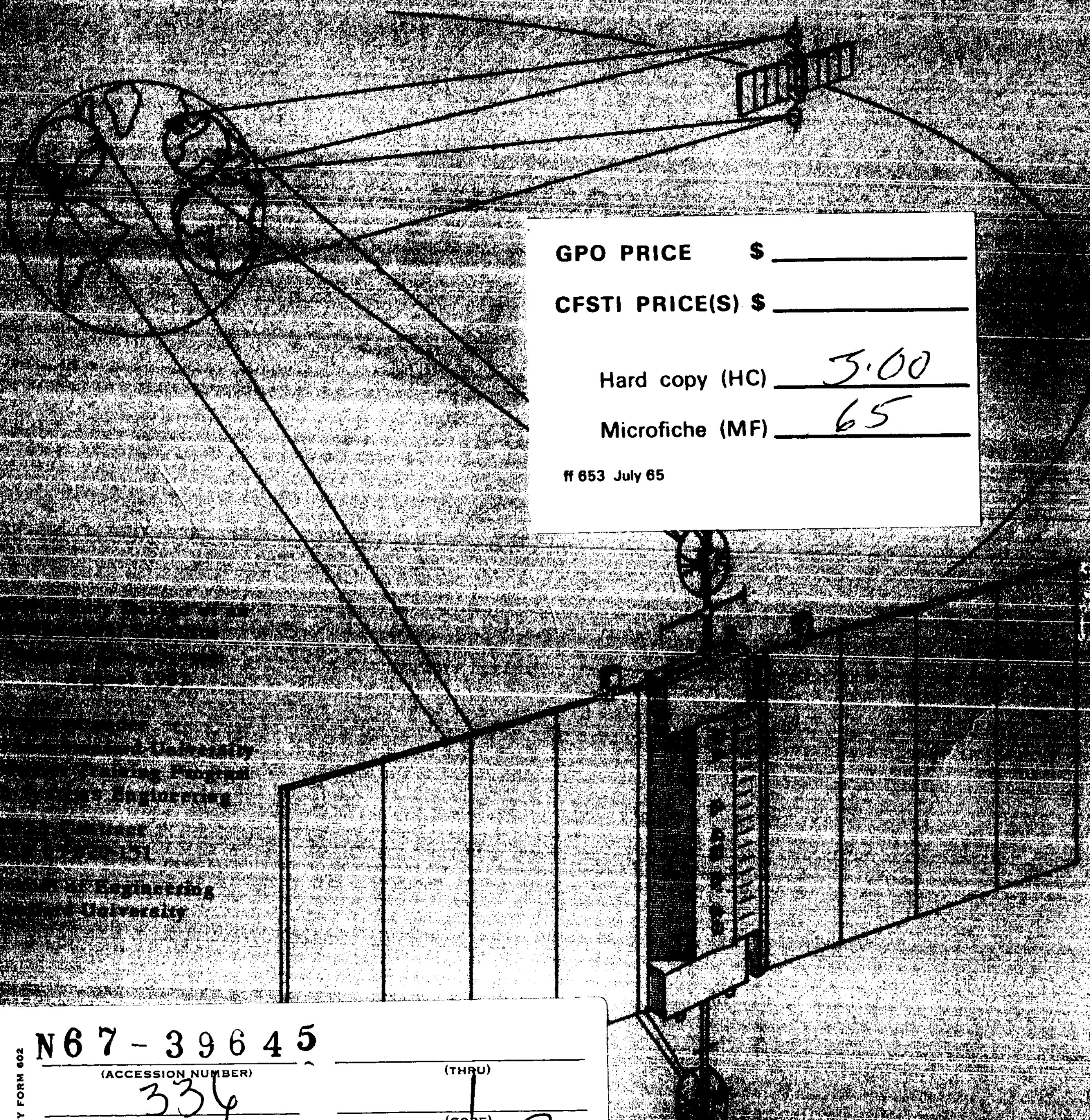

The SAINT Project

SATELLITE
ARRAY FOR
INTERNATIONAL AND
NATIONAL
TELECOMMUNICATIONS

NASA-STANFORD UNIVERSITY SUMMER
TRAINING PROGRAM IN SYSTEMS
ENGINEERING-AUGUST 1967

EUGENE V. SHAPARENKO - EDITOR

FOREWORD

This report presents the results of a preliminary design study undertaken by a group of thirty visiting professors at Stanford University as a part of the NASA-Stanford Summer Faculty Fellowship Program in Systems Engineering during the summer of 1967.

The purpose of the NASA Summer Fellowship Program was to acquaint the participants with the educational methods in space systems engineering used at Stanford in the hope that they would introduce similar techniques at their own universities. A second purpose was to acquaint the participants with NASA's activities in space technology and research, primarily to identify areas that are of importance to engineering students both in normal academic courses and in graduate research. The third purpose was the preliminary design of an optimum satellite system to meet world international telecommunications needs in the 1970s.

A project manager was chosen to oversee the overall design objective and to coordinate the three groups' activities. To allow more of the Fellows to participate in the direction of the project, the project manager and group leader responsibilities were rotated once during each of the three phases of the preliminary design.

The three project phases were as follows:

I June 19 - July 14 - Definition Phase, in which general background material was acquired and realistic alternative solutions were defined.

II July 17 - August 4 - Selection Phase, in which the best approaches were selected from the various alternatives.

III August 7 - August 24 - Completion Phase, in which detailed calculations were completed and the final presentation and the report were prepared.

A list of the groups, their members and group leaders, and their areas of responsibility follows.

Group 1

Primary and secondary satellite power systems; thermal design and integration.

Advisor: JAMES ADAMS

Fellows: ALLEN, C. W., Chico State College, Chico, California

*CHMIELEWSKI, W. S., West Virginia University, Morgantown, West Virginia

GILL, J. H., Western Michigan University, Kalamazoo, Michigan

HARDEN, D. G., University of Oklahoma, Norman, Oklahoma

*JANKOWSKI, F. J., Rutgers, New Brunswick, New Jersey

*MULLIKIN, H. F., Chico State College, Chico, California

**STEINHARDT, E. J., West Virginia University, Morgantown, West Virginia

YUEN, H. H., Villanova University, Villanova, Pennsylvania

Group 2

Launch vehicles and orbit analysis; attitude control and stabilization; physical configuration and structural design.

Advisors: JEAN MAYERS, WILLIAM BOLLAY, JAMES ADAMS

Fellows: AMER, SAAD, Chico State College, Chico, California

** *BOWEN, P. J., University College, London, Mullard Laboratory, Dorking, England

CASLER, M. S., University of Evansville, Evansville, Indiana

GONZALES, R. L., Bradley University, Peoria, Illinois

KAGAN, H. A., Rutgers, New Brunswick, New Jersey

LIU, Y. K., University of Michigan, Ann Arbor, Michigan

RED, D. D., University of Houston, Houston, Texas

RIGAS, A. L., University of Idaho, Moscow, Idaho

*SCHOENBERG, M. A., City College of New York, New York, New York

SHEWAN, W., Valparaiso University, Valparaiso, Indiana

** *SLATER, G. L., University of Cincinnati, Cincinnati, Ohio

VOLZ, C., Pennsylvania State University, University Park, Pennsylvania

SWIFT, W. A. C., University of Sheffield, Sheffield, England

Group 3

Ground and spacecraft communication systems and support subsystems; economic analysis of alternative ground/satellite systems.

Advisors: BRUCE B. LUSIGNAN, A. V. DA ROSA

Fellows: ADLER, R. W., Pennsylvania State University, University Park, Pennsylvania

*KIRK, D. E., Naval Postgraduate School, Monterey, California

KRAMER, F. W., Space Research Corp., Bad Godesberg, Germany

MEGGES, G., Institute for Space Systems Design, Oberpfaffenhofen, Germany

*MONSEES, L. R., South Dakota State University, Brookings, South Dakota

PERKS, N. W., Pennsylvania State University, University Park, Pennsylvania

SHAPARENKO, E. V., Purdue University, Lafayette, Indiana

VOCKE, M. C., Valparaiso University, Valparaiso, Indiana

*SCHERBA, M. B., Wayne State University, Detroit, Michigan

*Group Leader

**Project Manager

The course was conducted by Professors Bruce B. Lusignan and William Bollay, Stanford University.

The group appreciated the opportunity to visit several organizations concerned with various areas of the preliminary design. They are listed below.

Organization	Location
Electro-Optical Systems	Pasadena, California
General Electric Company	Pleasanton, California
Hughes Aircraft Company	El Segundo, California
Jet Propulsion Laboratory	Pasadena, California
Lockheed Missile and Space Company	Sunnyvale, California
NASA-Ames Research Center	Moffett Field, California
Stanford Linear Accelerator	Stanford, California
TRW Systems	Redondo Beach, California

During the course, several lectures by engineers and scientists in the aerospace industry were given on topics pertinent to the design objectives. A list of the speakers and their lecture topics appears below:

IMRE MOLNAR, Telecommunications Systems Research, San Francisco, California

D. HAVILAND, General Electric Company, Valley Forge, Pennsylvania, "Present and Future Telecommunications Systems and Plans"

J. V. FOSTER, NASA-Ames Research Center, Moffett Field, California, "Fund of Solar Supplies"

T. N. KOSTA, NASA-Ames Research Center, Moffett Field, California, "Synchronous Orbits"

DAN DEBRA, Stanford University, Stanford, California, "Fundamentals of Stabilization"

F. B. NELSON, Varian Associates, Palo Alto, California, "High Power Transmitters"

H. F. MATTHEWS, NASA-Ames Research Center, Moffett Field, California, "Program Management"

E. M. VAN VLECK, NASA-Ames Research Center, Moffett Field, California, "Cost Estimation" - In Mr. Van Vleck's absence, his notes were given by M. J. Zusman.

J. R. MULKERN, NASA-Ames Research Center, Moffett Field, California, "Reliability and Quality Assurance"

R. O. FIMMEL, NASA-Ames Research Center, Moffett Field, California, "On Board Data Handling"

J. E. GINGRICH, General Electric Company, Pleasanton, California, "Atomic Power Supplies"

F. KLAUSS, Lockheed Missile and Space Company, Sunnyvale, California, "Bearing and Slip Ring Application to Space Systems"

D. MAC QUIVEY, Stanford Research Institute, Menlo Park, California, "Present Telecommunications Systems and Plans"

The entire class is indebted to these people for the time and effort they spent in preparing and presenting these helpful lectures.

No report or book is written in a vacuum and it would not be practical to include the entire list of individuals outside the study group whose stimulating discussions and directive criticisms helped form this

report. Specific instances are cited in the text where appropriate.

The Editor wishes to thank Mary Egan and her faithful stenographic pool for their untiring efforts of typing and retyping the manuscript through its innumerable revisions. Mrs. Egan's coordination of the group's social functions was one of expertise and was sincerely appreciated by all. A special word of thanks goes to the Electronics Research Laboratories' drafting staff in helping to prepare the numerous figures and drawings for publication, and to Associate Editors W. A. C. Swift and D. G. Harden for their invaluable assistance. Cover design is by P. J. Bowen and E. V. Shaparenko.

Lastly, to our untiring project coordinators, Professor Bruce B. Lusignan and Professor William Bollay, and to the National Aeronautics and Space Administration, we owe an unrepayable debt for stimulation and opportunity.

E. V. Shaparenko

PREFACE

The primary requirement for a telecommunications satellite system is that it must satisfy existing communication requirements more economically than alternative systems with service of comparable quality. Such a satellite system requires the design of several interacting subsystems. The basic philosophy used in the preliminary design described in the following chapters was to enumerate alternative subsystems, and then to evaluate them using as criteria:

(1) the performance of the subsystem;

(2) the effect of the subsystem on the overall satellite and ground system; and

(3) the impact of the subsystem on other system configurations such as structures, power, communications, etc.

The present channel capacity available for international communication amounts to between three and four thousand equivalent two-way 4-kc circuits. These circuits are mainly in the form of submarine cables, the communications satellites in orbit at this time contributing less than 5 percent of the total capacity. The predicted number of circuits required by 1978, based on surveys by the American Telephone and Telegraph Company and Stanford Research Institute, is between ten and fifteen thousand; this figure does not include regional requirements within the United States or European areas.

The SAINT proposal is for an economical system of synchronous orbit satellites capable of handling the predicted world telecommunications needs in the late 1970s. The system consists of four satellites designed for international telephone and television communication, and four regional satellites covering the United States and Europe. The capacity of the international satellites has been oversized beyond the 1978 requirement by some 30 percent to take into account a possible increase in traffic due to a reduction in rates. It is planned to launch the international satellites in 1973, with a design life of five years. Two regional satellite launches are described, one for 1973 and an advanced regional orbiting network for 1978-1983.

The satellites will incorporate projected 1970 technology. They all require 3-kW input power, and are of the same basic design, except for antenna and associated communications electronics details. The communications frequencies proposed are mainly a 6-GHz uplink and 4-GHz downlink with regional satellites utilizing some additional spectra in the 4-10 GHz area. Single sideband modulation is used on the uplink, frequency division FM on the downlink.

Multiple access techniques will be used from the ground stations through to the satellite link. An economic survey has shown that it is preferable to use a moderately large number of stations with small ground antennae.

In view of the ten-week duration of the program, the selection of suitable boosters and the design of the actual spacecraft had to be commenced before the communications requirement was fully defined. The communications design called for an input power of 3-4 kW, and it was thought initially that multiple arrays of up to nine antennae per satellite would be required. A survey of power supplies suitable for use in space showed that only large area solar cell arrays would be sufficiently developed and tested by the early 1970s to be considered acceptable for the SAINT project. Assuming that the power output from a solar cell array is 8 W/ft^2 after five years, allowance being made in this figure for radiation damage, at least 400 ft^2 of solar panels are required. Also, the panel structure has to be oriented normal to the sun's direction for effective use of the solar cells. A conservative weight estimate based on the power requirement showed that each satellite might weigh up to 2000 lb, and the antenna array and solar panel requirements dictated the fully deployed geometry of the spacecraft.

In selecting a suitable booster, the cost effectiveness of launching four or more such spacecraft on a Titan III-Centaur, rather than individually launching them on Atlas-Centaur vehicles, soon became apparent. Because of the present limited reburn capability of the Centaur stage, additional injection rockets are proposed for use with both the above booster configurations. The final satellite design is based on a multiple launch of four spacecraft with a single Titan III-Centaur injection rocket booster, but single satellites of identical design could be launched on

Atlas-Centaur for replacement purposes at a later date. In the multiple launch mode, the folded spacecraft presents a highly efficient use of shroud volume, since each satellite requires over 400 ft^2 of solar panel area.

Active stabilization is employed to point the two linked antennae sections toward the earth, while the solar panels are oriented normal to the solar vector. Antenna pointing to $\pm 0.1^{\circ}$ arc is achieved by using a Polaris star tracker in conjuction with a radio interferometer and infrared earth sensors, although for some missions a lower accuracy compatible with using only the infrared sensors may be acceptable. The symmetrical antenna assembly design was adopted to minimize disturbance torques due to solar radiation pressure. Cesium ion thrusters with electrostatic jet deflection are used in conjunction with reaction wheels to provide attitude control torques. The antenna sections are rotated relative to the solar panels by means of a dc motor, and power and data are transferred across slip-rings.

It is planned to separate the four spacecraft simultaneously from the injection rocket assembly after injection into synchronous orbit. The solar panels and antenna masts are then deployed, and the spacecraft is oriented before a velocity increment is applied by means of hot gas thrusters. The satellites then "drift" in a near synchronous orbit to their orbital positions when a further velocity increment is applied to transfer the individual spacecraft to synchronous orbit. The four international satellites can be placed in their correct orbital positions in about twenty days by this technique. After the initial positioning, station-keeping is carried out by using the ion thrusters on the spacecraft. A position accuracy of $\pm 0.1^{\circ}$ arc relative to a ground observer is required. The structural design of the spacecraft incorporates extensive use of a high strength, light weight honeycomb sandwich construction exhibiting extreme rigidity. From a preliminary analysis, the structure weight is less than 17 percent of the estimated total satellite weight of 1250 pounds. Over 2 W of generated power per pound of spacecraft are obtained, and by using beryllium aluminum honeycomb sandwich substrate for supporting the solar cells, an output of 20 W per pound of panel is achieved.

In addition to the solar cells, nickel-cadmium batteries are carried in the satellite to give full operational capabilities when the satellite is in the earth's shadow. The spacecraft thermal design is a semi-active system with louvers being used to control the satellite temperature during the launch phase and while it is in shadow.

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SECTION I

THE SATELLITE SYSTEMS AND COMMUNICATION SUBSYSTEMS

INTRODUCTION

The six chapters devoted to the communications system contain descriptions of the alternatives that were investigated and core studies that were used to evaluate each of these alternatives. No attempt was made to assign a figure of merit to each of the configurations investigated; however, the system decisions were not made arbitrarily, but rather by objective evaluation of the information available on each of the alternatives. Refer to Fig. 1.

In Chapter 1, estimates of international telecommunication requirements for the year 1975 are presented. Using these predictions, a system of four synchronous satellites in equatorial orbit was designed to satisfy international needs (excluding traffic among European states). Because these four satellites primarily service international traffic, they are referred to as the "International System." The features of the international system design are uniformity of required satellite capacities, minimization of communications requiring excessive time delays, expandability of the system to satisfy increased traffic requirements, and the minimization of ground station costs in terms of the number of antennae required at each ground station. Each of the satellites provides telephone, television, and data transmission of various types.

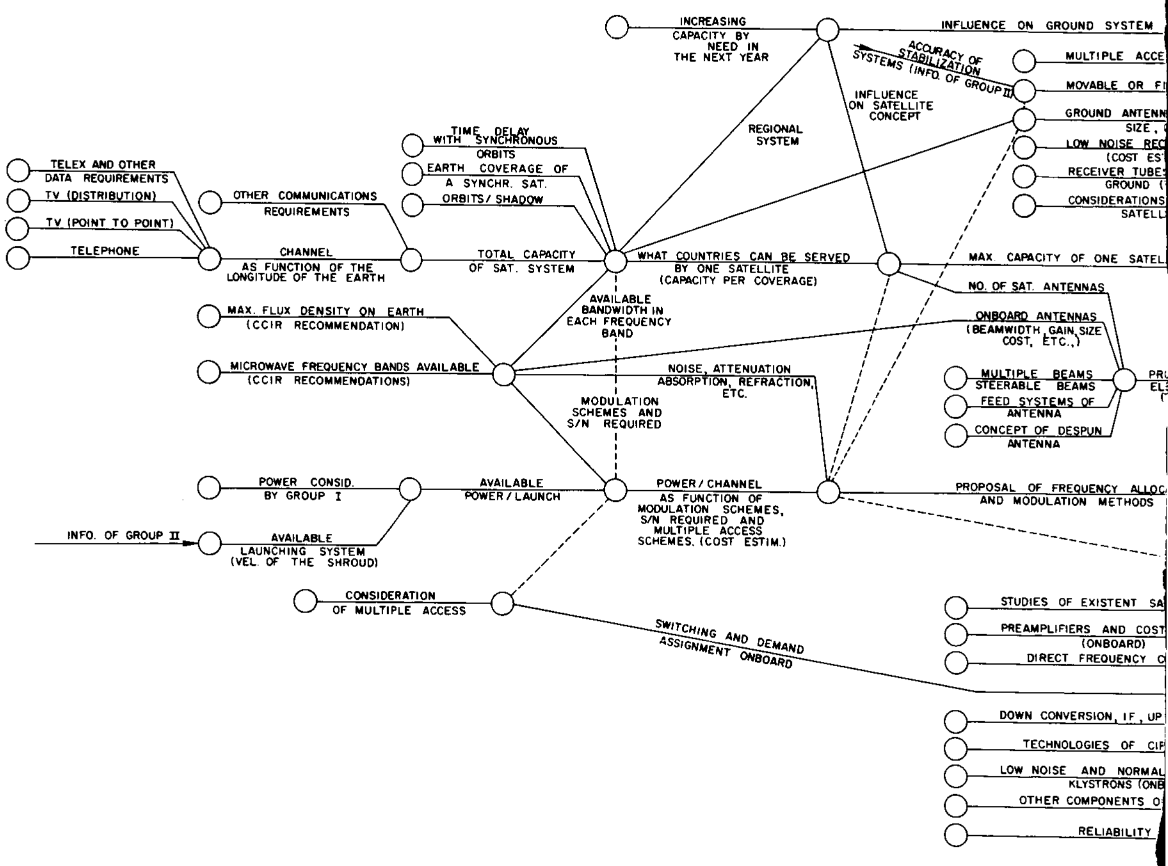
A complementary system of satellites to service telephone, television, and data traffic within the United States and within Europe is described in Chapter 2. This regional system consists of four synchronous equatorial satellites, two of which satisfy U.S. needs (one for TV exclusively, one for telephone and data), and an identical pair to provide for European traffic requirements.

In Chapter 3, the satellite communications subsystem is described for both the international and regional systems. Performance calculations for the single side band (SSB) uplink and the frequency modulated (FM) downlink are given; factors such as utilization of the available frequency spectrum, the effects of oxygen and water vapor absorption, and path losses due to heavy cloud and thunderstorms are also considered. The antennae configurations for each of the satellites are also described, and studies are presented that indicate the physical separation required of beams in order to be able to reuse frequencies. The section on transponders gives a description of the basic requirements, block diagrams of the transponders, and a typical interconnection diagram.

Chapters 4, 5, and 6 describe the design of the ground systems. In these chapters, the emphasis is on developing design procedures that can be applied to a variety of situations. Chapter 4 is devoted to the economic analysis of the ground stations, including the impact of the ground system parameters on satellite power requirements. The cost analysis is performed for two population density models: a uniform distribution and a population model with pockets of high density separated by large areas of sparse population. Costs included in the economic studies are for land, buildings, antennae, microwave links or cable connections to local telephone exchanges, cooled and uncooled preamplifiers, and channelizing equipment. The procedures developed can be used to determine the optimum number of ground stations for each of the satellite systems and the satellite power required. An economic analysis was also performed for a model that assumes that political factors require each country to have at least one ground station regardless of the economic feasibility.

In Chapter 5, four frequency division techniques for multiple access are described. A design procedure is proposed which provides fixed point-to-point frequency assignment between ground stations with high traffic densities, and fixed point-to-point frequency assignment with sharing of channel allocations among ground stations with low traffic densities.

Chapter 6 contains a block diagram of the ground station communication equipment. Several techniques (which are compatible with existing dialing procedures) for implementation of multiple access are presented. Consideration of factors that influence operation of a satellite while in the shadow of the earth and outages due to "blinding" of ground antennae by the sun are also included.


INCREASING CAPACITY BY NEED IN THE NEXT YEAR
INFLUENCE ON GROUND SYSTEM
ACCURACY OF STABILIZATION SYSTEMS (INFO. OF GROUP II)
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MOVABLE OR FI
INFLUENCE ON SATELLITE CONCEPT
REGIONAL SYSTEM
GROUND ANTENN
SIZE,
LOW NOISE REC
(COST ES
RECEIVER TUBE
GROUND
CONSIDERATIONS
SATELL
TIME DELAY WITH SYNCHRONOUS ORBITS
EARTH COVERAGE OF A SYNCHR. SAT.
ORBITS/SHADOW
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TV (POINT TO POINT)
TELEPHONE
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MICROWAVE FREQUENCY BANDS AVAILABLE (CCIR RECOMMENDATIONS)
NOISE, ATTENUATION ABSORPTION, REFRACTION, ETC.
MULTIPLE BEAMS STEERABLE BEAMS
FEED SYSTEMS OF ANTENNA
CONCEPT OF DESPUN ANTENNA
MODULATION SCHEMES AND S/N REQUIRED
POWER CONSID. BY GROUP I
AVAILABLE POWER/LAUNCH
POWER/CHANNEL AS FUNCTION OF MODULATION SCHEMES, S/N REQUIRED AND MULTIPLE ACCESS SCHEMES. (COST ESTIM.)
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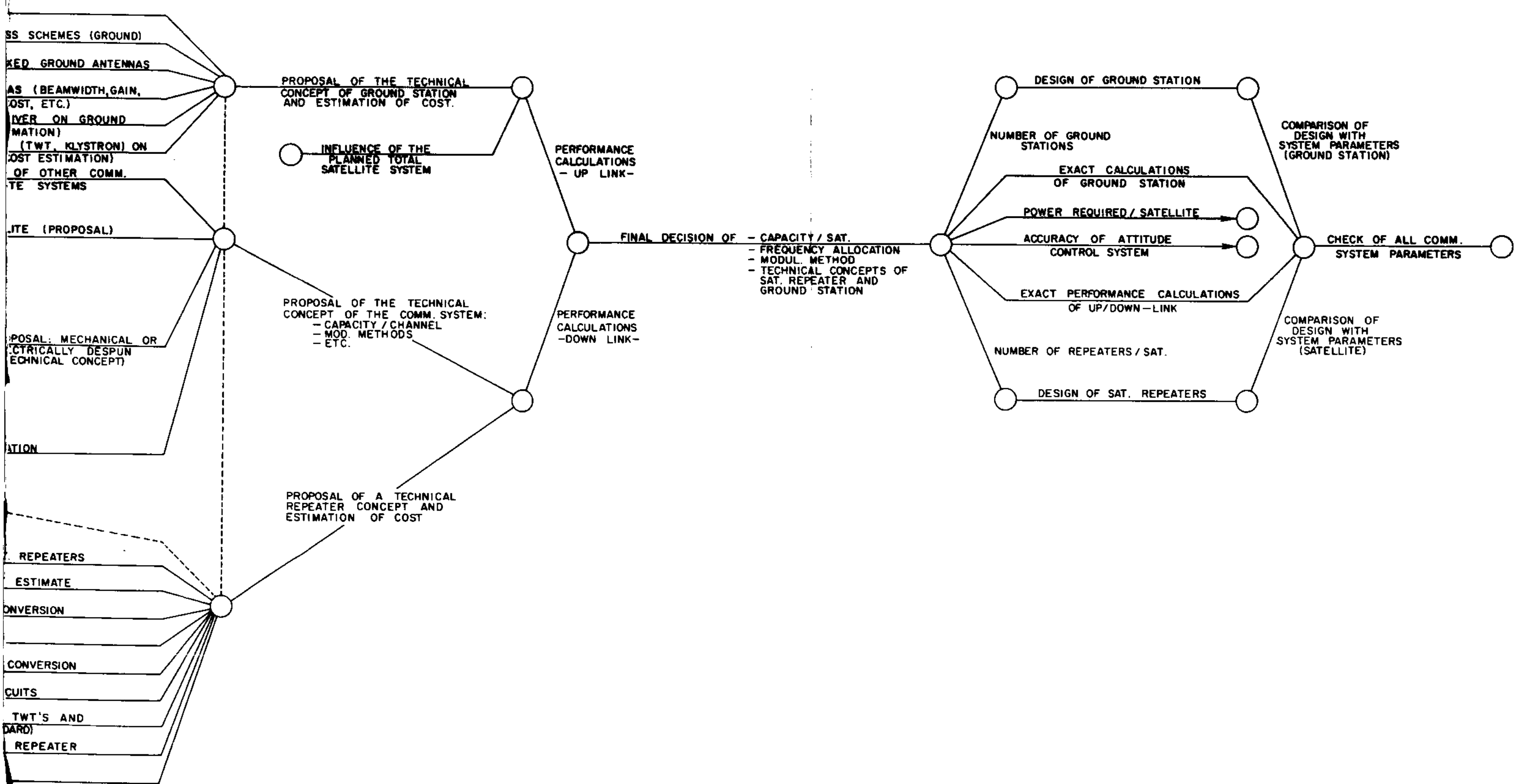

Fig. 1 ACTIVITIES AND DECISIONS OF GROUP III.

Chapter 1

THE INTERNATIONAL SYSTEM

1.1 Future International Telecommunication Requirements

The international telecommunications satellite network has been designed as a common carrier for the following types of services:

(1) two-way telephone;

(2) one-way telegraph, TELEX, and television;

(3) commercial and government leased telegraph, voice and alternate voice-data (AVD);

(4) wideband data.

Regional and international telecommunications forecasts for this study were derived from the following sources:

(1) Stanford Research Institute;[1]

(2) CCITT/CCIR Proceedings, Rome, 1963;[2]

(3) American Telephone and Telegraph;[4]

(4) Page Communications Engineers, Inc.[3]

The number of equivalent two-way voice circuits are enumerated in Section 1.5 on a regional basis according to the particular satellite furnishing the coverage.

The effects of any traffic increases, above and beyond normal international growth factors, due to rate reductions introduced by the implementation of a global satellite network have not been introduced into the international channel requirements. However, the system does have the ability to expand should such a situation arise. The service that appears to present a problem is the telephone: For example, a 25 percent decrease in international rates could precipitate an additional 33 percent increase in the demand for this service in 1975.[1] The demand for international telegraph services decreases under a rate decrease because the competitive price of other available services appears more attractive than telegraph itself.

1.2 Definition of International Traffic

The traffic estimates in Section 1.5 were collected by geographic zones, since tabulation of data is facilitated by use of regional rather than national divisions. The format for this tabulation is that prepared by the Secretariat of the International Telephone and Telegraph Consultative Committee (CCITT). A map showing the zonal divisions is presented in Fig. 2.

International traffic is defined as that traffic which takes place:

(1) on an interzone basis, or

(2) on an intra-zone basis when more than one country comprises a geographic zone.

The international system must provide access capabilities to all countries of the world that have an established telecommunications network or anticipate a future network.

Since the European countries constitute zones 1-12 of the international scheme, international traffic is extremely unbalanced. Therefore, the international satellite system was designed to exclude the traffic present within zones 1 to 12. However, since the study was one of solving international communications requirements, the European communication problem was treated on a regional basis. Therefore, in the four-satellite international network, described in Section 1.3, only traffic between zones 1 through 12 and the rest of the world was defined as international for that system. The European regional satellite system is described in detail in Chapter 2. Furthermore, since European regional communication requirements parallel those of the contiguous 48 states, solving the European problem as a part of the gross international system would essentially solve the regional U.S. satellite communication system and vice versa. This system is also described in Chapter 2. Since more information was available for U.S. traffic than for European, the system was modeled around U.S. growth factors.

1.3 The Four-Satellite Synchronous Altitude International Network

The decision to propose a four-satellite international system was prompted by the following factors:

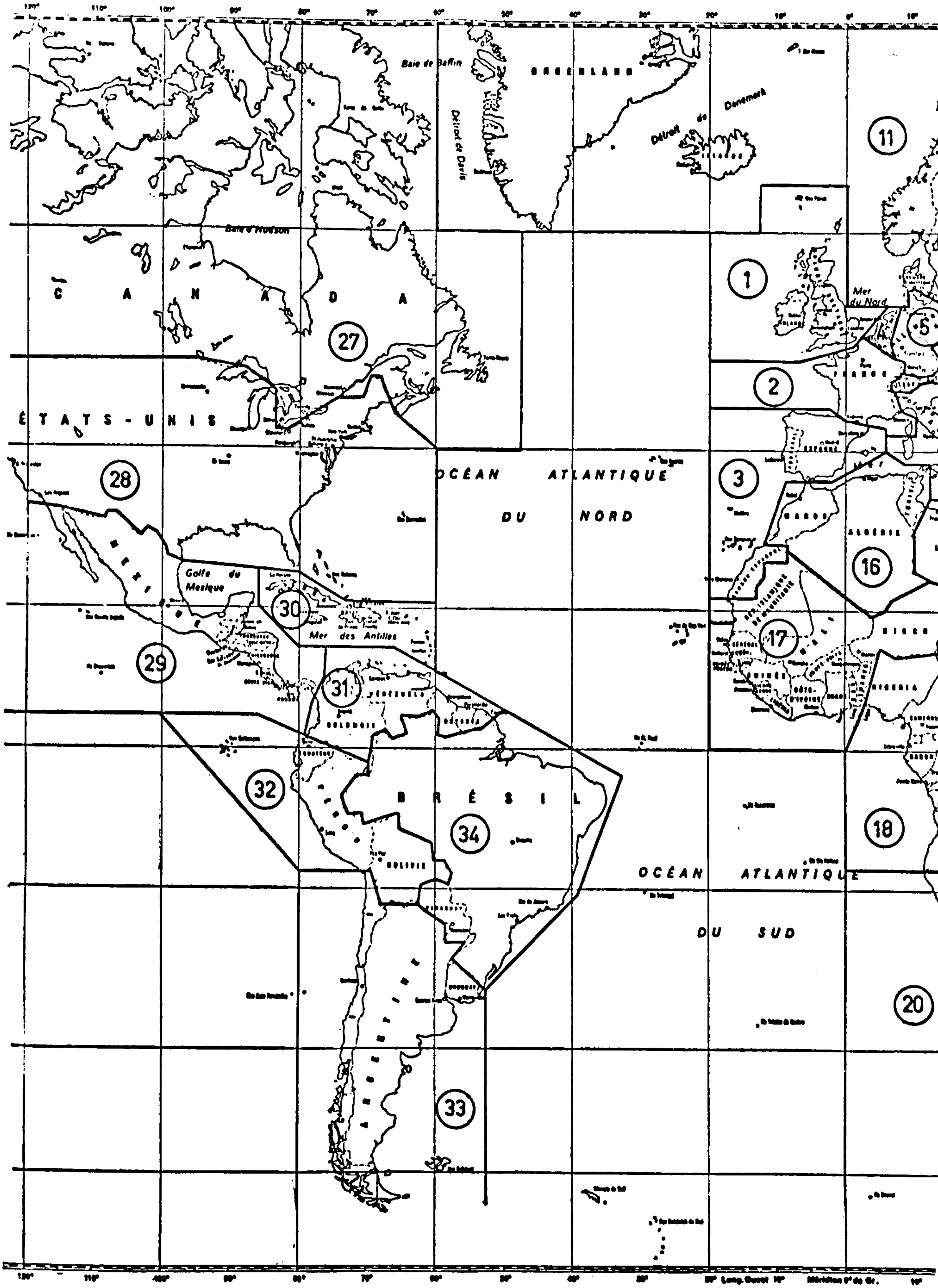
Baie de Baffin
Détroit de Davis
Détroit de Danemark
Baie d'Hudson
C A N A D A
ÉTATS-UNIS
Mer du Nord
OCÉAN ATLANTIQUE DU NORD
Golfe du Mexique
Mer des Antilles
B R É S I L
OCÉAN ATLANTIQUE DU SUD
BOLIVIE
1
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Fig. 2 INTERNATIONAL TELECOMMUNICATIONS ZONES.

(1) exploiting international load centers, which occur as a function of longitude;

(2) uniformity of satellite channel capacities;

(3) minimization of the number of channels requiring two-hop facilities;

(4) minimization of ground station costs in terms of the total number of antennae required for the ground system; and

(5) the capacity of the individual satellites to expand in order to accommodate normal and abnormal telecommunications growth within their lifetimes without exceeding power, bandwidth, and state-of-the-art limitations.

The equatorial positions of the four synchronous altitude satellites are shown in Fig. 3, together with their coverage areas as viewed from the North Pole. Three ground relay stations at San Francisco, Singapore, and Recife, Brazil, complement the system and provide links between ground stations that do not have a mutually visible satellite. The preference to provide a ground relay station as opposed to direct satellite-to-satellite communication links is discussed in Section 1.7.

1.4 Coverage

Approximately 98 percent of the earth's population is contained in a global band between 60° N and 40° S. Since this band also includes all of the largest cities, this is the area to be judiciously covered by communication satellite directive antenna patterns. At the expense of excluding certain geopgraphic areas with low population densities and very small international communication requirements, narrow beamwidth antenna patterns, centered on the high traffic density areas, were selected. This choice does not obviate the fact that the above mentioned low demand areas are entirely excluded from the international system; they may simply need more sensitive ground receiving systems to tie into the network than the high traffic density areas, because the antenna pattern contour at their locale presents a lower signal-to-noise level.

On the basis of this fact and the five criteria listed in Section 1.3, the four international satellites were allocated two earth-directed antenna each. The one-decibel contours for the satellite antennae are shown in Figs. 4-7.

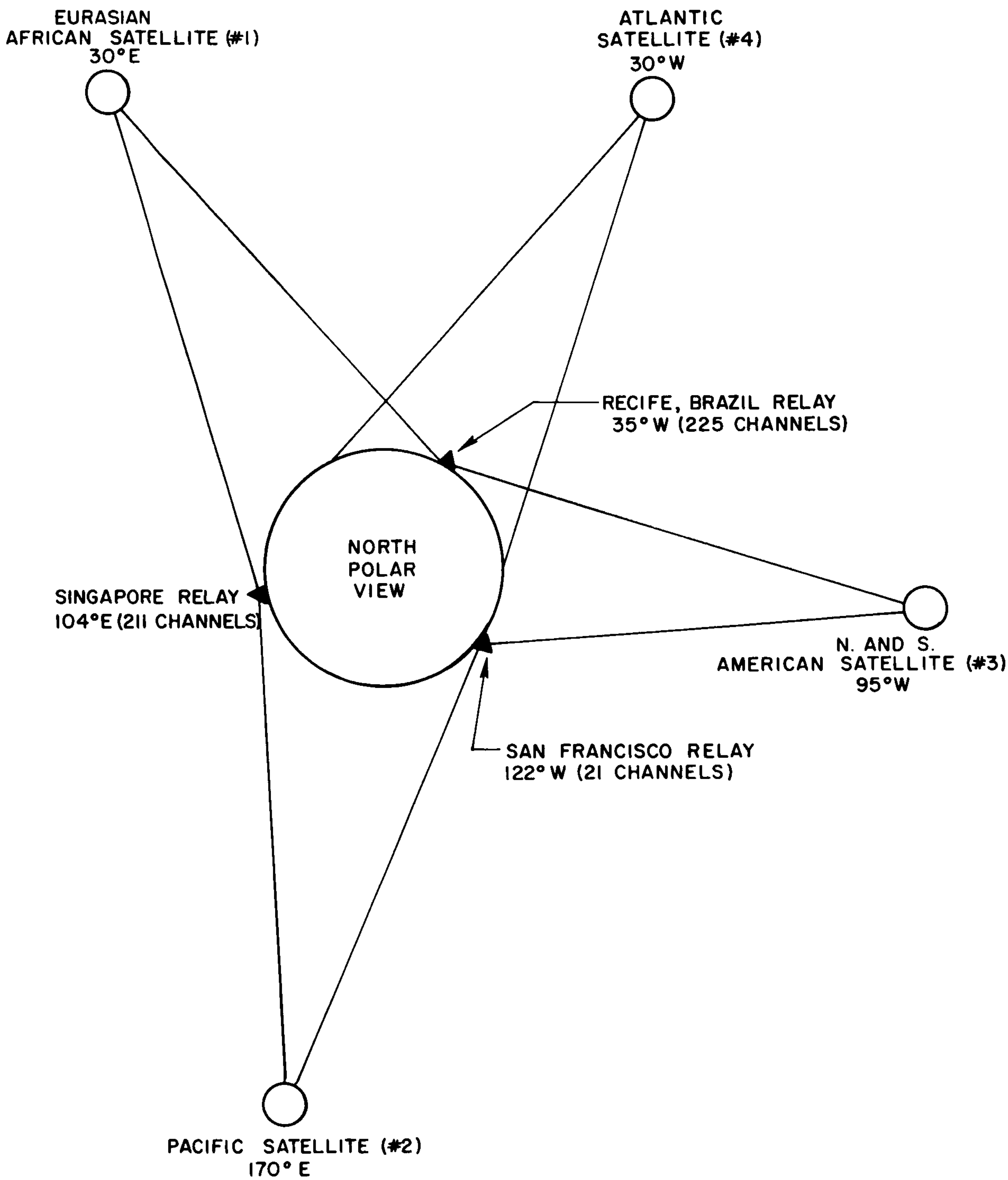


Fig. 3 EQUATORIAL POSITIONS OF FOUR SYNCHRONOUS ALTITUDE SATELLITES.

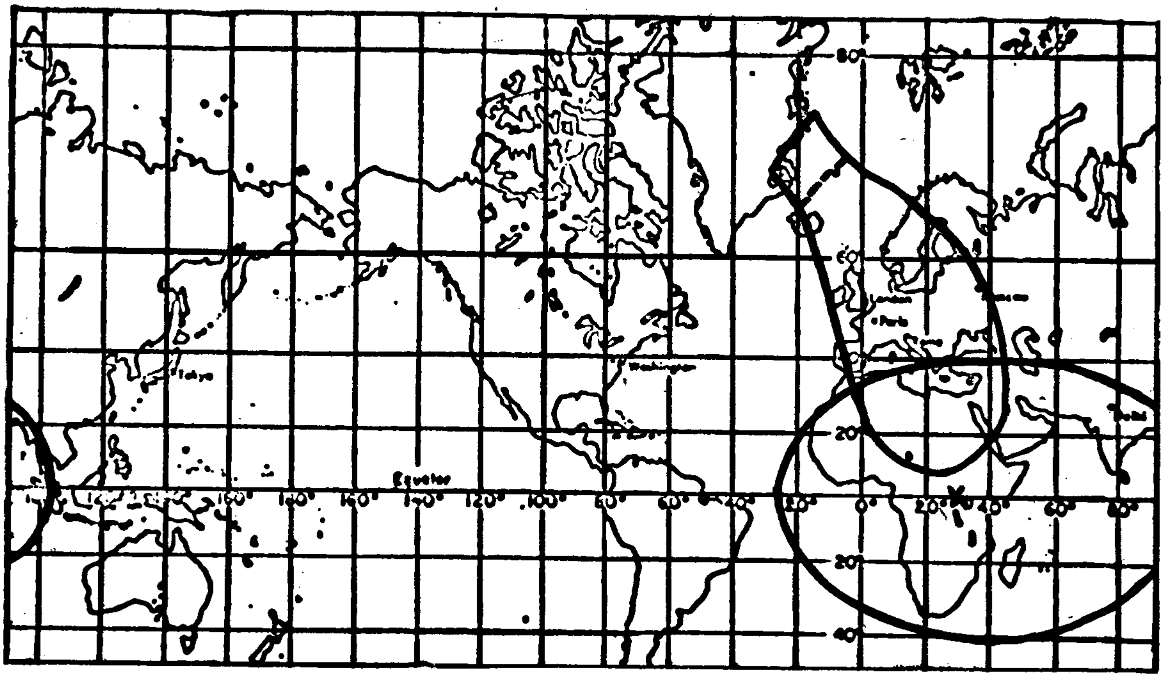


Fig. 4 EURASIAN AFRICAN SATELLITE COVERAGE - NO. 1 AT 30°E

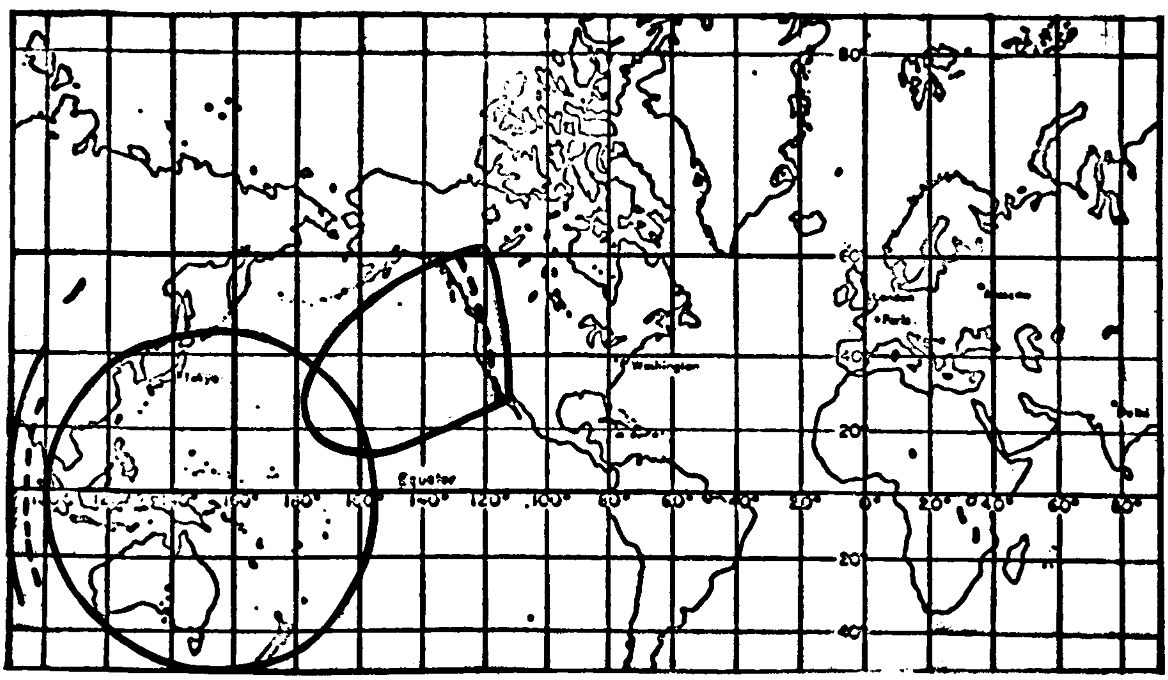


Fig. 5 PACIFIC SATELLITE COVERAGE - NO. 2 AT 170°E

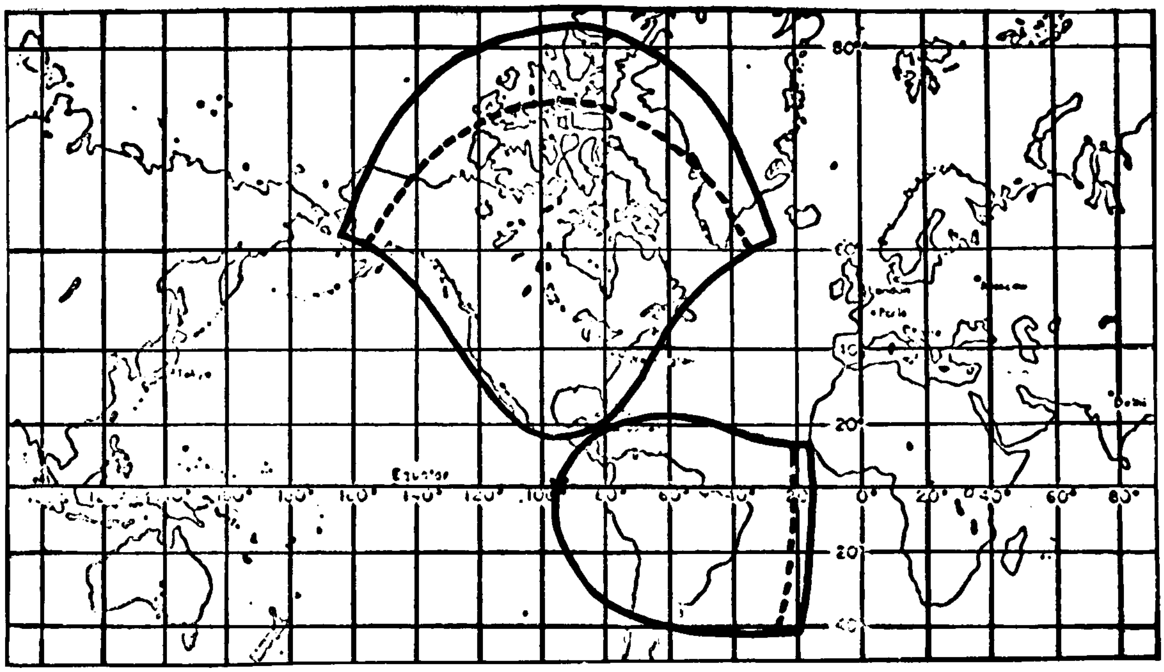


Fig. 6 NORTH AND SOUTH AMERICAN SATELLITE COVERAGE - NO. 3 AT 95°W.

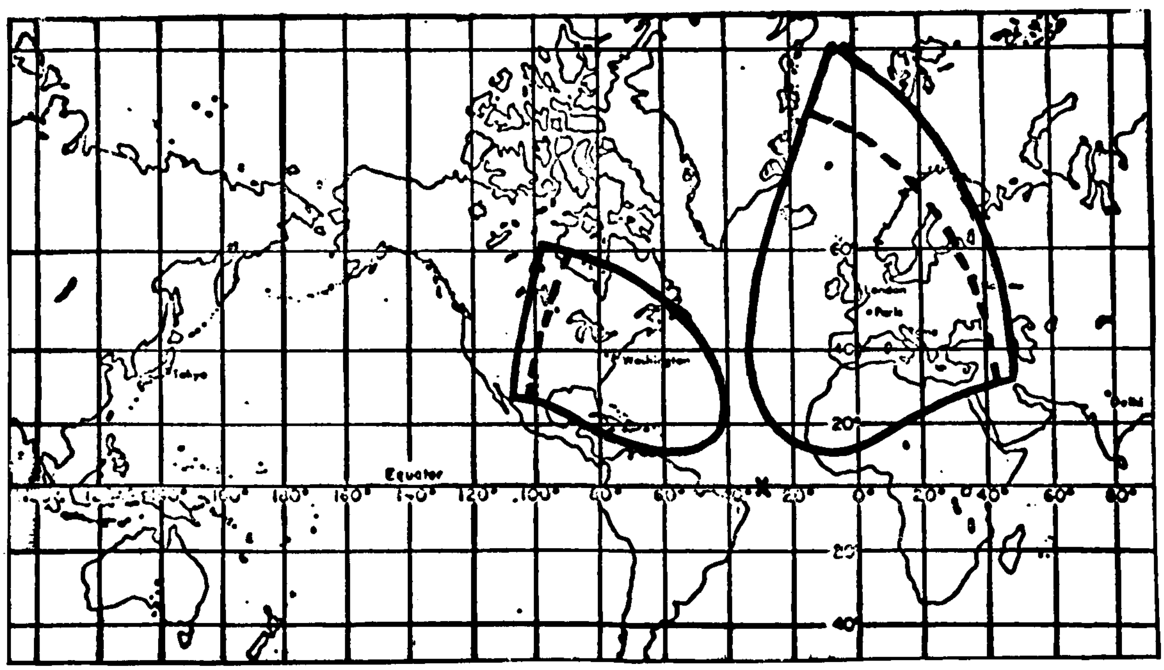


Fig. 7 ATLANTIC SATELLITE COVERAGE - NO. 4 AT 30°W

For any given satellite in the international system, beam overlapping does not present any reception problems since frequency diversification was not used on any of the satellites. Because of available bandwidth, modulation techniques, and channel capacities of individual satellites, there was no need to attempt reusing the spectrum on an international satellite.

For the international system (Figs. 4-7), the satellite antenna parameters are listed in Table 1.

Table 1

SATELLITE ANTENNA PARAMETERS

Satellite	Antenna Coverage	Beamwidth	Beam Pointing	Size*	Gain (dB)
1	Europe	7° × 7°	Offset 30° Azimuth 75°	18" × 18"	22.5
	Africa-Asia	11° × 16°	Offset 10° Due East	.875' × .625'	17.0
2	North America	6° × 6°	Offset 45° Azimuth 45°	21" × 21"	24.0
	Pacific and Far East	13° × 14°	Offset 10° Due West	.625' × .750'	16.75
3	North America	8° × 8°	Offset 45° Due North	15" × 15"	21.5
	South America	10° × 10°	Offset 30° Azimuth 15°	12" × 12"	14.5
4	North America	6° × 6°	Offset 45° Azimuth 45°	21" × 21"	24.0
	Europe	8° × 8°	Offset 45° Azimuth 60°	15" × 15"	21.5

*Elliptical or circular paraboloids

1.5 Satellite Channel Capacities

Satellite channel capacities were calculated on the basis of equivalent two-way 4-kHz voice circuits (4-wire circuits). The following conversion factors were used:

Voice circuit equivalents	Other services
1	4 telex
1	4 telegraph
1	22 teleprinter
12 one-way	1 broad band circuit
1200 (bandwidth) 1800 (power)	1 monochrome or color TV

The results of transposing all services to equivalent 4-kHz two-way circuits are shown in column 3 of Table 2. Columns 4 and 5 indicate the number of channels requiring ground relay to another satellite (east or west). Column 6 indicates the allocated international voice channels in the individual satellites under full load conditions. Thus, the difference in total channel allocations in column 6 and the estimated requirement in column 3 is the growth potential of that satellite beyond 1975 in handling international traffic.

Furthermore, once insuring that sufficient satellite capacity has been allocated for international telecommunications growth, the residual capacity of each satellite can be allocated to satisfy a portion of the regional (European and North American) telecommunications requirements. A listing of this residual capacity available per satellite which is allocated for regional traffic is also shown in Table 3.

Table 3 indicates that 19,600 one-way voice channels of the international system are available throughout the 1973-78 period for use in regional requirements. In Chapter 2, those 19,600 channels are integrated into the U.S. regional system.

1.6 Total Ground System Economy vs Two-Hop Requirements

In this study, a two-hop requirement was defined as a channel (two-way) between terminals which, in the four-satellite international system, did not have a mutually visible satellite. It is advantageous to expand on the term "visible." Although a person at a given ground station could "see" a given satellite, allocating a ground antenna at that terminal in order to make it visible to the ground station equipment may not be in the best economic interest.

Table 2
SATELLITE CHANNEL CAPACITIES

Column 1	2	3	4	5	6	7
From Zone (s)	To Zone (s)	Equivalent 4-kHz Two-Way Circuits (1975)	Relayed		1978 Satellite Allocations	
			East	West	Voice (Two-Way)	TV (One-Way)
		Eurasia-African Satellite (1)				
1-12	14-21	1378				
1-12	22	91				
16-20	16-20	200				
16-20	14,21,22	54				
22	23,26	36	X			
16-20	29-34	6		X		
16-20	27,28	55				
16-20	23-26	27	X			
1-12	23-26	148	X			
14	21,22	29				
1-12	29-34	128		X		
21	21	6				
29-34	22	4		X		
22	22	300				
27,28	22	32		X		
21	22	35				
					3600	3
Totals		2529	211	225	3600	3
Regional Provisions						
Europe					2700	
Africa-Asia					1200	
		Pacific Satellite (2)				
16-20	23-26	27		X		
1-12	23-26	148		X		
22	23-26	36		X		
23-26	29-34	21	X			
27	23-6	14				
28	Hawaii	700				
28	25	128				
28	26	190				
28	23,24	132				
28	23-26	1150 (leased)				
23-26	23-26	133				
26	26	52				
24-25	24-25	150				
23	23	20				
					4800	4
Totals		2901	21	211	4800	4
Regional Provisions						
Far East					1200	
North America					3000	
		North-South American Satellite (3)				
28	Alaska	240				
28	29-34	1243				
28	29-34	134 (leased)				
27	29-34	24				
31-34	31-34	411*				
31-34	30	50				
31-34	29	100				
29-34	23-26	21		X		
29-34	22	4	X			
16-20	29-34	6	X			
27-28	16-20	55	X			
27-28	22	32	X			
29-34	1-12	128	X			
					3600	4 TV
Totals		2448	225	21	3600	4 TV
Regional Provisions						
North America					1200	
South America					1800	
*See Reference 3						
		Atlantic Satellite (4)				
28	1-12,14	1606				
28	1-12,14	153 (leased)				
					4800	4
Totals		2759	0	0	4800	4
Regional Provisions						
Europe					3000	
North America					3000	

Table 3

SUMMARY OF INTERNATIONAL SATELLITE LOAD ALLOCATIONS

Satellite	Estimated 1975 Loading (2-way 4 kHz Ch)	Allocations (1978)		Regional Allocations		Total Voice Circuit Allocations
		Telephone	TV			
1	2,529	3,600	3	Europe 2,700	Africa 1,200	7,500
2	2,901	4,800	4	Pacific 1,200	USA 3,000	9,000
3	2,448	3,600	4	USA 1,200	S. Amer. 1,800	6,600
4	2,759	4,800	4	Europe 3,000	USA 5,400	13,200
TOTALS	10,637	16,800	15	19,500		36,300

TOTAL REGIONAL ALLOCATIONS FROM INTERNATIONAL SYSTEM:

USA - 9,800 two-way or 19,600 one-way
Europe - 5,700 two-way or 11,400 one-way

To minimize the total system cost, a certain portion of the international traffic was classified as two-hop. This amounted to $457/10637 \times 100$ or approximately 4-1/4 percent of the total international load. This 4-1/4 percent was then subjected to an additional relay, either via satellite-satellite relays or satellite-earth-satellite relays. The overall benefit of this additional delay was the following:

(1) All the following areas having satellite receive/transmit ground stations require only a single ground antenna: South America, Central America, Caribbean, Africa, Asia, Pacific, Alaska, Indian Ocean.

(2) These areas require a minimum of two ground antennae: continental United States, Canada, Europe, Near East (Zone 14).

This particular arrangement appears to have considerable merit for the following reasons:

(1) Those areas listed above in group (1) currently have the least developed terrestrial communication networks and would consequently be the candidates, cumulatively, for a much larger number of ground stations than group (2).

(2) Because areas in group (1) have more highly developed ground networks, fewer ground stations would be needed.

(3) Countries desiring admission to the international network, most likely within group (1), would not require more than one antenna to tie into the entire international system.

1.7 Two-Hop Delay and Ground Relays

Inspection of Fig. 8 reveals that the difference in path lengths between satellite-satellite and satellite-ground-satellite for two hops between satellites 1 and 2 and between 1 and 3, is not appreciable. Between satellites 2 and 3, there would be a marked difference in path length. However, the number of two-hop requirements between 2 and 3 is very low (21) and the separation in arc degrees is much smaller than the others. Round trip time delay is about 1.2 seconds for all two-hop traffic.

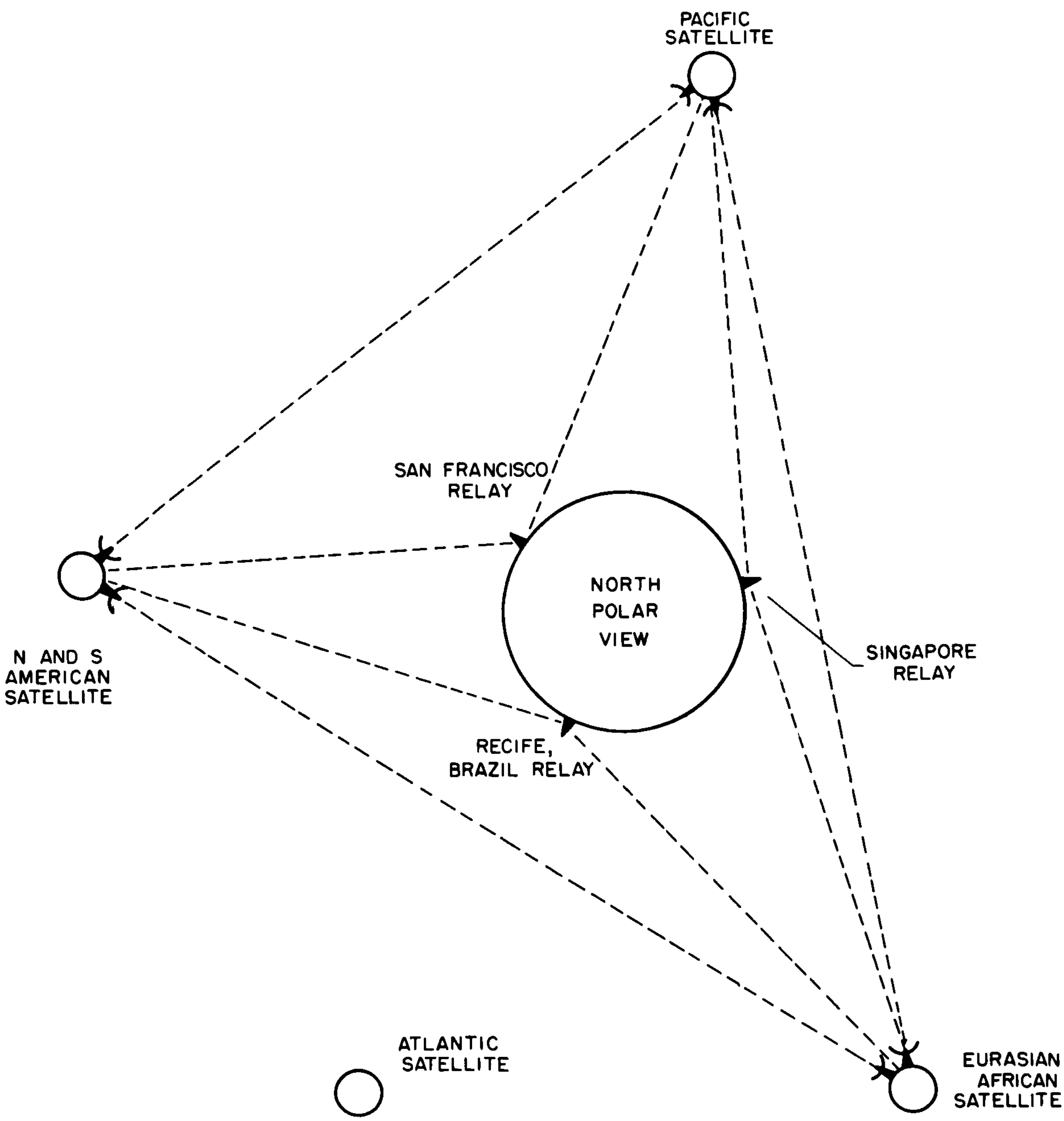


Fig. 8 PATH LENGTHS BETWEEN SATELLITE-SATELLITE AND SATELLITE-GROUND-SATELLITE.

Noting that the additional delay times, introduced by a ground relay in Fig. 8 are not appreciable, it becomes necessary to examine the economic trade-off of satellite-satellite relays and ground relays.

Since a ground relay station must communicate with at least two satellites, a minimum of two antennae and two transmitting and receiving systems is required for each station. Assuming that the chosen relay stations are a part of the international network initially, the total additional costs of providing three ground relay stations are summarized in column 1 of Table 4. Write-off time on the relay station is the same as that of ordinary ground stations--ten years. Choice of San Francisco as a relay station introduces no additional cost into the relay system since that city already communicates with the two pertinent satellites.

For line-of-sight relays directly between satellites, the four-foot unfurlable antennae that receive at 11 GHz and transmit at 18 GHz are needed for each satellite-satellite link. An average of 100 W dc is required on the communicating satellites for this link. Excluding costs of research and development for the higher frequency antennae and traveling wave tubes, optimistic figures for these items are \$150,000/antenna, \$50,000/TWT, and \$500/W dc power. The total cost of all satellite-satellite relays is summarized in column 2 of Table 4. Write-off time here is five years.

In addition to the lower cost/year of the ground relay system, the additional reliability of the ground relay equipment vs similar equipment in the satellites led to the choice of ground relays.

1.8 System Operation

This section describes the functional capabilities of each of the international satellites. Special characteristics of the traffic problems associated with each satellite and general ground station requirements are discussed.

1.8.1 Eurasian-African Satellite (1)

Traffic estimates show that this satellite will require 2529 equivalent two-way 4-kHz voice circuits in 1975. This includes all traffic originating in Europe and bound for Africa, South America, Asia, and the Pacific. Traffic bound for the United States is routed directly over the

Table 4

RELAY SYSTEM COSTS
(Thousands of Dollars)

Satellite-Ground-Satellite-Relay	Additional Costs	Satellite-Satellite Relay	Additional Costs
	Singapore (Satellite 1 to Satellite 2)		
Antenna	$200	Antenna (2)	$300
Transmitter	$150	TWT (2)	$100
Pre-Amplifier	$100	D.C. Power	$ 50
Broad-Band Amp.	$200		
	Recife, Brazil (Satellite 1 to Satellite 3)		
Antenna	$200	Antenna (2)	$300
Transmitter	$150	TWT (2)	$100
Pre-Amplifier	$100	D.C. Power	$ 50
Broad-Band Amp.	$200		
	San Francisco (Satellite 2 to Satellite 3)		
Antenna	- - -	Antenna (2)	$300
Transmitter	- - -	TWT (2)	$100
Pre-Amplifier	- - -	D.C. Power	$ 50
Broad-Band Amp.			
Total Ground Relay Cost	$1,300,000 or $130K/year	Total Satellite-satellite relay cost	$1,350,000 or $270K/year

Atlantic satellite (4). All internal African and Middle East traffic is handled on the larger beam (see Fig. 4). All traffic relayed east is picked up by Singapore; that going west, by the Recife, Brazil, relay. At synchronous altitude, approximately 163° of earth longitude is visible to the satellite so that the 139° separation between Singapore and Recife is sufficient for an average 13° visibility angle at the relay stations.

The 5° visibility contour of the European beam (dashed line) allows Iceland to participate in the international network. The eastern Mediterranean was given an additional antenna to monitor the Atlantic satellite, since a major traffic route is developing between the U.S. and the Near East.

Because of the small time differential between Europe and the other areas served by satellite 1, two one-way TV links are proposed for live distribution to Africa and the Indian Peninsula; one return link, from these areas to Europe.

1.8.2 Pacific Satellite (2)

The major portion of traffic on this satellite is east-west traffic from the U.S. mainland. The constraint of satellite visibility from the ground at Singapore allows only the west coast to monitor this satellite. If the traffic relayed from Europe and Africa via Singapore was channeled via microwave to a Manila relay, for example, much more of the mainland would be visible to the satellite, but traffic from Thailand and surrounding areas would have to be routed via satellite 1, thus creating more two-hop traffic from this area to Australia and points west. A vast area is open here for educational television, although the possibility of live broadcast is somewhat dubious because of time differential. Four one-way TV channels have been established to and from the U.S. mainland to cover the Pacific. Alaskan traffic is not routed via this satellite, although the erection of an additional antenna there would allow direct communication with the Pacific on the 3-dB contour of the $6^\circ \times 6^\circ$ U.S. beam. The San Francisco relay station would normally channel this traffic from the North American beam on satellite to the Pacific on a two-hop basis.

1.8.3 North and South American Satellite (3)

The original traffic estimates for this satellite in 1975 were 2448 two-way voice channels. Normal growth to 3600 voice channels in 1978 is foreseen. Live television broadcasts have great potential in this area. Two channels either way have been allocated for this purpose. Referring to Fig. 6, the Caribbean area can be served by whichever is most convenient in traffic loading. However, this must be determined prior to launch. The North American beam pattern for satellite 3 is similar to the proposed U.S. TV satellite coverage except that Alaska and Northern Canada are not included in the television beam.

1.8.4 Atlantic Satellite (4)

This satellite provides one-hop communication between a major portion of the United States, all of Europe, and the Near East. Two one-way television links are proposed, both east and west, on this satellite. No two-hop traffic is handled by satellite 4.

1.8.5 General Comments

The optimum satellite/ground system prescribed by the economic analysis of Chapter 4 coincides with the basic 1978 international telecommunications requirements. The additional flexibility of the international satellite system in handling regional and television requirements (Table 3) requires satellite power in excess of that indicated in Chapter 4.

1.9 International Back-Up Satellite (s)

A fortunate common denominator for all four of the international geostationary satellites is the frequency spectrum. This essentially means that, except for the unique directive antennae patterns from each satellite, any portion of the system of ground stations could use any one of the four satellites for telecommunications purposes. Hence, once the ground frequency allocations have been made, a 3-kW satellite with an earth coverage ($17^\circ \times 17^\circ$) antenna could serve as a direct replacement for any one of the existing four satellites. Thus, the two following alternatives for international telecommunications reliability are proposed:

(1) Satellite (s) having one earth coverage would be placed in a drifting, subsynchronous orbit in a fully deployed state. The difference in altitude of this satellite and a geostationary satellite should be such that on-board thrusters could place, on command, this back-up satellite into the proper longitudinal position. The power system should be operative for battery and communication subsystem thermal protection during occultation, and the telemetry-telecommand system should also be operative for tracking purposes. Since the proposed ground system does not allow full tracking capabilities, provision would be made for tracking via existing steerable antennae. A sacrifice of 3 dB on the earth coverage satellite antenna would have to be compensated for on the ground, if the basic communications and power systems were used on the back-up satellite (s).

(2) If the 3-dB sacrifice appears to be too large, the back-up satellite (s) could be placed on single or multiple launch vehicles on the ground, needing only tailor-made antennae. Reserving a special launch pad for such a purpose may prove too costly, whereas a 20- to 30-day preparation delay from satellite failure to actual launch may be prohibitive for continuity in telecommunications.

This back-up satellite could be fully operational on a time-shared basis when it operates in the drifting mode. Appendix C describes some scientific applications for such a satellite before and/or after it replaces one that might fail.

1.10 Other International Systems

This section enumerates and discusses the three other international satellite systems considered during the program.

1.10.1 Three-Satellite System

This particular system has the restriction that each ground station has only one antenna. The three satellites are

(1) Eurasian-Africa at 30°E,

(2) Pacific at 180°,

(3) North and South America at 97°W.

The satellite channel capacities, less TV and regional allocations, are approximately

(1) 5500 channels,

(2) 2900 channels,

(3) 7500 channels.

The total channel capacity of the system is larger than that of the proposed four-satellite network (Section 1.3) by about 50 percent because more traffic was now classified as two-hop. This amounts to about 33 percent. Furthermore, the 127°, 150°, and 53° satellite separations increase the (delay time × number of two-hop) product to an unreasonably high number. Although the ground system is now cheaper, the satellites are less uniform, more power is needed on the satellite itself, and the intersatellite communications power requirements are almost equal to the raw power allocated to the ground-directed antennae.

1.10.2 Four Satellites Plus Relay System

This system is nearly identical to the proposed four-satellite international system, with the exception of the "relay-only" satellite, positioned at 100° East to handle the two-hop traffic between the Eurasian-African satellite (1) and the Pacific satellite (2). The power requirement for this intersatellite relay satellite is on the same order of magnitude as the raw power requirements for the proposed system, since the intersatellite communications were at 11 and 18 GHz with narrow beam unfurlable antennae. This relay satellite would also, in addition to these two four-foot dishes, require an earth directed antenna for telecommand and tracking. This idea was discarded in favor of either a direct satellite-satellite relay between 1 and 2, or a ground relay station at Singapore. Refer to Section 1.7. The capacity of this relay satellite would be about 300 two-way 4-kHz voice circuits.

1.10.3 Five-Satellite System

The basic differences between this system and the proposed four-satellite system are:

(1) The traffic in the Pacific and Indian Oceans is split between an Indian Ocean satellite at 122°E and a Pacific satellite at 180°E. The Indian Ocean satellite serves as a "regional" sort of relay for Asian traffic and two-hop traffic from Europe and Africa. The Pacific satellite sees most of the U.S. mainland and serves primarily as a transoceanic relay rather than a regional and transoceanic satellite, as described in Section 1.3.

(2) The North and South American satellite is now located at 76° West and does not cover Alaska. (The Alaskan traffic to the mainland must go two-hop through the Pacific satellite.)

(3) The amount of two-hop traffic for the system is now somewhat less than the proposed four-satellite system, but now nearly all of the ground stations require a minimum of two antennae. This was the deciding factor for dropping this system.

The total system load was approximately the same as that of the four-satellite system because, even though the amount of two-hop traffic was the same, the additional satellite shared its load with the Pacific satellite.

The Indian Ocean satellite and Pacific satellite had a substantial common coverage area, and this did not seem necessary for that part of the world. The other satellites for this system were Atlantic at 21° West and Europe-Africa at 15° East.

Chapter 2

REGIONAL TELECOMMUNICATION SATELLITE SYSTEMS

The regional U.S. and European networks have been designed for the following services:

(1) Two way telephone

(2) Telegraph and TELEX

(3) Leased (commercial and government) voice and data

(4) Regional TV service as a separate pair of satellites.

Traffic forecasts are from the sources listed in Section 1.1.

The effects of rate reduction were not specifically included, although generous growth rates were allowed. Further variation of growth rates may be accommodated by shifting the time of second launch.

2.1 Regional U.S. Telephone Communications Satellite

Selection of a regional U.S. telephone communications satellite was made after consideration of a number of alternatives that are placed in four general categories in Table 5. Further variations of those four categories were considered. On the basis of detailed analysis of the facts summarized in Table 5, plan C (four-beam satellite) for 1973 launch was selected, with plan D chosen as highly promising for a replacement satellite in 1978, near the end of the expected five-year lifetime of the first launched satellite. The primary reason for a choice of plan D was that it met the regional U.S. needs at the lowest cost for its five-year lifetime. Its frequency usage factor is approximately two, yielding 7000 two-way channels.

A beam pattern of four elongated ellipses (Fig. 9), using the same frequency on alternate beams, was selected but found to be near marginal for separation of similar frequency beams. Therefore, a variation was devised in which the two center beams are separated north-to-south rather than east-to-west (see Fig. 10).

2.2 Regional European Telephone Satellite

Selection of a beam pattern for the regional European case was based on consideration of a number of possibilities. The one selected has a

Table 5

CONSIDERATION OF FOUR SYSTEMS FOR THE U.S. REGIONAL NEEDS
1st Launch, 1973; 2nd Launch, as Needed

Factor Considered	Alternative Plan			
	A. 1 beam/sat	B. 3 beams/sat	C. 4 beams/sat	D. 8 small beams/sat
Beams	$3^\circ \times 6^\circ$ covers U. S. continent	$2\text{-}3^\circ \times 1.3^\circ \times 3^\circ$ covers U. S. continent	$3^\circ \times 1.6^\circ$ each covers U. S. continent (variation)	0.8° beams
Antennae	1-1 m × 1.8 m solid structure (single feed)	1-1.8 m × 2.6 m (dual feed) 1-1.8 m × 1.5 m (single feed) (unfurlables)	2-1 m × 2 m (dual feed) 1-1.1 m × 2.3 m (dual feed) 1-2.3 m (dual feed)	2-6 m round unfurlable 4 feeds each
Channel Capacity	38 K	57 K	86 K	320 K
Limitation	power and bandwidth	power and bandwidth	power and bandwidth	bandwidth
Advantages	no feed problems no unfurling problems slightly cheaper satellite	state of art feeds tested size for unfurling approximately 1.5 frequency usage	same as B 2.0 frequency usage handles load for nearly 5 years	largest number of channels 8.0 frequency usage
Disadvantages	2nd launch needed in 1974 - requires $10-15 million added to ground facilities smallest capacity 1.0 frequency usage	2nd launch needed in 1976 - adds cost as in A	intermediate priced satellite	satellite dies before all channels are needed most expensive satellite 4 feed unfurlable antennas, well beyond present state of art

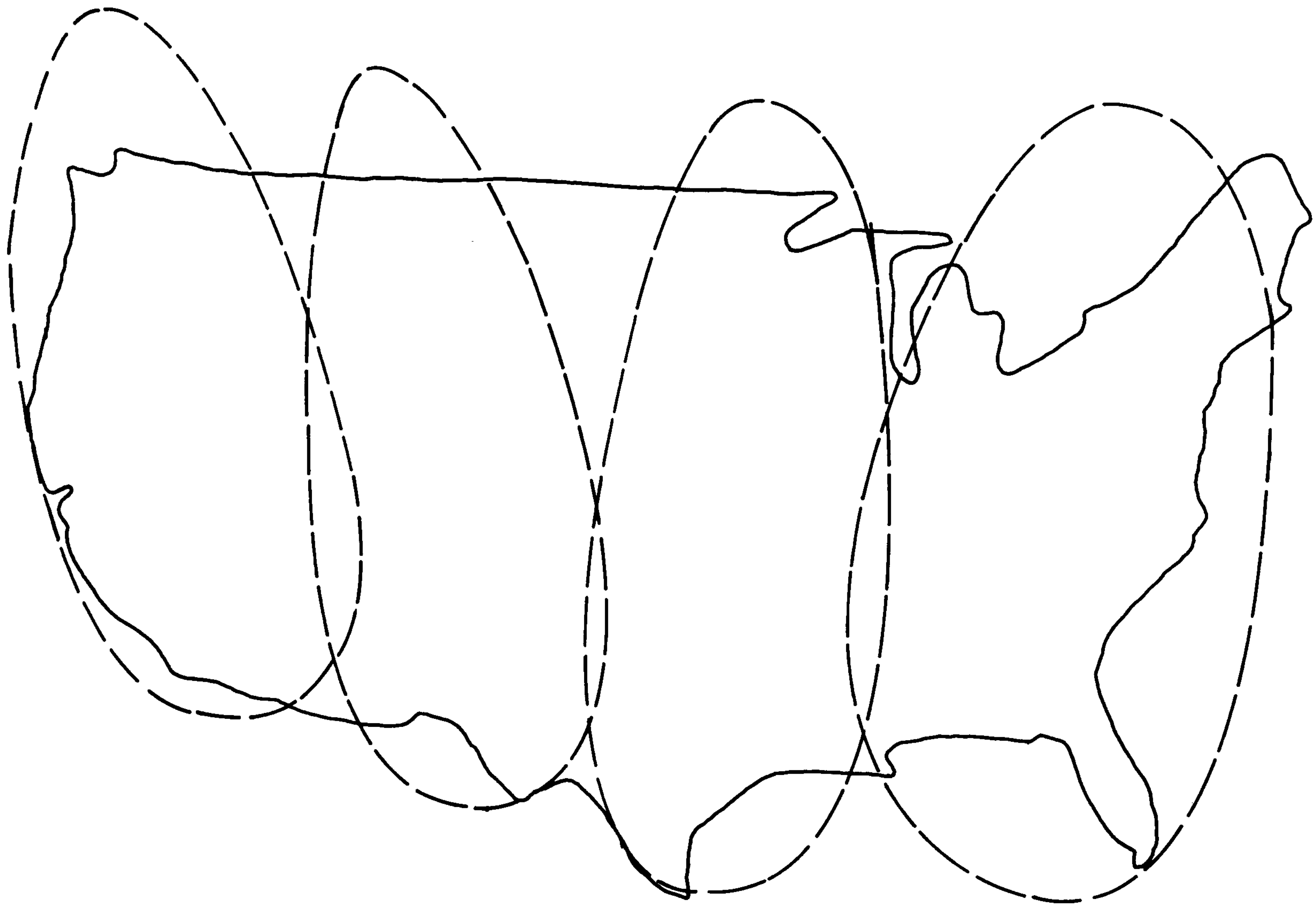


Fig. 9 A FOUR BEAM SYSTEM FOR REGIONAL U.S. TELEPHONE TRAFFIC.

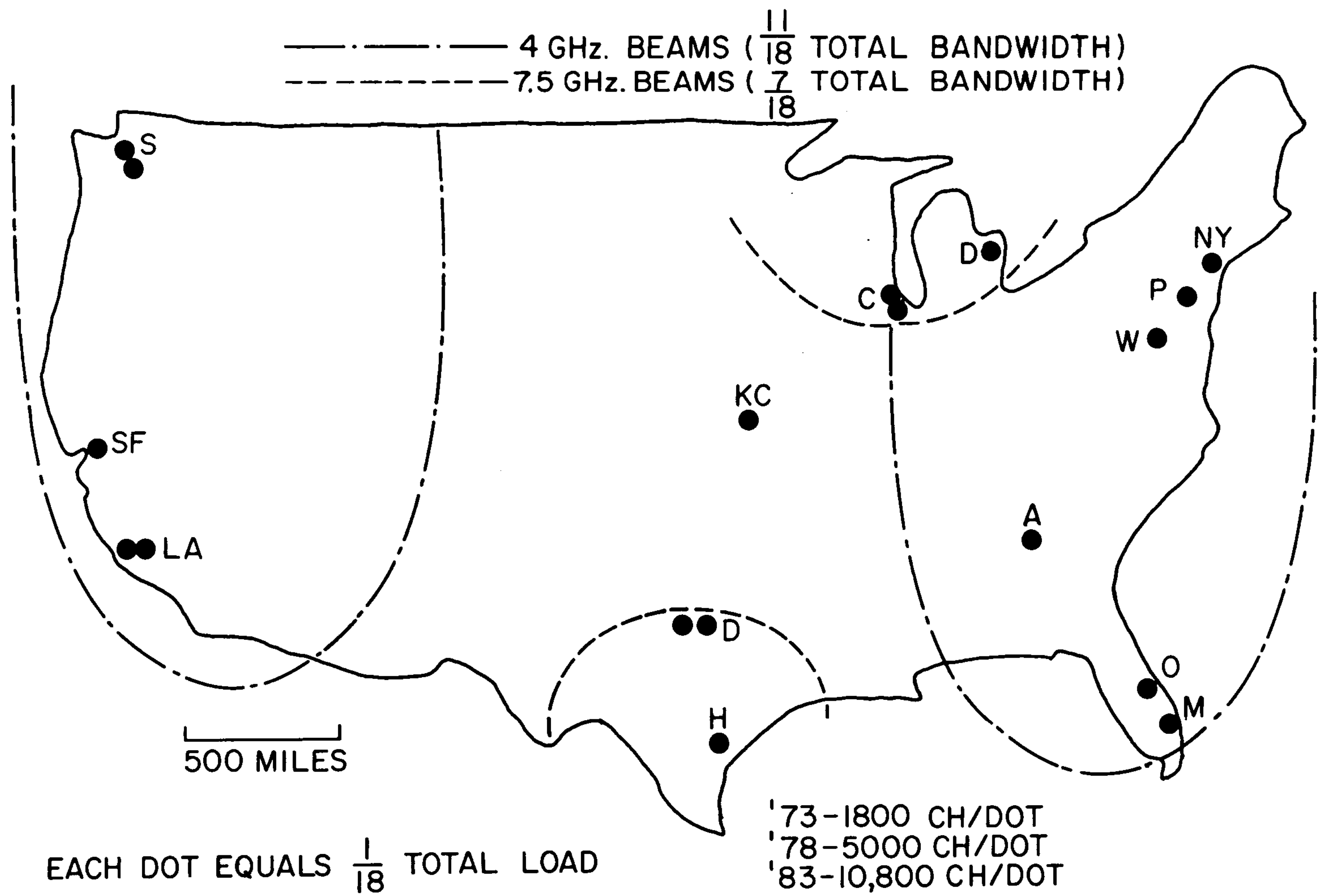


Fig. 10 U.S. REGIONAL TELEPHONE SATELLITE BEAM COVERAGE (1973-1978).

nearly identical telephone satellite beam pattern to that for the United States. This is fortunate, as design costs on a second type of satellite are then considerably reduced. The beam pattern for Europe is shown in Fig. 11. Another possible pattern configuration for Europe is one similar to that shown in Fig. 9.

2.3 Detailed U.S. Regional Traffic Consideration

Figure 12 indicates the maximum number of one-way equivalent telephone channels expected through 1983, based on traffic between an assumed 18 load centers in major U.S. cities. This type of division of traffic was used in the model in the economic analysis of Chapter 4. The maximum curve assumes a satellite system launched in 1968 with 5000-channel capacity, increasing to 65,000 by 1973. The dotted extension implies a growth rate doubling every five years. The minimum curve assumes no regional satellites launched until 1973; thus, it starts at zero. In the latter case, all long distance intra-U.S. traffic will be handled via conventional facilities. Neither of these two possibilities seems probable, so an average figure of 32,000 channels in 1973 was assumed, implying that some type of regional satellite system may be in operation by 1973.

The 18 assumed ground stations for the United States were divided in proportion to the bandwidth available (Fig. 10) in the two bands as follows:

(1) Eleven of the 18 stations are assigned to the 800-MHz band between 3400 and 4200 MHz on the downlink and part of the 5925-6425 MHz spectrum on uplink. See Section 3.4. Five of these stations are in the western United States; six of them, in the eastern United States.

(2) Seven of 18 stations are assigned to 400 MHz in the 7250 and 7750-MHz band on downlink and part of the 7900 and 8400-MHz spectrum on uplink. One possible division of these is to have three in the south central section, three in the north central section, with a special case at Kansas City, as discussed in the following section.

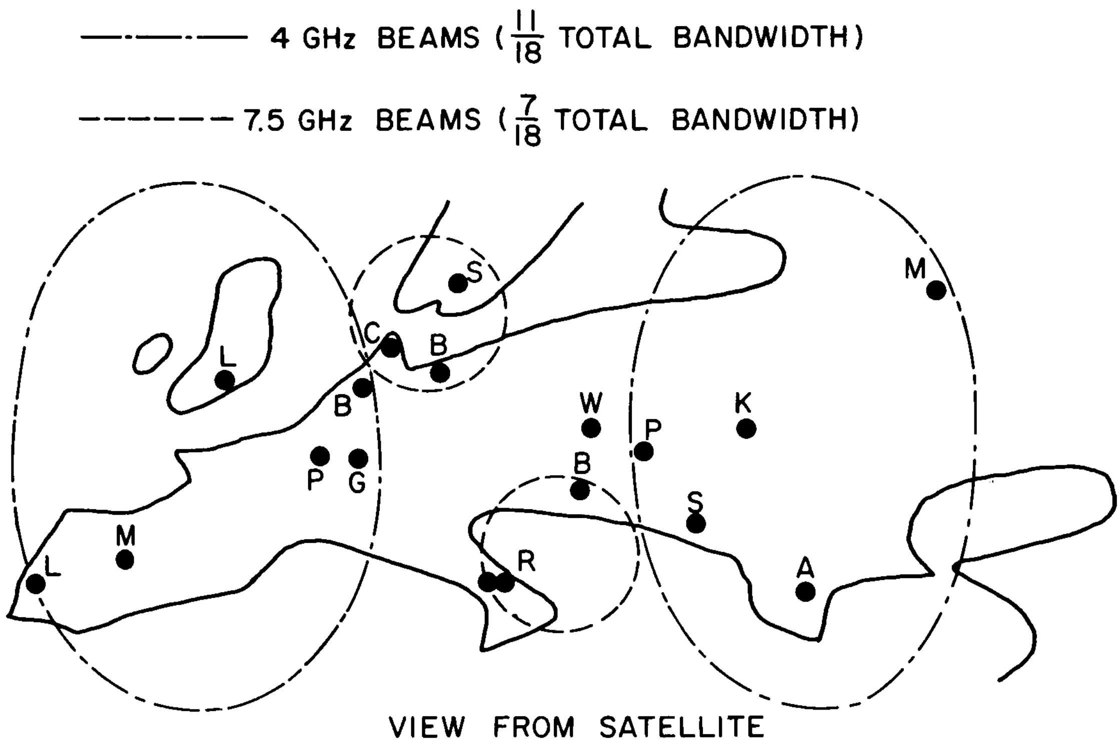


Fig. 11 BEAM PATTERN FOR EUROPE EUROPEAN REGIONAL TELEPHONE SATELLITE BEAM COVERAGE 1973-1978.

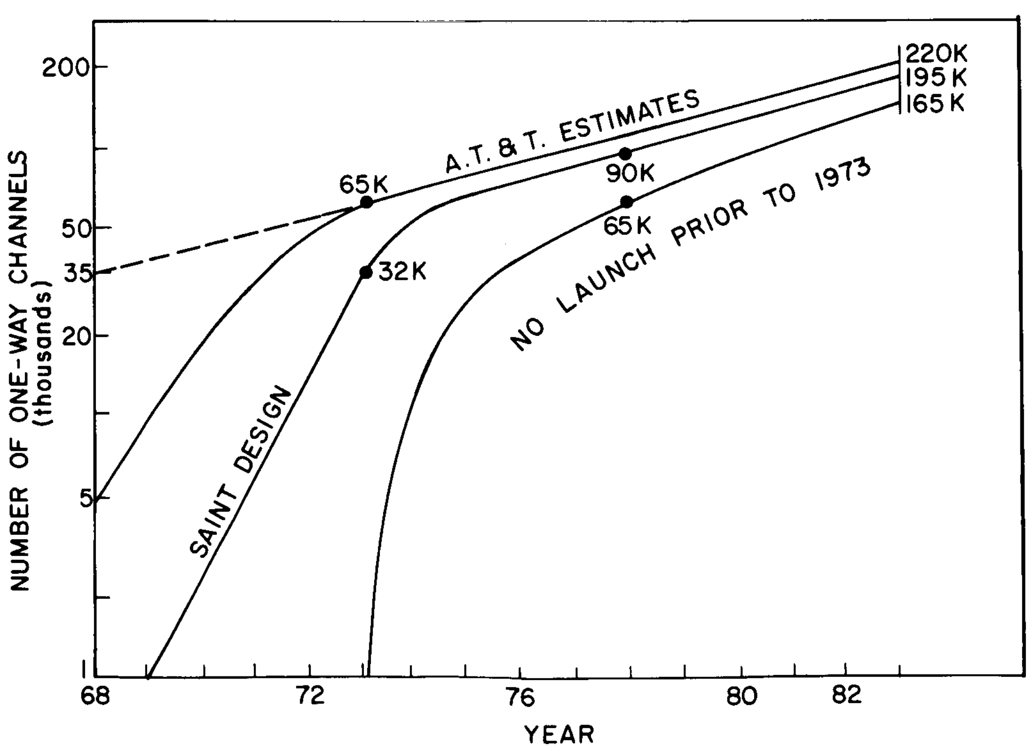


Fig. 12 MAXIMUM NUMBER OF ONE-WAY EQUIVALENT TELEPHONE CHANNELS EXPECTED IN U.S. THROUGH 1983.

Figure 13 shows the average load estimates of Fig. 12 and indicates that the initial international satellite system will handle a constant load of 19,600 channels of regional traffic during its five-year life, leaving the 1973-1978 regional satellite to relay a maximum of 71,000 one-way channels. Refer to Chapter 1, Section 1.5.

If the second launch of international satellites does not assume any regional traffic, the 1978-1983 U.S. regional telephone satellite will need a 195,000 channel capacity (one-way).

The four-beam regional telephone satellite is designed to meet an ultimate need of 7000 one-way channels near the end of its expected life. Using the total frequency available as 1250 MHz and a modulation index of 2.5, a total of 41,800 channels is available for each time that spectrum is reused; and using it twice gives a total of 83,600 channels. An allowance of 15 percent for guard bands still leaves an adequate bandwith to handle the traffic load for the expected satellite lifetime. The required C/N for 600 channel groups is 24.6 dB; for 1200 channel groups, it is 24.3 dB.

2.4 Beam Location

The initial U.S. regional satellite must reuse the available bandwidth twice through multiple beams. Figure 10 shows the location of 18 traffic centers of approximately equal loads and the beam coverage of the 4- and 7.5-GHz bands that will handle the load. Kansas City (midwestern loads) will not be covered by either main beam at 7.5 GHz. Therefore, Kansas City will rely on the international satellite system (3 and 4) to provide its regional traffic requirement. This amounts to 5000 one-way circuits split between the 10,800 one-way available on the Atlantic satellite and the 2400 one-way available at the North and South American satellite regional allocations. Refer to Chapter 1, Table 3.

The traffic load for the second generation U.S. regional satellite is large enough to require up to three traffic centers per beam, assuming each beam uses the entire available bandwidth.

Figure 14 indicates one possible beam coverage, using eight beams, which would satisfy the second generation needs. Antenna considerations for both regional systems are found in Chapter 3. This advanced regional

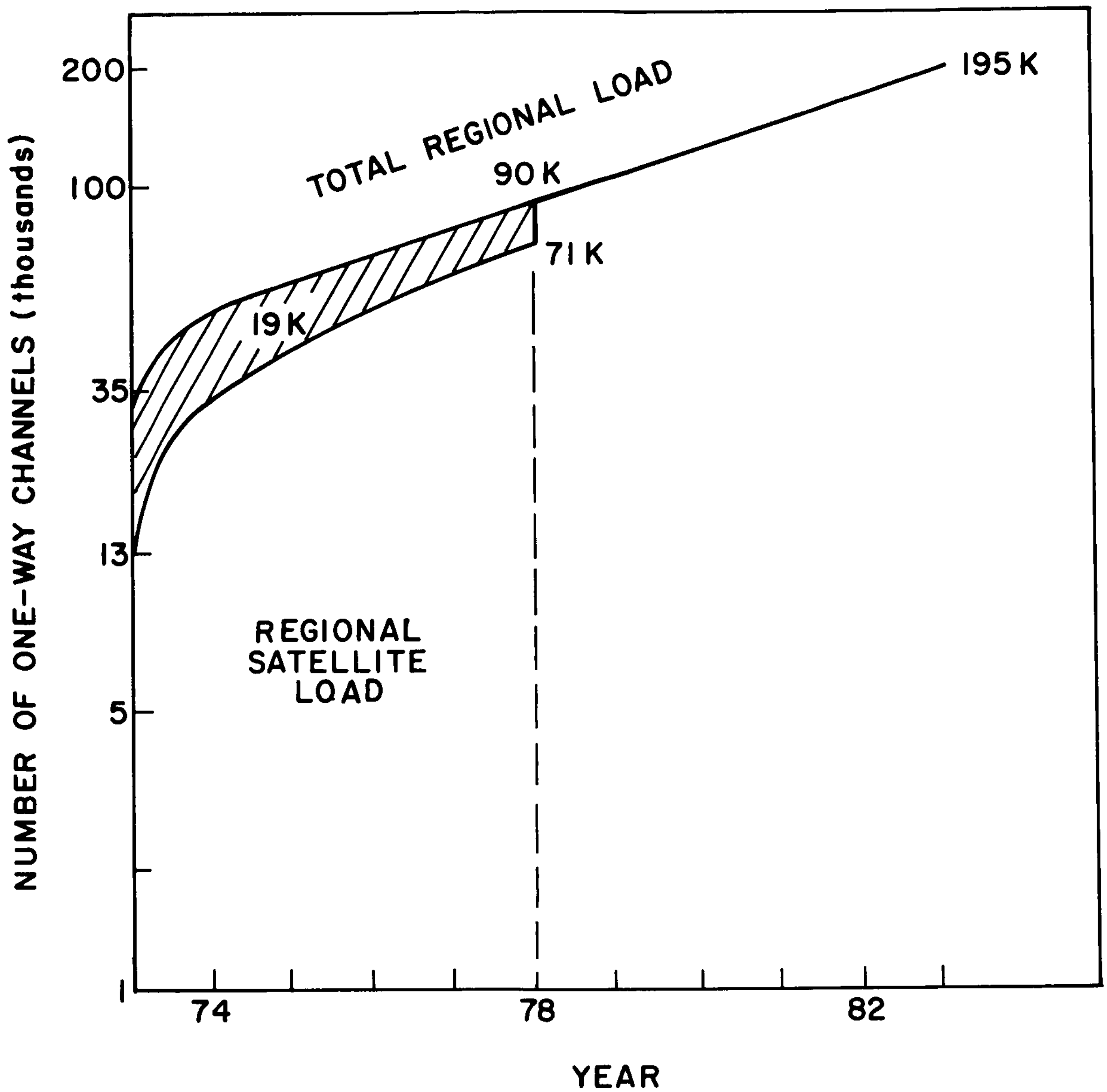


Fig. 13 AVERAGE LOAD ESTIMATES OF ABOUT 2-4.

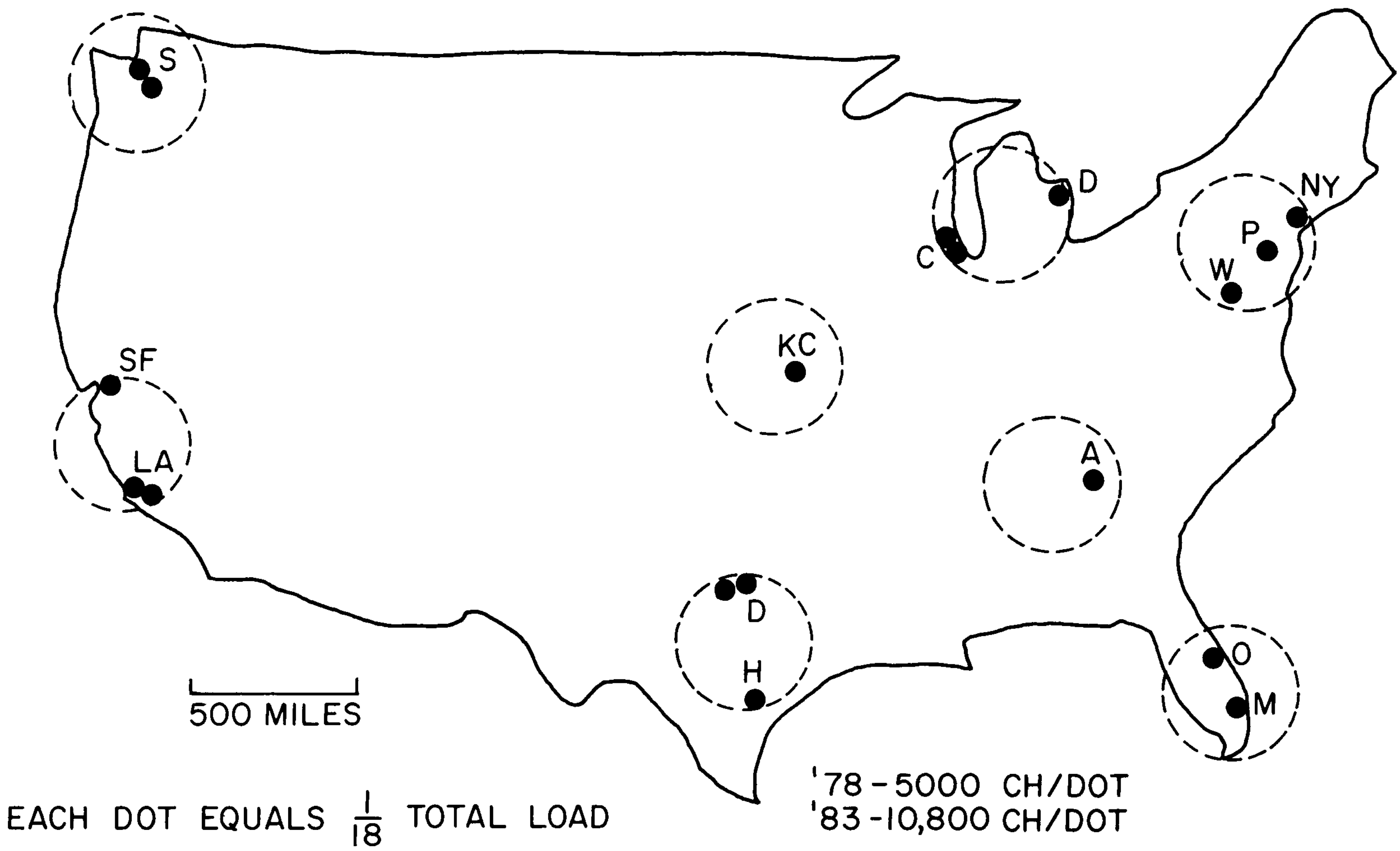


Fig. 14 BEAM PATTERN FOR U.S. REGIONAL TELEPHONE.

system requires less dc power than the first generation regional satellite and is essentially plan D of Table 5. Refer to Section 3.2 and Appendix C.

2.5 European Regional Telephone Traffic

The European regional telephone traffic estimates are not as extensive as those for the United States,[3] but the total load growth can be shown to be almost identical to that of the United States. Assumption of 18 traffic centers for Europe[2] suggests a beam pattern for the 1973-1978 system as shown in Fig. 11. A multibeam pattern for the advanced European 1978-1983 system, similar to the U.S. pattern of Fig. 14, could also be devised. Of course, political considerations can also affect location and numbers of ground stations; however, such considerations are beyond the scope of this portion of the report (see Chapter 4).

2.6 Regional Television System

On the basis of expected channel demand, television satellites, separate from the other services, were selected. Since bandwidth is adequate, single beam satellites were permissible.

Two single beam satellites will provide television distribution, one for Europe and one for North America. An anticipated need of 18 channels will require that 9 TWTs carrying two channels each be used. To fit 18 channels plus guard bands into the 4- and 7-GHz bands, 12 channels will be placed in the 4-GHz band and 6 channels in the 7-GHz band. A modulation index of 4 is necessary, and the required carrier-to-noise ratio is 20.9 dB. For uplinks, the 8.2- to 8.4-GHz spectrum is used for all 18 channels.

The U.S. TV satellite will handle the anticipated load (18 channels) throughout its lifetime, with the same frequency allocation. The European TV satellite is designed to allow the political broadcasting problems to be encountered separately. Antenna parameters may be found in Table 6. The $6^\circ \times 6^\circ$ Euorpean TV pattern from a satellite at 10°E is shown in Fig. 15. The dashed line represents the 5° visibility contour.

Table 6

ANTENNA PARAMETERS FOR THE REGIONAL COMMUNICATION SATELLITES

System	Beamwidth	Antenna Size	Shape	Number	Gain (dB)
Regional telephone for United States	$3^\circ \times 1.35^\circ$	$1\text{-}2.3 \times 1.1$ m	Elliptical	2	33.5
	1.35°	1-2.3 m	Round	2	37
Regional telephone for Europe	$3^\circ \times 1.25^\circ$	$1\text{-}2.5 \times 1.4$ m	Elliptical	2	34
	1.25°	1-2.5 m	Round	2	38
Regional TV for United States	$7^\circ \times 4.5^\circ$	0.5×0.8 m	Elliptical	1	25
Regional TV for Europe	$6^\circ \times 6^\circ$	0.5 m	Round	1	24

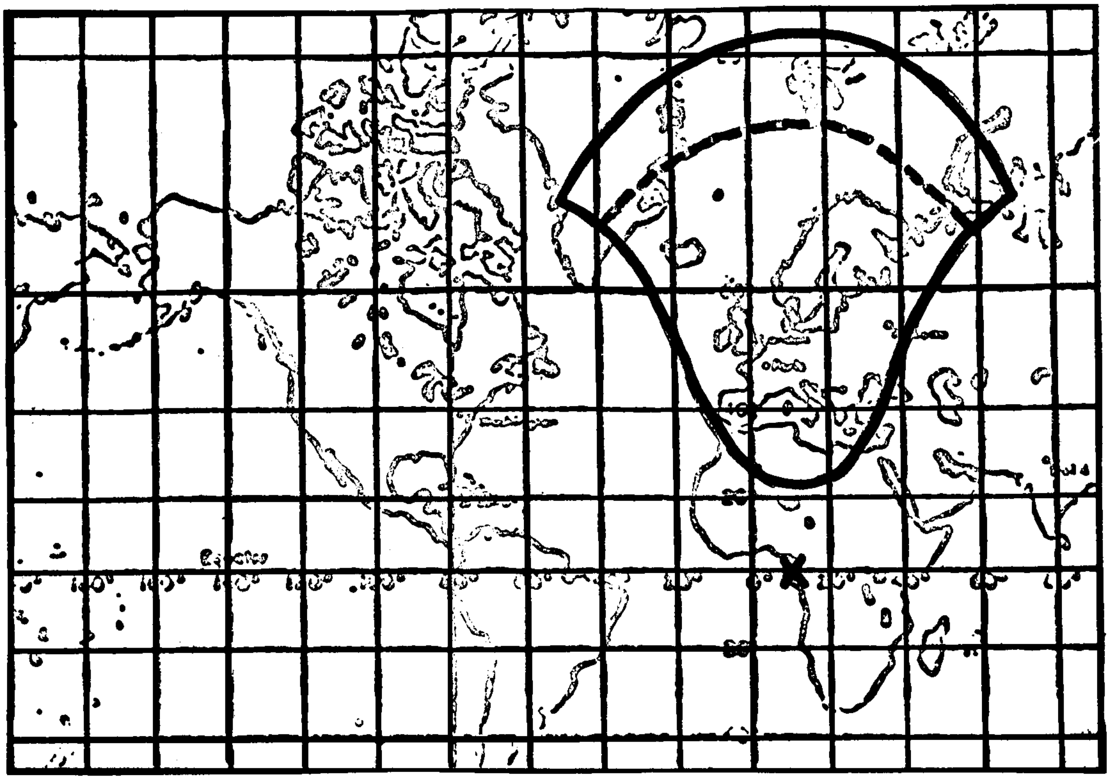


Fig. 15 EUROPEAN TV COVERAGE.

Chapter 3

ON-BOARD COMMUNICATION SUBSYSTEMS

3.1 Modulation Techniques

The types of modulation techniques that may be used in communication satellites must be considered in terms of the available bandwidth, the required transmitter power, possible types of interference, and the application to multiple access techniques. The three principal types that can be employed are single sideband, frequency modulation, and pulse methods.

For the downlink, satellite power limitations eliminate the use of single sideband methods for high capacity satellites. The advantages of pulse techniques over FM methods are not achievable without complex equipment at the transmitter, and if this equipment is required in the satellite, reliability for the life of the satellite would be lower. Where the available bandwidth is at a premium, the use of pulse technique offers little advantage over FDM-FM systems. Using the FDM-FM technique, the modulation index is easily adjusted for optimum use of the available satellite power and bandwith.

In determining the signal-to-noise ratios required, the statistics of multiplexed voice channels and the noise weighting factors used are those given in the CCITT. Recommendation G.223, and the calculations of satellite and ground station power requirements closely follow the procedures given in Doc.IV/1055-E submitted at the CCIR XI Plenary Assembly.

The required signal-to-noise ratio, using CCITT recommendations, is taken as 52.7 dB in the downlink. The carrier-to-noise ratio required to produce this signal-to-noise ratio is calculated and plotted in Fig. 16 as a function of the modulation index; the case of 600 channels per carrier is nearly optimal in that it results in almost maximum use of the satellite power. The power-saving resulting from use of one 600-channel carrier instead of ten 60-channel carriers is 5.8 dB, while the power-saving with one 1200-channel carrier instead of two 600-channel carriers is only 0.3 dB. Since most of the ground stations will require low channel capacities, the 600-channel carrier system will require less ground station equipment than the 1200-channel carrier, and the small saving in satellite power is offset by the increased ground station cost.

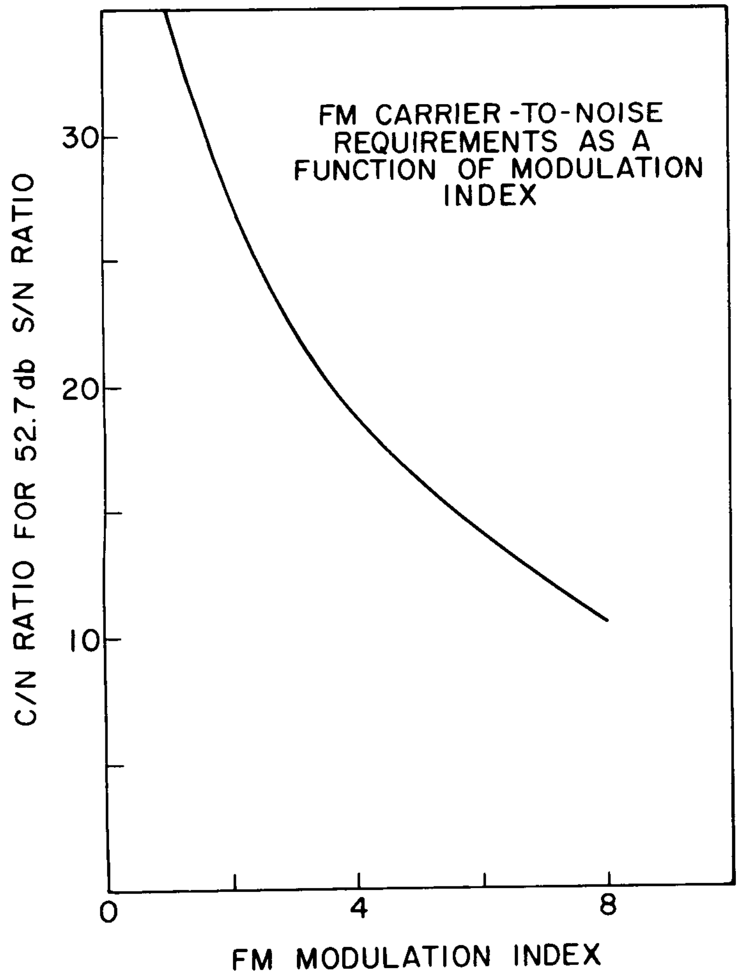


Fig. 16 FM CARRIER-TO-NOISE REQUIREMENT AS A FUNCTION OF MODULATION INDEX.

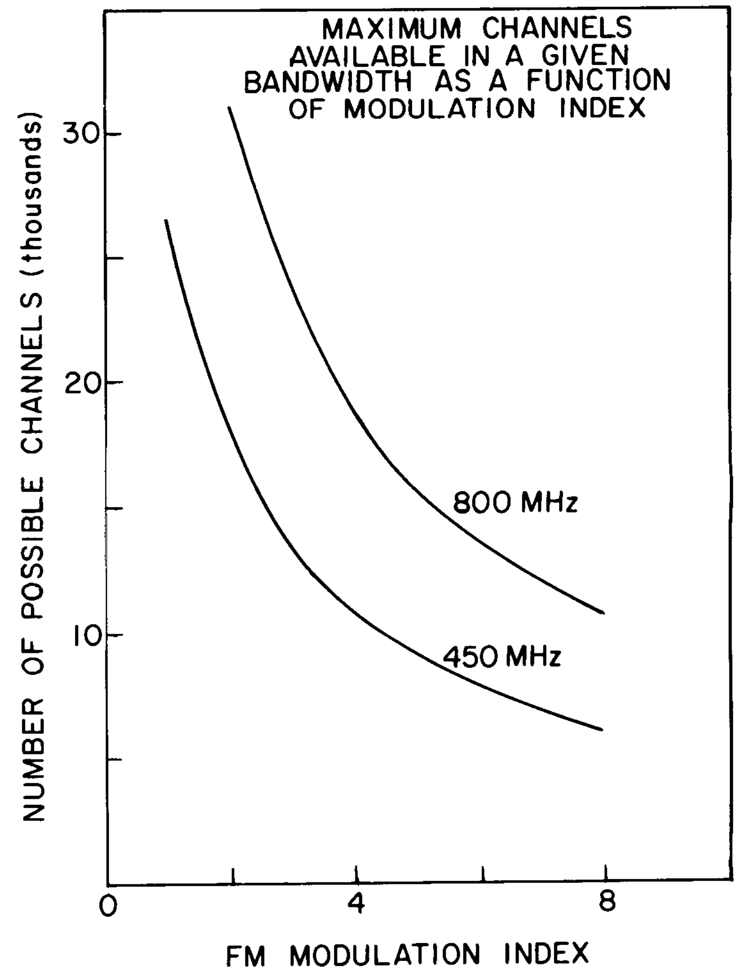


Fig. 17 MAXIMUM CHANNELS AVAILABLE IN A GIVEN BANDWIDTH AS A FUNCTION OF MODULATION INDEX.

The number of channels that can be accommodated in the available frequency bands specified for the satellite to earth link are calculated using Carson's rule to obtain the RF bandwidth, and plotted vs modulation index in Fig. 17.

To bring the several variables together for determining the required modulation index and satellite power for various channel requirements, a design curve is constructed for a 150°K ground system noise temperature, and a path loss of 197 dB. The curve gives the required sum in dB of the satellite effective radiated power and the ground station antenna gain as a function of modulation index. Thus, given a needed number of channels and the number of equivalent channels in guard bands, the required modulation index is obtained. For a given satellite beam coverage, the required ERP plus ground antenna gain value is read from the gain curve in Fig. 18. These values are then used in the cost analysis of the complete system.

The analysis of the uplink system is based on somewhat different types of limitations. Here the adaptability to multiple access systems is of prime importance, and the limitation of ground transmitter power can essentially be neglected. The process of transmitting a number of channels from several ground stations and assembling and transmitting them on a single carrier from the satellite can conceptually be easily done with single sideband uplinks properly assigned to produce the 600-channel group at the satellite in the correct frequency spectrum. Other methods of sending the channels to the satellite result in much greater complexity and little advantage in carrier-to-noise and power requirements. The difficulty in interference due to radiation off the main beam of the antenna can be minimized by careful frequency assignments when two wide beam satellites are visible from the same ground station. Since 105,000 channels are available in the 6-GHz uplink band, this assignment can be easily done. The interference is not a problem when the satellite has a narrow beam. Here the frequencies can be reused by stations outside the satellite beam (see Section 3.4).

Table 7 summarizes the effective radiated power requirements and the required ground station transmitter powers for different capacity ground stations. Tables 8 through 12 give typical up and downlink transmission path calculations for the 1973 regional satellites.

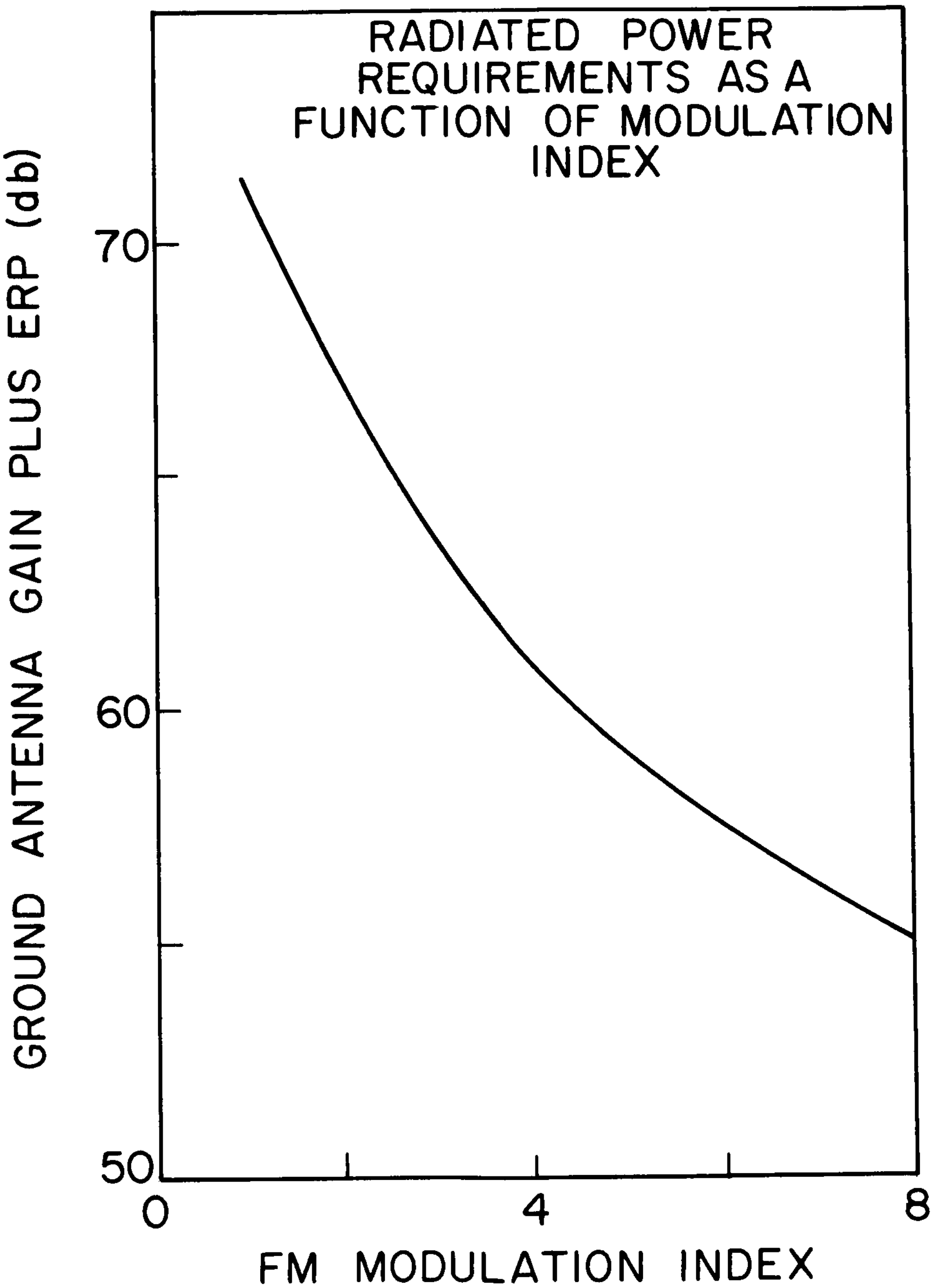


Fig. 18 RADIATED POWER REQUIREMENTS AS A FUNCTION OF MODULATION INDEX.

Table 7

SATELLITE ERP PER CHANNEL AND GROUND STATION TRANSMITTER POWER REQUIREMENTS

Satellite	Beam	Satellite ERP/Channel	Ground Peak - Power (kW)			
			60 Channel	240 Channel	600 Channel	TV
Atlantic	Europe	5.5	1.2	3.6	6.0	5.0
	North America	5.0	.68	2.0	3.0	2.8
Europe Asia Africa	Europe	4.7	.95	2.9	4.8	3.9
	Asia Africa	4.0	2.4	10.0	14.0	14.0
Pacific	North America	5.0	.68	2.0	2.8	2.8
	Far East	4.0	3.4	10.0	14.0	14.0
North America South America	North America	5.5	1.2	3.6	5.0	5.0
	South America	4.7	1.9	5.8	7.8	7.8
Regional		13.2	.07	.2	.34	2.8

Table 8

TYPICAL TRANSMISSION CALCULATION (DOWNLINK)
(Europe-Asia-Africa Satellite: Asia-Africa Beam)

Calculation	600 Channel Carrier	TV
TWT output	14.8 dBW	19.5 dBW
Feed losses in satellite	1.0	1.0
Radiated power	13.8	18.5
Satellite antenna gain	17.0	17.0
Effective radiated power	30.8	35.5
Path loss, at 4.0 GHz,	197.0	197.0
15^0 angle	-166.2	-161.5
Atmospheric loss	1.0	1.0
	-167.2	-162.5
Off axis loss at beam edges	2.5	2.5
	-169.7	-165.0
Ground antenna gain	52.0	50.0
	-117.7	-113.0
Feed losses in ground	1.0	1.0
station	-118.7	-114.0
Noise power at 85 ^{0}K	-133.8	-130.8
receiver		
Received carrier-to-noise		
ratio	15.1	16.8
Required carrier-to-noise		
ratio	14.0	15.8
Design margin	1.1 dBW	1.0 dBW

Table 9

TYPICAL UPLINK CALCULATION

(One Telephone Channel, Single Sideband)

Noise power in 4 kHz 450° noise temperature	-166.1 dBW
Psophometrical weighting factor	3.6
	-169.7
Signal-to-noise ratio	58.7
Required signal	-111.0
Satellite feed loss	1.0
	-110.0
Satellite antenna gain	20.0
	-130.0
Path loss at 6 GHz	200.0
	+70.0
Atmospheric loss	1.0
	71.0
Beam edge loss	3.0
ERP	74.0
Ground antenna gain	56.0
Radiated power	18.0
Ground system feed loss	1.0
Average transmitter output power	19.0 dBW

For 60 channels:	average increases	6.1 dB	25.1 dB (320 W)
	peak to average	10.2	35.3 (3400 W)
For 240 channels:	average increases	8.8 dB	37.8 dB (600 W)
	peak to average	12.2	40.0 dB (10,000 W)
For 600 channels:	average increases	12.8 dB	31.8 dB (1500 W)
	peak to average	11.5	42.3 dB (17,000 W)

Table 10

TYPICAL TELEVISION UPLINK CALCULATION

Calculation	TV
6-MHz bandwidth noise power 450^0 receiver	-134.3 dBW
Psophometrical weighting factor	9.2
	-143.5
Signal-to-noise ratio	52.7
Required signal	-90.8
Satellite feed losses	1.0
	-89.8
Satellite antenna gain	20.0
	-109.8
Path loss	200.0
	90.2
Atmospheric loss	1.0
	91.2
Beam edge loss	0.5
ERP	91.7
Ground station antenna gain	56.0
Radiated power	35.7
Ground station feed loss	1.0
Average transmitter power	36.7 dBW
Peak transmitter power	41.7 dBW

Table 11

REGIONAL SATELLITE DOWNLINK CALCULATION

Calculation	Telephone (1200-Channel Carrier)	TV
TWT output	10.0 dBW	14.8 dBW
Feed losses in satellite	1.0	1.0
Radiated power	9.0	13.8
Satellite antenna gain	34.0	24.0
Effective radiated power	43.0	37.8
Path loss	197.0	197.0
	-154.0	-159.2
Atmospheric loss	1.0	1.0
	-155.0	-160.2
Off-axis loss	2.5	2.5
	-157.5	-162.7
Ground antenna gain	52.0	52.0
	-105.5	-110.7
Feed loss in ground station	1.0	1.0
	-106.5	-111.7
Noise power at 85°K receiver	-133.8	-132.5
Received carrier to noise	27.3	20.8
Required carrier to noise	24.6	20.9
Design margin	2.7 dBW	-0.1 dBW

Table 12

ADVANCED REGIONAL SATELLITE
DOWNLINK CALCULATION

Total capacity of satellite: 192,000 channels
8 beams, 24,000 channels/beam
Modulation index of 4, using 1250 MHz bandwidth
Calculation is for a 1200 channel carrier, and an 85° receiver noise temperature

Noise power	-129.8 dBW
Carrier-to-noise ratio	19.1
Carrier power required	-110.7
Received feed loss	1.0
Carrier power received	-109.7
Ground antenna gain	52.0
	-161.7
Path loss	197.0
	+35.3
Atmospheric loss	1.0
	36.3
Beam edge loss	2.5
ERP	38.8
Satellite antenna gain	45.0
	-6.2
Satellite feed loss	1.0
Transmitter output	-5.2
Design margin	1.5
	-3.7 dBW
For 192,000 channels	22.0
Total satellite RF power	18.3 dBW
	(67 W)

3.2 Advanced Regional Satellite

The projected regional needs of 192,000 channels in the 1978-1983 time period require that the available frequency spectrum be reused many times on the advanced regional satellites for America and Europe. Therefore, a large number of small beams will be required to permit such reuse from a single satellite (see Sections 3.3 and 3.4).

The preliminary design of an eight-beam satellite covering the United States indicates that the basic satellite will be quite different from the international and first generation regional satellites. Due to the narrow beam requirements and associated high antenna gain, about 8 dB less power, i.e. 1/6, is required for the radio transmitters (Table 12). On the other hand, the channelizing equipment must have at least three times the capacity, which requires increased power, weight, and volume for this subsystem.

The antenna design for this satellite would require two 20-ft unfurlable antennae, each having four feeds, to produce a total of eight beams (see Section 3.3). These, of course, would be heavier than the systems required for the earlier satellites.

The satellite system total power would be between 1 and 1.5 kW, and its total weight would be similar to that of the basic SAINT satellite. Although a detailed study of the advanced satellite would be necessary, it is probable that a modification of the SAINT design, already developed for the 1973-1978 systems, would be most economical for the 1978-1983 regional satellites.

3.3 Antenna Requirements

All antenna sizes, both circular and elliptical paraboloids, are listed in Tables 1 and 6.

3.3.1 International Satellites

The size and positioning of antennae on the international satellites was determined by the land areas to be serviced by each satellite. Broad beam coverage, moderate gain paraboloids were chosen such that the edge of the beam corresponds to the 1 dB or 0.8 power beamwidth. This assures that the entire coverage area lies within the 3 dB beamwidth of the satellite antennae, even at the highest frequency.

On all satellites, each antenna will be used for both transmitting and receiving, requiring fairly broadband feeds. (If such feeds cannot be economically developed, one trade-off would be to double all antennae, separating the receiving and transmitting functions. Sufficient weight and space exist on all international satellites to achieve this.)

3.3.2 Regional Satellites

Since neither regional TV satellite is bandwidth-limited, a single, broad coverage beam is used. The same "1-dB beamwidth at 4 GHz and 3-dB beamwidth at 7.5 GHz" design philosophy as was used in the international system applies. The beams were chosen for full U.S. and full European coverage.

3.3.3 Regional Telephone Satellites

Since the telephone traffic loads described in Chapter 2 exceed the available spectrum space, the regional telephone satellites are characterized by moderate to large multiple beam antennae. Each beam can transmit and receive using the entire allocated bandwidth, if necessary. This allows the number of available channels to be multiplied as many times as is required to meet traffic demands. Such beams cannot be placed indiscriminately, but must be placed on a "noninterference-between-adjacent-beams" basis. In the Modulation discussion, Section 3.1, it is noted that a carrier-to-noise ratio of 25 dB is required for regional use on the FDM-FM downlink. This 25 dB must be maintained at all ground stations and implies that the undesired signal occupying the same bandwidth as received from an adjacent beam must be at least 25 dB below the desired beam's signal. This can be achieved, in part, by each of two techniques. First, adjacent beams must operate on oppositely rotating circular polarization. (Faraday rotation effects will not degrade the amount of isolation between opposite circular polarizations.) Feed technology can produce 10 dB of isolation between opposite circularly polarized feeds, when the feeds are part of a multiple feed system on a parabolic dish antenna.[6]

The second technique is simply to achieve an additional 15-dB isolation by spacing the half-power edges of adjacent beams, one half-power beamwidth apart. This can be derived by plotting concentric circles representing the power contours of a typical parabolic antenna. If the smallest circle of unit radius represents the 1-dB contour, succeeding circles, each of whose radius increases by a factor of unity, will represent 3, 6, 10, 19, and 36-dB contours.[7] See Fig. 19. The required spacing is easily seen to be a center-to-center beam separation of 2 half-power beamwidths. The interference region of less than 25-dB isolation lies in an "alley" between the two beam edges as indicated in Fig. 20. Any stations lying within this interference region will be served by one beam or the other, and the frequencies used for these stations cannot be reused on the second beam.

This basis was used for spacing beams, as outlined in Chapter 2, for the 1973-1978 European and U.S. Regional Telephone Satellites. Four beams are required, two at 4 GHz and two at 7.5 GHz. By distributing the traffic load in proportion to the bandwidth available at each frequency, it was decided that one elliptical unfurlable reflector at 4 GHz and one circular unfurlable dish at 7.5 GHz would fit within the space and weight constraints of the basic satellite package used for the international system.[8] Dual feeds are required, one for each beam, and this presents no unsolvable problems in the light of modern feed design capabilities.[6,9-11]

By the end of their 5-year life, the regional telephone satellites will be essentially filled to capacity, and the second generation 1978-1983 regional telephone satellites will have to have twice the 1973 capacity. This suggests a possibility of 8 beams per satellite, and might be achieved as outlined in Chapter 2, by using two 20-ft unfurlable paraboloids, each having four beams emanating from four feeds. The antenna can be packaged within the proposed configuration, but a four-beam feed system would require some extensive development. By 1973, this could most likely be achieved, thus enabling one to use the basic system design for ten years.

3.3.4 Ground Station Antennae

In an attempt to obtain a low cost, easily fabricated and erected ground antenna, a newly developed technique of using plastic foam and fiberglass for large parabolic dish construction was investigated.[12]

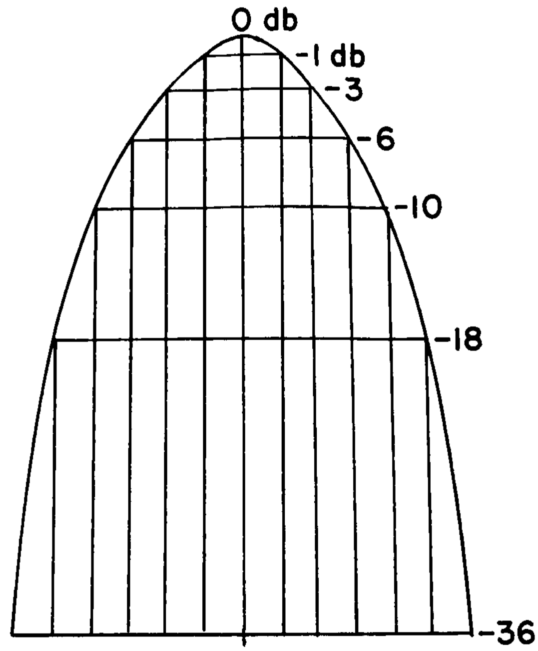


Fig. 19 TYPICAL PARABOLIC ANTENNA BEAM PATTERN.

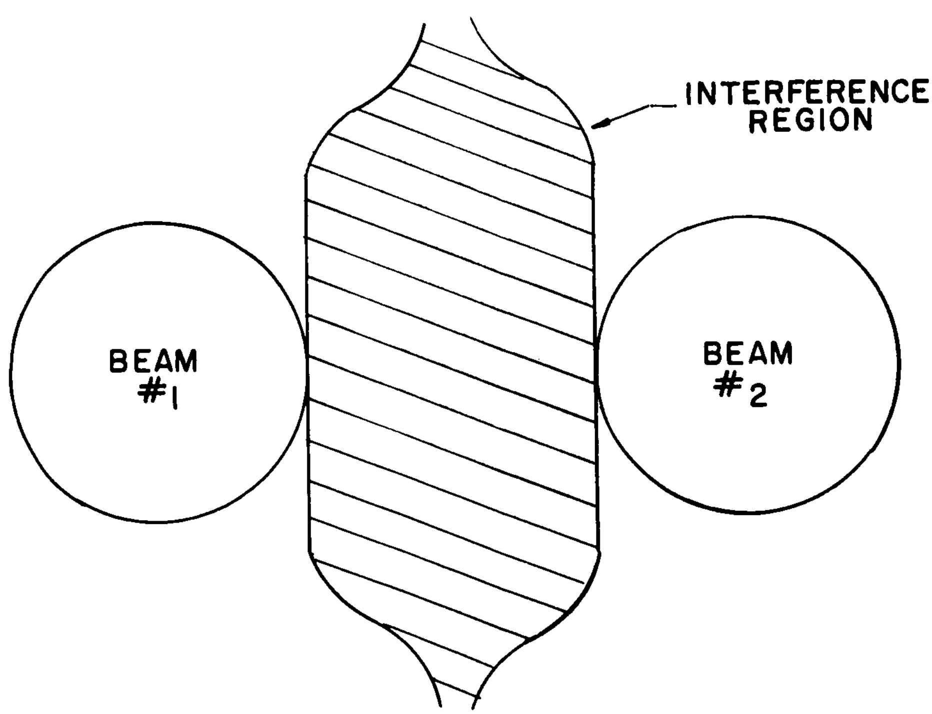


Fig. 20 TWO BEAMS SPACED AT MINIMUM ALLOWABLE SEPARATION.

On all of the above counts, the plastic ground antennae are superior: The cost is 60 to 70 percent less than that of conventional metallic structures; they also represent what appears to be a new standard for this type of application. The pedestal requirements are very modest due to the extremely light weight of these plastic structures.

With the $\pm 0.1^\circ$ station keeping requirements, the expensive tracking system and drive mechanism, characteristic of previous communications satellite ground stations, is replaced by a simple, manually adjustable pedestal.

3.3.5 Atmospheric Propagation Losses

Sources of attenuation at microwave frequencies occurring within the earth's atmosphere are

(1) oxygen and water vapor molecular resonance absorption,
(2) precipitation scattering and absorption,
(3) cloud droplet scattering and absorption.

These have been tabulated and are presented in Fig. 21 as a function of the ground stations elevation look-angle above the horizon. The 60° angle occurs for many U.S. and European stations and the 10° angle represents the minimum angle that was allowed when choosing satellite placement.

Oxygen and water vapor contribute a small amount at 4 to 7.5 GHz; these frequencies are sufficiently removed from the resonance frequency of 22 GHz.

Precipitation (rain, hail, and snow) causes a reduction in signal strength when it occurs along the propagation path. Hail produces attenuations of less than 1 percent of that due to rain and thus can be neglected.[13-15] Snow is considered with heavy cloud cover that accompanies it and the moderate rainfall attenuation presented in Fig. 21 is a result of combining several sources of information.[13,14,16] This "moderate" amount of rain (15 mm/hr) can be expected to be widespread in extent and occurs approximately 5 hr/year at temperate latitudes. Heavy cloud cover attenuation is a more serious problem than rain in satellite propagation, or than it is for land based propagation paths, for the

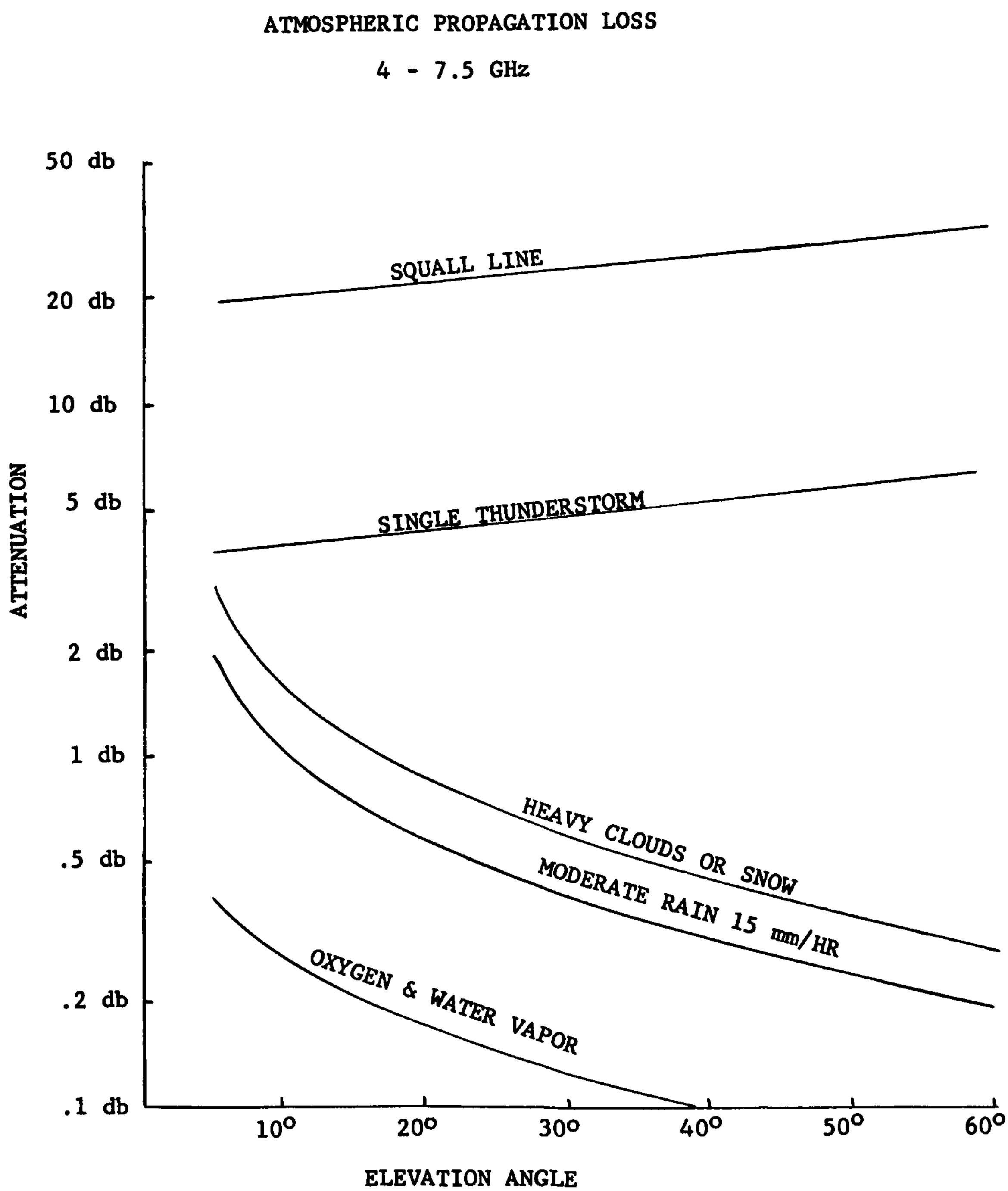


Fig. 21 ATTENUATION VS ELEVATION ANGLE.

obvious reason that the total path length containing high loss rain is much less when looking skyward. A model cloud/rain cover of 4 km high containing rain in the bottom 2 km, heavy cloud vapor over the entire 4 km, and extending uniformly in the horizontal place was used.[14] The average sum of the lower three attenuation effects (gaseous, moderate rain, and heavy clouds) was added to all path loss calculations as a constant 1-dB factor labeled "atmospheric losses" as shown in Tables 8-12.

The highest two curves in Fig. 21 show total loss through a typical thunderstorm 4 miles wide and 7 miles high.[17] Larger storms of 20-mile width occur, but less frequently.[18] The loss decreases at lower elevation angles because of the "necking" effect in thunderstorms which results in a base width smaller than the high altitude width. The squall line demonstrates the effect of a widespread belt of thunderstorms, typically 100 miles long and 20 miles wide with thunderstorms occurring over 20 percent of the length. A single thunderstorm of 5-dB attenuation can cause a noticeable reduction in signal-to-noise ratio, and a squall line occurring along the propagation path can result in 20-30 dB of additional loss, causing complete blackout.

The problem of predicting thunderstorm degradation and outage for each specific ground station is a difficult one at the present state of knowledge of thunderstorm distribution and occurrence. Available information[19,20] indicates the following general guidelines for estimating thunderstorm outage times:

(1) Thunderstorm-days occur at the rate of 20-40 days/year in the temperate latitudes and at the rate of 40-120 days/year in the tropical latitudes during the warmer months of the year.

(2) When thunderstorms occur in an area, approximately 20 percent of the area is under rainfall.

(3) A typical storm will be in the ground station's beam for 15-30 minutes as it passes.

The result of these considerations is

(1) U.S. and European ground stations can expect thunderstorm <u>degradation</u> of 15-30 minutes duration for 5 to 8 days a year during the warm season and one squall line <u>outage</u> of 15-30

minutes' duration each warm season _if_ a typical squall line occurs along the propagation path.

(2) Any back-up diversity stations should be over 20 miles apart and along a path perpendicular to normal squall lines.

Early in the design of this system, 18 and 30 GHz were briefly considered for regional use because of the larger available downlink bandwidth (1600 MHz vs 1300 MHz) and higher gain, smaller beamwidth satellite antennae. A quick survey of the propagation situation yielded these results for 18 GHz with respect to 4 GHz at a 30° elevation angle.

Typical antenna gain improvement	+14 dB
Free space path loss	- 9
O_2 and H_2O vapor loss	- 1
Moderate rain loss	- 6
Cloud loss	- 2
	- 4 dB

In addition, a single thunderstorm that caused only a 5-dB degradation at 4 GHz now produces a 135-dB blackout, necessitating diversity ground receiving sites over 20 miles apart. The fact that microwave technology at 18 and 30 GHz is far less advanced than at 4 and 7 GHz, combined with these propagation factors, eliminated the millimeter wave bands.

3.4 Frequency Utilization

Various techniques permit use of the available frequency spectrum over and over without causing interference. The signals can be isolated by use of different beams on the satellite to cover various parts of the earth, by use of multiple satellites in orbit to cover the same area of the earth, i.e., multiple beams from the ground stations, and by alternate use of right-hand and left-hand polarizations.

The degree of isolation required depends on the modulation technique used, which is different in the SAINT proposal for the up and downlinks. On the downlink, FDM-FM has been used throughout and the needed isolation is a minimum of 15 dB for the international system, and a maximum of 24 dB for the regional telephone satellites.

Section 3.3.3 describes the use of a combination of polarization isolation and satellite beam separation to reuse the satellite-to-ground frequency twice in the 1973-1978 regional telephone satellite, and up to eight times in the 1978-1983 satellites.

In addition, the SAINT proposal, in assigning downlink frequencies, takes advantage of satellite separation. The minimum spacing between satellites is 5° at the equator, representing at least five beamwidths of the ground antenna. This provides at least 30-dB isolation between satellites and allows the same spectrum, or portions of it, to be used for all eight proposed satellites on the satellite-to-earth link.

On the earth-to-satellite link, SSB-FM has been selected to allow multiple access to the satellite from many ground stations. This requires much less bandwidth per channel than does an FDM-FM system, but also requires more isolation to avoid interference. In the worst case, 59-dB isolation is necessary. This can only be accomplished with present technology by a combined satellite beam isolation and ground station beam isolation; that is, uplink frequencies can be shared only when there is more than one satellite (giving 30-dB isolation from ground antennae) and if the receiving beams on the two or more satellites cover different areas on the ground (adding 30-dB isolation in satellite antennae to give the desired total of 60 dB).

The isolation can also be enchanced by proper use of polarization. Using this for the uplink allows the spectrum to be used twice for the four regional satellites. In addition, the regional TV satellites for Europe and for the United States can share the same frequencies as can the two regional telephone satellites. In fact, in the future, any other TV or telephone satellites added to cover other continents would use the same assignment as the first satellites.

The proposed system plans to make use of the frequency spectrum available for satellite-to-earth transmission outlined at Geneva by the ITU in 1963. See Table 13. This source had divided the world into three regions, as shown in Fig. 22 and listed small bands according to their multiple assigned usage.

Nearly all of the frequencies assigned to communications satellites are shared with some other noninterfering applications. The bands used

Table 13

ITU FREQUENCY ALLOCATION - GENEVA 1963

	Region I Europe, Africa North Asia	Region II Western Hemisphere	Region III South Asia
Down	3400-4200 7250-7750	3400-4200 7250-7750	3400-4200 7250-7750
Up	4400-4700 5725-6425 7900-8400	4400-4700 5925-6425 7900-8400	4400-4700 5850-6425 7900-8400

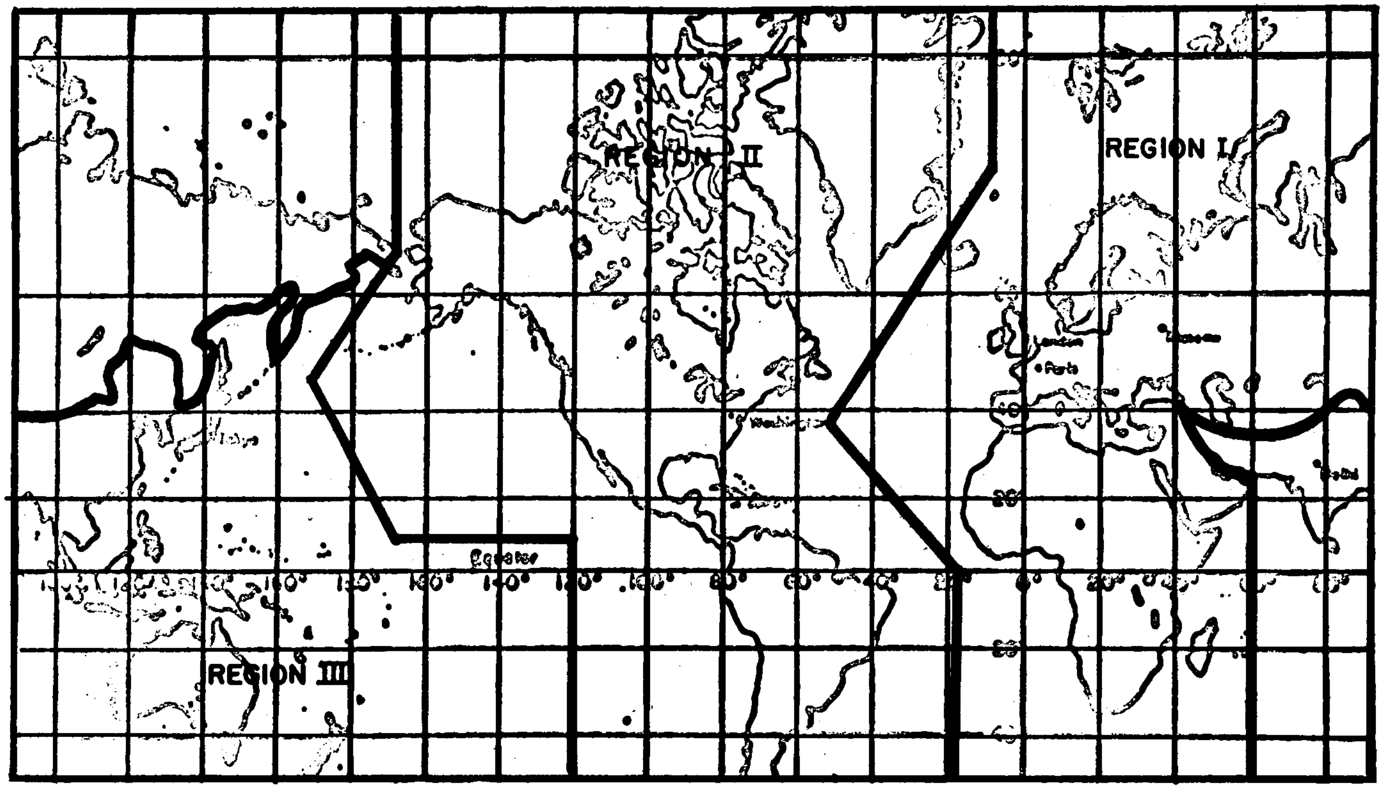


Fig. 22 REGIONAL FREQUENCY DIVISIONS.

extensively by the SAINT system are the 3.4- to 4.2-GHz and 7.25- to 7.75-GHz for satellite-to-earth. Since the military satellite system also uses the 7.25- to 7.75-GHz band, only 450 MHz of the available 500 MHz will be considered, giving a maximum downlink bandwidth of 1250 MHz.

Only three of the international satellites use the 3.4- to 4.2-GHz band; the fourth, the Atlantic satellite, also uses part of the 7.25- to 7.75-GHz region. The regional TV and telephone satellites each use the entire 1250 MHz available in the two bands.

For the earth-to-satellite link, the available frequencies would be used as follows: approximately one-third of the band from 5.925 to 6.425 GHz would be used for the four international satellites (note that in ITU Regions I and III an extra 200 MHz and 100 MHz are also available for this use). The remaining two-thirds, equivalent to more than 71,000 one-way voice channels, is used for the European regional telephone satellite and reused for the U.S. telephone satellite. Approximately one-fifth of the uplink band at 7.9 to 8.4 GHz is used for the European TV satellite and reused for the American TV satellite.

In the advanced regional telephone satellites for 1978-1983, all the 1250 MHz available for satellite-to-ground communications would be used. For ground-to-satellite use, if SSB-FDM is still used, two-thirds of the 5.925- to 6.425-GHz band, all of the 4.4- to 4.7-GHz band, and one-half of the 7.9- to 8.4-GHz band would be needed. Both regional satellites would use the same frequencies as would similar satellites serving other regions.

Interference with and from terrestrial microwave links is also considered in the placement of ground stations and frequency assignments. This is particularly important in the North American regional satellite in areas where there are extensive TD and TH microwave links.

3.5 Transponders

3.5.1 Transponder Scheme

The transponder scheme of the international and national satellite is essentially determined by three factors:

(1) the modulation technique chosen for the up and downlink,

(2) the types of information handled by the satellite, and

(3) the number of beams covering the earth per satellite.

In the present case, the decisions were made to use pure SSB-modulation for the uplink when transmitting voice information, and vestigial SB-modulation when handling TV signals. In both cases for the downlink, wideband phase modulation technique is proposed.

The consequence of this scheme is a repeater system which has to demodulate the received signals of several ground stations to baseband and remodulate them as a wideband phase modulated signal to be transmitted to the earth.

The number of antenna beams will chiefly influence the interconnections of subunits and units within the transponder. As a typical example, a diagram of the arrangement of traffic for the Atlantic satellite is shown (see Fig. 23). There are three different possibilities of arranging the traffic: the international, the international combined, and the regional traffic.

3.5.2 Description of the Transponder of an International Satellite (Atlantic: see Fig. 24)

The handling of voice and TV channels in the satellite repeater is in principle the same. Differences exist only in the kind of demodulation.

The received signal is amplified in an uncooled parametric preamplifier which requires a microwave circulator because of its negative resistance quality.

Down conversion in frequency is done by three stages, each with a local oscillator to minimize the requirements of conversion components referring to the slope of filters, etc. The conversion requires an accuracy in frequency of the first local oscillator of about 10^{-10} which can be realized by a temperature controlled crystal. The amplification of the intermediate frequency is split into two stages to satisfy the requirements of good linearity with the use of a SSB scheme. The gain of each stage should not be estimated too high. The amplification factor of the third IF stage is variable by an automatic gain control circuit, fed by the input of the phase modulator, which is eventually feeding the power stage. The AGC is used to obtain a fairly constant frequency

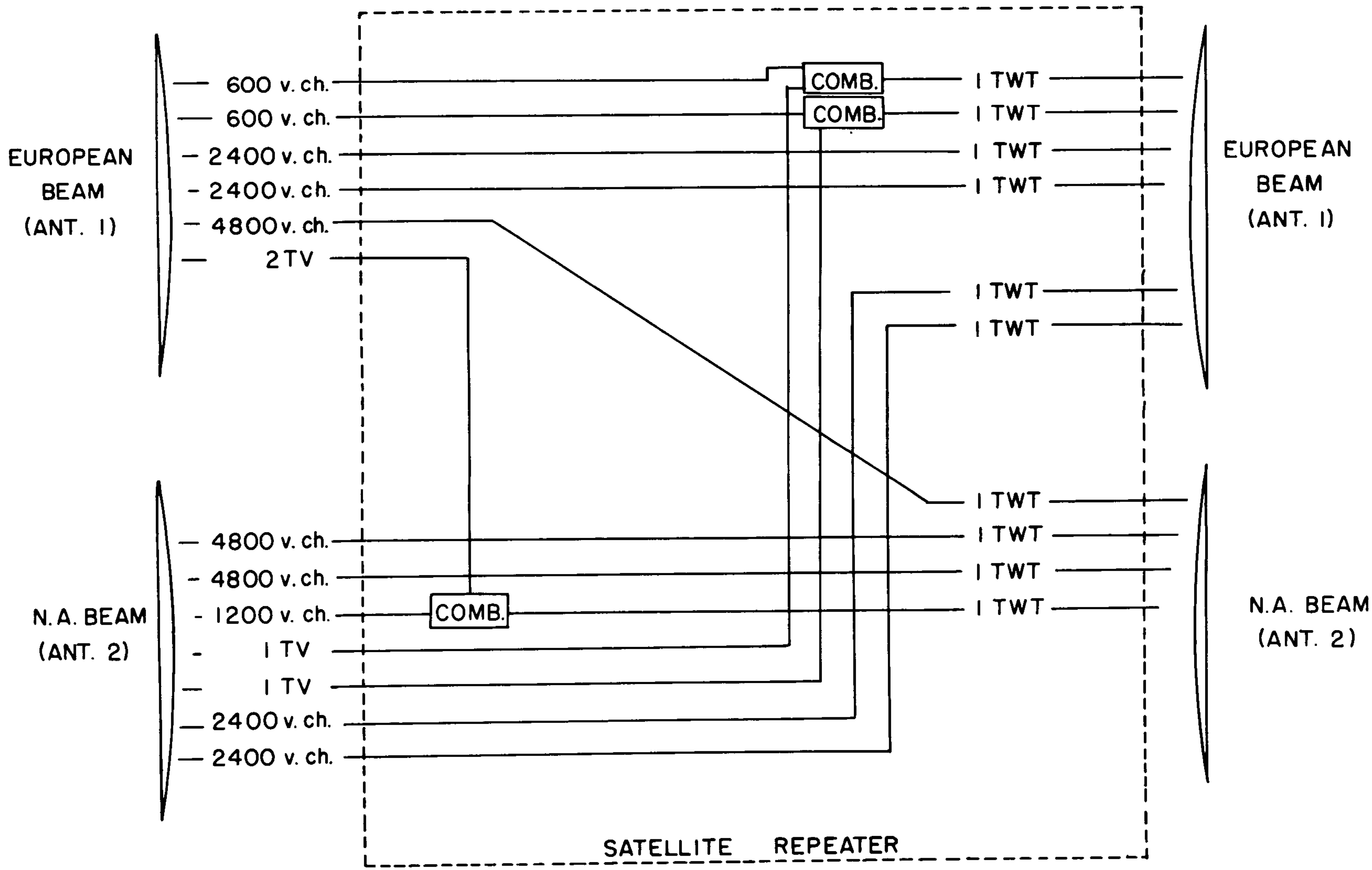


Fig. 23 ATLANTIC SATELLITE TRAFFIC ARRANGEMENT.

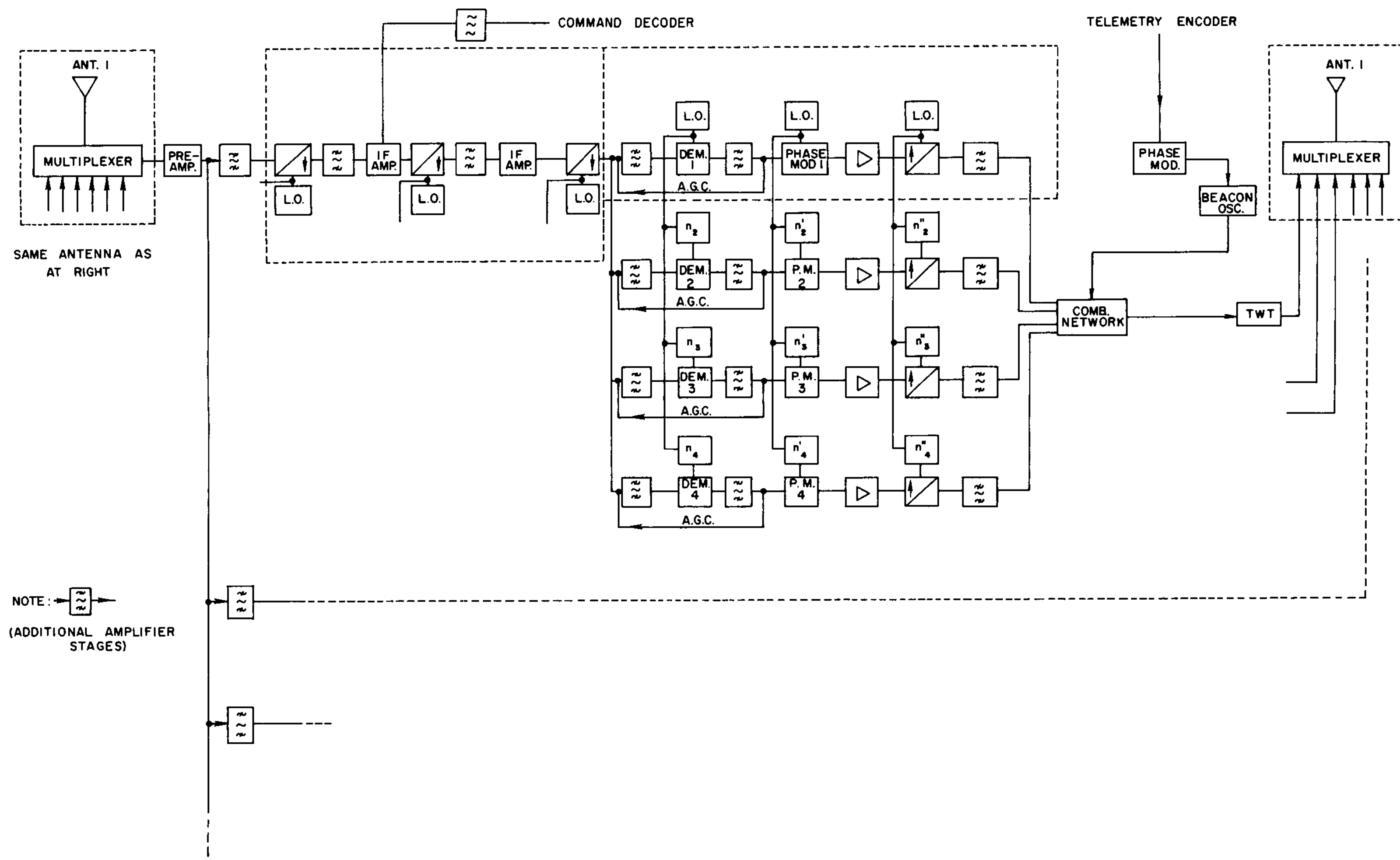


Fig. 24 ATLANTIC SATELLITE TRANSPONDER.

deviation for the phase modulators. The gains are fixed after the phase modulators to obtain a constant amplitude at the combining network.

The IF scheme was selected because the microwave frequency has to be demodulated down to baseband in any case, this IF method provides an easy implementation and modification when changing the gain per stage and the advantage of having the major part of the amplification at a relatively low level of frequency and power. Solid stage devices offer a good reliability at the present state of the art. The use of one or two TWTs instead of the IF amplification stages would surely increase the weight of the repeater while limiting the possibilities of changing the design.

An AGC system seems to be more adapted to an IF amplifier than to a TWT, and in the case of multiple access, the number of input signals may vary in a wide range, so that the AGC cannot be relinquished.

In demodulating the received signals, a differentiation is made between voice and TV. Temperature controlled, fixed local oscillators with sufficiently high accuracy are used for voice band demodulation to add the suppressed carrier. With TV transmission, a suppressed carrier is transmitted to lock the phase of the local oscillator on board to obtain phase coincidence (see Fig. 25).

The demodulating stage is followed by a phase modulating device, which drives a combiner to feed the TWT.

3.5.3 Standardizing of Units

The single components of the satellite repeater are combined in packages of equal or similar properties, building special units available for each type of the international and regional satellites.

The frequency down conversion and IF amplifying device is the same unit in all satellites. Voice band demodulating and modulating components are split in blocks handling voice channels. For one TV channel, similar blocks are used for 1200 equivalent one-way voice channels.

As an example, the completed block diagram of the Atlantic satellite is given (see Fig. 26) which shows the arrangement of traffic (see Fig. 23), and the statement of the interconnections of units and subunits mentioned above.

Instead of transmitting two groups of 2400 one-way voice channels in international traffic, it is proposed to handle each TV channel

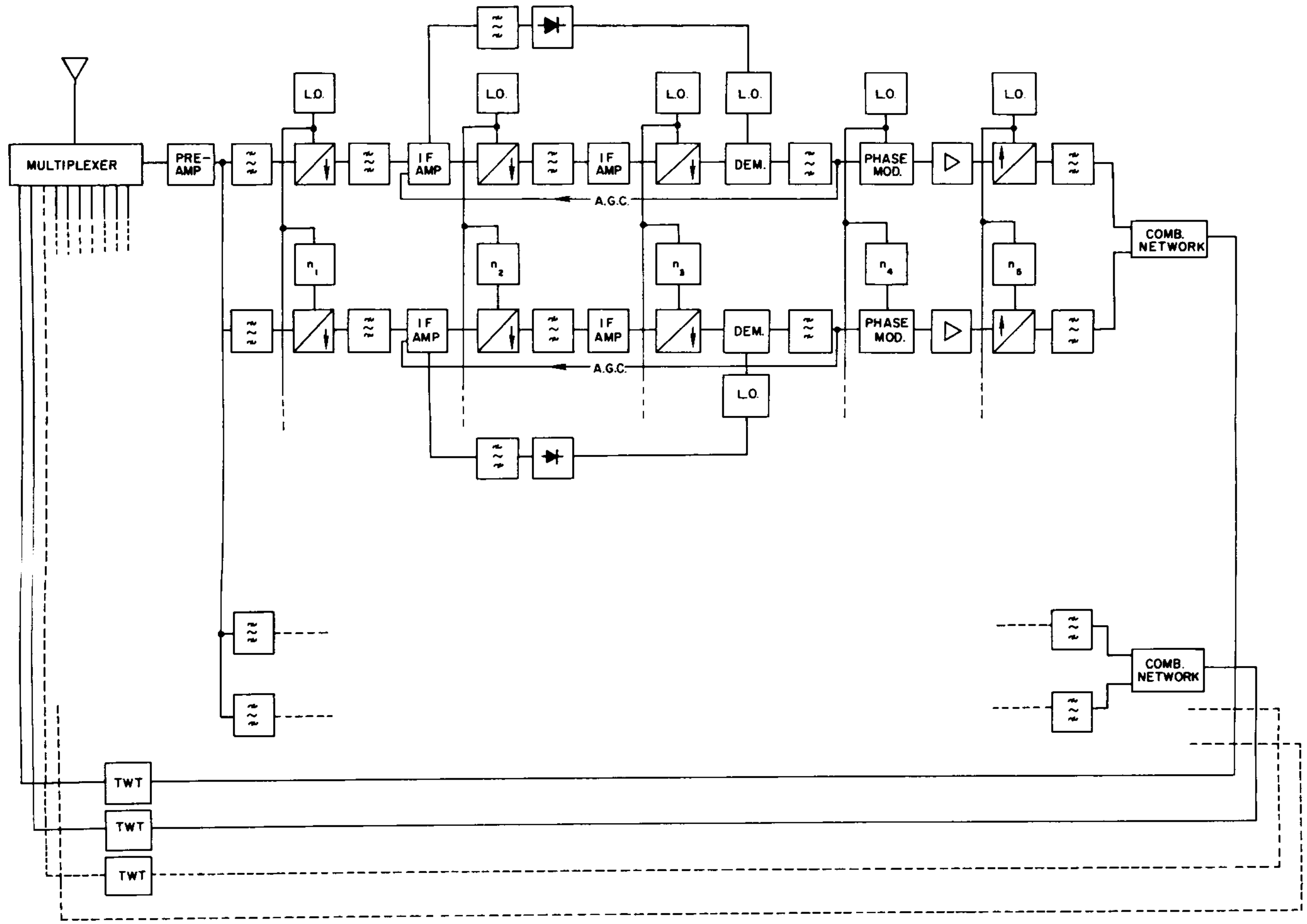


Fig. 25 REGIONAL SATELLITE TV TRANSPONDER.

(international) during times when the voice capacity is not loaded. In this case, the demodulating and modulating equipment is switched off whereas all other components might be used (see Fig. 26). This is an additional flexibility of the chosen transponder scheme.

3.5.4 Telemetry and Telecommand

It is proposed to use the microwave transponder as a telemetry transmitter and command receiver in the microwave band. The beacon signal is phase modulated by the output of the telemetry encoder. The command signals can be obtained from the first IF amplifier by a filter and a decoding device (see Fig. 24).

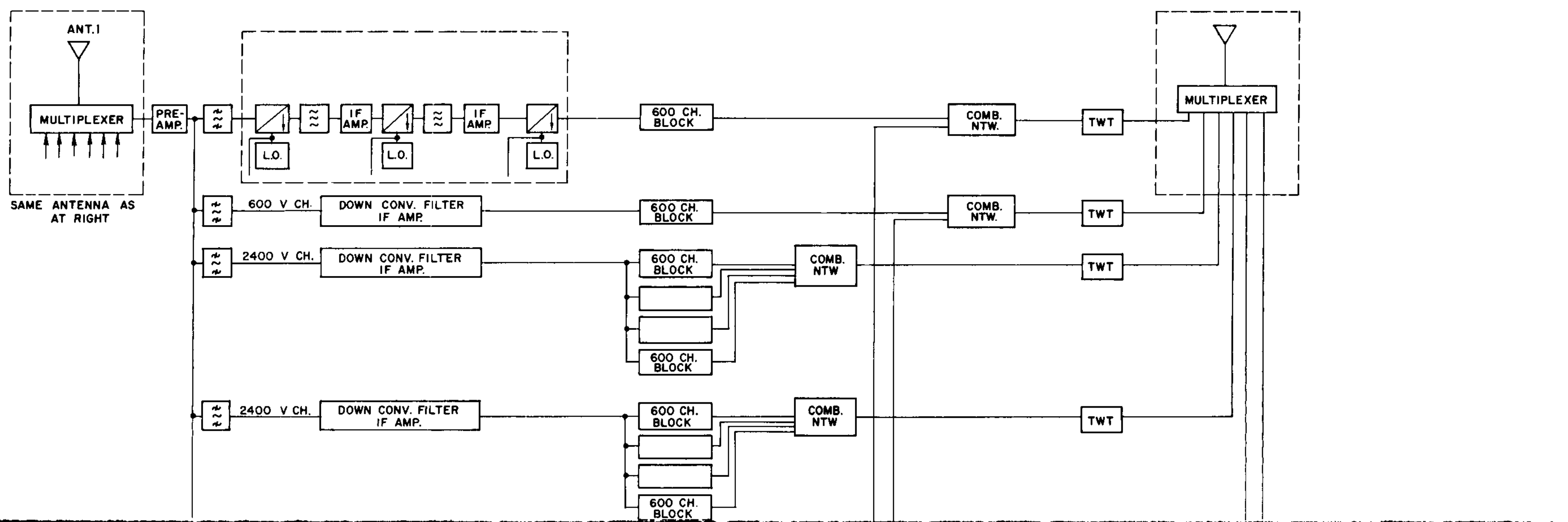
3.6 Satellite Traveling Wave Tube Requirements

The objective is to design the equipment to be essentially identical for the complete set of international satellites. The modulation index is specified by the maximum number of channels required in a single satellite beam. The satellite beam with the greatest number of channels is the North American beam of the Atlantic satellite. The number of one-way channels is 15,600 voice and 2 TV, each of which is equivalent to 1200 voice, for a total of 18,000 channels. Allowing guard band frequencies of 15 percent of the total, the required capacity is 21,200 channels. A modulation index of 6 is used, requiring a bandwidth of 1250 MHz.

The satellite power requirements are determined mainly by the satellite beams with the largest coverage: the Asia-African beam of the Europe-Asia-Africa satellite and the Far East beam of the Pacific satellite. From the model analysis which corresponds closely to the African case, the optimum ERP/channel is 3. The radiated power for 600 channels is 24 W. Two groups of 600 channels will require 48 W while a TV channel, taking a power requirement equivalent to 1800/voice channels, will require 72 W. A 90-W TWT will provide an additional margin of one dB, providing a radiated power of 60 W when operated 1.8 dB off saturation.

Using the same tube in the Atlantic satellite, the ERP/channel will be 5 for 4800 channels in a single TWT. The number of tubes for voice channels is then 3.25 and for the two television channels 0.75, requiring a total of 4 tubes in the North American beam. The beam gains give a

62


ANT.1
MULTIPLEXER
SAME ANTENNA AS AT RIGHT
PRE-AMP.
IF AMP.
L.O.
600 V CH.
2400 V CH.
DOWN CONV. FILTER IF AMP.
600 CH. BLOCK
COMB. NTW.
COMB. NTW
TWT
MULTIPLEXER

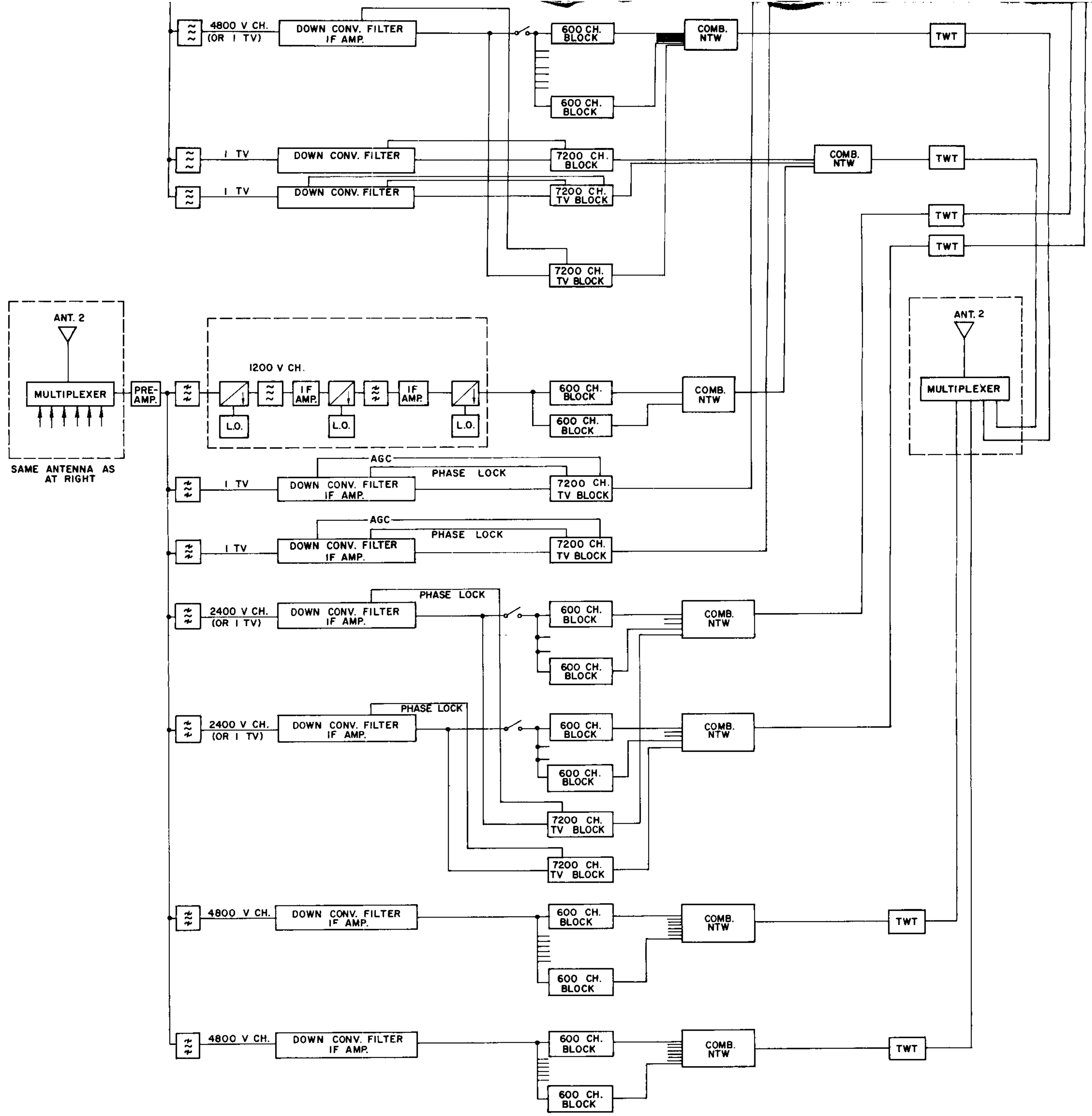


Fig. 26 ATLANTIC SATELLITE BLOCK DIAGRAM.

ratio of beam power of about 3/2; thus the European beam will need 6 tubes, giving a total of 10 traveling wave tubes.

3.7 Traveling Wave Tube Power Amplifier Characteristics

The radiated power for these high channel capacity satellites, in order to optimize the system costs, was found to be 90 W for the broadest satellite beam. Since the same output tube is to be used for all of the satellites, the bandwidth required for the maximum number of channels through a tube is 300 MHz. This high bandwidth and the long life reliability needed for a five-year satellite lead to the choice of a traveling wave tube as the output device. The available power and desired power output specified a tube with an output power of 90 W and efficiency of 39 percent.

Two types of tubes are required, the majority of which will be used in the 3.4- to 4.2-GHz band, and other types in the 7.25- to 7.75-GHz band. While the reliability of the TWTs is very high, a redundant system is desired. The difficulty in switching RF leads from a defective tube to a replacement led to the duplication of all TWTs. While no life figures are given for the associated power supply, failures are expected to be less than for the TWTs. From a space constraint on available volume in the communication platform, a package consisting of two tubes and a common power supply is proposed, with the power supply switched from one TWT to the other in the event of failure. The problems associated with high voltage switching are minimized by performing their operation with the power supply turned off. Using the pressurized package developed by EIMAC, a minimum amount of difficulty can be expected from this arrangement. Should further reliability data on TWTs indicate that the redundant tubes are not necessary, they can be readily left out of the package.

3.8 Telemetry and Control System

To operate a satellite such as is considered in this report, some commands must be sent from a control station on earth to the satellite. Also, certain metering measurements taken on the satellite are essential in deciding which commands to send, and must therefore be sent back to earth. The telemetry and command signals are handled through an omnidirectional antenna, during the initial phases and as part of the normal communications channels after earth acquisition.

A listing of the needed commands and telemetry signals (Table 14) shows a total of 75 commands and 113 telemetry signals. Since the requirements in this telemetry system do not exceed the present off-the-shelf hardware, a detailed block diagram of the telecommand system has not been prepared.

Table 14

ESTIMATE OF TELEMETRY AND COMMAND SIGNALS REQUIRED

Subsystem	Commands	Telemetry Signals Parameter	Sample Rate*
Power Supply	8 Temp cut off override 8 Battery turn (on/off)	8 Solar array temp	a,b
		4 Battery temp	a,b
		1 Solar array voltage	a,b
		1 Solar array current	a,b
		4 Battery voltage	a,b
		4 Battery current	a,b
		1 Regulator voltage	a,b
Attitude Control	6 Attitude thruster commands 2 Station-keeping thruster commands	1 Pitch	b
		1 Roll	b
		1 Yaw	b
		3 Reaction wheel r/min	a
		2 IR earth sensor	b
		1 Sun sensor	b
		1 Star tracker	b
		2 Interfer-ometer	b
		1 Rate gyro	b
		1 Temp control of electron-ic equipment	a
		1 Fuel meas	a
		1 Platform drive	b
		1 Motor speed	a
Communications	40 TWT Power and Heater (on/off) 10 TWT Changer (flip relay) (switches RF and TWT metering)	10 TWT heater voltages	a,b
		10 TWT anode voltages	a,b
		10 TWT collector voltages	a,b
		10 TWT helix voltages	a,b
		10 TWT helix current	a,b
		10 TWT cathode current	a,b
		10 TWT temp	a,b
Telemetry	1 Rate Control		a,b
TOTALS	75 Commands	113 Telemetry signals	

*Sample Rates: a. once per hr
b. once per min
a,b. capable of either rate.

Chapter 4

ECONOMIC ANALYSIS OF GROUND STATIONS

4.1 System of International Satellites and Ground Stations

This system consists of four satellites and a number of ground stations in several regions. The basis for selection of four satellites is given in Chapter 1. The system is more conveniently considered in terms of subsystems. A subsystem will consist of one satellite, the regions connected by the satellite, and the ground stations in the regions. Since the entire system will not be optimum unless the subsystems are optimal, the analysis will be with respect to the subsystems. Two types of analysis will be made. The first is based on a model which assumes that a given satellite is associated with two subregions. Two cases are considered. One considers a uniform traffic distribution; the other assumes "pockets" with concentrated traffic distributions. In this analysis, the number of ground stations that minimize the total cost of the system is determined.

In the second analysis, each region served by a satellite is assumed to have a given number of ground stations. This might be the case if, due to political reasons, each nation desires a ground station. Minimizing the cost of the entire subsystem leads to a description of each ground station and its cost.

4.2 Uniform Distribution Model

For definition, the model assumes that a need for 3600 two-way channels exists between two regions, each with an area of four million square miles. Internally, each region is assumed to have a uniform population density connected to ground stations by means of microwave links. The cost of the system is given by

$$C_s = C_{sat} + 2N(G + M) \qquad (1)$$

where

C_{sat} = satellite cost

G = ground station cost

M = microwave cost

N = number of ground stations in each region

C_s = system cost

This particular model is applicable to the North-South American satellite and to the Europe-Africa-Asia satellite. See Fig. 27.

Variables, parameters, and constants used to determine N, the number of ground stations to minimize cost, are defined as follows:

(1) The capacity $\underline{c}$ of each ground station equals the number of channels C divided by N.

(2) Each microwave link, associated with a ground station, carries $\underline{c}$ channels. The typical link for the case of only one ground station per region is defined as L and is 750 miles for a 2000 by 2000 mile region. For N ground stations, the typical link is $1/\sqrt{N}$.

For the case considered, $C = 3600$ two-way channels. The ground station costs used are summarized in Table 15.

Figure 28 shows the preamplifier, transmitter, and channelizing cost as a function of number of channels. Figure 29 shows the cost of the ground antenna as a function of size. The signal requirements for the FM modulation index selected in Section 3.1 set the ERP/channel plus antenna gain at 57.5 dBW. In determining the antenna and microwave preamplifier portion of the ground station cost, curves versus ERP/channel are plotted for the various preamplifier types. A composite curve showing the minimum cost is shown in Fig. 30. Based on satellite cost of $22 million and a total radiated power of 600 W, the cost per watt is $36,700.

The remaining item necessary to complete the cost analysis of the uniform distribution model is the microwave link cost. This cost may be represented by

$$M = (600 + 41)\underline{c} \qquad (2)$$

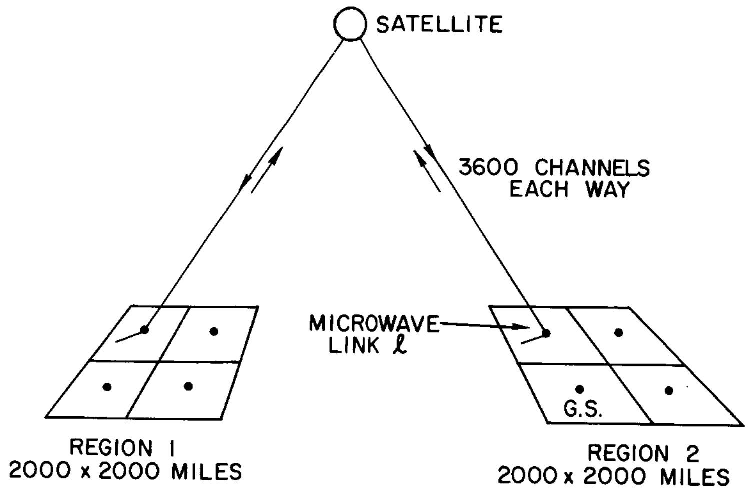


Fig. 27 TYPICAL PATH IN DISTRIBUTION MODEL.

Table 15

GROUND STATION COST*

Preamplifier	$100,000
Transmitter	150,000
Broadband Equipment	60,000 ($\underline{c}$)1/3
Hybrid Equipment	100 $\underline{c}$
Channelizing Equipment	1,000 $\underline{c}$ ($\underline{c}$ = no. of channels)

Ground Antenna Cost

Antenna Size	Gain	Installed Cost (Land, Bldg., Ant.)
15 ft dia	43 dB	$15,000
21	46	30,000
30	49	60,000
42	52	135,000
60	55	265,000

Additional Preamp. Cost (Front End)	Noise Temp.	Cost
Maser Amplifier	50° K	$350,000
12° Parametric Amplifier	85° K	60,000
77° Parametric Amplifier	110° K	40,000
Optimum Parametric Amplifier	150° K	10,000
Typical Parametric Amplifier	200° K	5,000

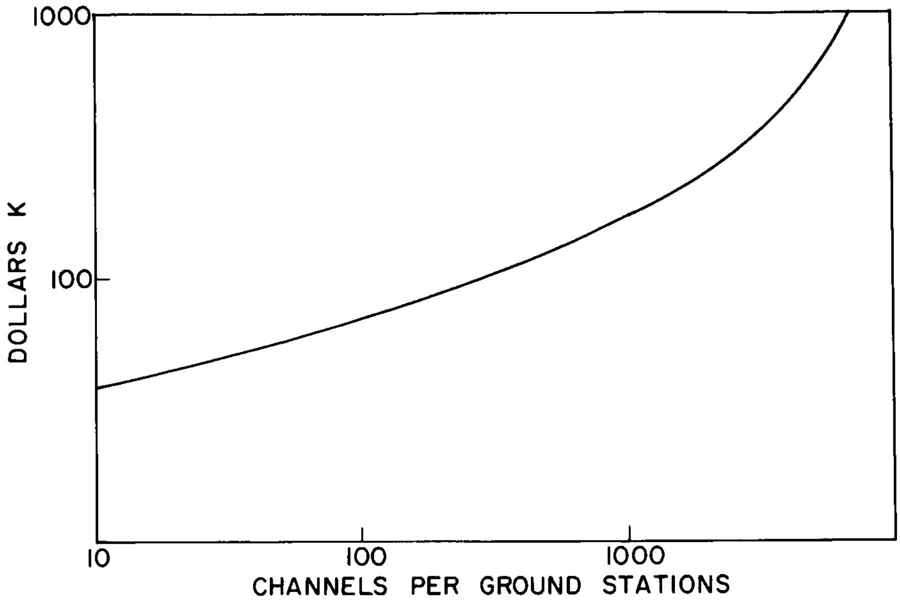


Fig. 28 COST OF PREAMPLIFIER, TRANSMITTER AND CHANNELIZING PER YEAR.

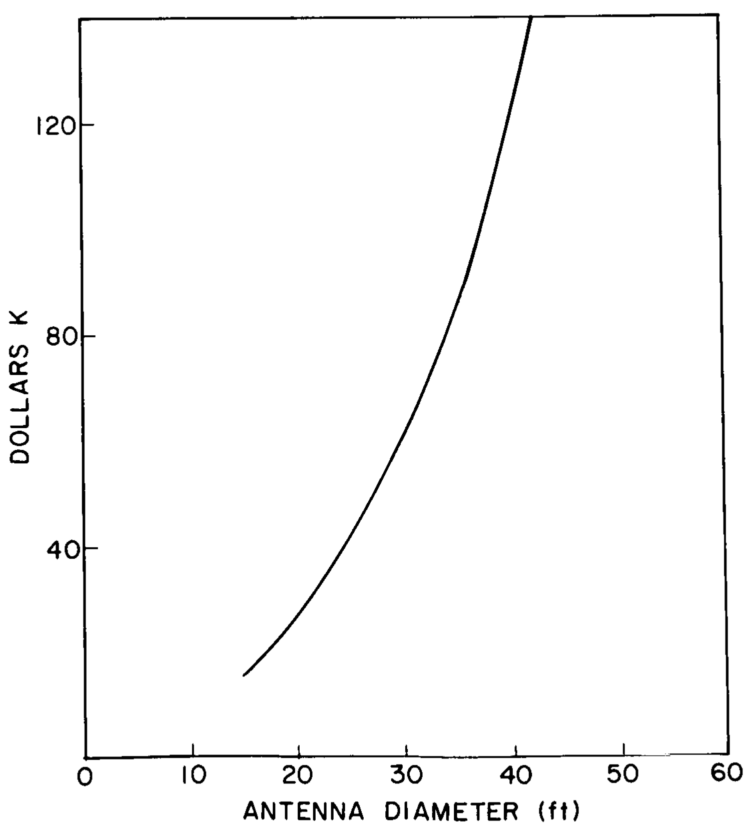


Fig. 29 COST OF GROUND ANTENNA SYSTEMS.

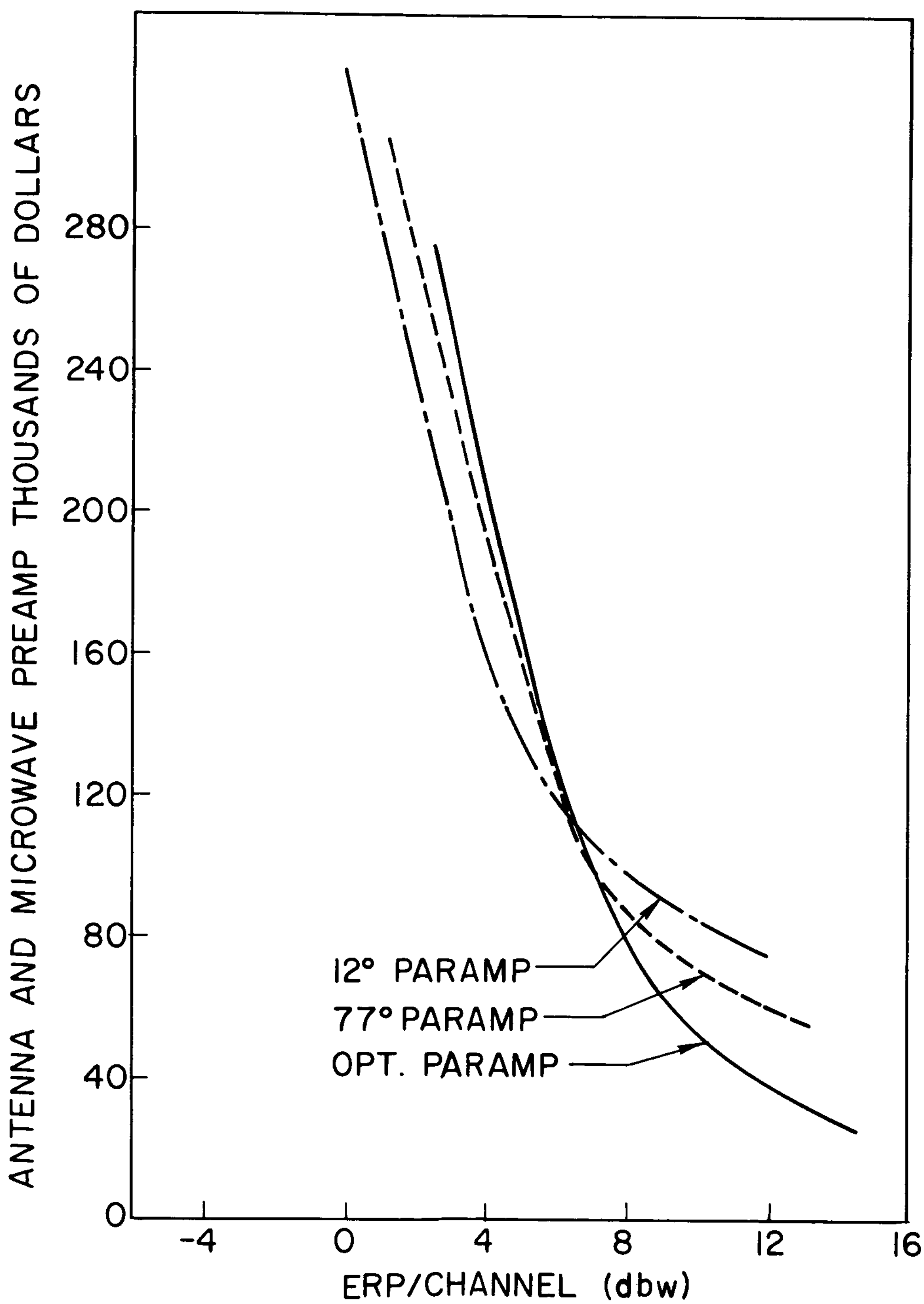


Fig. 30 GROUND STATION ANTENNA AND FRONT END COST COMPARISON.

where

M = cost in dollars per year of the microwave system

l = typical link length in miles

$\underline{c}$ = number of channels per ground station.

This cost equation assumes that international traffic shares the microwave links with local traffic. Detailed calculations are shown in the sections that follow. The optimum number of ground stations for this model will be 28 if the ERP/ch is 0 dBW and 32 if the ERP/ch is 10 dBW. Complete costs obtained from the sections that follow are shown in Fig. 31.

4.3 "Pocket" Distribution Model

In this model, the population density is assumed to be concentrated in pockets or centers. The microwave links in this case are much shorter. It is assumed that the microwave links are half as long as in the uniform density model. Cost figures and equations are the same as in the uniform distribution model. The optimum number of ground stations is 17 for an ERP of 0 dBW, and 23 for an ERP of 10 dBW.

4.4 Calculations for the Uniform Distribution Model

It is required that 3600 two-way or 7200 one-way channels connect two regions, 2000 miles by 2000 miles. The number of ground stations necessary to minimize the total system cost is desired. The satellite characteristics are assumed to be known. In this case, the radiated power is 600 W and the satellite antenna gain is 100(20 dB). $C = 7200$ channels and $L = 750$ miles (typical link for a one ground station case). The system cost is

$$C_s = C_{sat} + 2N(G + M) \ .$$

Since the satellite cost is fixed by the need, it is necessary to minimize the term $N(G + M)$. For $N = 1$ (one ground station per region),

$$\underline{c} = C/N = 7200 \text{ channels per ground station.}$$

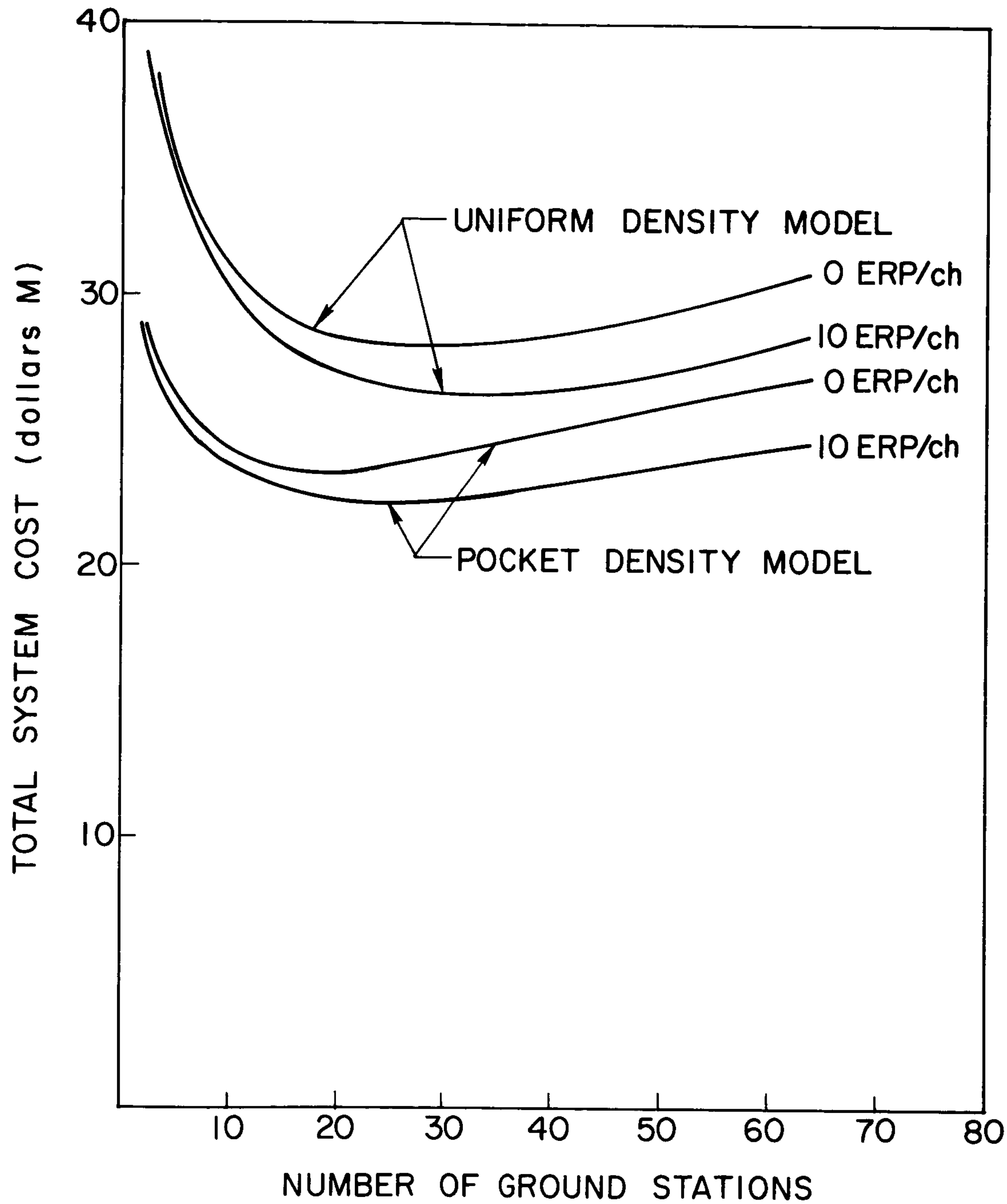


Fig. 31 COST OF UNIFORM DENSITY AND POCKET DENSITY MODEL.

The ground station cost (less antenna and front end) may be found from Table 15 or Fig. 28. In this case, the cost is \$9.55 million. The antenna and front end will cost \$0.325 million at an ERP/ch of 0 dBW, and \$.052 million at an ERP/ch of 10 dBW. Thus, due to the large channelizing cost, the ground station cost is almost independent of ERP/ch in this case. The microwave cost is obtained from Eq. (2).

$$M = (600 + 4 \cdot 750)\ 7200 = \$25.92 \text{ million per year.}$$

For

$N = 4$ (four ground stations per region)

$l = L/\sqrt{N} = 375$ miles

$\underline{c} = 1800$ channels per ground station

The cost of each ground station is

$G = \$3.325$ million at 0 ERP/ch, and \$3.052 million at 10 ERP/ch.

The microwave cost per year is

$$M = (600 + 4 \cdot 375)\ 1800 = \$3.78 \text{ million.}$$

Table 16 summarizes costs for various numbers of ground stations. The total ground station cost is divided by ten years to obtain the cost per year while the satellite cost is divided by five years to obtain the cost per year. The optimum cost for this model per two-way channel is approximately \$7800/year (\$28 million divided by 3600 two-way channels).

4.5 "Pocket" Distribution Model Calculations

As for the uniform distribution model, the cost is

$$C_s = C_{sat} + 2N(G + M) .$$

The only change in the procedure is the use of smaller typical link length. This will be taken as half the value used in the uniform density model. The requirements will be the same as those used for the uniform density model (3600 two-way channels and two 2000 by 2000 mile regions). Table 17 summarizes the results. The optimum cost for a two-way channel is approximately \$6400/year (\$23 million divided by 3600 two-way channels).

Table 16

SYSTEM COST AS A FUNCTION OF NUMBER OF GROUND STATIONS - UNIFORM DISTRIBUTION MODEL

(N)	(c) No. of Channels	(l) Miles	(G) 0 ERP/10 ERP	(M) Microwave Cost/Year per G.S.	Ground Cost/Year	Microwave Cost/Year	Satellite Cost/Year	Total Cost per Year
1	7200	750	$9.87M/9.62M	$25.92M	$1.974/1.924	$51.84M	$4.4M	$58.21M/58.16M
4	1800	375	3.32M/3.05M	3.78M	2.65M/2.44M	30.24M	4.4M	37.09M/36.08M
16	450	187.5	1.52M/1.25M	0.61M	4.86M/4.0M	19.5 M	4.4M	28.76M/27.90M
36	125	125	1.14M/0.87M	0.22M	8.20M/6.26M	15.8 M	4.4M	28.40M/26.46M
64	112	93.75	0.97M/0.70M	0.11M	12.4M/8.95M	14.1 M	4.4M	30.90M/28.65M

Table 17

SYSTEM COST AS A FUNCTION OF NUMBER OF GROUND STATIONS - "POCKET" DISTRIBUTION MODEL

(N)	Channels	(l) Miles	(G) 0 ERP/10 ERP	(M) Microwave Cost/Year per G.S.	Ground Cost/Year	Microwave Cost/Year	Satellite Cost/Year	Total Cost per Year
1	7200	375	$9.87M/8.62M	15.1 M	$1.974M/1.924M	$30.24M	$4.4M	$36.61M/36.56M
4	1800	187	3.32M/3.05M	2.43 M	2.65 M/2.44 M	19.5 M	4.4M	26.55M/26.34M
16	450	93	1.52M/1.25M	0.44 M	4.86 M/4.0 M	14.1 M	4.4M	23.26M/22.5 M
36	200	62	1.14M/0.87M	0.17 M	8.2 M/6.26 M	12.2 M	4.4M	24.8 M/22.86M
64	112	46	0.97M/0.70M	0.088M	12.4 M/8.95 M	11.2 M	4.4M	27.0 M/24.55M
100	75	37.5				10.8 M		

4.6 A Model To Determine the Optimum ERP/Channel

In this model, the number of ground stations for a given region is fixed. The microwave link distribution may be uniform or concentrated in "pockets." The objective will be to minimize the cost of the total system, where the communications requirements of the regions associated with a satellite are specified. This model will yield, in addition to cost, the optimum satellite power, optimum ground antenna size, and optimum microwave receiver.

4.7 Calculation of ERP/Channel Model

The need will be assumed as 3600 two-way channels. The regions will be the same as in the previous models (2000 by 2000 miles). It will be necessary to know the satellite cost as a function of power. This is shown in Fig. 32. It will be convenient to use effective radiated power per channel (ERP/ch) as a parameter. Figure 33 shows satellite cost per year for two-way channel requirements of 3600 and 7200. The ground station cost is a function of both the number of channels and the ERP/ch. The antenna and microwave preamplifier portion of the ground station is a function of ERP/ch, and this cost per year is shown in Fig. 34. This is the result of optimizing the curves of Fig. 30. As may be noted, if the ERP/ch is greater than 6.5 dBW, the minimum cost will be obtained by using the optimum parametric preamplifier, which has a noise temperature of 150°K. If the ERP/ch is less than 6.5 dBW, the 12°K parametric amplifier, which has a noise temperature of 85°K, is optimum. The channelizing and microlink cost is obtained in the same manner as previously. Results are shown in Table 18.

The 100 Ground Station System has an annual cost of approximately $7650 per two-way channel while the 36 Ground Station model has a cost of $5900 per two-way channel for the required 3600 channels. Note that most of this cost is in the ground system, dominated by microwave and channelizing equipment. The satellite cost per two-way channel per year is only $750 for the 100 Ground Station System and $510 for the 36 Ground Station System.

The incremental costs of Table 18 permit the determination of the optimum ERP/channel. Curves plotted from this data are shown in Figs. 35

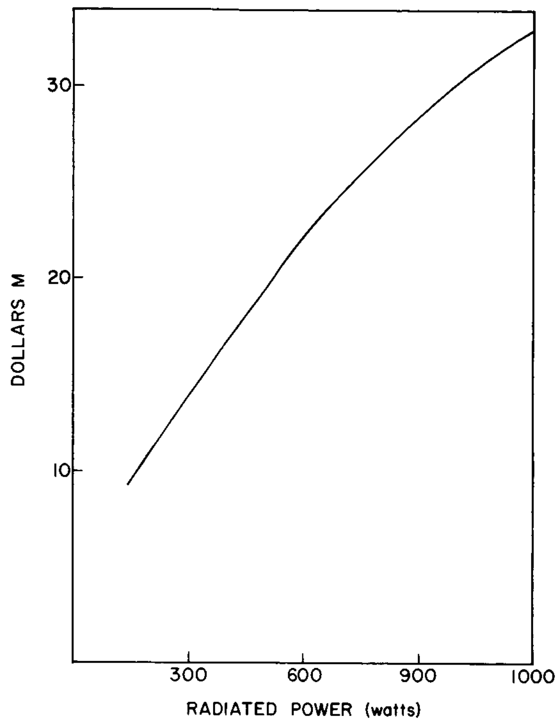


Fig. 32 TOTAL SATELLITE COST (INCLUDING LAUNCH).

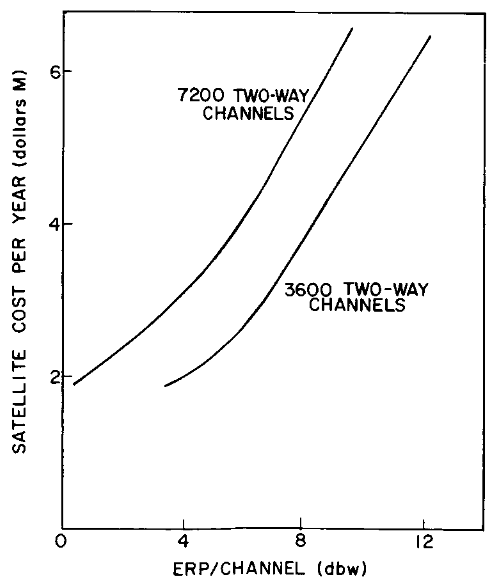


Fig. 33 SATELLITE COST VS ERP PER CHANNEL.

Table 18

SYSTEM COST - ERP/CHANNEL MODEL
(3600 Two-Way Channels)

Number of Ground Stations	ERP/Ch (dBW)	Satellite Cost/Year	Antenna and Front End Cost/Year	Channelizing Cost/Year	Micro-link Cost/Year	Total Cost/Year
100	0	$1.2 M	$6.5 M	$11.6 M	$10.8 M	$30.1 M
	3	1.84 M	3.9 M	11.6 M	10.8 M	28.14 M
	6	2.7 M	2.4 M	11.6 M	10.8 M	27.5 M
	9	4.6 M	1.3 M	11.6 M	10.8 M	28.3 M
36	0	1.2 M	2.34 M	5.84 M	12.2 M	21.58 M
	3	1.84 M	1.4 M	5.84 M	12.2 M	21.28 M
	6	2.7 M	0.86 M	5.84 M	12.2 M	21.61 M
	9	4.6 M	0.45 M	5.84 M	12.2 M	23.09 M

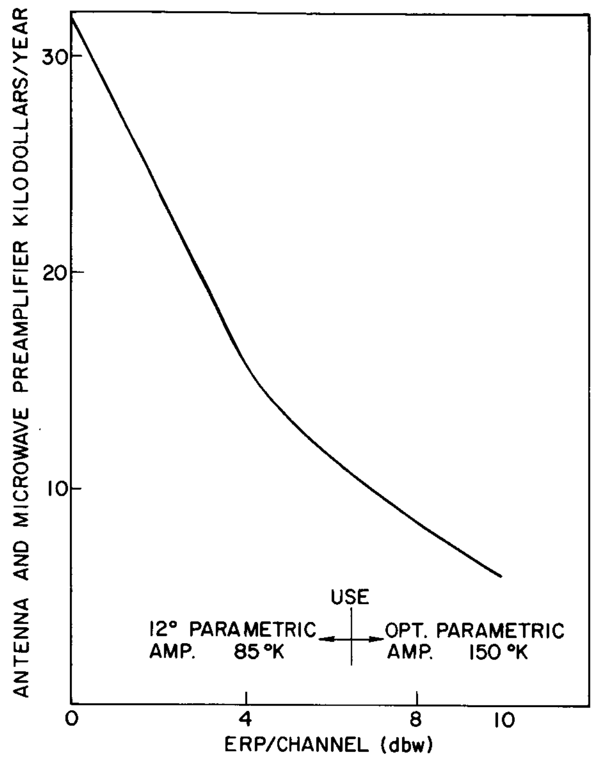


Fig. 34 OPTIMUM FRONT-END CHOICE.

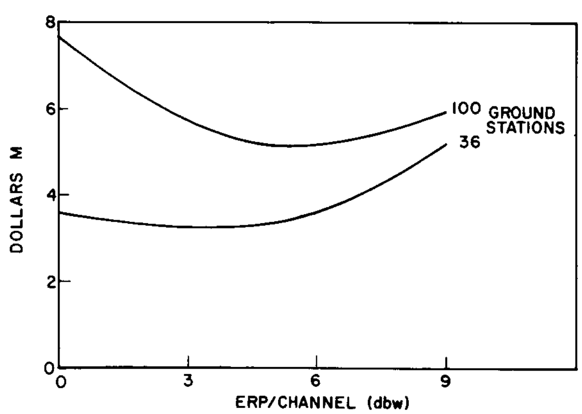


Fig. 35 ANTENNA AND MICROWAVE PREAMPLIFIER COSTS.

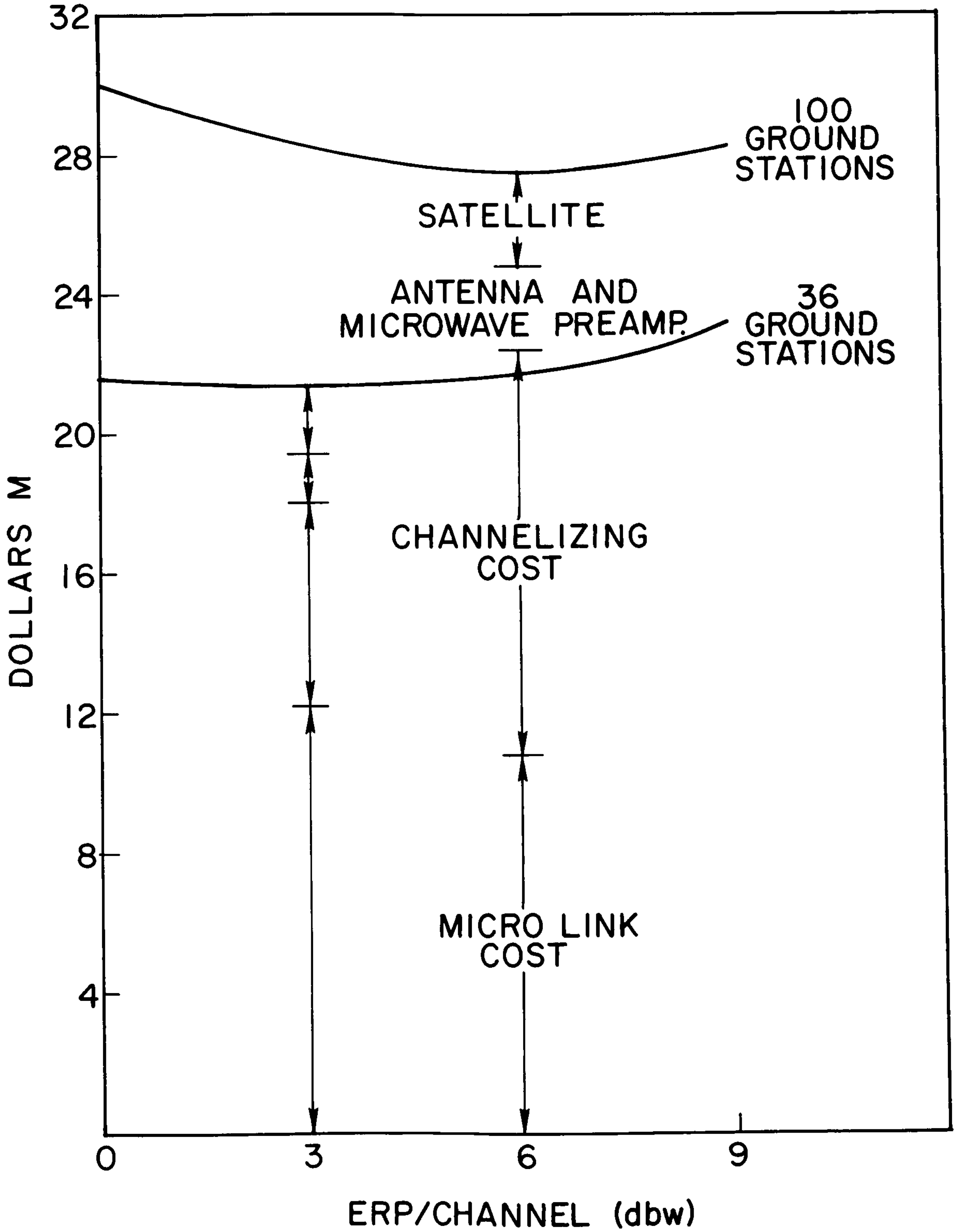


Fig. 36 TOTAL SYSTEM COST PER YEAR.

and 36. For 100 ground stations, the optimum ERP/ch is 6 dBW (four watts per one-way channel). Since the satellite antenna gain is 100 (20 dB), this corresponds to 288 W of radiated power. For 36 ground stations, the optimum ERP/ch is 3 dBW, corresponding to 144 W of radiated power.

If the capacity is doubled (7200 two-way channels), Table 19 summarizes the costs for a 100-ground station system. The results of this model are summarized in Table 20.

4.8 Minimum Distances for Regional Traffic via Satellites

In this analysis, the cost associated with a region (2000 miles by 2000 miles) using the satellite is compared with the cost of using microwave links. It will be assumed that there is a need for 7200 two-way channels. The operating ERP/channel will be taken as 3 dBW (2 W/channel or 288 W for 14,400 one-way channels). The cost will be a function of the number of ground stations. The model used in the analysis considers the cost via satellite to consist of satellite cost plus the cost of microwave links to each ground station. This cost is compared to the cost necessary to connect two adjacent ground stations by means of microwave links.

The results of this analysis, which are shown in Table 21, indicate that it is more economical to use the satellite system for distances greater than 260 miles for a two-way channel requirement of 7200. Also see Fig. 37. A similar analysis for 3600 two-way channels gives a break point of 320 miles. Refer to Fig. 38.

Table 19

SYSTEM COST - ERP/CHANNEL MODEL
(7200 two-way channels)

ERP/Ch (dBW)	Satellite Cost/Year	Antenna, Front End, Channelizing Cost/Year	Microlink Cost/Year	Total Cost/Year
0	$1.84 M	$29.7 M	$21.6 M	$53.14 M
3	2.7 M	27.1 M	21.6 M	51.4 M
6	4.6 M	25.6 M	21.6 M	51.8 M
9	6.3 M	24.5 M	21.6 M	52.4 M

The results of this model may be summarized by Table 20.

Table 20

SUMMARY OF POWER AND CHANNEL COST - ERP/CHANNEL MODEL

Number/ Ground Stations	Preamp	Ground Antenna	Two-way Channels Required	ERP/Ch for Min-Cost	Watts One-way Channel	Total Watts Radiated	Cost Two-way Ch/Year
36	12°K Para	42 ft dia	3600	3 dBW	2	144	$5900
100	12°K Para	30 ft dia	3600	6 dBW	4	288	7650
100	12°K Para	42 ft dia	7200	3 dBW	2	288	7130

Table 21

COST VIA SATELLITE SYSTEM AND VIA MICROWAVE SYSTEM

Link (miles)	No. of G.S.	Distance Between G.S.	Sat. Cost/ Year	Ant. & Micro Pre Amp Cost/Year	Channel-izing Cost/ Year	Micro-link Cost/ Year	Via Satel-lite Cost/ Year	Via Micro-wave Cost/ Year
250	4	1000	$2.7M	$0.08M	$1.2M	$30.2M	$34.18M	$66.24M
187	16	500	2.7M	0.32M	1.9M	19.4M	24.32M	37.40M
200	36	333	2.7M	0.72M	2.9M	15.8M	22.12M	27.80M
112	64	250	2.7M	1.28M	4.6M	14.0M	22.58M	23.00M
75	100	200	2.7M	2.00M	5.8M	13.0M	23.50M	20.20M

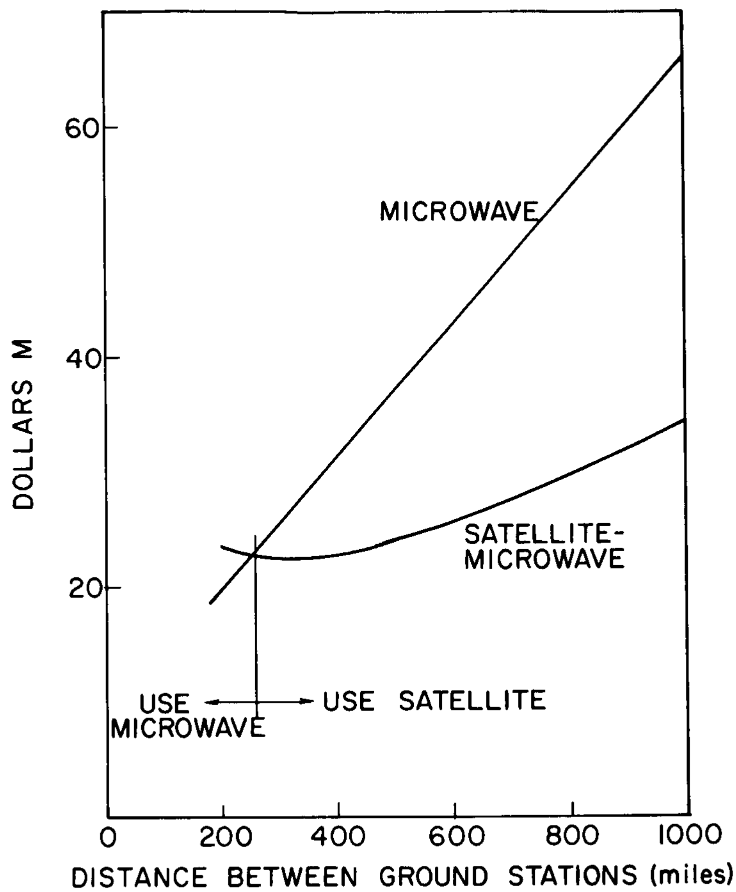


Fig. 37 COMPARATIVE COSTS OF MICROWAVE AND SATELLITE SYSTEMS TO HANDLE 7200 TWO-WAY CHANNELS.

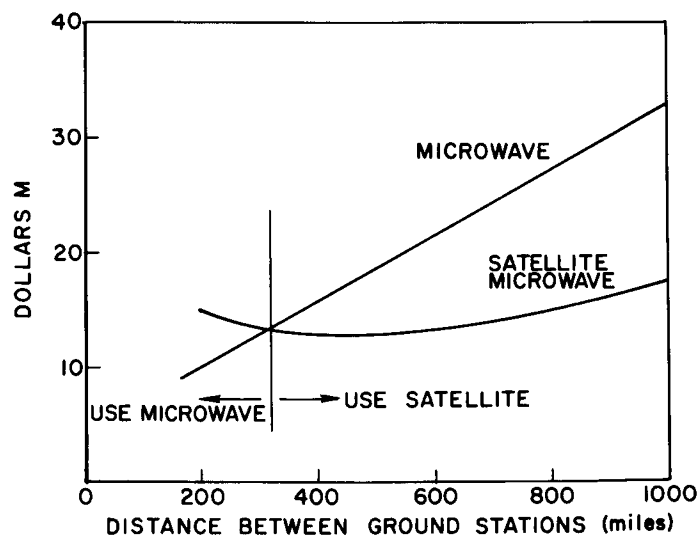


Fig. 38 COMPARATIVE COSTS OF MICROWAVE AND SATELLITE SYSTEMS TO HANDLE 3600 TWO-WAY CHANNELS.

Chapter 5

MULTIPLE ACCESS

5.1 Techniques for Multiple Access

When several ground stations communicate with one another through a satellite, each of the ground stations must have access to the facilities of the satellite. If the number of ground stations is relatively small (fewer than 20) as in present and projected INTELSAT systems, providing access for each of the ground stations is not very difficult. However, to allow hundreds of ground stations to communicate with one another may require more than routine methods. To see why this is the case, one need only to look at the number of voice circuits required for N ground stations to talk with one another. It is easy to show that the total number of connections required is

$$\text{number of connections} = \frac{N(N-1)}{2} , \tag{3}$$

and if each connection carries M channels, then

$$\text{number of circuits} = \frac{N(N-1)M}{2} . \tag{4}$$

The number of circuits varies approximately at the square of the number of ground stations. The principal difficulties occur when service is to be provided to a large number of ground stations because, as the number of ground stations becomes large (assuming a constant number of conversations), the traffic density, at each ground station drops accordingly. As shown in Section 5.2, low traffic densities require a disproportionate channel allocation to maintain the probability of losing a call at a level of .03.

Three general types of multiple access were investigated: frequency division multiple access (FDMA), time division multiple access (TDMA), and spread spectrum multiple access (SSMA). Assuming that a large number of ground stations would require access, TDMA and SSMA were rejected because they are difficult to synchronize and because the equipment complexity is prohibitive economically for a small (light traffic) ground station. The remainder of this chapter will be devoted to four FDMA schemes that were considered.

First, the simplest technique, fixed point-to-point channel assignment, was considered. Assuming a total available frequency bandwidth of 1300 MHz and 65 kHz required for the FM downlink, using Eq. (3), the maximum number of ground stations each with M/2 one-way channels can be determined. The maximum number of ground stations that can be provided access for various values of M are summarized below.

M	N max
1	141
2	100
4	71
10	45
20	32
50	21
100	15
1000	5

In addition to fixed point-to-point access, three other techniques were considered: fixed point-to-point frequency assignment with groups of ground stations sharing a common spectrum, assigned receive frequencies, and demand assignment of frequencies.

Fixed point-to-point assignment with sharing is accomplished by assigning a frequency band to a group of stations for communication through the satellite to another group of ground terminals. Communication among members of the same group can also be accomplished by sharing a frequency band.

By assigned receiver frequency, each ground station is assigned one or several receive-transmit frequency pairs on which calls initiated by some other ground station are carried. To initiate a call, a ground station must tune his transmit-receive frequency pair to the frequencies which correspond to one of those assigned to the destination of the call.

Demand assignment is accomplished by having a central computer that controls the access of each of the participating ground stations. To complete a call, the initiating ground station notifies the central computer of its desire to make a call and the destination of the call. Demand assignment with the computer on board the satellite was also considered, but abandoned when it became clear that the equipment involved was essentially of the same complexity as a ground based computer without any additional flexibility.

These four methods for multiple access were compared, using as criteria:

(1) the number of frequency allocations to satisfy a specified traffic demand

(2) the complexity of the equipment required for implementation.

In Section 5.2, a probable argument is presented which allows the determination of the number of frequency allocations required to satisfy a specified traffic demand with a probability of .03 that a call is rejected because all frequencies allocated are occupied.

5.2 The Relationship Between Loads and Channel Requirements

For given call arrival rates, the number of two-way voice channels required to handle the traffic was obtained by a consideration of the stochastic process. Since the arrival rate varies in time, and hence leads to a nonstationary process, only the arrival rate at the peak traffic hours was considered. Since the channel allocation must handle the peak load with only 3 percent call rejection, only the peak load was considered to be independent and the number of arrivals in any time period independent of the number of arrivals in any other nonoverlapping time period. These assumptions lead to the Poisson distribution of arrivals, where the probability of the number of arrivals in period T, $P_n(t)$ is given by

$$P_n(T) = \frac{(\lambda T)^n \quad e^{-\lambda T}}{n!} \qquad \begin{matrix} 0 \leq n \\ \\ 0 \leq T \end{matrix}$$

where λ is the mean number of arrivals/time. Call lengths were considered to follow an exponential density function given as follows:

$$p(x) = \mu e^{-\mu x} \qquad \begin{matrix} \mu > 0 \\ x \geq 0 \end{matrix}$$

where

$$\mu = 1/\text{mean length of calls} = 1/12 \text{ min}^{-1} .$$

The state probabilities of the queueing system were calculated (see Appendix B) for delayed calls lost rather than waiting. The probability of a call's not being immediately serviced was calculated and set equal to .03, and from this $K(\lambda)$ was determined, where K is the number of channels required for 3 percent loss. Figures 39-42 and Table 22 show K as a function of Q, where

$$Q = \frac{\lambda}{\mu} \quad \text{for} \quad P \approx .03 . \tag{5}$$

These calculations are for steady state conditions, which can be obtained if the peak traffic is achieved gradually, as is generally the case.

Using the K, Q relationship, we can convert a point-to-point traffic matrix of K_{ij} to Q_{ij}. Then if Q_T

$$Q_T = \sum_i \sum_j Q_{ij} \qquad \text{for} \quad i > j ,$$

and then

$$K(Q_T) = K_0 \sum_i \sum_j K_{ij}, \qquad \text{where} \quad i > j .$$

K corresponds to the demand access channel requirements and

$$\sum_i \sum_j K_{ij} ,$$

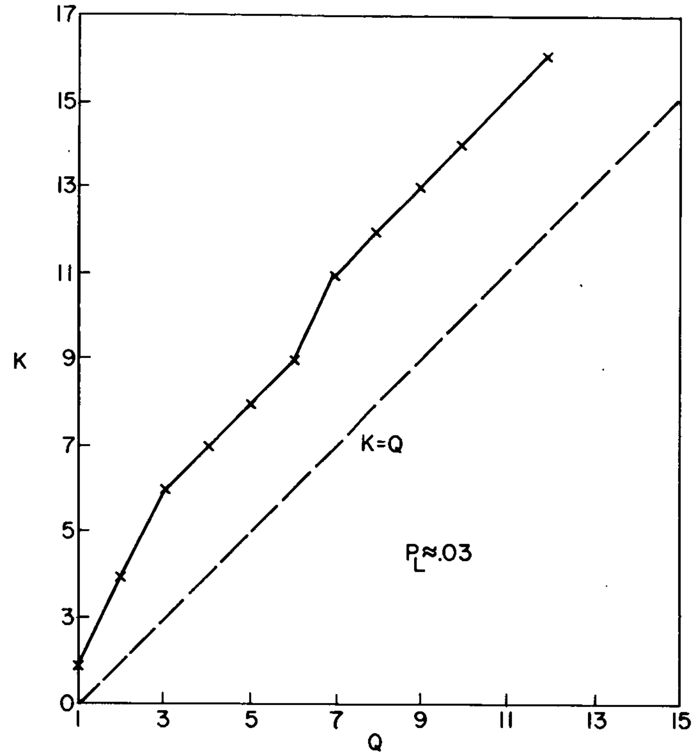


Fig. 39 REQUIRED NUMBER OF CHANNELS VS ARRIVAL RATE.

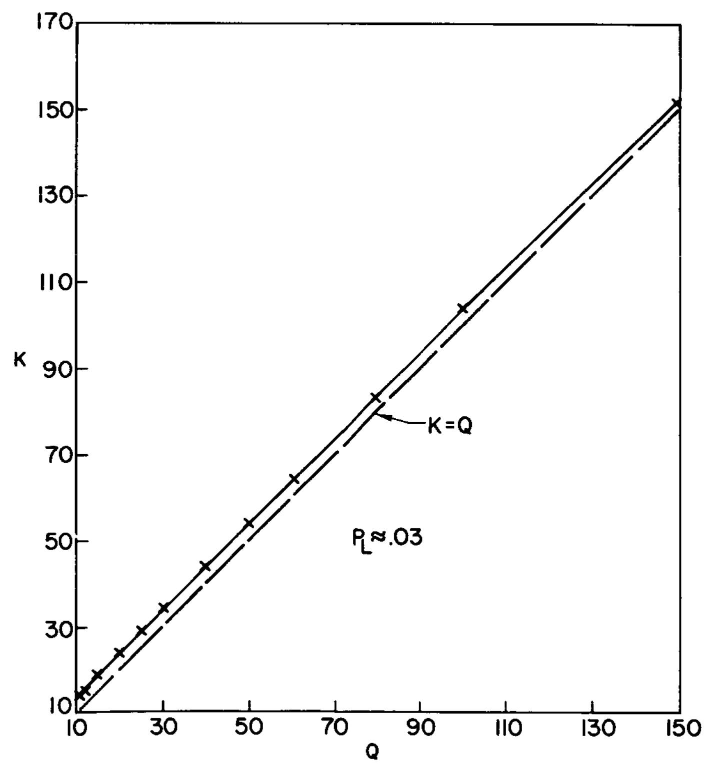


Fig. 40 REQUIRED NUMBER OF CHANNELS VS ARRIVAL RATE.

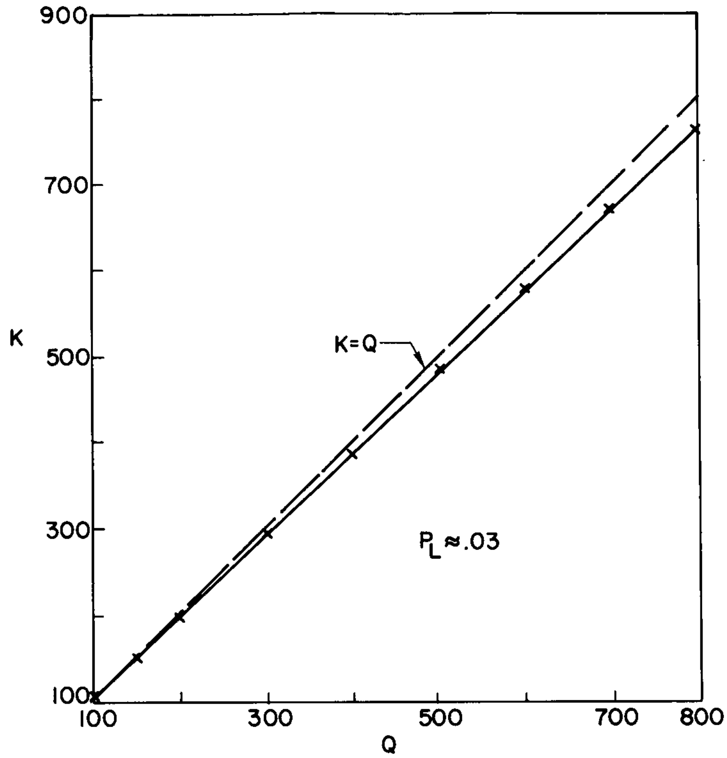


Fig. 41 REQUIRED NUMBER OF CHANNELS VS ARRIVAL RATE.

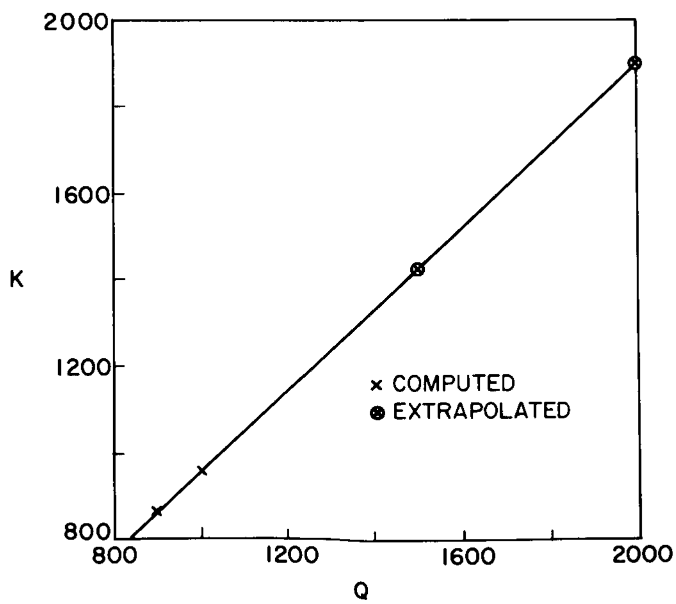


Fig. 42 REQUIRED NUMBER OF CHANNELS VS ARRIVAL RATE.

Table 22

NUMBER OF CHANNELS (K) FOR 3 PERCENT CALL LOSS

Q	K	Q	K	Q	K
	2	12	16	150	151
2	4	15	19	200	198
3	6	20	24	300	293
4	7	25	29	400	388
5	8	30	34	500	483
6	9	40	44	600	577
7	11	50	54	700	672
8	12	60	64	800	766
9	13	80	83	900	860
10	14	100	103	1000	954

Table 23

k* MATRIX: A HYPOTHETICAL TRAFFIC LOAD

Station	1	2	3	4	5	6	(i)
1	X	11	8	18	6	4	
2	11	X	9	8	11	50	
3	8	9	X	4	2	40	
4	18	8	4	X	1	22	
5	6	11	2	1	X	17	
6	4	50	40	22	17	X	
(j)							

*k = number of conversations at peak load.

Table 24

Q MATRIX: NUMBER OF CALL ARRIVALS PER UNIT TIME X AVERAGE CALL DURATION TIME (q)

Station	1	2	3	4	5	6	(i)
1	X	7	5	14	3	2	
2	7	X	6	5	7	46	
3	5	6	X	2	1	36	
4	14	5	2	X	.2	18	
5	3	7	1	.2	X	13	
6	2	46	36	18	13	X	
(j)							

for $i > j$ is the point-to-point requirement. K_0 is smaller because advantage may be taken of the statistics associated with the larger Q, and the 3 percent loss for the point-to-point assignment is cumulative and the safety factor is redundant.

This technique may also be applied to grouping of point-to-point assignments where the total Q is equal to the sum of individual Qs, and the total K would be less than the sum of the individual Ks.

5.3 A Case Study of Several Multiple Access Techniques

To compare the four FDMA techniques described in Section 5.1, each of these methods was applied to some traffic patterns and also examined to determine the complexity of the equipment required for implementation.

To determine the frequency allocations required to satisfy a specified traffic pattern, a table of values of k as a function of q obtained from Eq. (5) was used. To see how this table is employed, we will consider the hypothetical traffic matrix shown in Table 23. The ij-th element of the matrix is the number of conversations between ground stations i and j at the peak period of traffic. Notice that the traffic matrix is symmetric. The matrix of values of q which corresponds to this traffic matrix is shown in Table 24. It is assumed that the traffic peaks all occur at the same time--a worst case situation.

To find the number of frequency/allocations required for fixed point-to-point access, one simply sums the above diagonal elements of the K matrix to obtain the result that 211 channels are required. Notice that even if the traffic matrix contained several zeros, these would require the allocation of one channel; otherwise access between these stations would be impossible.

The number of channels required for fixed point-to-point access with sharing is found by partitioning the Q matrix into submatrices of dimension equal to the group size. For example, suppose that each ground station shared circuits with two other ground stations. Assume that the groups are arbitrarily designated as stations 1-2-3 (Group I) and stations 4-5-6 (Group II). To find the number of channels required for calls within Group I, sum the above diagonal elements of the 3×3 upper left submatrix of Q to obtain 18. This number is proportional to the number of calls initiated within Group I destined for other members of Group I. The number of channels allocated for calls within Group I is the value of k which corresponds to $q = 18$. This value is found by an interpolation in Table 22 to be 22. For calls between Groups I and II, the elements of the 3×3 submatrix located in the upper right corner of the Q matrix are summed to give 116. Converting this value to the number of channels required gives 119. Using the elements above, the main diagonal of the lower right 3×3 submatrix of Q gives a "q" summation of 31.2, which indicates that 36 channels are required for calls within Group II. Thus, by sharing frequency allocations, the total number of channels required is $22 + 119 + 36 = 177$. A group size of three may not be optimum, or perhaps the members of a group could be selected more judiciously, but even such an arbitrary grouping reduces the channel requirements from 211 to 177.

The third FDMA technique considered is the assignment of receiving frequencies. To calculate the number of frequencies which must be assigned, first form a new K matrix by halving each of the elements of the original K matrix. This is required because it is assumed that half of the calls engaged in by two ground stations are initiated by each of the participating stations. The adjusted K matrix and the corresponding Q matrix are shown in Table 25 and 26, respectively. To find the value of q corresponding to the arrival of incoming calls at station 1, sum the elements of the first row (or first column) of the adjusted Q matrix. This yields a sum of 13.25, which corresponds to 18 (rounded to next highest integer) channels required for calls destined for station 1. Continuing this procedure yields the required incoming channels for stations 2 through 6 as 35, 26, 21, 15, 55, respectively, or a total of 170 required assigned frequencies. Twice this number of channels are required for the conversations, but only half of these require preassigned frequency channels.

Finally, consider the system which is the ultimate in flexibility, but the most complicated in terms of implementation: demand assignment of frequency channels. To find the required number of channels sum the above-diagonal elements of the Q matrix in Table 24 to obtain 165.2, which indicates that if demand assignment is used, 167 frequency channels are required. The required frequency assignment for each of the four techniques with the assumed traffic pattern is summarized below.

Table 25

ADJUSTED K MATRIX

Station	1	2	3	4	5	6
1	X	5.5	4	9	3	2
2	5.5	X	4.5	4	5.5	25
3	4	4.5	X	2	1	20
4	9	4	2	X	.5	11
5	3	5.5	1	.5	X	8.5
6	2	25	20	11	8.5	X

Table 26

ADJUSTED Q MATRIX

Station	1	2	3	4	5	6
1	X	2.75	2	6	1.5	1
2	2.75	X	2.25	2	2.75	21
3	2	2.25	X	1	.2	16
4	6	2	1	X	.1	7
5	1.5	2.75	.2	.1	X	5.5
6	1	21	16	7	5.5	X

Access Technique	Number of Assigned Frequency Channels
Fixed point-to-point	211
Sharing (group size = 3)	177
Assigned receiver frequencies	170
Demand assignment	167

The procedure illustrated in the preceding example was also applied to some predicted traffic patterns. Fixed point-to-point assignment and demand assignment are extreme cases of point-to-point assignment with sharing. Sharing with a group size of one is fixed point-to-point assignment, and sharing with a group size of N (where N is the total number of ground stations) results in demand assignment. Demand assignment requires the minimum number of frequency channels regardless of the traffic pattern and fixed point-to-point assignment requires the maximum number of channels.

The selection of one of the four access techniques involves a tradeoff between conservation of available frequency spectrum and the complexity of equipment required. The intuitive notion that efficient utilization of the channel capacity of the satellite requires a large expenditure for ground equipment can be readily verified by listing the necessary equipment. The question the case studies attempts to answer is whether or not the cost of sophistication required for ultimate efficiency (demand access) is justified by the resulting saving in frequency allocation. To illustrate the extremes involved, the traffic patterns used were estimates for regional African traffic in 1975[1] characterized by many (40) ground stations with light traffic and European regional traffic in 1968--28 ground stations with heavy traffic.[1] The results of applying the foregoing procedure to the traffic patterns suggested by CCIR/CCITT to African and European traffic are summarized below.

Africa:

Table 27

NUMBER OF FREQUENCY ASSIGNMENTS REQUIRED FOR AFRICAN REGIONAL TRAFFIC IN 1975

Access Technique	Number of Assigned Frequency channels
Fixed point-to-point	944
Sharing (groups of 5)	357
Sharing (6 groups of 6, one group of 4)	345
Assigned receive frequencies	346
Demand assignment	212

Table 28

NUMBER OF FREQUENCY ASSIGNMENTS REQUIRED FOR EUROPEAN REGIONAL TRAFFIC IN 1968

Access Technique	Number of Assigned Frequency Channels
Fixed point-to-point	20,038
Demand assignment	19,013

For all of the preceding cases, it was assumed that the peak loads occur at the same time. However, if this is not the case, then each element of the Q matrix should be multiplied by a factor β, $0 < \beta \leq 1$, which accounts for nonsimultaneous peaks. For example, if $\beta = .8$, then in the demand assignment example, the total q value is $165.2 \times .8 = 132.2$ which corresponds to a channel requirement of 135. For most cases, β will be nearly 1.0 since the traffic maxima are quite broad and time shifts of two or three hours will not greatly reduce the required peak loads.

The results given for African traffic for sharing were found without strenuous efforts to optimize group sizes or members of groups. For the European traffic, only fixed point-to-point assignment and demand assignment were considered; the other two schemes would yield numbers which are between these two extreme cases.

In the African example, the inefficiency of fixed point-to-point access is apparent; however, as the density becomes large, as is the case for the European example, even fixed point-to-point access, the crudest of the techniques considered, becomes feasible. An additional factor is that European regional traffic in 1968 is more typical of the type of traffic patterns which will emerge as the underdeveloped nations enter into more commerce. This trend indicates that the frequency savings resulting from complete demand assignment will be of marginal value as communication requirements grow.

5.4 Equipment Required for Implementation of Multiple Access

In the following sections, the differences in the ground equipment required for each of the four multiple access techniques are described. Once again, the techniques considered are

(1) fixed point-to-point frequency assignment,

(2) fixed point-to-point frequency assignment with grouping of the ground stations,

(3) assigned receive frequencies,

(4) demand assignment.

The additional channelizing costs for multiple access equipment for the ground stations using techniques 2, 3, and 4, will be referenced to the cost of implementing the fixed point-to-point frequency assignment technique (1). The cost of technique 1 is taken as unity.

5.4.1 Fixed Point-to-Point Frequency Assignment with Grouping of the Ground Stations

The additional expenditure over a fixed assignment is due to

(1) A signal oscillator that will transmit a tone outside the voice information band (less than 300 Hz but higher than 100 Hz. Refer to Section 6.2). This tone is a characteristic sign of the special ground station; it changes from ground station to ground station, and serves as a busy signal for all other ground stations of the same group.

(2) A signal combiner that adds the voice signal and the tone signal to feed the modulating equipment.

(3) A filter that picks up the busy signal of other ground stations at a frequency band described above.

(4) Equipment that informs the user of the busy status of the ground station he wants to use for a call. (This is a device which is distinct from the busy equipment that is active when the receptor of the call is busy.)

An estimate of the cost of the above equipment would add between 5 and 10 percent to the cost of a fixed assigned nonshared channel. Thus the cost of this scheme is given by

$$C_{(2)} \cong [1.05 - 1.1] \cdot C_{(1)} .$$

5.4.2 Assigned Receive Frequencies

In this case, each ground station has a set of frequencies on which this station can be called.

The additional equipment is

(1) Several transmitter-receiver pairs that have access to the satellite and that may be either tuned continuously or selected by switching. (The complexity is less when using transmitter-receiver pairs with fixed frequencies which are switched.)

(2) A device that will analyze the dialing numbers to check what ground station is being called and what frequency pair is allocated to this ground station.

(3) Equipment that will check the frequency status of the transceivers at the calling ground station and that will perform a frequency comparison between this and the above mentioned information according to frequency.

(4) A logical decision box that will cause the tuning or switching of the several transceivers of the calling ground station.

(5) A busy tone analyzer that causes tuning or switching to occur until a free frequency pair is found or gives a busy signal to the user when all frequencies or receptors are busy.

An estimate of this cost in comparison to the electronic equipment of ground station number 1 is

$$C_{(3)} \approx [1.3 - 1.4] \cdot C_{(1)} .$$

5.4.3 Demand Assignment

This is the most complex case of the techniques examined. Channel allocation is made only upon request. The consequence is that the call information has to be stored in a central computer that decides upon the frequency allocation. In this computer, the frequency status of all ground stations must be known.

The additional equipment is, in principle, the same as that mentioned in the "assigned receive frequency pair" scheme except now both caller and called must have frequency-variable transceivers. In addition, a central computer or one computer per ground station is needed. The central computer scheme seems to be less complex than the solution with a computer per each ground station.

An estimate of this cost relative to the electronic equipment of ground station number 1 is

$$C_{(4)} \approx [1.8 - 2] \cdot C_{(1)} .$$

5.5 Conclusions

Based on the necessary frequency allocation and the equipment required for implementation, fixed point-to-point frequency assignment is considered most advantageous between ground stations with high density traffic requirements, while group sharing is worthwhile for stations with low traffic densities. The small increase in frequency utilization is not warranted by the large cost of channelizing equipment for the more efficient assignment schemes. The recommended procedure for servicing a specified traffic pattern matrix is

(1) to remove all point-to-point connections requiring more than 5 circuits from the traffic matrix and to allocate these on a fixed point-to-point basis.

(2) to partition the reduced traffic matrix into groups of size S and determine the circuit requirements of each group.

The parameter S can be optimized by assigning a channel allocation cost factor, C_k, and a cost factor for the equipment required for sharing capability, C_s and minimizing the total access system cost with respect to S. Typical values for S are in the range $5 \leq S \leq 10$.

Chapter 6

THE GROUND COMMUNICATION SUBSYSTEMS

6.1 Electronic Equipment

6.1.1 Ground Station Equipment

The ground station equipment (Fig. 43) is similar to the transponder design, in that it is determined by both the modulation techniques for up and downlink and the types of information to be relayed by the satellite.

Because of the proposal to use a wideband phase modulation technique for voice and TV transmission, the receiving part of the ground station is identical in both cases. Single variations exist in processing the video signals, which are more complex with voice transmission.

The equipment in the transmission section of the ground station is relatively extensive because of the SSB modulation technique. Receiving and transmission units are split in blocks to handle voice and TV transmissions because of different types of video band processing and different sideband modulation techniques.

6.1.2 Description of the Signal Flow

The received signal will be amplified in a cooled parametric preamplifier stage and split into different blocks according to several phase modulated carriers. The frequency of a local oscillator of high accuracy will be used to feed a frequency converter to bring down the signal to a reasonable intermediate frequency. A demodulating stage produces the baseband of the signal which will be brought directly to the standard ground equipment or will be processed in the voice baseband analyzer followed by a hybrid network and the standard terrestrial equipment.

For transmitting the voice baseband, information is modulated in SSB by using a modulation stage and a highly accurate frequency generated by a local oscillator. In a baseband combiner, one sideband of each

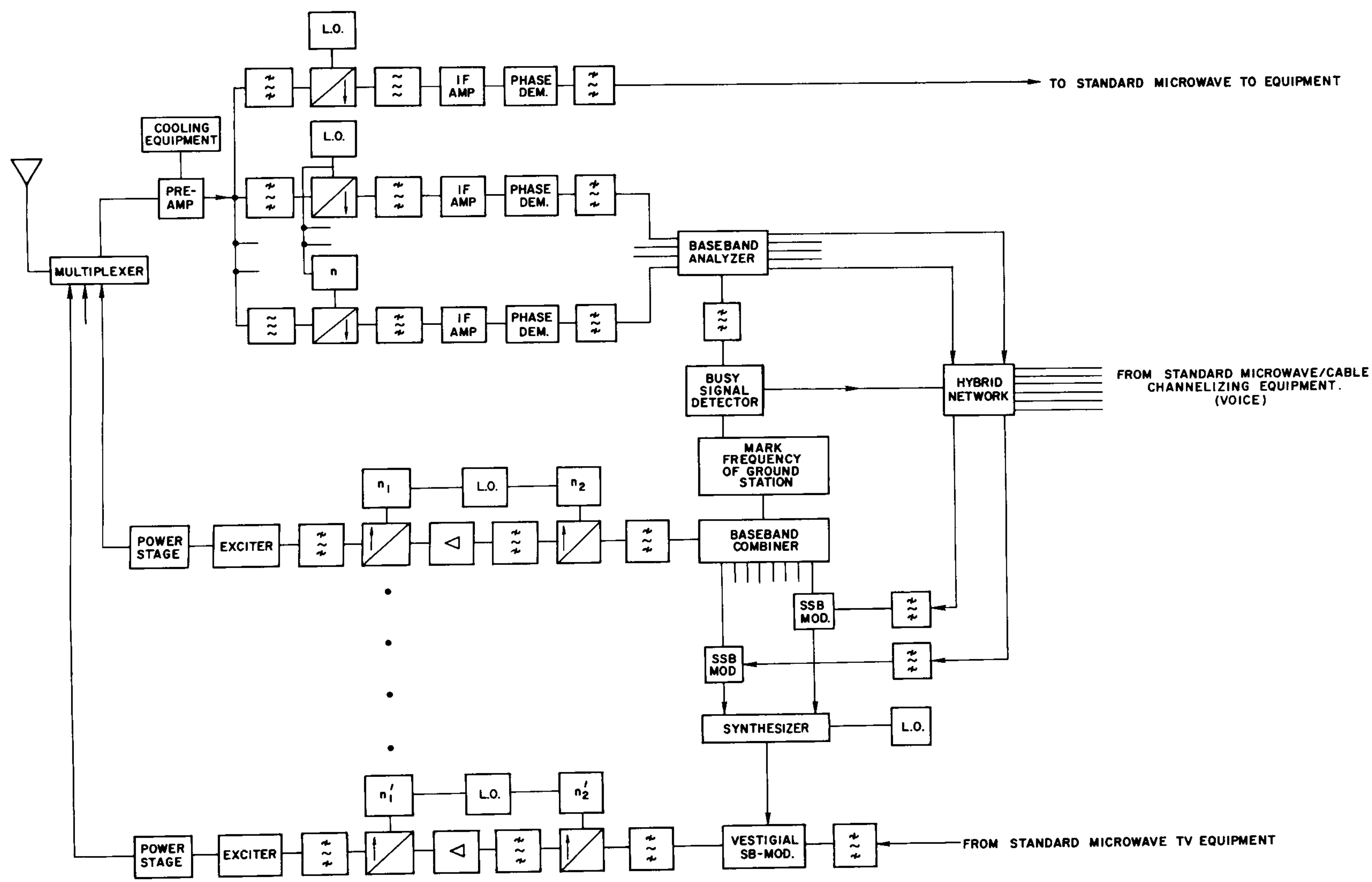


Fig. 43 GROUND STATION (INTERNATIONAL/REGIONAL TRAFFIC).

carrier is suppressed and linearly combined with others. The total baseband signal will be upconverted in frequency in two stages to minimize the requirements of filters. Conversion from baseband directly to RF is difficult because then the desired sideband can scarcely be separated from the undesirable band.

The final power stage is a klystron. The transmission of a TV baseband is in principle the same. Instead of a SSB technique, a vestigial sideband modulation scheme is applied because the suppressed carrier is to be used in the satellite repeater to obtain coincidence in phase of the transmitted and onboard generated carrier. This coincidence in phase is not necessary with the transmission of voice information.

6.1.3 Equipment for Multiple Access to the Satellite

For the selected method of multiple access, which operates with fixed frequency allocation, grouping certain ground stations, certain additional equipment must be implemented at the ground stations:

(1) A signal oscillator (mark frequency generator), which transmits a tone outside the voice information band. This signal oscillator is used specifically for busy tone detection. The mark frequency is fed into the baseband combiner.

(2) For receiving the busy tone, a filter is to be used to channel this signal to the standard equipment and the user.

6.2 Terrestrial Network Interfacing

This section discusses the alternatives available for interfacing the terrestrial communication network with a system of ground stations served by a given satellite. The primary problem is one of implementing a given multiple access scheme in the most economical and efficient manner, using existing and/or easily modified equipment which can be available by the anticipated launch date, 1973. Three separate alternatives for land-line channel terminations and allocations will be discussed. From Chapter 5, these are

(1) fixed point-to-point channel allocation;

(2) fixed point-to-point frequency assignment with groups of ground stations sharing a common spectrum or channel group; and

(3) assigned receive frequencies.

Demand assignment involves computer supervision and is not discussed in this section.

In the discussion to follow, certain conventions are introduced for convenience in presenting the three alternatives. The channel frequencies are denoted as follows:

du ≐ dialing station, uplink (outgoing)

dd ≐ dialing station, downlink (incoming)

cu ≐ called station, uplink

cd ≐ called station, downlink

Note that the information is the same in the channel pairs du-cd and dd-cu, according to the speech directions. However, if the dialing and called stations are in the common coverage area of a given satellite beam, separate sets of frequencies have to be used to prevent interference.

A basic review of telephone exchange operation is now given. Telephone dialing systems use commutators for dialing, i.e., electromagnetically driven rotary switches scanning in steps across banks of output contacts. The dialed number dictates the stops made by the commutator. A commutator (cr) and its subcommutators are normally one mechanically connected unit; the output contact banks of the subcommutators are arranged above each other in different levels. The dial controlled steps move the contact fingers of the commutator up to the selected output group, where they rotate around over all outputs. The standard commutators have ten levels, each with ten outputs. Model motor driven high speed commutators have all outputs in a circular arrangement in one bank. When dialing, the contact fingers make steps over ten outputs at each digit, then single stepping forward automatically to find a free line. Ground lines transfer the dialing contacts thus chosen by three wires, two for the speech, one for dc control signals as checking for free, entering and locking. The satellite cannot do so since there is only one speechband per communication contact. In this case, discrete code frequencies inside or below the speechband are transmitted. N different frequencies may form a code containing 2^N-1 signals.

The following signals will be needed:

(1)	entering a channel, locking	dialing station (outbound)
(2)	double occupation	
(3)	dial pulses	
(5)	receiver called	called station (response)
(6)	receiver busy	
(7)	receiver picks up	

Three frequencies provide eight signals, which will meet the needs of telephone signaling. They may be arranged below the speechband between 0 and 300 Hz.

In general, referring to Figs. 44-46, the rectangles above the horizontal lines represent speechbands: Bands above the lines are transmitted; those below are received. Increasing frequency is from left to right.

6.2.1 Fixed Point-to-Point Allocation

This scheme, as described in Chapter 5, allocates a unique receive-transmit channel pair between each and every subscriber ground station desiring communication. For N conversations between two ground terminals, N unique channel pairs are allocated.

Figure 44 shows in channel group du three separate channels (4, 5, and 6) reserved to call stations A, C, and D. Station C uses channels 8, 9, and 10 to contact stations A, B, and D, respectively. Finally, station D calls stations A, B, and C via channels 10, 11, and 12. Regarding equipment expense, this scheme is very simple. To select the called ground station, only the corresponding du channel will be selected. That can be done easily by the ground telephone commutators. Prevention of other users from entering the same channel is also no problem because of the unique channel pair. The needs of bandwidth, however, are very high. This fixed point-to-point access scheme gives a definite advantage to high traffic ground stations.

Figure 47 shows how to contact one out of five other stations B to F by station A, when this scheme is used. To each station, five channels are available. In this example, five different calls may be

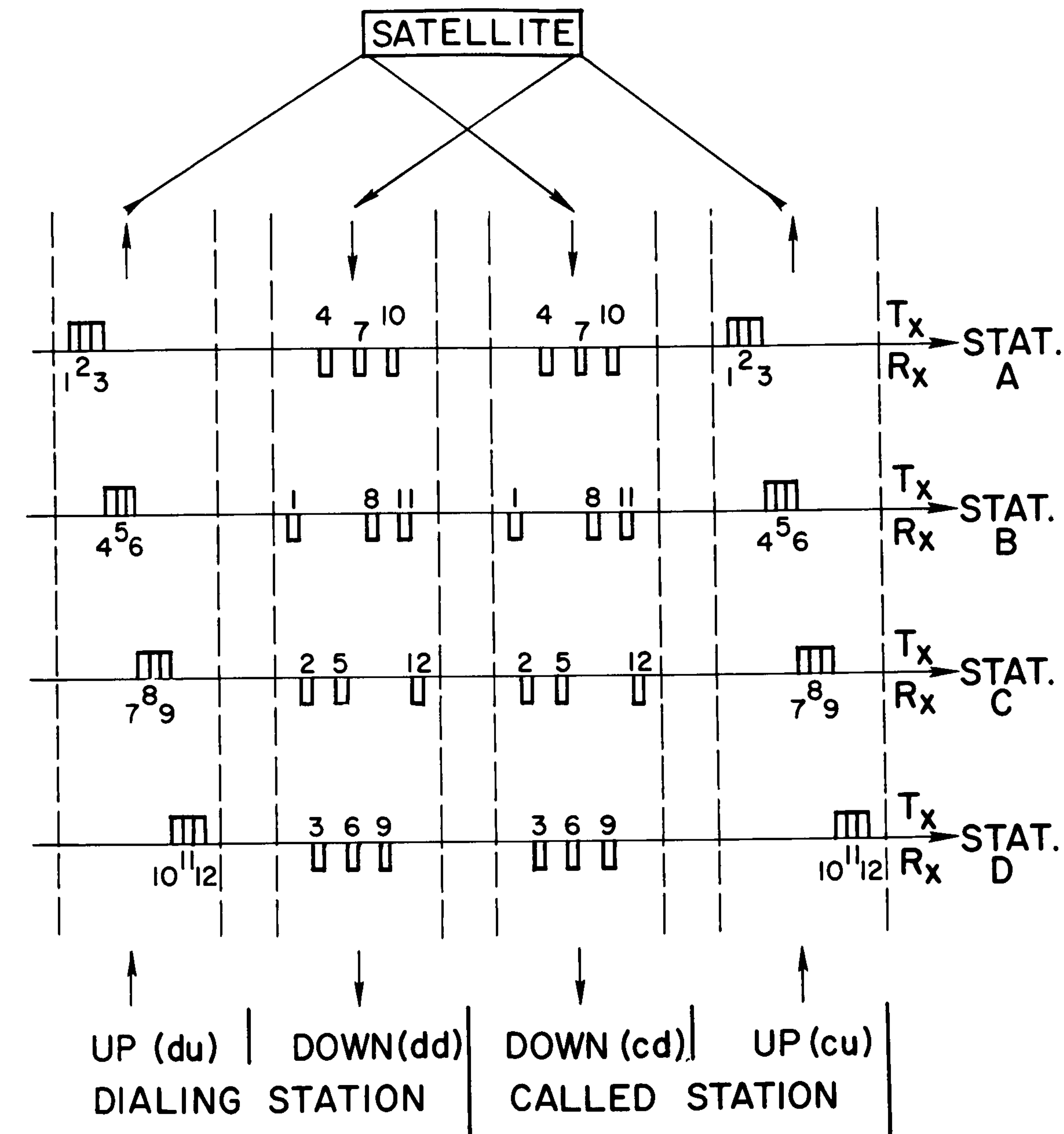


Fig. 44 CHANNEL ALLOCATION-SCHEME 1.

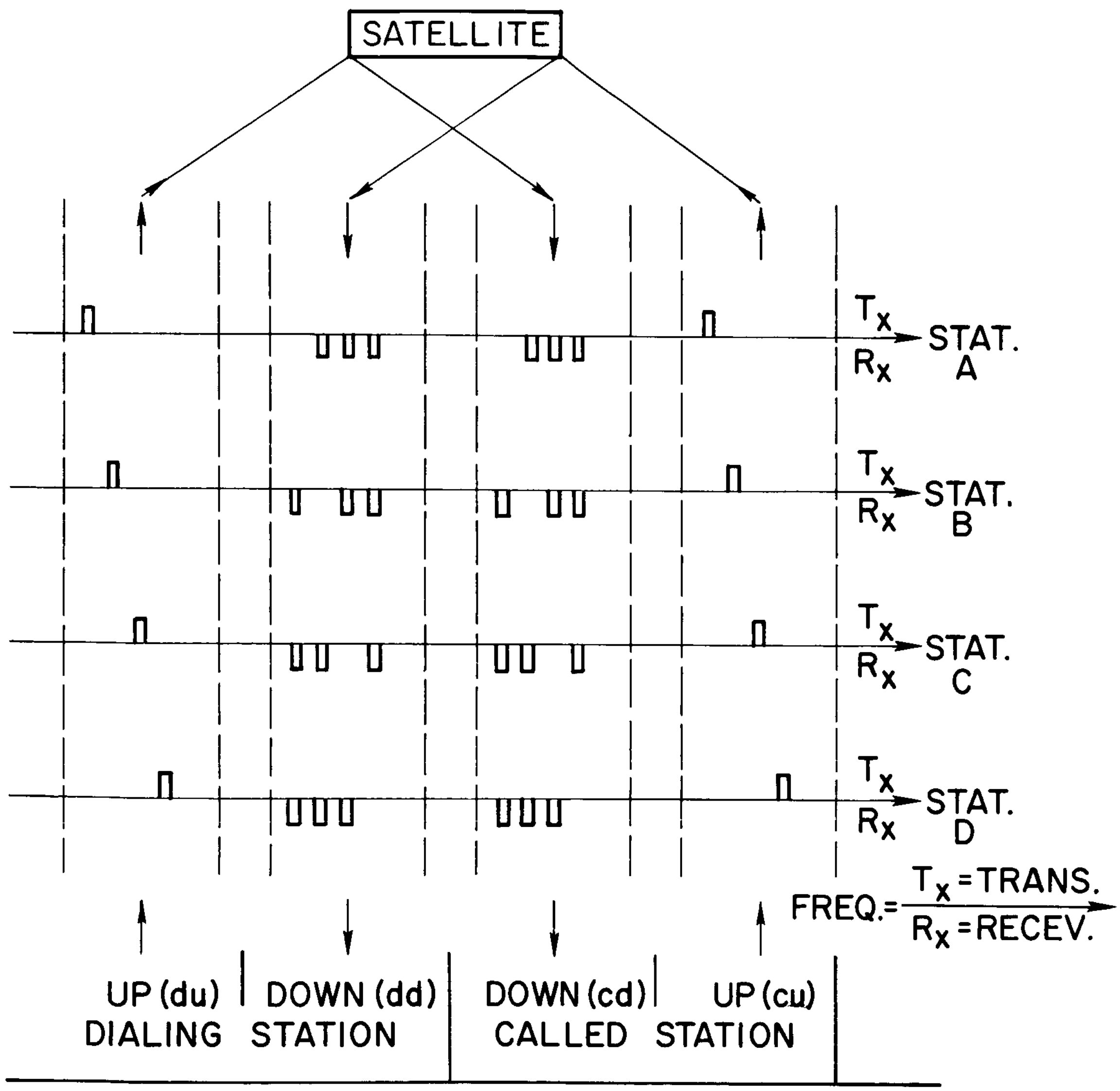


Fig. 45 CHANNEL ALLOCATION-SCHEME 2.

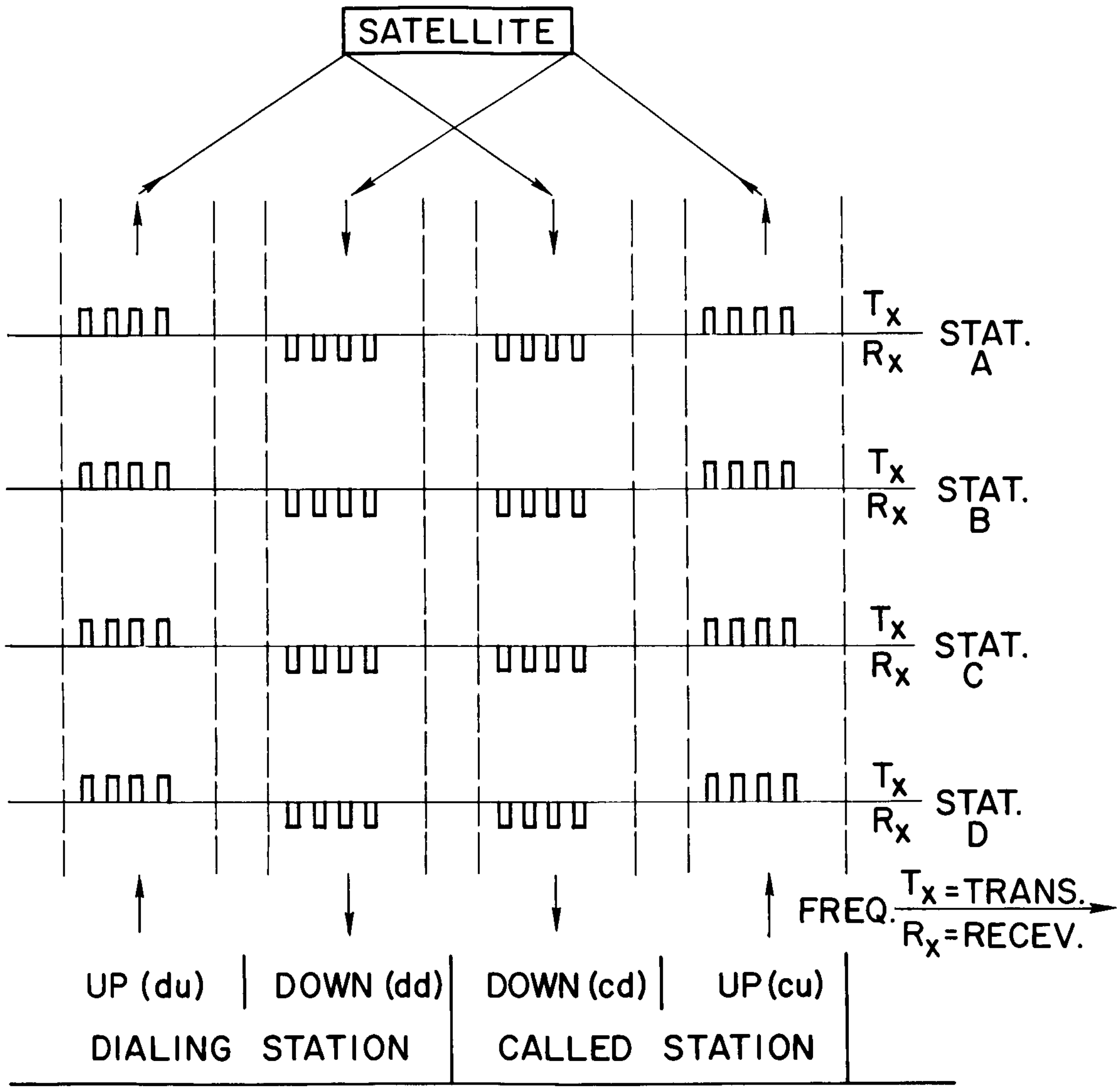


Fig. 46 CHANNEL ALLOCATION-SCHEME 3.

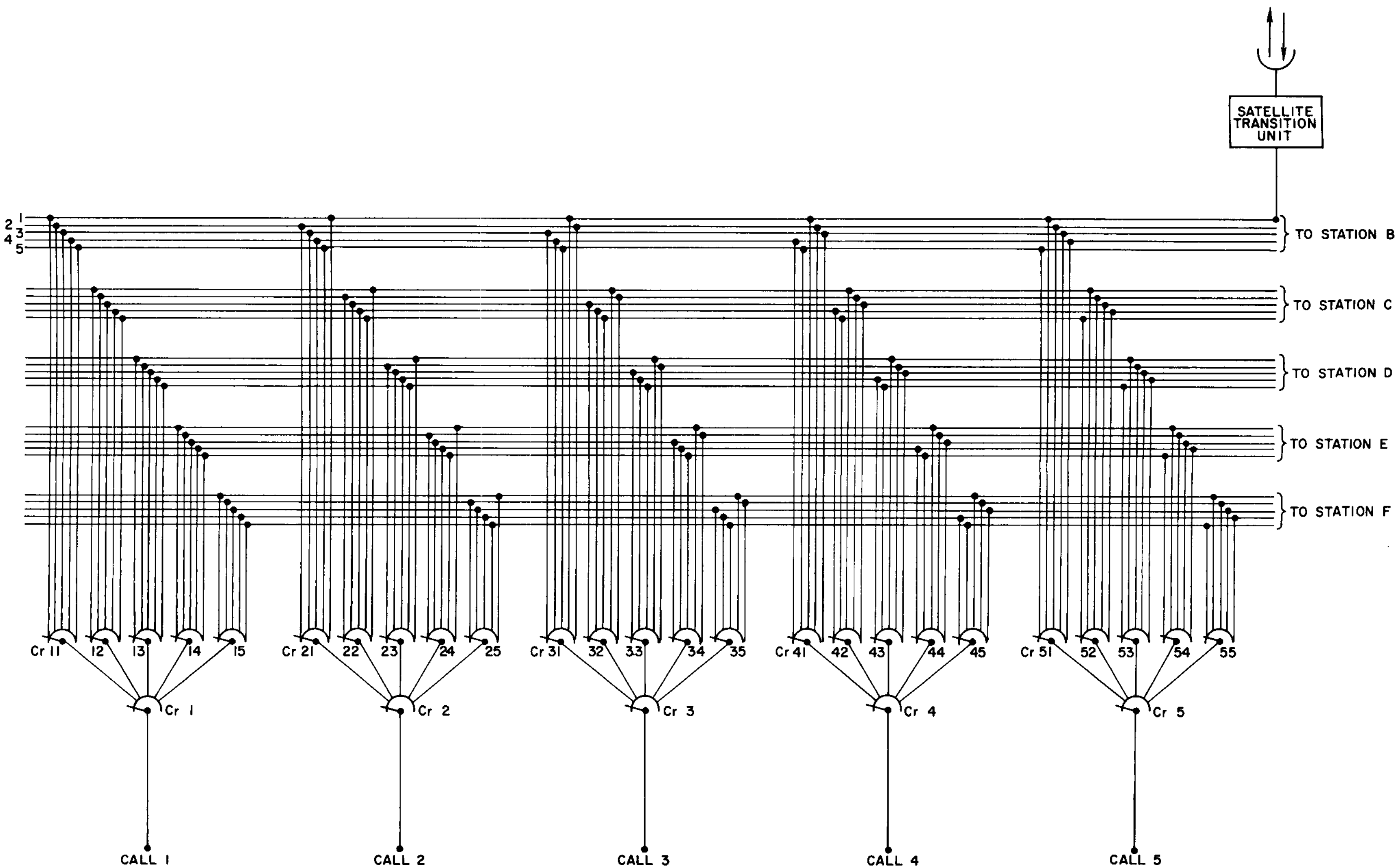


Fig. 47 CHECKING AND LOCKING.

served at the same time. The lines, Call 1 to Call 5, are equivalent, regardless of call destination. The telephone is first attached to a commutator, say cr2. Following the dialed number, the commutator steps around to one output line, dialing one, reaching subcommutators cr21, dialing 2, cr22, and so on. Each number indicates one ground station. When the subcommutator is chosen, it scans automatically over all its output lines and rests on the first free one. It is obvious that all five lines have all five channels accessible to all ground stations. The subcommutators are connected in parallel at their outputs, but in a circular shift manner. For example, commutator cr11 checks the lines from 1 to 5 sequentially. Commutator cr21 begins with lines 2, 3, 4, 5, and ends at 1. In this manner, all five lines pick up at random the same traffic, since the ground installations also are mixed in such a way as to bring in nearly the same number of calls across each line.

In Fig. 47, only one switching line per channel is shown. In reality, there are three wires switched simultaneously by three mechanically combined contact fingers. Wires a and b transmit the speech, and c in common with ground is a signal line. Figure 48 gives a detailed example for free-checking and locking an output. When commutator 1 reaches a free output, its P-relay finds a potential across line c. In the transmission equipment, relay c activates, starting further operations. In the commutator, the P-relay opens, connects the speech lines and shorts a high resistance position of its coil. The check potential drops sufficiently so that commutator 2 cannot seize the channel. Commutator 2 then continues scanning to find another free circuit.

6.2.2 Fixed Point-to-Point with Sharing

This method, also described in Chapter 5, allocates a common spectrum to be used between two groups of ground stations. Refer to Fig. 45. Here the uplinks are unique but the downlinks are shared among the group or parts of the group of ground stations. Entering a channel does not yet decide which station is called, so that both dialing and called stations need additional equipment. The dialing station has to transmit a series of dial pulses, which are identified by counters in all other ground stations. Only the called one is adjusted to this

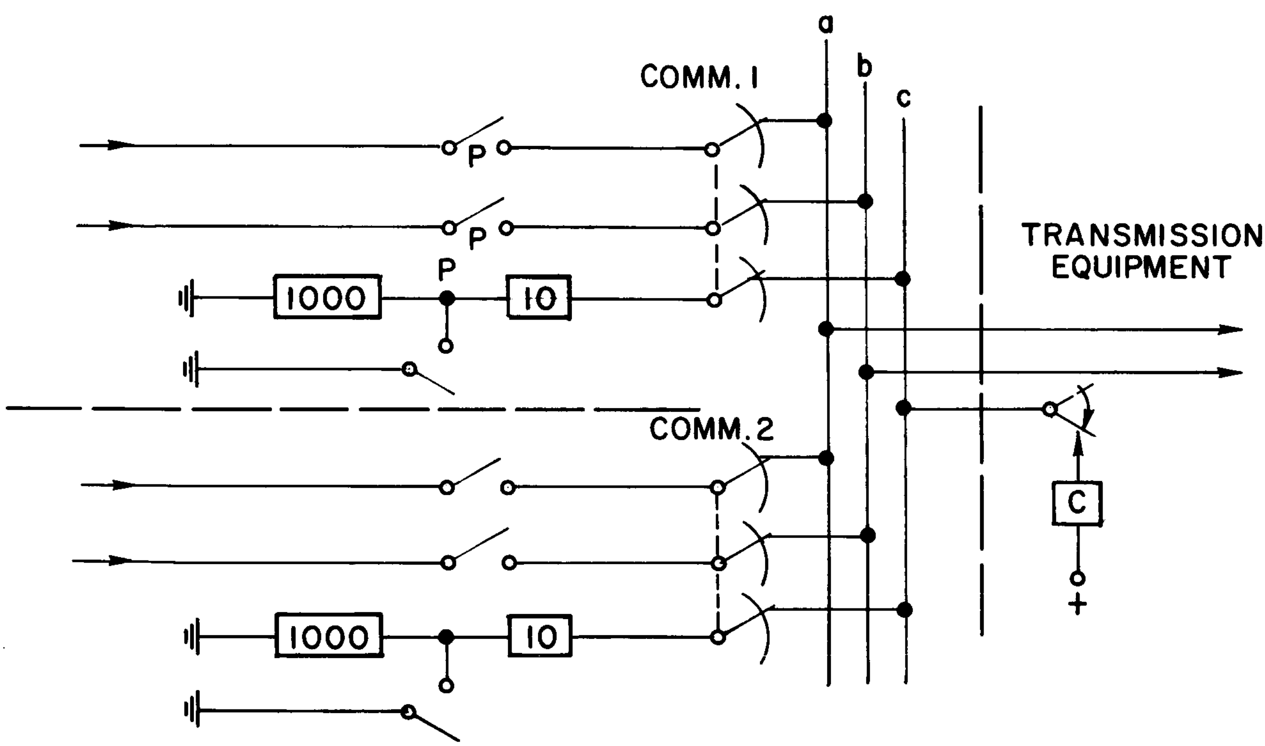


Fig. 48 TRANSMISSION INTERCONNECTIONS FOR FIXED POINT-TO-POINT ACCESS (Station A to Stations B-F).

number and will attach the call to outgoing ground lines. This scheme requires less bandwidth, particularly when routing the traffic between many stations without many calls. The station equipment, however, is more complicated.

6.2.3 Assigned Receive Frequencies

This method allocates a limited spectrum to a given ground station X over which his incoming calls must be monitored. Hence, the other ground stations in the system, having a need to converse with X, must tune to his receive frequency and then attempt to acquire a channel through to X. When entering or attempting to seize a channel to X, a given ground station must indicate to the rest of the network that he is attempting to lock onto a channel to X. As in the method of Section 6.2.2, a series of dialing codes must be transmitted and monitored by the network, indicating to them that a particular channel is now unavailable for their possible communication needs to X. Refer to Fig. 46. Due to the propagation delay in transmitting this code via the satellite to the rest of the ground station network, some stations may not be sufficiently forewarned that this channel is occupied and will try to seize it themselves. Hence, this scheme requires extremely complex equipment in each station, but it does conserve bandwidth. This method is recommended for groups of stations with low traffic requirements and consequently lower probability of simultaneous channel seizure. The next subsections describe some of the entering and locking procedures.

6.2.4 Double Occupation

A channel used by another ground station cannot be entered by a checking commutator, because the potential at line c in Fig. 49 is removed. However, the information that a channel is occupied by a certain station reaches the others only after considerable delay, due to the propagation time of the waves. Meanwhile, any other ground station may enter the channel, also. Therefore, a time comparator must notify the second station to leave the channel

Refering to Fig. 50, station A enters a channel at time A_o. It sends a unique code to all other stations which may arrive at station

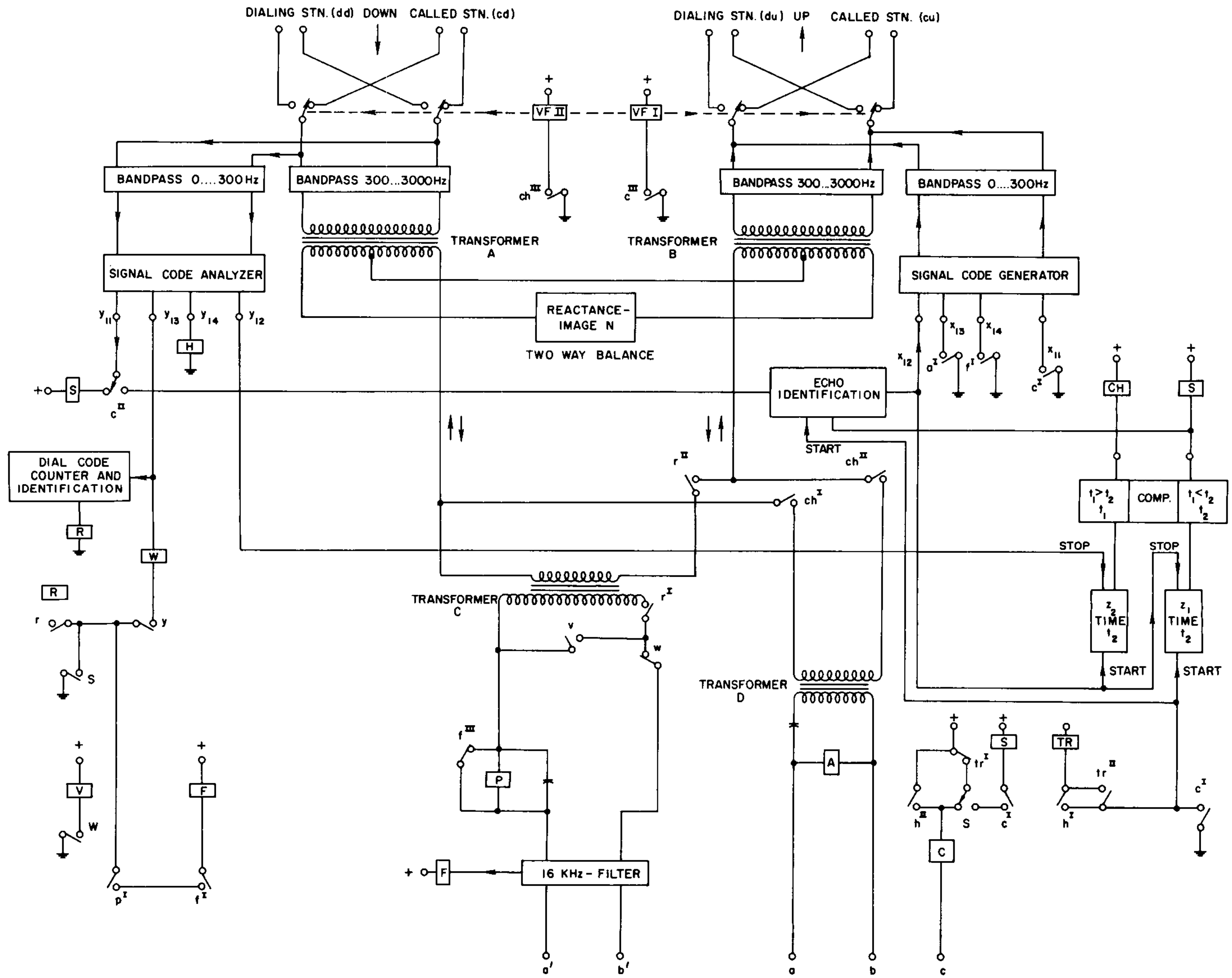


Fig. 49 INCOMING AND OUTGOING ENTERING AND LOCKING.

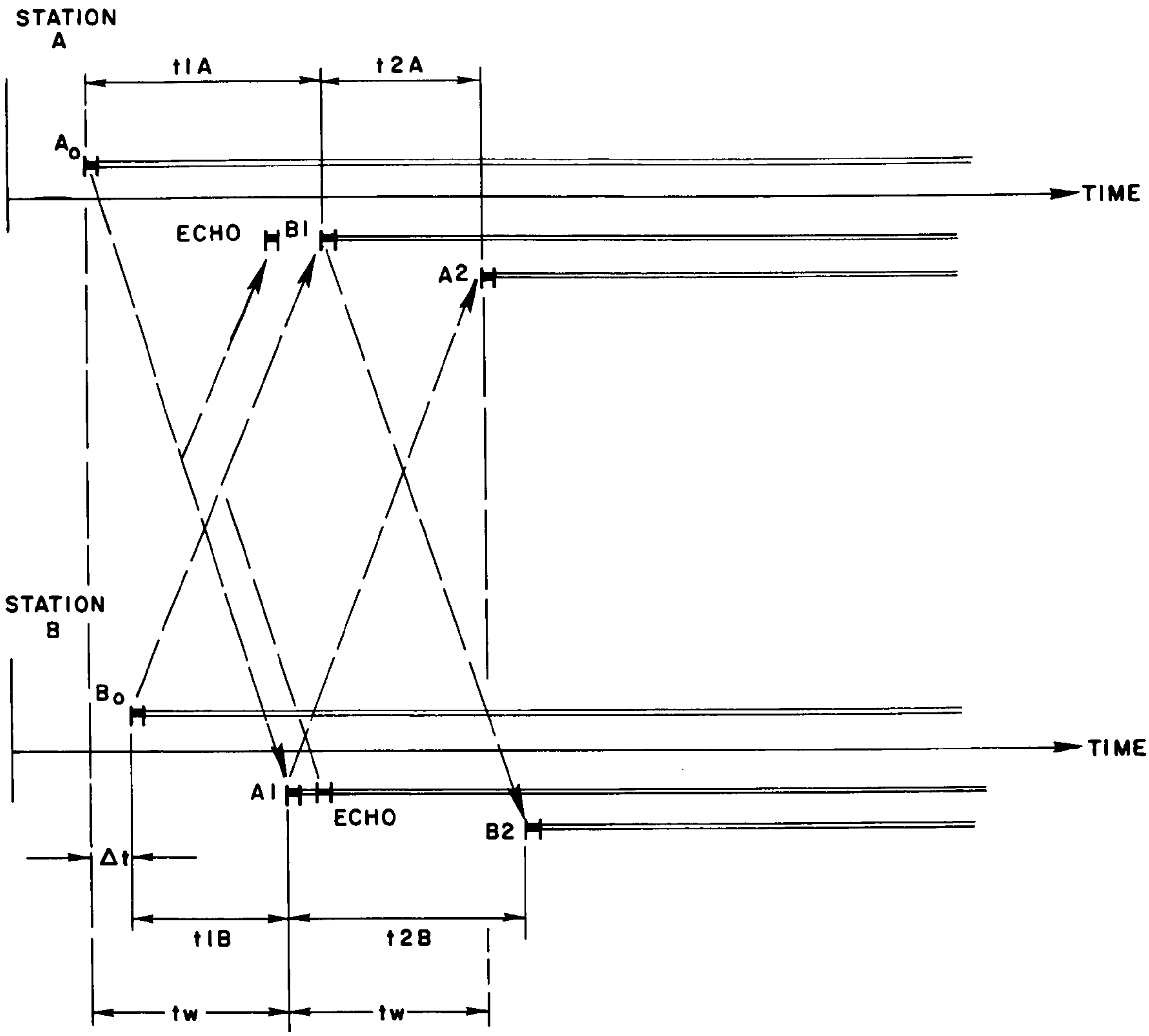


Fig. 50 ECHO AND DELAY.

B at time A_2 (delayed by propagation time t_w). But at time B_o, station B also had entered the same channel, at a later date than station A. In this case, station B gives a double occupation signal back to station A. Station A channel receives the occupation signal from station B at time B_1, and retransmits to B, arriving at time B_2. The time differences t_1 and t_2 in Fig. 50 are significant for what station was first in the channel. In the same way, however, the echo of the occupation signal of the station itself comes back. But this echo is constant in both time delay and amplitude, and will be suppressed by an identification comparator.

Station A gives its occupation signal at time A_o to station B, arriving there after delay time t_w. About Δt later than station A, station B also had entered the channel at time B_o. It had started a time counter Z_1 at B_o. When the double occupation signal arrives, counter Z_1 is stopped and another counter Z_2 starts. It stops when the double occupation signal comes back from station A. The same operation is completed in station A. Referring to Fig. 50, the following times t_1 and t_2, respectively, are stored in the counters:

Station A:

$$t_1 = t_w + \Delta t , \qquad (t_1 + t_2 = 2t_w) , \qquad t_2 = t_w - \Delta t$$

Station B:

$$t_1 = t_w - \Delta t , \qquad (t_1 + t_2 = 2t_w) , \qquad t_2 = t_w + \Delta t$$

In both station A and station B, a time comparator checks which time is longer. When $t_1 > t_2$, station A was first in the channel. Relay CH of Fig. 49 is activated. When station A was later, $t_1 < t_2$.

Relay S is significant for other reasons. Either achieved directly by code yll or later after double occupation has been rechecked, it cuts the entrance of the unit. Any commutators resting on the input are thereby removed. This double occupation system is a high precision version designed for fast switching equipment (data transmission). In any case, where the traveling delays between the ground stations do not differ more than negligibly, it is sufficient to watch only if there is an occupation signal before the echo. The station from which this comes has priority. However, when the distances to the satellite are considerably different (when standing near the horizon) and the priorities are to be identified precisely, the former method should be preferred. In this case, the echo could be identified more accurately. Both signal and echo come from the same signal oscillator, such as to provide a constant phase shift. Other signals drop in at random, and the phase angle is incoherent to the oscillator.

Occupation: When the channel is free for the station relay, CH activates. Its contacts ch^{I} and ch^{II} connect the input a,6 to the uplink radio equipment. Contact ch^{III} operates a second HF-Relay VF II to switch over the downlink channel into the dialing station.

Dialing: Relay A pulses when the caller dials. Its contact a^{I} gives a dial code X_{13} following the pulse train.

6.2.5 Inbound Traffic (Fig. 49)

All stations activate their relays when any one station sharing the channel enters (code y11). When this station dials, all other stations receive the dialing code and count it in a dial code counter unit. Only the called station will respond when its dial code is identified. The other stations will not answer. The called station turns on its relay R. This connects the entire unit to a pair of outgoing lines by contacts r^{I} and r^{II}. Further dialing pulses then will operate relay w to pulse out the dial by breaking the outgoing lines. Relay v shorts the transformer while the pulses are running.

When the called receiver picks up his telephone, his exchange gives a short time 16-kHz pulse back to the lines a' and b'. A 16-kHz filter identifies the pulse and activates relay F. Relay F opens a contact f^{III} which was suppressing a relay P in the outgoing circuit. P also activates, holding F-relay by means of its own contacts f^{II} and p^{I}. Contact f^{I} gives a receive-code X14 to the calling station, which is there picked up as Y14.

Code Y14 opens relay H in the calling station. H activates relay TR. When the called receiver hangs up, the H-relay opens. TR holds itself by contact tr^{I}. In this position, the c-line is open again;

i.e., contacts h^{II} and tr^{I} are both opened. That makes the commutator locked on this channel reset to zero.

6.2.6 Outbound Traffic

Entering and locking a channel: A three line switch rests on terminals a, b, and c. A relay checks at line c if potential is to activate. If the channel is free, relay c is activated. The other relay in the switch shorts the high resistance position of its coil, thus removing the potential from all other switches. See Fig. 49. The C-contact gives a signal code, i.e., a distinct frequency below the speechband, to all remote ground stations sharing the same channel. Their own transmission units receive and identify this code Y11. It activates relay S, which opens the inbound checkline C. S holds as long as the channel is otherwise occupied.

6.3 Shadow Operation and Sun Blind Antennae

6.3.1 Occultation

For any satellite using solar cells as a power source, the possibility of passing through the shadow cast by the earth (occultation) must be considered. If a synchronous equatorial orbit is used, the period during which there is no sunlight to illuminate the solar cells occurs only during the two equinoctial periods. The maximum daily shadow time, shown in Fig. 51, is 72 minutes; the number of days per year is 92: 46 in the spring and 46 in the fall. The daily eclipse time is evenly dispersed about midnight, which, fortunately, is a time of low demand for most telecommunications.

Without sunlight, the solar cells cannot provide power, and no transmission is possible, unless a source of power other than solar cells is provided. Thus, the major alternatives are:

(1) to provide no communications during occultation or

(2) to provide communications during occultations by including a redundant nonsolar power supply.

The second alternative may be further divided according to the degree of operation required or anticipated during occultation. First, however, consideration will be made of the basic differences in alternatives 1 and 2.

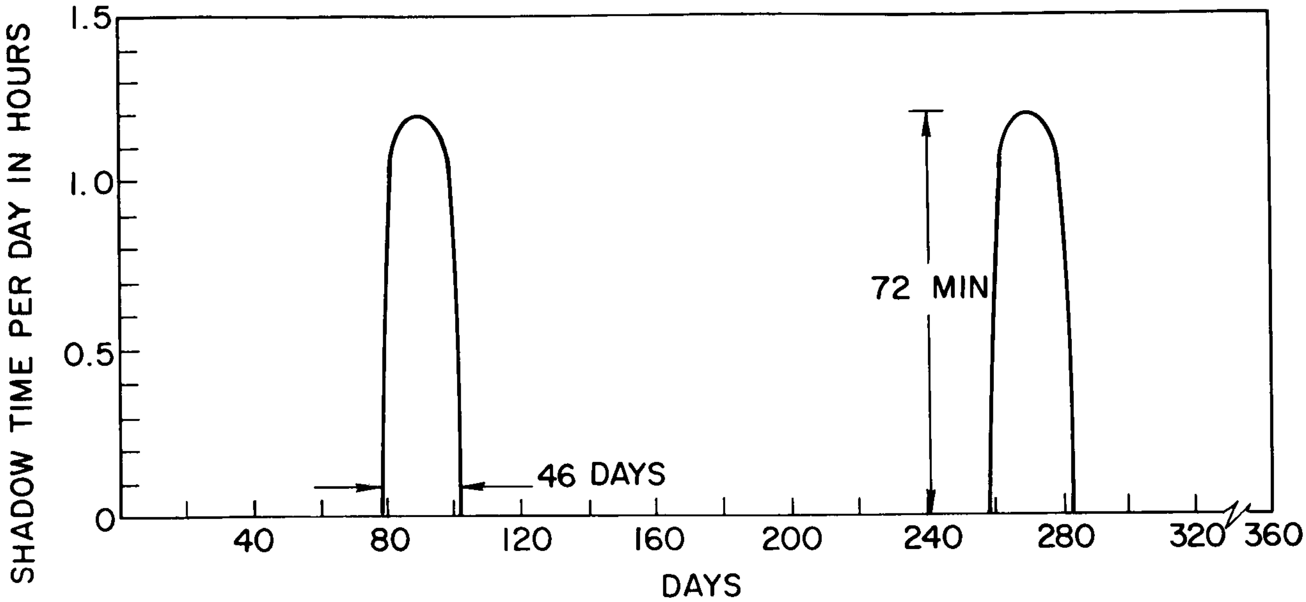


Fig. 51 TIME IN ORBIT FOR SYNCHRONOUS EQUATORIAL ORBIT.

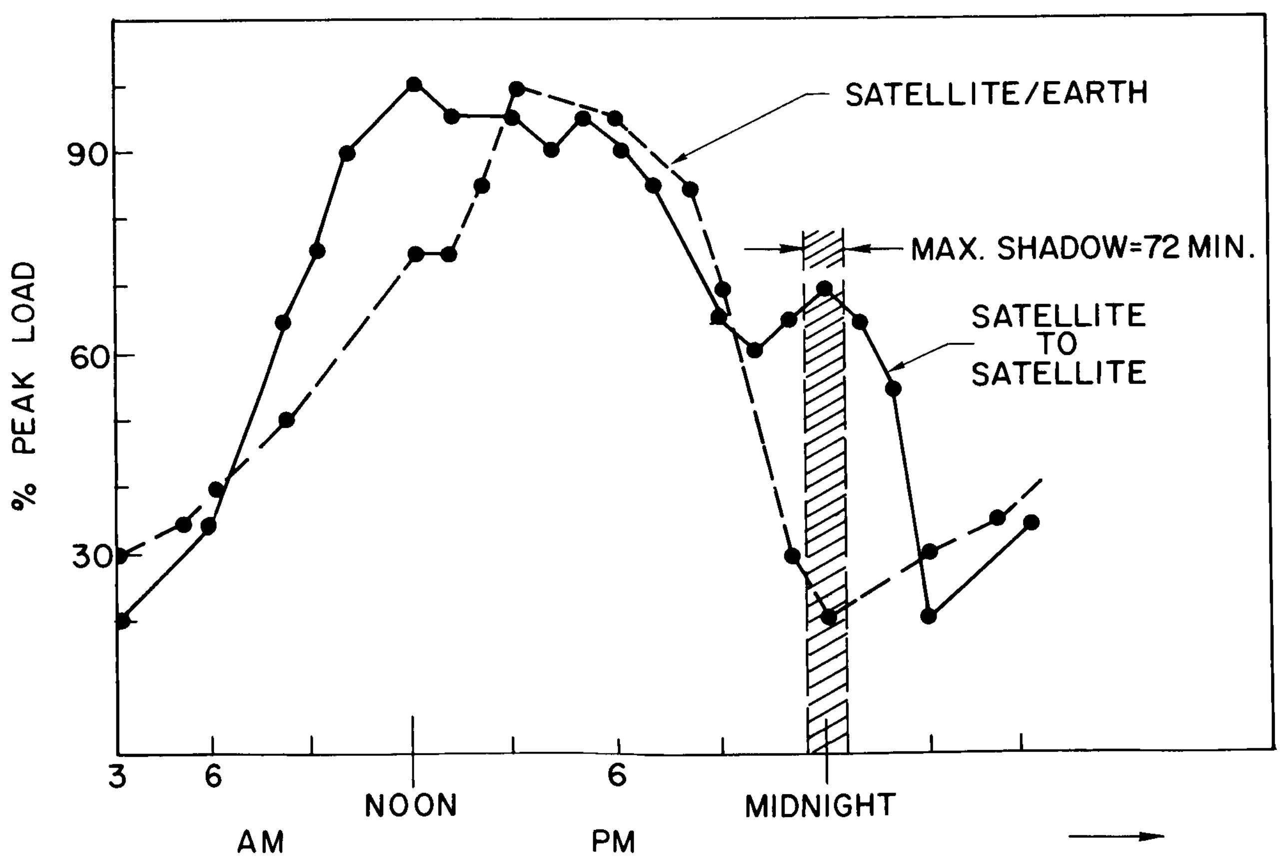


Fig. 52 HOURLY TRAFFIC DEMAND AS A PERCENT OF PEAK LOAD.

Providing no power during occultation has advantages in weight and space; hence launch costs and station-keeping costs may be reduced, or the satellite may carry other hardware. Some of the weight and space advantage is lost when the change of temperature of components is considered. When fully in the sun and at maximum operation, the satellite must dissipate a considerable amount of heat. (Refer to Chapter 11.) With power off and in shadow, the temperature of the satellite and the components therein would drop rather drastically toward absolute zero. Batteries may rupture, and semiconductors may drop below their operating temperature range, and may not be able to retain their normal characteristics. The mean life of transmitting tube filaments is decreased by repeated power shutdown, so they would be better off left on. Hence, zero transmitted power does not mean zero required dc power, as the components must be kept warm. In addition, power may be needed for station keeping. Careful estimates indicate that about 20 to 25 percent of full power would be required even if no RF power were transmitted.

Full operation during shadow has obvious communications advantages over no operations, in that communication is possible. The primary disadvantage is the weight and space required for full power redundancy in batteries. Partial transmission during the eclipse period looks interesting when probable daily traffic cycles are considered. Figure 52 shows the relative traffic density for a homogeneous distribution of calls over 10 time zones as handled by one satellite over a 24-hour period. If fewer time zones are considered, the peaks become higher and the nulls become lower. On the same graph is the daily loading for international traffic for time differentials of 8 to 12 hours between terminals. Traffic is shown as a percentage of peak load on the respective curves.

Figure 52 indicates that approximately 33 percent operation during shadow could be considered. This partial operation would be best accomplished by dividing the satellite output channels so that the desired percentage goes through a single transmitting tube for each antenna beam. For 33 percent operation, about 55 percent of the total dc power is required in batteries. For 50 percent operation, about 70 percent of the total dc solar power must be available from batteries. This makes partial operation in shadow less desirable.

Any shadow time operation other than 100 percent complicates power tube redundancy and sorting by frequency filtering problems, and also lowers the reliability of the system due to the additional switches.

As a final consideration, the size of launch vehicles is evaluated. Because the payload is a bulky one due to large solar cell and antenna packages, a large launch vehicle is required. The spacecraft configuration allows sufficient room and launch capability for 100 percent battery power. Hence, full operation will be available during shadow operation, and the same batteries can be used for power during the time period between launch and unfurling of solar cell arrays.

6.3.2 Sun Blind Antennae

When a receiving antenna beam is focused on, or very near the sun, the noise level is increased because the antenna sees a high background temperature. This increased noise level may obscure the desired signal, interfering with intended communications.

This situation can occur in two ways in an earth-satellite communications system:

(1) The satellite antenna may look at the sun.

(2) The earth antenna may look at the sun.

The first case occurs just before and just after the satellite is eclipsed by the earth's shadow (occultation). Its severity is determined by the beamwidth of the satellite's receiving antenna, and by how near the earth's horizon that beam is focused, as shown in Fig. 53. The sun subtends an angle seen by earth as 32.25' or about 0.5°.

The geostationary satellite system may be considered as a rigid system changing in angle with the sun at a rate of 360°/24 hr (60 min/hr) or 0.25°/min. The effective noise temperature seen by the satellite antenna is

$$\frac{10^{5}\,{}^{\circ}\mathrm{K} \times (\text{area of sun seen by antenna in square degrees})}{(\text{satellite antenna beam area in square degrees})} .$$

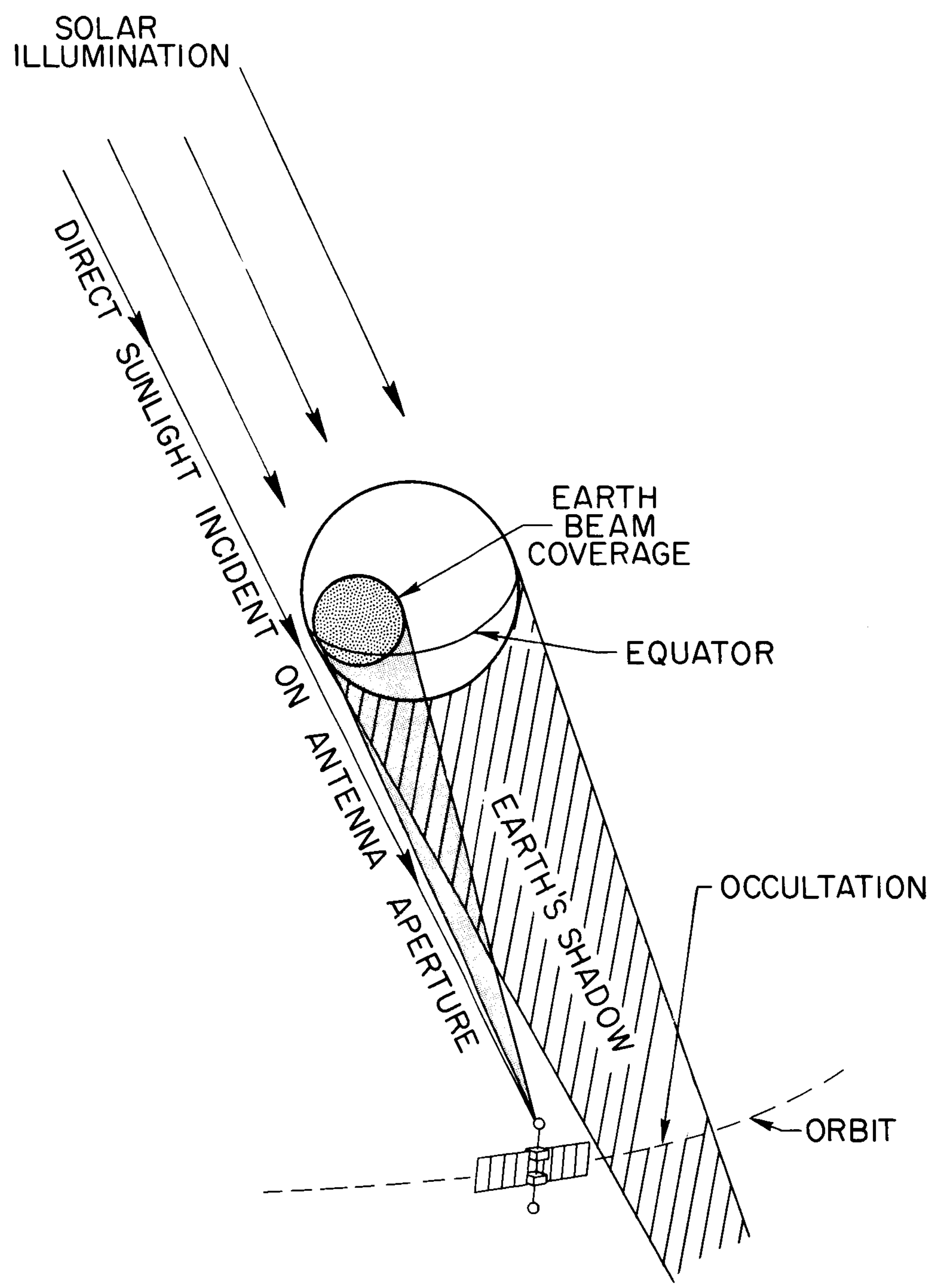


Fig. 53 SUNLIGHT AND SATELLITE BEAM CO-LINEARITY OCCURRENCE.

The relative antenna gain (maximum of unity) must also be considered. For a worst case of a 6° satellite antenna beam with its 0.9 power point at the earth's horizon, the noise temperature is

$$T_N = \frac{10^5{}^\circ K \, (.5^\circ)^2}{(6^\circ)^2} (0.9) = 625^\circ K \; .$$

This raises the effective noise level by 3.7 dB. About 19 minutes later (or 12 minutes earlier), the noise temperature increase is reduced to about 160°K and the noise added is slightly less than 1 dB. This 19-minute period occurs twice each day for 96 days per year. It does not force discontinuation of communications, but may give a barely noticeable increase in noise.

The second case, with earth antennae looking at the sun, is more severe. The satellite does not cast a significant shadow on the earth antenna, and the peak noise temperature seen by a 0.3° earth antenna beam is effectively 10^5°K. This increases the noise level by 30.5 dB and communication during this condition is then impossible. The maximum noise temperature which the downlink will tolerate is 800°K, based on the ground terminal receiving system noise temperature of 85°K. The output signal-to-noise ratio is normally 52 dB, and it will drop by approximately 10 dB when the 800°K noise temperature is reached. The minimum separation between beam center and sun center to guarantee that the antenna sees less than 800°K is one which gives 21 dB reduction in signal strength, or 1.3 beamwidths (see Fig. 54). This means that there can be ground station blackout of approximately four minutes. This problem occurs for any ground station antenna aimed at a synchronous satellite. Its duration and the time of year it occurs vary with the latitude of the ground stations. The time of day it occurs varies with the longitude of the ground stations relative to the longitude of the satellite, i.e., the ground station's longitudinal pointing angle.

The latitudinal angle of the sun with respect to a point on the equator varies at a rate

$$\sin^{-1} [\sin \epsilon \; \sin 2\pi t]$$

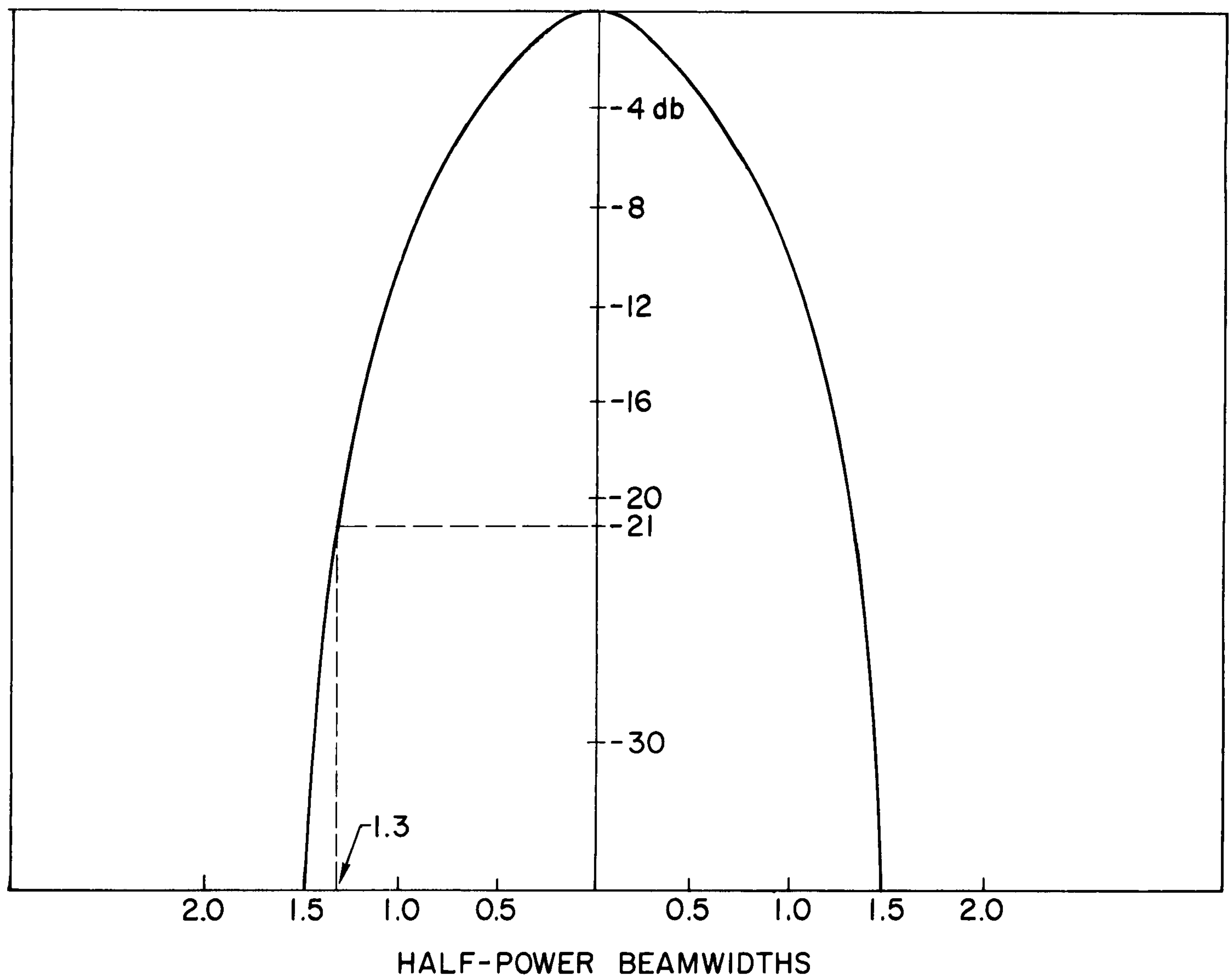


Fig. 54 RELATIVE GAIN FOR A PARABOLIC REFLECTOR.

where

ϵ = the obliquity of the equator = 23.5°, and

t = the time in years measured from the spring equinox.

Then,

$$\dot{\theta} = \frac{(0.4)\ 2\pi \cos 2\pi t}{\left[1 - (0.4 \sin 2\pi t)^2\right]^{\frac{1}{2}}}$$

which has an equinoctial maximum of 0.8π radians per year or 39°/day. For a ground station on the equator with a 0.3° beam, the number of days per year that sun blindness occurs is either five or six. For the same ground antenna located at 60° latitude, the two periods per year when sun blindness occurs are 15 or 16 days each, for a total of over 30 days.

One way to overcome this problem is to use diversity antennae. However, the antennae would have to be so far apart (5° of earth surface) that it is not a practical answer. It appears at this time that the system will have to "live with" such blackouts as are caused by earth station antennae pointed at the sun.

SECTION II

SATELLITE STRUCTURAL ANALYSIS, ATTITUDE CONTROL AND LAUNCH VEHICLES

INTRODUCTION

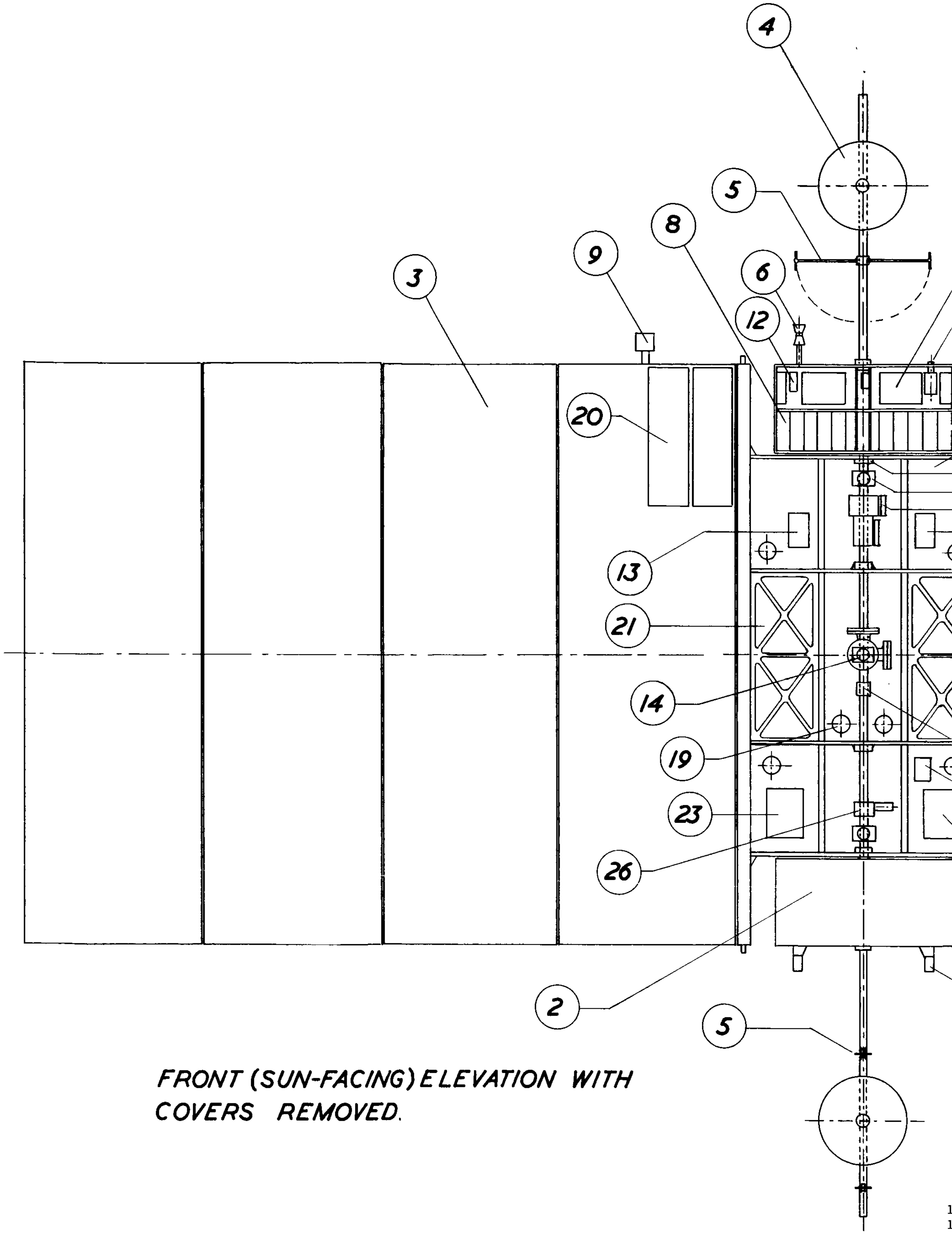
To meet the telecommunication needs of 1973 and beyond, several satellites carrying the necessary electronic equipment are to be placed in synchronous orbit. The mechanical and structural design of these spacecraft is the subject of Chapters 7, 8, and 9.

What is desired is a platform, accurately positioned in space and accurately pointed at the earth, that could supply the relatively high power (3 kW) required. Further, it is felt that the design should be such as to permit the use of the basic structural and mechanical configuration for many different communications needs that would require in the neighborhood of 3 kW of raw power. Thus, in the 1970s, many such spacecraft would be built, effectively lowering the cost per unit. Also, provision is to be made for the launch of several satellites with one booster, to take advantage of the much lower cost per pound placed in orbit, with the larger rockets. The capability of a single launch, however, for a replacement or supplemental satellite is to be maintained. These factors, along with the decision that power is to be provided by solar cells, guided the overall spacecraft design.

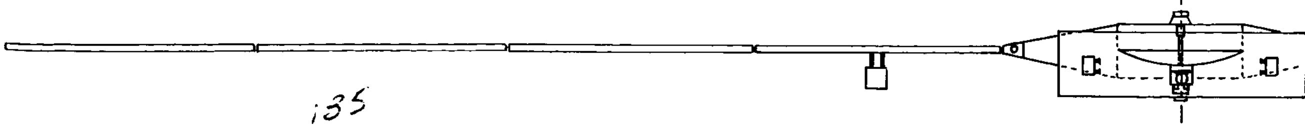
The power required from the solar cells dictates large solar array areas. These are to be unfolded in space and pointed at the sun, and they are the dominating physical aspect of the spacecraft. Refer to the spacecraft configuration in Fig. 55. Between the solar cell arrays is a flat core section which carries stabilization equipment, batteries, and other equipment that does not have to be in close proximity to the antennae system. With the solar arrays folded, the satellite has the general shape of a right triangular prism which allows the packing of four satellites within the roughly cylindrical booster shroud.

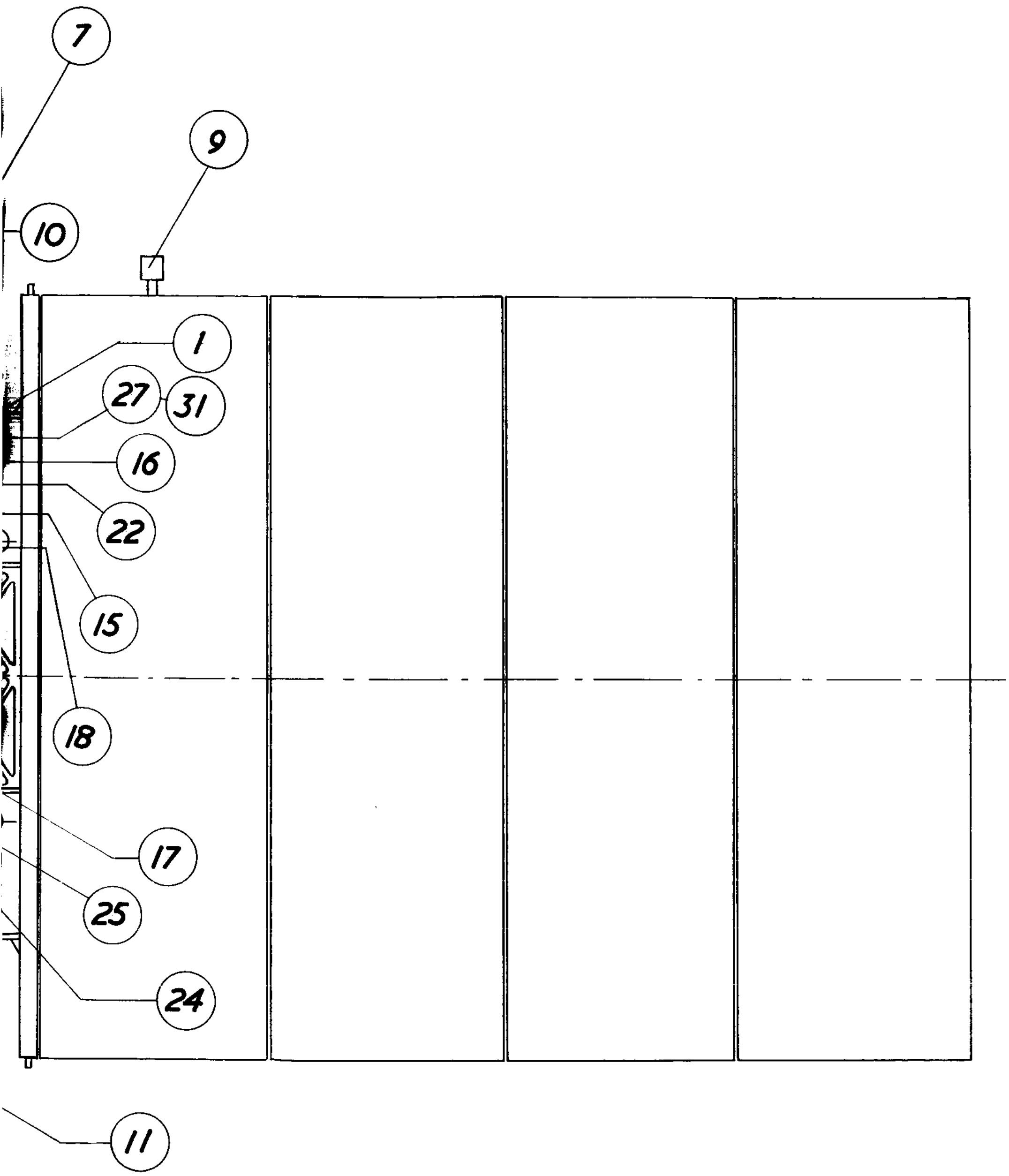
The antennae are unfolded top and bottom, and positioned on two connected platforms, one on top and one on the bottom of the central core section, and the entire antenna platform assembly can rotate with respect to the core section, with the antenna pointed at the appropriate position on earth. The antenna system, pointed at the earth, rotates once per day with respect to the rest of the satellite while the satellite itself, with solar cells pointed at the sun, rotates once per year with respect to inertial space.

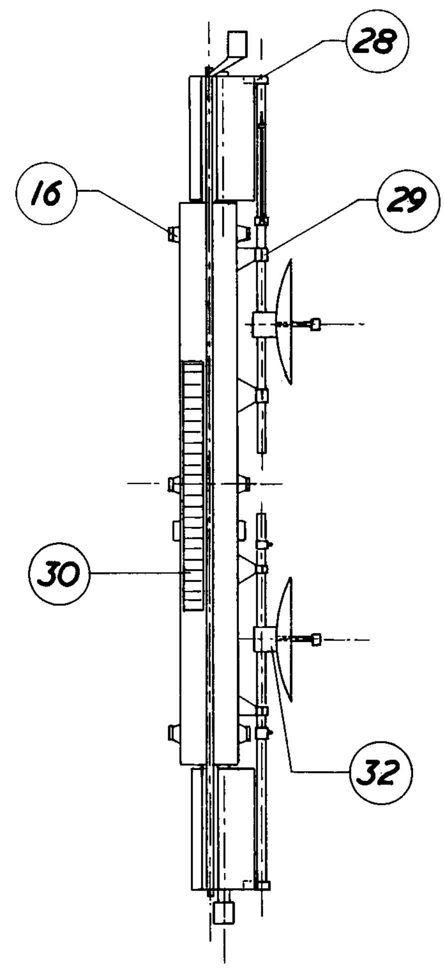
The satellite has a fully active attitude control system using reaction wheels and ion thrusters, which are also used for station keeping. The attitude control system can be adjusted for whatever accuracy in pointing that is desired, using only more fuel for greater accuracy. In addition, a hot gas jet system is on board, used initially and solely for correcting errors on injection into orbit.



FRONT (SUN-FACING) ELEVATION WITH
COVERS REMOVED.





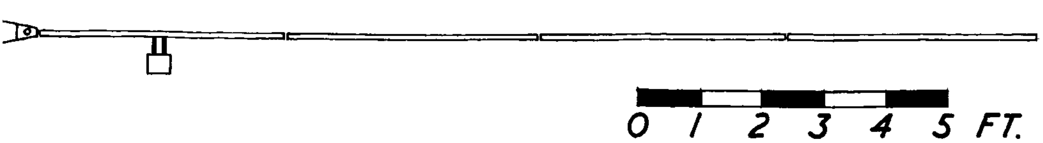


SIDE ELEVATION SHOWING ANTENNAS FOLDED.

. CONTROL SECTION
. COMMUNICATION ELECTRONICS SECTIONS (2)
. SOLAR CELL PANELS
. ANTENNA ASSEMBLIES (2)
. INTERFEROMETER ANTENNAE (2)
. TELECOMMAND ANTENNA
. TRAVELING WAVE TUBES (5)
. TRANSPONDERS
. SUN SENSORS (2)
. POLARIS STAR TRACKER
. INFRARED EARTH SENSORS (2)
. PITCH RATE GYRO
. RATE GYRO
. REACTION WHEELS (3)
. ATTITUDE LOGIC ELECTRONICS
. ION THRUSTERS (6)

17. HOT GAS THRUSTERS (4)
18. CESIUM STORAGE BOTTLES (4)
19. GAS STORAGE BOTTLES (2)
20. SOLAR CELL MODULE ARRANGEMENT
21. BATTERIES (4 UNITS) AND POWER REGULATION EQUIPMENT
22. POWER AND DATA SLIP RINGS
23. TELEMETRY UNIT
24. COMMAND SYSTEM
25. SEQUENCER TIMER
26. ANTENNA SECTIONS DRIVE MOTOR
27. BEARINGS (4) SUPPORTING DRIVE SHAFT (ITEM 31)
28. ANTENNA MAST HINGE
29. RETAINING CLAMPS (4) FOR ANTENNA MASTS
30. THERMAL CONTROL LOUVERS
31. SEE ITEM (27)
32. PREAMPLIFIERS (2)

Fig. 55 SATELLITE STRUCTURE.



Chapter 7

PHYSICAL CONFIGURATION AND STRUCTURAL DESIGN

7.1 The Launch Structure

During launch, and up to the time of injection into synchronous orbit, the four satellites are attached to the injection rocket support structure by means of the framework shown in Fig. 56. Each satellite is located between the upper and lower cruciform members by four explosive bolts set in the ends of the central section beryllium tube thrust members. The four spacecraft together thus form a bending torsion box structure. The folded solar panel assemblies rest against the foam plastic lined flanges and webs of the cruciform and central upright of the launch structure. Launch loads on the panels are due only to their own weight, as critical acceleration loads on the main spacecraft structure are not reacted through the panels. Because the total payload weight is not critical, the cruciform structure shown in Fig. 56 has been sized to demonstrate a high margin of safety relative to providing strength and stiffness for supporting the satellites through the critical design environments.

The satellites are centrifugally separated from the launch structure because of the residual spin of a few r/min of the injection rocket after firing and partial despin. The retaining explosive bolts fire simultaneously, and a spring mechanism ejects the upper cruciform structure forward from the central support tube. Springs on the flanges of the lower cruciform raise the spacecraft sufficiently to insure that they will not catch on the structure as they move outward from the framework.

The spinup and despin rocket motors, gas supplies, timers and power supplies are carried on the injection rocket support structure.

Weight Estimate

Upper cruciform	55 lb
Central upright	50
Lower cruciform	70
	175 lb

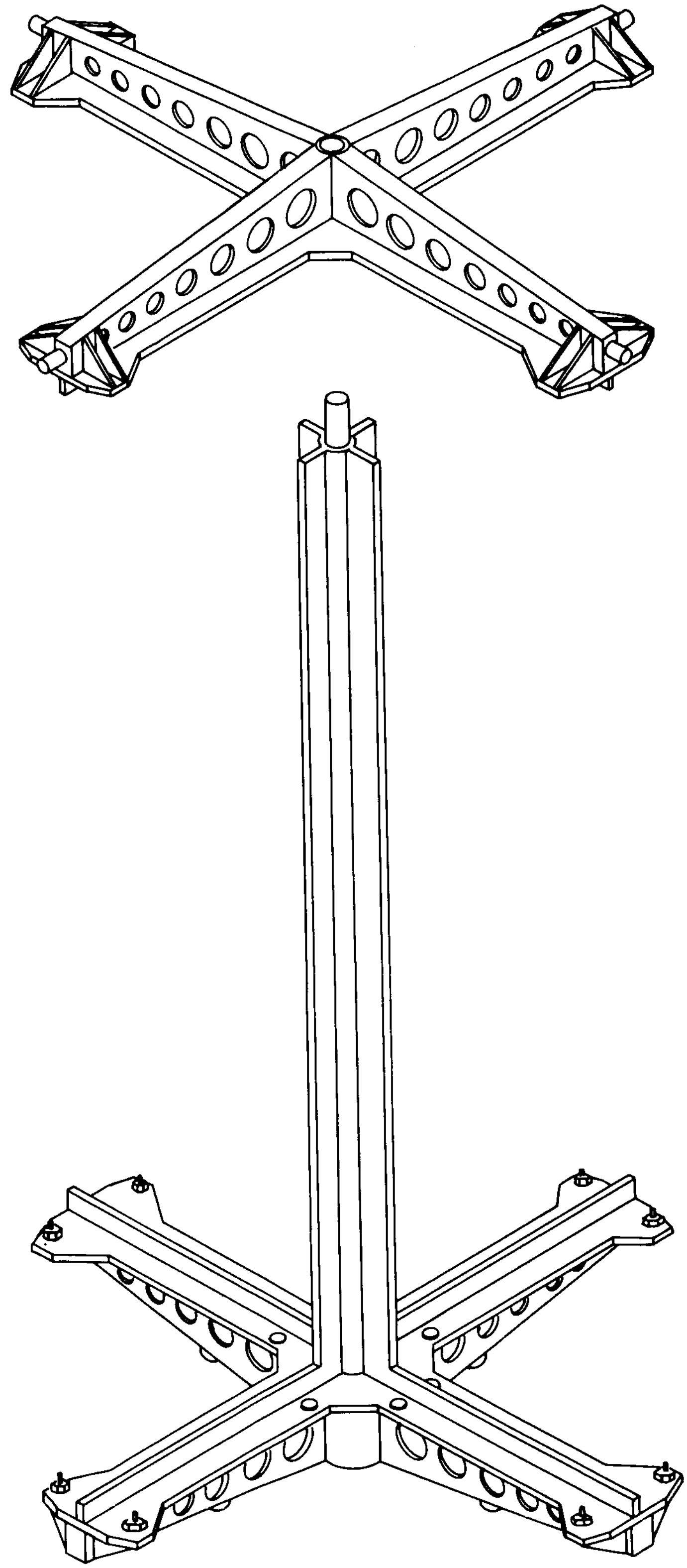


Fig. 56 CRUCIFORM LAUNCH STRUCTURE.

With injection rocket attachment structure, timers, spin and despin systems, and wiring, the total weight may be near 210 lb.

7.2 Spacecraft Deployment after Ejection from Cruciform

The post ejection deployment procedure is initiated by pyrotechnically releasing the solar array constraint bands after a predetermined delay, allowing the hinge springs to unfold the arrays. Frictional dampers on the hinges limit the rate of array deployment to avoid inducing dynamic load when the panels reach their stops at the full open position. Limit switches indicate completion of solar array deployment and allow the start of antenna erection. Drive motors simultaneously raise the antenna masts, completing the deployment sequence. Spacecraft orientation proceeds from this point.

7.3 Structure

7.3.1 Configuration

The configuration of the satellite is shown in Figs. 57-59. It consists of three major sections: the solar array, the antenna turret assembly, and the control section assembly. The shape and size of the configuration were dictated primarily by multiple launch considerations, power requirements, and attitude control constraints. (See Table 29 for weight estimate.)

The decision to use a multiple launch of four satellites on a Titan III C Centaur booster provides a space per satellite of a quarter section of a circle with a radius of 54 inches. Fitting a satellite into this space suggests that it be essentially a triangle when viewed from the normal to the payload platform. This would permit maximum satellite volume in the space provided. The power requirement of 3 kW at the solar array demands an area of 400 square feet which can be satisfied by a solar array consisting of folding panels. By designing the satellite as described herein, it has been possible to retain a high degree of symmetry. As a consequence, the center of gravity is essentially coincident with the geometric center. Adjustments in the actual location of the center of gravity can be made by shifting the position of the batteries in the central bay.

The satellite structure is indicative also of extremely efficient design concepts. Judicious use of beryllium and aluminum in

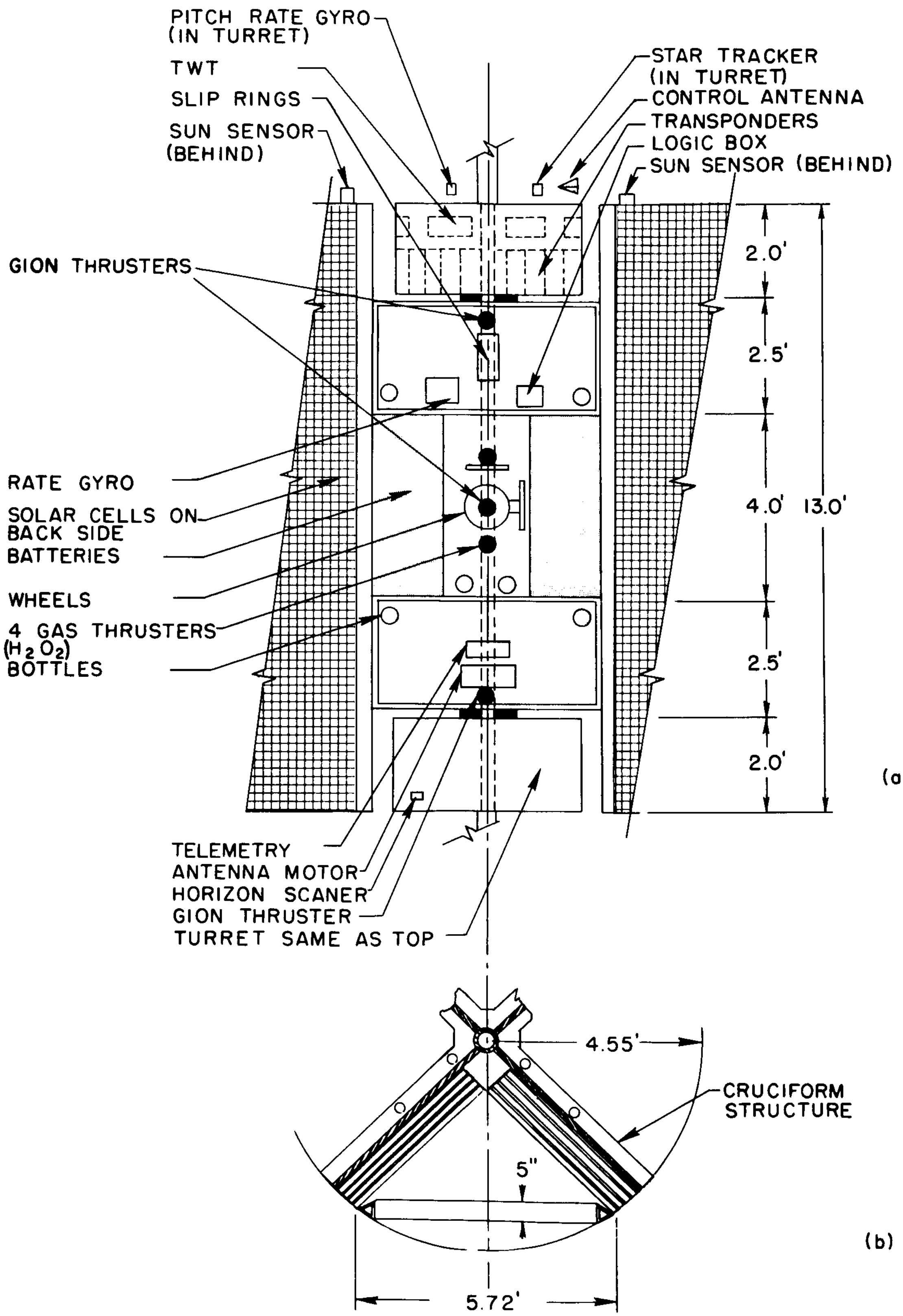


Fig. 57 CONTROL PACKAGE.

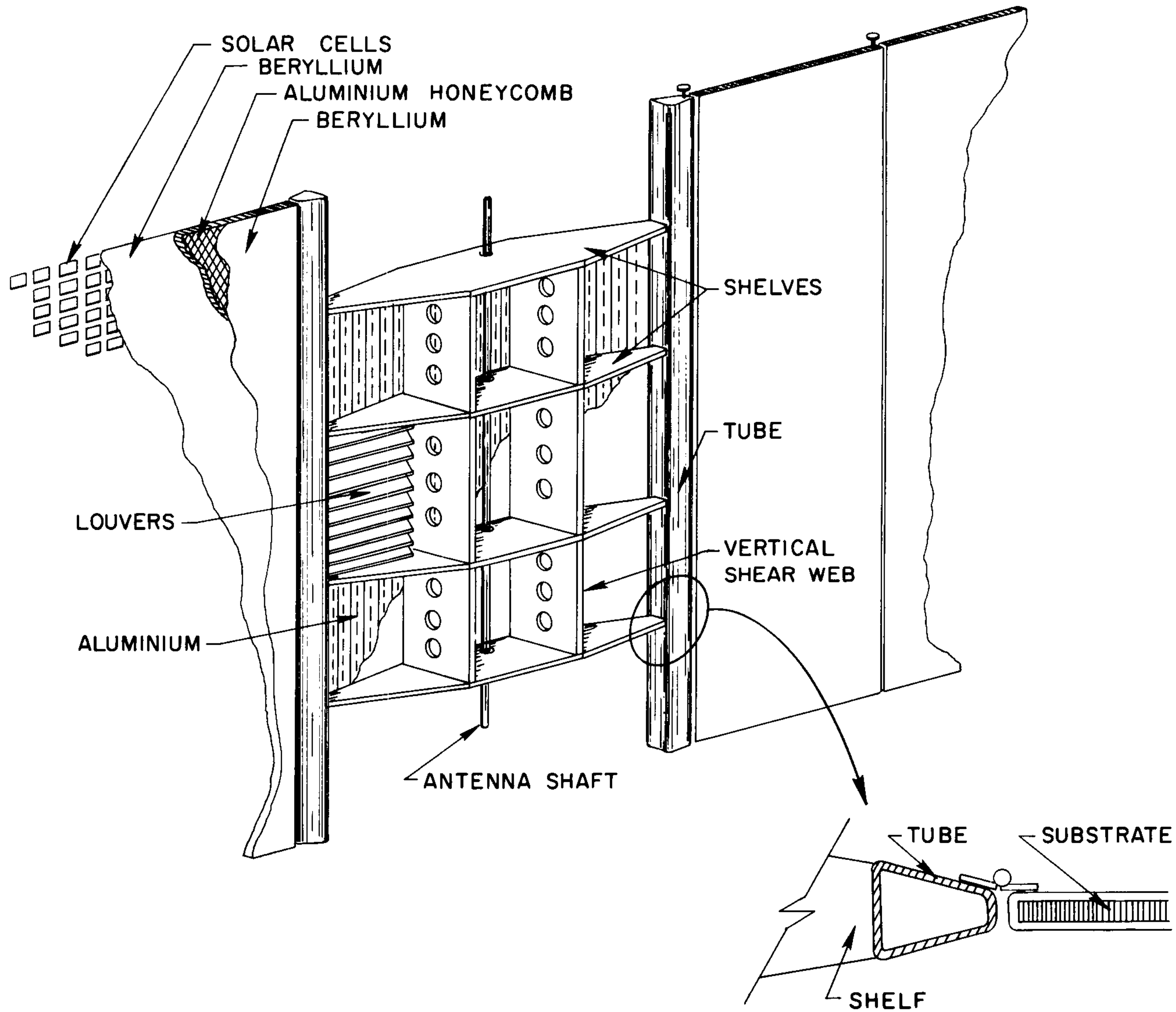


Fig. 58 CENTRAL STRUCTURE.

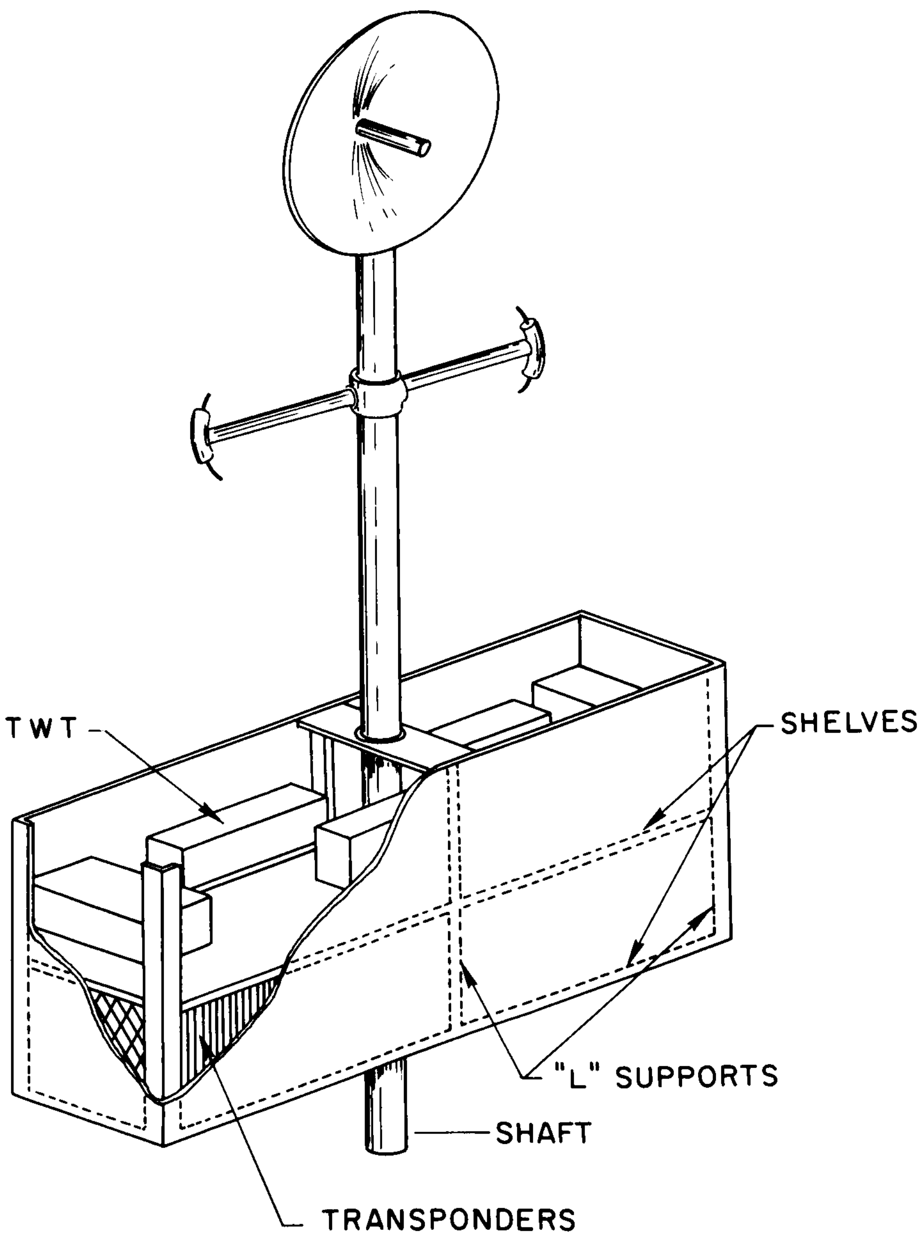


Fig. 59 ANTENNA STRUCTURE.

Table 29

WEIGHT ESTIMATE FOR ONE SATELLITE
(Pounds)

I.	Antenna			
	1. Preamplifiers		1.0	
	2. Interferometer		1.0	
	3. Interferometer electronics		3.0	
	4. Regional antenna		8.0	
	5. Structure		18.0	31.0
II.	Communications package			
	1. TWTs		120.0	
	2. Transponders		300.0	
	3. Structure		30.0	450.0
III.	Control package			
	1. 6 Ion thrusters		6.0	
	2. 4 Fuel bottles		40.0	
	3. 4 Gas thrusters and tanks		30.0	
	4. 3 Reaction wheels		19.0	
	5. Rate gyro		2.0	
	6. Logic box		5.0	
	7. Telemetry and command		10.0	
	8. Batteries		300.0	
	9. Control subsystem			
	Sun sensors	4.0		
	Star tracker	4.0		
	Pitch rate gyro	2.0		
	Horizon scanner	13.0	23.0	
	10. Control subsystem electronics		10.0	
	11. Antenna motor		2.0	
	12. Shaft and bearings		7.0	
	13. Rings		15.0	
	14. Fuel		30.0	
	15. Structure		36.0	533.0
IV.	Solar array			
	1. Cells		88.0	
	2. Structure		96.0	184.0
V.	Miscellaneous wires, hardware, etc.		50.0	
Total				1248.0 lb

conventional and honeycomb sandwich elements results in spacecraft structural weight that is approximately 17 percent of the total spacecraft weight. The consideration of structural efficiency coupled with multiple payloads and the higher power available makes this spacecraft system extremely attractive with respect to cost effectiveness.

7.3.2 Solar Array

The solar array was designed in the form of eight panels, each four feet wide and thirteen feet long. These are divided into two groups with four panels each, hinged together along the thirteen-foot dimension with one panel hinged to the frame of the control section. When folded, each group forms one leg of a triangle and the control section forms the third. When deployed, each group forms a rectangle 13 feet by 16 feet.

During launch, the panels are folded and secured to the payload cruciform and platform. They are supported on resilient material at the bottom to relieve stress on the hinges and to damp longitudinal vibration. Intermediate resilient spacers on approximately one foot centers between panels prevent scuffing of the solar cells and allow the folded array to behave essentially as a single laminated unit of significantly greater stiffness than an individual panel.

After ejection from the cruciform, the constraint on the panels has been removed and deployment begins. Deployment is accomplished by means of springs attached to the hinges which tend to unfold the array. Dampers are used to limit the rate at which the hinges open and prevent undue terminal load when the array attains the "fully deployed" position. Stops are provided to limit opening so that a flat array results. The springs then provide holding force to keep the array flat during its useful life. The solar panel hinges would be designed to provide metal to metal connections between adjacent members. The hinges include core inserts which serve to diffuse loads and improve damping qualities through the medium of shear. The dampers do not inhibit folding the array. The structural analysis of the spacecraft including panels is given in Appendix E.

7.3.3 Antenna Turret Assembly

The triangular arrangement for the folded satellite also provides space for storage of the antennae during launch. These are mounted on "fold-down" masts, made of aluminum tubing, and occupy the inner volume of the folded satellite during launch. The masts are deployed after the solar panels are deployed. Since the satellite is actively stabilized, it is possible to deploy two antenna masts, one on each end of the control section.

Figure 57 shows that there are two antenna turrets. These are joined on a common hollow aluminum shaft and rotate as a unit with respect to the control section. Each turret houses the communication electronics equipment including the traveling wave tubes, which furnish the radio frequency power for the antennae. This arrangement allows hard line coupling between the TWTs and the antennae. Power for the communications equipment is carried to the turrets over silver plated slip rings with silver molybdenum disulfide brushes on the turret shaft. The earth sensors are also mounted on the turrets and the circuits are carried to the stabilization logic unit in the control section over separate slip rings.

The satellites will differ in the specific number, size, and shape of antennae, but the basic turret is used in all cases. The maximum number of TWTs that may be carried by a single turret is five; however, in some cases, fewer may be used. The turret structure is a basic beryllium space framework closed by beryllium cover sheets. Two of the covers, for structural efficiency and thermal considerations, serve as outside walls of the TWT packages. The additional electronics are packaged on two beryllium aluminum honeycomb sandwich shelves. Preamplifiers are mounted on the mast directly behind the antennae. During launch, the antenna masts are folded and locked in place on brackets attached to the control section housing. Upon initiation of deployment, the masts are released and deployed by a dc motor drive through a gear train. When the mast reaches the "fully deployed" position, it is locked in place and the motor is de-energized. This arrangement is shown in Fig. 59.

7.3.4 Control Section

The control section of the satellite is also the main structural frame of the spacecraft. It houses all control equipment except the earth sensors, which are located on the turrets, and the sun sensors, which are located on the solar panels. The frame is made from two extruded beryllium tubes, shaped as shown in Fig. 58, and joined by four aluminum honeycomb sandwich shelves with front and back housing covers of aluminum sheet. The shelves are stiffened by two aluminum vertical shear webs, providing an efficient structure for resisting both bending and torsion.

The front of the satellite shall be defined as that surface which faces the sun. The front covers are permanently mounted to the frame and serve as mounting surfaces for control system components. The back covers are bolted to the frame but are removable to provide access to all internal components. By this arrangement, complete access to all circuits and components is available when the satellite is in the folded condition in place on the launch platform. It also provides access during various phases of fabrication and test.

The use of four shelves divides the volume of the control section into three bays: the upper, lower and central bays. Refer to Fig. 57. The central bay houses the batteries in two groups, one on each side. In the center, the three control stabilization reaction wheels are mounted so that each lies on its respective control axis. All are kept as close to the center of gravity of the satellite as possible (within a few inches). In addition, the central bay houses control thrusters mounted on the yaw axis for station-keeping. Two of these serve a secondary function of supplying heat for the battery compartment during launch and during the shadow time of the orbit.

The upper and lower bays house the fuel bottles for the control system as well as various other components such as the control logic unit, the telemetering equipment, etc.

The shelves provide a mounting for the turret shaft which passes vertically through the control section riding in bearings on each shelf. These bearings carry the thrust load during launch and provide proper alignment of the turret assembly. Two slip ring assemblies are used to carry circuits to the turrets. One of these is a nine-ring unit which will carry dc power to the communication equipment. The other carries all data links between the turret and control sections. These are mounted on the shaft in the upper bay. The drive motor for controlling the turret position is located in the lower bay.

Chapter 8

ATTITUDE CONTROL SYSTEM

8.1 Mission Requirements

The satellite considered in this study is a synchronous orbit communications satellite. Some of the antennae are required to point at certain specific locations on earth with a high degree of accuracy. The mission requires that orientation of the satellite satisfies the following stringent constraints.

(1) orientation of the antennae with respect to the earth station to be maintained with $\pm 0.1^\circ$ accuracy

(2) orientation of the solar panels with respect to the sun.

In addition to the above requirements the satellite is to be designed to be operational for a minimum period of five years, and to be capable of generating 3 kW of solar cell power at the end of that period. The control system therefore must be capable of acquiring the correct attitude as soon as possible after deployment, and also of maintaining this attitude during the anticipated five-year life of the satellite. The control system should be efficient so that no unnecessary fuel is consumed. Once the satellite is correctly aligned, the control system must be able to continuously correct for errors induced by the perturbing torques acting on the satellite.

8.2 Choice of Control

When a satellite is placed into orbit, it is necessary to acquire a specific attitude. There are three basic methods of controlling the attitude of a satellite:

(1) passive control

(2) semipassive control

(3) active control.

In this study, all three of these basic approaches are considered.

(1) A satellite is said to be passively controlled if the design is such that there is a natural tendency for the satellite to maintain a desired orientation. The main environmental factors considered in this study were the gravity gradient torques. These torques act about the satellite center of mass whenever the mass center, gravity center, and earth center are not collinear. In the calculations made for this study, an eccentricity of 0.5 ft was used. Should this not be the case, appropriate corrections can be applied to the calculations. The gravity gradient torques are inversely proportional to the cube of the distance from the center of the earth. For a synchronous orbit, these torques are very small compared with other perturbing torques. To increase the magnitude of these small torques, long booms with large tip masses (on the end of the booms) must be used. By making the weight at the tip of each boom approximately 10 percent of the weight of the satellite, the length of each boom must be 1300 ft to provide gravity gradient control. The choice of the value of 10 percent is purely arbitrary. In the case of a satellite weighing 1600 lb, three 1300-ft booms with a 160-lb tip weight on each are required to provide gravity gradient control. The use of such long booms in synchronous orbit presents problems, and it is doubtful if the required degree of pointing accuracy could be attained; therefore, the gravity gradient control system was rejected.

(2) A semipassive control system may be defined as one which makes use of the basic passive environmental factors, but may use sensors and thrusters to acquire and maintain the desired orientation. The principal method of attaining this type of control is the well known method of spin stabilization. In general, a spin stabilized satellite must have a cylindrical or "pancake" shape, and it must be spinning about an axis of maximum moment of inertia. These constraints limit both the length/diameter ratio and the size of any despun sections. Also, since launch vehicle shrouds are generally longer than

they are wide, the "pancake" shape constraint may result in a nonoptimum use of the shroud. Because of the high power requirements of the satellites, it was considered that the cylindrical or "pancake" shape was not feasible in this design. To achieve the high power requirement with solar cells, it is proposed to make use of expandable solar panels which will give a total area, when deployed, of 400 sq feet. Taking this into account, and also the fact that one or more antennae must be pointed accurately toward the ground station, it appears that the mass of the satellite capable of being spun is small. For this reason, spin stabilization as such is not an attractive system. However, a configuration in which part of the satellite spins and controls a large despun platform has been proposed, and is known as the Hughes Gyrostat System. The Gyrostat System is not constrained to have a "pancake" shape, or a perfectly rigid rotor as is the case with the "standard" spinning concept. The Gyrostat concept is a broad application of spin stabilization, and may be used to provide a fully stabilized platform while maintaining desirable spinning concept features. This concept was very seriously considered in this design, and is proposed as an alternative attitude control system. The main reasons that the Gyrostat System was rejected as the primary control system in this design are as follows:

(a) The configuration of the present design which meets the required communications and power constraints is such that the weight of the spin rotor is very small compared with the total weight of the satellite. Hence, the rotor spin moment of inertia will be very small. Therefore, to achieve large gyroscopic stiffness, which is desirable for accurate control, large spin rates must be used.

(b) The communication electronics, traveling wave tubes, and associated electronic equipment are mounted on the despun platform. Therefore, with the proposed configuration, advantage cannot be taken of the passive thermal control system used with spinning satellites.

(c) Multiple launching of four satellites at a time without individual apogee kick motors will be used. Therefore, the advantages claimed to exist by spinning are lost.

(d) Since the antennae are housed on the despun platform, and must be pointed toward the earth with an accuracy of $\pm 0.1°$, complex attitude sensors must be used even with the Gyrostat System.

(3) To meet all the mission requirements and confine the control system within the specified satellite configuration, a fully active attitude control system was chosen. There are three main advantages to be gained by the use of this system:

(a) It is more versatile.

(b) It provides a high degree of accuracy.

(c) It is more adaptable to complicated maneuvers.

The design provides for considerable redundancy, which increases the long life reliability of the active control system. Reaction wheels will be used to minimize the inherently large amplitude limit cycle oscillations. Six ion thrusters are proposed. Since the ion beams can be vectored by at least 30°, the 6 ion thrusters will produce decoupled control thrusts and torques in all directions and with considerable redundancy. On the other hand, at least 14 cold gas jets would be necessary. Because of their high specific impulse, propellant weight is much less for ion thrusters than for gas jets.

8.3 Attitude Acquisition

The Attitude Control System (ACS) is required to keep the body yaw, z-axis, of the spacecraft (S/C) aligned with the earth's local vertical within $\pm 0.1°$ at all times following acquisition. Pointing accuracy of the antennae is capable of being maintained to within $\pm 0.1°$. The solar array faces must align with the sun. Figure 60 defines the S/C body coordinate system.

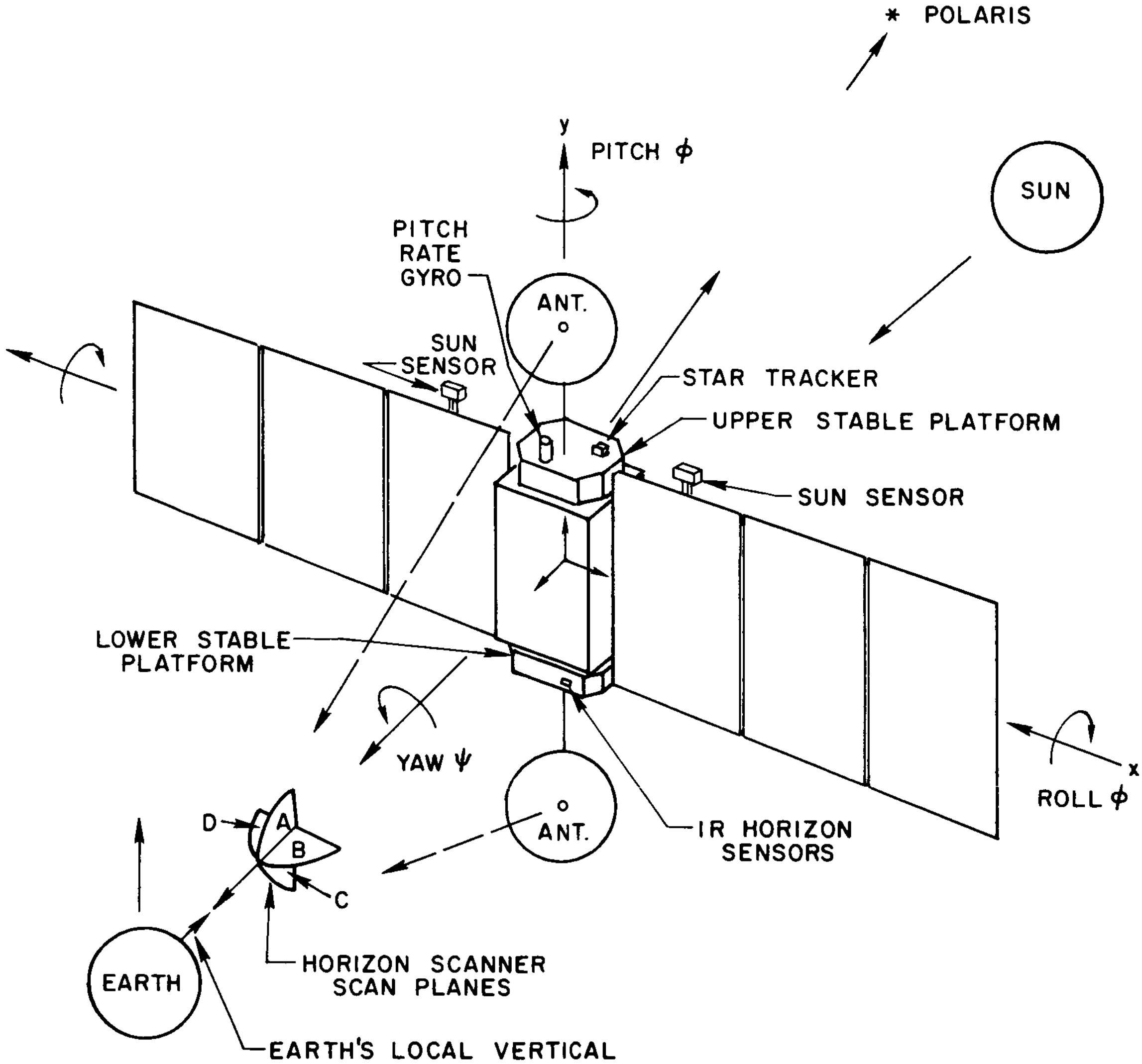


Fig. 60 SPACECRAFT COORDINATE SYSTEM.

Sensor requirements include:

One star tracker

Two dual IR earth horizon scanners

Two sun sensors

Two RF interferometers.

A system of IR earth horizon scanners is required to provide an error signal when the z-body axis is not aligned with the earth's local vertical resulting in pitch and roll control. The sun sensors are mounted on the array to provide vehicle initial acquisition and to orient the solar array panels.

The field of view of the sun sensors encompasses two steradians (1) and indicates the deviation of the perpendicular to the array from the sun. The resultant axis information provides coarse error signals which control the vehicle yaw angle. After acquisition of Polaris, the star tracker locks the spacecraft into a precise vertical position with respect to the earth's local vertical. The RF interferometer provides antenna pointing error signals for fine control as well as some redundancy in the overall sensing capability.

Body control torques are provided by means of either gas jets or ion thrusters for angular momentum removal and a reaction wheel system for momentum storage. The thruster assembly is so designed as to produce couples about the x, y, and z axis in either a positive or negative sense during both the acquisition mode and the normal mode of operation. Likewise, the reaction wheel assembly will also provide couples.

In the acquisition phase, the ACS is capable of orienting the vehicle to required references when given angular errors and body angular rates about all three axes. The acquisition phase starts following the multiple ejection of the four spacecrafts into synchronous orbit from the second stage booster. As each S/C is ejected, it is given a slight turning torque about its pitch axis as well as variable ΔVs to aid in drift to its desired position in synchronous orbit. After a short predetermined period following ejection, the solar arrays are deployed and the sun acquisition mode starts. A programmed sequence requires completion of array articulation and antennae unfurling. The desired orientation is achieved in four modes: sun acquisition, earth search, star fixation, and RF-interferometer pointing control.

The sun acquisition mode orients the yaw axis along the sun vector, and aligns the pitch axis (y-axis) so that it is parallel to the earth's polar axis. The error signals during the sun acquisition mode are provided by the sun sensors and the pitch rate gyro. The latter is oriented parallel to the pitch axis on the upper stable platform. Initially pointing the S/C body to the sun assures early battery charging and proper orientation for temperature control. The fields of scan of the two dual IR horizon scanners intersect on the +z-axis which is to be oriented toward the earth's local vertical.

During the earth's search mode, the S/C is oriented the same as in the sun acquisition mode and IR horizon scanner axis sweeps out a plane. The stable platforms may be driven to aid in the search and this plane must intersect the earth. When this occurs, almost normal control is achieved. The vehicle either drifts into proper orbit position or is driven via ground telecommand signals.

During S/C positioning within synchronous orbit, the star tracker locks onto Polaris, precisely positioning the craft to the earth's local vertical. When orientation is within required accuracy in both the pitch and roll axis, the RF-interferometer mode results in fine control. Once the RF-interferometer mode is complete, the pitch rate gyro is inhibited.

The ACS is designed with a considerable amount of redundancy in order to meet variable pointing accuracies as required for either regional or international satellite application. The sensor requirements as outlined are typical of the OGO system developed by TRW. Accuracy capability of SAINT is enhanced by the star tracker and RF-interferometer systems. During normal mode operation, either the IR horizon sensors or the RF-interferometer system may be inhibited, depending on accuracy requirements of the particular satellite. Furthermore, precise control can be maintained during shadow period by use of the RF-interferometer system. State of the art listings of typical sensors are listed in Table 30.

8.4 Operational Mode

The design philosophy of ACS operation centers about the earth's local vertical and the sun vector as primary reference axes with the star,

Table 30

ATTITUDE CONTROL SENSORS

Type	Weight (lb)	Power (W)	Resolution	Cost	Acquisition Technique
Star tracker	4.0	5	0.025 to 0.1 minutes of arc	\$25,000 to \$50,000	Optical
Infrared horizon scanners	6.5	7	0.1 to 0.8 degree of arc	\$25,000 to \$60,000	Prism scan
Sun sensor	2.0	3	Precision: 0.6 to 1.8 sec of arc ±20 min cone Fine: 1.1 to 10 milliradian ±17 degree cone 8 to 100 milliradian ±70 degree cone	\$ 5,000 to \$50,000	Solar cells to combinations of solar cells and radiation tracking transducers (RTT)
Radio-frequency-interferometer	3.0	10	0.05 degree of arc	\$30,000	---

Polaris, as a precise reference to permit antenna pointing with respect to the yaw axis.

The control and stabilization subsystem consists of sensors, electronics, control torque sources, and electromechanical drive mechanisms. The star tracker locks onto Polaris and precisely assures antenna pointing accuracy with respect to yaw. Sun sensors are used to obtain error signals which orient the face of the solar array with respect to the sun. Horizon scanners which track the temperature gradient between the earth's horizon and space are utilized in normal control to provide the pitch and roll error signals, indicating the deviation of the yaw axis from the earth's local vertical. A single degree of freedom rate-integrating gyro is used to provide pitch angular rate reference information which aids in establishing inertial reference during acquisition. Electronic and logic circuitry provide the amplification, signal conditioning, and logic required to sequence and actuate the torque sources and drive mechanisms.

Torques for vehicle attitude control and station-keeping are provided by six ion thrusters and inertia flywheels. Control signals are "bang-bang," either full on positive or full on negative, resulting in a simple reliable control system. Each channel in roll, pitch, and yaw consists of a reaction wheel which operates in parallel with the ion thrusters to keep the S/C in normal control position. It is assumed that the thruster deadband can be set at about twice the reaction wheel deadband. The two stable platforms are provided with a servo system and drive mechanism for proper positioning with respect to the S/C axes. The torque for drive is provided by a small servo motor geared through a hermetically sealed spur gear train, which in turn drives an exposed wobble gear drive.

8.4.1 Star Tracker

Star trackers are available with accuracies of 0.025 to 0.1 minute of arc or better, and fields of view of from 0.2° square to 5° square. One will be needed on SAINT to precisely orient the S/C to the earth's local vertical.

8.4.2 Horizon Scanner

The horizon scanner system is comprised of two dual infrared sensors which perform the function of generating pitch-and-roll error signals and earth edge position signals which are processed by the sensor electronics and logic assembly. The accuracy of typical horizon sensors ranges from 0.1° of arc to 8 milliradian. The horizon scanner system operates as follows.

Four infrared search-track units track points separated in vehicle azimuth by 90° on the earth's horizon. Signals generated by each tracker are proportional to the angular displacement of the horizon from the nominal vertical of the S/C. Each dual-tracker-head assembly contains two trackers which scan in the same plane, but whose fields of view are separated 180° in azimuth. Each head is mounted on the vehicle such that the trackers scan in the vertical planes with respect to vehicle coordinates. The heads are mounted at right angles so that the four scan planes sweep out a pattern as shown in Fig. 60.

8.4.3 Sun Sensors

The sun sensors are used to orient the vehicle during initial acquisition and to orient the solar array panels toward the sun at all times. The sun sensor system is comprised of two identical +X and -X units mounted approximately 13 feet apart on the solar array panels. The accuracy of typical sun sensors ranges from 0.6 second of arc to 100 milliradian and are classified precision, fine, and coarse. Coarse accuracy sensors are adequate for the needs of SAINT. The outputs of each of these detectors are parallel summed at the sensor electronics and logic assembly.

8.4.4 Pitch Rate Gyro

The pitch rate gyro assembly establishes a reference angular rate about the S/C pitch axis during sun and earth acquisition. In the normal mode of S/C control, it is inhibited. The pitch rate gyro assembly is comprised of a subminiature rate gyroscope, signal amplifying and demodulating electronics, and telemetry processing electronics. External power supplies furnish direct and square wave voltages as needed.

8.4.5 RF-Interferometers

There should be two interferometers with the dipoles spaced at 1 meter (20λ @ 6 GHz). The interferometers should be located on either of the antenna array areas so that they will not be blocked from viewing the earth. The pitch interferometer dipoles should be parallel to the pitch axis and in the plane of the face of the antenna array. The general arrangement is shown in Fig. 61. The roll interferometer dipoles should be parallel to the roll axis and in the same plane as the pitch dipoles. The operation of the interferometer system is described in Appendix H. The total weight of the system would be about 3 lb; with power requirements not exceeding 10 W. Associated equipment would be a power supply and logic unit which would accept output signals from the interferometers and activate proper corrective torques.

8.5 Mass Expulsion Control System

Cesium bombardment ion thrusters were chosen as the source of thrust for both attitude control and station-keeping. The major reasons for this choice are

(1) high specific impulse ≈ 2000,

(2) placement requirements due to spacecraft structure and retractable antennae,

(3) low fuel consumption.

In addition, the vectoring capability of ion engines allows one microthruster to replace a cluster of gas discharge jets. At the present state of the art, ion thrusters may be vectored by 30°; thus, one may attain both torques and translation thrust with one ion engine. The selection of ion engines permits the use of 4 microthrusters to obtain decoupled translation and rotational forces. In contrast, a conventional gas jet system would necessitate 12 gas jets to accomplish the same decoupled conditions. To provide redundancy, the ion engine system chosen uses 6 ion engines. The location and operation of the thrusters is shown in Fig. 62. The pair of redundant thrusters centered on the control package will provide decoupled translation and rotation in all modes except yaw, but at a lower thrust and torque level.

The specific ion thrusters chosen deliver a thrust of 150 micropounds at a specific impulse of 2000. Tests have shown that the present life of ion thrusters is at least 20,000 hours. Improvements in ionizer design are such that at least a 10-year life is to be expected in the near

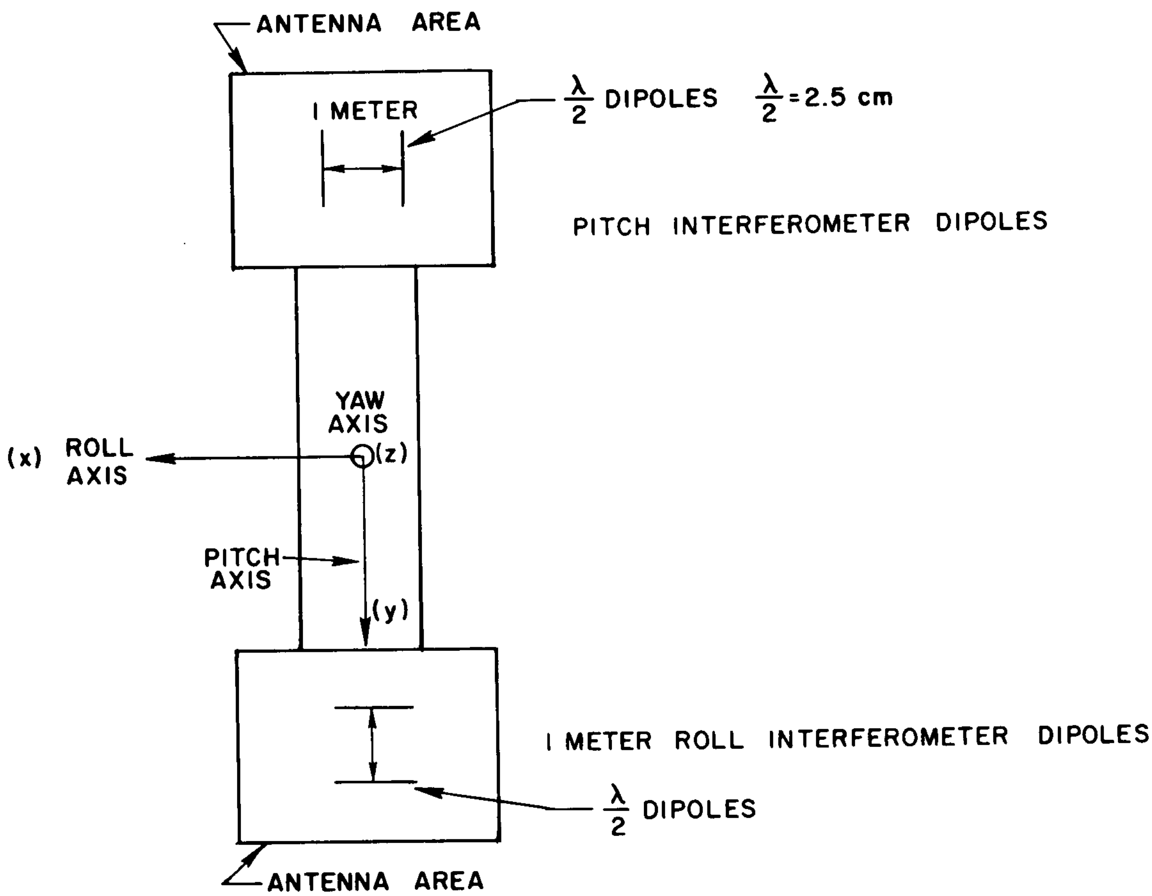
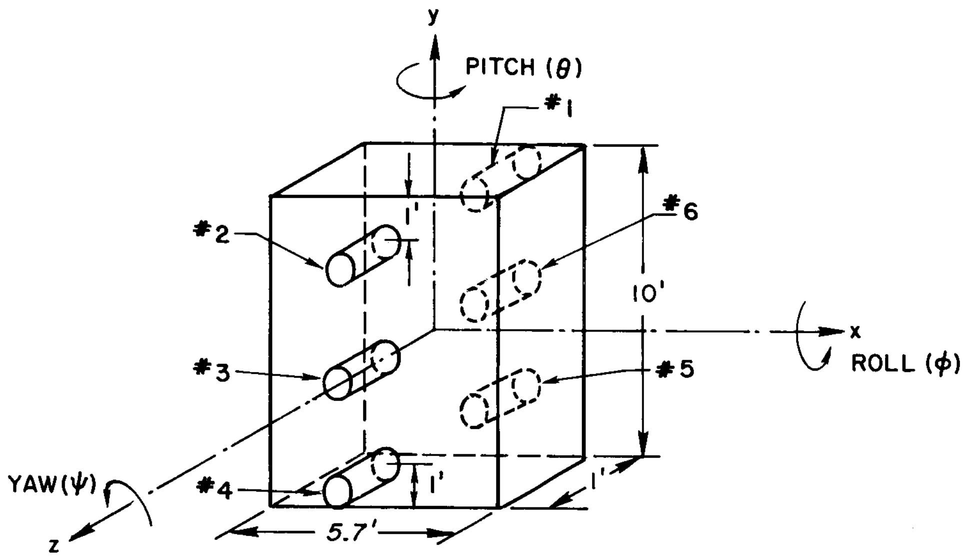
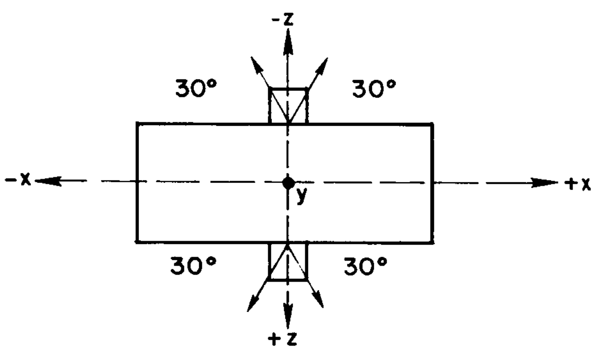


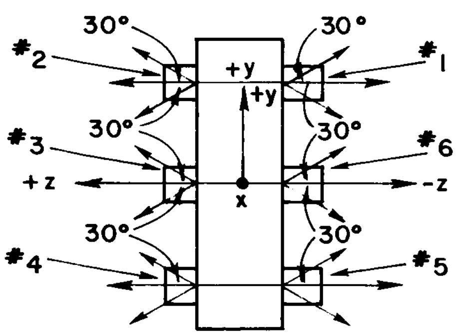
Fig. 61 INTERFEROMETER.





TRANSLATION

DIR.	THRUSTERS	VECTOR
+ x :	1, 2, 4, 5	30°, − x
− x :	1, 2, 4, 5	30°, + x
+ y :	1, 2, 3, 4, 5, 6	30°, − y
− y :	1, 2, 3, 4, 5, 6	30°, + y
+ z :	1, 5	0°, − z
− z :	2, 4	0°, + z



ANGULAR TORQUES	THRUSTERS	VECTOR
PITCH:		
+ θ	1, 5	30°, + x
	2, 4	30°, − y
− θ	1, 5	30°, − x
	2, 4	30°, + x
ROLL:		
+ φ	1, 5	30°, − y
	2, 4	30°, + y
− φ	1, 5	30°, + y
	2, 4	30°, − y
YAW:		
+ ψ	1, 2	30°, + x
	4, 5	30°, − x
− ψ	1, 2	30°, − x
	4, 5	30°, + x

Fig. 62 THRUSTER OPERATION.

future. The ion system is the lightest weight available for long life station-keeping and attitude control. In addition, it has the smallest minimum impulse bit. Ion engines are to be flight tested on the ATS-D and E flights in NASA's Applications Technology Satellite Series.

Using 150-μlb thrusters and 30° vectorings, the maximum translation forces and rotational torques are

± X direction	=	450 μlb
± Y direction	=	450 μlb
± Z direction	=	450 μlb
± Pitch	=	360 μft lb
± Roll	=	1320 μft lb
± Yaw	=	1200 μft lb

The antennae must be pointed toward the earth with an accuracy of ±0.1° to ±0.5°. Sensors placed on the antennae platforms transmit antennae position data to the control logic unit. In addition, the angular position of the antenna platform relative to stable platform is fed into the control logic unit as well as signals from the solar sensors located on the solar panels. The logic unit decodes these signals and triggers the appropriate thruster bias and ignition control circuit as well as the solar panel drive motor. A block diagram of the control loop is shown in Fig. 63.

The ion thrusters require as many as six different voltages (ac and dc), ranging in magnitude up to about 5 kV. Some of these voltages must be accurately controlled in response to closed loop signals from sensing element in the thrusters themselves. Recent developments include lighter power conditioning and control circuits. The technology for systems requiring up to 10 kW is now well in hand. A conservative estimate of the electrical power requirements for the ion thruster system is 180 W/millipound. Thus, each 150-μlb thruster will require about 27 W. The total power required for the thrusters is thus 162 W.

Based on 150-μlb ion thrusters with a specific impulse of 2000, calculations indicate that continuous operation for five years of one thruster would require 12 lb of cesium fuel. Thus, a very conservative estimate of fuel requirements for the entire system would be 70 lb. The total system weight including power conditioning and control is estimated at 120 lb.

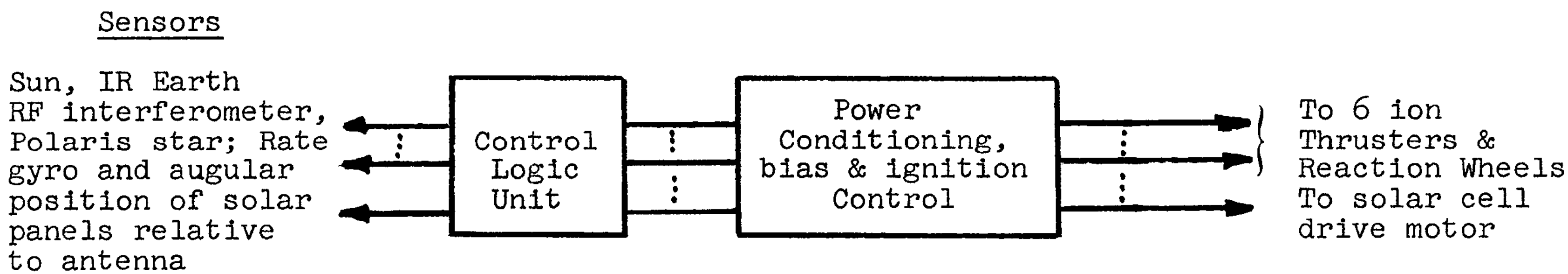


Fig. 63 BLOCK DIAGRAM OF CONTROL.

8.6 Momentum Transfer Reaction Wheels

The reaction wheels will be similar to those used on the OGO satellites, i.e., motor driven inertia wheels in hermetically sealed cases filled with argon gas. The motor and wheel are built as one unit, the motor being an "inside out" two-phase induction motor. The reaction wheels apply torques for controlling angular positions about the pitch, roll and yaw axes. Three reaction wheels are mounted so that their spin axes are aligned with the pitch, roll and yaw axes, respectively, of the solar panels as shown in Fig. 64. Note that all wheels are mounted as near to the center of gravity as possible. As power is applied to a reaction wheel motor, the motor torque accelerates or decelerates the wheel, changing its momentum. The reaction to the motor torque acting through the motor mounting rotates the spacecraft about that particular axis. The power for driving the motors comes from a 400 c/s, two-phase inverter source.

The roll and yaw axes of the stable platform periodically interchange positions as the spacecraft moves around the earth. See Figs. 65 and 66. Therefore, the momentum storage capacity of the yaw and roll reaction wheels must be the same. The cyclic torques about the pitch, roll, and yaw axes plus the capability of storing secular momentum change until it is expelled by the gas system, were used in sizing the wheels. The difference between the roll and yaw axes in Fig. 64 of the spacecraft and the panels is due to their relative motion about the pitch axis.

Table 31 summarizes the results and calculations in Appendix J.

Associated equipment required includes a two-phase inverter to provide 125 V at 400 c/s. Logic units are required for driving motors in proper direction, and to convert the error signals of coordinates referred to the stable platform to coordinates referred to the solar panels or where the wheels are mounted. The wheels can be made with integral tachometers for measuring speed and direction of rotation.

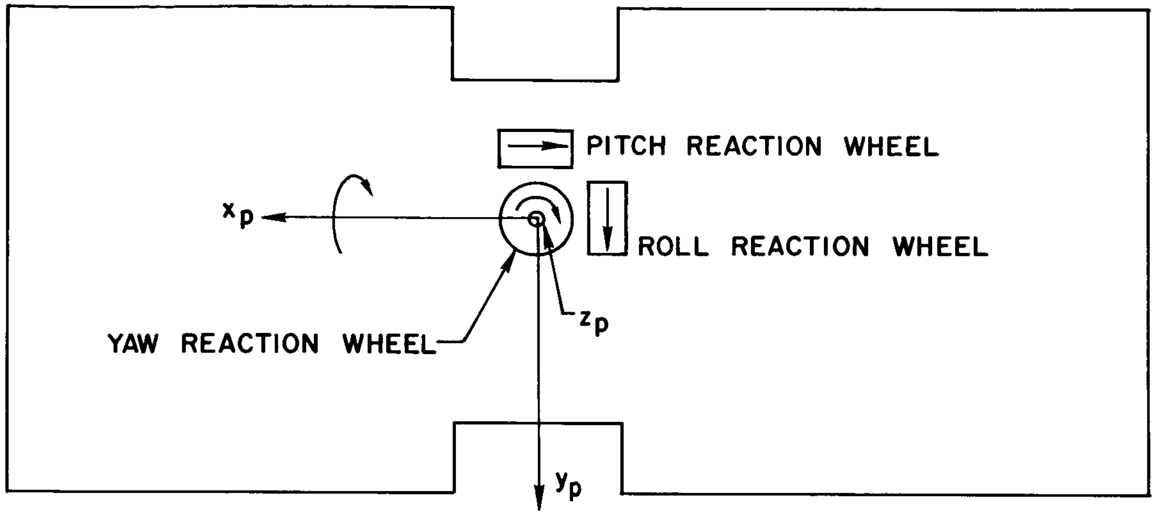


Fig. 64 AXES OF PITCH, ROLL, AND YAW FOR SPACECRAFT AND SOLAR PANELS.

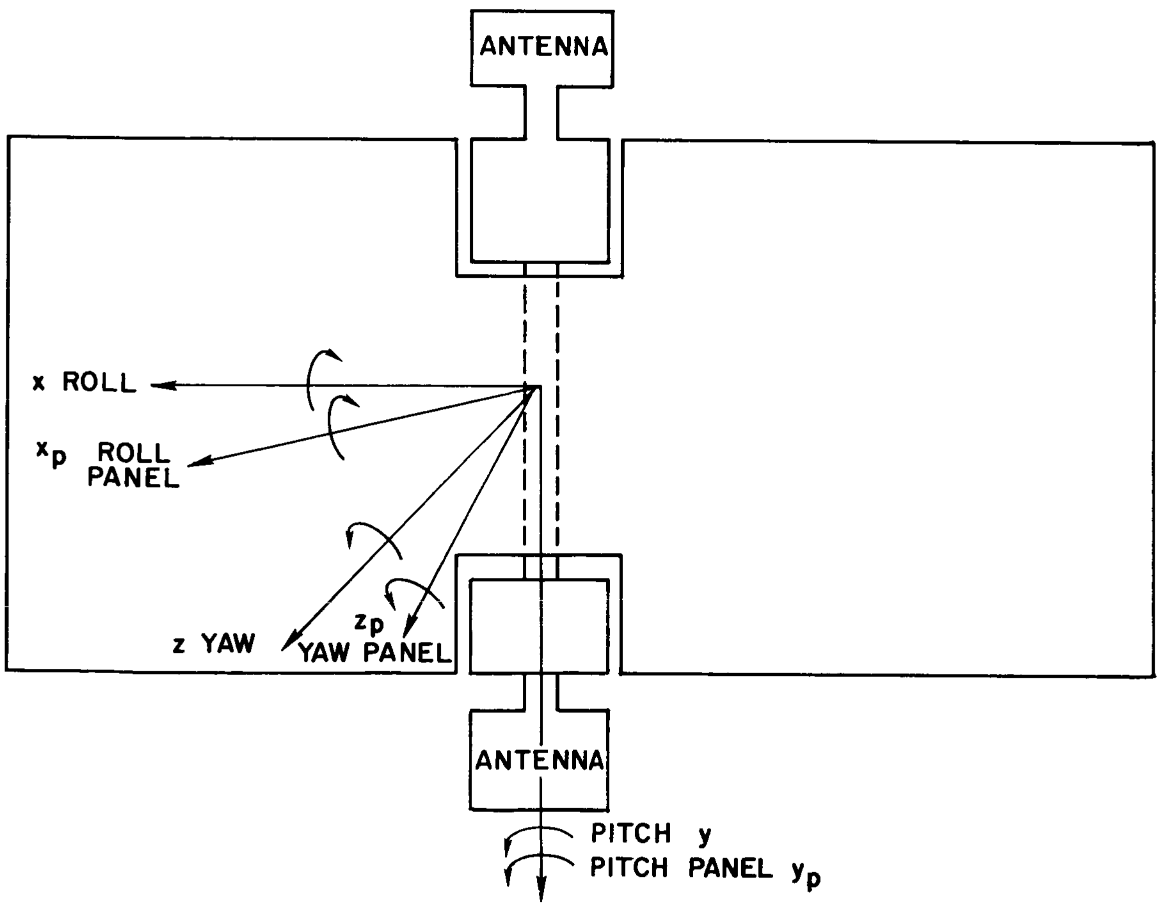


Fig. 65 LOCATION OF REACTION WHEELS.

Table 31

REACTION WHEEL PARAMETERS

Control	Weight (lb)	Radius of Gyration	Power (watts)	Diam.	Height
Pitch	10 (8.5)	3"	8 (40-stall)	7"	4"
Roll	4.5 (3)	3"	5 (20-stall)	7"	4"
Yaw	4.5 (3)	3"	5 (20-stall)	7"	4"

Note: Weight in parentheses is wheel weight less case

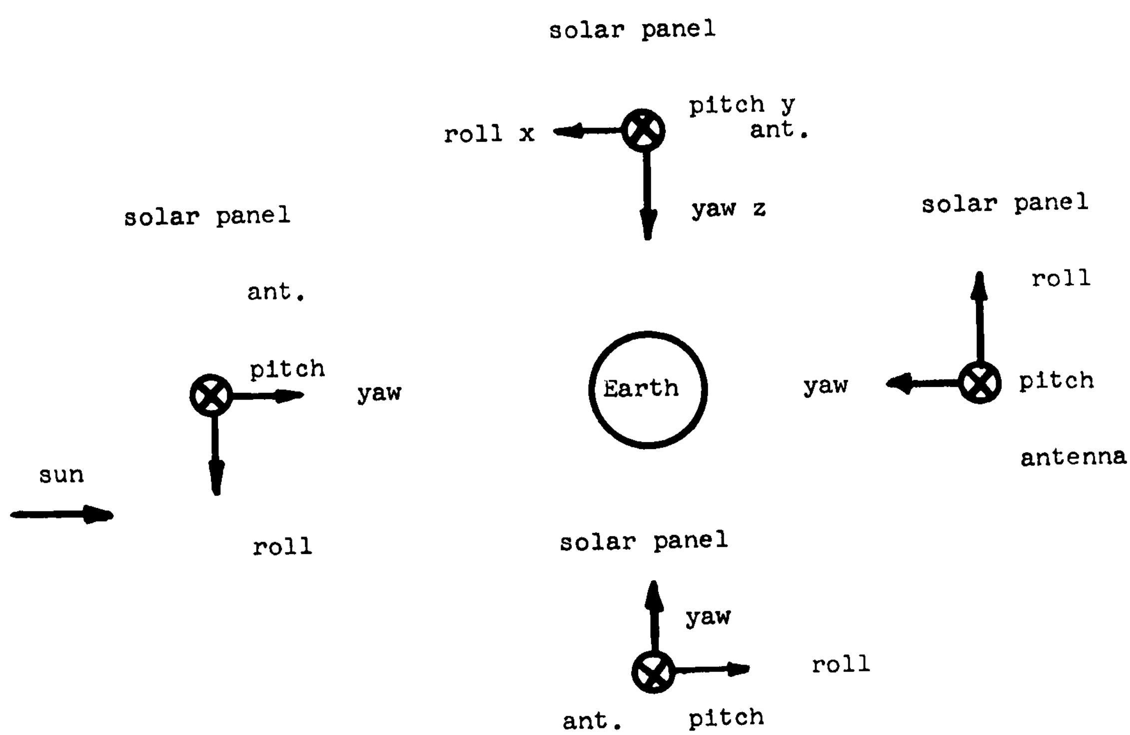


Fig. 66 DAILY MOTION OF SATELLITE AROUND EARTH.

Chapter 9

ORBIT ANALYSIS AND LAUNCH VEHICLES

9.1 Multiple-Launched Satellite Spacing

From the cost effectiveness standpoint, it appears obvious that payloads of more than one satellite per launch result in substantial savings. The following sections consider the problem of the spacing of a group of satellites injected with one booster. The problem was previously considered by Kaplan.[46]

9.1.1 Mechanics of Drifting

It is desired to place a satellite into circular orbit of radius r_c. The typical sequence is illustrated in Fig. 67.

(1) The vehicle is launched from point L, into a ballastic trajectory.

(2) When point P (the perigee) is reached, the vehicle is given another acceleration so as to coast along the transfer ellipse.

(3) Upon reaching apogee A with velocity V_a, the injection motor (also called the apogee-kick motor) is then fired to circularize the orbit at r_c with circular velocity V_c.

From elementary orbit theory,

$$V_c = (\mu/r_c)^{\frac{1}{2}} \tag{6}$$

$$V_a = \{\mu[(1/r_c) - (2/a)]\}^{\frac{1}{2}} = [2\mu\ r_p/r_c(r_c+r_p)]^{\frac{1}{2}} \quad , \tag{7}$$

where

μ = gravitational constant of the earth,

a = semimajor axis of the ellipse.

The velocity difference between V_c and V_a accounts for the difference between the circular and elliptic orbits. However, the periods of the transfer ellipse and circular orbit, T_t and T_c, respectively, are given by

$$T_t = 2\pi\left[(r_c+r_p)^3/8\mu\right]^{\frac{1}{2}} , \tag{8}$$

$$T_c = 2\pi\left[r_c^3/\mu\right]^{\frac{1}{2}} , \tag{9}$$

or

$$\frac{T_t}{T_c} = \left[(r_c+r_p)/2r_c\right]^{\frac{1}{2}} . \tag{10}$$

Equations (6) through (10) are valid regardless of the location of the ellipse, i.e., whether it is inside or outside the circle. In order to emphasize this fact, define as positive the perifocus direction and the quantities Δr and ΔV as

$$\Delta r = r_c - r_p \tag{11}$$

$$\Delta V = V_c - V_a . \tag{12}$$

Note that r is positive in Fig. 67 and negative in Fig. 68. Regrouping Eqs. (6) through (10), using (11) and (12),

$$\frac{T_t}{T_c} = [1 - (\Delta r/2r_c)]^{3/2} \tag{13}$$

and

$$\frac{\Delta V}{V_c} = 1 - \left[2 - \frac{1}{1-(\Delta r/2r_c)}\right]^{\frac{1}{2}} . \tag{14}$$

Equations (13) and (14) are fundamental to all spacing schemes in multiple launched satellites using discrete-thrust injection.

In order for the satellites to be spaced relative to each other in respect to the earth, one must be able to calculate their drift rates. Thus, for each orbit of a satellite in the transfer ellipse, the

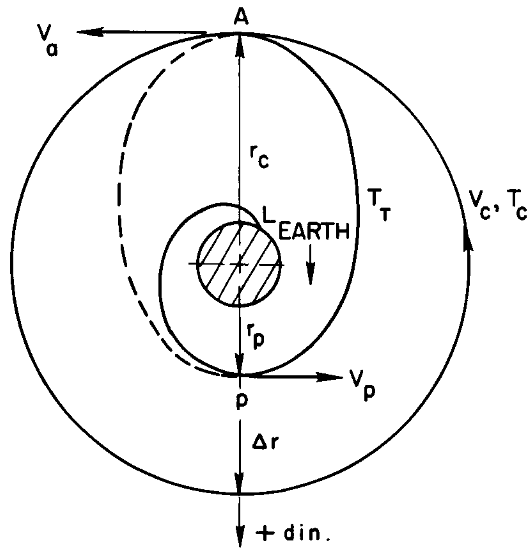


Fig. 67 HARMONIC INJECTION.

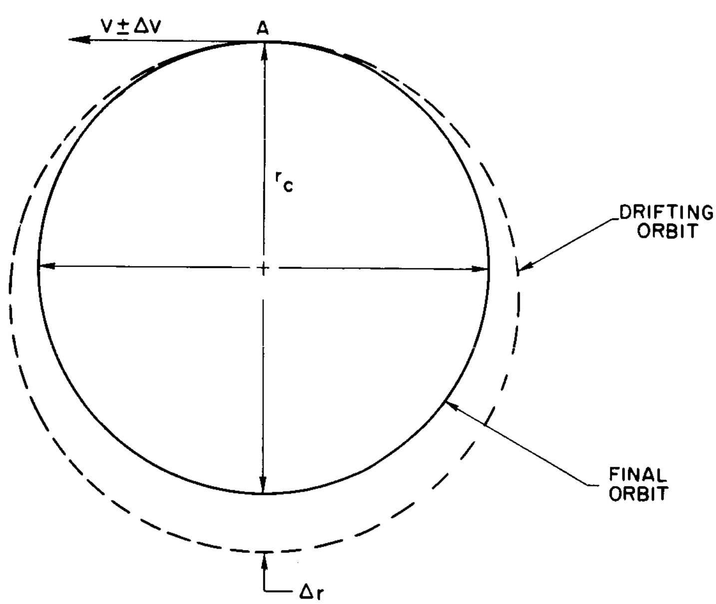


Fig. 68 DRIFTING INJECTION.

tangency point appears to have drifted eastward by the angle θ_E with respect to an inertial observer, i.e.,

$$\theta_E = (2\pi/T_c)T_t \ .$$

Hence, the satellite has a total rotation, $(\Delta\theta)_T$ westward in one elliptic orbit:

$$(\Delta\theta)_T = 2\pi - (2\pi/T_c)T_t = 2\pi\left\{1 - [1 - (\Delta r \ 2r_c)]^{3/2}\right\} . \qquad (15)$$

The <u>average</u> angular velocity of the transfer ellipse is

$$(\dot{\theta})^T_{avg} = (\Delta\theta)_T/T_t = (2\pi/T_c)\left\{[1 - (\Delta r/2V_c)]^{-3/2} - 1\right\} . \qquad (16)$$

The angular change per circular orbit is

$$(\Delta\theta)_c = (\theta)^T_{avg}/T_c = (2\pi/T_c)\left\{[1 - (\Delta r/2r_c)]^{-3/2} - 1\right\} . \qquad (17)$$

We would now like to relate ΔV to the angular change in one circular orbit, $(\Delta\theta)_c$. Equating (14) and (17),

$$\Delta V/V_c = 1 - \left\{2 - [1 + (\Delta\theta)_c/2\pi]^{2/3}\right\}^{1/2} . \qquad (18)$$

The total time for each spacing, n, in number of days, for a total angular drift change of Θ is related as

$$n = \Theta/(\Delta\theta)_c$$

or Eq. (18) becomes

$$\Delta V/V_c = 1 - \left\{2 - [\Theta/2\pi n + 1]^{2/3}\right\}^{1/2} . \qquad (19)$$

It is to be emphasized that all the equations up to this point are valid for <u>any</u> pair of circular and elliptic orbits. A Maclaurin series expansion of Eq. (19) yields:

$$\Delta V/V_c \simeq (1/2\pi) \left[\frac{1}{3} (\Theta/n) - \frac{1}{9} (\Theta/n)^2 + \ldots\right] . \qquad (20)$$

For the special case of <u>synchronous</u> orbit, $T_c = 1$ day and $\Theta_{max} = \pm\pi$, and

$$\Delta V/V_c = 1 - \left\{2 - [(\pm 1/2n) + 1]^{2/3}\right\}^{1/2} , \qquad (21)$$

where the plus and minus signs are for eastward and westward drift respectively.

From elementary rocket theory, it is well known that

$$\Delta m/m = (\Delta V/V_c) \; (V_c/I_{sp}g) \quad , \qquad (22)$$

whereas $\Delta m/m$ is the ratio of ejected mass (considered as small) to the total mass and I_{sp} is the specific impulse of the fuel. For <u>synchronous</u> orbit, the relationship between $\Delta m/m$ and V/V_c, assuming a specific impulse of 215 sec, is approximately

$$\Delta m/m = (\Delta V/V_c) \; (10{,}087/215 \times 32.2) \approx 1.5 \; (\Delta V/V_c) \quad . \qquad (23)$$

9.1.2 Techniques of Spacing

All techniques of multiple-launched satellite spacings are based fundamentally on the mechanics of drifting given in the previous section. Three types of discrete-thrust spacing techniques will be examined. They are

(1) harmonic injection,
(2) drifting injection,
(3) last stage discrete thrust injection (LSDTI)

Their relative merits will be compared.

(1) Harmonic Injection

In this scheme for spacing, the individual spacecraft is fitted with its own injection (or apogee-kick) motor. The cluster of satellites arrive at apogee (r_c) along its elliptical coasting trajectory. Precisely at apogee, the injection motor on satellite No. 1 is fired to achieve circular synchronous orbit. The rest of the satellite package continues to coast along its elliptical trajectory. When the package next crosses its apogee after M transfer orbits, No. 2 injection motor is fired to achieve circularization, while the remainder of the cluster continues along the ellipse. The process is continued until the last satellite is deployed in position. As a typical example, consider the problem of launching a spacecraft due east into a parking orbit at a height of about 100 nautical miles and then onto synchronous orbit. From Eq. (13),

$$T_t = T_c\left[1 - (\Delta r/2r_c)^{3/2}\right] = 24\left[3.82/6.611\right]^{3/2} = 10.52 \text{ hr.}$$

For one orbit of the transfer ellipse, the satellite appears to have drifted eastward by

$$\theta_E = (T_t/T_c)2 = (10.52/24) \times 360° \approx 158° .$$

If the perigee radius is not prescribed, then it is easy to show from Eq. (10),

$$(r_p/r_c) = 2(T_t/T_c)^{2/3} - 1 . \qquad (24)$$

The drift rate is fixed by the (T_t/T_c) ratio which fixes the (r_p/r_c) ratio. For an equally spaced network, the ratio T_t/T_c must be a rational fraction of the form

$$T_t/T_c = A/MN , \qquad (25)$$

where A is any whole number less than the denominator which has no common factors with the denominator, M is the number of transfer orbits completed between satellite injections, and N is the number of satellites. For a given spacing requirement, the parking orbit required by harmonic injection is higher than an optimum one. Generally, the higher the perigee, the more the booster payload capability will be reduced.

The total time, T_s, required to accomplish spacing is approximated by the formula

$$T_s \simeq T_t[M(N-1) + 1] , \tag{26}$$

where the time from launch to perigee is estimated (conservatively) as $T_t/2$. The booster attitude control system must remain active until all satellites have been ejected. Furthermore, the satellite must remain in an undeployed configuration until after the injection motors have been fired, which could entail some thermal control problems.

(2) Drifting Injection

Just prior to achieving apogee, the spacecraft is given a spin. The spin serves to give gyroscopic rigidity to the proposed injection direction furnished by the last stage guidance system. Shortly thereafter, one injection-kick motor simultaneously places the entire satellite package into final circular orbit. The package is next despun to a few r/min. The slight rotation is used to expedite the separation of the satellites from the injection-motor subassembly. Once freed from each other, each satellite is given either a forward or retro burn in the tangential direction by the satellite on board propulsion system. The satellites drift with respect to earth according to Eq. (19) or Fig. 69. When the satellite has drifted approximately the correct amount, a velocity correction in the opposite direction is made when the satellite next crosses its final orbit. Spacing is then considered as completed. The total deployment time, T_s, is the sum of the longest drift time plus the launch to ejection time, i.e.,

Table 32

INJECTION METHOD COMPARISON

Method	Advantages	Disadvantages
Harmonic injection	Quick spacing (within 15°) Little on-board propellant	Individual kick motors Extended booster life Thermal problems
Drifting injection	One kick motor needed Quick deployment of solar panels	High elapsed time for spacing High use of on-board fuel
LSDTI (last stage discrete thrust injection)	Quick, accurate spacing No kick motors Little use of on-board fuel	Needs development High booster fuel consumption Thermal problem

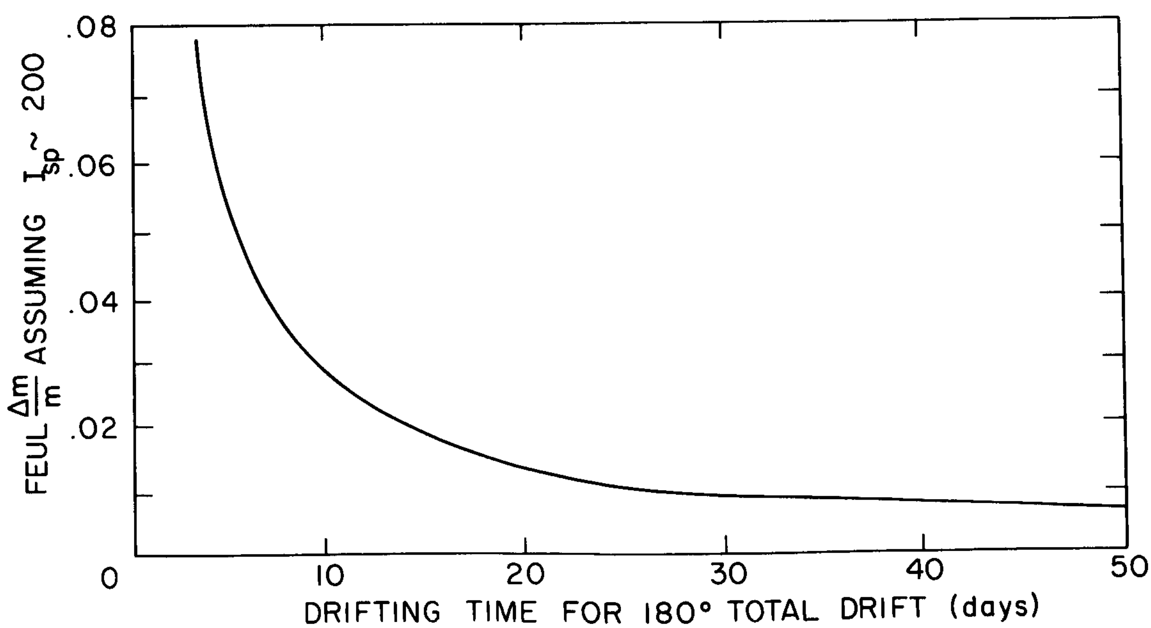


Fig. 69 DRIFTING INJECTION.

$$T_s = T_{max\ drift} + T_t/2 \ . \tag{27}$$

The longest time for maximum drift is found from Fig. 69. A fast spacing time involves a large initial expenditure of fuel.

(3) Last Stage Discrete Thrust Injection (LSDTI)

This technique is essentially the same as drifting injection except that all the velocity corrections are performed by the last stage (e.g., the Centaur). A typical sequence would be as follows:

(1) perigee burn;
(2) coast to apogee, thrust to circularize orbit;
(3) drop off satellite No. 1;
(4) thrust again to achieve desired drift rate;
(5) at appropriate point on the ellipse, application of a retro-burn to recircularize the orbit and complete desired spacing;
(6) repetition of procedure for No. 2;

Two burns are required to space each satellite. The LSDTI scheme is not available using the present technology, since restart of the Centaur vehicle must be accomplished 45 minutes after the previous burn. Furthermore, the number of restarts is restricted to three. (See comments under Booster Section.) The method is very wasteful of total fuel consumption, but has the advantages of quick spacing and complete elimination of the apogee-kick motor or motors.

9.1.3 Comparisons and Conclusions

The comparisons of the method of injection are summarized in Table 32. Harmonic injection is not recommended because of the need for individual injection motor for each satellite and the unequal spacing desired for SAINT. The LSDTI scheme is very desirable but not available at present. The drifiting method is recommended since spacing time and the amount of fuel carried is not critical in the proposed design.

9.2 Actual Spacing Scheme, Taking Injection Errors into Account

9.2.1 Launch

Since drifting velocity relative to earth is very small for reasonable amounts of fuel, it is desired to minimize the distance the satellite drifts. From the placement of the four international satellites, at 30°E, 30°W, 95°W, and 170°E, it is obvious that the drifting phase, which occurs shortly after the apogee kick motor is fired, should begin at 80°W, to give a maximum drift of 110°. See Fig. 70. If the satellites are launched due east from Cape Kennedy, longitude 80°W, into a 100 nautical mile parking orbit, and then the transition is made to transfer ellipse on the ascending mode, (i.e., on the second equatorial crossing), the satellite traverses a total angle of 450°E from launch to apogee while the earth rotates 95.5°E in that time, putting the apogee of the transfer ellipse at 85.5°W, an angle $\psi = 5.5°$ from where it is desired to have the apogee. This gives a maximum drift of 115.5°, roughly 5 percent further than necessary.

To remedy this situation, the satellites are to be launched $\alpha°$ north of due east, and is given by

$$\tan \alpha = \frac{1}{2} \sin 2\theta \tan\left\{\frac{\psi}{.94}\right\} , \tag{28}$$

where the factor .94 comes from the fact that the earth rotates while the satellite traverses the distance ψ relative to the earth. For small α and ψ,

$$\alpha = \frac{1}{2} \sin 2\theta \times \frac{\psi}{.94} = 2.4° ,$$

where θ is the launch latitude (i.e., 28.5°N at Cape Kennedy).

The angular inclination, i, of the orbit, is given by

$$\tan i = \frac{\tan \psi}{\cos (\psi/.94)} = \frac{\tan 28.5°}{\cos 5.8°} ; \tag{29}$$

note that for $\psi = 0$, $i = \theta$ and that for small ψ, $i \cong \theta$ as in this case.

For the four regional satellites where two are grouped at 10°E and two at 105°W, the apogee shall occur at 47.5°W, giving a maximum drift of 57.5°. However, to effect this, $\psi = 38°$ and $i = 35.5°$ (i.e.,

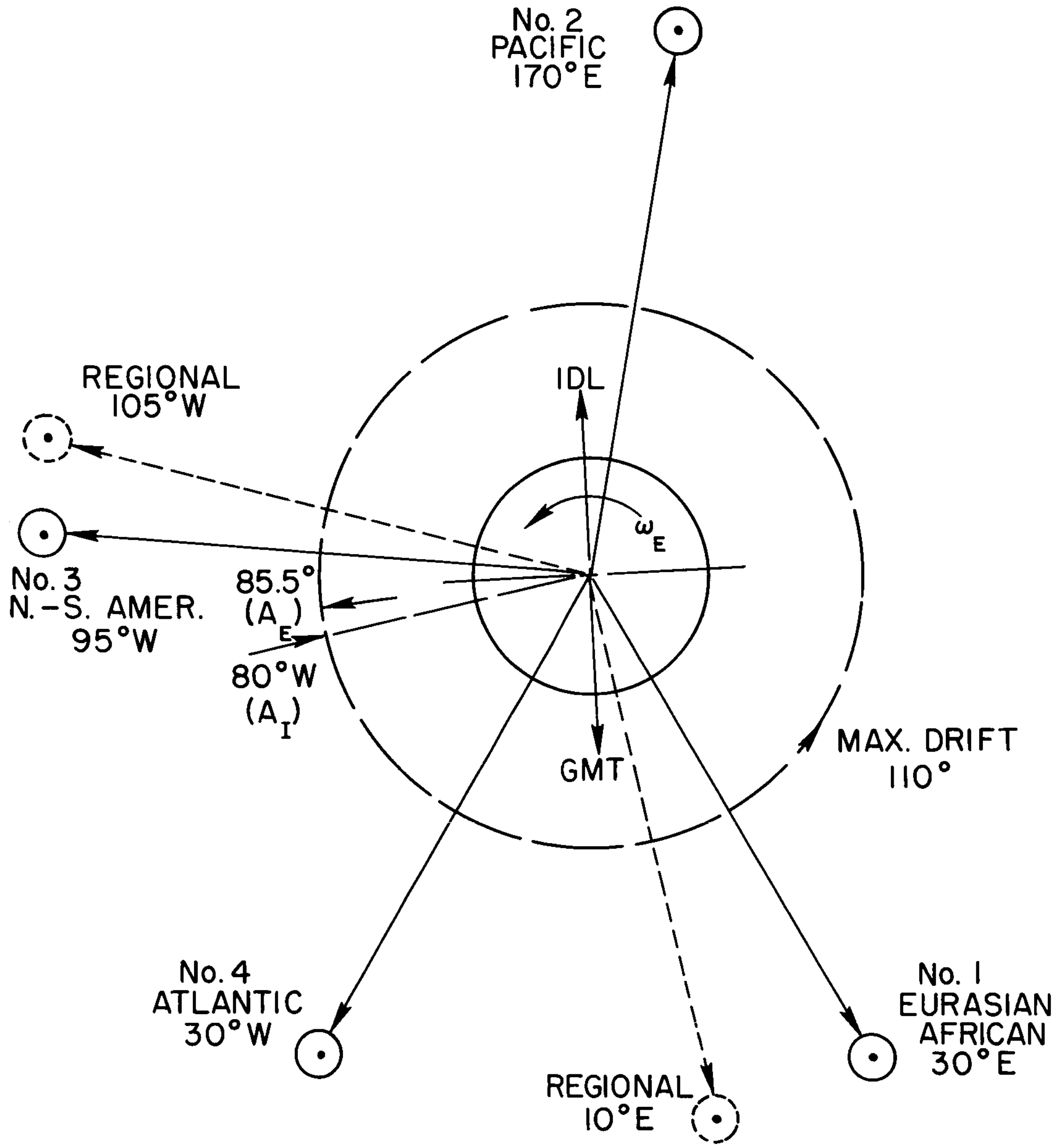


Fig. 70 REGIONAL AND INTERNATIONAL SATELLITE SPACING.

the inclination of the parking orbit) would have to be increased an additional 7° (Eq. 29) and since the maximum drift, even with a due east launch and transferring on the ascending mode, is only 95.5°, it is not worthwhile to shift the orbit the entire 38° (a shift of 15° can be made at a cost of only 1° increase in inclination or orbit).

9.2.2 Spacing

The sequence of events that takes place in spacing the satellites in synchronous orbit is as follows:

(1) The Centaur guidance system orients the bus carrying the four satellites for firing of the apogee kick motor.

(2) The entire satellite package is dropped off at apogee and spun to 60 r/min by means of two cold gas jets on the bus, to even the impulse from the apogee kick motor. This motor then fires ($\approx$ 1 min) to circularize the orbit and immediately thereafter, the bus is despun to one or two r/min by a yo-yo mechanism. If the Centaur booster can fire at synchronous altitude, then it will be used to circularize the orbit. Further, if the Centaur can be fired eight times over a period of two days, then it will be used to place the satellites in position.

(3) The satellites are released from the bus by explosive bolts and given a push by two very soft springs and they separate from the bus and from one another by "centrifugal effects."

(4) The solar cells are deployed, the spacecrafts are oriented, and the velocities are computed on the ground.

(5) Four hot gas jets are then used to eradicate injection errors and to give the satellites the necessary velocity increment to enable them to drift to position in reasonable time (three weeks maximum).

9.2.3 Description and Use of Hot Gas Jet System

The hot gas system is to be used for acquisition of position for each satellite. There are four 3-lb thrusters arranged to give both in-plane and out-of-plane thrusts (Fig. 71). These thrusters have vanes

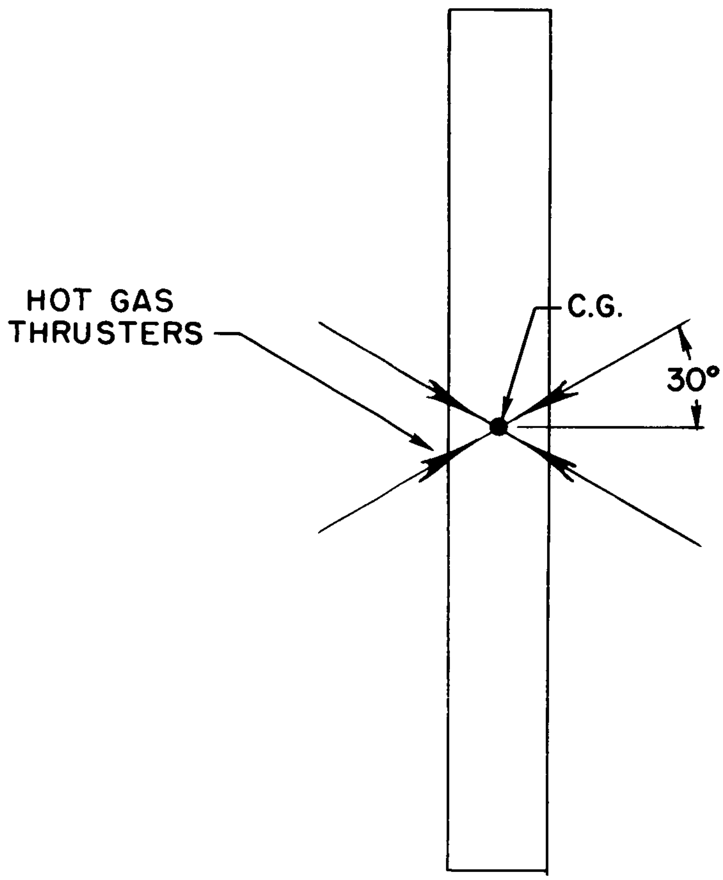


Fig. 71 HOST GAS THRUSTER SYSTEM IN CENTRAL CORE SECTION.

attached to deflect the exhaust gases so that the thrust is directly through the center of gravity. The vanes are controlled automatically by a feedback loop from the reaction wheels which sense when and if the thrusters start to change the attitude of the spacecraft.

In-plane thrusts can be made in an arbitrary direction due to the rotation of the satellite relative to the coordinate system defined by the radial and tangential directions. Out-of-plane thrusts to correct orbit inclination can be made with either the top or bottom sets of thrusters, (see Fig. 62), depending on whether the corrections are made on the ascending or descending node. Thus the satellite has complete capability for uncoupled translational motion even with one of the thrusters inoperative.

For the maximum drift of 110° with a drift time of 20 days, the spacecraft needs $\Delta V/V_c = .0051$ where V_c is the synchronous velocity. V positive (in the direction of motion) gives westward drift and V negative (retro) gives eastward drift. To obtain this ΔV for a spacecraft of 1450 lb, a tangential thrust of 5.2 lb from two 3-lb thrusters at a 30° angle, from the equatorial plane is required lasting for 15.7 minutes. The amount of fuel needed, assuming an $I_{sp} = 200$, is 27 lb for the two impulses required (to shart and stop drifting). The maximum acceleration which the satellite receives in its deployed state is about .004 g.

At apogee, there is a liquid fuel rocket firing to give the satellites the necessary V_c and to make the necessary plane change into the equatorial plane. For the sake of the discussion, the percent error in the ΔV will be assumed to be .1 percent, which is poorer than can be expected with liquid fuel rockets. Also, an error of ±1° in the alignment of the satellite package from the Centaur will be assumed.

Of the total 29° plane change required, the Centaur at perigee changes its position for the transfer ellipse by 8°, leaving a change of 21° at apogee. This gives $\Delta V = .002 \times 5540$ ft/sec at an angle of 80° from the equatorial plane as the worst error that can be expected. To correct this error, about 5 lb of fuel are required for each satellite. Thus, the total fuel requirement is about 32 lb. When the satellite is on station, the requirements of the hot gas system are completed and the ion engines take charge of station-keeping.

9.3 Station-Keeping Requirements

9.3.1 Orbit Perturbations

Once positioned in synchronous orbit, a certain amount of fuel must be expended over the satellite lifetime to keep the satellite in its designated position over the equator. There are three main perturbations that tend to move the satellite from this position:

(1) Gravity perturbations due to earth's gravity field,
(2) gravity perturbations due to the moon and sun, and
(3) solar radiation pressure.

These perturbations will now be discussed.

(1) Earth Perturbations

Perturbations from the earth's gravity field arise from the fact that the potential field of the earth is not a simple $1/r$ type field. Instead, the earth's potential is normally represented as an infinite series of Legendre Polynomids (spherical harmonics) and terms of $1/r_n$ (see any text for the solution to Laplace's equation in spherical coordinates). The spherical harmonics are generally broken into two categories: (a) the zonal harmonics which are the latitude dependent terms, and (b) the tesseral harmonics which are both longitude and latitude dependent. Due to the fact that the stationary orbit is directly over the earth equator, the zonal harmonics have only a very slight effect on the orbit. Generally speaking, the second (and other "even harmonics") will merely change the orbit radius slightly, while the third harmonic (and other "odd harmonics") will displace the orbit slightly parallel to the equator. These effects do not cause any station-keeping requirement.

The longitude dependent terms in the earth potential do cause a significant tendency for the satellite to drift in its orbit, and extensive work has been done in computing these effects. Generally speaking, the longitude terms tend to cause the satellite to oscillate along the equator about various "potential wells." Thrusting must be done to counteract the drift. For example, Early Bird required a ΔV of 0.36 ft/sec every 1½ months or about 3 ft/sec/yr to maintain its position to within 0.1°.[47] For different positions, different ΔVs would be required with the maximum about 6 ft/sec/yr.[47]

(2) Lunar-Solar Perturbations

The lunar and solar disturbing accelerations cause changes in virtually all of the orbit parameters . Their primary effect is to cause a precession of the orbit plane, thereby producing an inclination angle between the equator and the orbit plane.

The sun causes an inclination change of about 0.27°/yr while the moon's effect varies between 0.48°/yr and 0.68°/yr due to the 18.6-year period of precession of the lunar orbit plane. In 1973, the total effect will be about 0.90°/yr and this gradually reduces to about 0.85°/yr in 1978.[47] In addition to this primary effect, there will also exist a periodic drift term bounded by 0.1° as well as a small secular term of about 0.0046°/day.[48] These effects are negligible in computing fuel requirements.

(3) Solar Radiation Pressure

The radiation flux from the sun acts as an effective force tending to perturb the satellite from its orbit. The magnitude of this acceleration is given by

$$a = \left[\frac{A}{m}\right] P$$

where P is the radiation pressure constant (about 10^{-7} lb/ft^2 at the earth's radius) and A/m is the area-to-mass ratio of the satellite. For the chosen configuration, this is about 7 ft^2/slug. This tends to displace the satellite slightly along the orbit and to cause a small change in the orbit eccentricity. These effects are much smaller than for the first two items, however, and generally can be ignored in computing station-keeping fuel requirements.

9.3.2 Fuel Requirements

To maintain the satellite on station with ±0.1° for five years requires propulsive effort to counteract these disturbing forces. The 0.90°/yr inclination change is equivalent to a ΔV of 150 ft/sec/yr in the North-South direction, while to counteract East-West drift a conservative ΔV of 10 ft/sec will easily fulfill requirements.

The basic rocket equation relation ΔV to fuel used is

$$\Delta V = -I_{sp} q \ln \left[1 \frac{\Delta m}{M_I} \right]$$

where

I_{sp} = specific impulse

g = acceleration of gravity at surface of earth

Δm = fuel used

M_I = initial satellite mass.

If $\Delta m/M_I << 1$ the approximate equation holds

$$\frac{\Delta V}{I_{sp} q} \cong \frac{\Delta m}{M} \quad .$$

The thrust level and thrusting time can be varied independently, depending on the type of system chosen. Because of the desire for high specific impulse, the ion jet was chosen for a combined station-keeping and attitude control scheme. Since the ion jet must operate at very low thrust levels, the thrusting time must be relatively high. For example, for a 1500-lb spacecraft, a N-S thrust of about 225 μlb is required continuously to offset the lunar-solar perturbations. For the East-West control, 15 μlb continuously would counteract any drift tendency. The thrusting time would decrease proportionately with an increase in thrust level, so normally, a higher thrust level than the minimum values will be chosen. The thrusters are described in the attitude control section.

9.3.3 Control Scheme

The station-keeping system is designed to be operated by means of ground command. The on-board thrusters have a capability of 450 μlb thrust in three directions. This means that thrusting for the North-South control is required 12 hours per day, while the East-West control is required about 50 minutes per day.

For East-West control, it is recommended that this thrusting period be applied at 6:00 o'clock in the morning or evening to take advantage of the favorable thruster orientation (i.e., thrusters facing directly in the East-West direction). The North-South control would also be ground initiated and could be initiated at any convenient time.

Because all signals originate on the ground, the station-keeping scheme is necessarily very simple. There is also a great deal of flexibility in choosing thrust times and thrust levels. In the event of thruster failures, there is sufficient redundancy to still allow position correction. In the critical North-South direction, three of the six thrusters could be fired continuously to give complete position control. Refer to Fig. 62.

9.4 Launch Vehicle

9.4.1 Primary Requirements

(1) The launch vehicle must transport its payload without damage to a specified position in space and impart to it a predetermined velocity and attitude.

(2) The interface between payload and launch vehicle (payload here includes the upper stages of the launch vehicle and the satellite itself) must allow provision for the necessary electrical power, communications, environmental control, and support for the payload.

(3) The launch vehicle itself should conform to existing practice as far as launch procedure and ground equipment are concerned.

9.4.2 Primary Considerations

(1) Development costs should be kept to a minimum which implies:

(a) minimum modification of existing boosters,

(b) minimal development of payload adapters, and

(c) apogee kick motors obtained by only slight modifications of existing rockets, whenever possible.

(2) The factors influencing the availability of launch vehicles are the following.

(a) Considering the system as a whole, recourse may be made to either (1) a number of single satellite launches or (2) a smaller number of multiple launches or (3) a combination of (1) and (2).

In any case, the launch vehicle availability is dictated by weight and volume requirements. In designing the system, therefore, the various launch vehicle combinations already available must be examined as to their capability to place a payload into synchronous orbit either with or without an injection rocket. Also, the vehicle combinations currently being proposed must be considered whether or not they ultimately become a reality. Furthermore, possible development must be projected to include the proposed launch date, and beyond, since expansion and updating of the system is envisioned and replacement of individual satellites may be necessary.

(b) The launch position affects the selection of the launch vehicle. For launching into synchronous orbit, it is most likely that the facilities at Cape Kennedy would be used. It is possible that other launch sites might become available for a project such as the one now under consideration, but it is highly unlikely and it is assumed that the E.T.R. will be used for the launch.

9.4.3 Launch Vehicle Availability

A number of boosters capable of placing a synchronous satellite in orbit already exist. This number can be increased by anticipating present and future development of boosters and booster combinations. At present the launch vehicles available for the types of payloads envisioned are

(1) Atlas-Agena-Kick for payloads weighing up to about 1100 lb,
(2) Atlas-Centaur-Kick with suitable apogees motor for payloads up to about 2000 lb,
(3) Titan III C-Kick for payloads up to 2200 lb,
(4) Titan III-C-Centaur-Kick for payloads up to about 8500 lb.

9.4.4 Payload

A preliminary study established that a payload on the order of 4 $\times$ 2000 lb satellites would be desirable, single and multiple launch techniques and costs having been considered. Refer to Table 33. It was evident early in the study that the payload volume would probably be the dominant factor rather than payload weight. It was known at the start of the project that a number of composite launch vehicles were under consideration elsewhere. Also, it was anticipated that certain boosters might be uprated or replaced prior to the anticipated launch date of 1973. It was considered, therefore, that a first estimate of maximum payload (i.e., satellites in orbit) capability of 8000 lb was not unreasonable.

9.4.5 Probable Launch Vehicle Configurations

Table 33 lists those launch vehicle combinations most likely to be developed together with their payload capabilities (useful weight into a 24-hr synchronous orbit) and their estimated costs.

9.4.6 Present Restrictions

(1) The main restriction imposed on the launch and injection sequence is the present limitation of the Centaur. The time interval between burns may not presumably exceed one hour, but it is anticipated that, with the superior insulation materials now being developed, this interval might be extended

considerably. With this restriction, the Centaur main engines cannot be used for the injection into synchronous orbit from the apogee of the transfer ellipse; hence, the requirement of an injection rocket or apogee kick motor.

Should this restriction be removed, then it is suggested that injection into synchronous orbit and deployment of satellites might be accomplished using the Centaur booster. In addition to this, there is the possibility of developing a new booster of a type similar to the Centaur but capable of being mated directly with Saturn IB and also capable of direct injection into equatorial synchronous orbit. Such a vehicle would probably cost about \$10 million compared with \$6 million for the Centaur (total vehicle cost \$53 million), and the payload weight might be approximately 20,000 lb. This would fit into the range of boosters between the Titan III C-Centaur and the Saturn V.

(2) The maximum weight injected into equatorial synchronous orbit by the Atlas-Centaur-Injection combination would be about 2100 lb, (useful satellite weight would be approximately 1450 lb) assuming that development of the Atlas follows the anticipated pattern and neglecting development of the Centaur. Unfortunately there are no booster combinations available for multiple launches into synchronous orbit of 8000 lb other than the Titan III C-Centaur-Injection and Saturn IB-Centaur-Injection.

9.4.7 Costs

The system cost is dealt with elsewhere but the launch vehicle plus satellite cost plays an important role in deciding which booster system to adopt. Assuming a communications satellite cost of \$10,000/lb, the various booster costs are shown in Table 33 and compared further in Table 34 and Fig. 72. In the case of Saturn IB, the increased volume capability will allow the consideration of six satellites per launch. Table 34 does not make allowance for the escalation of costs which might be on the order of 10 percent per annum and it should be noted that the cost data are approximate but are based on information gathered from

Table 33

LAUNCH VEHICLES

Vehicle	Weight (lb)	Useful Volume (ft^3)	Cost ($M.)
Atlas-Centaur-Injection†	2100	1300 at 109"d	14.4*
Titan III C (120-7)-Centaur	9200	1300 at 109"d	23.6
Titan III C (120-7)-Centaur-Injection	12,000	2990	37.2
Saturn IB-Centaur-Injection	8500	1300 at 109"d	25.0*

*Includes an estimated cost for injection rocket

†For specification of injection stage see Appendix M

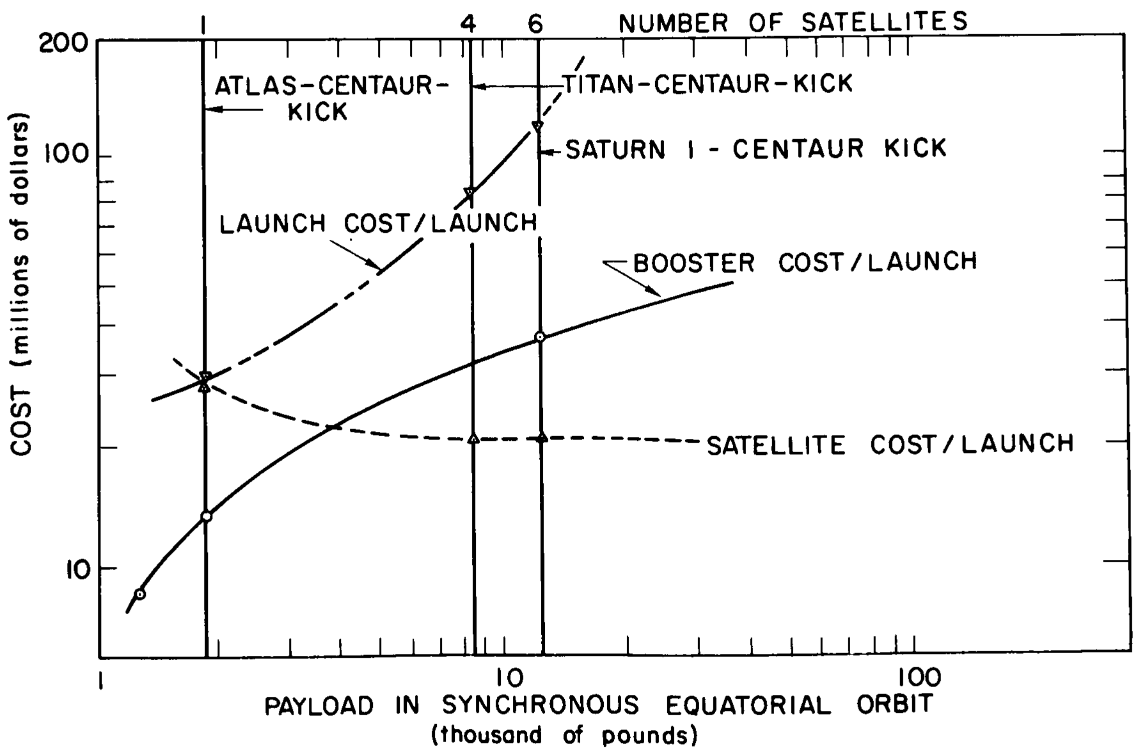


Fig. 72 PAYLOAD COST COMPARISON.

Table 34

BOOSTER COMPARISON

Launch Combination	Weight (lb)	Cost ($M)		Total Cost ($M)
Atlas		7.4		
Centaur		6.0		
Liquid injection rocket (estimated)		1.0		
Payload	1800			
Attitude control fuel (17%, see Appendix M)	350			
Useful payload (1 satellite)	1450	14.5		
				28.9
Titan III C		17.6		
Centaur		6.0		
Injection rocket		1.4		
Payload	7200			
Fuel (17%)	1400			
Useful Payload (4 satellites)	5800	58.0		
				83.0
			Cost/sat	20.75
Saturn IB		29.8*		
Centaur		6.0		
Injection rocket		1.4		
Payload	10,800			
Fuel (17%)	2100			
Useful payload (6 satellites)	8700	87.0		
				123.8
			Cost/sat	20.64

*A better estimate might be $40 M.

reliable sources. The large difference in the costs of Titan III C and Saturn IB, which are vehicles having similar performances, is accounted for by the fact that research and development costs of the Titan III C are borne elsewhere.

9.4.8 Development

Many studies are in progress aimed at improving the performance of boosters. The results of these studies are not yet available and, therefore, what follows is merely anticipation of extrapolation of the performances one might expect in the not too distant future.

(1) It appears that the Centaur is a useful "second-stage" booster and that the space communications program requires a vehicle capable of operation throughout the transfer period from parking orbit to synchronous equatorial orbit. The present limitation of the time interval between reburns of the Centaur main engine(s) appears to be due to propellant boil-off; i.e., it is a thermal insulation problem. This limitation has already been eased slightly, resulting in an increase from 20 minutes between burns to 60-65 minutes.* It is known that "superinsulating" materials are becoming available and it can be anticipated that the time between burns might be 12 hours or more by 1973. It is anticipated also that the control system will be fully operational during the whole 12-hour period. (In this case, the trans-stage of the Titan III C also might have an extended operating period, which would allow it to operate at synchronous altitude if required.)

(2) Improvement or replacement of the Atlas and Agena vehicles is anticipated and a new version of Titan III C has been announced. These improved versions might well be available before 1973. Thus, provided that the "matching" capability of the boosters is maintained, a new series of combinations will become available for launching payloads into equatorial synchronous orbit. Of these, the Atlas-Agena, the Atlas-Centaur, the Titan III

*Information supplied by Centaur Project Office.

C-Centaur, and the Titan III G are of particular interest. The payload capabilities (into synchronous equatorial orbit) will probably be on the order of 1500 lb, 2000 to 3500 lb, 10,000 lb, and 16,000 lb. The two values for the Atlas-Centaur imply that two versions of the Atlas might become available.

9.4.9 Titan III C-Centaur

This combination of boosters appears to form the vehicle most likely to be acceptable for use if the system under study is to become a reality. It will be capable of launching four 1450-lb** satellites into synchronous equatorial orbit with its present characteristics if provision is made for apogee injection and provided that the shroud can be extended by 15 ft. There appear to be no serious objections to this latter proposal.

If development of the system is taken into account, then the following predictions can be made:

(1) The insulation problem of the Centaur should be solved in the near future, enabling the reburn capability to be extended considerably, certainly for a period sufficiently long to enable Centaur to be used for direct synchronous injection and deployment of several satellites. This implies that guidance over the same period will be available and that the main engines can be used a number of times. (A three-burn capability is claimed to be feasible.)

(2) The thrust engines of the Centaur are at present capable of being uprated to levels in excess of 20,000 lb per engine. It is not unreasonable, therefore, to expect thrust levels in the region of 30,000 lb to be attainable in the future.

(3) Titan III C already exists in two versions by virtue of the fact that it may have either five or seven segment strap-on solid boosters. It is possible that further slight

** The present design of satellite has a weight of slightly less than 1300 lb. The value of 1450 lb was chosen (a) to allow for growth, (b) with "cold gas" station-keeping. 350 lb of station-keeping fuel are required, giving a total satellite weight of 1800 lb which is quoted as the maximum payload for the Atlas-Centaur vehicle.

modifications would enable Titan to inject a Centaur booster and its payload into a 90 or 100 nmi parking orbit without a Centaur burn. This facility would possibly allow some improvement to be made in the expansion ratio of the Centaur nozzles with a consequent increase in payload capability.

(4) It is not clear at present exactly what form the final Titan-Centaur will take, but the possibilities are that Titan III C (120-7) without the trans-stage will form the base vehicle. A two engined Centaur, each with thrust levels of 25,000-30,000 lb, carried in place of the trans-stage, should enable payloads weighing about 10,000 lb to be injected into a synchronous equatorial orbit.

A second variation would be to drop the second core stage of the Titan as well as the trans-stage. This would result in a three stage instead of a four stage configuration. At the present time, this is considered the more acceptable combination since a three stage system is simpler than a four stage system. This vehicle should have the capability of placing in synchronous equatorial orbit a payload of from 8000 to 10,000 lb, depending on the number of engines installed in the Centaur and the thrust levels attained.

9.4.10 Preferred Vehicles

For the single satellite launch, the Atlas-Centaur-Injection rocket system appears to be the best choice of vehicle from those available at present. With a solid propellant rocket for apogee injection, e.g., a slightly modified TE-M-364-3, the maximum weight of the payload injected into synchronous equatorial orbit could be approximately 1800 lb.* In this instance, the plane change in the orbit would be carried out by the Centaur. If the injection rocket capability is improved, it should be possible to attain the correct attitude at the apogee by yawing the Centaur in the transfer ellipse, the plane change and circularization of the orbit then being carried out by the injection rocket. The most economical method would

*Compared with the value of 1620 lb obtained in Appendix M.

be to optimize the plane change to be carried out by the Centaur at the perigee of the transfer ellipse.

For the multiple launches, a Titan III C-Centaur-Injection rocket system would be used. The specification of the injection rocket required is given in Appendix M. Figure 73 illustrates the assembled system. The shroud for use with both vehicles would be an extended version of the Surveyor shroud. An extension of 15 ft reduces the maximum payload capability by about 35 lb.

9.4.11 Launch Procedure

The launch procedure for the Titan III C-Centaur-Kick vehicle would probably follow the accepted pattern using a 90° launch azimuth (Fig. 74):

	Operation	Approximate Time (sec)
1.	Titan ignition state 0 - vertical lift-off	T
2.	Attain programmed trajectory	T + 5
3.	Jettison stage 0 - first stage ignition	T + 110
4.	Jettison shroud of 350,000 ft	T + 180
5.	Jettison stage 1 - Centaur ignition	T + 260
6.	Fly optimum trajectory to coast into 100 nmi parking orbit and first burn	T + 600
7.	Yaw Centaur to give optimum plane change during second burn	T + 1600
8.	Centaur second burn enter transfer ellipse perigee 100 nmi, apogee 19,323 nmi	T + 2100
9.	After Centaur second burn yaw to attitude required for apogee injection	
10.	Fire retrorockets on Centaur and separate payload	
11.	On approach to apogee spin up payload	
12.	Fire apogee motor	T + 20,500
13.	Despin payload to low r/min	T + 20,560
14.	Eject satellites	T + 20,570
15.	Initiate acquisition program	T + 20,600

This procedure is illustrated in Fig. 74.

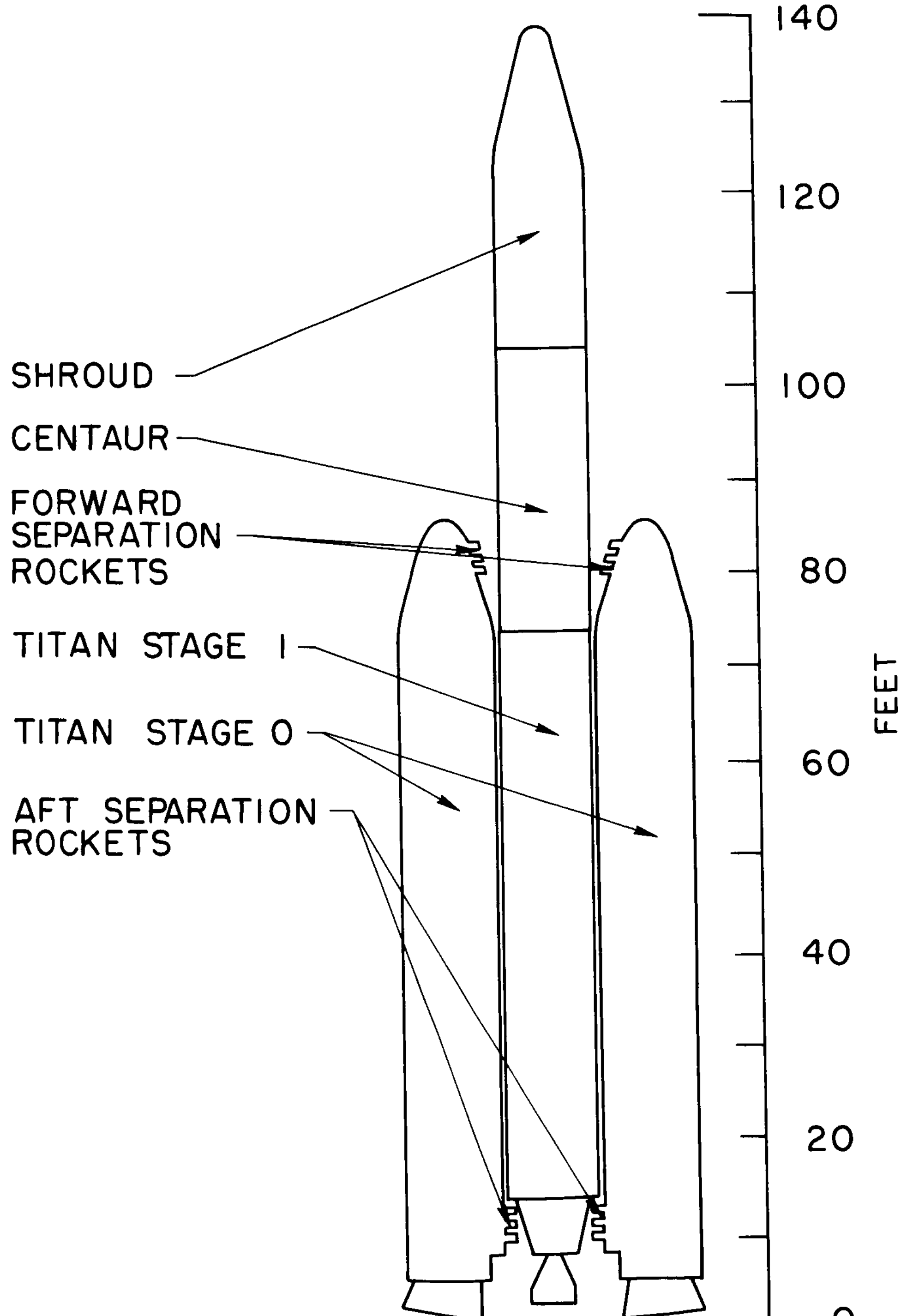


Fig. 73 TITAL III C CENTAUR INJECTION ROCKET ASSEMBLY.

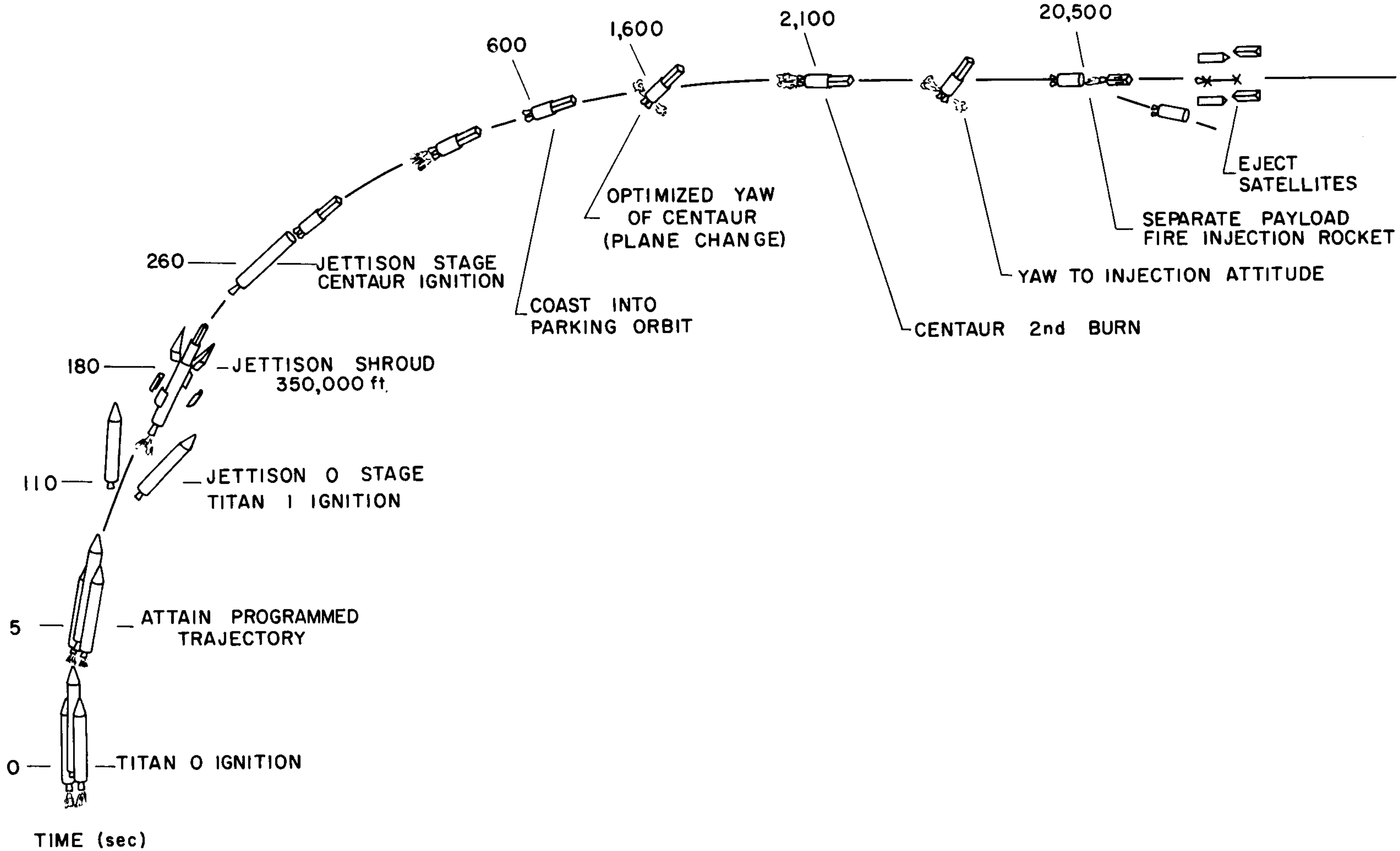


Fig. 74 LAUNCH AND INJECTION SEQUENCE.

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SECTION III

PRIMARY AND SECONDARY SATELLITE POWER SYSTEMS

INTRODUCTION

The satellite power supply provides energy for all electrical loads of the communication satellite for its orbital life. It also provides all the electrical energy required by the satellite during the launch and deployment phases of the system. The electrical energy is converted to suitable forms by the appropriate conversion equipment which is an integral part of the power supply.

The basic contraints on the power supply for the study presented here are

(1) a power requirement of 3 kW with continuous operation, including eclipses;

(2) a five-year orbital design life; and

(3) a 1973 launch with 1970 technology.

The eclipse occurs 92 times a year with a maximum length of 72 minutes.

From a parametric analysis of various power supplies, discussed in Section 10.1, a solar cell configuration is proposed for normal operation with a nickel-cadmium battery providing 100 percent power during eclipse. The main factors in the selection were weight, cost, durability, and state of the art of power system development.

The solar cell configuration consists of two-200 ft^2 expandable arrays constructed of N/P silicon solar cells mounted on an aluminum beryllium substrate. This design tends to maximize cell efficiency by minimizing the operating temperature of the cells.

No orientation of the solar arrays is made for the ±23°, 5° yearly declination of the sun from the plane of the equatorial orbit. The maximum solar energy incidence on the cells occurs during the season when the satellite experiences solar eclipse. At this time, the excess power supply capacity during the noneclipse portion of the orbit is used to help recharge the batteries.

Sections 10.1, 10.2, and 10.3 discuss in detail the subassemblies of the power supply.

Chapter 10

SATELLITE POWER SUPPLY

10.1 Main Power Supply Selection

The types of power supplies theoretically applicable to satellite systems include chemical units, batteries, fuel cells, solar powered units, isotopes, nuclear reactors, magnetohydrodynamic power generation, and other devices. Zwick and Zimmerman[49] have reported an excellent parametric study of a number of these systems for space applications. Furthermore, an updated state of the art for these power systems has been reported by Szego,[50-51] Shair et al,[53] Rappaport,[54] Elliot,[55] and Becker.[56] Table 35 is a list of the advantages and disadvantages of types of power supplies considered for the study.

Figure 75 is a plot of the specific weight vs the power level for the four most likely prospects for power supplies for communications satellites (nuclear reactor, solar cells, isotope and solar concentrator systems).

Considering Table 35 and Fig. 75, a solar cell battery combination is selected for the power supply for the study. The basic reason for this selection is the known technology and extensive experience with solar cells vs the development and state of the art of other systems. A detailed description of the solar cell array is given in Section 10.3; a detailed description of the battery is given in Section 10.2

As power requirements for satellites increase, power systems other than solar cells must be considered. The weight for extremely large arrays becomes excessive and the deployment problems are formidable. Nuclear reactors appear to have the most potential for space applications at high power levels, but 8-10 years of development are required for satisfactory service. The current design studies of reactors for space applications consider fast-fission reactors, and the systems may be very costly. Also, political problems due to potential radiation hazards can be formidable.

Table 35

SUMMARY OF ADVANTAGES AND DISADVANTAGES OF VARIOUS POWER SUPPLIES

Advantages	Disadvantages
Chemical Units (Open or Closed Cycle)	
1. Known technology	1. Refueling required
2. No orientation required	2. Excessive specific weight when fuel is included
3. Operates in eclipse	3. Vibrations, gyroscopic effects
Batteries	
1. Reliable for short term	1. Temperature degradation
2. Static	2. High specific weight
3. Operates in eclipse	3. Limited life
4. No orientation required	4. Development required
	5. 5-Year operation not probable for main power
Fuel Cells	
1. Static	1. Excessive specific weight when fuel included
2. No orientation required	2. Considerable development required: (not available for 5-year operation)
	3. Regenerative energy source required for adequate life
Solar Powered Units: (a) Photovoltaic Cells	
1. Known technology	1. Large areas and weights required for large power
2. Reliable	2. Degradation due to radiation
3. Static	3. Subject to environmental damage (meteorites), etc.
4. Flexibility in construction	4. No power during eclipse
5. Increase in power by increase in area	5. Launch problem due to packaging (vibration and g-loading)
6. Some improvement in efficiency probable	6. Deployment problems for large arrays
	7. Orientation required
	8. Long term development limited
	9. High manufacturing cost
	10. Availability problem
	11. Low efficiency
Solar Powered Units: (b) Solar Collectors (Mirror) Thermoelectric	
1. Static	1. No power during eclipse
2. Power increase probable	2. Launch problem due to packaging (vibration and g-loading)
	3. Deployment problems
	4. Orientation requires greater accuracy than photovoltaic cells
	5. Low efficiency
	6. Development required: (uneven temperature problems - not available by 1970)
Solar Powered Units: (c) Solar Collectors (Mirror) Dynamic System	
1. Relatively high efficiency	1. Launch problems due to packaging (vibration and g-loading)
2. Integral secondary power system possible (with energy storage other than battery)	2. Deployment problems
	3. Orientation required
	4. Possibility of no power during eclipse: (if no energy storage system provided)
	5. Dynamics of coolant may lead to satellite stability problems
	6. Reliability of dynamic system for 5 years
	7. Development required: (not available by 1970)
Isotope	
1. No orientation required	1. Launch complications cooling required, shielding may be required
2. Continuous operation through eclipse	2. Availability and cost
3. Small relative volume, high energy density	3. Development required 2 orders of magnitude of development required for 1970
4. High reliability	4. Possible radiation hazard
	5. Political considerations
Nuclear Reactors	
1. No orientation required	1. Further development required for 1970 state-of-the-art: a. Reactor for 5-year design life; b. Associated equipment
2. Continuous operation through eclipse	2. Launch complications remote control of reactor
3. Small relative volume, high energy density	3. Relatively heavy for small (1-6 kW) systems
4. Relatively long life power system with little degradation in energy output possible	4. Dynamics of systems could lead to satellite instability
	5. Heavy shielding required to protect sensitive instruments
	6. Problem of spent fuel in reactor possible radiation hazard
	7. National and international political implications

Magnetohydrodynamic Power Generation: 10 - 20 years of development required.

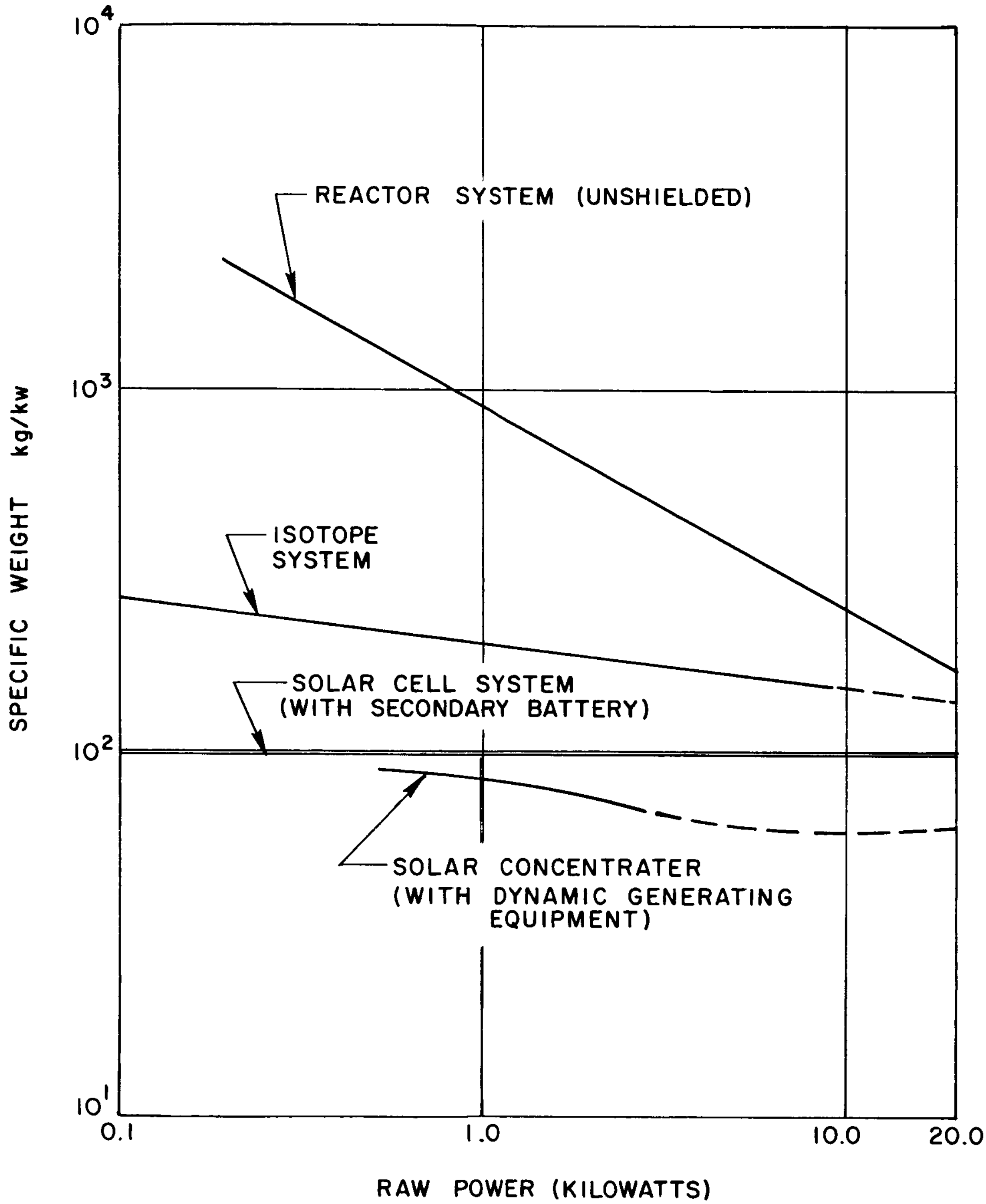

Fig. 75 SPECIFIC WEIGHT VS DC POWER FOR FOUR TYPES OF SATELLITE POWER SYSTEMS.

Isotope power systems do not appear to have much merit for large power applications because of cost and lack of availability of long-lived isotopes. This is shown in Table 36, where the cost and production of 5 isotopes are given.[50,56] Further discussion of nuclear systems is in Appendix N. A solar concentrator power system has great potential for space application, but state of the art does not allow a 1973 launch using such a system.

10.2 Secondary Power Supply

The secondary power supply is required for the periods when the satellite is in the earth's shadow, and for the launch and orbit acquisition phase. The requirements are given in Table 37. Full power is provided during eclipse. The largest single quantity of power needed during launch phase is for two ion engines, which are turned on to provide heat energy to maintain the battery temperature at the proper level.

Orbit insertion is planned for the first apogee. The secondary power supply has sufficient capacity to permit orbit insertion as late as the third apogee if this should be necessary.

10.2.1 Choice of Battery Type

The battery types considered as the secondary power supply are shown in Table 38, along with some of the factors which are important to this satellite design. The number of charge-discharge cycles the batteries must sustain is approximately 500. Of the four types considered, only the AgZn is questionable on its ability to withstand cycling.

Of the three remaining types, only NiCd currently has a shelf life adequate for the design life of the satellite. Because of the small number of cycles, the battery system can withstand a depth of discharge of 70 percent, giving a useful specific power of 12 Wh/lb.

The H_2O_2 rechargeable cell is in an early stage of development and considerable improvement can be expected. It can withstand large depth of discharge, on the order of 90 percent, for a large number (thousands) of such cycles. It is expected to be competitive for low orbit satellites which have periods of a few hours, and go through thousands of cycles. For the few cycles encountered in the synchronous orbit, where

Table 36

COST AND PRODUCTION OF FIVE ISOTOPES FOR SPACE APPLICATIONS

Isotope	Half-Life	Cost ($/W)	Availability
Pu-238	86 yr	500 - 900	500 thermal kW by 1980
Cm-244	18.4 yr	700	3-4 kg for current efforts
Po-210	138 days	30 - 190	Deferred pending mission requirement
Pm-147	2.67 yr	93 - 490	10 thermal kW/yr
Sr-90	28 yr	24 - 78	No present interest

Table 37

SECONDARY POWER SYSTEM DEMANDS

Demand	Power	Time
Eclipse (92 days per year)	3 kW	72 min (max)
Launch-to-apogee	120 W	6 hour
Orbit insertion	130 W	2 min
Deployment	320 W	2-5 min

Table 38

SECONDARY BATTERY COMPARISON

Factor	NiCd	AgCd	AgZn	H_2O_2
Shelf life, years @ 25°C	8	2	1	~1
Cycle life, cycles				
25°C; 65% d.o.d.*	2,500	1,800	50	>1000
25°C; 25% d.o.d.	10,000	12,000	300	>1000
Capacity, watts-hr/lb;				
predicted	17	30	80	15
current	15			10

*d.o.d. - depth of discharge

Table 39

SECONDARY POWER SUPPLY CHARACTERISTICS
OF NiCd BATTERY SUPPLY

Characteristics	Each Battery	System
Number of batteries	--	4
Number of cells	23	92
Voltage (nominal)	28.5	28.5
Depth of discharge, percent (maximum)	70	70
Capacity, amp-hour	45	180
watt-hour	1300	5200
Weight, pounds	75	300
Volume, $feet^3$	0.6	2.4
Internal resistance, ohms	0.025	---
Current, amps, normal discharge	26	105
charge	3	12
safe continuous over-charge at 77 °F	4.5	18
Charging efficiency, percent	40	40
Power dissipation in batteries, watts		
discharge	17.5	70
charge/overcharge	0.025/100	1/400
Temperature, °F normal	90	90
range	70-100	70-100

NiCd can be used with a large depth of discharge, the H_2O_2 cell has a small weight advantage. Countering this advantage is a lower charging efficiency in the H_2O_2 cell, and the uncertainty as to whether adequate life will be obtained in time for this project.

The AgCd cells would be restricted to a smaller depth of discharge to produce a reliability equal to that of a NiCd system. The weight saving in using AgCd would be on the order of 20-30 percent (60-90 lb saving on a 300 lb system). Again the life of the batteries is inadequate.

The NiCd cell meets all the requirements for this satellite, and has been proven in service. A modest improvement in the Wh/lb is assumed in this satellite design.

10.2.2 Battery Design

For redundancy, the secondary system has four batteries. The characteristics of this battery system are given in Table 39, where the nominal and limiting operating characteristics of the secondary power supply are listed.

10.2.3 Battery Charging

The batteries will be charged direct from the solar panel primary supply. A control unit will limit the charging current to 3 A (safe as a continuous overcharge current if the cell temperature is maintained above 50°F), and it will cut off the charging current if the battery temperature goes above 120°F. The charging current will also be cut off during the periods of the year when the satellite does not enter the earth's shadow. The slow charging rate (approximately 10 percent of the discharge rate) results in a low charging efficiency of about 40 percent.

The solar panels will have sufficient capacity to provide full satellite power at maximum inclination to the sun. However, the eclipse only occurs when the inclination is between 0 and 8°, and full eclipse occurs only at zero inclination. At zero inclination, the solar panels will have an excess generating capacity of 9 percent of the minimum generating capacity. This excess capacity will be more than sufficient to charge the batteries for more than four years. Since full charging capability is required at the end of the fifth year, approximately 100 W of additional solar cell generating capacity were added for this factor.

10.2.4 Bus Power

The space bus which will hold the multiple launch satellites in a single package requires a modest amount of power for timing, logic, spin-up, despin, and separation. This power demand will vary from 0-10 W over an interval of 2-5 minutes. The power will be taken from one of the satellites. The battery drain will be negligible.

10.3 Technical Details of Power Systems

10.3.1 Power Requirements

The electrical power subsystem of the satellite is designed to provide 3 kW at 28 V for a satellite life of five years. The power is supplied by a sun-oriented solar cell array and, while in the shade, by nickel cadmium batteries. The power requirements are shown in Table 40.

10.3.2 Silicon Solar Cells

Solar cells of single crystal silicon of N/P type have been used extensively in spacecraft. The advantages of solar cells are mainly their reliability, long and relatively stable cell life, and no need for fuel. Their disadvantages to date are high cost, radiation damage, large area required, and the need for a storage device. Recent advances in solar array design and fabrication make it practical to consider large lightweight deployable array for power levels of up to 50 kW or more.

The performance of a solar cell operating in space (air mass zero) is dependent upon the following parameters:

(1) illumination,

(2) temperature, and

(3) radiation degradation.

For an earth-orbiting satellite, the solar constant remains close to 1400 W/m^2. The principal variation in illumination is due to the divergence between the angle of incidence and the normal of the solar

Table 40

POWER REQUIREMENTS
(Watts)

Subsystem	Launch	Deployment	In Orbit
TWT			2300
Transponder			500
Radio beacon	10	10	
Deployment motors		20	
Telemetry and command	10	10	10
Attitude control			
6 ion thrusters		220	220
3 wheels		24	24
1 rate gyro		8	8
2 sun sensors			
4 earth sensors			
1 star tracker		28	28
1 logic unit			
2 antenna drives			
Power conditioning		40	40
Totals	20	360	3130

cell. For small angles, the power output is directly proportional to the cosine of the angle of incidence. The maximum expected angle of incidence is 23.5° due to the inclination of the earth's axis to the ecliptic. This will result in a maximum reduction of 8 percent in power. But the temperature of the cell is lower, and the actual reduction of power is therefore less than 8 percent.

Silicon solar cells are usually rated at 28°C and decrease in efficiency at the rate of about 0.45 percent per degree Centigrade rise in temperature above 28°C. Computation shows that the temperature of the solar panel is 58°C so that the degradation due to temperature will be about 13.5 percent.

Radiation degradation for synchronous satellite arises principally from solar flares. The effects of the Van Allen belt are rather small. A conservative estimation of 20 percent degradation in power in five years is obtained from existing data. This degradation is manifested by approximately the same percentage degradation in the open circuit voltage and the short circuit current as shown in Fig. 76.

New cells being developed aiming at reducing radiation damage are the "drift-field" solar cell and the "wrapped-around" cell. The dendritic cell grown in thin sheets aims at reducing the cost of fabrication, but the reduction in cost has not been large enough to stimulate future use.

The lightweight thin-film cell is made by depositing thin layers of a semiconductor such as cadmium sulfide onto a flexible metal or plastic substrate. The thin-film cell with 4-8 percent efficiency and poor stability will not reach a satisfactory stage of development by 1970.

A panel of solar cells with V-ridge reflectors is structurally quite good but has certain disadvantages, such as higher cell temperature and thus lower efficiency and degradation of reflector surface.

Heating and cooling the degraded cell to regenerate it to high efficiency has been proposed, but it does not seem practical in a spacecraft.

Modern silicon solar cells are usually 2 $\times$ 2 cm N/P type, 8 mm thick, with 3 mm cover glass. The principal parameters for such a cell are listed in Table 41.

Table 41

SOLAR CELL PERFORMANCE
(2 × 2 cm N on P Silicon Cell)

	Value		
	Initial	Initial	After 5 yrs
Inclination	0°	23.5°	23.5°
Parameter			
Open circuit voltage (volts)	0.50	0.52	0.45
Short circuit current (amps)	0.150	0.140	0.125
Maximum power (watts)	0.050	0.047	0.038
Voltage at maximum power (volts)	0.38	0.39	0.35
Current at maximum power (amps)	0.130	0.120	0.110
Temperature ($^{\circ}$C)	58	52	52

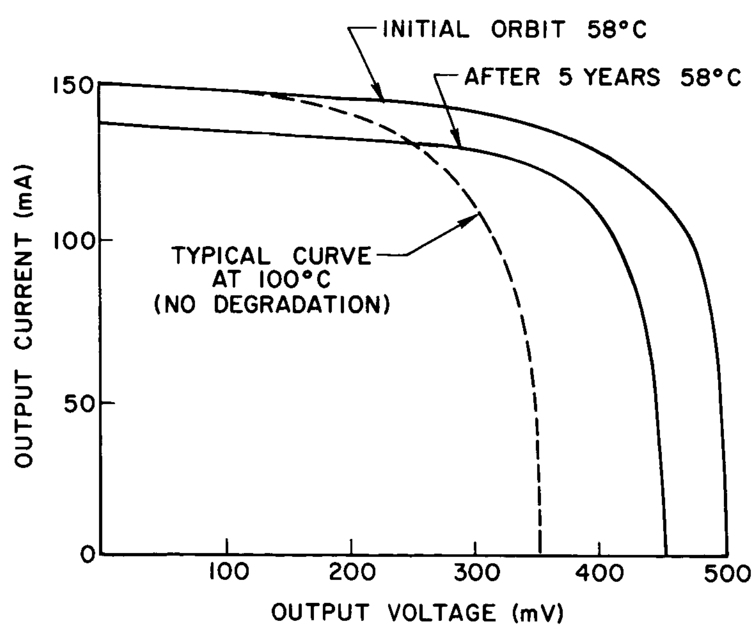

Fig. 76 TWO X TWO cm N/P SILICON SOLAR CELL CHARACTERISTICS BASED ON ILLUMINATION OF 140 WATTS/SQ METER.

10.3.3 Solar Cell Array

The required voltage is 28 V. To supply 28 V at the power bus, a 28.5-V battery is required. A voltage drop of 1.5 V occurs in the battery charger; approximately a 1-V droppage occurs in the diodes protecting the solar panels. Therefore, 31 V have to be supplied from the solar array. After five years of operation, the voltage drops about 10 percent, and an initial voltage of 34.5 V should be provided from the solar array.

NiCd battery	28.5 V
Voltage drop in battery charger	1.5 V
Voltage drop in two diodes	1.0 V

Reduction in voltage in five years (10 percent)

Initial voltage required from solar array $= \frac{31}{9} = 34.5$ V

The solar array consists of 8 panels, each of which contains 16 modules. Each module contains a string of 92 groups in series and each group contains 7 cells in parallel as shown in Fig. 77. A diode is connected between each module and the bus to protect the array in the event that a short circuit occurs in the modules. See Table 42.

Details of the mounting and interconnection of the solar cells are shown in Fig. 77. The modules are connected to the busses by copper strips. The busses are one inch wide and 0.031 inches thick copper, stacked behind the panel with insulation between the panel and bus and between each bus.

10.3.4 Power Conditioning

The approximate distribution of the power is shown in Table 40. A major portion of the power is required for the TWTs, and since these have their own power conditioning, the power supplied is raw power from the solar array. However, because of the relative rotation of the antenna with respect to the solar array, a rotary electrical coupling is required between the array and the antenna. The power required to charge the batteries is taken from the bus through a battery charger which performs regulation, switching, and protective functions for the battery. The remainder of the power is required for the ion engines and

Table 42

SOLAR ARRAY DETAILS

	Value			
	Initial		After 5 yrs	
Inclination	0°	23.5°	0°	23.5°
Parameter				
Open circuit voltage (volts)	46.0	48.0	40	41.5
Maximum power (watts)	4215	3870	3410	3132
Voltage at maximum power (volts)	34.5	35.5	30.5	31.5

Number of panels	8
Number of modules/panel	16
Number of groups in series	92
Number of cells in parallel/group	7
Total number of cells	82,432

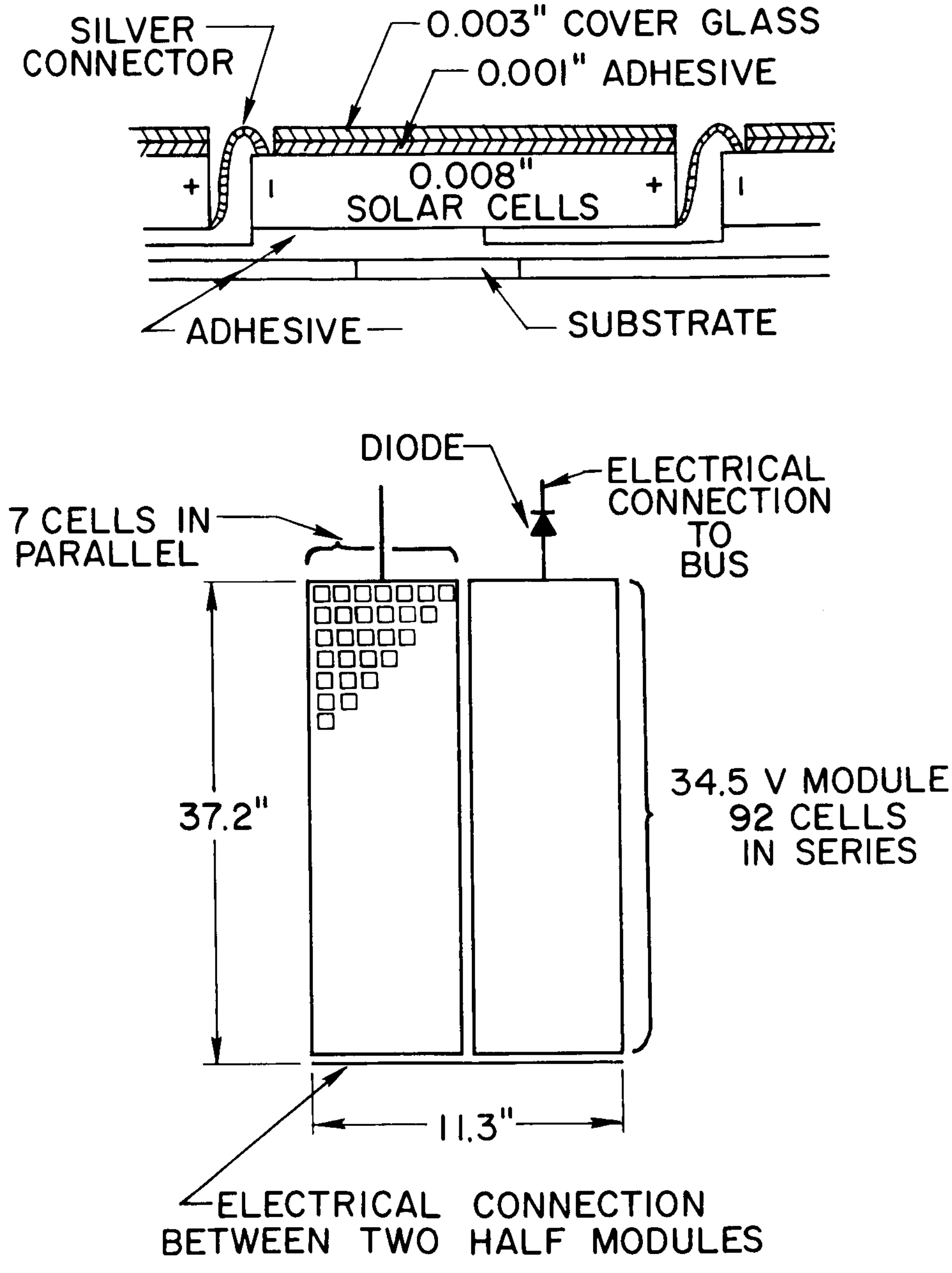

Fig. 77 SOLAR CELL ARRANGEMENT.

miscellaneous tasks, and is regulated to 28 V ± .5 V by a central pulse-width-modulated bucking regulator. Figure 78 is a schematic of the power distribution system. Details of possible power systems configurations are outlined in Ref. 57.

10.4 Power Transmission to a Rotating Antenna

The amount of power to be transferred from the sun-oriented solar array to the earth-oriented antenna package is 2700 W which, at a voltage of 31.5 V, requires a current of 85 A. The relative rotation is very slow, one revolution per day, and there are three possible methods of coupling:

(1) slip rings,
(2) wind-up cable and rapid despin, and
(3) rotary transformer.

10.4.1 Slip Rings

The first choice is to use silver slip rings with sintered silver molybdenum disulphide brushes. Recent tests[63] have shown that such rings can operate at low speed in a high vacuum (10^{-9} torr) for 650,000 revolutions at a current density of over 100 A/m^2 with very little wear. This test included periods of two hours' duration, during which the assembly was at rest but carried full current. Slip rings carrying currents of 2-4 A have performed successfully in space for several years of the OSO series of satellites.

Since the entire satellite operation is dependent upon the slip ring performance, a redundant system is used in which the current is shared between a total of 12 brushes in parallel on each bus. There are three separate slip rings on each bus with four brushes per ring. Depending upon the results of tests within the next few years, it may be deemed necessary to incorporate a release for each brush to be operated on command from the ground.

10.4.2 Wind-up and Despin

The power connection to the antenna may be made by cables which are allowed to wind-up during the orbit, and then the antenna

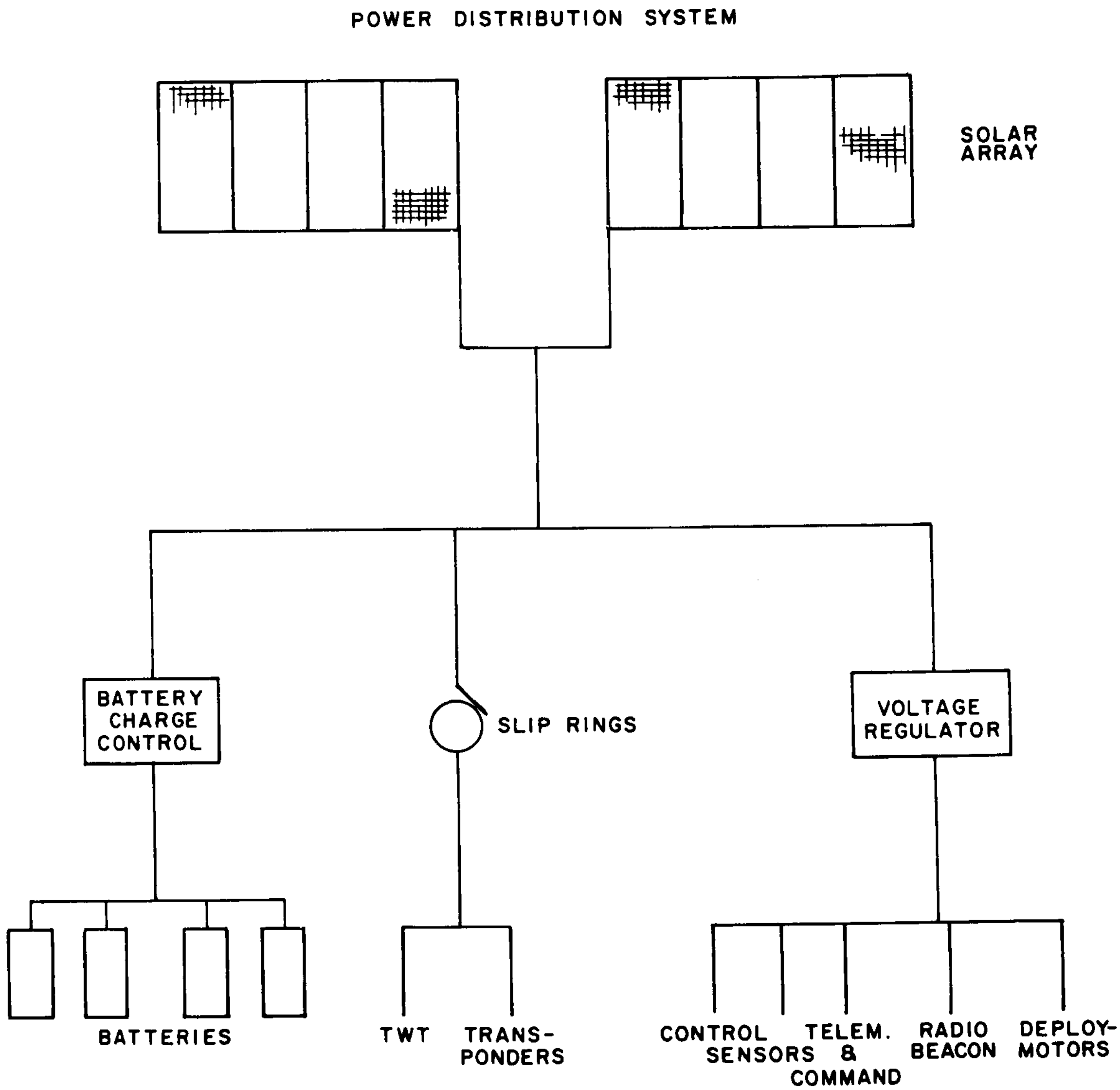

Fig. 78 POWER DISTRIBUTION SYSTEM.

assembly is rapidly rotated back through one revolution once every 24 hours. This provides a very efficient electrical coupling; however, during the unwinding period (perhaps one minute), transmission would be interrupted and reorientation would be required.

10.4.3 Rotary Transformer

The rotary transformer consists of stationary and rotating yokes separated by a small air gap. The primary is wound upon the stationary portion and the secondary on the rotating portion. The principal disadvantage is the necessity for an inverter to provide ac to the primary winding. Other disadvantages are the reluctance of the air gaps and the difficulty of laminating the rotating yoke.

Chapter 11

THERMAL ANALYSIS

The final design of a spacecraft for temperature control is a difficult problem, the solution of which requires extensive analysis and test simulation. This problem has been met and solved often enough, though, that considerable confidence can be placed in new designs based on existing techniques. With this background, it is possible to make a preliminary analysis of a new spacecraft design without too extensive or detailed calculations.

The three primary areas in terms of thermal control in the proposed satellites are, referring to Figs. 79 and 80: the solar cells (1), the battery and control systems package (2), and the communications package consisting of traveling wave tubes (3A), transponders (3B), and the preamplifier (4). The satellite geometry is such that there is little or no interaction between these packages. Details of these packages are now discussed.

11.1 Communications Thermal Package

11.1.1 TWT Thermal Package (3A)

The thermal energy to be dissipated by the traveling wave tubes (TWTs) and the associated power conditioning equipment represents the largest fraction of the total spacecraft energy budget. The radiated power from the satellite is to be accomplished, using TWTs with a maximum radiated power (ERP) of 80 W/tube. TWTs of this size are not yet in existence, but it is expected that they can be developed at a cost of $250,000. There appears to be no major technical problem except that of qualifying these new tubes for long, reliable operation. For design purposes, the 80-W ERP tube will be sized, based on a linear scale-up of the 56-W ($\times$ 1250) Eimac TWT.[58] This scale-up results in the specification of Table 43.

The information of Table 43 for the thermal energy to be rejected is the primary data requirement for the thermal design of the TWT package. It should be noted that the ERP requirement of 80 W at 35 percent efficiency and saturated operation sets the total power requirement of 230 W dc, but the off-saturation operation of 60 W ERP sets the maximum on the

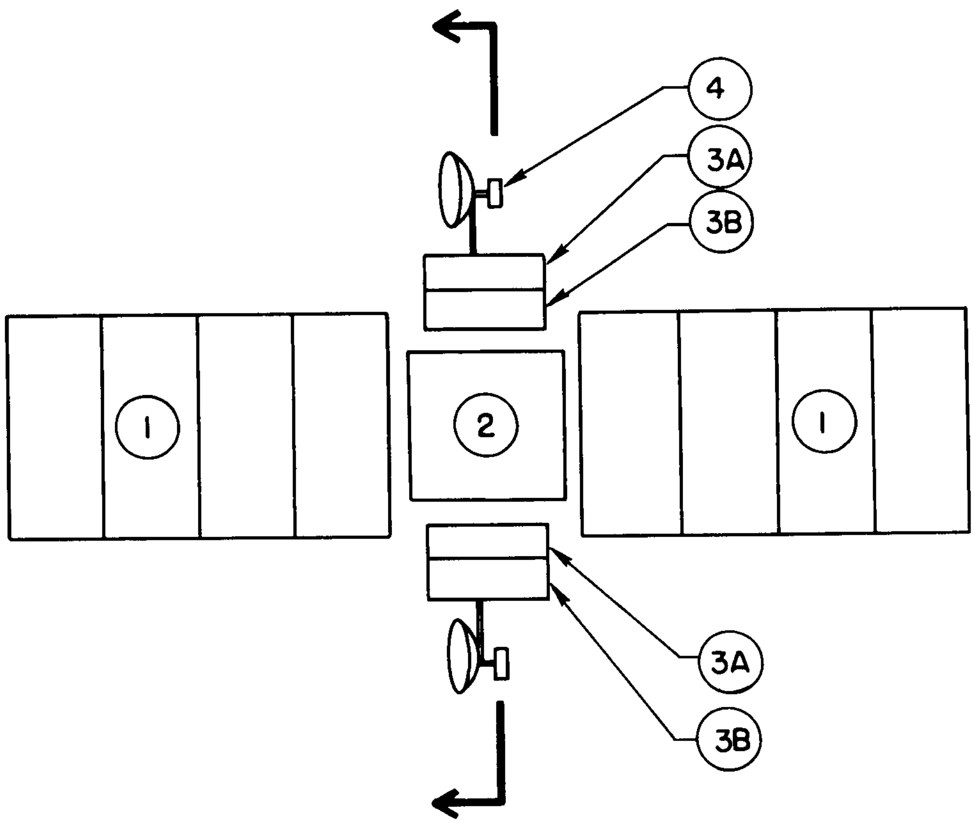

Fig. 79 COMPONENT BREAKDOWN FOR THERMAL ANALYSIS.

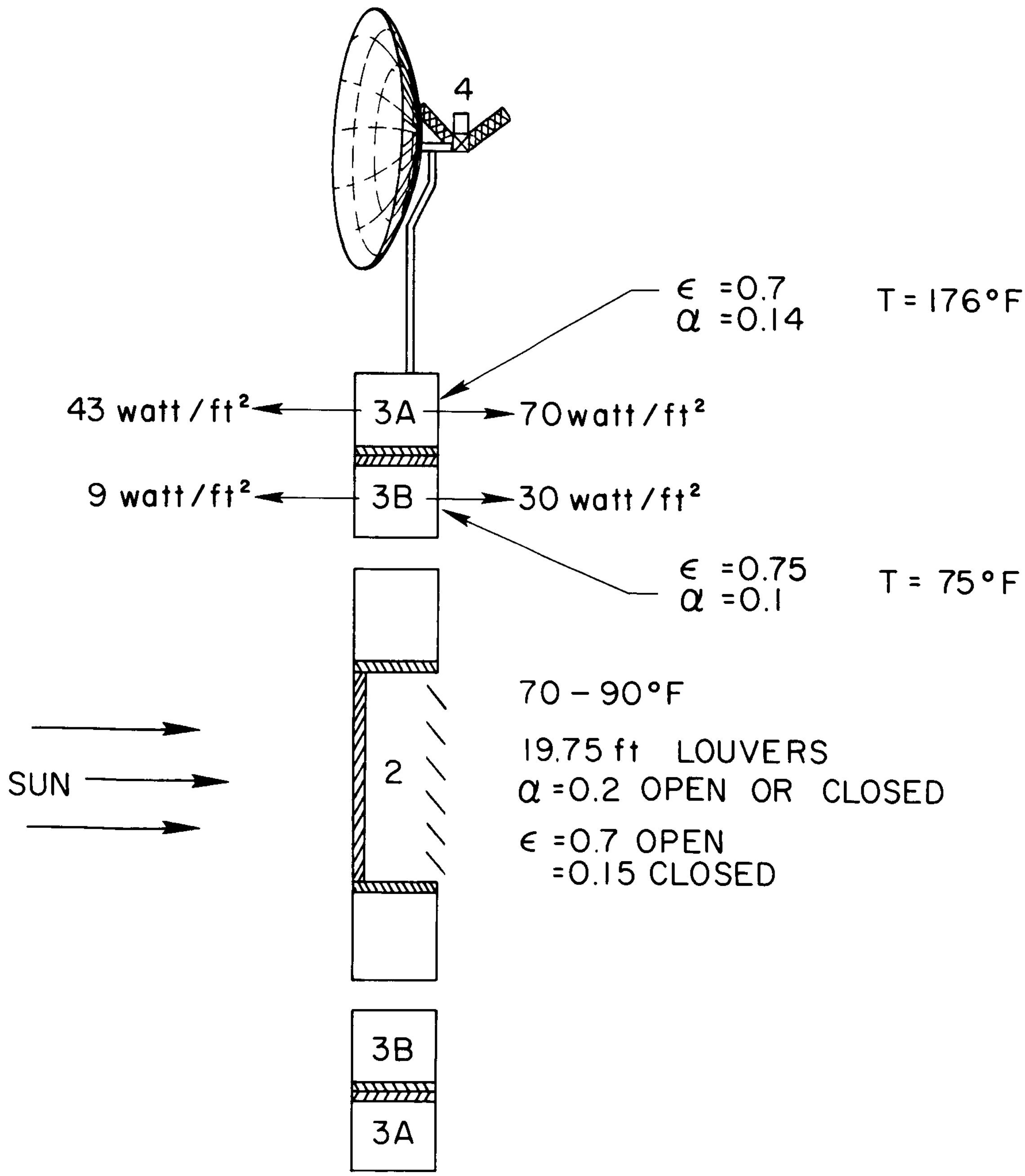

Fig. 80 HEATING AND LOSSES ON CENTRAL SATELLITE SECTIONS.

Table 43

THERMAL DESIGN OF THE TWT PACKAGE

Design Factor	Eimac × 1250	Proposed TWT Size	
		Saturated Operation	Off Saturation Operation
TWT beam power	150 W	203 W	203 W
TWT heater	4 W	6 W	6 W
Power supply losses (at 88 percent)	21 W	21.0 W	21.0 W
Total TWT power	175 W	230 W	230 W
Typical RF out	56	80	60
Typical efficiency	32.0%	35.0%	26.0%
Thermal energy to be rejected	119 W	150 W	170 W
Dimensions	12" × 5" × 3"	12" × 8" × 3"	12" × 8" × 3"
Weight	8 lb	12 lb	12 lb

thermal energy to be rejected. Characteristically, 90 percent of the energy to be rejected is at the collector end of the TWT. The energy transfer is accomplished by conduction to the TWT baseplate and then by radiation to space. The other energy losses are handled in a similar manner. There is a temperature limitation of about 250°C on the copper collector. Allowing for a 50°C temperature drop through the conduction path gives a temperature of 200°C maximum for the TWT base. Provided the TWT baseplate can be mounted on an area sufficient to radiate the necessary energy at or below this temperature, the thermal problem is solved.

The off-saturation operation mode requires 170 W of dissipated power. The area required for this thermal load depends on the surface temperature, the surface coating, and the energy input from the sun. The worst case is with the radiating surface facing the sun and low surface temperature. Assuming 75°C as the low temperature, and an α/ϵ of 0.2 with $\epsilon = 0.9$, gives a net heat radiating potential of approximately 55 W/ft^2 (see Appendix Q). The area required is then 170 W/55 W/ft^2 $\cong$ 3.0 ft^2. The baseplate area made available for the TWTs is the sides and top of a box 4 ft $\times$ 1 ft $\times$ 1 ft or 14 ft^2. With 5 TWT's, the area required is 15.0 ft^2. Thus, sufficient area is available to provide the necessary cooling. Optimization in the face, or less uncertainty, would allow a reduction in the size of the box sides.

As only one side of the box will be facing the sun, and the others will be facing cold space, it will be advantageous to provide conduction paths from the warm to cold sides to equalize the temperature. Since the cold side has a radiating capacity of 75 W/ft^2 for the same surface temperature, it will be possible to reduce the emissivity to about 0.7 while holding $\alpha/\epsilon = 0.2$. This gives a radiating potential on the sun side of 40 W/ft^2 and on the shade side 65 W/ft^2. Averaging these numbers over the radiating area gives a radiating potential of 810 W. The total required is 170 $\times$ 5 = 850 W. Thus, there is sufficient area to insure operation at a temperature around 75°C with an $\alpha/\epsilon = 0.2$ and $\epsilon = 0.7$.

11.1.2 Transponder Thermal Package (3B)

The portion of the communications package containing the transponders is required to dissipate 250 W. The surface area available is 14 ft^2 which gives the number of 250 $W/14\ ft^2 = 18\ W/ft^2$. Assuming 75°F as the upper limit on temperature and the surface facing the sun, this heat can be dissipated if the surface has a coating with an α/ϵ of 0.1 and $\epsilon = 0.9$ (see Appendix Q).

The surface facing away from the sun has excess radiating capacity (40 W/ft^2 at 75°F) and will cool down below 55°F. As this is undesirable, it will be necessary to provide conduction heat paths between the warm and cold sides to equalize the temperature and make the box sides as nearly isothermal as possible, and to reduce the emissivity below 0.9. With reduction of the emissivity to 0.5 with $\alpha = 0.1$, the side facing the sun would be able to radiate only 6 W/ft^2, and the side away from the sun would be able to radiate 20 W/ft^2. Since an average of 18 W/ft^2 is required, this shows that the 2 W/ft^2 of excess radiating capacity on the shade side will be sufficient to handle the reduced capacity of the hot side. For total shade operation, the radiating capacity is 20 W/ft^2, which is 2 W/ft^2 more than is required. The temperature drop due to this excess capacity will not be significant over the shade period of 1.2 hr.

Insulation of the TWT thermal package from the transponder will be necessary because of the close proximity and differences in temperature. The thermal interaction with the solar cells and the center compartment will be negligible.

11.1.3 Preamplifier (A)

It is desirable to locate the preamplifier near the antenna and operate it at as low a temperature as possible. The preamplifier is approximately a cube 4 inches on a side, with less than 0.5 W of dissipated power. By shielding with superinsulation, the solar heat input and heat input from the spacecraft components can be minimized. There will be two metal waveguide connections which will conduct heat to the lower temperature preamplifier. The operating temperature of the preamplifier will be that necessary to reject the heat conducted in along the waveguide, and that generated internally. Neglecting the solar input by virtue of the insulation, the energy balance is

$$q_{waveguide} + q_{gen.} = q_{rad.} = A\sigma\left(T_{equil.}\right)^4$$

or

$$T_{equil.} = \sqrt[4]{\frac{q_{waveguide} + q_{gen.}}{A\sigma}}$$

Neglecting for the moment the heat input from the waveguide, the minimum equilibrium temperature necessary to dissipate the 0.5 W of internally generated heat is approximately 170°K. Adding in the heat load coming in from the waveguide will increase this temperature.

It is difficult to estimate the magnitude of the conduction heat level because of the uncertainty in the size and geometry of the waveguide installation. If an upper limit of 300°K on the preamplifier temperature is assumed, 4.4 W can be dissipated. This allows a maximum of 4 W into the preamplifier by conduction. As this is not a large figure, it appears that extreme care will be necessary to provide thermal isolation even for a temperature of 300°K.

11.2 Battery and Control Systems Thermal Package (2)

This center compartment houses the attitude control equipment, the telemetry and command equipment, and the battery. It is desired to maintain these components between 21°C and 32°C under all conditions. These conditions include orbital operation (sun and shade) and ascent into orbit.

The approach used in thermally controlling the compartment is to limit the range of heat loads absorbed by the compartment and the heat loads generated within the compartment. If this range is sufficiently limited, a louver mechanism can be used to control the temperature. The sun heating load is kept small so that there is little load variation from sun to shade. All major components of the satellites: solar cell panels, antennae, electronics compartments, and center compartment are thermally isolated to reduce interaction uncertainties. Both of these requirements are accomplished by use of a total reflector coating ($\alpha = 0.1$, $\epsilon = 0.1$) on all sides of the compartment except the side that faces away from the sun (the louvered side). These sides are also insulated with 1.0 cm of super insulation [$K = 2.2 \times 10^{-6}$ W/(Cm - °K)].

During normal orbit operation, the heat generated within the compartment is distributed as follows:

	Sun		Shade
	Battery Charging	Battery Overcharging	Battery Discharging
Battery	1 W	400 W	70 W
Power Cond. Equipment	40	40	40
Thursters	60	60	60
Stabilizing wheels	0-24	0-24	0-24
Gyro	8	8	8
Telemetry and Command	10	10	10
Totals	119-143 W	518-542 W	118-212 W

During the ascent into orbit, the thermal load depends upon whether or not the sun impinges on the louvered side of the compartment. As a multiple launch is anticipated with immediate shroud ejection, one satellite may be exposed to continual sun load while others have no sun load or partial sun load.

The only heat that is generated within the compartment during ascent is 11 W from the telemetry and command and power conditioning. To maintain ascent loads with the range of orbit loads, a maximum sun load of 513 W may be absorbed by the louvers. To maintain a satisfactory minimum loading during ascent, a resistance heater, capable of providing 108 W (if necessary) is provided in all satellites.

Thus the heat load imposed upon the compartment is

Total heat to be dissipated:	119-542 W
Heat absorbed from sun:	0-513 W

These loads can be satisfactorily eliminated by a louver area with the following characteristics:

Temperature	70°F		
Area	19.75 ft^2		
Effective	= 0.2,	= 0.7	(opened)
Characteristics	= 0.2,	= 0.15	(closed)

11.3 Temperature of the Solar Array (1)

The available solar energy is 130 W/ft^2, of which about 8 to 10.5 W/ft^2 are actually converted to electrical output. Therefore, the remainder must either be reflected or radiated as heat. When an antireflecting coating is used, the reflected energy is about 5 percent of the incident energy, which leaves about 112 W/ft^2 to be radiated.

The solar cells, cover glasses, and adhesive have a total thickness of 15 mils. The panel itself is a combination of beryllium and aluminum, both of which are good thermal conductors, and calculations show the panel to be practically isothermal (within 5°C).

Using an emissivity of 0.84 for the solar cells, and 0.95 for the rear of the panel, the cell temperature is 58°C when the panel is normal to the sun's rays, and drops to 52°C when the inclination is 23.5°.

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APPENDICES

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Appendix A

A PROPOSAL FOR A RADIO AMATEUR SCIENTIFIC/COMMUNICATIONS SATELLITE PACKAGE

It is proposed that space and power be allocated for a radio amateur scientific/communications package aboard at least one of the communications satellites proposed for geostationary orbit in 1973, preferably one over the American continent. The international standby satellite might be most appropriate. See Chapter 1. It is anticipated that said equipment, exclusive of antennae, can be contained in a total volume less than one cubic foot and can be broken into smaller pieces. The total weight is anticipated to be less than 50 lb and the power drain less than 50 W. All operation will be on a noninterference basis to the communications satellite.

By allocating this space, the sponsors may expect to obtain:

(1) a large quantity of scientific data that would be impractically expensive to collect otherwise;

(2) the interest and training of a large number of radio amateurs in satellite communications, measurements, and ionospheric sciences; and

(3) a contribution to the ionospheric sciences by making a collection of measurements never before combined in a geostationary satellite.

The Communications Package

It is proposed that a cross-band translator similar to the ill-fated Oscar IV[21] be used. The input frequencies would be a 50 kHz band centered on 145.95 MHz. The output would be 434.95 MHz ±25 kHz. As many as 100 radiotelegraph (CW) communications or 25 single-side-band (SSB) voice communications could be handled simultaneously, assuming optimum channel loading.

Since the satellite would be about five earth radii distant from all stations, the dynamic range requirements are not severe, and 20 dB should cover all possible serious users. Limited AGC will be employed to prevent cross-modulation and distortion. The translator transmitter would be designed for about 10 W peak input power. A directive transmitting antenna (a yagi) will provide about 13 dB of gain and will easily fit in the proposed satellite. The total improvement over Oscar IV should be 20-30 dB, opening communications possibilities to many more radio amateur stations.

The Scientific Package

It is proposed that four harmonically related stable frequencies be transmitted in the 144, 432, 1296, and 2304 MHz amateur bands from the earth-oriented packages. Also, two frequencies in the amateur 14 and 50 MHz bands would be transmitted from the sun-oriented package.

It is proposed that the refractive index of the surrounding medium be measured by an impedance probe at frequencies between 100 Hz and 120 kHz. The measurement would use a sweeping oscillator to obtain data on the variation of the refractive index with frequency, which would be compared with the Appleton-Hartree equations to obtain plasma and gyrofrequency variations with time of day and solar activity.

Stationary satellites offer unparalleled opportunities for continuous observation of various ionospheric and exospheric parameters. The ATS-1 satellite presently over mid-Pacific has added a significant amount of knowledge on the behavior of the upper ionosphere and exosphere.[22] The equipment necessary to make these measurements was simple and inexpensive, and involved just measuring the polarization angle of the received signal as a function of time to obtain the integrated ionospheric electron content. One experiment that would have added even more information was shut off at the last minute, however, in order not to cause interference to a radio astronomy observatory. That experiment was a third harmonic transmission.

While space communications and telemetry bands may not be harmonically related, many of the amateur bands are. The following frequencies in amateur VHF/UHF bands are harmonically related: 144, 432, 1296, 2304 MHz. These are easily obtained from a standard oscillator, which for this experiment would be offset about 1 part in 105 to keep the transmissions inside the 144 MHz band. The conversion oscillators in the translator would be derived from the same frequency chain. The frequency separation between the 144 MHz beacon and an input signal would be equal to the frequency separation between the 432 MHz beacon and the translated output frequency.

Measurements of the received polarization will indicate the integrated electron content of the ionosphere. Any ambiguity in the total polarization shift can be resolved with measurements on additional frequencies. Measurements of the differential doppler between harmonically related carriers will indicate the integrated electron content of both the ionosphere and exosphere. Again, ambiguities can be resolved by use of additional frequencies. By subtracting the two measurements of electron content, the integrated electron content of the exosphere can be estimated. If the electron content in the vicinity of the satellite is measured, a much better approximation to the actual variation of electron density with height can be obtained. Harmonically related beacons aboard a geostationary satellite are just the solution recommended by Da Rosa[23] to the problem of measuring the temporal variation of transport of ionization between ionosphere and exosphere.

Whistler studies[24] indicated that the electron density in the vicinity of the satellite may be 10-100 per cc. The corresponding maximum plasma frequency is about 90 kHz. The gyrofrequency is about 2 kHz. These parameters are found to vary with solar activity. Theory indicates large changes in the refractive index at frequencies near the plasma frequency. Some variation may be expected near the gyrofrequency as well.

It is planned to use a sweeping beat frequency oscillator to generate a 0.1-120 kHz sweep in approximately 120 seconds. Ten kHz markers will allow frequency calibration. The sweeping oscillator will be applied to a capacitative voltage divider, one element of which is an impedance probe. The analog outputs of amplitude and phase across the probe will be converted to variable frequency audio tones for telemetry on the 50 MHz beacon.

Antennae

The following appendages are contemplated:

A. On the earth-oriented package

(1) Two 19-inch tape whips for transmission and reception

in the 144 MHz band;

(2) One yagi antenna for 432 MHz, elements 12 inches long, boom length 4 foot or as long as possible;

(3) One two-foot parabolic dish with two cross-polarized feeds, one for 1296 MHz, and one for 2304 MHz.

B. On the sun-oriented package

(1) One 55-inch tape whip for transmission in the 50-MHz band;

(2) One 55-inch tape whip for use as an impedance probe;

(3) A shunt feed for the solar cell assembly to resonate at 14 MHz.

Appendix B

DERIVATION OF QUEUEING SYSTEM STATE EQUATION

We may write the state equation as follows:

$$P_o(t + \Delta t) = P_o(t)\ (1 - \lambda\Delta t) + \mu\Delta t P_1(t)$$

$$P_1(t + \Delta t) = P_1(t)\ [1 - (\lambda + \mu)\Delta t] + \lambda\Delta t P_o(t) + 2\mu\Delta t P_2(t)$$

$$\vdots$$

$$P_n(t + \Delta t) = P_n(t)\ [1 - (\lambda + n\mu)\Delta t] + \lambda\Delta t P_{n-1}(t) + (n + 1)\mu P_{n+1}(t)$$

$$\vdots$$

$$P_k(t + \Delta t) = P_k(t)\ [1 - (\lambda + k\mu)\Delta t] + \lambda P_{k-1}(t)$$

After rearranging, these become

$$\frac{dP_o(t)}{dt} = \lim_{\Delta t \to 0} \frac{P_o(t+\Delta t) - P_o(t)}{\Delta t} = \mu P_1(t) - \lambda P_o(t)$$

$$\frac{dP_1(t)}{dt} = \lim_{\Delta t \to 0} \frac{P_1(t+\Delta t) - P_o(t)}{\Delta t} = \lambda P_o(t) - (\lambda + \mu)P_1(t) + 2\mu P_2(t)$$

$$\vdots$$

$$\frac{dP_n(t)}{dt} = \lim_{\Delta t \to 0} \frac{P_n(t+\Delta t) - P_n(t)}{\Delta t} = \lambda P_{n-1}(t) - (\lambda + n\mu)P_n(t) + (n + 1)\mu P_{n+1}(t)$$

$$\vdots$$

$$\frac{dP_k(t)}{dt} = \lim_{\Delta t \to 0} \frac{P_k(t+\Delta t) - P_n(t)}{\Delta t} = \lambda P_{k-1}(t) - k\mu P_k(t)$$

Since steady state conditions are assumed

$$\frac{dP_i}{dt} = 0 \quad \text{for} \quad 0 \leq i \leq K$$

and $P_i(t)$ may be denoted by P_i.

Solving these equations by repeated substitution, we have

$$P_n = \frac{\left(\frac{\lambda}{\mu}\right)^{\eta} \frac{1}{n!}}{\sum_{i=0}^{k} \left(\frac{\lambda}{\mu}\right)^{i} \frac{1}{i!}}$$

and the probability of losing a call is the probability of being in state K and obtaining a new arrival before a termination, or

$$P_L = P_n \cdot \frac{\lambda}{\lambda + k\mu}$$

$$P_L = \frac{\frac{1}{(k/Q + 1)} \cdot \frac{Q^k}{k!}}{\sum_{i=0}^{k} \frac{Q^i}{i!}}$$

where $\lambda/\mu = Q$.

Appendix C

CALCULATION OF ERROR PROBABILITIES IN THE DETECTION OF A BUSY CHANNEL

For a given channel allocation in a stated multiple access scheme, it is necessary to evaluate the interlocking aspect of the group of ground stations sharing a common frequency spectrum. This interlocking and consequently sharing technique must provide a sufficient confidence level to the individual ground stations for correctly detecting a free channel and then seizing it.

In order to avoid a centralized control station for allocation of channels on demand, each ground station may search the appropriate channel space and seize any free channel. This implies the necessity of detecting a sinewave-plus-noise signal. The carrier is transmitted at a reduced level when the channel has been seized whether there is voice activity at that moment or not. The detection of a sinewave in noise is well documented. Refer to Fig. C-1.

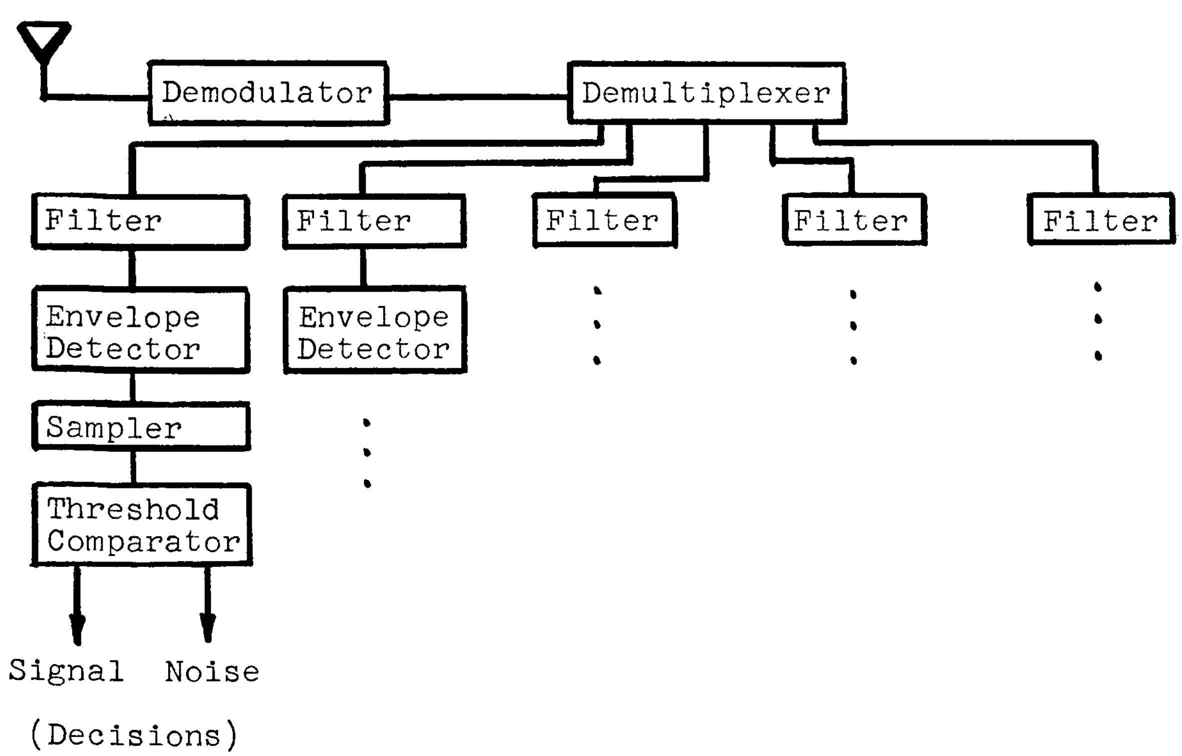


Fig. C-1.

The probability density of the envelope (r) of a signal wave in noise is

$$p(r) = \frac{r}{N} \exp\left[- \frac{(r^2 + As^2)}{2N}\right] I_0\left(\frac{rAs}{N}\right) \qquad 0 \leq r$$

where N is the noise power

$I_0(x)$ is the modified Bessel function of the first kind

A_s is the sinewave amplitude

For the special case where noise alone is present

$$p(r) = \left(\frac{r}{N}\right) \exp(-r^2/2N$$

which can be easily identified as the Rayleigh density function.

Given a threshold (T), the probability of observing a value of $r > T$ so that the hypothesis-signal-present is inferred, when actually the null hypothesis--noise only--is in fact correct, is given by

$$P_{FA} = e^{-T^2/2N}$$

For large signal-to-noise ratios, i.e., $A_s^2/2 \gg N$

$$I_0\left(\frac{rAs}{N}\right) \rightarrow \frac{e^{rAs/N}}{\sqrt{2\pi Asr/N}}$$

and therefore

$$p(r) \rightarrow \left(\frac{r}{N}\right) \exp\left[- \frac{(r^2 + As^2)}{2N}\right] \frac{e^{rAs/N}}{\sqrt{2\pi As/N}}$$

or

$$p(r) = \sqrt{\frac{r}{2\pi AsN}} \exp\left[-\frac{(r - As)^2}{2N}\right]$$

and if rAs/N is large $r \approx As$ and then

$$p(r) = \frac{1}{\sqrt{2\pi N}} \exp\left[-\frac{(r - As)^2}{2N}\right]$$

This can easily be identified as the Gaussian density function.

Using the substitution

$$y = \frac{(r - As)}{\sqrt{N}}$$

we have

$$p(y) = \frac{1}{\sqrt{2\pi}} e^{-y^2/2}$$

and the probability of a false rest, i.e., $y < T$ when the signal is present, is given by $(T - As)/\sqrt{N}$

$$P_{FR} = \frac{1}{\sqrt{2\pi}} \int_{-\infty}^{\{(T - As)/\sqrt{N}\}} e^{-y^2/2} dy$$

the signal-to-noise ratio is given by

$$\left(\frac{S_s}{N}\right)_{out} = 3(\gamma\beta)^2 \left(\frac{B_{if}}{B}\right)\left(\frac{S_c}{N}\right)_{in}$$

where S_c/N is the FM carrier-to-noise ratio

S_s/N is the subcarrier-to-noise ratio

γ = ratio of deviation for the subcarrier to the max deviation

B_{if} = the IF bandwidth

B = sinewave detector bandwidth

β = modulation index for the FM signal.

If $\gamma = .1$

$S_c/N = 20$ dB

$B_{if} = 14 \times 4$ kc $= 56$ kc

$B = 200$ cps

$\beta = 6$

$$S_s/N_{out} = 3[(.1)(6)]^2\ [56000/200]\ (100)$$

$$S_s/N_{out} = 3(36)(280) = 30{,}200$$

$$S_s/N_{out} \approx 45 \text{ dB}$$

For $S_s/N_{out} = 45$ dB, the probability of either type of error is negligibly small, i.e., $<10^{-8}$.

Appendix D

CALCULATION OF THE PROBABILITY OF THE SAME CHANNEL BEING SEIZED BY TWO DIFFERENT CALLS

If an unoccupied channel is located by the technique of channel searching and detection of a sinewave in noise, then the possibility of the same channel's being seized by another call is not zero. There is a propagation delay time of approximately .25 seconds single hop or .5 seconds double hop. During this propagation delay time to the satellite and back, another call may try to seize the same channel as the previous one, since it cannot detect any carrier to indicate channel occupancy.

The probability of such an occurrence is

$$P_{ss} = \left(\frac{1 - e^{-\lambda t}}{k}\right) \sum_{n=0}^{k-1} (n+1)\, P_n$$

where P_n is the state probabilities given in Appendix B and T is the delay time. These probabilities are shown in Fig. D-1, where

$$Q = \frac{\lambda}{\mu} .$$

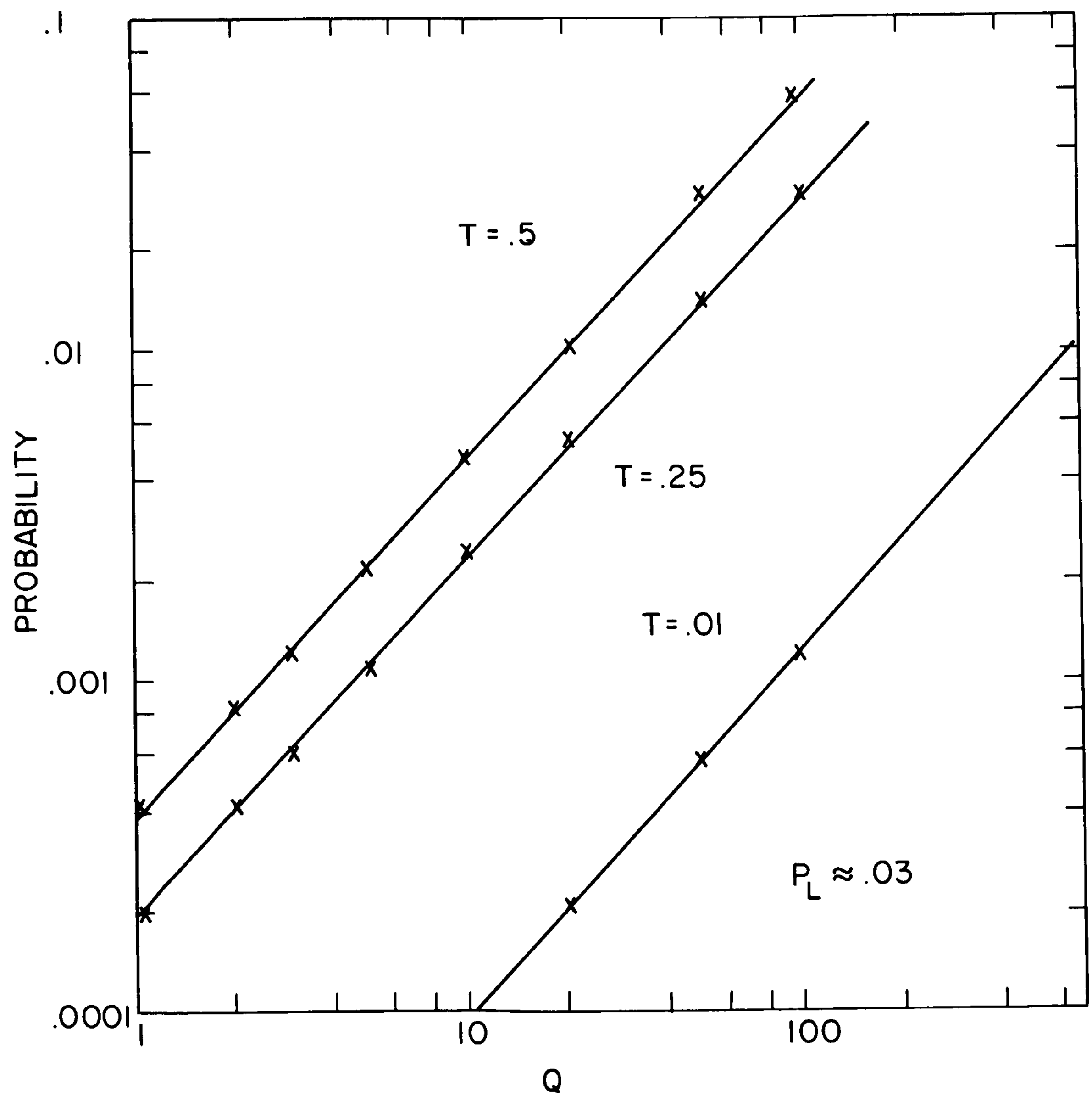


Fig. D-1 Q VS PROBABILITY OF TWO OR MORE CALLS SIMULTANEOUSLY SEIZING THE SAME CHANNEL AT PEAK HOURS.

Appendix E

OPTIMUM SIZE OF GROUP FOR SHARED CHANNELS

Assume that the point-to-point matrix of arrival rates for calls is homogeneous such that

$$q_{ij} = q_o \quad \text{for} \quad i \neq j$$

$$q_{ij} = 0 \quad \text{for} \quad i = j$$

A cost functional can be expressed which relates the cost of scanning channels for access and the cost of the equipment for access to the number of channels required. This should result in a group size factor which could be optimized for minimum cost. The tradeoff occurs when one observes that by increasing n (the group size), the cost of the necessary scanning equipment is greater because of the increased number of shared channels. However, if n is reduced, the searching factor cost is lower, but there are more total channels required, since for each group of shared channels, the load factor (Q) is lower.

Such a cost functional is as follows:

$$\frac{C}{N} = C_s\left[K(Q_1) + K(Q_2)\right] + \left(\frac{C_k}{n}\right)\left[\left(\frac{N-n}{2n}\right)\left(K(Q_1)\right) + K(Q_2)\right]$$

where

$$Q_1 = n^2 \, qo$$

$$Q_1 = \frac{n(n-1)}{2} \, qo \ .$$

Q_1 and Q_2 results from grouping the homogeneous matrix with element value qo into groups of size n, and where

C_s is the cost of scanning one channel

C_k is the cost of providing one channel

N is the total number of ground stations

C/n is the cost per ground station

Figures E-1 through E-3 show c/n versus n for c_s/c_k = .01, .1, and 1.0 for qo - 1, 2, 5. For C_s/C_k = .1, a minimum appears at $n = 1$ or 2.

Obviously, the restriction of a homogeneous load matrix may be difficult to achieve in practice. However, this type of analysis can be useful as a comparison with actual group cost for determining how low a minimum cost can be achieved.

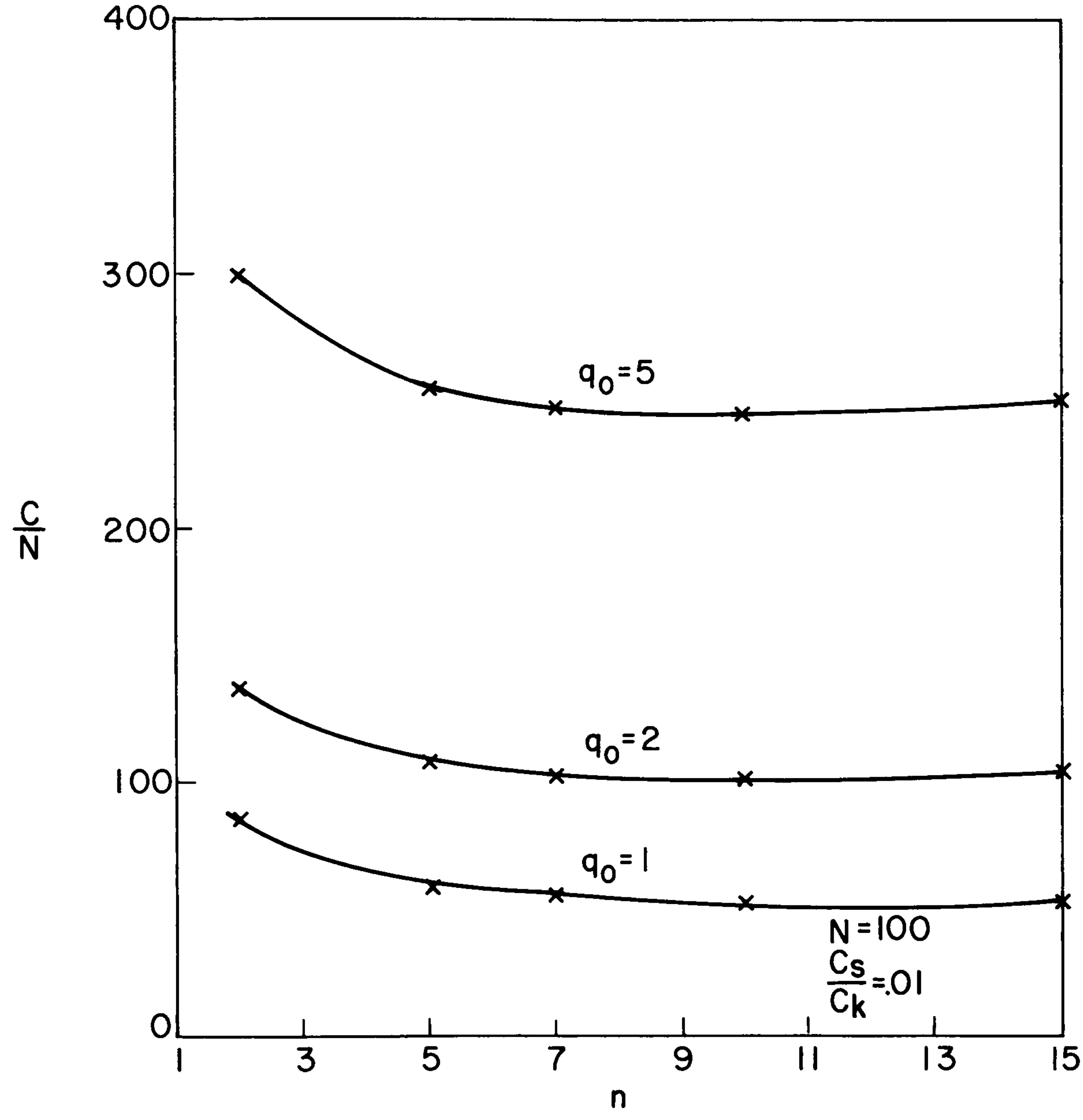


Fig. E-1 CHANNEL SHARING COSTS.

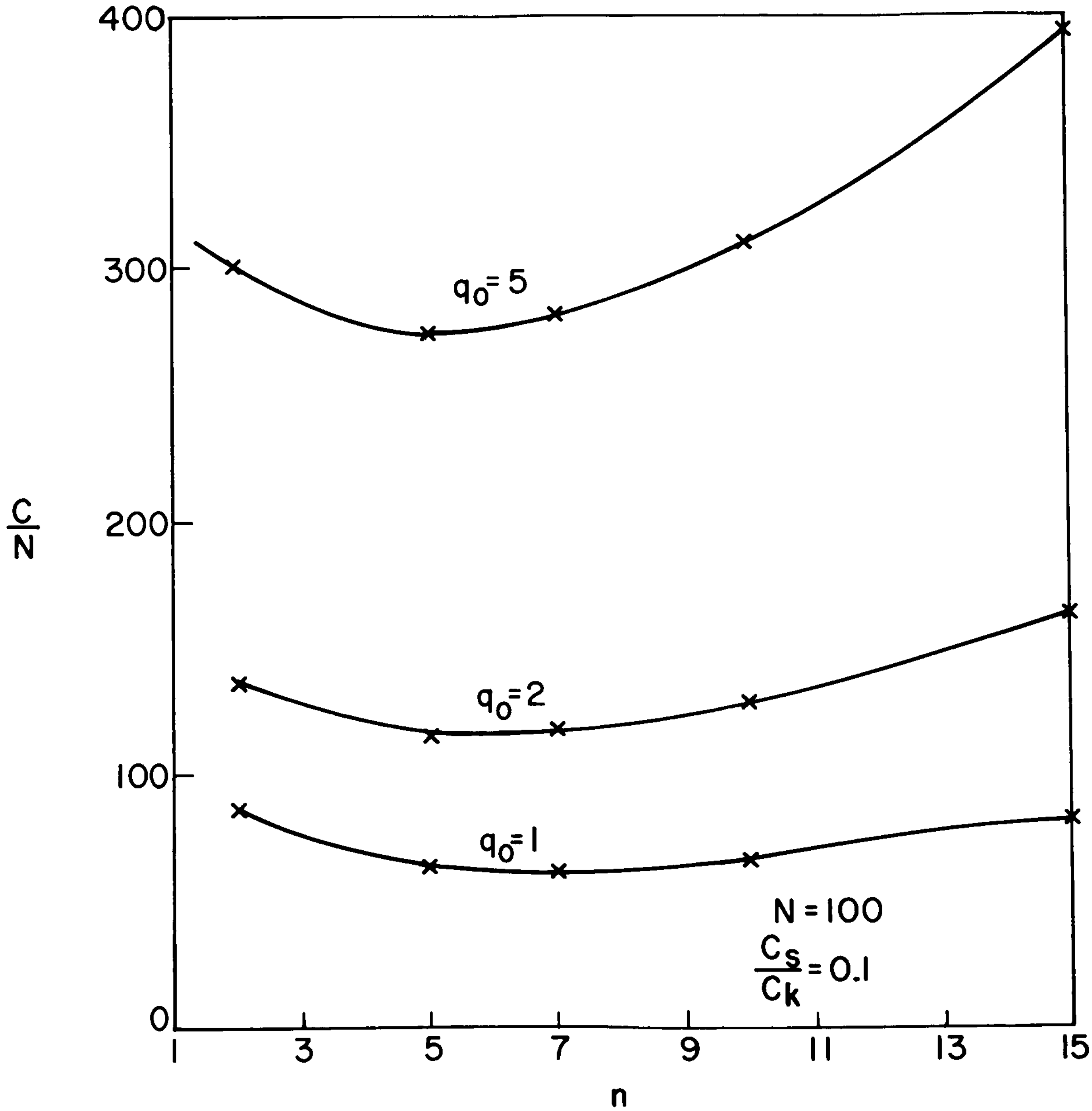


Fig. E-2 CHANNEL SHARING COSTS.

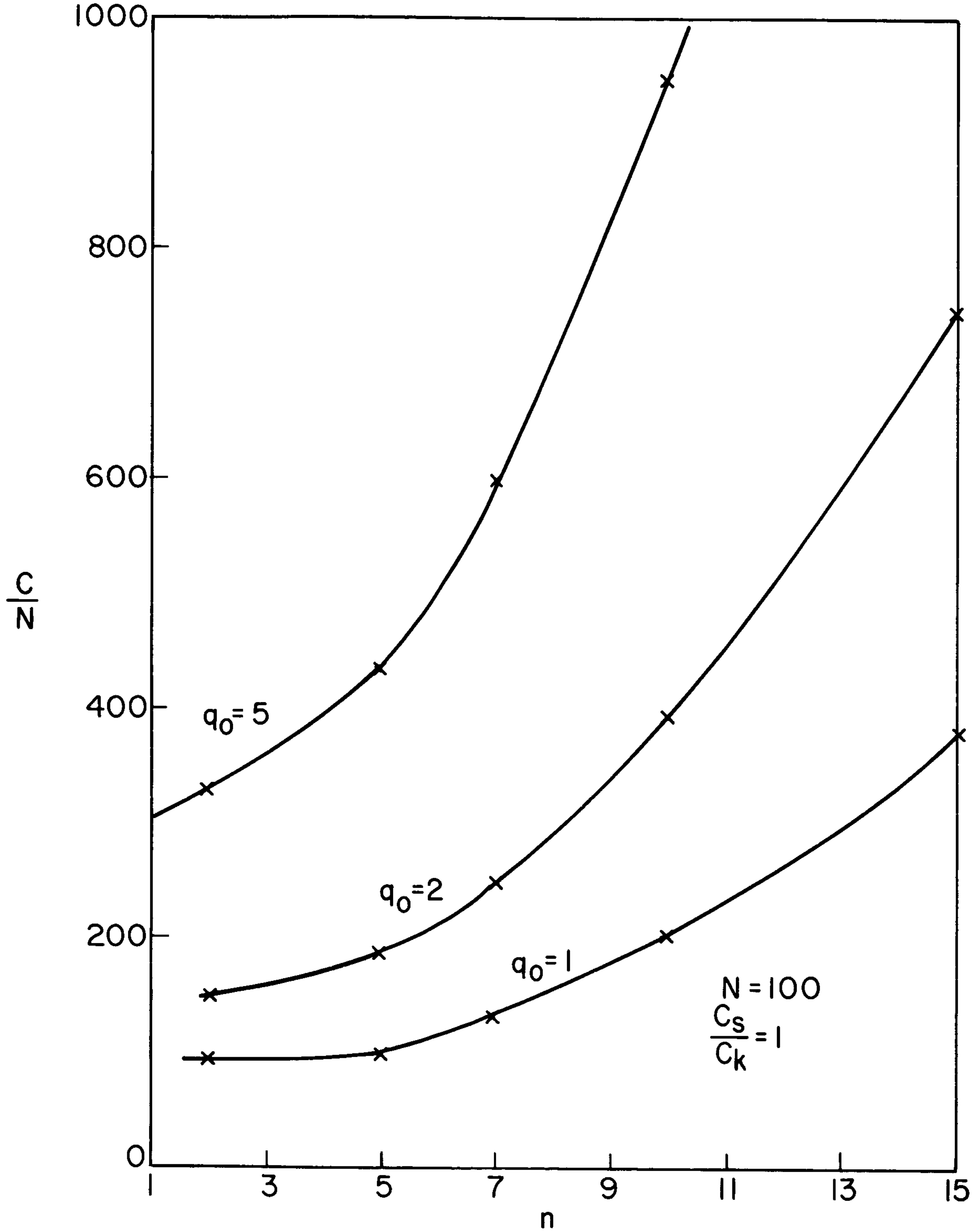


Fig. E-3 CHANNEL SHARING COSTS.

Appendix F

STRUCTURAL ANALYSIS

Analysis of Solar Array Structure

The compact folded arrangement of the solar panels enables them to restrain one another during the critical boost phase so that buckling and vibration of the panels are not severe design constraints. The presence of intermediate supports in the form of inter-panel spacers helps resist any tendency for long wavelength buckling and vibration of the panels.

Design Goal for Array Weight

In a preliminary report of the design of a large solar cell array, Boeing[1] is working toward a design figure of 0.2 lb/ft^2 for structural weight. The Boeing design is not completed, and various problems have arisen requiring the addition of stiffening structure. The 0.2 lb/ft^2 figure was taken as the design goal for a significantly less complex configuration.

Cases Governing Design

(a) With buckling and vibration considerations minimized, the only other service load to consider would be concentric load in the plane of the panel during boost. For this case:

Panel area	= 4' × 13' = 52 sq ft
Wt. of solar cells = 0.212 lb/sq ft × 52	= 11 lb
Structural wt. (assumed) - 0.2 lb/sq ft × 52	= 10.4
Total wt.	= 21.4 lb

Launch environment: 8g × 1.5 (safety factor) = 12g

Assume total load = 12 × 21.4 = 256 lb

Define: $\sigma = \frac{P}{A}$

where:

σ = unit stress in compression (psi)

P = applied concentric load (lb)

A = cross sectional area (sq in)

σcy = compressive yield stress

σcy = 50 ksi for Be

$$A = \frac{P}{\sigma cy} = \frac{256}{50 \times 10^3} = 5.14 \times 10^{-3} \text{ in}^2$$

$$A = \text{width} \times \text{thickness} = 48 \text{ inches} \times t$$

$$t = \frac{5.14 \times 10^{-3}}{48} = 1.09 \times 10^{-4} \text{ inches}$$

A sheet of beryllium 48 $\times$ (1.09×10^{-4}) will satisfy this design requirement.

(b) A more severe design constraint is placed upon the panel by handling during fabrication. It will be specified that the panel be supported so as to remain undeflected during installation of the solar cells. However, it is conceivable that after solar cell installation the panel may be lifted at either end, behaving as an end-supported plate beam of 13 foot span. For this case, a deflection constraint will govern the design. A deflection on the order of 1 foot at the center of the 13 foot span should not damage the solar cell installation, and is selected as the allowable deflection during ground handling for sizing the plate structure.

Three alternative plate configurations were considered:

(i) Solid beryllium plate

(ii) Beryllium sandwich with vented aluminum honeycomb core

(iii) Integrally stiffened beryllium plate produced by chemically milling or machining out material to produce the stiffeners.

(i) Solid beryllium plate:

Define: $w_{max} = \frac{5}{384} \frac{pl^4}{D}$ (Ref. 2) where:

w_{max} = maximum plate deflection

p = transverse plate load

l = span length

D = plate bending stiffness = $\dfrac{Et^3}{12(1-\nu^2)\nu}$

E = Young's modulus

t = plate thickness

ν = Poisson's ratio

So that: $$w_{max} = 12'' = \frac{5}{384}\left\{.067\ t + \left[\frac{\frac{(0.212)}{(144)}(13 \times 12)^4}{\frac{(42)(10^6)\ t^3}{12\ (1-\nu^2)}}\right]\right\}$$

$\nu \approx 0$ for Be

Solving for t:

$$t^3 - (1.23 \times 10^{-2})t - 2.74 \times 10^{-4} = 0$$

$$t \approx 0.17 \text{ inches}$$

This is a solid beryllium plate which satisfies the deflection constraint. This plate would weigh 1.64 lb/ft^2 which is considerably in excess of the design goal.

(ii) Honeycomb sandwich plate:

Define: $$w_{max} = \frac{5}{384}\frac{pl^4}{D_f} \qquad \text{(F-1)(Ref. 3)}$$

Where: D_f = Bending stiffness of face plates about the neutral surface of the section

$$= \frac{E_f\ t_f\ h^2}{2(1-\nu^2)} \qquad \text{(Ref. 3)}$$

Solving for D_f:

$$D_f = \frac{5}{384}\left\{\frac{\left[2(.067t_f) + \frac{0.212}{144}\right](13 \times 12)^4}{12}\right\}\left\{\left[\frac{E_f\ t_f\ h^2}{2(1-\nu^2)}\right]\right. \quad \text{(F-2)}$$

$$E_f = 42 \times 10^6 \text{ psi}$$

$$t_f\ h^2 = 0307\ (.134\ t_f + .00147) \quad \text{(F-3)}$$

For practical honeycomb sandwiches: $20 \leq h/t_f \leq 40$. If it is assumed that $h = 30\ t_f$, then a solution of Eq. (F-3) results in a t_f of .008" and an h of .24 inches. Using a lightweight aluminum honeycomb core because no buckling problem exists, a core weight of 4 lb/ft^3 should be adequate. The structural weight will then be 0.234 lb/ft^2 which is slightly above design goal but is satisfactory. Checking w_{max} in Eq. (F-1) using the higher structural weight, w_{max} is found to be 16.8 inches, which is an acceptable figure. Checking the bending stress in a plate of such dimensions:

$$\text{Define: } \sigma_{bending} = \frac{MC\ E_f}{D_f} \quad \text{(F-4)}$$

Where: M = maximum bending moment

C = distance to face plate from neutral surface

$$\sigma_{bending} = \frac{pl^2}{8}\ \frac{h}{2}\ \frac{E_f}{E_f t_f}\ \frac{(2)}{h^2}$$

= 2350 psi (sufficiently low to insure against dimpling)

Checking deflections due to shear deformations using the relationship

$$\frac{dQ_x}{dx} = -p \quad \text{(F-5)}$$

where Q_x = unit shear force on an x-face of the plate it is found that for a uniform transverse load p applied to the plate, no shear correction is necessary. The sandwich plate therefore meets desired deflection, stress and design weight constraints.

(iii) Integrally stiffened beryllium plate

$$\text{Define:} \quad w_{max} = \frac{5}{384} \frac{pl^4}{D_{eff}} \tag{F-6}$$

where: D_{eff} = effective bending stiffness of stiffened plate

$$= \frac{Et^3}{12(1-\nu^2)} \left[1 + \frac{E_s I_{so}}{Et^3} \frac{12(1-\nu^2)}{d} \right] \qquad \text{(Ref. 3)}$$

E = Young's modulus of plate

E_s = Young's modulus of stiffeners = E

d = centerline distance between stiffeners

I_{so} = moment of inertia of a stiffener about the centroidal axis of the top plate.

Solving (F-5) for D_{eff} and equating it to D_{eff}, an equation in terms of various plate parameters is obtained:

$$\left(\frac{b_s}{d}\right)\left(\frac{h_s}{t}\right)^2 \left[12 + 13\left(\frac{h_s}{t}\right)\right] = \frac{.0131}{t^3}$$

$$\left[\frac{W_{sp}}{W_p}(.067) + .00147\right] - 1 \tag{F-7}$$

where b_s = width of stiffeners

h_s = depth of stiffeners

t = thickness of top plate

W_{sp} = weight of top plate plus stiffeners

W_p = weight of top plate only

The constants of Eq. (F-6) arise from the specific loadings, materials and deflection constraints of the particular array. Through a trial and error procedure, parameters were selected so that the plate was sized as shown in Fig. F-1. This plate has a weight of 0.23 lb/ft^2 and deflects no more than 12" under load with a two-point support.

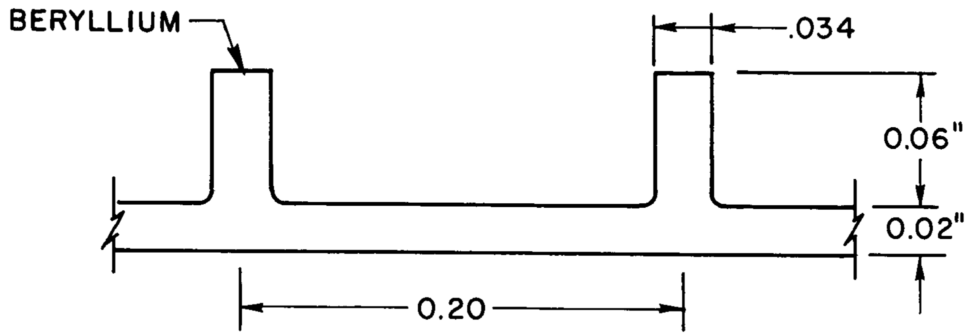
The honeycomb and integrally stiffened designs were examined for selection of the better alternative. Both designs weigh almost the same. The integrally stiffened panel was somewhat stiffer. However, it requires chemical milling or machining (both processes are expensive and wasteful of the material between stiffeners, which has to be discarded). The honeycomb design only requires beryllium plate stock and off-the-shelf aluminum honeycomb. Joining of the aluminum to beryllium would be by bonding. Some penalty would be paid in terms of higher operating temperatures of solar cells, but this would only be a few degrees. Therefore, essentially on a cost basis, the honeycomb sandwich was selected in preference to the integrally stiffened design. Figure F-1 shows a comparison of the alternative designs.

Interpanel Stiffeners

It is important that when the panels are in the folded position the solar cells do not contact adjacent panel surfaces. However, there must be intermediate supports to minimize buckling and vibration effects. To meet these considerations, 1 inch cubes of a resilient material will be used as spacers on 1 foot centers over the front surfaces of the panels. Materials such as urethane foam or silicone rubber foam should be suitable, provided it has been seen that the materials have been outgassed. The loss of solar array area will be:

$$3 \times 12 \times 1 = 36 \text{ sq in. per panel}$$

i.e., a total of 2 sq ft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.034
0.06"
0.02"
0.20

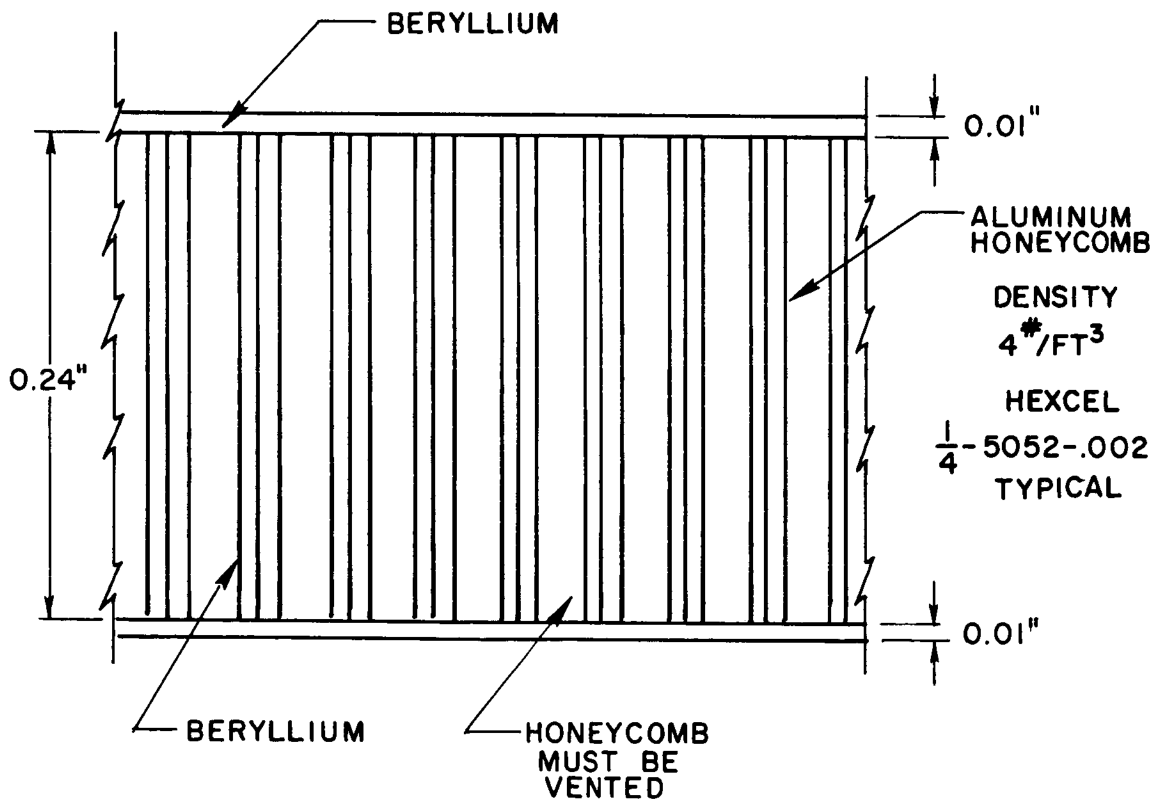
BERYLLIUM
0.01"
ALUMINUM
HONEYCOMB
DENSITY
4#/FT3
HEXCEL
1/4-5052-.002
TYPICAL
0.24"
0.01"
BERYLLIUM
HONEYCOMB
MUST BE
VENTED

Fig. F-1.

Analysis of Central Structure (Bending-Torsion Box)

Only the major structural members are sized in this section. Structural detailing is not within the scope of this preliminary study. A description and illustration of the layout of this portion of the spacecraft is contained in Section 7.1.

a. Main load carrying members (Outer tubes). These two members are to be extruded beryllium of the shape previously shown in the main body of the report. Assuming a spacecraft weight of 1600 lb, each member supports 800 lb and is subjected to a total loading of 9600 lbs. Assuming sufficient constraint against buckling from adjacent structure, the required cross sectional area is:

$$A = \frac{P}{\sigma_{cy}} = \frac{9600}{50 \times 10^3} = .193 \text{ sq in}$$

Perimeter of extrusion: approximately 13"

$$\text{Required thickness} = \frac{.193}{13} = .015 \text{ inches}$$

To satisfy handling and fastening requirements: recommend a 0.05" wall. The weight per member then becomes:

$$(.05 \text{ ft}) \times (13 \text{ in}) \times (13 \text{ ft}) \times (12 \text{ in/ft})$$
$$\times\ (.067 \text{ lb/in}^3) = 6.8 \text{ lb}$$

b. Equipment Shelves

Various sizes of equipment will be placed on the different shelves; the shelves, in turn, serve as bulkheads to rigidize the entire box. As a preliminary sizing exercise, use the same material for all shelves. Try all aluminum honeycomb sandwich with 0.005" plate over a .25" core of 8.1 lb/ft^3 density. The highest load would be from the batteries which are supported between the two center shelves. Assume 75 lb per shelf uniformly distributed, which is actually a conservative figure. The shelves are essentially clamped ended.

Shelf area supporting batteries ≈ 108 inches

Load per square inch $= \frac{75}{108} = 0.7\ \text{lb/in}^2$

Moment of inertia of shelf for 6" width

$$= \frac{6 \times 0.25^3}{12} = .0078\ \text{in}^4$$

$$\frac{Mc}{I} = \frac{wl^2}{12}\frac{C}{I} = \frac{(.7 \times 6)(18)^2(.13)}{12(.0078)} = \frac{\begin{matrix}1940\ \text{psi}\\ \times\ 12\ \text{g's}\end{matrix}}{\begin{matrix}23{,}000\ \text{psi}\\ \text{in bending}\end{matrix}}$$

Analysis of Shelf Face Plate Buckling:

Define:
$$N_{x_{cr}} = \frac{\dfrac{4\pi^2\ E_f\ t_f\ h^2}{2(1-\nu^2)\ l^2}}{\left[1 - \dfrac{4^2}{2(1-\nu^2)}\ \dfrac{E_f\ t_f\ h}{G_c\ l^2}\right]} \qquad \text{(F-8)} \quad \text{(Ref. 3)}$$

Where:

E_f = Young's modulus of face plates

t_f = thickness of face plate

h = distance between face plates

G_c = shear modulus of core

For:

t_f = 0.005 inches

average plate width - 6 inches

l = 18 inches

$E_f = 10^7$ psi

$G_c = 6 \times 10^4$ psi

$\nu = 0.33$

$N_{x_{cr}}$ = 204 lb

$\sigma_{x_{cr}} = \frac{204}{2 \times .005} = 20{,}400$ psi too low

Since $\sigma_{x_{cr}} \propto t_f$, take t_f as 0.01 inches

$$\sigma_{x_{cr}}\ (\text{for } t_f = 0.01) = 40{,}800\ \text{psi}$$

Margin of safety then is + 0.77

Use t_f of 0.01", h = .25 inches, core density of 8.1 lb/ft^3

The shelf weight is:

$$\text{Honeycomb weight} = 2 \times t \times \rho = 2 \times .01 \times .101 \times 144 = .265\ lb/ft^2$$

$$= \frac{.25}{12} \times 8.1 = \underline{.168\ lb/ft^2}$$
$$0.433\ lb/ft^2$$

Shelf area = 3.16 ft^2

Weight per shelf = 1.37 lb

c. Vertical Shear Webs

These are assumed fabricated from aluminum sheet stock in the shape of channels. Channels must be 10" deep, based on the equipment packaging constraint. Use 0.05" gage sheet selected on the basis of fastening and forming requirements. Channel weights are then:

$$.05'' \times 10'' \times 9' \times 12''/ft \times 0.1\ lb/in^3 = 5.40\ lb$$

less 20% for lightening holes = <u>1.08</u>

weight per channel = 4.32 lb

d. Cover Plates Front and Rear

Very little direct load is applied to these plates. They serve mainly to close and stiffen the box structure. Aledad plate, .008" thick, should be adequate. Local stiffening can be provided for panel mounted equipment.

Weight per plate: $.008'' \times 9' \times 5'' \times 144\ in^2/ft^2 \times 0.1 = 5.2\ lb$

Weight of central structure:

Outer tubes	13.60 lb
Shelves	5.50 lb
Vertical shear webs	8.64 lb
Cover Plates	10.40 lb
Total	38.14 lb

All fastening of the structural components shall be bonding with a suitable epoxy cement. This will insure structural integrity and continuity as well as improved structural damping qualities.

Analysis of Communication Package

a. Turret Shaft Connecting Rotating Antenna Packages

All thrust is taken by bearings. The shaft only rotates the packages at an extremely slow rate. Two inch outer diameter specified on a space basis. Using a .05 inch wall thickness aluminum tube provides a shaft weighing 3.44 lb total.

b. Antenna Mast

The mast shall be stowed during the boost phase and held by several intermediate clamps to prevent long column buckling.

If a 2 inch diameter aluminum tube of .05 inch wall thickness is used, the maximum compressive stress is:

$$\sigma = \frac{31 \text{ lb} \times 12 \text{ g}}{0.314 \text{ sq in}} = 1190 \text{ psi} \qquad \text{(satisfactory)}$$

The weight per 6 ft length is 2.25 lb.

c. Communications Box

This 1 ft × 4 ft × 2 ft box contains 150 lb of transponders on its lower shelf and 90 lb of TWT's and antenna on its upper shelf. A plate and angle construction is suggested with local reinforcement around the lower bearing attachment and the antenna attachment. The loads are assumed as uniformly distributed over the shelf surfaces. The shelves are taken as plates with minimum rotational restraint provided along the 4 sides to establish conservative estimate of the shelf thickness.

$$\text{Top shelf load:} \quad \frac{90}{1 \times 4 \times 12} \times 12 = 22.5 \text{ psi}$$

$$\text{Bottom shelf load:} \quad \frac{150}{1 \times 4 \times 12} \times 12 = 37.5 \text{ psi}$$

If intermediate channel or box frames are used on the box on 1 ft centers, the maximum plate spans are 1 ft × 1 ft. From Reference (2), the maximum moment in the lower plates will be 259 in/lb per inch. After several iterations, a beryllium aluminum sandwich having a t_f of 0.015 inches and an h of 0.5 was selected.

From Eq. (F-4),

$$\sigma_{bending} = \frac{M}{t_f h} = 34{,}500 \text{ psi} \qquad \text{(Satisfactory)}$$

A comparison with the calculation of $N_{x_{cr}}$ for the equipment shelves using Eq. (F-8) indicates that for the thicker face plates and shorter span of this plate, the buckling stress will be in excess of 41,000 psi provided a core density of 8 lb per cubic foot is maintained. The weight per shelf is then 4.7 lb. Beryllium channel or box sections will be used to form the framework. The box sides will be enclosed by removable beryllium plates used as shear panels between frames. The chassis of the TWT's located in the upper section of the box will be used to strengthen the side walls to which they are mounted. This appears to be an advantageous integration of structure and electronics.

Calculation of Plate Natural Frequency

$$\omega^2 = \frac{D_{eff}\left(\frac{\pi}{L}\right)^4 \left[1 + \left(\frac{L}{b}\right)^2\right]^2}{\rho}$$

$$D_{eff} = \frac{E_f t_f \; h^2}{2} = \frac{42(10^6)(.015)(.5)^2}{2} = 78{,}500 \text{ in}^2$$

$$\rho = \frac{2t_f \; \delta_f}{g} + \frac{\delta_c k}{g} = \frac{2(.015)(.067)}{196} + \frac{(8)(.5)}{196 \times 1728}$$

$$= (1.02 + 1.18)10^{-5} = 2.2 \times 10^{-5}$$

$$\frac{L}{b} = 1 \qquad L = 12''$$

$$\omega^2 = \frac{(7.85 \times 10^4)(4.8 \times 10^{-3})(4)}{2.2 \times 10^{-5}} = 111 \times 10^6$$

$$\omega = 10.5 \times 10^3 = 10{,}500 = 2\pi f$$

$$f = 1680 \text{ cps}$$

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1. "Large Area Solar Array Design," Second Quarterly Report, The Boeing Company, Seattle, April 1967.
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3. Mayers, J., "Unified Theory of Structural Analysis for Present and Future Aerospace Vehicles," Presented at 1st AIAA Professional Study Series, April 1-2,1956, California Institute of Technology, Pasadena.
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Appendix G

WRAP-AROUND SOLAR PANEL SCHEME

Among various satellite configurations considered before selection of the proposed design, one that deserves special mention is the wrap-around concept for solar panels. The concept involves mounting the solar cells on a multijointed substrate which is wound around a drum-like core. Figure G-1 illustrates the principle. Spring loaded hinges at the substrate joints unwind the panels once restraining bands are released when in orbit. Latches lock the joints in place and provide rigidity between substrate panels. The panels themselves are of a lightweight honeycomb, which imparts sufficient rigidity to the unfurled arrays. The panels are always sun-oriented, requiring either an active stabilization system or despining if spin stabilization is used.

The great advantage this technique of packaging the solar arrays has is the extremely efficient use of existing booster space. Wrapping the arrays around a central core allows the interior space to be utilized, in this case, by the antennae. The arrays would be extremely thin and even when wrapped in two layers would not occupy more than a few inches in thickness. The substrate and rigidizing system is very lightweight, yielding a low weight per kilowatt ratio.

Several factors contributed to the rejection of the wrap-around technique and no further development of the concept was made. It was decided that the triangular packaging arrangement worked better with the articulated rather than the wrap-around array. This provided more efficient use of booster space than the circular packaging. The access problem weighed heavily in favor of the triangular shape. Once the panels are wrapped up, access to the interior becomes difficult. This problem can be overcome by the use of access holes in the solar arrays, but less area is then available for solar cells. Further study of access improvement could overcome the problem. Finally, further work on design details and testing of prototype hardware would be necessary before the concept could be proved, and some idea of its reliability obtained. The technique is novel and untried while the hinged, accordion-like unfolding of the triangular array utilizes concepts that have been flight-proven. The structures subgroup favors this wrap-around concept quite strongly and believes it to have merit for consideration in future satellite designs.

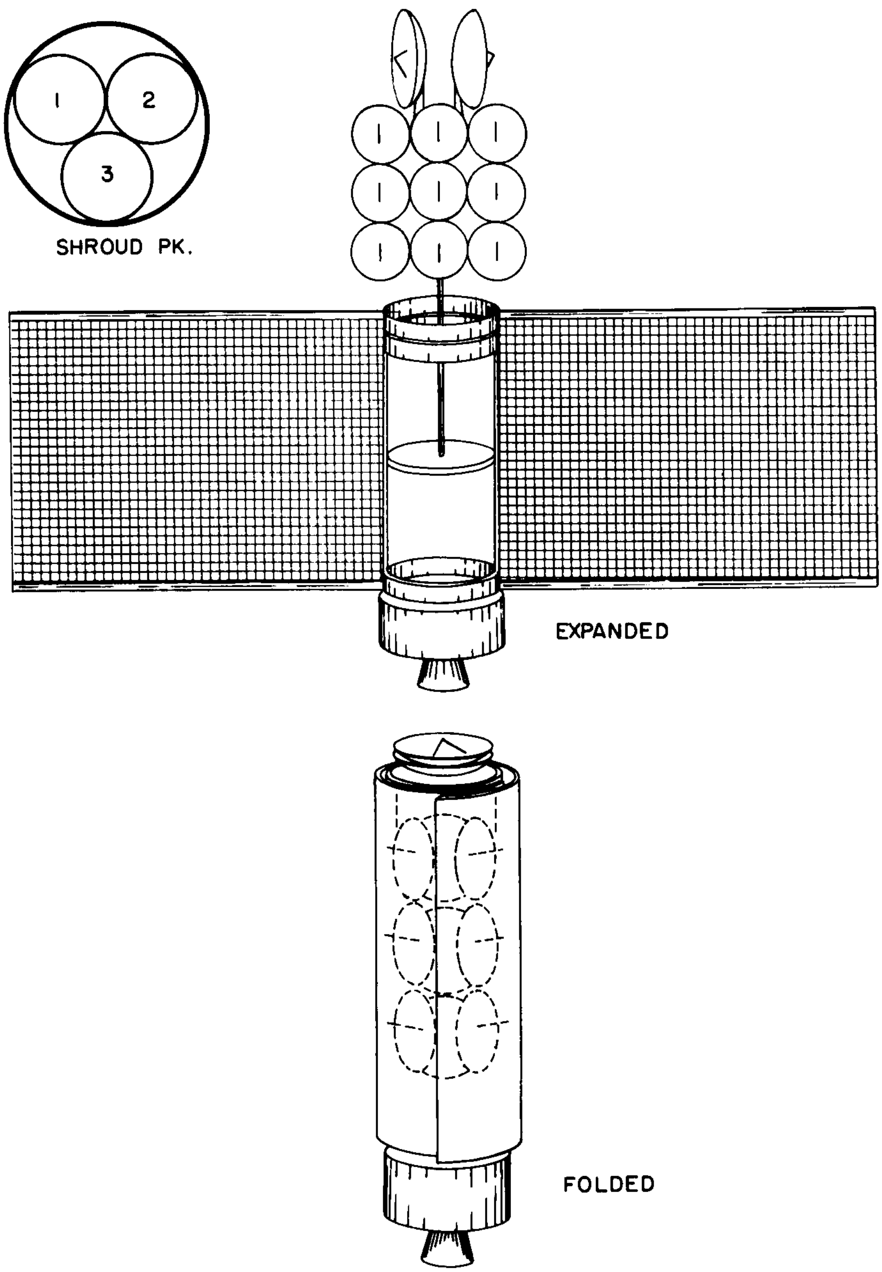


Fig. G-1 PROPOSED WRAP-AROUND SOLAR CELL ARRAY.

Appendix H

PROPOSED SYSTEM FOR DIRECTION FINDING USING RADIO TECHNIQUES

Two different separations were investigated for the direction finding dipoles λ and $\lambda/4$. It was found that for elimination of ambiguities of pointing direction for a null output signal, the separation of the dipoles should be $\lambda/4$ rather than λ. This separation results in a relatively insensitive system for the detection of angles of pointing of the order of $\pm 0.1^\circ$. An output signal of the order of 0.04 V is about all that can be obtained with separation of $\lambda/4$.

Increased sensitivity can be obtained by increasing the separation of the dipole to a number of wavelengths greater than one, but then the problem of ambiguity becomes considerable. It is proposed that a coarse sensor could be used to eliminate the ambiguity and a fine interferometer sensing system with the separation of the antennae by several wavelengths be used for detecting angular positions of the order of $\pm 0.1^\circ$. The coarse sensor could be of the same interferometer type as the fine sensor but with the separation of antennae being $\lambda/4$. For an angular error of pointing of $\pm 1.5^\circ$, the $\lambda/4$ separation system would give an output of about 0.6 V. If it is thought this signal is not strong enough, the coarse sensor could be a horizonal scanner.

The geometry block diagram and equations for the proposed system are shown in Fig. H-1.

ϕ = angle of lag of signal at A with respect to signal at B

$$\phi = \frac{2\pi}{\lambda} h = \frac{2\pi}{\lambda} l \sin E$$

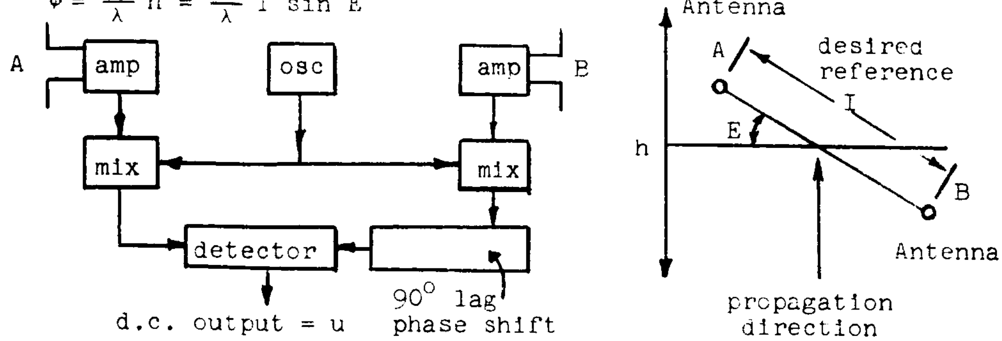


Fig. H-1. DIRECTION FINDING APPARATUS.

The output signal $u = \mu/\pi \sin, \phi = (\mu/\pi)\sin[2(\pi/\lambda)I \sin E]$, where μ is the amplitude of the sine wave signals out of the mixers. If the signal at A lags signal at B, u is positive; if the signal at B lags signal at A, u is negative.

If separation $I = 10\lambda$, the output signal from detector is 1.74 V for angular error of pointing $E = 0.1°$, as compared with 0.04 V for $I = \lambda/4$. If $I = 20\lambda$, the output signal would be ≈3.5 volts for $E = 0.1°$, but the coarse sensor would have to be more sensitive. With $I = 10\lambda$, the coarse sensor needs to bring E down to about 1.5°, while with $I = 20\lambda$, the coarse sensor would have to bring E down to about 0.75°. See Fig. H-2.

Another consideration in the separation of the antennae is the frequency used and the dimensions of the spacecraft. Since there will be a 6000 MHz signal used on the uplink to the satellite, it would be advantageous to use it.

For f = 6000 MHz

$$\lambda = \frac{300}{f_{MHz}} = \frac{300}{6000} = \frac{1}{20} = 0.05 \text{ m} \quad \text{or} \quad 5 \text{ cm}$$

or

$$\lambda = \frac{5}{2.54} = 1.97 \text{ inches}$$

If $I = 10\lambda$, I = 19.7 inches or 50 cm.

If $I = 20\lambda$, I = 39.4 inches or 1 m.

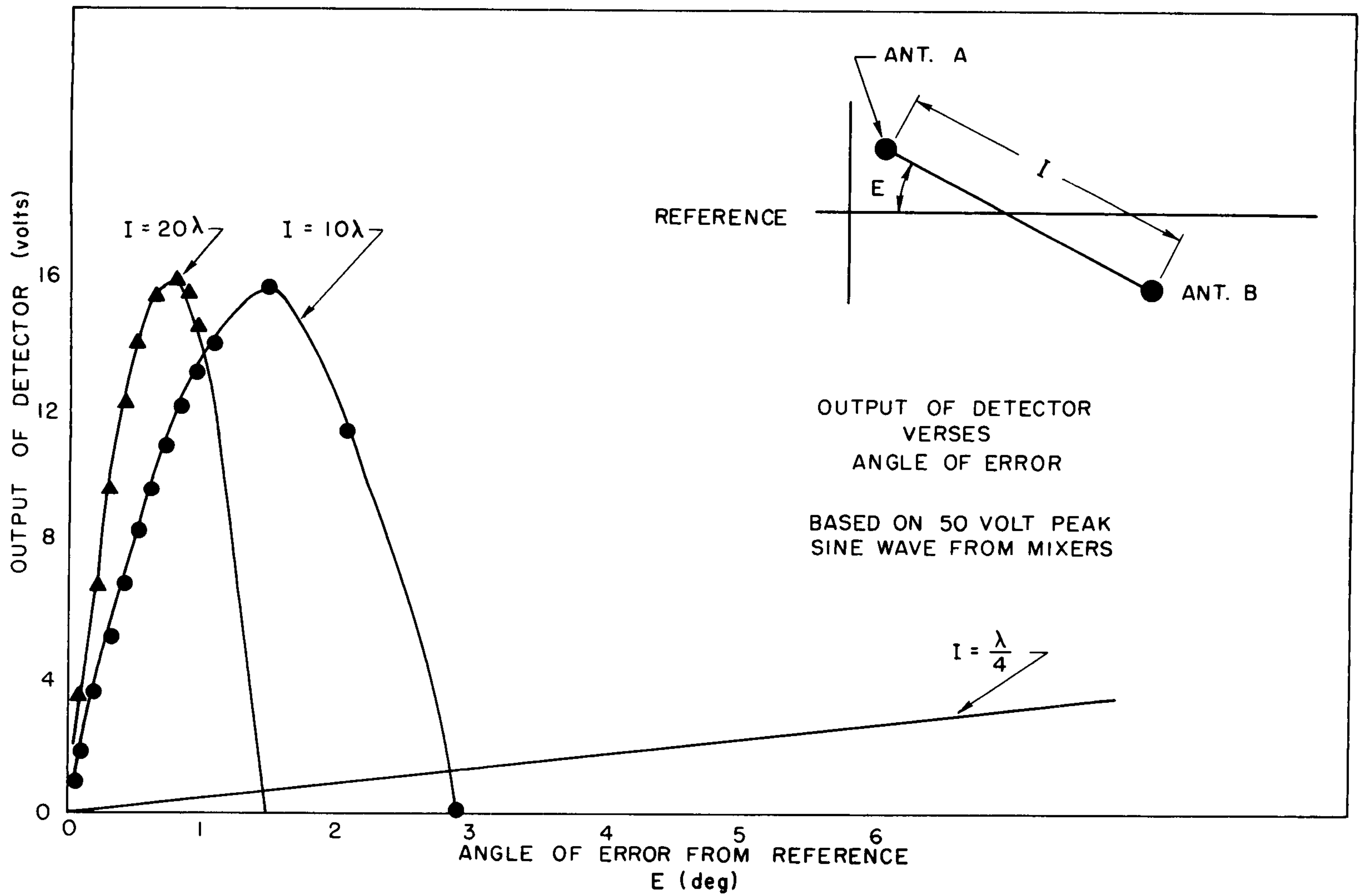


Fig. H-2 OUTPUT OF DETECTOR VS ANGLE OF ERROR
(Based on 50 Volt Peak Sinewave from Mixers).

Appendix I

MOMENTUM WHEEL CALCULATIONS

Cyclic torque - roll and yaw (acts on antenna and solar panels)

$A = 500 \text{ ft}^2$ $\qquad$ $L = 1/2$ ft (assumed lever arm)

$$T = 500 \times 0.94 \times 10^{-7} \times 1/2 = 2.35 \times 10^{-5} \text{ lb ft}$$

Period: $2\Delta t = 24$ hr

$$\Delta H = T\Delta t = 2.35 \times 10^{-5} \times 1/2 \times 3600 \times 24 = 1.015 \text{ lb ft sec}$$

r = radius of gyration = 1/4 ft (max speed = 2000 r/min)

$$m = \frac{\Delta H}{r^2 \omega} = \frac{1.015}{(1/16) \times \frac{800\pi}{12}} = 7.25 \times 10^{-2} \text{ slugs wheel mass}$$

$$W = mg = 7.25 \times 10^{-2} \times 32.2 \approx 2.5 \text{ lb wheel weight}$$

Noncyclic torques due to gas leak, etc.

$T \approx 0.1$ T solar pressure

$$T \approx 2.35 \times 10^{-6} \text{ lb ft}$$

Store momentum for 1 day

$$\Delta H = T\Delta t = 2.35 \times 10^{-6} \times 3600 \times 24 = 2.03 \times 10^{-1} \text{ lb ft sec}$$

$$m = \frac{\Delta H}{r^2 \omega} = \frac{2.03 \times 10^{-1}}{(1/16) \times \frac{800\pi}{12}} = 1.45 \times 10^{-2} \text{ slugs}$$

$$W = mg = 1.45 \times 10^{-2} \times 32.2 \approx 0.5 \text{ lb}$$

Make roll and yaw wheels 3 lb each

Cyclic torque - pitch (acts only on antenna)

$$A = 100 \text{ ft}^2 \text{ (ant)} \qquad L = 1/2 \text{ ft}$$

$$T = 100 \times 0.94 \times 10^{-7} \times 1/2 = 4.7 \times 10^{-6} \text{ lb ft}$$

Daily cycle

$$\Delta H = T\Delta t = 4.7 \times 10^{-6} \times 1/2 \times 3600 \times 24 = 0.203 \text{ lb ft}$$

$$r = \text{radius of gyration} = 1/4 \text{ ft (max speed} = 2000 \text{ r/min)}$$

$$\omega = \frac{800\pi}{24}$$

$$m = \frac{\Delta H}{r^2\omega} = \frac{2.031 \times 10^{-1}}{1/16 \times \frac{800\pi}{12}} = 1.55 \times 10^{-2} \text{ slugs}$$

$$W = mg = 1.55 \times 10^{-2} \times 32.2 \approx 0.5 \text{ lb}$$

Secular torque - pitch

$$A = 400 \text{ ft}^2 \text{ (solar panel)}$$

$$L = 1/2 \text{ ft}$$

$$T = 400 \times 0.94 \times 10^{-7} \times 1/2 = 18.8 \times 10^{-6} \text{ lb ft}$$

To store secular momentum change for 1 day

$$\Delta H = T\Delta t = 1.88 \times 10^{-6} \times 3.600 \times 2.4 = 1.624 \text{ lb ft sec}$$

$$m = \frac{\Delta H}{r^2\omega} \frac{1.624}{1/16 \times \frac{800\pi}{12}} \approx 12.4 \times 10^{-2} \text{ slugs}$$

$$W = mg = 1.24 \times 10^{-1} \times 32.2 \approx 4 \text{ lb}$$

Make pitch wheel 4.5 lb.

Torquing of solar panel drive motor around pitch axis and friction of bearings and slip rings may produce another 18.8μlb ft of torque which means another 4 lb of pitch wheel to store this secular momentum change for one day.

Make pitch wheel 8.5 lb.

Appendix J
GRAVITY GRADIENT CALCULATIONS

Referring to Fig. J-1,

$$F_c = F_g$$

$$(m_A + m_B)\omega_o^2 R = \frac{GM(m_A + m_B)}{R^2}$$

$$\omega_o^2 R^3 = GM$$

where

M = mass of the earth

G = universal gravitational constant

ω_o = satellite angular velocity.

Gravity gradient torque:

$$T_g = F_\theta L = m_B\left[\omega_o^2(R + L\cos\theta) - \frac{GM}{(R + L\cos\theta)^2}\right] L\sin\theta$$

$$= m_B\left[\omega_o^2(R + L\cos\theta) - \frac{GM}{R^2}\left(1 + \frac{L}{R}\cos\theta\right)^{-2}\right] L\sin\theta\ .$$

To a first approximation, $\frac{L}{R} = 1$. Therefore,

$$T_g = m_B\left[\omega_o^2(R + \omega_o^2 L\cos\theta) - \frac{GM}{R^2} + 2\frac{LGM}{R^3}\cos\theta\right] L\sin\theta\ .$$

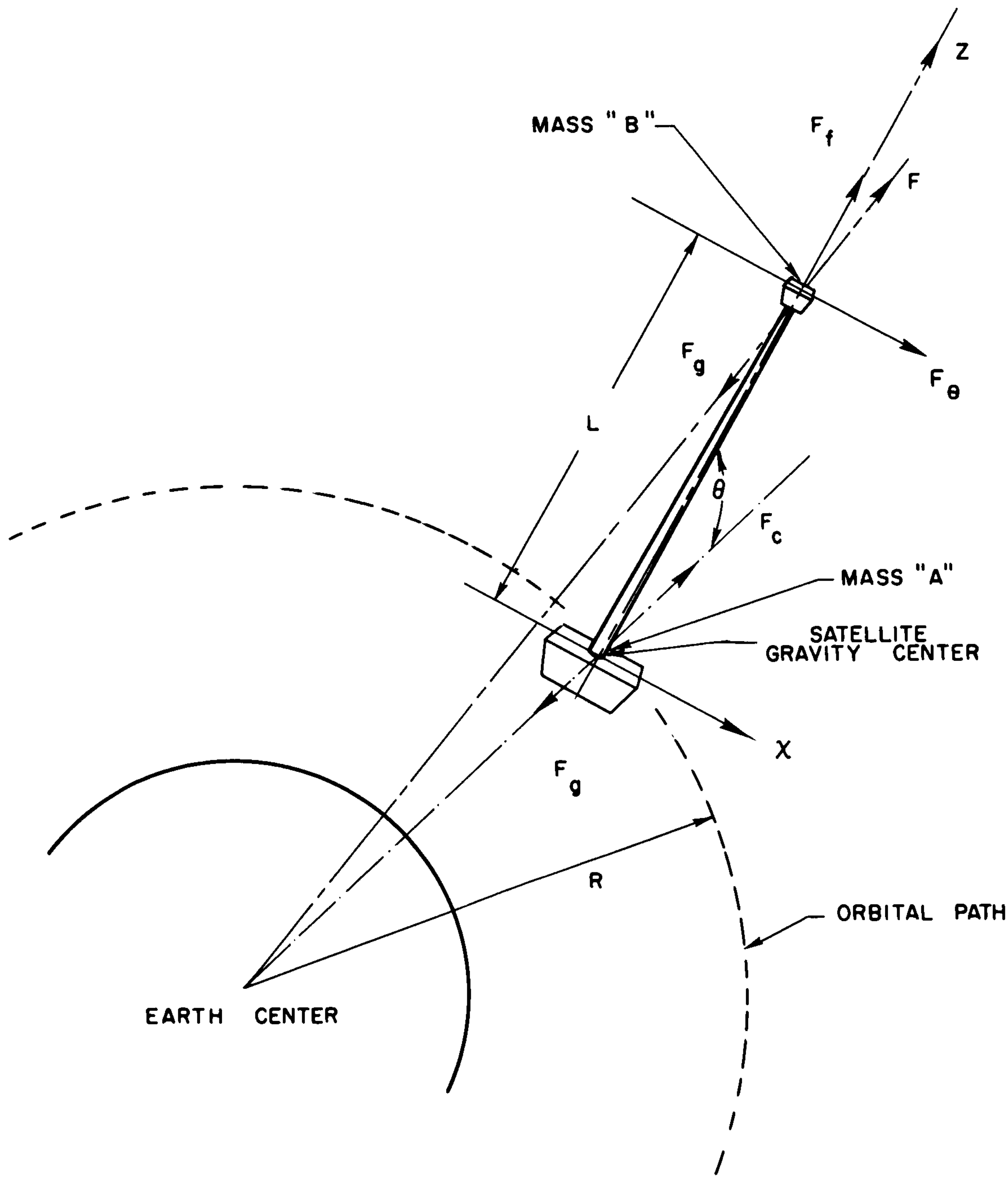


Fig. J-1 GRAVITY GRADIENT FREE-BODY DIAGRAM.

Since

$$\omega_o^2 R = \frac{GM}{R^3} ,$$

$$T_g = m_B\left(3\omega_o^2 L^2 \cos \theta \sin \theta\right) = \frac{3}{2}\left(\omega_o^2 L^2 \sin 2\theta\right) .$$

An alternate expression for the gravity gradient torque is

$$T_g = \frac{3}{2} \omega_o^2 (I_x - I_z) \sin 2\theta$$

where

I_x and I_z are the moments of inertia of the vehicle about the x and z axes.

Considering the expected length of the boom and that the mass center is located within the mass m_A, the following assumptions could be made:

$$I_x \gg I_z \quad \text{and} \quad I_x = m_B L^2 .$$

Therefore,

$$T_g = \frac{3}{2} \omega_o^2 I_x \sin 2\theta = \frac{3}{2} \omega_o^2 m_B L^2 \sin 2\theta .$$

Solar Radiation Pressure and Torque

$$\text{Solar pressure} = 0.94 \times 10^{-7} \text{ lb/ft}^2.$$

For 500 ft^2 exposed area of solar arrays and antennae, and an assumed 0.5 ft eccentricity between the center of pressure and the center of mass, the solar pressure torque

$$T_{solar} = 0.235 \times 10^{-4} \text{ ft lb.}$$

For an effective gravity gradient stabilization, it is assumed that the gravity gradient torque should be about an order of magnitude higher than the solar radiation torque.

$$T_g = 0.235 \times 10^{-3} \text{ ft lb.}$$

For an orientation error of ±0.1° maximum for the regional satellites,

$$\theta = 0.1^\circ = 1/573 \text{ radian}$$

$$\sin 2\theta \cong 2\theta = 2/573.$$

For synchronous orbit,

$$\omega_o = \text{one revolution per day}$$

$$= \frac{2\pi}{24 \times 3600} \text{ rad per second}$$

$$0.235 \times 10^{-3} = \frac{3}{2} m_B \left(\frac{2\pi}{24 \times 3600}\right)^2 L^2 (2/573).$$

Therefore,

$$m_B L^2 = 8.37 \times 10^6.$$

For a satellite weight of 1600 pounds,

$$m_A + m_B = \frac{1600}{32.2} = 50 \text{ slugs.}$$

Assuming one can detach 10 percent of the mass to the tip of the boom, $m_B = 5$ slugs. Therefore,

$$L^2 = 1.674 \times 10^6 \text{ ft}^2$$

or a 1300 foot boom is required. For an orientation error of ±0.5° maximum for the international satellites, a 580 foot boom is required.

Propellant Weight for Attitude Control

Spin Stabilization

The part of the vehicle that could be spun is the control part containing the batteries, propellant tanks, thrusters, logic units, etc., which weigh a total of 300 lb. If spin stabilization is used, this central part would be cylindrical with a radius of two feet. For a uniformly distributed mass,

$$I_{spinner} = \frac{1}{2} mR^2 = 18.65 \text{ slug ft}^2.$$

If the total mass could be concentrated at the rim,

$$I_{spinner} = mR^2 = 37.3 \text{ slug ft}^2.$$

It is reasonable to assume that the mass could be distributed to obtain

$$I_{spinner} = 25 \text{ slug ft}^2.$$

Assuming a spinner rpm of 160,

$$\omega = \frac{320}{60} = 16.7 \text{ rad per second.}$$

Solar pressure torques are

$$T_{solar} = 0.235 \times 10^{-4} \text{ ft lb.}$$

By locating the thrusters on the rim of the two foot radius spinner, the total five year impulse is

$$S = \frac{0.235 \times 10^{-4}}{2} \times 5 \times 365 \times 24 \times 3600 = 1850 \text{ lb sec.}$$

For cold gas thrusters, assume a specific impulse of 200 seconds.

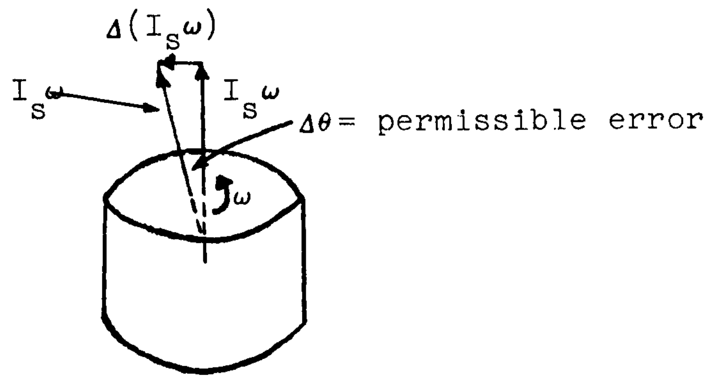
$$\text{Weight of gas} = \frac{1850}{200} = 9.25 \text{ lb.}$$

For ion thrusters, assume a specific impulse of 2000 sec.

$$\text{Weight of cesium} = 0.925 \text{ lb.}$$

<u>Time for Error Correction</u>

Referring to Fig. J-2,



$$\Delta(I_s\omega) = I_s\omega\sin\Delta\theta = I_s\omega\Delta\theta$$

$$\Delta(I_s\omega) = T_{solar}\ t$$

Fig. J-2 SPIN STABILIZATION.

where t is the time for the permissible error to occur under the influence of the solar pressure torque or

$$t = \frac{I_s\omega\Delta\theta}{T_{solar}} .$$

For a permissible error in orientation of ±0.1°,

$$t = \frac{(25)\,(16.7)\left\{\frac{0.1}{57.3}\right\}}{0.235 \times 10^{-4}} \begin{array}{l} = 31{,}000 \text{ sec} \\ = 8.6 \text{ hr.} \end{array}$$

For a permissible error in orientation of ±0.5°,

$$t = 155{,}000 \text{ sec} = 43 \text{ hr.}$$

Active Stabilization

Time for Error Correction

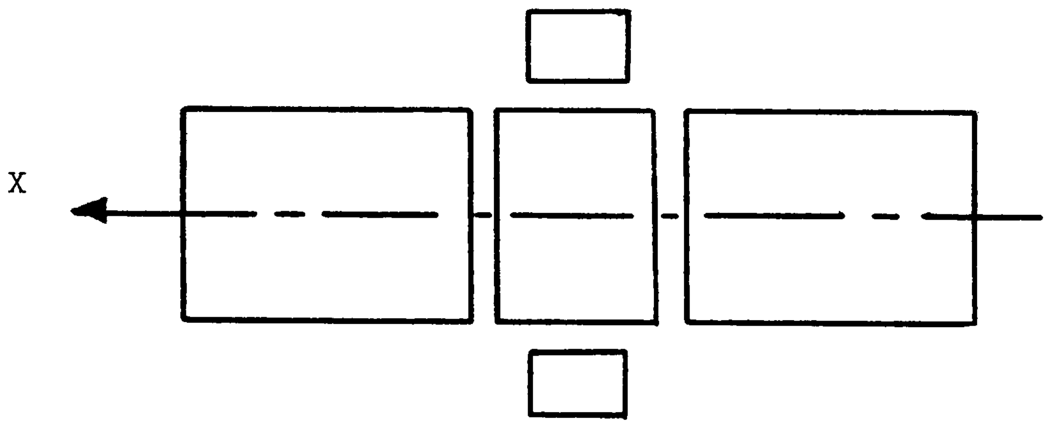


Fig. J-3 ACTIVE STABILIZATION.

Assume:

$$I_{x-x} = 1000 \text{ slug ft}^2$$

$$T_{solar} = I_{x-x}\alpha$$

$$0.235 \times 10^{-4} = 1000\,\alpha$$

$$\alpha = 0.235 \times 10^{-7} \text{ rad per sec}^2.$$

For a $\pm 0.1^\circ$ maximum error,

$$\Delta\theta = \frac{1}{2}\alpha t^2$$

$$t^2 = 2 \times \frac{0.1^\circ}{57.3} \times \frac{1}{0.235 \times 10^{-7}} = 15 \times 10^4$$

or

$$t = 387.5 \text{ sec} = 6.5 \text{ min.}$$

For a $\pm$ 0.5° maximum error,

$$t = 865 \text{ sec} = 14.5 \text{ min.}$$

Attitude Control by Mass Expulsion

Calculations for the worst propellant demand

$$w_p/w = \frac{16\ r^2\ \theta}{gI_{sp}\ L\ t_c} \quad \text{(weight fraction per cycle)}$$

where

w_p = propellant weight (lb)
w = spacecraft weight (lb)
r = radius of gyration (ft)
L = radius of jet centerline from c.g. of spacecraft
θ = amplitude of motion (maximum error in radians)
t_c = period of motion (sec)
I_{sp} = specific thrust (sec)
w = 1600 lb.

Mass of spacecraft $\cong$ 50 slugs

$$r^2 = \frac{1000}{50} = 20$$

L_{min} = 0.4 ft for ion thrusters vectored
θ = either 0.1° or 0.5°
t_c = either 1550 or 3460 sec
I_{sp} = 2000 sec for 150 μlb ion thrusters.

Therefore,

$$w_p/w = \frac{(16)(20)(\theta)}{(32.2)(2000)(0.4)t_c} = \frac{\theta}{80t_c} \text{ per cycle.}$$

For 0.1° error,

$$w_p/w = 14 \times 10^{-9} \text{ per cycle}$$

For the five year lifetime,

$$w_p/w = 1.425 \times 10^{-3}$$

or

$$w = 2.3 \text{ lb}$$

Propellant for attitude control about the three axes is about seven lb of cesium for an error of ±0.1° or ±0.5°.

Power Requirements

Each 150 μlb ion thruster requires 40 W of power or a total of 240 W for the six microthrusters.

Power requirement on the order of 180 W per mlb of thrust is to be expected. For the 150 μlb thrusters, power requirements of 30 W per thruster is reasonable or a total of 180 W.

Appendix K

FEASIBILITY OF SOLAR "TRIM TABS" FOR SPACECRAFT STABILIZATION

Since the primary disturbance torques on the spacecraft will be those due to solar radiation pressure, it is meaningful to consider the possibility of using this radiation to help stabilize the vehicle. There are two primary ways to use this radiation pressure.

(1) Use "solar trim tabs" on the solar panel arrays to cause a restoring torque; and

(2) Cant the panels such that a restoring torque is developed if they do not face directly into the sun.

These two cases are now analyzed.

<u>CASE I: (Trim Tabs)</u>

Assume that the tabs reflect spectularly. α is the error angle. See Fig. K-1.

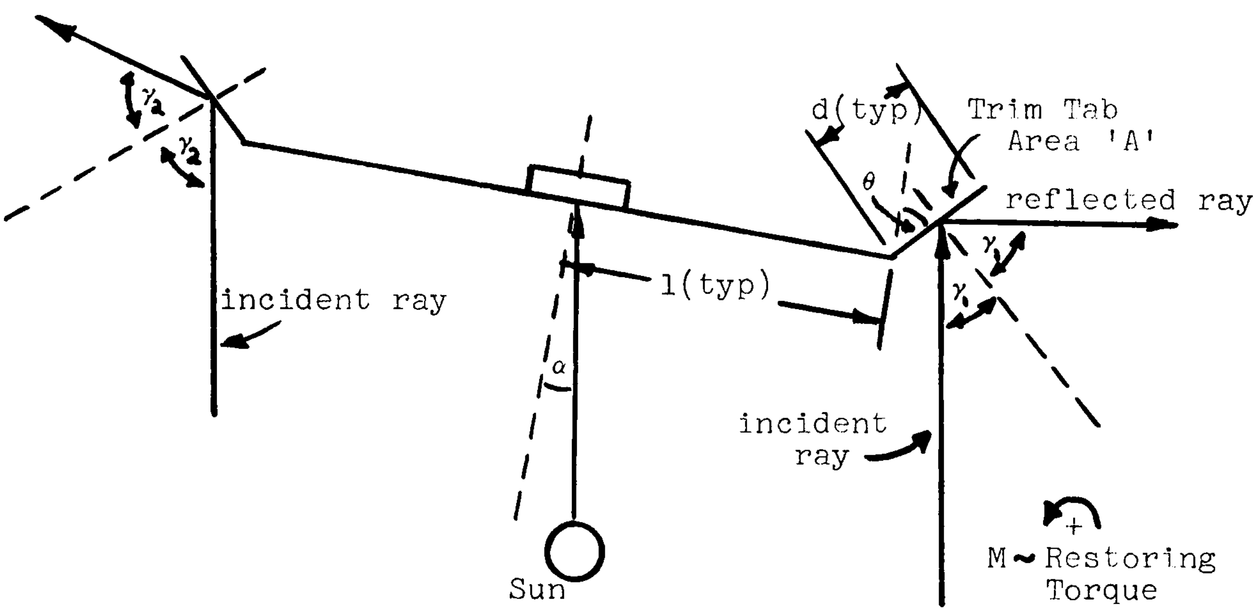


Fig. K-1. TRIM TAB ORIENTATION.

The torque 'M' developed by the radiation pressure on the trim tabs is

$$M = (2pA \cos \gamma_1) \cos\left(\frac{\pi}{2} - \theta\right)l - (2pA \cos \gamma_2) \cos\left(\frac{\pi}{2} - \theta\right)l \ .$$

From the geometry,

$$\gamma_1 = \frac{\pi}{2} - (\theta + \alpha)$$

$$\gamma_2 = \frac{\pi}{2} - (\theta - \alpha) \; .$$

Hence,

$$M = 2pAl \sin\theta[\sin(\theta + \alpha) - \sin(\theta - \alpha)]$$

$$\left.\begin{aligned} M &= 2pAl \sin 2\theta \sin\alpha \\ \frac{dM}{d\alpha} = M_\alpha &= 2pAl \sin 2\theta \cos\alpha \end{aligned}\right\} \qquad (K.1)$$

If α is a small angle, i.e., $\alpha \ll 1$,

$$\left.\begin{aligned} M &\cong (2pAl \sin 2\theta)\alpha \\ M_\alpha &\cong 2pAl \sin 2\theta \end{aligned}\right\} \qquad (K.2)$$

From Eqs. (K.1) and (K.2) it is obvious that the optimum value of θ is $\theta = \pi/4$, hence $\sin 2\theta = 1$. Numerically,

$$p \cong 10^{-7} \text{ lb/ft}^2$$

$$l \cong 20 \text{ ft}$$

$$A \cong 12d \text{ ft}^2 \; .$$

Using these figures and converting M into torque/degree:

$$M_\alpha = 8.35 \times 10^{-7} \text{ ft-lb/deg} = 0.835d \; \mu\text{ft-lb/deg} \; .$$

Hence, d must be extremely large to compensate for any possible disturbance. Note that there are approximately 500 ft^2 area exposed to the sun. A separation of 1/5 inch between the center of mass and the center of pressure causes a torque of one micro-foot-pound.

CASE II: Canted Panel

To evaluate the second configuration, assume the entire panels are tilted back by an angle ϕ. Equation (K.1) still holds if (a) replace θ

with $(\pi/2 - \phi)$, (b) divide by two, since the panels can be assumed to be reflecting diffusely. See Fig. K-2.

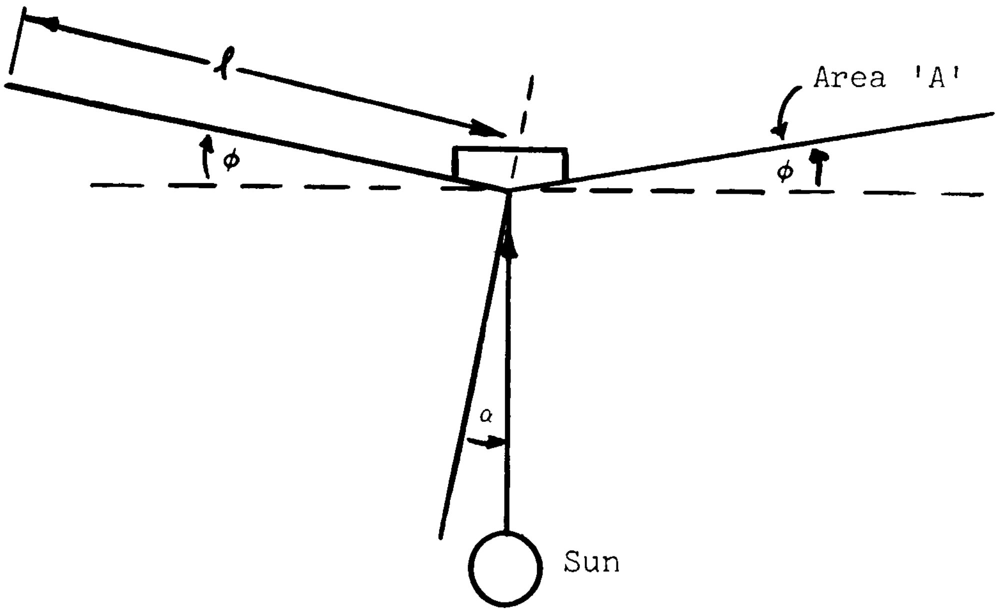


Fig. K-2. CANTED PANELS.

Then,

$$M = pAl \sin 2\phi \sin \alpha$$

$$M_\alpha = pAl \sin 2\phi \cos \alpha$$

or, if $\alpha \ll 1$,

$$M = (pAl \sin 2\phi)\alpha$$

$$M_\alpha = pAl \sin 2\phi \ .$$

For this case,

$$p \approx 10^{-7} \text{ lb/ft}^2$$

$$A \approx 192 \text{ ft}^2$$

$$l \approx 10.8 \text{ ft.}$$

Then,

$$M_\alpha = 414 \sin 2\phi \ \mu\text{ft-lb/rad} = 7.21 \sin 2\phi \ \mu\text{ft-lb/deg} \ .$$

Obviously, the optimum value of ϕ is $\pi/4$ radians, but in this case, the angle ϕ also causes a reduction in power available to the spacecraft. These values are shown in Table K-1 for various values of ϕ.

Table K-1

RESTORING TORQUE AND POWER LOSS FOR VARIOUS PANEL ANGLES

ϕ (deg)	M(μft-lb/deg)	Percent Total Power
5	1.25	99.5
10	2.48	98.5
20	4.63	94.0
45	7.21	70.7

The results of this study indicate that this method of passive control can be effective on this rather large vehicle only if the spacecraft is extremely well balanced. These completely passive schemes appear to be, at best, rather marginal in their ability to compensate for any disturbance torques. The possibility still exists for incorporating these same concepts into a more complex control, utilizing movable members. Here, only the first case (trim tabs) appears feasible. To evaluate the feasibility of this system, assume the "best" system to be one which moves one trim tab squarely into the sun and removes the second panel from the sun's view entirely. If one does this, the torque

$$M = 2pAl$$

is obtained, independent of the angle α. For the numbers presented in CASE I, this gives a restoring torque of

$$M = 48d \ \mu\text{ft-lb}$$

This torque is substantially higher than previous numbers and is of the magnitude needed for proper control.

Appendix L

STATISTICAL ANALYSIS OF MULTIPLE LAUNCH

Given that a system of satellites is to be injected into orbit using the multiple launch concept, the question arises as to whether the probability of a launch failure could offset the direct cost advantage found in the multiple launch concept. One can analyze this from a statistical standpoint as follows:

Define "success" to mean that m successful launches have been made to place the total satellite system in orbit.

Let n be the number of launches required to obtain m successful launches $(n \geq m)$.

P is the probability of a successful launch (constant).

$Q = 1-P$ is the probability of failure.

Finally, let c_n be the cost of success on the n^{th} launch.

With these assumptions, the probability of success on the n^{th} launch is

$$P_n = C \binom{n-1}{m-1} P^m Q^{n-m}$$

$$P_n = \frac{(n-1)!}{(n-m)!(m-1)!} P^m Q^{n-m} \tag{L-1}$$

(This is simply the binomial probability for $m-1$ successes in $n-1$ launches with the m^{th} success on the n^{th} launch.)

One can then define the statistical parameters:

<u>Expected Cost</u>

$$\overline{c_n} = \sum_{n=m}^{\infty} c_n P_n$$

Second Moment of Cost

$$\overline{c_n^2} = \sum_{n=m}^{\infty} c_n^2 P_n \tag{L-2}$$

Standard Deviation

$$\sigma_n = \left[\overline{c_n^2} - \overline{c_n}^2\right]^{1/2} \tag{L-3}$$

If one assumes that each launch costs an amount c_m, then c_n becomes

$$c_n = nc_m \tag{L-4}$$

and Eqs. (L-2) and (L-3) become

$$\overline{c_n} = \frac{c_m P^m}{(m-1)!} \sum_{n=m}^{\infty} \frac{n!}{(n-m)!} Q^{n-m} \tag{L-5}$$

$$\overline{c_n^2} = \frac{c_m^2 P^m}{(m-1)!} \sum_{n=m}^{\infty} n \frac{n!}{(n-m)!} Q^{n-m} \tag{L-6}$$

By using the series expansion for $(1-Q)^{-1}$ and successively differentiating, it can be shown that

$$\sum_{n-m}^{\infty} \frac{n!}{(n-m)!} Q^{n-m} = \frac{m!}{(1-Q)^{m+1}}$$

Hence, Eq. (L-5) becomes

$$\overline{c_n} = \frac{mCm}{P} \tag{L-7}$$

Unfortunately, the second series has no simple analytic form and must be evaluated numerically (see Table L-1).

The implication of Eq. (L-4) is that the "expected cost" is the system cost with no failures divided by the probability of success of a single booster. If all boosters have an equal probability of success, this means the "best" system (in this restricted interpretation of "best") is the one which has the lowest cost, assuming no failures. However, because we are dealing with discrete systems, this statistic can be misleading.

For example, assume one can have a single launch for $100 million or four launches for $30 million each. This results in a $20 million savings if there are no launch failures. If each booster has a P = 0.90, then the expected costs are $111 million and $133 million, respectively. Actually, one time in ten, the first system will cost at least $200 million, whereas the second system will cost as much as $200 million.

The expensive system ($120 million) has a 35 percent chance of at least one failure (thereby boosting cost by at least $30 million), whereas the multiple launch system has only a 10 percent chance of costing an additional $100 million. All this points out that we are dealing with real dollars and not statistical averages, and a decision cannot be made on this expected value criteria alone, but that other facts should enter into the decision, e.g., the arguments previously noted as well as the question of spare spacecraft, time to put the system in orbit, etc.

The crux of the problem is that the multiple launch concept can save considerable dollars but at a certain risk of losing a very large amount of money in the event of a launch failure. Using single launches, one expects to spend more money but a launch failure will no longer cause a catastrophic increase in system cost. It appears obvious that the multiple launch concept should be used only if the launch vehicle has a well proven reliability and if the launch checkout system is comparable to that now given only to manned missions.

Table L-1

EXPECTED COST AND STANDARD DEVIATION/UNIT LAUNCH COST*

No. of Launches	P	$\bar{C}_n$	σ_n
1	.99	1.01	0.101
	.95	1.05	0.235
	.90	1.11	0.351
	.85	1.18	0.456
2	.99	2.02	0.143
	.95	2.10	0.333
	.90	2.22	0.497
	.85	2.35	0.644
3	.99	3.03	0.175
	.95	3.16	0.407
	.90	3.33	0.609
	.85	3.53	0.789
4	.99	4.04	0.202
	.95	4.21	0.471
	.90	4.44	0.703
	.85	4.70	0.911
5	.99	5.05	0.226
	.99	5.26	0.526
	.90	5.55	0.785
	.85	5.88	1.019
6	.99	6.06	0.247
	.95	6.32	0.577
	.90	6.66	0.861
	.85	7.06	1.112

*All numbers generated by computer evaluation of infinite series.

Appendix M

SPECIFICATIONS OF APOGEE ROCKETS

Liquid Fuel

Assume N_2O_4 = MMH[1] propellant of specific impulse I_{sp} = 340 sec and $W_b/W_e = 9$, where W_p = weight of propellant and W_e = weight of empty rocket case.

There is an optimum value for the plane change to be carried out at the perigee of the transfer orbit, but a fairly detailed knowledge of the system is required to evaluate this. As a first approximation, it is assumed that the entire plane change is made at the apogee by yawing the Centaur the correct amount before firing the injection rocket. Then, by [2]

Total velocity requirement to attain apogee = 33,561 ft/sec

Velocity increment required at apogee, ΔV = 6,009 ft/sec

and

$$\Delta V = (I_{sp} g) \operatorname{Log}_e \frac{W_e + W_p + W}{W_e + W} \qquad (M.1)$$

According to predicted values, the Titan IIIC-Centaur-Kick vehicle should be capable of giving a payload of about 16,000 lb, a velocity of 33,561 ft/sec.

The jet velocity for the injection rocket is given by

$$V_j = I_{sp} g = 340 \times 32.2$$
$$= 10,920 \text{ ft/sec}$$

Substitution of values in Eq. (M.1) gives:

$$600\ q = 10,920 \ln\left(\frac{16000}{1600 - W_p}\right)$$

or

$$\frac{16000}{16000 - W_p} = e^{\frac{6,009}{10,920}} = 1.734$$

and

$$W_p = \frac{0.734}{1.734} \cdot 16,000 = 6,760 \text{ lb}$$

$$W_e = \frac{W_p}{q} = 750 \text{ lb} .$$

The total weight of the injection rocket = 7510 lb. Therefore, the weight of useful payload injected into orbit = 8490 lb.

Assuming a constant jet velocity during burning the total impulse required is given by:

$$\frac{W_p}{q} \cdot V_j = \left(\frac{6760}{32.2}\right)(10,920) = 2.3 \times 10^6 \text{ lb sec}$$

For a burn time of 30 sec, the thrust would be 77,000 lb and the maximum longitudinal acceleration 77,000/9240 g = 8.3 g which is acceptable.

The dimensions of the space available for the injection rocket are shown in Fig. M-1.

In the case of a single launch, the Atlas-Centaur-Kick vehicle should be capable of giving a payload of 4000 lb [3] and a velocity of 33,561 ft/sec. In this case, the injection rocket would weigh 1880 lb of which 1690 lb would be fuel. The maximum weight of payload injected, therefore, would be 2310 lb (useful payload 2120 lb). Also:

$$\text{Total impulse} = \frac{1690}{32.2} \cdot 10,920 = 0.574 \times 10^6 \text{ lb sec}$$

For a burn time of 30 sec, the thrust would be 19,150 lb and the maximum longitudinal acceleration 8.3 g as before. The space available for this injection rocket is about the same as that for the multiple launch system.

Solid Fuel

For solid propellant rockets, $W_i/W_a = 0.405$,[2] where W_i = weight of payload injected, W_a = weight at apogee. In the case of the

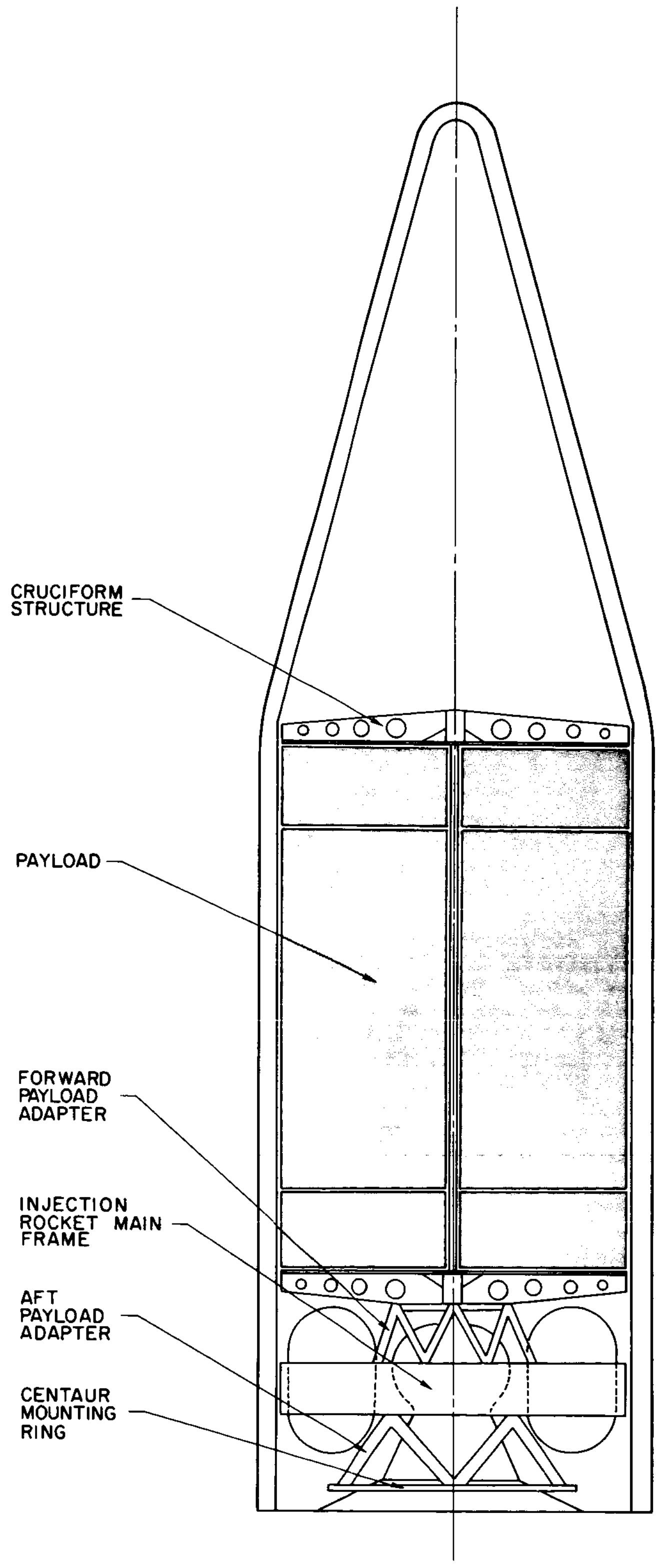


Fig. M-1 CENTAUR SHROUD.

Titan-Centaur-Kick vehicle $W_a = 16,000$ lb, $W_i = 6480$ lb, and $W_p = (.047)(W_a) = 7520$ lb. For a specific impulse of 280 sec,

$$\text{Total impulse} = 7520 \cdot 280 = 2.11 \times 10^6 \text{ lb sec},$$

Longitudinal thrust $\cong$ 53,000 lb for a burn time of 40 sec,

Maximum longitudinal acceleration = 6.2 g.

This performance could be accomplished using a spherical motor case of approximately 60 inches diameter and a nozzle approximately 40 inches diameter with an expansion ratio of about 21. A more convenient size would be obtained using a higher expansion ratio and a higher specific impulse.

For the single launch:

$$W_a = 4000 \text{ lb} \quad W_p = 1880 \text{ lb} \quad W_i = 1620 \text{ lb}$$

$$\text{Total impulse} = 1880 \cdot 280 = 0.526 \times 10^6 \text{ lb sec}$$

The TE-M-364-3 motor used in connection with the Delta booster almost gives this performance. An improved version would fulfill all of the requirements, especially if the optimum plane change is made at the perigee of the transfer ellipse.

References

[1] "Rocket Propellant Characteristics," T.R.W. Space Data, 1967.

[2] Bollay, Dr. William, "Fundamentals of Rocket Boosters for Educational TV."

[3] "OSSA Launch Vehicle Estimating Factors."

Appendix N
SPACE ENERGY SYSTEMS

Evaluation of Nuclear Power

"Nuclear power" for satellite application has the connotation of either an isotope or reactor system. Nuclear power offers several attractive potential advantages: high energy content per unit mass, reliability and compactness of system. A satellite with a nuclear power supply would require little or no articulation, no orientation, and the power supply would operate through the eclipse. Shielding requirements, materials limitations, and low energy conversion efficiencies prevent the full attainment of these advantages at this time, and development is continuing in the government SNAP series, the General Electric STAR system, and other efforts. The nuclear isotope systems are now competitive in the low power applications and the nuclear reactor systems are expected to become competitive for the larger power levels.

Nuclear power supplies were evaluated for the SAINT communication satellite. The basic conclusions are reported in Section 10.1.

State of the Art

Radioisotope Energy Sources

Radioisotope power sources of 30 W and less are supplying energy to satellites in space. Larger sizes are in various stages of development. SNAP-29 is a 500 W electrical supply for space. It will be useful for lunar missions to supply power during the lunar night and possibly for manned missions. It is to be powered by a short-lived isotope which will make it unsuitable for communication satellite power. To use a long-lived isotope will require a new development effort, because the long-lived isotopes produce much lower power per unit volume.

NASA Lewis Research Center is developing a 2-10 kwe turbo-generator unit operating on the Brayton cycle. Both a solar collector and an isotope energy source are being developed. The isotope source is scheduled for test in 1971; flight-rated units will require a few years of further development and testing.

Reactor Energy Sources

One reactor-thermoelectric power supply has been flown. SNAP-10A was launched, started in space, and was operating successfully when a spacecraft failure on the 44th day interrupted communications with the satellite. The nuclear power unit appears to have been a success.

Development of reactor power supplies is being carried on at a low level of support, primarily due to lack of definite mission requirements. Development to increase efficiency and to reduce weight, and further testing will require several years before flight-rated supplies are available.

Comparison

The high cost of nuclear research and development and the lack of specific missions requiring nuclear energy has resulted in a small number of power units produced for test. When the units which have been launched are compared to photovoltaic solar cell panels at a similar stage of development, nuclear power compares favorably.

Comparison of Space Power Supplies
at an Early Stage of Development

Case	Type Power Supply	Power Output (W)	W/lb
OSO	Oriented solar cells	31	0.86
Relay	Solar cells	35	0.65
Tiros	Solar cells	51	0.79
Transit 4A, 4B	SNAP-3B(mod)	2.7	0.59
Transit (1963)	SNAP-9A	25	0.93

The reason solar cells do not appear better is that for the complete system, a secondary battery power supply is necessary. For the satellite in this project, the solar panels weigh 250 lb, and the batteries 300, a total of 550 lb, plus controls, power conditioning, wiring, etc. A comparable isotope power supply would weigh approximately 1500 lb (more optimistic prognosticators would estimate 1000 lb), almost a factor of 3 (or 2) times as heavy as the solar panel secondary battery supply. A nuclear reactor supply would be expected to weigh about the same.

Competitive Position

For an orbiting satellite requiring secondary power, nuclear reactor units will increase in weight with increasing output at a slower rate than will solar panel secondary battery supplies. A crossover should occur in the region of 10-100 kW electrical. If secondary batteries are not needed, or if only a small fraction of the total power is needed from secondary batteries, nuclear supplies may be heavier than solar panels in all sizes.

It is technically feasible to unfurl (or unfold) large areas of solar arrays, and further improvement in power per unit weight can be expected. As these solar arrays grow larger, the problems of packaging for launch, articulating in space, ground testing, and maintaining reliability become larger. At some point, nuclear power may be attractive even with its weight penalty.

In summary, considering the factors in the paragraph above, nuclear space power supplies, when developed, will be competitive with other power sources in all sizes. In and above the 10-100 kwe range, nuclear reactors may have a weight advantage also.

Problem Areas

Radiation-Isotope Sources

Some alpha-emitting isotopes have little gamma activity associated with them. The packaging materials supply most of the shielding needed. These are ideal energy sources and are being developed in sizes to 10 kwe, and could be made larger. The disadvantages of the two principal isotopes are that Pu-238 is too expensive and in short supply, and Po-210 has too short a half-life for many applications.

Beta emitters are being used for terrestrial supplies. Because of the Bremsstrahlung gamma radiation, shielding is needed. A study considering the isotope to be a point source pessimistically predicts 16-20 cm lead shielding needed for a one kwe source; when the self-shielding of the isotope is considered, the shielding reduces to 2-3 cm. Considering that in space, shadow shielding is adequate, and radiation can be reduced by extending the power supply on a boom, the radiation problem is not particularly serious.

On the launch pad, shadow shielding is not sufficient, and isotopes cannot be turned off. Shielding of equipment and personnel before launch can be a serious problem.

Gamma-emitting isotopes can be used as energy sources. To absorb the energy the source must not be too small (over 1 kwe), and must be compact. This will require a circulating coolant to remove the heat. The radiation problem is similar to those encountered with beta emitters.

Radiation - Reactor Sources

Nuclear reactors present the same problems in space as isotopes. Shadow shielding is needed; remoteness from the payload is desired. The activation of the coolant is an additional design or shielding problem, but a much smaller problem than the direct radiation.

Launch Safety

In case of launch pad fire or explosion, the presence of radioactive material must not create a severe hazard. All isotope sources are designed and tested to withstand these catastrophes.

Nuclear reactors produce less launch pad hazard because there is a relatively small amount of radio activity present until the reactor has operated for a period of time. Reactors are started in space, after launch has been successful.

Re-entry Safety

In cases of failure to attain orbit, or low orbit, the satellite may return to earth. It may burn up in the upper atmosphere, or return intact. If a reactor has not operated, burning up in the upper atmosphere does not create a serious hazard. Further evaluation of this situation is required.

A nuclear reactor power supply, after operating for a period of many months, will contain a large quantity of radioisotopes. The current method of handling this is to place the satellite in a high orbit which will not decay for several hundred years. Whether this is a satisfactory procedure for the future must also be examined.

Radioisotope sources in larger sizes (over 1 kwe) may produce a significant hazard if dispersed in the atmosphere. These sources are being designed to re-enter intact, which adds to their weight.

Forecasts

Radioisotope over 1 kwe. Now under development; available about 1975.

Reactor -- thermoelectric. One test flown; further development and testing required for a practical system. A system may be ready about 5 years after a need for such a system is established.

Reactor -- thermionic. This requires further improvement in thermionic converters, and integrating them into a nuclear system. A practical system probably will not be ready until the 1980's.

Nuclear -- turbo-generator. These systems can be built. Reliability and lifetime are problems. Corrosion, material deposition, bearing reliability, and integrity of fluid systems are problem areas. One year lifetime might be achieved in the late 70s if development continues.

Appendix O

POSSIBLE SATELLITE GEOMETRIES FOR SOLAR CELL UTILIZATION

After the decision that the primary electric power is to be supplied by solar cells, the solar cell panel configuration must be selected. Three configurations appear feasible and are: sun-oriented flat panel, cylindrical panel, and triform flat panel. These are shown in Fig. O-1. Each has its own weight, reliability, deployment, and cost advantages.

The cylindrical panel and triform panel require no orientation toward the sun. They only require that the satellite antenna point toward the earth. Per unit panel area, the cylindrical panel generates 36.2 percent of the power generated by the flat panel. The triform generates 28.1-32.4 percent. The values consider solar cell geometry, thermal efficiency, and radiation damage effects. Assuming that the backing is the same for all of the panels, the weight analysis for the 3-kW power supplies is

	Flat Panel	Cylindrical Panel	Triform Panel
Panel Weight	136 kg	360 kg	320 kg
Drive Motor, bearing, and sun sensors (needed for sun orientation)	23 kg	---	---
Totals	159 kg	360 kg	320 kg

The reliability of systems is difficult to assess. Obviously, the cylindrical and triform arrays which require no sun orientation are most reliable. The reliability of the additional components needed for the flat panel array must be thoroughly analyzed. Deployment of the power system favors use of the flat panel array as the deployment complexity can be equated to the total panel area.

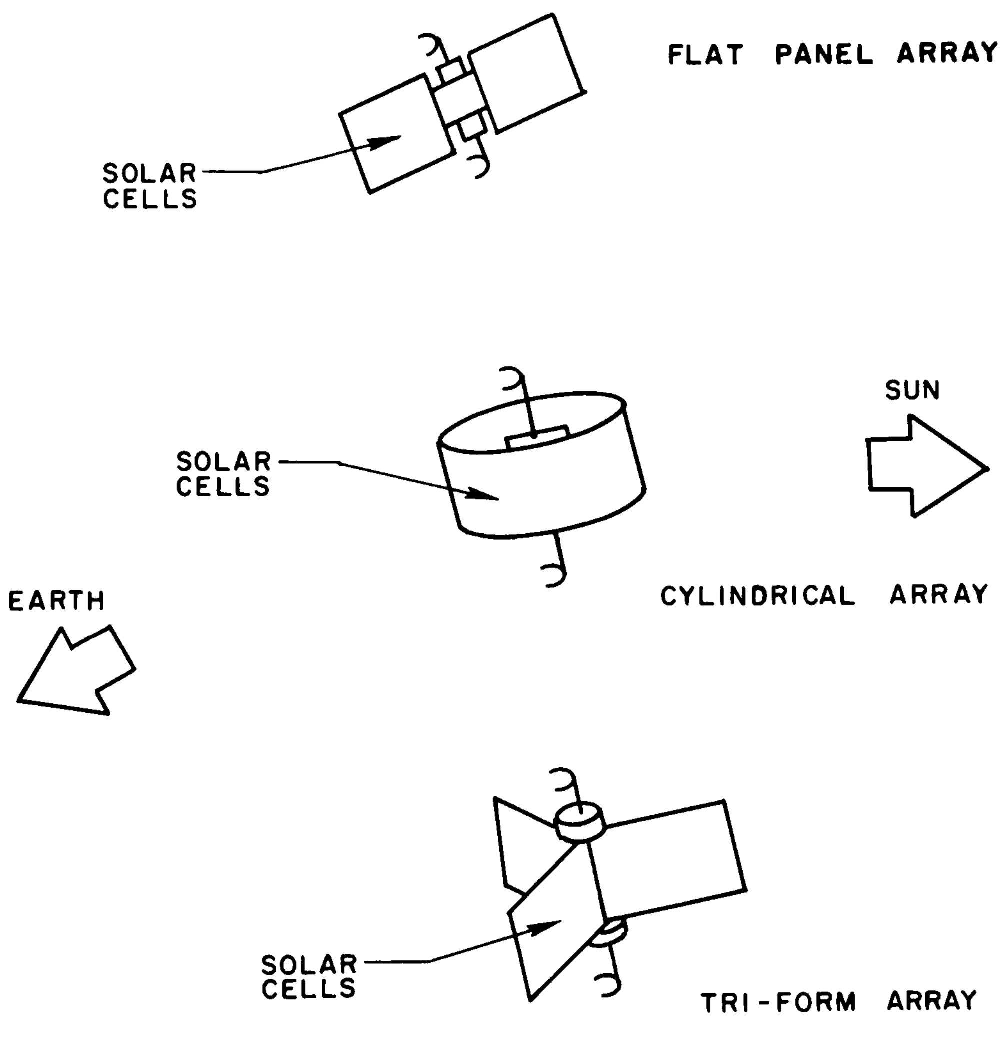


Fig. 0-1 FEASIBLE SATELLITE GEOMETRIES FOR SOLAR CELL UTILIZATION.

Appendix P

GENERAL COMMENTS ON FUTURE SPACE POWER SYSTEM DEVELOPMENT

When considering space electrical power generation systems where life of one year and longer is required, there are at the present time only two feasible energy sources, nuclear and solar.

Nuclear Systems

Nuclear systems consist of two basic types, the radioisotope and the reactor type. Either static or dynamic electrical conversion systems may be used with either the reactor or radioisotope systems. The more common systems are as follows:

Static	Dynamic
Thermoelectric	Rankine Cycle
Thermionic	Brayton Cycle
	Sterling Cycle

Solar Systems

Solar systems extract energy from the sun and are also of two basic types: The static systems and the dynamic systems. These systems were not described in detail because of the large volume of literature which adequately details the various systems. Again, the more common, and the solar systems with the most potential at the present time are as presented below:

Static	Dynamic
Solar Cell	Rankine Cycle
Thermoelectric	Brayton Cycle
Thermionic	Sterling Cycle

Recommendations for Future Development

Component Elements

Due to the inherent reliability of static systems because of the lack of moving parts, solar cells, thermoelectric elements, and thermionic elements will be utilized in space secondary power systems of the future. Each of these elements has a unique characteristic which makes it more applicable for a particular mission or power range than the others. Therefore, it is recommended that all three of the static system components be developed further to improve their efficiency, reliability, and durability.

Solar cells have been the main source of secondary power for most satellites that have been launched where life of over several weeks was expected. In fact, solar cells have been chosen for the mission of the satellites of this projects. Since the original use of solar cells in space, the basic efficiency has been increased from 5 to 11 percent and material improvements have been made to improve the resistance to particle radiation. However, since there will be more future satellites launched with solar cells than any other conversion system, a major development effort should be carried out in this area.

Radioisotopes

At the small wattage level, radioisotope systems using either thermoelectric or thermionic units can have considerable utilization for space and earth applications. Some of the earth applications may include remote, unatten ed weather stations and ocean buoys. Therefore, it is recommended that radioisotope systems in the small wattage levels using static electrical conversion systems be continued to be developed to a greater degree of efficiency and reliability.

Nuclear Reactor Systems

At the present time, due to the progress that has been made with other types of electrical space power generation systems for the lower wattage levels, it appears that the nuclear reactor systems should be limited to the high kilowatt levels. Hence it is recommended that development of this type of system, the nuclear reactor, with its attendant radiation problems, be limited to those sizes where other systems will find difficulty because of their size. This would seem to be above 100 kW of useful power.

Dynamic Solar Systems

In the range of 10 to 30 or more kilowatts, the dynamic solar systems can show certain advantages of weight and size over the static systems. In addition, these systems can be designed to eliminate the batteries which are required for the solar cell systems for shadow-of-the-earth operations. Therefore, systems development in the Rankine and Brayton Cycle systems should continue, with possibly a re-evaluation made on the Brayton Cycle systems competitively at continuous intervals because of the heavier radiator weight than Rankine Cycle systems.

Appendix Q

ANALYSIS OF HEAT RADIATION TO SPACE

Net heat radiation to cold space is given by the Stefan-Boltzman radiation law

$$\frac{\dot{Q}}{A} = eC\left[\left(\frac{T}{100}\right)^4 - \left(\frac{T_\infty}{100}\right)^4\right] \frac{\text{Btu}}{\text{hr sq ft}}$$

where:

$\dot{Q}$ = radiation per unit time (Btu/hr)

A = projected area perpendicular to sun (sq ft)

e = emissivity of body (dimensionless)

C = 0.172 Btu/hr ft^2 R^4

T = temperature of body ($^{o}R = t^{o}F + 460$)

T_∞ = temperature of space = $0^{o}R$

If the sun radiates to the body, the T_∞ term is replaced by a(130) where

a = absorptivity (dimensionless)

130 = solar radiation constant (130 W/sq ft)

the equation then becomes:

$$\frac{\dot{Q}}{A} = eC\left(\frac{T}{100}\right)^4 - a(130) = \frac{e(.127)}{3.413}\left(\frac{T}{100}\right)^4 - a(130)\ \frac{W}{\text{sq ft}}$$

where 3.413 = 3.413 Btu/W hr.

Plots of this equation for various emissivities and absorptivities are shown in Figs.

The cost of the power system also favors use of the flat panels. The panel cost may be estimated at $500,000/kW for a sun-oriented panel. If the cost per solar cell area is constant, a 3.0-kW flat panel array would cost $1.5 million, a cylindrical array $4.1 million, and a triform array $4.0 million. (It is assumed that an array with solar cells on both sides costs only 50 percent more than a regular panel.) The development cost of the deployment mechanism is difficult to predict, but it should be related to the area and geometry of the panel deployed. An estimate of $100,000 for the flat panel and triform panel and $300,000 for the cylinder is given. Drive motors and slip rings for space applications have caused great concern and a development cost of $1.0 million is predicted.

Upon reviewing the above factors, it was decided to use the sun-oriented flat panel. It has the least weight and cost, and the least deployment difficulty. It is anticipated that its development will yield a system with satisfactory reliability.

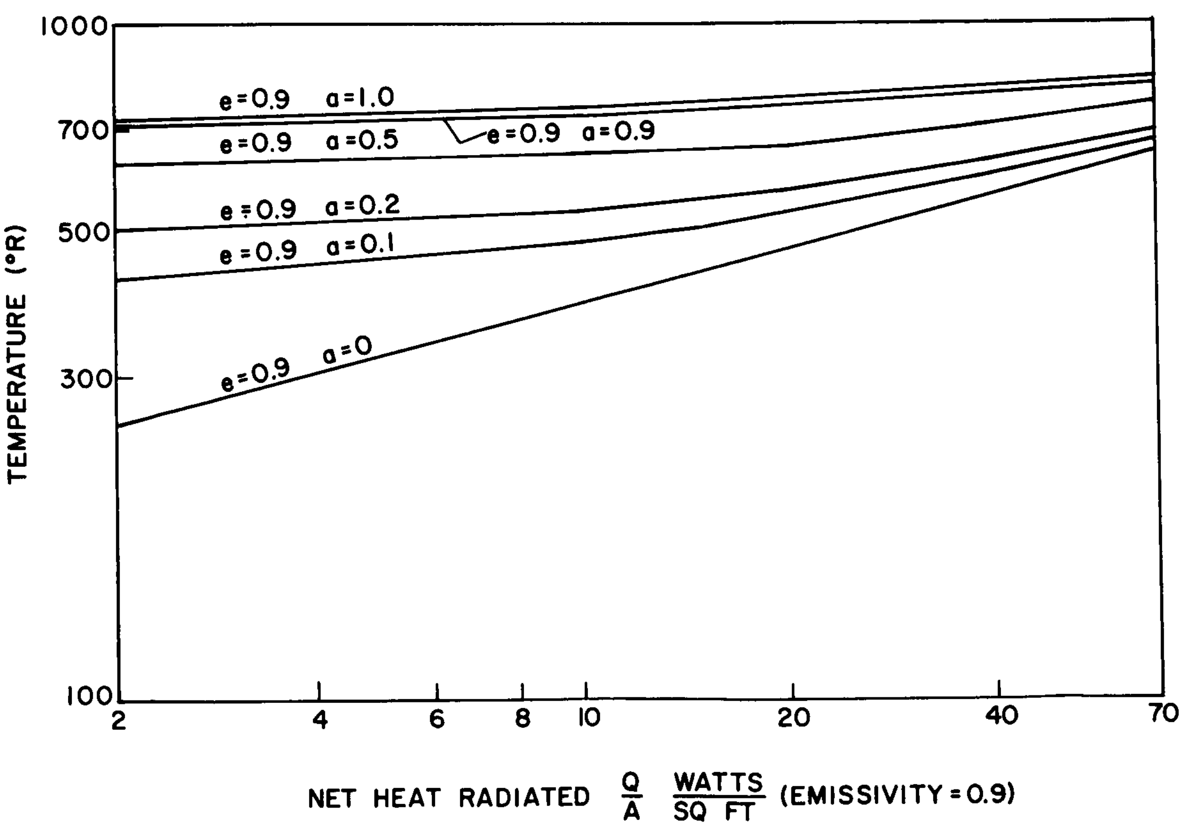


Fig. Q-1.

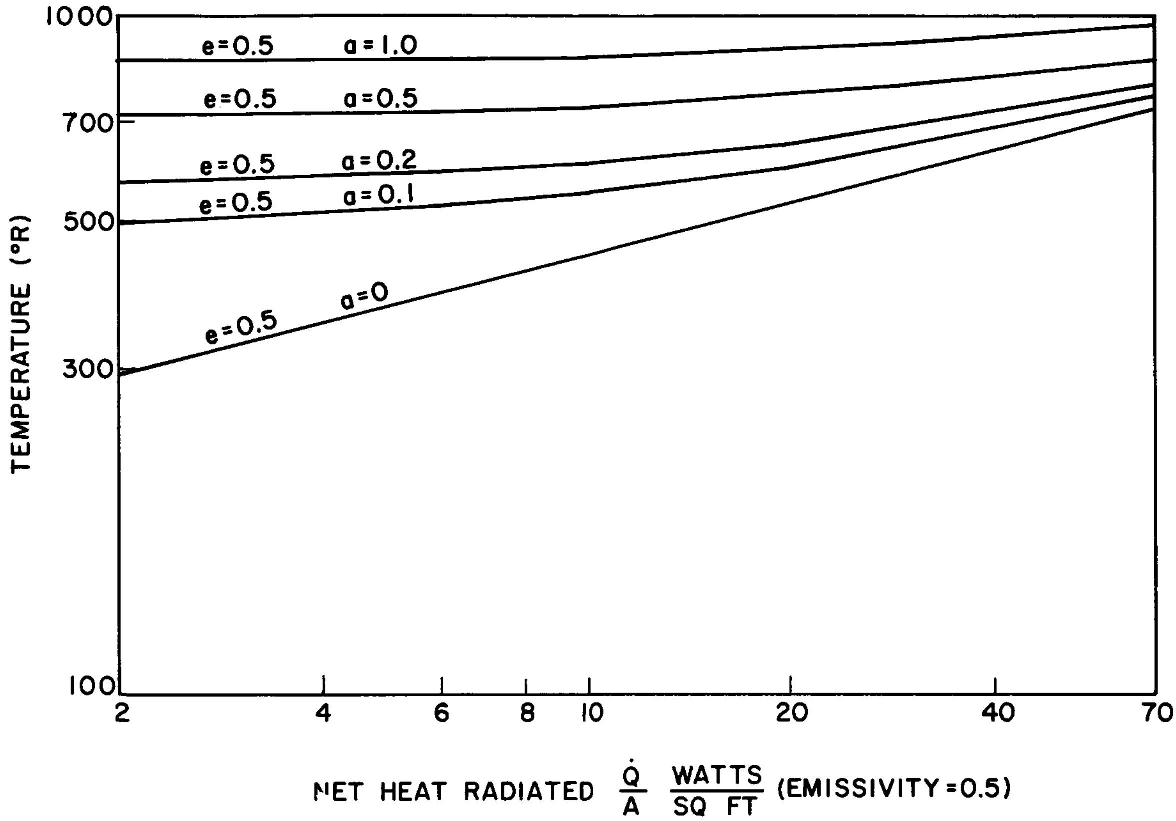

1000
700
500
300
100
TEMPERATURE (°R)
e=0.5 a=1.0
e=0.5 a=0.5
e=0.5 a=0.2
e=0.5 a=0.1
e=0.5 a=0
2
4
6
8
10
20
40
70
NET HEAT RADIATED $\frac{\dot{Q}}{A}$ $\frac{\text{WATTS}}{\text{SQ FT}}$ (EMISSIVITY=0.5)

Fig. Q-2.

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