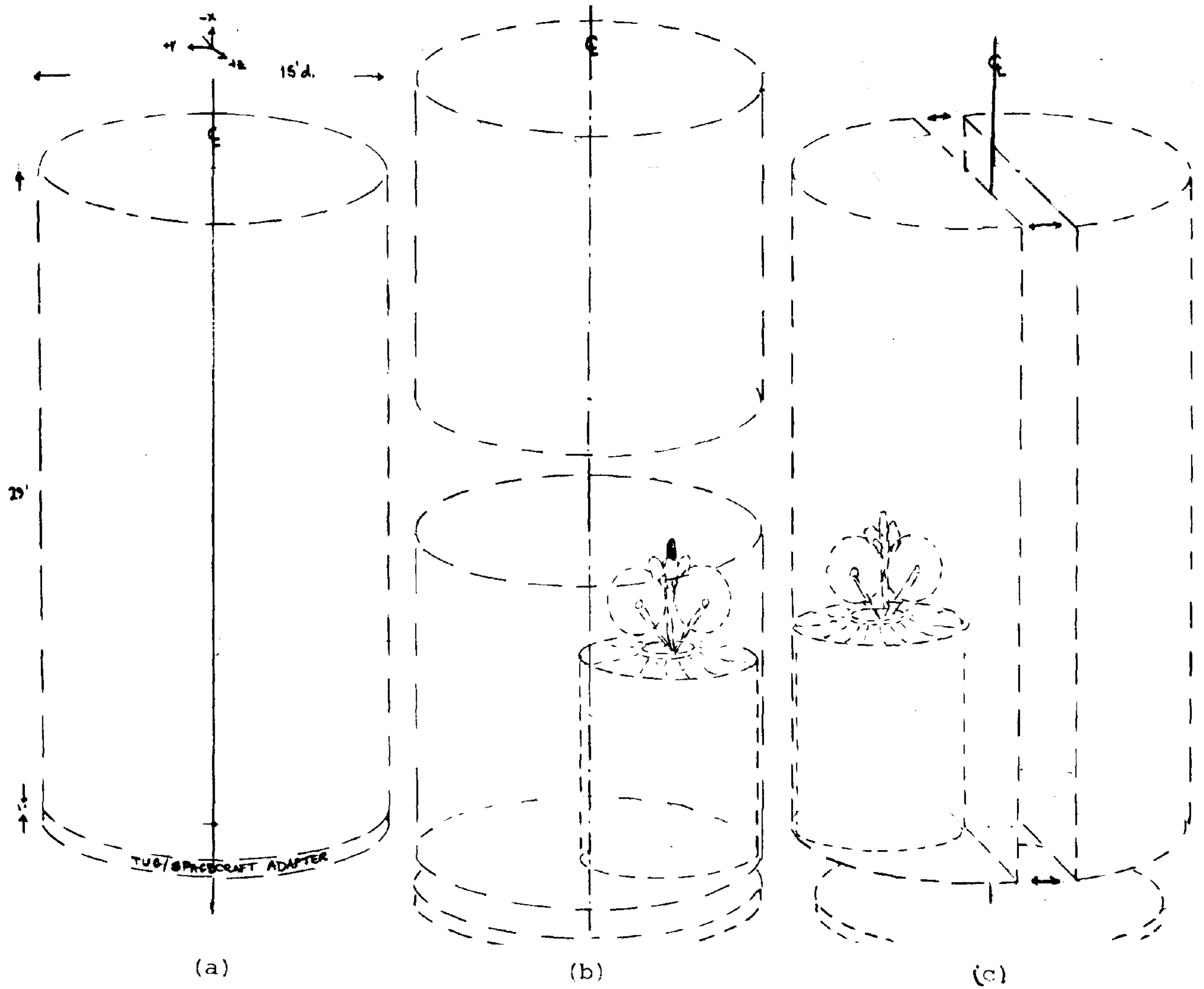


Figure 24. Volume for Satellites

For scale purposes the INTELSAT IV-A, COMSTAR (AT&T), and TACSAT are similar in size and mass. The INTELSAT IV is slightly larger in diameter than the radius of the cargo bay. It is also slightly higher than half of 29 ft. Since some redesign would be needed for a tug launch (e.g., removal of the AKM, and little or no need for transfer orbit beacon and telemetry data), these problems could be overcome.

NUMBER OF SATELLITES PER LAUNCH

The reported weight of INTELSAT IV F-3 on May 15, 1971, was 1601 lb. Removing the AKM casing and attaching hardware saves 131 lb, of which some must be replaced by structure. Based on an assumed weight of 1500 lb for an INTELSAT-IV-type satellite and the current NASA values [1] for the geosynchronous tug capabilities, the number of satellites per launch has been estimated in Table 11. With the exception of the case in which the tug exchanges a new satellite for an old one (deploy and retrieve), multiple INTELSAT IV class satellites can always be launched.

Table 11. Shuttle/Tug Launch Capabilities

	Delivery Only	Retrieval Only (full satellites)	Deploy and Retrieve (each way)
Weight (lb)	6000-8000	3000-4000	2070
Number of Satellites			
1500-lb satellite (INTELSAT IV)	4-5	2	1
2000-lb satellite	3-4	1-2	1
2500-lb satellite	2-3	1	--
Average Density*			
(lb/ft^3)	1.2-1.6	0.6-0.8	0.4
(gm/cm^3)	0.019-0.024	0.01-0.012	0.0064

*The tug lifting capacity divided by 5000 ft^3.

If less expensive satellites can be built by using heavier but less expensive components (e.g., batteries and RF filters), the total mass of the satellite may increase to 2000 or 2500 lb. In most of these instances, there is still enough mass to carry two, three, or more satellites per tug. Therefore, it is concluded that each tug is likely to carry at least two satellites. Figures 24b and 24c indicate that there is more than enough volume to house several INTELSAT IVs.

The question of how to handle two (or more) satellites is important since it affects the size and, more importantly, the shape of each satellite. The shape of the overall satellite influences the modularization, which is in turn part of the orbital servicing study.

SATELLITE SHAPES

Despite the cylindrical envelopes of existing launch vehicles and the cargo bay, there is a strong trend toward the boxy body-stabilized structure. COMSAT's studies (see Executive Summary Vol. I) have evolved toward studies of rectangular boxes which make module exchange easier. Fortunately the round hole (cargo bay cross section) is large enough to accommodate the square or rectangular peg-like satellites envisioned.

The choice between Figure 24b (cylindrical sections) and Figure 24c (pie-shaped wedges) is somewhat dependent upon the type of satellite and the tug limitations. As cylindrical sections are dispensed, the tug/payload center of mass moves along only the X axis (assuming balanced sections). Dispensing wedges shifts the center of mass along the Y or Z axis, as well as the X axis, resulting in a more complex tug stabilization requirement.

Figure 25 shows a rectangular spacecraft to scale for the sectional dispensing approach. The 15-ft dimension has been

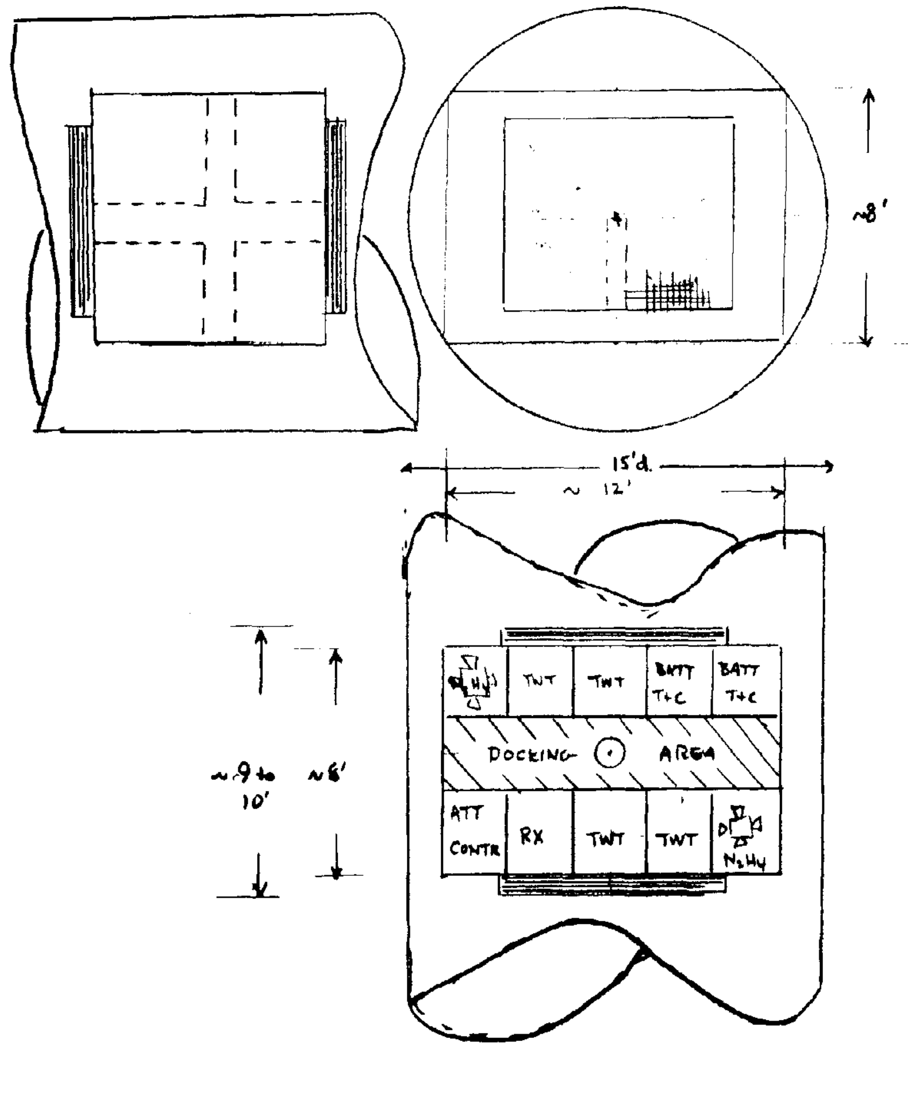


Figure 25. Rectangular Spacecraft in Shuttle, Solar Arrays Facing End

reduced to 12 ft to fit into the 15-ft-diameter envelope. Some of the lost radiator area may be regained by letting the maximum dimension be 15 ft and rounding the corners to form a chord with a 7.5-ft radius. Two or three such spacecraft may fit into the cargo bay.

Turning the spacecraft on end produces the configuration shown in Figure 26. The maximum dimension is 29 ft divided by the number of spacecraft. For two spacecraft this is about 14 ft (allowing for a separation section between the satellites).

UTILIZATION OF THE VOLUME

Each 15- × 8- × 8-ft satellite occupies about 1000 ft^3, or utilizes about 20 percent of the total volume. This suggests that a more optimum spacecraft configuration should be possible. It may be desired to increase the radiator and the module access areas. This would result in a satellite with a larger volume (but less density) and hence should reduce the satellite cost (due to relaxed packaging and layout constraints).

DENSITY

Figure 27 shows the results of a study of spacecraft densities. It indicates that today's spinning satellites with body-mounted solar cells fall into the 0.05- to 0.1 g/cm^3 class. Thus, a 15- × 8- × 8-ft conventional satellite of this class would be expected to weigh between 3000 and 6000 lb. This is not the case, however.

The body-stabilized satellites of the 1970's generally fall closer to 0.13 gm/cm^3. The 15- × 8- × 8-ft satellite would weigh about 8000 lb. Since there is apparently a large amount of

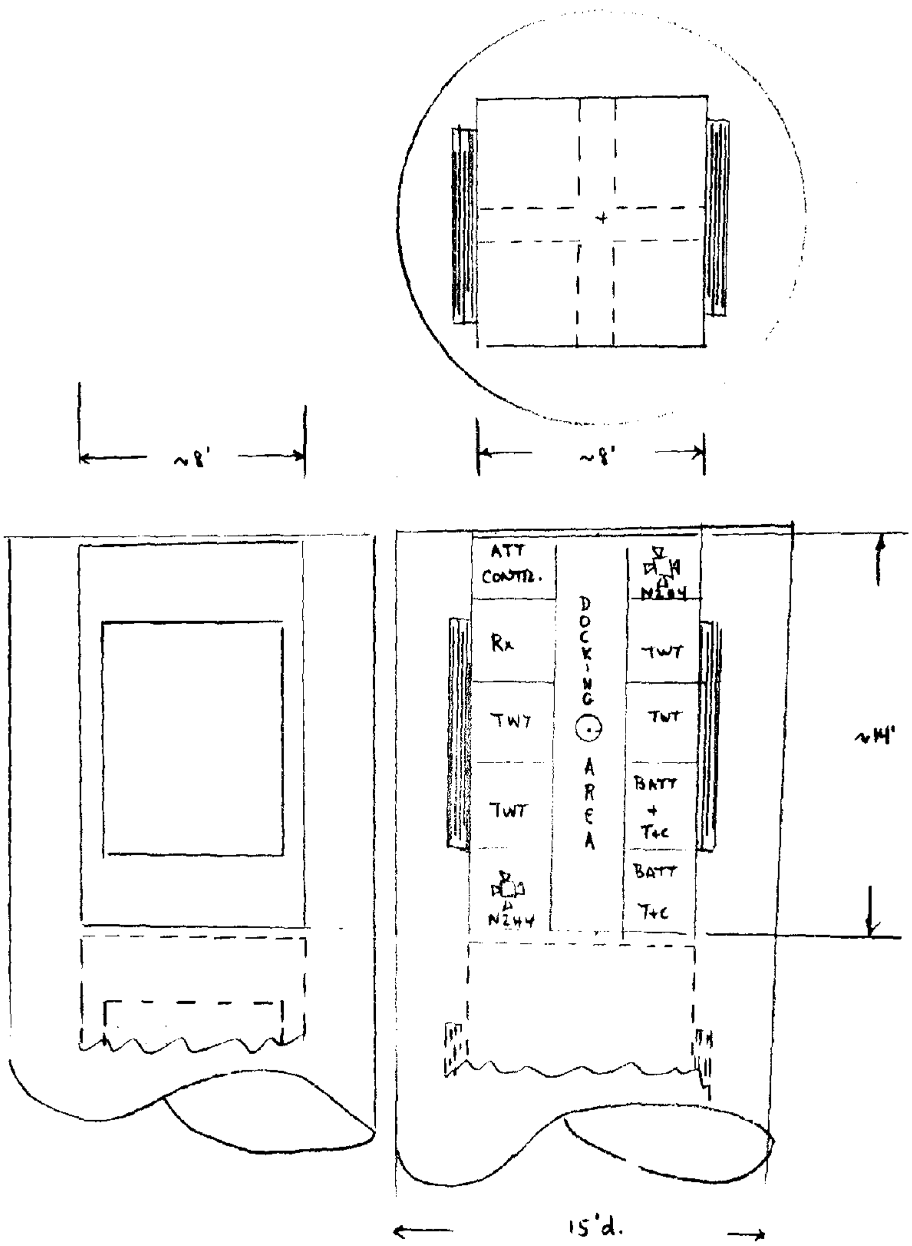


Figure 26. Rectangular Spacecraft in Shuttle, Solar Arrays Facing Sides

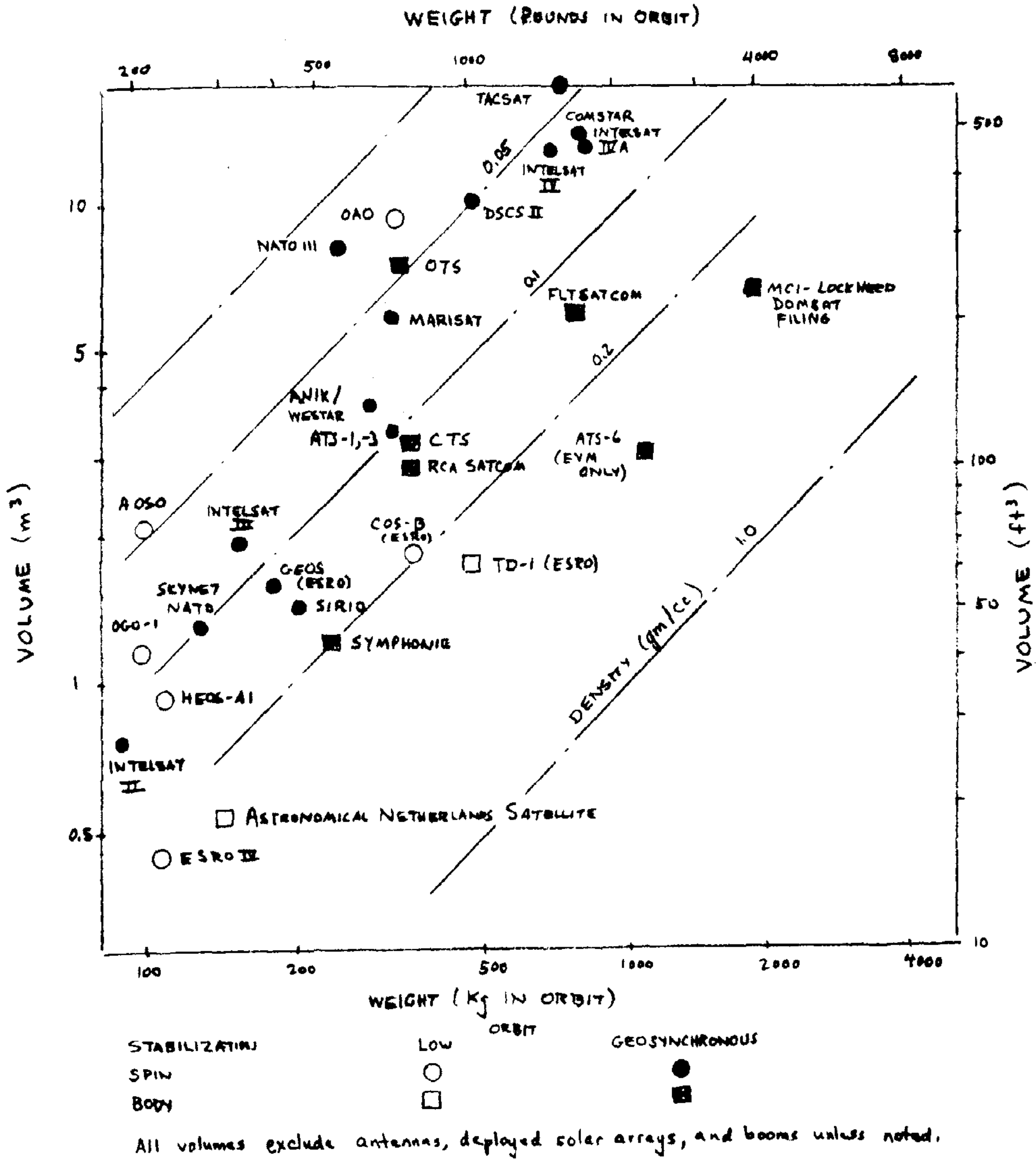


Figure 27. Spacecraft Densities

of space in the satellites of the shuttle/tug era, a much less dense design appears not only desirable but necessary to avoid excessive weight. This is not an argument for more lightweight materials, but rather an observation that the spacecraft equipment may be spread out over a large volume because the satellite is easier (less expensive) to build, operate (thermal), and service in orbit.

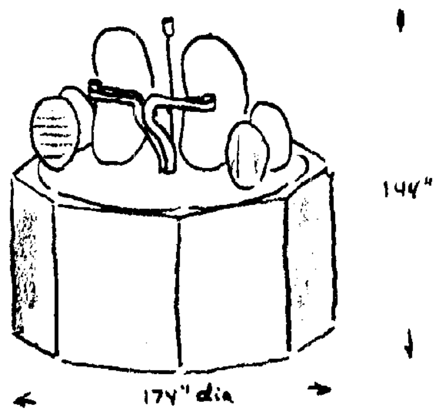
If the 15- × 8- × 8-ft satellite weighed 1500, 2000, or 2500 lb, the density would be 1.6, 2.1, or 2.6 lb/ft^3 (or 0.025, 0.033, and 0.041 gm/cc), respectively. This indicates that the densities may be one-quarter to one-half of contemporary values.

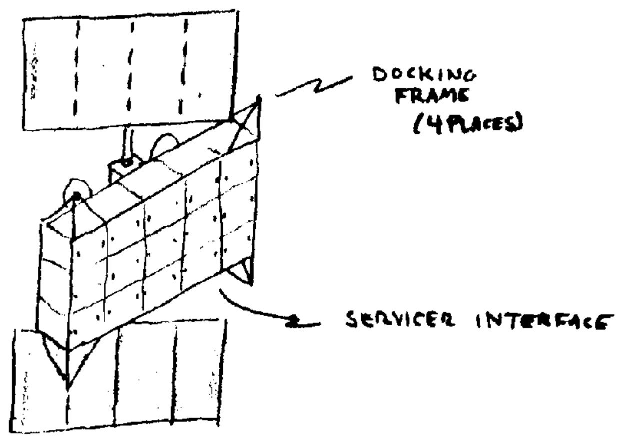
DATA FROM FAIRCHILD

For comparison purposes a 1974 EASCON paper from Fairchild is considered.[9] This paper describes two shuttle/tug satellites (see Figure 28). The first has a drum spinner with a 174-in. diameter and a height of about 6 ft with an assumed weight in the 2000 lb class. The other is a serviceable satellite for which few details have been given. Figure 29 attempts to scale the satellite on the basis of the scanty information available and to fit it into the 15-ft-diameter cargo bay.

Several observations may be made from the Fairchild study:

a. The densities are indeed lower than those of current practice (roughly 0.013 to 0.03 gm/cm^3) and in line with COMSAT's independent estimates.
b. About two spinning satellites fill up the available space (but not mass). The body-stabilized satellite appears to be very thin (a few feet). Seven to ten





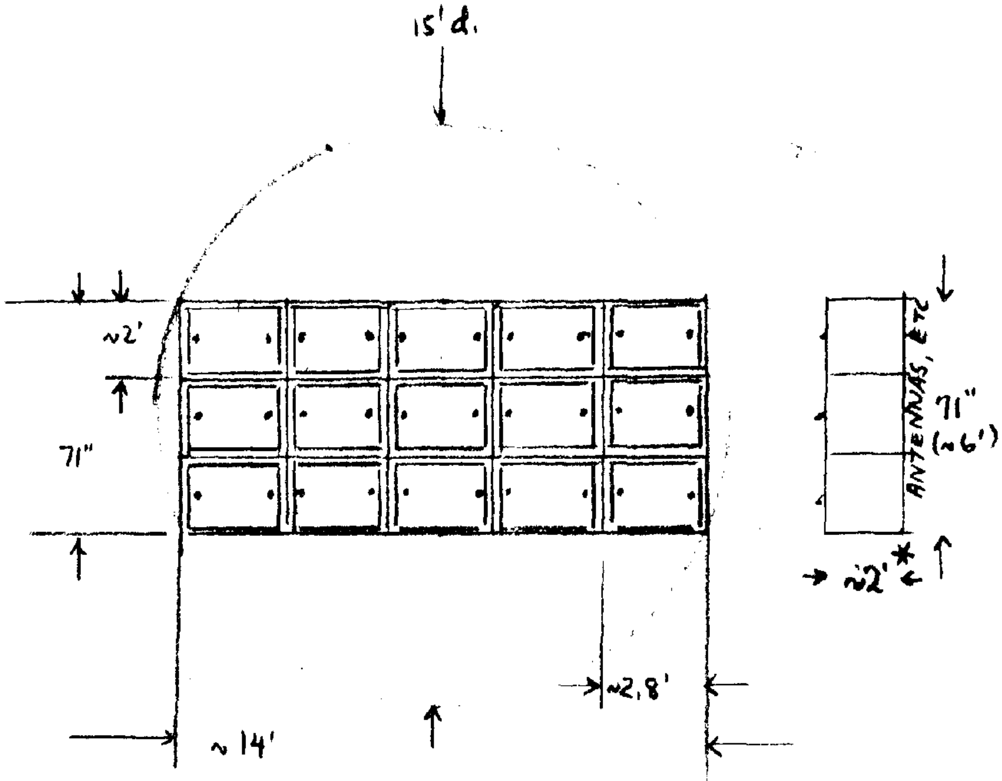
Module	Qty.	Mass (lb)	
		Each	Total
1. TT&C	2	39	78
2. RCS	2	160	320
3. ACS	2	79	158
4. Input MUX and TWTAs	4	58	232
5. Batt. and Pwr. Dist.	2	89	178
6. RX and Divider	2	33	66
7. Output MUX and Feeds	1	52	52
Module Total			1084 lb

Launch Configuration:

Diameter = 174 in
Length = 71 in
Weight = 1344 lb

1	4	5	4	3
2	6	7	6	2
3	4	5	4	1

Figure 28. Fairchild Shuttle-Launched Satellites[9]



$$6' \times 2' \times 14' = 168 \text{ ft}^3$$

$$\text{Density} = \frac{1344 \text{ lb}}{168 \text{ ft}^3} = 0.80 \text{ lb/ft}^3 = 0.013 \text{ gm/cm}^3$$

*May be substantially deeper than 2 ft when antennas, waveguides, backboard wiring, coupling, and structure are added.

Figure 29. Fairchild Serviceable Satellite in Cargo Bay[9]

spacecraft might be stacked within the cargo bay if the mass limit were not in the range of one to five (Table 11).

CONCLUSIONS

It appears very unlikely that the weight and volume constraints will occur simultaneously. The average packing density (Table 11) differs, by an order of magnitude in several cases, from current practice (see Figure 28). Figure 28 excludes the antenna farms which may currently demand a fairing volume as large as the rest of the satellite. Their inclusion will reduce actual densities by as much as a factor of two (but not ten) except possibly in the case of extremely complex, large reflector satellites which are not presently envisioned.

It may be preferable to make satellites thicker (and thus make the module "drawers" longer) to better utilize the available volume.

Because designers have been so long constrained by Centaur and Delta fairings, round satellites (or hexagonal approximations) topped with antenna farms are taken for granted. The shuttle/tug offers large cylindrical sections (or portions thereof, as shown in Figure 24).

The large lifting capacity of the tug suggests multiple satellites (Table 11). Since it is unlikely that one user (e.g., COMSAT, INTELSAT, Western Union, or NASA) will always want to use the full capacity of the tug, some form of standardization among the users is likely to evolve. This will take the form of common separation/attach fittings, incremental volumes, and eventually means for on-orbit servicing (even though the modules may not be electrically interchangeable because of mission-unique requirements).

Provision must be made for attaching the satellites to the tug. This may be done by stacking them one on top of the other (as in some of the military satellites), side by side (DSCS-II), or in a dispensing fixture (IDCSP). With solar panels and thermal radiators demanding the north and south surfaces and antennas the earth-viewing face, the only remaining face pair is the east and west surfaces. East/west panels could be used both for stacking satellites (enroute to orbit) and for docking (in-orbit servicing). This argument favors the configuration shown in Figures 24 and 25 over the Fairchild approach[9] if Figure 24b is used. A single attach fitting (on the east, west, or possibly the anti-earth panel) would favor the Fairchild concept using Figure 24c.

VII. GLOSSARY FOR ON-ORBIT SERVICING

This glossary has been prepared for use with geostationary satellites and their communications systems. Many of the terms are unique to these applications. In instances in which a term may be defined in several ways, depending upon the satellite orbit or application, the geostationary communications satellite usage has been emphasized.

AKM	See apogee kick motor.
Agena	An upper stage propulsion unit.
AM	Amplitude modulation.
Antiearth	Surface or direction away from the earth's surface.
Antisolar	Surface or direction away from the sun, as on the dark side of the satellite or of the earth.
Apogee kick motor (AKM)	A rocket used to convert from the elliptical transfer orbit into the circular orbit. This motor is fired at or near apogee and is used for plane changing and/or orbit circularization by raising the perigee to equal the apogee altitude.
Attitude	The orientation of the spacecraft, which may be expressed in terms of pitch, roll, and yaw (or i, b, or n).
Axis	A straight line about which a body rotates, or one of a set of reference lines for a coordinate system.
Body-stabilized satellite	Generally a satellite which is stabilized by reaction wheels, momentum wheels, or jets.
Centaur	The name of a second stage used for launching.

Characteristic velocity	The energy required to overcome gravity and place an object into orbit; see also orbital velocity.
Channel	See half-circuit.
Circuit	A 2-way telecommunications loop consisting of two channels (also called two half-circuits). The ends of the loop are at terrestrial locations.
Cross strap	A form of switched redundancy in which the outputs of two (or more) devices may be switched to the inputs of two (or more) subsequent devices.
Delta	Name of a launch vehicle, also called Thor-Delta; more specifically, the third stage of the Thor-Delta vehicle.
Disturbance torques	The effect of the earth's environment (magnetic field and gravity gradient), solar pressure, and internal rotating parts upon the spacecraft attitude.
Earth-lock	The process of locking the spacecraft axes to the local vertical and its maintenance.
Earth-stabilized satellite	A spacecraft which can keep one of its axes to the local vertical and its maintenance.
Earth station	A terminal on the earth for communication to a satellite or another body.
Earth synchronous orbit	An orbit whose period is the same as the earth's rotation time or 1436.1 minutes. The orbit may be circular ($h_a = h_p$ = 22,300 mi.).
Effective radiated power	Radiated power (e.r.p.). The product of the power supplied to an antenna and the antenna gain relative to a half-wave dipole (+2.15 dB), particularly in those frequency bands where a dipole or an array of dipoles is a

	useful antenna. Also the product of the power supplied to an antenna and the antenna gain relative to an isotropic radiator. (NOTE: The preferred term is e.i.r.p. See CCIR XIIth Plenary Session, Vol. IV, Part 2, p. 239, New Delhi, 1970.)
Equatorial orbit	An orbit in the plane of the earth's equator.
Equivalent isotropically radiated power (e.i.r.p.)	The product of the power supplied to the antenna for an emission and the antenna gain relative to an isotropic antenna.
FDMA	Frequency-division (or domain) multiple access.
FM	Frequency modulation.
Fuel	Propellant for a rocket or a satellite (including attitude and stationkeeping control).
Full Circuit	A communications system or equipment capable of simultaneous transmission in two directions.
Geostationary satellite	A geosynchronous satellite having an equatorial ($i = 0$) circular ($h_a = h_p$; $e = 0$) orbit with the same sidereal period as the earth (23 hr, 56.1 min) so that the satellite appears to remain fixed in relationship to the earth. See also stationary orbit.
Geosynchronous satellite	A satellite for which the mean sidereal period of revolution about the earth is equal to the sidereal period of rotation of the earth about its own axis (23 hr, 56.1 min). Also a satellite whose period is synchronous with the earth's rotational period. See also geostationary satellite.
Geosynchronous satellite orbit	The orbit of a geosynchronous satellite. This orbit has the following unique properties: P = 23 hr, 56.1 min. NOTE: The orbit may be elliptical and still be synchronous.

Half-circuit	A communications system capable of alternate transmission but not simultaneous transmission in two directions.
Hydrazine (N_2O_4)	A liquid propellant fuel used for spacecraft.
Nutation	Wobble of the satellite about its spin axis.
Occulation	The interruption of light upon one body (e.g., the spacecraft) by the intervention of another (e.g., the earth).
PCM	Pulse code modulation.
Pitch axis	The axis normal to the plane of the flywheel. When the spacecraft is properly aligned, the pitch axis and orbit normal are coaxial.
Pitch axis control	The control used to maintain spacecraft orientation about the pitch axis so that the principal axis and the local vertical are coaxial throughout the orbit.
Redundancy	The inclusion of an extra (spare) element available in a system for use in the event of a failure (or removal from service) of a similar element Also the portion of the total information contained in a message which can be eliminated without loss of essential information.
Spin stabilized	Applies to a satellite which is stabilized by spinning its body so that the axis of rotation remains pointing in a given direction.
Synchronous orbit	See earth synchronous orbit.
TWT, TWTA	Traveling wave tube and traveling wave tube amplifier, respectively.
Up-link	The earth-to-satellite link.

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