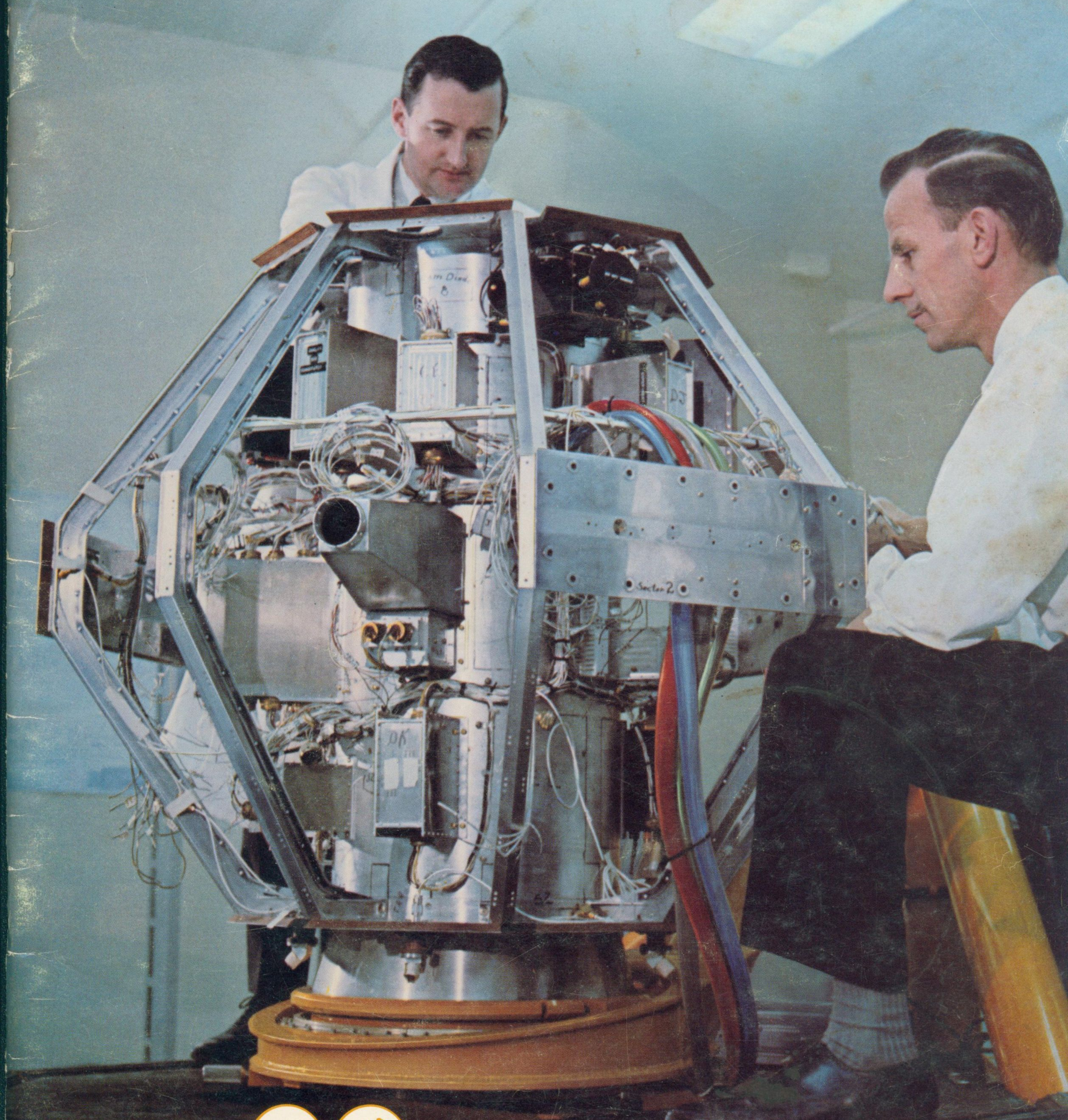
# Canadian Satellites and Space Communications



RCA Victor Company, Ltd.,
1001 Lenoir St., Montreal, Canada

# Canadian Satellites and Space Communications

Our guiding principle of developing specialized skills in areas which complement rather than duplicate those of other parts of RCA has led toward an emphasis on telecommunications. Among the many modern trends in electronics, telecommunications provides great opportunity for the growth and future development of technical products in RCA Victor Company, Ltd.

We had already earned a position of leadership by 1964 in this field through the successful completion of several major microwave systems, including one spanning the continent between Montreal and Vancouver. These systems use equipment developed by our Montreal engineers and engineers of the Broadcast and Communications Products Division in Camden. Since then, our engineers have completed development of a new series of solid-state, RF, and modulator-demodulator equipment for use in both overland and satellite telecommunications systems. This experience, plus our expanded knowledge of antennas and related items, led toward emphasis on the satellite earth stations described in this issue.

Thus, space communications via satellites continues to be a major influence in determining our research and engineering development programs. We have worked closely with Astro-Electronics Division in developing and manufacturing the transponder for RELAY I and II satellites, and since then have taken on the overall job of designing and manufacturing the Canadian ISIS A scientific satellite, a follow-on from the successful ALOUETTE I and II ionospheric sounding satellites. We served as prime contractor in the supply of Canada's first satellite communications earth station located in Nova Scotia; this facility is now engaged in commercial transatlantic service and will later become a part of NASA's advanced technological satellite program.

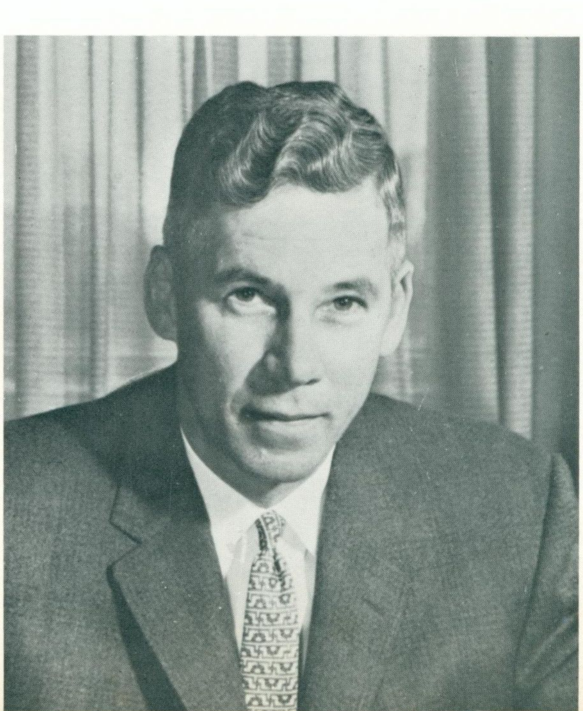
To foster a continued growth of our chosen areas of specialization and achieve a competitive position in world markets, we must further develop our skills in dealing with the many problems and opportunities unfolding the world over. We are grateful for the close association between Montreal engineers and their associates in the U.S.A. An interchange of technical information has, through the years, been of immense benefit to us. Our goal is to steadily increase this interchange and enhance further our contribution to the advancement of technology; in so doing, we strive to play an active role in the growth of RCA's total business.

John D Houlding

*J. D. Houlding, President and Director*
*RCA Victor Company, Ltd.*
*Montreal, Canada*

**OUR COVER**

". . . . . . . . the Engineering Model of the ISIS A spacecraft now undergoing systems testing in the Aerospace Engineering Laboratory, RCA Victor Company, Ltd., Montreal, Canada. Carl Gaul, RCA Victor technician (left) and Dave Lambert, de Havilland Aircraft of Canada Designer, are making adjustments. Construction and test of the flight model has started and will continue until the Spring of 1968. The spacecraft will then be shipped to Goddard Space Flight Center, Washington, D.C., for environmental tests by NASA and launch from the Western Test Range at Vandenburg AFB, California.



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# SPACE ACTIVITY IN CANADA

The investigation of space, and the exploitation of it for the betterment of man, has occupied the attention of increasing numbers of scientists, engineers, lawyers, politicians and others not only in the United States of America and the U.S.S.R. but throughout the world. In Canada, the Engineers at RCA Victor Company, Ltd., have supported such space programs as RELAY, ALOUETTE, and ISIS with equipment and detailed studies. They have also designed and built Communications Satellite Earth Stations, and are presently competing with several other companies to obtain a solid position in the world market for these stations.

**G. B. MacKIMMIE, Mgr.**
*Space Systems*

**J. A. COLLINS, Mgr.**
*Space Systems Marketing*

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Current space activity can be divided into two broad classifications: (1) investigation and exploration of the space environment including the reactions of men and materials to it, and (2) the use of space to serve man's immediate needs. This paper and several of the following papers describe work being done by RCA Victor Company, Ltd., Montreal, Canada under both of the above classifications.

### HISTORY OF RCA VICTOR SPACE ACTIVITY

Under the heading of investigation and exploration of the space environment, Canadian scientists had, even before the advent of orbiting satellites, attained a position of leadership in ionospheric studies. The ionosphere is a logical subject of specialization for Canada because of its proximity to the *Aurora Borealis* an important phenomenon associated with the ionosphere. Furthermore, Canada, because of its sparse population and vast area, has a great need of reliable and inexpensive communications, the future development of which may well depend on increased knowledge of the ionosphere. Out of this interest and background, the Alouette project was born in 1959. Canada took advantage of an offer by the U.S.A. to provide launch facilities for Alouette I, the first of a series of Canadian ionospheric sounding satellites, and a forerunner of Alouette II and Isis A. Alouette I and Alouette II, launched in 1962 and 1965 respectively, have been outstandingly successful satellites. Isis A is to be launched in 1968.

RCA Victor first became involved in the Alouette-Isis program when the Government Laboratory constructing the satellite sought and obtained assistance in supplying an FM telemetry transmitter for Alouette I. This first order led to a subsequent contract, calling for progressively increased participation and responsibility on the part of RCA Victor for the Alouette-Isis program and its hardware.

Falling under the heading of serving man's immediate needs comes the subject of Communications Satellites, with their associated Earth Stations and terrestrial interconnections.

Studies during the 1950's culminated in the decision by NASA to construct the Relay experimental communications satellite, intended to establish the feasibility of satellites for transatlantic transmission of wideband message traffic and television. Relay I was launched in December, 1962, and after a difficult beginning, became a great success. RCA Victor, Montreal, held a subcontract from the Astro-Electronics Division, Princeton, N.J., for the development and construction of the all-solid-state transponder for the Relay satellite.

While the work was proceeding at AED and at RCA Victor on the Relay project, senior officials of Government and the Communications Industry were busily working on a bill which was subsequently passed by Congress and became known as the "Communications Satellite Act of 1962." The aim of this act is lucidly expressed in one of its opening paragraphs:

> "The Congress hereby declares that it is the policy of the United States to establish, in conjunction and in cooperation with other countries, as expeditiously as practicable a commercial communications satellite system, as part of an improved global communications network, which will be responsive to public needs and national objectives, which will serve the communications needs of the United States and other countries, and which will contribute to world peace and understanding."

One result of the Communications Satellite Act of 1962 was the creation of the Communications Satellite Corporation (ComSat), whose purpose it is to implement the policy defined in the Act. International cooperation and participation was arranged through the formation in July 1964 of IntelSat. Nineteen countries were original signatories to the IntelSat agreement, amongst which was Canada with a 3.75% interest. ComSat became IntelSat's manager for the space segment (the satellites).

While IntelSat was being formed, Canada took the decision to proceed with the construction of an experimental Communications Satellite Earth Station originally intended to participate in NASA's Applied Technology Satellite program. RCA Victor received a work definition contract in the summer of 1963 for this station and shortly thereafter a contract for the supply of the station itself.

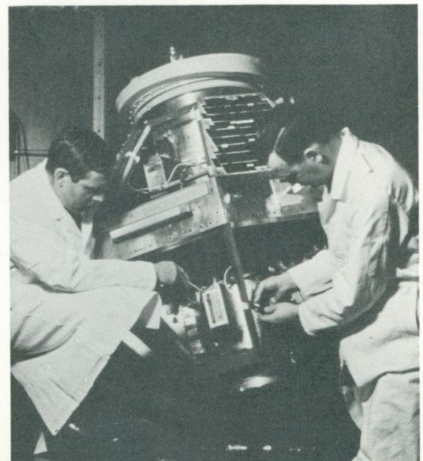
Thus, by late 1963, RCA Victor was solidly into a scientific space program, Alouette-Isis, and a Communications Satellite program—Canada's Earth Station at Mill Village. These contracts were won only by demonstration of engineering competence in such specialties as wideband microwave communications technology, RF optics of large antennas (including feed systems), and systems engineering. This competence grew out of earlier in-house development work in microwave radio relay, and work done under various NASA or NASA-sponsored projects. Furthermore, RCA Victor Research Laboratories had been active in many areas of space technology, and was ready to support the engineers as required.[1]

### DESCRIPTION OF PRESENT PROJECTS

#### International Satellites for Ionospheric Studies (ISIS)

The third satellite of the Alouette-Isis series of ionospheric sounding satellites, Isis A, is now under construction at RCA Victor. As is the case for Alouette I and II, the principal experiment aboard Isis A is the ionospheric sounder. Sounding is accomplished by a pulsed swept-

L. A. Keyes and H. Hore inspecting engineering model of ISIS A.



*Final manuscript received May 31, 1967.*

**G. B. MacKIMMIE** received the B. Eng. (Communications) degree from McGill University in 1943. In 1945, after two years with the Royal Canadian Navy as an electrical officer, he joined RCA Victor where he was engaged for several years in the field of antenna design. From 1956 to 1958 he was Manager of the Broadcast and Antenna group in the Engineering Department. From 1958 to 1960 he was Manager of the Engineering Department, Technical Products Division, and between 1960 and 1966 Mr. MacKimmie was Chief Engineer of Technical Products. In 1966 Mr. MacKimmie was appointed Manager of Space Systems of RCA Victor Company, Ltd. The activity of Space Systems encompasses the development and supply of communication satellite earth stations for the global market and of scientific satellites and subsystems thereof. Mr. MacKimmie is a member of the Corporation of Engineers of Quebec.

**J. A. COLLINS** received the B.Sc. Degree from the University of British Colombia in 1942. He joined RCA Victor in the same year and for a period of seven years was engaged in various design, development programs covering high power LF transmitters, multiplex equipment and instrument landing systems. For two years he served with the Broadcast Engineering group on design of antenna matching networks and installation and proof of performance of broadcast transmitters across Canada. In 1951, Mr. Collins was transferred to Marketing operation of Technical Products for the company, in which capacity he handled sales of radio relay systems, mobile radio, and special defence projects. In 1952 he was loaned to the newly formed Department of Defence Production in Ottawa as a production officer to assist in the military preparedness program, which embraced the Mid-Canada Early Warning Line, the Pinetree Line, and a total step-up in Canada's defence efforts. Upon his return to RCA Victor in 1954, he served as manager of Government Contracts Department. He was actively associated with the ASTRA weapons system contract for the CF-105 project, other major projects, and the build-up of research and development capability within the company. In 1966, Mr. Collins assumed the position of Manager of Marketing for the newly formed Space Systems operation of Technical Products. He was instrumental in bringing about the build-up of capability for communication satellite earth stations. He is currently involved in studies of the application of satellites to serve Canada's growing requirements for Educational TV, Broadcasting, and general Communications.

frequency signal of 100-microseconds duration and 400-watts peak power. The frequency coverage is from 0.1 MHz to 20 MHz. The swept frequency system is supplemented by a fixed frequency sounder operating at six selected frequencies between 0.25 and 9.3 MHz.

A unique feature of the ISIS A satellite is the long antenna associated with the sounder experiment. Two dipoles, one having a length of 240 feet and the other a length of 61½ feet, are used to cover the complete frequency band. The dipoles are extended from the spacecraft when in orbit and, in space, are self-supporting.

Other experiments carried aboard ISIS A are:

- VLF, to measure incidence of lightning induced waves (whistlers) at very low frequencies;
- Cosmic Noise;
- Energetic Particle Detector;
- Langmuir Probes, to measure electron density;
- Ion mass spectrometer;
- Ion probe, to measure ion density;
- Soft Particle Spectrometer;
- Beacon, to measure scintillations, particularly in the auroral zone.

The ISIS A satellite will carry a tape recorder for data storage: a new feature not provided in its predecessors.[2] An engineering model of ISIS A has undergone complete electrical tests, and the flight-unit structure has been fully tested and approved. Flight unit electrical subsystems are now being assembled and engineers are, at the time of writing, beginning the integration of hardware onto the flight structure. At present, approximately forty RCA Victor Engineers with supporting staff are engaged in this activity.

### Communications Satellite Earth Stations

Following the successful completion of Canada's experimental Earth Station, a second station was needed for handling commercial transatlantic traffic at the same location. Accordingly, the Canadian Overseas Telecommunications Corporation solicited international tenders in late 1966, and after evaluation, chose RCA Victor as the supplier. The design of the new earth station differs from the completed experimental one in the following main respects:

- No radome is used;
- Antenna diameter will be 95 feet;
- The interconnection between antenna and control building is accomplished at 4 and 6 GHz, instead of at baseband or video frequencies; and
- The station will be equipped for simultaneous reception of nine separate RF carriers and transmission of two carriers.

Both stations are equipped with automatic tracking antennas which can lock on medium altitude as well as synchronous satellites, although for the immediate future only synchronous satellites will be available for commercial traffic.

In addition to the work which is now beginning on Canada's commercial earth station, Engineers at RCA Victor are busy proposing similar stations to other countries. The international bidding for earth stations is highly competitive, with the leading companies eagerly seeking to obtain a solid position in what undoubtedly will become a substantial market in years to come. RCA Victor Engineers have recently lent assistance to RCA Communications in the establishment of the latter's temporary earth station in Thailand.

### THE FUTURE

In Canada, the direction of space activity during the next ten years might well be towards the employment of satellites for TV and telephone service to the northland, for augmenting existing east to west communication facilities, and for educational TV (a particularly attractive application considering Canada's scattered rural population).

A fundamental question which must be resolved is whether Canada should own synchronous satellites for expansion of its domestic communications, or whether it should rely on renting circuits from a satellite system which is owned by International or U.S.A. interests. In either event, Engineers at RCA Victor look forward to active participation in whatever program is chosen to serve Canada's domestic needs.

As far as programs outside Canada are concerned, it is hoped that there will be a continuation and growth of the kind of cooperation with other parts of RCA which led to the highly successful RELAY I and II satellites. Space communications, by its very nature, extends far beyond national boundaries. The opportunities for service to mankind in this field are unlimited for those who by training and experience can make useful contributions. It is hoped that Canadian Engineers, along with those of the U.S.A. and other nations, will make the best of these opportunities.

# SYSTEM DESIGN AND RELIABILITY CONSIDERATIONS FOR THE ISIS A SPACECRAFT*

The purpose of the ISIS A satellite is to provide data which will aid in the solution to the problem of radio-wave propagation in the upper atmosphere; the main experiments are topside ionospheric sounding with VLF and VHF equipment and direct particle samplings made within one meter of the spacecraft. The choice of such satellite subsystems as telemetry, power, command, and attitude sensing and control are described. Reliability goals, and the means available to meet them, are discussed. A reliability flow diagram with estimated yearly failure rates is developed showing unit and system redundancies from which satellite mission success probabilities are calculated.

**L. A. KEYES, Ldr.**
*Systems Design*
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*Montreal, Canada*

**and**

**Dr. W. R. ATKINS, Mbr.**
*Scientific Staff Systems*
*and Operations Laboratory*
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THE ISIS A spacecraft will be the second of four satellites used in a program of upper-ionospheric measurements sponsored jointly by the Canadian and United States Governments. The National Aeronautics and Space Administration (NASA) Goddard Space Flight Center is responsible for the launch, tracking, and data acquisition through the NASA STADAN network. The Defence Research Telecommunications Establishment (DRTE) of the Canadian Defence Research Board is responsible for the design, construction, and testing of the spacecraft as well as the control of the satellite and data acquisition from telemetry stations in Canada. RCA Victor Company, Ltd., Montreal, is the prime contractor and de Havilland Aircraft of Canada is the associate contractor for the spacecraft which will be launched by a Delta vehicle from the Western Test Range at Vandenberg, California.

The purpose of the spacecraft is to continue a series of topside ionospheric sounding begun by ALOUETTE I (1962) and ALOUETTE II (1965) and to complement this with simultaneous direct particle measurements within one meter of the spacecraft. The series of ISIS (International Satellites for Ionospheric Studies) satellites is planned to provide measurements over a large part of a solar cycle.

The 500-pound spacecraft has units mounted within and on a central thrust tube and equatorial panels (Fig. 1). The spacecraft is to be spin stabilized in a polar orbit with a 3500 km apogee and 500 km perigee.

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## SYSTEM REQUIREMENTS

The ten complementary experiments on board ISIS A largely dictate the system requirements. In these experiments, the composition of the ionosphere is examined using two methods: (1) direct measurement of the energy and flux of particles within one meter of the spacecraft and (2) remote measurement by swept-radar sounding techniques and by beacon observation. The data obtained are to be accurately correlated, in time, to other ground based observations.

The five direct measurement experiments are:

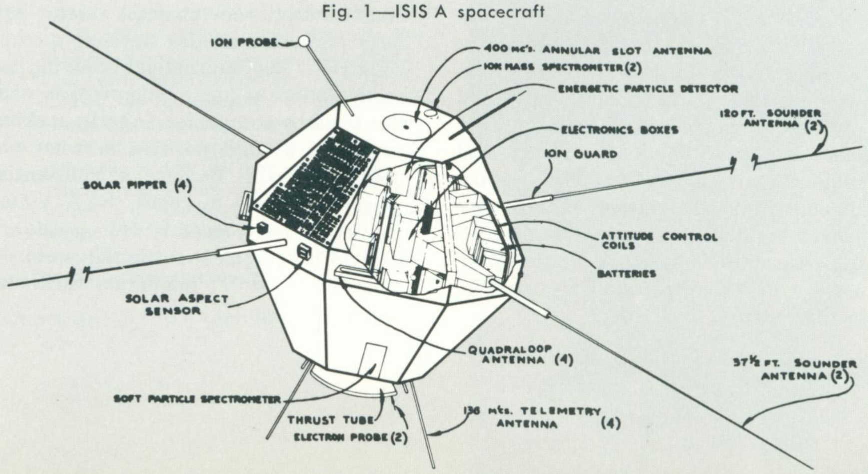
1) An *electron (Langmuir) probe* in which a slowly varying voltage is applied to two electrodes and the resulting volt-ampere curve is used to determine electron temperature and density.
2) A *spherical electrostatic analyzer probe* which applies a slowly varying voltage between concentric, spherical meshes. From the resulting current, information is gained about the energy and mass of the ambient ions.
3) An RF *quadrupole ion mass spectrometer* from which the charge-to-mass ratio of the surrounding ions (mass numbers from 1 to 20) is determined.
4) An *energetic particle detector* which is capable of measuring energies from 3 keV to 50 MeV. It consists of scintillation counters, Geiger counters, and solid-state detectors.
5) A *soft particle spectrometer* measuring in the range from a few eV to 10 keV.

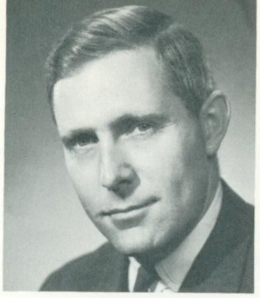
These experiments are supplied as packages ready for integration except for probe mounting and the necessary power, telemetry, and command inputs.

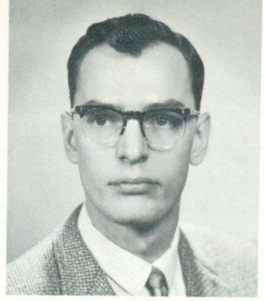
The remote experiments are:

1) A *swept frequency sounder* (the principal experiment) which is a pulsed radar system sweeping over the range 100 kHz to 20 MHz in about 30 seconds, transmitting a 100 microsecond pulse at 30 or 60 pulses/second. Energy is radiated and received by two sets of crossed dipoles: one set 240 feet tip-to-tip and the other 61.5 feet tip-to-tip. Echoes from the ionosphere as a function of frequency give a measure of ionosphere electron density from which ionograms or density profiles can be made.
2) A *fixed frequency sounder* which uses the same basic apparatus as the swept sounder to perform finer spatial resolution measurements at one of five fixed frequencies.
3) The AGC *voltage* of the sounder receiver which is used to measure cosmic noise as a function of frequency.
4) A detector, which is basically an AGC *audio amplifier*, is used to measure VLF radiations and whistlers in the range 50 Hz to 30 kHz. This detector

Fig. 1—ISIS A spacecraft







**LORNE KEYES** received the B.Sc. in Electrical Engineering from Queen's University in 1954. He has taken a number of graduate courses at McGill University. He has been employed at RCA Victor Company, Ltd., Montreal, since 1954 on a variety of telecommunication problems including airborne VHF transceivers, UHF troposcatter telephony, and heavy route microwave telephony. He was engaged in design and system integration of the NASA RELAY communications transponder and recently participated in communications satellite proposals to COMSAT Corporation. He is a member of the Quebec Corporation of Professional Engineers, the Engineering Institute of Canada and the IEEE. He is currently Spacecraft Systems Engineer for the ISIS A Program.

**W. R. ATKINS** received the B.Sc. in Electrical Engineering from the University of Alberta in 1956, and remained there the following year for post graduate studies while holding a lecturer's assistantship. He obtained an Athlone Fellowship to study in England for the period 1957 to 1959, receiving a Diploma from Imperial College and an M.Sc. from the University of London for work on transistor analog multipliers, and gaining experience in analog computers and transistor pulse circuitry. From 1959 to 1960, he was employed by Canadair working on pulse multiplexing equipment of quasi-static low-level information and environmental testing of transistor equipment. In 1960, he joined RCA Victor Research Laboratories, working on studies connected with radar under ECM conditions and investigating behaviour of noise at very low levels. In 1961, he was awarded a David Sarnoff Fellowship to proceed with his studies at the Imperial College of the University of London under RCA Corporate Sponsorship, receiving his Ph.D. in 1965 for work on sequential decoding techniques using decision theory. Since returning to the RCA Victor Research Laboratories, he has been advising on technical exhibits for EXPO '67, and more recently, concerned with spacecraft reliability and systems studies.

is connected to one set of sounder poles and may also be used with a swept exciter to measure ion-resonance effects.

5) A VHF *beacon* propagation experiment which determines total electron content and scintillations in the ionosphere between the satellite and a receiving station.

Because coverage beyond that available in real-time telemetry and command is required, a tape recorder and a clock and programmer are included for remote control and data gathering. A minimum of four hours per day of sounding should be possible at the end of one year in orbit, and it should be possible to operate continuously for two consecutive pole-to-pole passes (half orbit) in the same direction.

The requirement for a reasonable probability of operation of the spacecraft for longer than one year is embodied in the spacecraft system reliability goals. The most important of these goals is a 0.8 or better probability (60% confidence) of completely satisfactory operation at the end of one year in orbit as based on 2000 hours of main mode operation during that year. (Completely satisfactory operation is defined as a fully operational condition in all equipment with the exception of an on-board tape recorder and certain non-essential monitoring channels, solar cells, and programming capability; only one of completely redundant equipments need work. Completely satisfactory operation is a much more stringent criterion than mission success, which is more difficult to specify.) Another important goal is that system design is to be such that the probability of a first failure causing the loss of useful data from more than three experiments should not exceed 0.05.

## DESIGN APPROACH

### Structure

For the required orbit, the Delta launch vehicle allows a payload weight of 500 pounds. Based on the limits of the maximum diameter inside the spacecraft shroud and the power requirement of 50 watts (average) body-mounted solar cell arrays were chosen instead of the mechanically more complex erectable-paddle arrays. The requirement for nearly equal power output at all solar aspects dictated the roughly spherical shape: this shape results in an average power output, for random spin-axis orientation, within a few percent of optimum and has satisfactory thermal and structural properties. The thrust-tube mounting of components provides good accessibility for test and installation, except for those mounted inside the tube. To maintain dynamic stability, it is necessary to locate some of the boxes on the equatorial panels, thus maintaining maximum moment of inertia about the spin axis. A principal feature of the mechanical design is the great length of the sounding antennae: 240 feet tip-to-tip for one set and 61.5 feet for the other. The large change in moment of inertia (as the pre-stressed tape antennae are extended) produces a despin from 90 r/min (at separation) to 3 r/min.

Thermal control is passive by choice of surfaces and by means of aluminized mylar heat shields which are placed over the ends of the thrust tube.[1] Major heat flow is along the tube and the mylar blankets introduce a thermal resistance which reduces this heat flow and, as a result, reduces the temperature gradient along the tube. The high temperature extreme for end-on illumination and the worst-case combination of a fully charged battery and full sunlight is +40°C. The low temperature extreme of −5°C occurs for a discharged battery and end-on illumination in minimum sun (eclipse). The average temperature for random orientation varies by about 30°C. The thermal blankets are beneath the outer covering of the spacecraft which must appear as a closed electrically conducting sphere for RF continuity. The surface of the spacecraft must also present a minimum of 1000 square inches of free metallic area to satisfy experimenters needs.

The spherical electrostatic probe and one of the Langmuir probes are mounted on erectable booms at the top end of the spacecraft. This position is considered necessary to keep the probes from being influenced by the plasma sheath which forms around the spacecraft as a result of voltage induced in the sounder poles as they cut the earth's magnetic field (V x B effect). This sheath represents an equilibrium condition between the resulting collection of highly mobile electrons and less mobile ions.[2]

### Telemetry

The choice of transmission bands is determined by compatibility with the existing 136-MHz and 400-MHz NASA STADAN facilities.

The sounder data in natural form is a composite video waveform consisting of calibration, synchronization and echo return pulses in a spectrum from DC to 20 kHz. This bandwidth makes PCM processing impractical. Although it contains much redundant information and could possibly be processed on board into a more compact form, the simple FM analog transmission system proven on ALOUETTE I and II was adopted. This system matches the existing facilities for ionogram preparation, 136-MHz transmission frequency, and 100-kHz channel bandwidth.

The VLF signals in the band from 50 Hz to 30 kHz are readily accommodated on the same transmission link either on a time-shared basis or by straight summation. The latter technique is possible because the sounder signals are analyzed mainly on a time-domain basis while the VLF signals are analyzed mainly in the frequency domain. The interfering sounder spectra which contains a large proportion of energy at harmonics of the pulse rate can be filtered out or largely ignored. Separate but simultaneous VLF and sounder can be transmitted at the expense of extra power consumption by putting the VLF on the 400-MHz link.

A standard 30-kHz IRIG FM subcarrier modulates the main carrier and is pulse-amplitude modulated (PAM) by the output of a 30-channel essential housekeeping commutator. This is multiplexed with a pulse-width-modulated time code to satisfy a data processing requirement that spacecraft time be available on all data links. The PAM essential housekeeping system is used mainly by the satellite controller for quick-look monitoring of such essential parameters as battery currents and voltages, which are also carried on the PCM system.

Transmission of most direct particle measurement data could be accomplished simply either by an analog system of several FM subcarriers on a single carrier or by time multiplexing on a single carrier. However, due to the widespread use of the recommended NASA PCM time-multiplex system and the compatibility with present data processing systems, a PCM system was chosen. This system lends itself readily to various data rates by use of super- and subcommutation but it necessitates complex synchronizing and decommutation equipment on the ground. The soft-particle spectrometer and the energetic particle detector have data in digital form which can be conveniently multiplexed into the bit stream. The format chosen is a 24-word frame 60 times per second using an 8-bit word. Quantitizing errors are consistent with the 1 percent accuracy aim for the telemetry system.

Some data are sampled twice per frame (120 words/sec) while others are sampled once per second by one of two 60-channel subcommutators. The synchronizing pattern comprises the first 16 bits in a pseudo-random pattern. One bit of the frame carries time code in a non-standard 60 pulse/second BCD format; one bit carries the output of a 60-channel flag subcommutator.

The PCM transmission scheme is a split-phase format modulating a PM transmitter, as this system gives the minimum transmitter power (2W) for the marginal bit error rate of $10^{-4}$. The modulation index is adjusted to leave ten to fifteen percent of the nominal power as a residual carrier to permit good carrier lock for coherent reception.

Transmitter powers and deviations are based on the principle of limiting marginal performance at maximum slant range. Assuming the lowest performance telemetry receiver system operating at 15-degree antenna elevation, marginally satisfactory ionograms can be made with a video S/N ratio of 20 dB.

For both FM and PM systems, average performance will be much better than the marginal limits, since marginal conditions occur for only a short time.

Both FM and PM transmitters are diplexed together and duplexed with the command receivers to feed a four-pole turnstile antenna symmetrically mounted around the bottom end of the satellite. This antenna produces an almost omnidirectional circularly polarized radiation.

The beacons radiate through four equatorially mounted quadraloop elements fed in quadrature to produce a nearly omnidirectional circularly polarized radiation pattern.[3]

The tape recorder requires four channels: two analog for the sounder and VLF, one digital for the PCM, and one analog for time code. The record and playback speeds are interrelated and are ultimately controlled by the bandwidth and power available for the replay link as well as by the time available for readout. The 400-MHz band was the only feasible choice for the replay link and a 500-kHz channel with a transmitter power of 4 watts allows 65 minutes of recording and a 4:1 speed-up on playback. This permits a full tape dump on an overhead apogee pass. Normal record periods are 16 minutes, but successive record periods can cover a single pole-to-pole pass.

The 400-MHz link can also be used as a real-time back-up for the 136 MHz FM and PM. The 400-MHz transmitter has both FM and PM modulation capability. The sounder, or VLF signals, frequency modulate the carrier while a 93 kHz subcarrier normally carrying the PCM data, phase modulates the carrier. Time code in the form of amplitude modulation on a second subcarrier also phase modulates the transmitter. The transmitter radiates from an annular slot antenna, flush mounted at the top end of the satellite, thus there are no protrusions near the particle probes that might affect the plasma. Although it would have been desirable not to have any RF field in the region, it was not structurally feasible to put the antenna on the opposite end. In any case, the RF field produced probably has a negligible effect in comparison with the 400-watt sounder pulse. To provide some control of the plasma effect produced by the sounder poles near the spacecraft, loose insulating boots or sleeves are fitted over the first few feet at the base of the sounder poles.[2] A DC bias may also be applied to the sounder poles.

#### Command System

Based on the large number of experiments and the required operational flexibility, over 100 separate commands were needed. For compatibility, only NASA command standards were considered. A system that exceeds the 70-command capacity of the tone Digital System and yet is simpler than the PCM Instruction Command System is an allowable variant of the Tone Command System. This system transmits single tones sequentially as AM on a VHF carrier. One of fifteen tones is assigned as a unique spacecraft address and is transmitted first. The execute tones follow to produce a particular command function and may consist of up to 3 tones. The system chosen uses six of the possible seven execute tones in a 3-tone sequence to give $6^3$ or 216 commands. Essential commands are redundant in the decoder, reducing the possibility of losing both due to decoder faults.

For command switching, each converter, or separate load, has a latching relay in series with it which is operated according to command pulses fed through a diode steering matrix. The various modes of operation are described below:

> *Main* modes activate the experiments and initiate data transmission or recording in one of the most commonly used experiment configurations (e.g., "All Experiments On"). These main modes require only a single command which is storable for execution at a stored time. This is achieved by a clock and programmer which can store up to five turn-on times in a twenty-four-hour interval and any five of the ten *main-mode* commands. Spacecraft *main modes* are reset to a standard off condition either by a spacecraft manual turn-off or an automatic turn-off.
>
> *Spacecraft sub-modes* normally provide major modifications to *main modes*. The modifying commands, which set magnetic latching relays as do most of the commands, are reset to the standard *main-mode* condition when the spacecraft is turned off, either directly or by automatic timer.
>
> *Equipment sub-modes* are used to switch equipment into relatively permanent states of operation (e.g., to engage a redundant unit or to turn off a faulty unit). These commands are not negated by turn-off.
>
> *Experimental sub-modes* select the internal modes in which a particular experiment will operate. These remain fixed until modified by a further experimental *sub-mode* command.

The command system consists of redundant receivers feeding a command decoder which distributes command pulses to the command switching unit and to other units (Fig. 2). A redundant clock and programmer supplies timing signals and time code to other subsystems and provides remote turn on capability. An automatic turn-off unit (ATO) with internal redundancy prevents accidental discharge of the batteries.

#### Power System

Power for the spacecraft is generated by an array of 11,130 solar cells arranged to form 174 separate circuits with 62

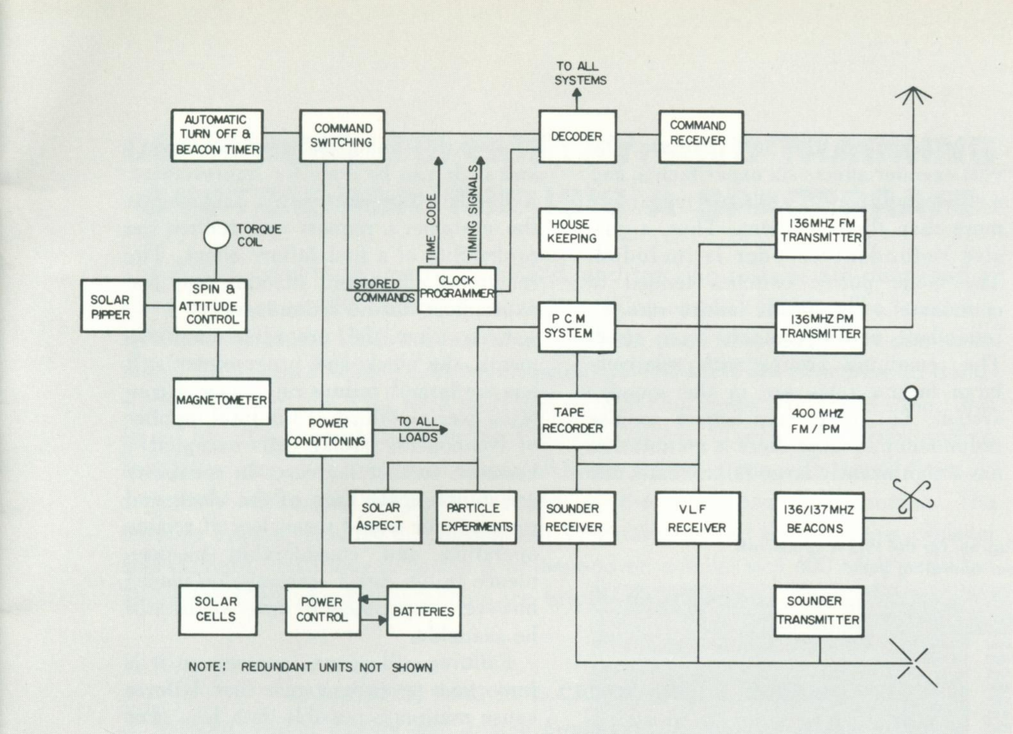



Fig. 2—ISIS A overall system

cells in series. The circuits are mounted on 16 solar cell panels which are placed symmetrically around the spacecraft so that a uniform number of strings are illuminated for all sun orientations. The circuits are combined into three separate charging rails, each capable of producing a 1-amp current early in life. The effects of charged-particle radiation and micrometeroid erosion are reduced by a 0.030-inch fused-silica cover glass; the current from each charging rail will be 700 mA after one year in space. The N-on-P cells have a resistivity of 10 ohms-centimeter with a blue-shifted spectral response.

There are three batteries of 17 nickel cadmium cells, each having 8.5 ampere-hour capacity; discharge voltage is $+21.5 \pm 2.15$ volts. In normal operation, each battery can withstand continuous overcharge at one ampere. If any battery fails, its loads and charging currents are switched equally to the remaining batteries. The charging current is then controlled by a coulometer which integrates the battery current and signals a fully charged condition at which time the charging current is reduced to a safe level. The minimum expected period of daily operation is still met at the expense of increased depth of discharge.

The power system loads are divided into permanent and switchable categories. Permanent loads such as command receivers, decoder and clock are fed from a common diode rail (CDR) which can be fed by any or all of the batteries or a charge rail. Some switchable loads are switched directly to batteries while others are fed by DC-DC converters of which there are five in the power conditioning system, one for each of the three telemetry systems and two for the sounder experiment.

These converters reduce interference by operating in the 30-40 kHz range which is above the highest VLF and below the lowest sounder frequency. They provide overload protection and voltage limiting as a protection against failures in the primary power system such as a battery open circuit on charge. Experimental loads connected to batteries are protected by series circuits against over voltage or over current, or in some cases both.

The typical mode of operation requires 90 watts of power which can be delivered for 13 hours per day in early life in full sunlight, and for a minimum of 6 hours per day in minimum sun after 1 year in orbit.

### Attitude Sensing and Control

The attitude of the satellite is measured by a solar-aspect sensor and a 3-axis flux-gate magnetometer. The solar-aspect sensor gives an eight-bit reading of the angle between the spin axis and the sun line. This value is read out once per second and is updated at each spacecraft revolution. Magnetometer readings taken once per second are compared to the known magnetic field at the satellite position. These readings and the solar aspect are used to compute the spin-axis direction. The phase of rotation is given by a separate pulse from the solar-aspect sensor when it passes through the plane containing the spin axis and the sun. A schedule of spin and attitude system operation is arrived at by computer reduction of the spin and aspect data, and manual commands are issued to the spacecraft.

Control torque for spin and attitude is provided by passing current through a pair of air-core coils which are wound about a spacecraft meridian to produce an azimuthal magnetic moment. The spin rate, nominally 3 r/min can be increased or decreased by reversing the coil current in the correct phase. The system is capable of about 0.1 r/min change per orbit and is turned off automatically after one hour of operation.

Attitude is controlled by switching the coil current under the control of four solar sensors. These sensors are mounted on the equator and give a pulse each time they pass through the sun spin-axis plane. The system is capable of about 2.5-degree change per orbit at nominal spin rate in 100% sun. Either system is switched off automatically to conserve power if the earth's field is less than one-tenth oersted.

### Reliability

The system reliability goals were translated into unit reliability goals by apportionment in ratio to the number of components in each unit. Generic failure rates for the components were used to establish relative failure rates, thereby setting a realistic goal for each unit. (A unit is defined as an engineering module or subdivision uniquely concerned with a particular combination of experiments.) Grouping all units affecting the same combination of experiments permits the construction of a reliability flow diagram; the flow diagram for the final system is shown in Fig. 3, where housekeeping is treated as another experiment with a subdivision for essential housekeeping. Fig. 3 is simplified in that it does not show those relationships having a negligible effect on reliability, and some of the equipment groupings are rather coarse. With this diagram it is convenient to assess the relative criticalness of the groups and determine system failure propagation.

Neglecting the redundancy indicated in Fig. 3, it is obvious that all equipment at the top of the diagram affects all experiments, and in addition their failure rates are relatively significant. Therefore redundancy is introduced to improve overall reliability and reduce the probability of a first failure effect. Thus, the command receiver, clock and programmer, and ATO unit have redundant units. The back-up clock and programmer, which can be made operational by command, is a simpler unit than the original with less flexibility and fewer programming stores. Although not shown in the simplified diagram, most of the critical units have considerable redundancy. For example the power system requires only two out of three batteries, and the Command Decoder has redundant codes for essential commands with additional safety in the command flexibility. Loss of spin and attitude control would not jeopardize all experiments provided that attitude data were available. Loss of the 400-MHz transmitter would still leave all data available in the direct transmission mode; the 400-MHz transmitter can act as an alternate path for real-time data transmission, but this is not shown in Fig. 3.

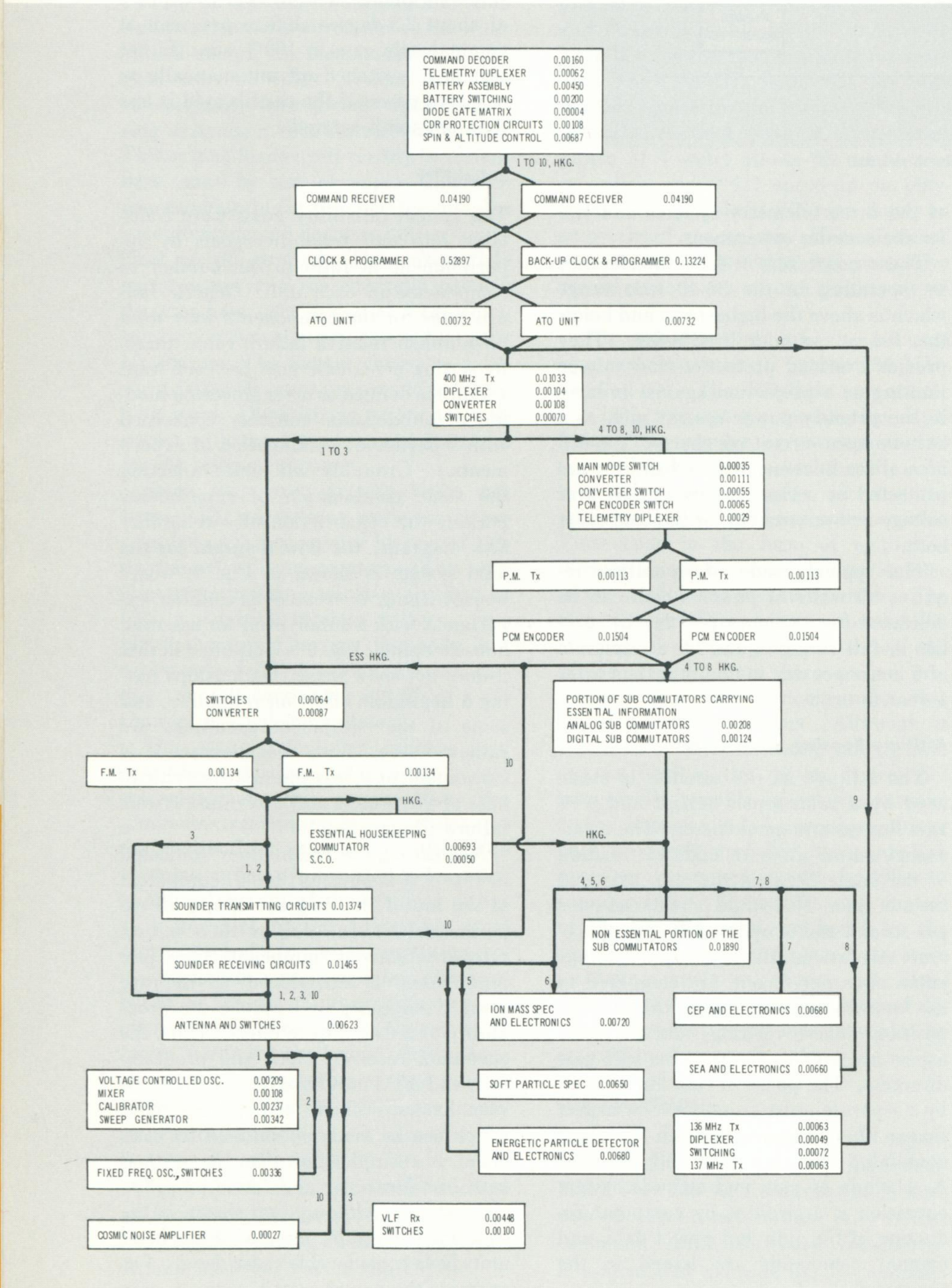
Further down the flow diagram, the PCM encoder affects six experiments, and a single failure will probably jeopardize more than three of them. Thus, a passive redundant encoder is included. The mode power switches tended to contribute a significant failure rate, so redundant active contacts were used. The remaining groups with relatively large failure rates are in the, sounder system, but it is impractical to use redundancy because there is no unit that has a significantly large failure rate; the complexity is inherent in the experiment and little can be done for improvement. Redundant transmitters are included at the customer's request and reduce the probability of a first failure effect. The remaining equipment affecting only one experiment has no redundancy.

Even after the preceding improvements, the clock and programmer still has the largest failure rate, since it contains over one-third of the total number of components (solar cells excepted); however, further increases in reliability are impractical. Loss of the clock and programmer would mean loss of remote operation and considerable inconvenience in the direct transmission mode; however, experimental data would still be available.

Failures will occur, however, so it is important to design such that failures cause minimum possible data loss. For example, the command decoder can affect all experiments, but a single gate failure affects only a localized area of circuitry. Therefore the command allocations should be such as to place commands affecting a single experiment in this localized area so that a failure affects only one experiment (with the exception of redundant commands). A similar approach was used in the design of the commutating circuits. Alternatively when such a separation cannot occur, the design should be such that a failure will only reduce flexibility rather than cause complete data loss.

With the preceding improvements, and the redundancy as indicated in Fig. 3, the calculated probability of completely satisfactory operation at the end of one year in orbit is 0.813, and the probability that a first failure would affect more than three experiments is 0.041. These figures show that the Isis "A" system meets the reliability goals, provided that the units meet their individual reliability goals. Even if the tape recorder were included in the reliability calculations, the probability of completely satisfactory operation is still 0.79. When a mission success is defined to require experimental data in the direct transmission mode only, then the clock and programmer and associated remote operation equipment do not enter reliability calculations. The probability of a mission success at the end of one year in orbit is then 0.89 with a 60% confidence level.

Fig. 3—Simplified reliability flow diagram for the ISIS A spacecraft showing failure rates per operating year.

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# COMMAND, TELEMETRY, AND TRACKING SYSTEM FOR THE ISIS A SPACECRAFT

The ISIS A satellite telemetry, command, and tracking systems are described in some detail as are the antennae and units making up the systems. The telemetry system includes two 136-MHz real-time data links, one FM for sounder video transmission, and one PM for PCM transmission. A 400-MHz link of increased bandwidth is used to playback tape recorded data at four times the record speed and to act as a real time back-up for the 136-MHz system. Commands corresponding to main modes of operation of the satellite are stored, along with stored times, in a content-addressed magnetic memory in the clock and programmer. The antenna system includes a four-element VHF turnstile fed by a low-loss diplexer and duplexer featuring strip-line couplers and an annular slot 400-MHz radiator. The extendible sounder antennae are briefly described.

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The Isis A ionospheric sounding satellite is now under construction at RCA Victor. The principal experiment aboard this satellite is the ionospheric sounding which is performed by a pulsed swept-frequency signal. The frequency coverage is from 0.1 MHz to 20 MHz. The swept frequency system is supplemented by a fixed frequency sounder operating at six selected frequencies between 0.25 and 9.3 MHz. A unique feature of Isis A is the antennae associated with the sounder experiment.

### TELEMETRY AND TRACKING SYSTEM

The telemetry and tracking system (Fig. 1) consists of two main subsystems: one for use in direct real-time transmission and the other for transmission of tape recorded data with a speed-up playback (4:1). The first subsystem operates in the 136-MHz telemetry band,[2] and the second operates in the 400-MHz band.

The 136-MHz portion consists of two redundant pairs of transmitters: one pair of FM transmitters with a 100-kHz transmission bandwidth and one pair of PM transmitters with a 50-kHz transmission bandwidth. Either FM transmitter is selectable on command and is modulated by a sounder video waveform which is an analog signal (DC to 15 kHz) consisting of synchronizing and calibrating pulses at 30 or 60 pulses/second plus echo returns. It may be modulated also by the output of a VLF receiver in the 50-Hz to 30-kHz band, either on a time-shared basis with the sounder or combined with the sounder. The carrier is continuously modulated by a 30-kHz IRIG FM subcarrier[3] which is normally modulated by the output of a 30-channel essential housekeeping commutator. The commutator output is replaced by clock and programmer data (60 pulse/second BCD code) for five seconds at the beginning of each minute. The FM transmitter output is fed to a diplexer which feeds a four-element turnstile antenna.

Data from all experiments other than the sounder and VLF are time multiplexed and encoded using one of a redundant pair of PCM encoders[4]. The output of the encoder in split-phase format is used to phase modulate one of the PM transmitters. The transmitter output is duplexed with the command receivers and feeds the same duplexer and turnstile antenna as the FM transmitter. All designs must meet specifications over a temperature range of $-5$ to $+40°C$ and must operate over the range $-50$ to $+75°C$.[5]

#### 136-MHz FM Transmitter

The 136 MHz FM transmitter is a four-watt unit with a basic design that has proven successful on Alouette I and II. It consists of a 17-MHz crystal oscillator employing crystal pulling for frequency modulation. This method requires careful control of spurious modes of crystal oscillation to control stability and modulation distortion. The oscillator is followed by three stages of buffering, and doubling by varactor multipliers. The final amplifier is a single 2N3375 transistor feeding the output via a band-pass filter which is necessary to meet the NASA spurious emission requirements.[2] The design employs conventional discrete components and the construction is by a series of individual compartments milled from an aluminum block.

#### 136-MHz PM Transmitter

The 136-MHz PM transmitter is a two-watt unit consisting of a 34-MHz crystal oscillator with two sages of buffering and varactor doubling, followed by a final amplifier and filter. The phase modulator is a two-varactor high-pass low-pass interstage filter design. In this arrangement, the incidental amplitude modulations are complementary and so tend to cancel, while the phase modulations add and some of the modulation non-linearities cancel. The design of the transmitters is similar to that of the FM units.

#### PCM Encoder

The PCM encoder employs a frame of 24 words of 8 bits each (192 bits), with 60 frames per second (11,520 bits/sec).[4] Some data are in the form of slowly varying voltages in the range 0 to +5 volts with frequency components not over 25 or 50 Hz. These are sampled once or twice per frame, according to the bandwidth required, by a commutator consisting of analog gates connected to a common output. These gates use field effect transistors (FET) as analog switches, and are opened in a sequence defined by a counter using integrated logic circuits. The samples are held by a sample and hold circuit and then encoded by an A/D converter of the cascaded feedback type. The encoder output is stored in a shift register and clocked out at the appropriate time with the most significant bit appearing first in the output. Some data, which are already in the form of bits stored in experiment shift registers, are fed word sync pulses at the appropriate time and bit sync which is then used to clock out the contents of the shift registers. During these intervals, the commutator and A/D converter are inactive.

Two words are available as 16 parallel access bits for a digital flag subcommutator and various high speed flags such as a solar aspect sensor and a 60 pulse/second BCD time code. Two analog words are used to carry the output of two 60-channel analog subcommutators. Frame sync consists of the first sixteen bits in a pseudo-random pattern. Inputs to the redundant encoders are normally fed by separately buffered outputs but in some cases may be connected in parallel. The encoder design attempts to minimize the probability of loading parallel inputs under

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failure conditions. The inactive encoder as well as all inactive units using FET's as series switches are maintained in the pinched-off condition by a permanent "keep alive" voltage. The design uses discrete components and integrated circuits on single-side printed circuit boards mounted in sandwich style.

#### Analog Subcommutator

Each analog subcommutator scans 60 channels/second, giving an individual channel sampling rate of one/second. The commutators are not redundant because the experimental data on either subcommutator are not carried by both units nor by other means. Thus, the commutators are designed as five-layer trees to reduce the possibility of catastrophic failure. A combination of relatively low source resistance for experimental outputs ($\sim$ 10K) and a high input resistance (1 Meg) in the PCM encoder ensures that the 1% PCM accuracy aim is not compromised by the series elements. Fifteen channels on each commutator arranged in 5 groups of 3 are available as low-level inputs (100 mV full scale). These channels are amplified by a single differential amplifier after commutation. This method minimizes the number of amplifiers used, at the expense of lower accuracy in these channels due to amplifier input leakage current flowing through series FET switches. Subcommutator synchronization is achieved by resetting the counters to the "channel one" condition with a separate divide-by-60 counter which also provides subcommutator sync indication as a "true" condition on the first three bits of the sixteen parallel-access PCM bits. The design uses both discrete and integrated circuits and construction is by single-side printed-circuit mother boards mounting "flat pack" logic integrated circuits and "cordwood module" discrete components.

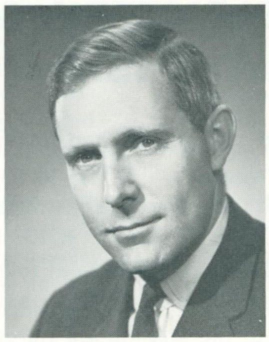
#### Digital Subcommutator

The digital subcommutator is a 60-input unit using gated counters scanning a 5x12 matrix. It is driven and synchronized in the same way as the analog subcommutators to give a channel information rate of one bit/second. Inputs are compatible with flat-pack logic (0 and +5 volts) or with relay contact closures (short or open) and are designed for greatest noise immunity.

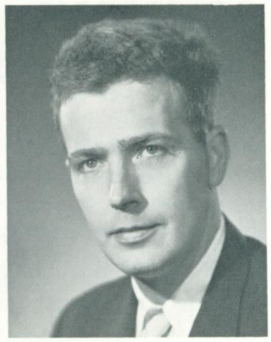
#### Tape Recorder

The tape recorder is a four-track unit in a pressurized container which records sounder video, VLF, PCM and time code on separate tracks. The sounder video modulates the period of an oscillator in the recorder whose output gives

JOSEPH ZURAN



LORNE KEYES



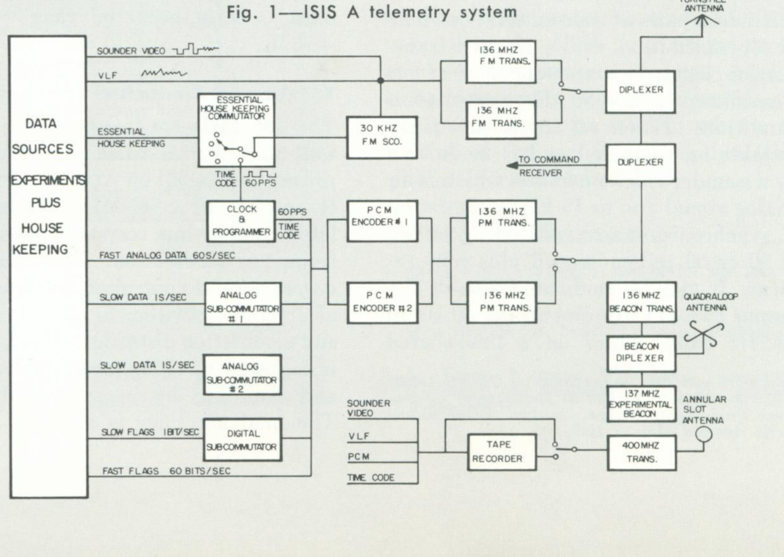
P. T. CADEN

**JOSEPH ZURAN** received the Dipl.ING. degree in mechanical engineering from the University of Belgrade in 1941 and the M.Sc. degree in electrical engineering from the University of Ljubljana in 1949. During World War II, he served with the RAF in the Middle East. From 1946 to 1949, he was Assistant Lecturer at the Faculty of Electrical Engineering, University of Ljubljana. From 1949, he was chief of transmitter group of the Institute of Telecommunications, Ljubljana, and in December 1954 joined the Marconi Wireless Telegraph Co., Chelmsford, England, working on advanced high power transmitter designs. From 1958, he was in charge of the Antenna Development Section at the Marconi Research Laboratories, Gt. Baddow, Essex. He joined RCA Victor Company, Ltd., Montreal, Canada as specialist engineer in 1960. Since then he has been responsible for the development of RCA Victor log-periodic antennas, wideband high power transformers, and multi-channel TV antennae for Canadian Broadcasting Corporation and is presently with Aerospace Engineering, responsible for spacecraft antenna systems and design. He holds several patents on high power transmitters, helical resonators and wideband antennas. Mr. Zuran is a member of the IEE (Great Britain), the IEEE (US) and a member of the Corporation of Professional Engineers, Quebec, Canada. He is also a member of Professional Groups on Antennas and Propagation and Microwave Theory and Techniques.

**LORNE KEYES** received the B.Sc. in Electrical Engineering from Queen's University in 1954. He has taken a number of graduate courses at McGill University. He has been employed at RCA Victor Company, Ltd., Montreal, since 1954 on a variety of telecommunication problems including airborne VHF transceivers, UHF troposcatter telephony, and heavy route microwave telephony. He was engaged in design and system integration of the NASA RELAY communications transponder and recently participated in communications satellite proposals to COMSAT Corporation. He is a member of the Quebec Corporation of Professional Engineers, the Engineering Institute of Canada and the IEEE. He is currently Spacecraft Systems Engineer for the ISIS A Program.

**P. T. CADEN** received the BA in mathematics and physics from the University of Toronto in 1954. From 1954 to 1958, Mr. Caden was an Engineer with the de Havilland Aircraft Co. of Canada where he assisted in the simulation of terminal phase of interception of the AVRO ARROW using CEAC computer and in the analytical design of space-stabilized gymbal system for SPARROW II infrared homing system. In 1958, he joined Avro Aircraft Co. of Canada as an aerodynamicist and from 1959 to 1962, he worked for Canadian Aviation Electronics where he was a Systems Engineer in charge of development of a Sonar System of ARGUS ASW trainer; development of 2-$\mu$sec cycle time asynchronous computer (first of its kind), and design of a CAE telepath translater. From 1962 to 1966, he was a consultant to the Sperry Gyroscope Co. where he participated in system planning for the UMAC-5 computer. He also developed software packages of modular nature to run UMAC-5 computer system. In March 1966, he joined RCA Victor Co., Ltd., where he is presently leader of the Digital Systems and Equipment Group, Aerospace Engineering. Mr. Caden is a member of the Canadian Association of Physicists.

Fig. 1—ISIS A telemetry system



saturation recording on one track; VLF signals are recorded on a normal AC-biased track. PCM signals are provided in NRZ-C format but are converted by the recorder before recording, to a split phase format. Time code in a 60 pulses/second BCD pulse-width format is used to amplitude modulate a reference PCM bit rate signal which is recorded on the fourth track. The tape recorder is controlled remotely by commands stored in the clock and programmer and may record for several periods for a total of 65 minutes. The normal record period is 16 minutes at 5 in/sec and the playback is in reverse order at four times this speed. A period discriminator restores the sounder video waveform and the split phase recorded PCM data is restored to NRZ-C format. VLF and time code are available without special processing in the tape recorder. The outputs are played back over the 400-MHz link.

#### 400-MHz Transmitter

The 400-MHz transmitter is a four-watt unit with both FM and PM modulation capability and contains a latching relay switching unit for input signal switching of played-back data or real-time data. The RF portion of the transmitter consists of a 17-MHz crystal oscillator, capable of frequency modulation by crystal pulling, feeding a buffer and varactor quadrupler. For this application, the crystal must be free of spurious resonant modes over an unusually large band because of the modulation frequencies involved which can be up to 80 kHz. This stage feeds a buffer and tripler circuit via a single tuned phase modulator interstage network. A further buffer and doubler circuit provides 2 watts of power at the output frequency which is then used to drive the final amplifier to the required four-watt output level.

A final filter is required to keep crystal harmonics from exceeding the specified output level. The power output is fed directly to an annular slot antenna mounted on one end of the spacecraft. As well as providing input signal switching, the transmitter also contains a non-standard 93-kHz SCO which may be modulated by various played-back or real-time data. The SCO and the played-back time code which appears as a 240 pulse/second modulation on a 46,080 Hz carrier produce phase modulation of the 400-MHz carrier while the sounder and VLF frequencies modulate the carrier. The transmitter construction is nearly identical to that of the 136-MHz FM and PM units.

#### Experimental and Tracking Beacons

These units which are identical except for their frequency, radiate 100 mW of unmodulated power and together provide stable frequencies at about 136 and 137 MHz. Their outputs are diplexed together and radiate through four equatorially mounted quadraloop elements. One frequency satisfies the requirements for a tracking beacon compatible with NASA STADAN[2] facilities while the two together are used to study ionospheric irregularities.

The beacon design consists of a seventh-overtone crystal oscillator operating at the output frequency, followed by a buffer and final amplifier. Particular care is taken in the design to minimize the effects of temperature and voltage fluctuation on frequency. Low power dissipation in the crystal also minimizes long term aging effects. Worst case combination of voltage fluctuation and temperature shock still result in stabilities better than $10^{-6}$ without the use of crystal ovens. Tests at atmospheric conditions have shown that the difference frequency between beacons may vary as little as $10^{-7}$ for periods of 10 minutes during thermal transients. This is probably because of nearly identical construction and frequency-temperature characteristics.

### COMMAND SYSTEM

The command system accepts approximately 150 commands via radio link from ground stations, decodes these commands into control signals for spacecraft equipment either immediately on reception or after a command delay, and ensures that spacecraft equipment is turned off after a suitable operating period in the event of no reception of a deliberate turn-off command.

Reliability and system design of the spacecraft system and experiments dictate different reliability requirements for different commands and functions of the command system. The command switching design separates the commands into four categories: *main modes, spacecraft submodes, equipment submodes* and *single commands.*[1]

The command link to the spacecraft consists of an amplitude modulated VHF carrier, which carries a 7-ary series of tones to the spacecraft in groups of four tones. The first tone is an address tone, and the next three represent one of 216 6-ary characters.

The commands decoded by the command decoder affect operating spacecraft units directly. However, the clock and programmer is an alternative source for some commands (main modes). The clock and programmer is a simple computer with a content-organized memory of ten words. Each memory word can contain a number from one to ten (ten main modes) and a time of day. The memory contents are continually being scanned and compared to a clock; when the clock time agrees with the time stored in a memory location, the associated stored main mode command is executed. The memory of the clock and programmer is filled by simple commands designated for this purpose.

As the capacity of the spacecraft equipment to dissipate power exceeds the capacity of the solar cells, the main

Fig. 2—Command system

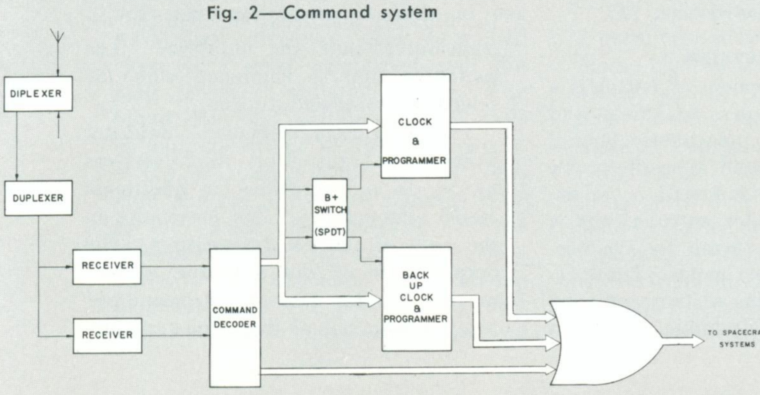
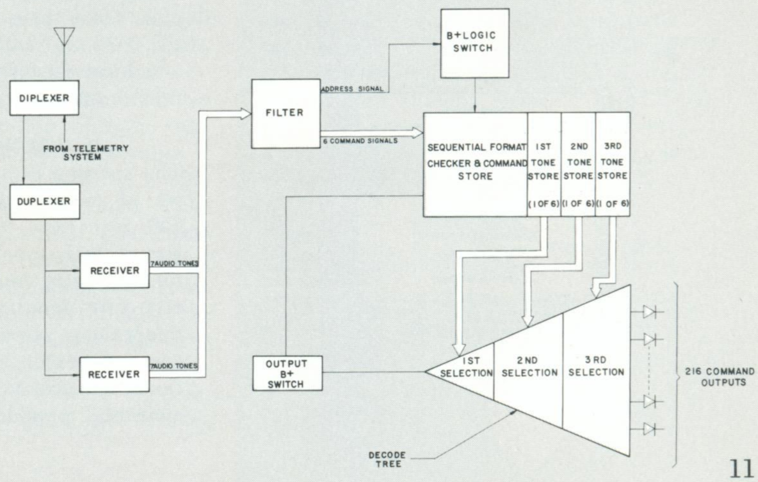



Fig. 3—Command decoder



power drain of the spacecraft is called upon intermittently and is drawn from storage batteries. This arrangement requires that the storage batteries be never fully discharged; the automatic turn-off unit is designed to guarantee this condition. When certain spacecraft functions are turned on by command, the automatic turn-off unit is turned on also, and (after a suitable interval) causes the appropriate spacecraft function to turn off. Thus power consumption is limited in the event of failure to receive a transmitted turn-off command. Fig. 2 is a block diagram showing the interconnection of the units in the command system.

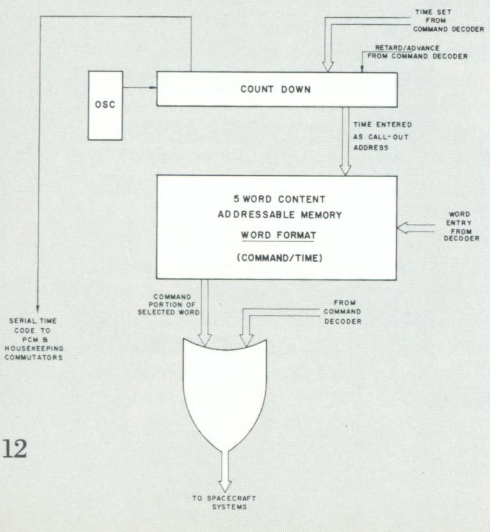
The command receiver decoder combination is redundant providing two receiver decoder pairs, each handling 108 different command codes. Those commands requiring redundancy for enhanced reliability are made up of outputs associated with two command codes, one from each receiver decoder pair, combined in a diode "or" circuit.

Some of the decoded commands, redundant and single, go to the clock and programmer, and some to the other spacecraft units. Delayed mode selections originating in the clock and programmer provide alternates for ten of the forementioned commands.

#### Command Decoder

Each section of the command decoder (Fig. 3) includes equipment performing the following functions: filtering, format checking, memory, and decoding. The detected signal from the command receiver is presented to seven filters, where the tone is classified as "address" or some "number" one through six. If the tone is an "address", power is applied to the decoder logic; "number" tones following the address tones are stored as decoded lists in flip-flop registers. Upon reception of the third "number" tone, if the message has consisted of one address and three numbers, and if the durations and spacings of the tones have complied with the required format, voltage is applied to the top of a three-layer tree of transistor switches, 3 in the top layer, 6 in the next, and 108 in the bottom layer. The appropriate switches are held open by the contents of the flip-flop memory, resulting in a current flow from one of the 108 outputs corresponding to the command code received.

Fig. 4—Clock and programmer




#### Clock and Programmer

The clock and programmer (Fig. 4) consists of an accurate clock, a program store, command inputs, command outputs, serial data outputs, timing pulse outputs, and control logic. The clock is an accurate (0.5 part in $10^7$) temperature compensated crystal oscillator, whose output is divided down by a counter to provide a time code. The program store is a content-addressed magnetic memory of ten words, each word capable of storing a time of day and one of ten characters corresponding to the ten main modes. The contents of the program store is defined by a series of commands from the command decoder.

The time of day portions of the program store words are being continually compared to the time code, and when agreement is observed, the associated mode command is issued, and the time portion of the word is replaced by an inadmissible time (29:00 hours).

The clock and programmer is paralled by the back up clock for reliability. These two units receive power via a type-C magnetic latching relay.

#### Automatic Turn-Off Unit

The automatic turn-off unit is essentially a redundant pair of clocks timing the turn-off of the spacecraft equipment. The main unit consists of a magnetic shift register, counting minute pulses provided by the clock and programmer. This counter is turned on automatically, and times for sixteen minutes. By command, this time can be changed to eight or twenty-four minutes. The main unit is paralleled by the back-up automatic turn off unit, which consists of an oscillator and thirty-minute magnetic counter.

### RADIATING SYSTEM

The antenna complement of Isis A is a set of crossed dipoles, 61.5 feet and 240 feet long for ionospheric sounding, a VHF broadband turnstile array for telemetry and command, a telemetry UHF annular slot antenna, and a quadraloop antenna array for the beacon and beacon experiments. The locations and orientations of these antenna elements provide optimum radiation patterns at minimum mutual coupling.

The VHF telemetry and command systems share the same antenna. In orbit, the attitude of the spin-stabilized spacecraft continuously changes with respect to earth and a nearly isotropic radiator is required for continuous telemetry and command operations. A broadband circularly polarized monopole array in turnstile configuration provides the best solution at VHF frequencies. The array is located around the end of the spacecraft thrust tube which is attached to the launching rocket by a marman clamp. The sleeve-type monopole elements of the turnstile array have a broadband characteristic with 30% frequency bandwidth and have a nominal input impedance of 50 ohms. The PM and FM transmission use opposite rotations of polarization controlled by the phasing characteristic of the telemetry diplexer. The design of the telemetry antenna system is governed by bandwidth requirements; isolation requirements between transmitters and receivers; attenuation of harmonics; and high isolation between transmitters to prevent cross modulation between the two transmitters.

The heart of the telemetry antenna system is the S-type diplexer which combines a broadband characteristic with a low insertion loss, low VSWR, and high isolation between transmitter input ports. The S-diplexer, which is a 3-dB strip-line coupler using Faraday screening, offers particular advantages over the conventional coaxial hybrid when antenna arrays are to be connected in phase quadrature over wide frequency bands. The diplexer provides an isolation in excess of 45 dB between telemetry transmitters at an insertion loss below 0.2 dB. The VSWR at the input terminals of the diplexer is less than 1.05. The power ratio at the two antenna ports is within $\pm 0.25$ dB. The deviation from phase quadrature at transmitter and receiver frequencies is within $\pm 2$ degrees.

The main and standby FM and PM transmitters are connected to the switching unit, which is operated on command. A system of SPDT latching relays connects the selected FM transmitter to the diplexer and feeds the selected PM transmitter into the duplexer. The switching unit is combined with the telemetry/command diplexer.

The telemetry/command duplexer isolates the transmitters from receivers by 50 dB and provides an additional 50-dB attenuation of 2nd harmonics in the path of the PM transmitters. The output port of the duplexer is connected to the common transmitter/receiver terminal of the telemetry/command diplexer. In the command antenna system, the RF signal which is intercepted by the turnstile antenna is fed into the diplexer and picked-up at the common transmitter/receiver terminal of the diplexer. From this terminal, the receiver signal is fed into the duplexer which provides a 3-dB power split with an isolation of 40 dB. The two command receivers which are connected to the duplexer feed the command decoder.

The duplexer is designed as a combination of a folded comb-line filter and an S-type 3-dB strip-line coupler using the same basic design as the telemetry/command diplexer. The height of the resonators of the comb-line bandpass filter is less than one-sixteenth of a wavelength at 136 MHz, thus resulting in a compact lightweight unit.

The broadband metallic matching sleeves of the turnstile monopole elements are attached to base-mounts by threaded fittings enabling the antennas to survive the high vibration levels at launch. The nominal input impedance (50 ohms) of the monopoles permits connection of dummy loads at the base of antenna mounts for rapid check-out of the telemetry system prior to launch.

The broadband diplexer provides high stability of operation at wide temperature variations from −50°C to +70°C. Due to the inherent phase quadrature at the ports of the diplexer, the RF harnesses feeding each pair of monopoles are identical. As a result, a small number of soldered cable junctions is used, reducing the harness insertion loss and increasing the reliability of operation of the system.

The 400-MHz telemetry antenna system utilizes an annular slot antenna with a back-up cavity. A broadband radial transmission line transformer, forms an integral part of the cavity and matches the antenna to the transmitter over a wide frequency band. The antenna system is trimmed for minimum reflection (1%) at the transmitter frequency using a shorted coaxial stub.

The annular slot antenna is designed to operate as an electrically short radiator near the lower cut-off frequency with surface currents flowing across the near-spherical spacecraft body of one wavelength diameter. The combined effect is a good omnidirectional, linearly polarized, radiation pattern. The broad doughnut pattern has two nulls in both directions of the spin axis.

The supporting booms for the Langmuir and ion probes are mounted close to the annular slot antenna and are decoupled from the antenna surface currents by a system of ferrite chokes mounted along both booms. The effect of the metallic booms on the antenna radiation pattern is thus reduced.

The antenna surfaces are highly polished and form a part of the heatshield of the spacecraft. For the same reason, the antenna back-up cavity is embedded in a thermo-insulated container.

The beacon radiating system has manifold functions. The antenna array radiates a 136-MHz CW signal for tracking purposes in conjunction with a 137-MHz beacon for the study of ionosphere irregularities and electron content of the ionosphere.

The University of Western Ontario is the principal user of the beacon facility for studying isolated inosopheric irregularities. Small-scale variation in the total electron content of the ionosphere are studied using the differential polarization method (measuring the instantaneous difference angle between polarization received at the two beacon frequencies).

The closely spaced frequency method is used to determine the total electron content of the ionosphere averaged over several degrees of orbit path. Simultaneous amplitude and phase scintillation measurements of the received signal provide information for establishing the ionic inhomogeneities responsible for these scintillations.

Depending on the experiments, either circular or linear polarization is used. The antenna provided for these experiments is an array of four loop antennas mounted on the equatorial panels between the sounder antenna monopoles. The antennas are fed in phase quadrature, providing a circularly polarized pattern in the direction of the spin-axis of the spacecraft and mainly linear polarization in the direction perpendicular to the spin axis. The quadraloop antennas are shorted quarter-wave transmission-line radiators, capacitively loaded at the open ends by slug-type capacitors.

The individual quadraloops have a nominal input impedance of 50 ohms and are fed by a coaxial-cable phasing network. The common cable junction has an input impedance of 12.5 ohms. The transformation to 50 ohms is achieved by two quarter-wavelength 50-ohm cable sections in parallel. Thus, the same cable is used throughout.

The beacon antenna system features a novel type of strip-line diplexer, combining the two transmitters operating at closely spaced frequencies into a common 50-ohm antenna terminal. The diplexer consists of a half wavelength 3-dB coupler with Faraday shielding posts, assisted by two helical resonators. With matched output terminals, the diplexer provides isolation between transmitters in excess of 45 dB at 1% frequency separation over a temperature range from −50°C to +70°C. The insertion losses are 0.9 dB and 0.7 dB between the output and the two input ports of the diplexer. The VSWR at input ports does not exceed 1.1:1.

The sounder antenna system is essentially the same as those for ALOUETTE I and ALOUETTE II. The sounder array has a crossed dipole configuration mounted in the equatorial plane of the spacecraft. The dipole radiation pattern is derived from two pairs of monopoles, each pair driven in antiphase by a quasi-coaxial feed system. The pairs operate sequentially in two bands over the frequency range of 0.1 MHz to 20 MHz, with a crossover frequency at 5 MHz. The length of the shorter dipole and the crossover frequency are determined by the pattern requirements allowing a maximum side-lobe level of 10 dB below the main lobe and also to ensure antenna efficiency at the lower end of the high frequency band. The antenna elements are made of preformed tubular beryllium copper tape, and stored in the antenna module before and during launch. The antenna elements are individually extendible in orbit by means of an electric motor mechanism activated on separation from the launching vehicle or on command.

The antenna modules are equipped with speed control circuits which maintain correct de-spin rate of the spacecraft during deployment of antenna elements. The modules are double shielded coaxial units with RFI shielding in excess of 100 dB.

The four modules are radially mounted on the thrust tube of the spacecraft and fed via a combining crossover network mounted inside the thrust tube. The crossover network consists of a low-pass high-pass filter and a TR switch. In orbit, the long sounder antennas cut the lines of earth's magnetic field and build up a potential proportional to $\mathbf{V}\mathbf{x}\mathbf{B}\cdot L$, where **V** is the spacecraft velocity, **B** is the earth's magnetic field and $L$ the length of antennas. To free the spacecraft from the ion sheath generated by the $\mathbf{V}\mathbf{x}\mathbf{B}\cdot L$ effect, all four antenna modules are equipped with ion guards in the form of cylindrical 3-foot-long silicone rubber sleeves mounted at the root of each antenna element. In addition, the antennas are coupled capacitively to the crossover network to prevent DC from reaching the spacecraft.

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# THERMAL DESIGN OF THE ISIS A SPACECRAFT

The thermal design of the ISIS A spacecraft was dictated by details of the structure, orbit mission, on-board temperature limits, and payload power dissipation. Consideration of these factors showed that 'passive' thermal control of the spacecraft was feasible. A computer programmed multi-node heat balance analysis is described which has been developed and used to predict spacecraft flight temperatures and evaluate design parameters. The scope of a thermal-vacuum test program required to establish design integrity is outlined.

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All the experiments and electrical systems are on the Isis A spacecraft thrust tube and equatorial panels. (A complete description of the spacecraft is given in Reference 1). Because Isis A will be spin stabilized (with a design spin rate of 3 r/min) net heat inputs will tend to be axially symmetric for any sun-spin vector orientation. As the spin stabilization does not prevent long-term large-amplitude spin-axis excursions all sun-spin vector orientations must be considered. Thus sixteen sloping trapezoidal solar cell panels were used to provide fairly constant solar cell aspect ratio for the complete range of sun-spin vector orientations, without allowing the absolute minimum aspect ratio to fall significantly below the mean. (The solar cell aspect ratio is the projected solar cell area in any direction divided by the total solar cell area of the spacecraft.) Solar aspect ratio of a sphere is 0.25; values for Isis A range between 0.215 and 0.260.

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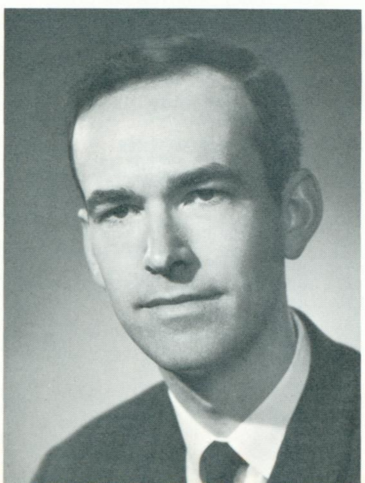


The estimated electrical power dissipation for the electrical systems is 60 watts continuous over an entire orbit. Depending upon the mode of equipment operation, the instantaneous dissipation range can be from 120 watts to nearly zero. The required flexibility is provided by 51 nickel-cadmium storage cells grouped into three 17-cell batteries; these can supply power in addition to the conversion rate at peak demand and low solar input, and store energy during low demand and high solar input.

The temperature constraints of various components of the spacecraft, including the storage batteries, are $-5°C$ to $+40°C$. These limits must prevail for all satellite orbit and attitude configurations. This includes the launch phase during which time the spacecraft will not be spinning and for which a special ascent heating calculation must be performed.

## THERMAL CONSIDERATIONS

The thermal design techniques used for Alouettes I and II were developed further for Isis A.

The average surface properties of the spacecraft are based on the maximum and minimum heat inputs averaged over an orbit. The inputs come from the sun, the earth, and from spacecraft internal heat dissipation (Fig. 1).

There are two heat inputs from the earth: infrared radiation due to the earth's internal temperature, and reflected radiation from the sun (albedo). The earth infrared input to the spacecraft is a function of altitude, while the solar reflected radiation is a function of both altitude and the angle between the earth-sun and earth-spacecraft vectors. For the nominal Isis A orbit, the average infrared input is about 10% of the solar input, and the reflected solar radiation can vary from a maximum average over one orbit of about 10% of the solar input to zero, depending on the angle between the sun-earth vector and the orbit plane. The maximum instantaneous value of reflected solar radiation is almost 50% of the direct solar input. The extreme conditions possible are listed in Table I.

In addition to variations in heat flux, there is the possibility of variations in spacecraft-sun aspect. Due to gravitation forces, the orientation of the spacecraft spin axis can change through a large amplitude for periods of several weeks. Thus, the surface of the spacecraft must be designed so that the heat intake is, as near as possible, independent of the spin-axis orientation.

For Isis A, the solar cells are on panels which are part of the external shell. The distribution of solar cells on the surface is dictated by the necessity for having a constant rate of conversion of solar power, independent of the spin-axis orientation.[3] Fortunately, this tends to agree with the thermal need for a constant $(\alpha/\varepsilon)$xArea, independent of orientation.

In the direction normal to the spin axis, the average heat intakes are:

$$\textit{Direct solar radiation} = A_H\,\bar{\alpha}_H\,SP \quad (1)$$

$$\textit{Earth reflected radiation} = A_H\,\bar{\alpha}_H\,R \quad (2)$$

$$\textit{Earth infrared radiation} = A_H\,\bar{\varepsilon}_H\,E \quad (3)$$

$$\textit{Payload power dissipation} = P' \quad (4)$$

The heat loss is

$$A\bar{\varepsilon}\,\sigma T^4 \quad (5)$$

Thus, the heat balance is

$$A\bar{\varepsilon}\,\sigma T^4 = A_H\bar{\alpha}_H\,(SP + R) + A_H\bar{\varepsilon}_H E + P' \quad (6)$$

In general, similar equations can be written for any orientation of spin axis. Therefore, the spacecraft mean temperature is a function of $\alpha/\varepsilon$ for the various external surfaces, and, to some extent, is a function of $|\bar{\varepsilon}|$.

Thus, temperature of an earth satellite is affected by surface configuration, radi-

**LIST OF SYMBOLS**

- $A$ Surface area
- $A_H$ Horizontal projected area
- $E$ Earth infrared heat flux
- $F_{ij}$ Black body radiation shape factor from node $i$ to node $j$
- $M$ Thermal mass
- $P$ Percent sun
- $P'$ Payload power dissipation
- $R$ Earth reflected solar radiation
- $S$ Heat flux from sun
- $T$ Absolute temperature
- $t$ Time
- $\alpha$ Solar absorptivity
- $\beta$ Solar aspect ratio
- $k_{ij}$ Conductance from $i$ to $j$
- $\eta'$ Node solar radiation conversion efficiency
- $\varepsilon$ Infrared emissivity
- $\overline{G}$ $\frac{1}{A}\int_A \frac{\exp(-\eta^2)}{2s\sqrt{\pi}}\,dA$
- $\overline{F}$ $\frac{1}{A}\int_A \frac{\eta(1-e\gamma f\eta)}{2s}\,dA$
- $C$ Accommodation coefficient
- $N$ Number of molecules/unit volume
- $\sigma$ Stefan Boltzmann's Constant
- $T_o$ Free stream temperature
- $u$ Mass velocity
- $\overline{Q}$ Average heat transfer/unit area
- $\gamma'$ Ratio of specific heats
- $T_w$ Temperature of wall
- $s$ Ratio of most likely molecule speed to spacecraft speed
- $\eta$ $S\sin\theta$
- $\theta$ Angle of incidence
- $\gamma$ Earth aspect ratio

Average values denoted by a bar.

| | RADIATION ARRIVING AT SPACECRAFT | | | |
|---|---|---|---|---|
| | SOLAR BTU/HR | REFLECTED SOLAR BTU/HR | EARTH INFRARED BTU/HR | PAYLOAD HEAT BTU/HR |
| 100% SUN | 5050 | 350 | 590 | 340 |
| 67% SUN | 3330 | 360 | 590 | 0 |

Table I—Heat inputs to spacecraft.

ative properties of the surface, internal heat dissipation, spacecraft orientation, and orbit parameters.[4]

The heat dissipation term in the heat balance equation is smaller than the other terms but is significant in establishing the radiation coupling required between the spacecraft interior and the surface. For Isis A, high emissivity surfaces were required.

Passive thermal control is used in the Isis A spacecraft. A passive design is one in which the selection of surface radiation characteristics is such that the spacecraft will operate within an acceptable range of temperature limits. The alternative, active control, implies the use of devices which will function in space to control a heater, cooler, spacecraft attitude, radiation characteristics of surfaces, or any combination of the above. Passive control is inherently simple and the reliability problem associated with actuated components is not present; however, no measures can be taken to compensate for changes in surface properties with time and variations in spacecraft attitude or orbit parameters. In this connection it is virtually impossible to effect a totally passive design for space missions where the distance from the sun to the spacecraft will vary appreciably during the mission. With active control, it is possible to compensate for variations in the factors mentioned above. Most earth-orbiting scientific satellites have passive temperature control designs, whereas deep space probes have semi-active or fully active designs and manned spacecraft have complex active control.

## CALCULATIONS

### Orbital Temperatures

The thermal design based on average absorptivity, emissivity, areas and heat fluxes produced a temperature range of $+2°C$ to $+35°C$. As the desired temperature range is $-5°C$ to $+40°C$, there is a little in reserve, so knowledge of variations of temperature with location in the spacecraft, and with time, was needed. The heat-flow equation (6) can be expanded to include the interactions of thermal masses and the dynamic effects:

Fig. 1—Radiation inputs to spacecraft.

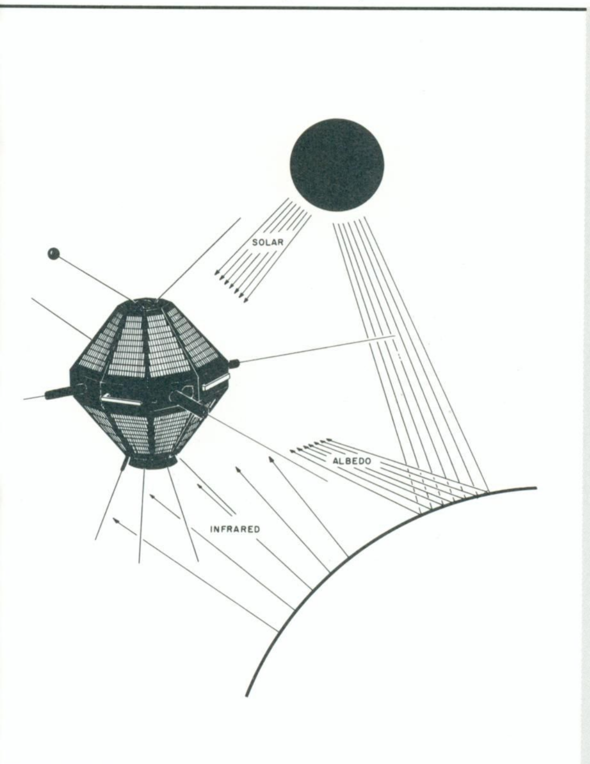



Fig. 2—Insulation.

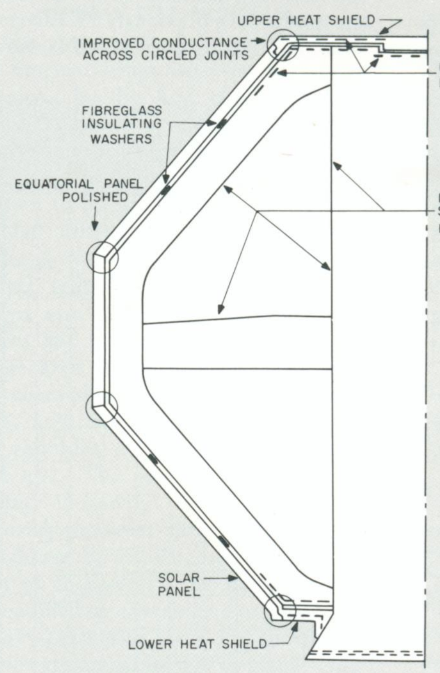
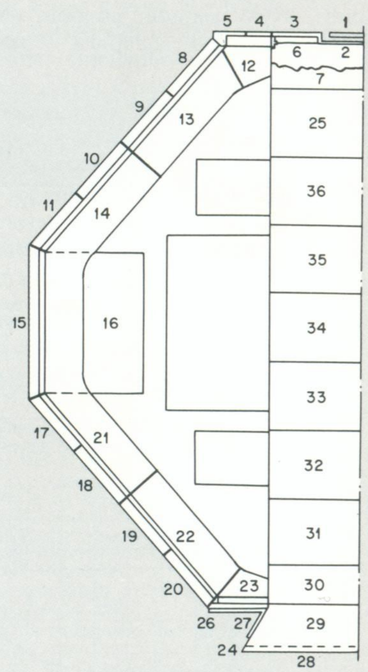



Fig. 3—Nodal divisions.



**ISIS A SURFACE PROPERTIES**

| | | $\alpha$ MIN | $\alpha$ MAX | $\epsilon$ | $\frac{\alpha}{\epsilon}$ MIN | $\frac{\alpha}{\epsilon}$ MAX |
|---|---|---|---|---|---|---|
| END PLATES | 86% EVAPORATED ALUMINUM | ·10 | ·15 | ·025 | ·81 | 1·27 |
| | 14% WHITE PAINT | ·20 | ·35 | ·85 | | |
| EQUATOR | POLISHED ALUMINUM | ·15 | ·25 | ·03-·04 | 3·7 | 8·3 |
| SOLAR PANELS | | ·735 | ·77 | ·7 | 1·05 | 1·10 |

Table II—ISIS A—surface properties.

$$M_i \frac{dT_i}{dt} = A_i\alpha_i(1-\eta_i)[S^{(t)}\beta_i^{(t)} + R^{(t)}\gamma_i^{(t)}] + A_i\gamma_i^{(t)}\varepsilon_i E^{(t)} + P_i' - \kappa_{ij}(T_i - T_j) - A_i F_{ij}\sigma(T_i^4 - T_j^4) - A_i\varepsilon_i\sigma T_i^4 \quad (7)$$

The equation is one of a number of first-order non-linear simultaneous differential equations with variable coefficients. This equation assumes that the solar cells have a constant conversion efficiency, which is approximately true if there is no overchange limiter. On Isis A there is no overcharge limiter except in certain failure conditions.

A set of equations similar to Eq. (7), can be solved numerically. If the variable coefficients change cyclically with time, the numerical procedure is iterative; if the initial conditions are unknown and the steady state for a given environment is sought, the calculation is again iterative. If the starting conditions are known, the temperature-time history can be followed. The calculation method is to start with a set of temperatures and calculate the resulting heat fluxes. From these, the rate of change of temperature with time can be found, and so the new temperatures at the end of the short time interval can be used for the integration.

A computer program was written to solve these equations. It has capacity for 100 nodes, each connected to all the other by conduction and radiation. The program can be used for predicting:

1) temperature distribution in a spacecraft;
2) temperature distribution in a piece of electronic equipment in vacuum; and
3) temperature distribution of a spacecraft during thermal vacuum and solar-simulation testing.

The spacecraft can have time-varying earth-infrared and albedo inputs, solar and earth aspect ratios, and payload heat dissipation. It can be spin stabilized, gravity gradient stabilized, or programmed for an arbitrary sequency of orientations. Active open-loop or closed-loop temperature control can be simulated, where surface properties change at pre-set times or temperatures. It follows that the program can be used for many different types of spacecraft employed in various missions. The use of this program showed that three types of insulation had to be added for high thermal resistance between the exterior "polar regions" and the thrust tube (Fig. 2). A heat shield was added over the end decks, with the annular slot antenna in the center. The surface properties of the heat shields were kept the same as the decks were previously. Radiation insulation blankets were added as shown in Fig. 2. They are made up of many layers of aluminized mylar, crinkled to minimize the contact areas between layers, and so eliminate the need for fiberglass spacing sheets which were used in previous designs. Fiberglass insulating washers are fitted between the panels and ribs.

To simulate this design, 36 nodes were used (Fig. 3). With these subdivisions, the estimate of internal radiative heat flux becomes important. The problem is greatly simplified by painting all internal surfaces, thus providing them with an infrared emissivity of about 0.85. Hence emission is diffuse, reflections are assumed zero, and all that remains is the calculation of the radiation interchange shape factors. Painting all exposed inside surfaces also reduced thermal resistance from the thrust tube to the shell and the effect of variations in payload heat dissipation.

The final surface was defined as in Table II. These finishes are assumed to be relatively stable in space, with the amount of expected variation shown as a tolerance on absorptivity.

Absorptivity variations in solar panels are not as well known as the variations in white or black paint or polished or evaporated metals. This is partly because a solar panel is a complicated composite surface. For the surface finishes of Table II, the calculated temperature extremes are now $-5°C$ to $+40°C$. These extreme temperatures occur in different orbital situations. The 40°C case is 100% sun, maximum albedo, maximum power, with the spin axis pointing at the sun. The $-5°C$ case is 66% (minimum) sun, minimum power and albedo, with the spin axis pointing at the sun. The variation in each orbit is shown in Fig. 4. The effect of the albedo radiation on the cold side of the spacecraft can be seen clearly.

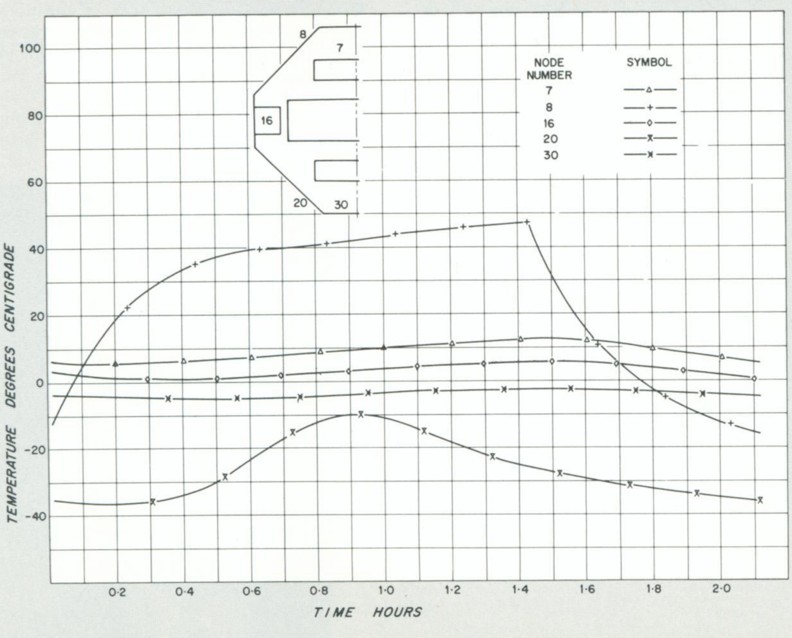
The heat fluxes used in temperature estimates are found using computer programs which calculate percent sun, earth albedo radiation, and infrared radiation with respect to time.

Quadratures for radiation coupling coefficients were evaluated numerically using an IBM-7044 digital computer. The spacecraft is assumed to be a cylinder contained inside a sphere as shown in Fig. 5. Radiation heat exchange between two surfaces in a black enclosure is a function of the surface temperatures and a geometric shape factor. The shape factor between two area elements is given by:

$$dF_{ij} = \frac{\cos\beta_i \cos\beta_j dA_i}{\pi L^2} \quad (8)$$

where $\beta_i$ and $\beta_j$ are the angles between the area element normals and the line $L$ joining the two area elements. Integra-

Fig. 4—Minimum sun temperature history.



tion over the two areas yields the exchange factor between the two finite areas. The equation in Fig. 5 gives the interchange integral equation between thrust tube and external panels. Additionally the interchange factors are required for nodal areas of panels since they "see" each other and themselves. For this case, the shape factor between two circumferential segments of a sphere is simply:

$$\frac{\phi_o}{2\pi}(\cos\theta_{jA} - \cos\theta_{jB}) \qquad (9)$$

where $\theta_{jA}$ and $\theta_{jB}$ are the values of $\theta$ which define any area $A_j$ (Fig. 5). This stems from the fact that any finite area inside a sphere $A_j$, 'sees' all other area elements under the same solid angle.

The same method of numerical temperature prediction has been applied to some electronic boxes in the spacecraft e.g., the converter box. If the battery short circuits, this converter must dissipate 18 watts. The simulation was done by assuming a radiation sink temperature to correspond to the average interior temperature of the spacecraft, and a conduction sink temperature corresponding to the average thrust tube temperature. Sixteen nodes are used to make a numerical model of this box. The two transistors mounted on the flange, which is integral with the base, produce the heat. Hence, the main heat path is through the base to the thrust tube. However, the alternative paths to the sides and top of the box, and to the spacecraft shell by radiation, have an appreciable effect on the design. This type of analysis is used to decide the thickness of the flanges and base of boxes which are required to dissipate a large amount of heat.

#### Launch Window Temperatures

The launch window is set by dynamic and thermal considerations (only the latter being considered here). A certain range of "percent sun" is permissible; thus, a program has been written to calculate percent sun for a launch at any time during any day of the year. This is a calculation of the points of intersection of the orbit plane with the earth-shadow cylinder, and the corresponding time in shadow, for the full range of possible launch times.

Ascent heating has to be calculated, to see what combination of times spent in sun and shadow are necessary for the spacecraft temperature to stay within specified limits. The ascent heat calculation is in three parts: 1) the heat fluxes from the earth and sun with corresponding view factors, 2) aerodynamic heating, and 3) calculation of heat fluxes and temperature within the spacecraft.

The earth and solar heat flux calculations are similar to those already described for inputs to an orbiting spacecraft. The solar aspect ratio is now a function of the time of day, the day of year, and the position in the launch trajectory. As the spacecraft may not be spinning for some time after shroud release, it is necessary to calculate solar and earth aspect ratios without assuming axial symmetry. This means a large number of nodes must be used to describe the circumferential and axial heat flow.

The aerodynamic heating considered is in the free molecule flow region only as, in general, the spacecraft is protected by a shroud until this region is reached. The equations governing the heat transfer are:[6,7]

$$\frac{\overline{Q}}{CN\sigma T_o u} = \frac{\gamma'+1}{2\gamma'-2}(\overline{G}+\overline{F})\frac{T_w}{T_o} - \left(s^2 + \frac{\gamma'}{\gamma'-1}\right)(\overline{G}+\overline{F}) + \frac{\overline{G}}{2} \qquad (10)$$

The computer program for this calculation will include the 1959 ARDC Model Atmosphere, and the input will include a description of the launch trajectory. At shroud separation, spacecraft temperature distribution is known. The radiative earth and solar heat fluxes, the earth and solar aspect ratios, and the free molecule flow heating (which is dependent on the spacecraft temperature and velocity, and the local atmospheric properties) are calculated. This information will then be fed into the multi-node thermal program, which will calculate new temperatures for the next time interval. This will be repeated, providing a history of temperature with time for each node.

### THERMAL TEST PROGRAM

The test program to check out the spacecraft thermal design involves three main activities. In the first check, exterior surface finishes are selected by spectral reflectance measurements. Where necessary, samples are exposed to ultraviolet radiation to investigate possible changes in solar absorptivity. For the solar panels, it was not possible to use the spectral reflectance technique because of their size (approximately 18 inches square) and the composite surface. The surface area is made up of 74% solar cells covered by adhesive, cover glass, filters, and an anti-reflective coating; 13% sandblasted aluminum; and 13% black RTV. The solar absorptivity and infrared emissivity of this panel was measured in a solar simulation chamber at Goddard Space Flight Center. As a result of those tests, the panel surface was modified by covering an exposed area of epoxy glass with black RTV and changing the existing white RTV to black. Subsequent to this, it has been discovered that the RTV used (GE-511 and GE-577)

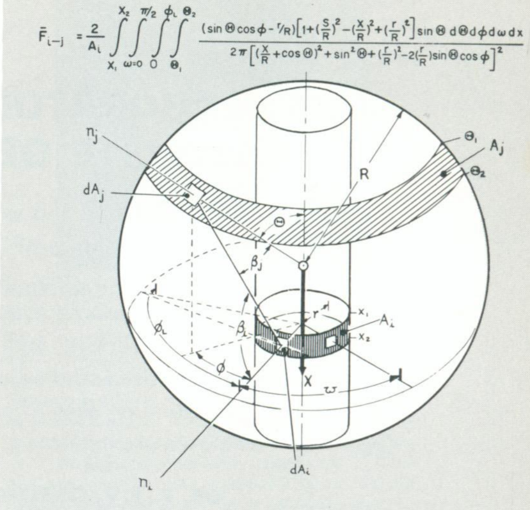



Fig. 5—Internal shape factors.

gives off up to 5% by weight of silicon oil in vacuum. Temperature cycling techniques to minimize this outgassing are currently being investigated. Meanwhile spectral reflectance measurements on polished aluminum coated in silicon oil are being conducted to establish the effect of the silicon oil on the thermal design.

The second type of thermal testing which has been undertaken is to investigate regions in which the heat flow paths are doubtful. One region of doubt is where sheet metal or thin sections are bolted together. The only satisfactory method of establishing the contact thermal resistance value is by measurement. This has been done on a sample solar-panel-to-equatorial-panel bolted joint. The effect of adding indium foil was measured, and it was found to be an appreciable advantage. The test was done by measuring heat flow and temperature drop across a joint in vacuum.

The thermal design will be acceptance tested in a solar simulation facility at Goddard Space Flight Center. This will check out the spacecraft at the extremes of orientation and percent sun. The resulting spacecraft temperatures will not be directly representative of what is expected in space. As a result, spacecraft temperatures in the solar simulation chamber will have to be predicted, and compared with measured results.

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# LABORATORY STUDIES OF SATELLITE DESIGN PROBLEMS

This paper considers some of the satellite engineering problems and constraints which develop as a consequence of the orbital environment, the experiments, and portions of the spacecraft system. With reference to topside sounding satellites such as ISIS A, potential problems associated with spacecraft geometry, solar-cell panels, DC-to-DC converters, and the experiments themselves are discussed. These problems have been studied under scaled laboratory conditions, contributing design information on alleviation techniques where necessary. Other uses of simulation in this area are indicated.

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In considering a spacecraft to be used as an earth satellite for scientific investigations, the various design areas may be conveniently classified under five very general headings:

1) Mechanical provision for experiments;
2) Communications system for command and data transmission;
3) Power;
4) Spacecraft orientation (attitude) monitoring and usually control; and
5) Spacecraft environment.

The first four of these classifications involve what may be termed (for the purposes of this article) conventional space engineering; the last classification —the problems associated with the environment—includes many problems that are also the province of the engineer: thermal balance throughout the vehicle, behavior of components under space conditions; environmental changes during launch. As a subdivision within this general classification, however, there aie many potential problems arising from the satellite motion through a tenuous magnetoplasma, i.e. through an electrically neutral system of positive and negative charged particles (usually positive ions and electrons) that includes an associated magnetic field. Such problems are generally within the realm of the plasma physicist.

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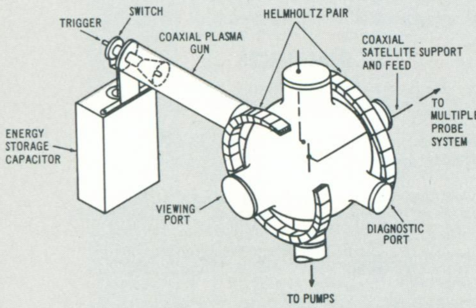



Fig. 1—Schematic diagram of apparatus.

Since the satellite is moving through a magnetoplasma, interactions between the two will occur, which may have deleterious effects. These interactions will usually give rise to some form of plasma sheath about the vehicle, i.e. a region wherein the plasma is perturbed.

As an illustration of this effect consider what happens when a body is placed in a plasma. The plasma consists of approximately equal concentrations of (generally) positive ions and electrons. These particles have equal energies and, since there is a large discrepancy in their masses, this means that the electrons have, by comparison, a very high mobility. Thus, more electrons will strike the body than positive ions. Since the body is not allowed to act as either a source or a sink for electric charge, the electrons build up a negative charge just sufficient to decrease the flow of further electrons so that it equals and cancels the ion flow, and equilibrium is established. However, the potential of this electrically floating body is now slightly negative with respect to the plasma (i.e. the 'floating potential' is negative with respect to the 'plasma potential'). The decrease in the electron flow caused by this potential difference, or electric field, means that the region around the body has fewer electrons than the normal plasma.

Now it can be seen that if this body has some form of plasma measuring device mounted on its surface, it is possible for this diagnostic to monitor the conditions prevailing inside the plasma sheath, instead of measuring the parameters of the ambient plasma.

Generally speaking, a sheath such as that described is of small dimension and it is possible to design diagnostic devices such that they effectively probe through and investigate conditions beyond the sheath. However, it must be appreciated that the situation described, though accurate in itself, is very much a simplification of the satellite situation. Consider a small spherical satellite in earth orbit in the upper ionosphere. First it may be said that the vehicle will be surrounded by a 'floating' sheath, as described above. However, since the body is now a satellite, account must be taken of the high relative velocity between the satellite and the plasma. Usually this velocity is high with respect to the ion thermal velocity (at least for near earth orbits) and low with respect to the electron thermal velocity. Thus the front surface of the satellite 'sweeps up' a 'ram current' of ions, leaving a complex wake in the rear of the vehicle, and so modifying the simple sheath picture. To this picture must be added the effects produced by the interaction of the moving, conducting satellite with the earth's magnetic field **B**.

These effects become most pronounced when the satellite velocity vector **V** and the magnetic field **B** are perpendicular, when an electric field **E** is produced perpendicular to both **V** and **B** (the **VxB** effect). This produces a sheath analogous to the floating sheath, but since the potential difference between the body and the plasma now varies across the surface of the body (parallel to **E**) the sheath so formed also varies.

It must be realized that none of the above sheaths are produced as a steady phenomenon, but rather as effects which vary depending upon variations in a number of parameters including satellite orientation, magnetic field, and velocity.

## LABORATORY EXPERIMENTS

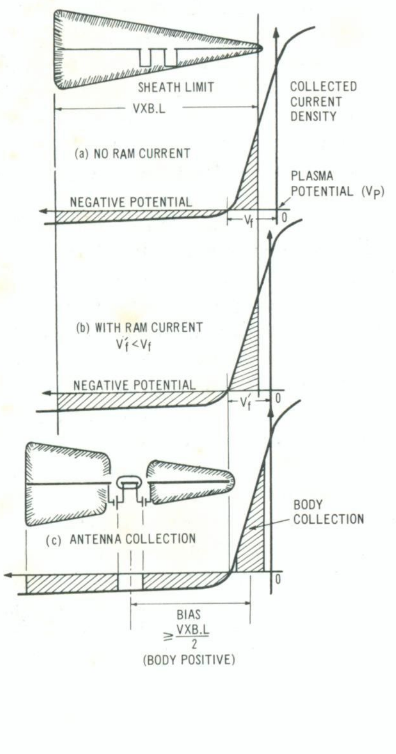
The two experiments to be described both deal primarily with aspects of the **VxB** sheath: in the first instance, a technique is developed whereby the effects of the sheath on measuring equipment can be minimized; in the second, it is shown how this **VxB** effect can lead to interference in the operation of the spacecraft.

### VxB Effect

Since this work was performed for the Defence Research Telecommunications Establishment of the Defence Research Board of Canada, the interest was biased towards the problems of the Canadian satellites of the ALOUETTE and ISIS series. The primary experiments associated with these satellites are the sounding experiments whereby the upper ionosphere is investigated by probing with electromagnetic waves (100 kHz to 20 MHz), in a technique somewhat similar to radar. Satellites are used for these measurements because of their mobility and accessibility. Since the upper levels of the ionosphere respond to electromagnetic waves of a frequency which will not pass through the layer situated below these upper levels they cannot be investigated from the ground, and so must be probed from above, or the topside, producing the generic name 'topside sounder' for such satellites. A characteristic of such topside sounder satellites is that they carry very long sounding antennas (Isis A has approximately a spherical body of one meter diameter, and main antennas 36.6 meters long, i.e. ~73 m tip-to-tip). Thus, because the total **VxB** voltage varies directly as $L$ (~70 m for Isis A), even though the magnetic field $B$ may be small, the velocity **V** is large and under certain conditions of satellite orientation and velocity the total **VxB**·$L$ voltage developed can be as much as 20 volts. It can be shown that the maximum potential difference occurs at one end of the antennas. However, a potential difference of 10 volts developed about the middle of the antennas (i.e., on the body) is still sufficiently large that it can produce a sheath capable of disturbing the direct measurement experiments located on or within ~1 meter of the body.

Such a situation was studied in the laboratory using an apparatus in which the plasma/satellite interaction is simulated. Models of the satellite were suspended in an evacuated interaction chamber and a plasma produced by a two-electrode coaxial plasma gun was allowed to flow down a flight tube into the interaction chamber and around the model. A magnetic field was produced by a Helmholtz coil arrangement set up external to the interaction chamber, as shown in Fig. 1. Thus, a plasma flow was produced with a magnetic field **B** arranged to be perpendicular to the plasma velocity vector **V**, producing an electrical field **E** perpendicular to both **V** and **B** and parallel to the antennas of the model satellite. This arrangement produced the field **E** in the plasma, leaving the satellite a unipotential body, a situation analogous to the space condition of an electrically field-free plasma and a potential gradient restricted to the vicinity of the satellite.[1]

Fig. 2—Langmuir probe analogy.



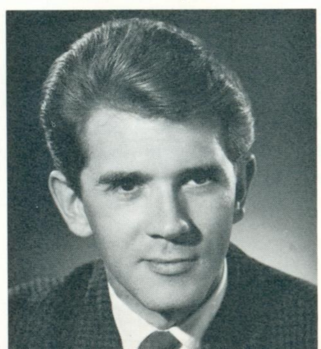

The first consideration with such an apparatus, since it is not a true but a scaled simulation, is how accurate is the scaling? To evaluate this it is necessary to determine first the scaling laws that apply and then determine values for a number of laboratory plasma parameters and compare them with the predictions derived from the scaling laws. These scaling laws were investigated by Shkarofsky[2] who showed that the plasma parameters required to simulate a typical orbit were a density of $\sim 10^{10}$ to $10^{11}$ cm$^{-3}$, at a temperature of <1 eV, and a velocity of ~2 cm/μs through a magnetic field of ~350 gauss. These requirements were met reasonably well.[1]

The techniques used to determine the plasma parameters included Langmuir probes to determine the plasma temperature and also, from a variation in the floating potential across the stream, the **VxB** field; 9-GHz microwaves for the variation of plasma density with voltage on the plasma gun; and, using a gaussmeter to measure the magnetic field, the previous electric-field measurement was used to measure plasma velocity.

Having established the parameters of the pulsed plasma stream and ensuring that they met the scaling requirements, the model satellite (consisting of a spherical body with two extended antennas) was placed in the interaction chamber. Considering the Langmuir characteristic of the plasma and assuming the satellite to act in an analogous fashion led to the conclusion that a bias voltage applied between the body and antennas would minimize the sheath problem in the body region (Fig. 2). For simplicity, the body and the antennas are considered to be of the same cross sectional area. The varying potential difference between the body and the plasma, represented by the absissa, is considered as a gradient along the body, and the current density

**FRELEIGH J. F. OSBORNE** was educated at the Royal Canadian Naval College (1946-48); obtained his B.Sc. at McGill University in 1950 and was awarded his M.Sc. at Laval University in 1951 for a thesis on Application of the Secondary Electron Emission Multiplier to a Mass Spectrometer. In 1954 he received a D.Sc. from Laval University for a thesis on Secondary Electron Emission of Beryllium Copper. He joined the Research Department of Canadian Marconi Company in 1954 as a Senior Physicist, working primarily in the fields of component reliability, systems, and instrumentation. In 1956, he was made a supervisor and as such directed a variety of projects. He transferred to the Electronic Tube Plant where he developed an S-band Electron Beam Parametric Amplifier. In January 1961, he joined the Research Laboratories of RCA Victor as a Senior Member of Scientific Staff. He has since been particularly active in the area of plasma measurements and techniques, leading and contributing to programs on plasma diagnostics, plasma loaded microwave structures, laboratory simulation of geophysical phenomena, and laboratory studies of the interactions between a satellite and its plasma environment. In 1965, he became Director of the Microwave and Plasma Physics Laboratory, now the Plasma and Space Physics Laboratory. He was active in the Royal Canadian Navy (Reserve) from 1946-1956, is a member of the Canadian Association of Physicists, the American Physical Society, and the American Geophysical Union. He is Chairman of Commission IV of the Canadian National Committee of the International Scientific Radio Union (URSI), a member of the National Research Council of Canada Associate Committee on Radio Science and the Defence Research Board of Canada Advisory Committee on Physics Research. He has been associated with McGill University teaching a class in plasma physics, and is listed in American Men of Science.

**MICHAEL KASHA** received the B.Sc. in Physics (Honours) in 1957 from London University. After a short time as a Lecturer in Mathematics and Physics at the Harrow Polytechnic, he joined the United Kingdom Atomic Energy Authority (UKAEA) at Harwell, to work in the Controlled Thermonuclear Reactions Division. There he was engaged on plasma physics studies, working on the Spider experiment, an investigation of shock waves in a plasma. This work entailed the design, construction and operation of 40 kV, 20 kilojoule device. In 1961 he transferred to the Culham Laboratory of the UKAEA, to work on Tarantula, a similar experiment using a 100 kV, 100 kilojoule machine. He was concerned with the development of a 100 kV, 1.4 million ampere fast-acting (20 nsec) switching system and various other aspects of the design and construction. He devised and developed various diagnostic devices (e.g., multicoil magnetic probes, optical interferometers, and time resolved spectrometers). Mr. Kasha joined the Research Laboratories of RCA Victor Company, Ltd. in 1964, since which time he has been engaged in studies of satellite interaction with the ionosphere. He is an Associate of the Institute of Physics in London.



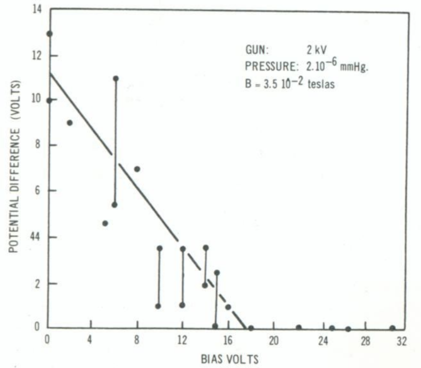



**Fig. 3—Variation of plasma/vehicle potential difference with bias.**

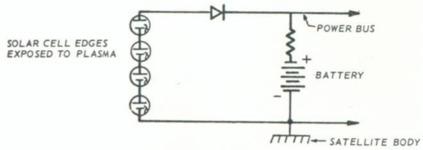



**Fig. 4—Power supply.**

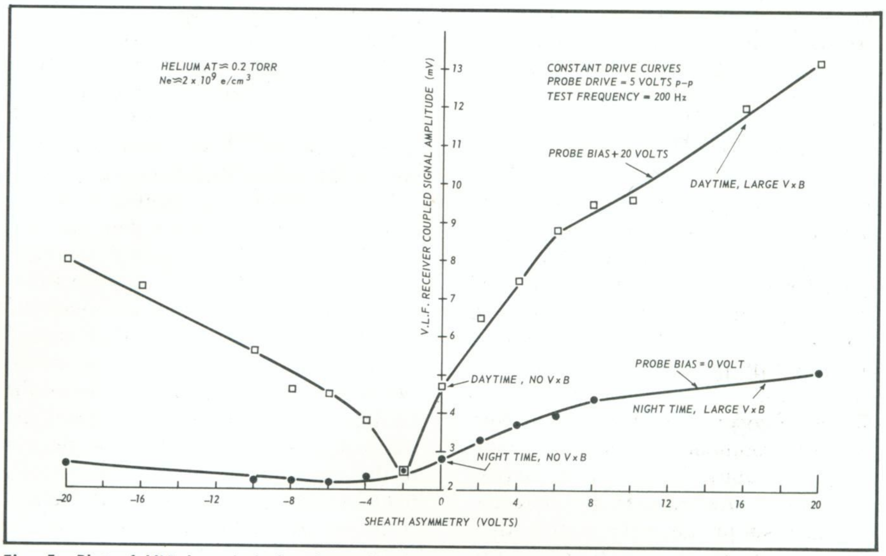



**Fig. 5—Plot of VLF 'coupled signal amplitude' versus sheath asymmetry voltage and probe bias, under constant amplitude probe drive voltage.**

collected at each part of the body is represented by the ordinate (electron current shown as positive). The addition of a ram-ion current, caused by the 'sweeping up' of ions discussed previously, has little effect on this curve.

It can be seen that the smallest sheath is associated with the smallest body/plasma potential difference, and with the electron collection region. Thus a bias applied as shown, causing the body to become the electron collecting portion of the system, should ensure that the body has the smallest sheath

To test this concept, a model satellite was constructed so that a variable potential bias could be applied between the body and the antennas. As was mentioned above, the plasma sheath is associated with the potential difference between the plasma and the body, and so this was used as a diagnostic for the sheath. The plasma potential (in fact 'floating' potential) was measured by a probe located in the plasma, near the model but outside any possible sheath effects. This potential was measured and compared with the potential of the body of the satellite. The difference between them was plotted as a function of the body/antenna bias (Fig. 3), and it can be seen that this difference goes to zero, signifying a minimum sheath for a bias $\geq 18$ volts which corresponds to $\mathbf{VxB}\cdot L/2$ as predicted in Fig. 2.

The experiment thus shows that the bias concept is effective, and calculations of the DC-to-DC converters used on the given in Osborne and Kasha[1] show that the system is easily applicable to a satellite of the ALOUETTE or ISIS type, for a very small power cost ($\sim$20 mW). This technique is being applied in the ISIS A spacecraft.

### VLF Interference

Analysis of the output of the VLF receiver on ALOUETTE I showed the presence of signals at the frequency (and harmonics) satellite power supply. This is an obvious form of interference but, since it has been observed to mix with the natural signals and form combination tones, and since the level of the automatic gain control of the VLF receiver can be set by the level of interference signals, it is virtually impossible, at times, to discriminate between the true signal and the interference, or even to know the correct signal level when this discrimination is possible. Inductive interference, or 'pick-up' is of course a common problem in circuit design of all kinds, and this was borne in mind during the design of the satellite system, and was moreover checked out before launch, when it was found to be detectable, but at a level considered to be not important. It appears that the interference really developed when the satellite was in orbit. It must also be noted that this interference did not occur at a constant level, but showed a definite variation. Examination of the data from the ALOUETTE led to certain patterns being established for the behaviour of this interference: variations associated with satellite being in the dark or light part of its orbit with a sharp discontinuity at satellite dawn, and also variations which could be linked to the satellite orientation with respect to a plane defined by the satellite's velocity vector and the earth's magnetic field, i.e. a possible dependence on the presence of a **VxB** effect. Such patterns led to the conclusion that the interference was in some way associated with the plasma around the vehicle, and hence a possible mechanism, based on such an assumption, was developed and tested.

Specifically, any proposed solution had to explain the following observations:

1) With the satellite in the dark, aligned so that no $\mathbf{VxB}\cdot L$ potential gradient exists along the antennas, no converter interference is found in the VLF receiver output;
2) With the satellite in the dark but with **VxB** gradients, negligible effects were noted;
3) With the satellite in the light but no **VxB**, no interference; and
4) With the satellite in the light and aligned for a **VxB** gradient, large interference signals are found.

Fig. 4 shows a simplified power supply system for ALOUETTE I. A series-parallel array of solar cells with their terminals exposed to the surrounding plasma supply (via a diode) power to the batteries and various satellite systems. Where the power supply impedance is not zero, the chopping frequency of the electrical DC-to-DC converter will appear on the main power bus. Now when the satellite is in the dark, the solar cells are at the body potential and the diode is biased off to prevent battery drain, and so any AC signal on the power bus is unable to reach the solar cells. When the satellite is illuminated, however, the diode has a low impedance, allowing the converter signal to appear across the solar cells, which are now raised to various positive DC potentials. It remains to determine how this signal will appear on the VLF receiver antenna with **VxB** modulation.

The VLF receiver antenna is a large dipole, coupled via a transformer, to the VLF receiver in such a way that any alternating current of the appropriate frequency passing through the transformer between the two halves of the antenna, or unequally between each half antenna and the satellite body, will appear at the VLF receiver input. Now a satellite can have associated with it asymmetric **VxB** plasma sheaths, in such a way that all the electron collection required to keep the system electrically floating takes place at one point, usually one end of the antenna array (Fig. 2). Now if there is an AC signal on the exposed edges of the solar cells, which are themselves raised to a positive potential by their operation, the solar cells will collect a varying electron current. Then, to maintain the floating condition, a variation of opposite phase must occur in the electron current collected on the antenna, which immediately introduces an asymmetric signal through the transformer to the body, as discussed above, and consequently introduces an interference signal at the input of the VLF receiver. In the light of this proposed mechanism, consider now the four conditions previously set forth.

1) Satellite in the dark, with no **VxB** gradient: no converter signal is present on the cells since the diode is biased off. Any residual signal would affect the antennas symmetrically and there is thus no interference at the receiver.
2) Satellite in the dark, with **VxB** sheaths: The **VxB** gradient is such that the body (and cell) potential is approximately **VxB**·$L/2$ below the plasma potential and the AC signal is not sufficient to raise the solar cell potential into the electron collecting region, and thus the cells are free to change their potential without affecting the current collection pattern.
3) Satellite in the light, but no **VxB** effects. The solar cells are now at a positive potential, and so a variation of electron collection develops on the antennas, but it does so symmetrically and so no interference is found.
4) Satellite illuminated and suffering **VxB** potential gradients produces the correct asymmetry for the converter signal, to affect the electron collection and thus induce a modulation in the electron current collected by one antenna and so produce an asymmetric signal which can appear at the VLF receiver. The converter signal appears on the exposed edges of the solar cells which are now at a large positive potential with respect to the plasma potential.

Such a situation was simulated in the laboratory, where an equivalent satellite system, effectively containing a transmitter transformer, antennas, variable **VxB** antenna sheaths, and an exposed solar cell terminal (which could be raised to a DC potential and have impressed upon it a converter signal) was placed in a stationary helium plasma. The results of this simulation are shown in Fig. 5 which shows the magnitude of the detected interference signal for the four conditions discussed above. The lack of symmetry and the fact that the minimum does not occur at zero **VxB** potential are due to inherent coupling introduced by the model and circuitry unbalance. Such a picture illustrates well the operation of the proposed mechanism, and agrees well with the spin modulation data from the satellite.[3]

## SUBSIDIARY WORK

The previous section has dealt in some detail with two laboratory investigations of satellite/plasma interactions to give some idea of the method of approach used and the techniques employed by the plasma physicist in dealing with such problems. The techniques can of course be applied to a broader field than that implied by the above examples.

Kasha and Johnston[4] have used very similar techniques to investigate the response of a plasma diagnostic device mounted on a satellite as a function of satellite orientation. A gold plated sphere was used to simulate a satellite, with a total current monitor, or collector, mounted on the surface, the whole being situated in a pulsed flowing plasma. By rotating the sphere around an axis perpendicular to the plasma velocity vector, the response of the current monitor was determined as a function of satellite orientation in such a manner that the effects of the ram current and the wake found behind the simulated vehicle could be clearly seen. As a further experiment a magnetic field was produced over the volume of the plasma stream/satellite model region, parallel to the axis of rotation of the model. This produced a **VxB** potential gradient effect which was now superimposed onto the previous response curve. This second response curve, now highly asymmetric where before it had been symmetric about the plasma velocity vector, was determined experimentally, and found to compare well with that of the EXPLORER VIII satellite.

Other aspects of this satellite work have included a study by Graf and Jassby[5] of the behaviour of antennas in a plasma. The impedance of an antenna is usually determined by its geometry and the dielectric constant of the surrounding medium. Should this antenna be mounted on a satellite, it will find itself in a plasma, i.e. a medium whose dielectric constant may change, either with antenna position (different locations within the satellite orbit) or time. Obviously such changes will bring about changes in the antenna impedance which are related to the plasma parameters.

Such studies as this are useful both to the engineer, since they illustrate such phenomena as a varying antenna impedance, and to the physicist, since they indicate new methods for determining the parameters of the space plasma.

At present the RCA Victor Research Laboratories are conducting some preliminary investigations into the phenomenon of 'transmitter-induced sheaths'. These plasma sheaths are formed around transmitting antennas under the influence of the antenna operation, i.e. the presence of a pulse of varying high voltage. The main point of interest is not so much the creation of such sheaths, but the prospects for their persistence once the generating signal has been removed, since they can be of such magnitude as to seriously perturb the operation of the satellite or its experiments. The sheaths, and their relaxation times, are found to depend markedly on the transmitter frequency and on the ambient plasma. Preliminary results indicate that it may be possible for the relaxation time, under certain conditions, to overlap the time period when the local-plasma diagnostic experiments may be activated, a time of the order of a millisecond. The determination of these relaxation times is complicated by the internal circuitry of the particular satellite under consideration and therefore the study is being continued with the aid of an equivalent circuit, with which it is hoped to simulate, though not directly, the action of the ram ion currents, electron currents, etc.

## CONCLUSION

In this article one of the parts played by the plasma physics laboratory in the essentially engineering problem of designing a satellite has been described. It can be seen that the general approach adopted in the laboratory to these problems is that of simulation, either direct or scale model experiments. The particular scaling laws that must be used for such experiments have been well established, and the techniques that are used to produce plasma of the correct, scaled parameters are being continually improved. Indications have also been given as to the various other fields where work has been, or is, being performed with a view to acquiring a greater understanding of the interactions that occur between a fast moving satellite and the ambient magnetoplasma.

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# ELECTRICAL AND MECHANICAL DESIGN OF THE MILL VILLAGE ANTENNA-RADOME COMPLEX

Requirements for a satellite communication ground station must be determined from the optimum conditions of the global ComSat system. The most important characteristic is the gain-to-noise-temperature ratio (and its fading) which must be optimized for a given cost. Additional requirements are the compatibility with other ground stations, communication reliability, minimum design risk, easy operability and maintainability, low operational cost, and long life. Special requirements were presented for the Mill Village station by the Canadian environment. This paper discusses the electrical and mechanical design of the inflated radome and antenna. The system has full-angle and polarization tracking capability, including orientable linear polarization. The system has been built and has met all the original design and communication performance objectives.

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Ever since the usefulness of microwaves for point-to-point communications was recognized, designers have searched for large antenna gain at economical cost. The advantage of large gain as a means of reducing transmission losses was obvious; nevertheless (for economical reasons), the desired antenna gain had to be balanced with the subsystem characteristics of the entire network. Thus, the development of communication antennas was limited not only by their own problems, but by the characteristics of the related equipment.

#### INITIAL MILL VILLAGE ANTENNA DESIGN CONSIDERATIONS

Orbiting satellites, carrying at least one of the repeater stations, removed the larger part of the propagation path from the vicinity of atmospheric and ground-noise sources. In the case of the down link, the inherent geometry eliminated most of the background radiation from the sensitivity region of the ground-based receiving antenna. The drastic reduction of external noise levels gave tremendous impetus to the development of compatible low-noise antennas and receivers. Under these new circumstances, the antenna gain ($G$) was no longer the overwhelmingly important characteristic, and the gain-to-system-noise-temperature ratio ($G/T_s$) was introduced.

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Usually, the occurrence of a new problem is followed by an early solution based more or less on brute force methods. Earth station antennas were no exception, and some large-gain, low-noise antennas were built, where economy (though important) was of secondary consideration. However, the situation quickly reversed, and new characteristics were defined to measure the cost effectiveness of the design. This effectiveness can be expressed as the $G/T_s$ ratio relative to the weight of the antenna.

During the past decade, the state-of-art in satellite communication has favored a large ground antenna, close to 60-dB gain. The frequency bands have to satisfy the following conditions:

1) for reception—giving minimum combined atmospheric and galactic noise radiation; and
2) for transmission—high enough to permit adequate receive-transmit isolation and low enough to avoid excessive antenna tolerance problems.

When hemispherical angular coverage and high-precision tracking (1/10 of the 3-dB beamwidth) are added to the above defined requirement of effectiveness, the possible antennas are reduced to a very limited number.

In these antennas, without exception, the large aperture area ($\approx$ 6000 ft$^2$) is realized by a parabolic reflector illuminated by a spherical wave. The effectiveness of such an antenna is determined by the shape of the main reflector. As the cost of the large, precision reflector and its associated structure is usually far more than the cost of the source (feed system) a great effort went into the design of source systems which produce nearly ideal aperture distributions.

In a global communication system with many satellites and earth stations, the standardization of the $G/T_s$ ratio is highly desirable. Thus, effective radiated power of the satellite can be selected optimally to obtain economical advantages in the construction and operation of Earth stations.

#### EARTH STATION OBJECTIVES

In 1963, the Canadian Department of Transport decided to build, for Canada, an experimental earth station terminal. The declared objectives of the station were:

1) Create a tool by which specifications and design data for operational ground stations can be generated;
2) Permit participation with other nations in satellite communication experiments;
3) Aid participation with other nations in the development of international operating procedures and standards;
4) Develop Canadian knowledge in supplying operational earth stations for Canada and for international market;
5) Train staff for operational use; and
6) Conduct experiments on various hardware units and system aspects in order to permit adaptation of the station for future operational use.

#### RADOME-ANTENNA REQUIREMENTS

On the basis of the above considerations, the following requirements were established for the antenna.

1) Frequency Band—3960 to 4200 MHz receive, 6160 to 6400 MHz transmit. In actual practice, these frequency bands (limited by the feed system only) were exceeded by a fair margin.
2) Gain—More than 58 dB (in the receiver frequency band) to assure 600 voice channel or one color-TV channel capability, with currently used satellite power. (The gain achieved was approximately 1-dB higher).
3) System Noise Temperature—Less than 65°K at 7.5° elevation above the horizon at 4GHz in clear weather. The measured value on the completed system was a few degrees better than the specification.
4) Angular Tracking—Tracking is generally characterized by angular coverage, accuracy, speed, and acceleration.

**TABLE I—Measured Antenna Gain of the Mill Village Antenna System**

| | Gain (dB) | |
|---|---|---|
| *f (GHz)* | *factory* | *field* |
| 3.9 | 58.6 | 58.0 |
| 4.0 | 59.0 | 59.1 |
| 4.1 | 59.1 | 59.2 |
| 4.2 | 59.3 | 59.4 |
| 6.0 | 61.2 | 59.9 |
| 6.1 | 61.2 | 60.6 |
| 6.2 | 61.1 | 59.6 |
| 6.3 | 61.5 | 59.7 |
| 6.4 | 61.2 | 59.6 |

**TABLE II—Calculated and Measured Antenna Noise Temperature Versus Elevation Angles**

| | Noise Temperature (°K) | |
|---|---|---|
| *Θ (degrees)* | *Calculated* | *Measured* |
| 90 | 28.0 | 34.2 |
| 60 | 28.5 | 33.8 |
| 30 | 29.5 | 36.1 |
| 10 | 39.0 | 44.0 |
| 7.5 | 41.5 | 47.6 |

f = 4081 MHz

The smallest angular coverage, and minimum speed, are required for syncronous satellites. For medium-orbit satellites, full beam steering capabilities are desirable. Therefore, tracking speeds and accelerations were specified according to these types of satellites. The rms tracking accuracy had to be less than 1/10 of the 3-dB beamwidth to make tracking loss negligible (equivalent to about 0.02°).

5) Polarization Matching—To avoid excessive signal loss, the polarization of the "receive" antenna had to be matched to the incoming polarization. Similarly, the transmitter antenna had to be matched to the "receive" polarization of the spacecraft. Generally, orthogonal polarizations were needed between transmit and receive modes to obtain additional isolation between these frequencies. Changeover time from circular to linear polarization was not critical, but alignment of the attitude of linear polarization had to be instantaneous.

6) Weather Protection—The ideal weather protection is a function of the site, but generally the effects of weather cannot be completely eliminated. The magnitude and time distribution of the weather-originated fading (about 3 to 6 dB) can be influenced by the type of antenna protection. In systems under radomes, the main source of fading is precipitation, while in systems without radomes, the main sources are precipitation and wind. The choice of radome-protected or exposed antennas was not clearcut on the basis of $G/T_s$ ratio; the choice was based on operational and life-time considerations.

## ANTENNA ELECTRICAL DESIGN

Effectiveness of the antenna system as previously defined is maximum for symmetrical structures (axially symmetrical paraboloid) which are the most commonly available design. Therefore, it was decided to tailor the rest of the optical system to fit this arrangement; furthermore, the choice of a far-field Cassegrainian feed system was selected on the basis of its advantages in equipment layout, simplicity, and high $G/T_s$ ratio (Fig. 1).

When the Mill Village program began, an RCA experimental feasibility study of a multimode wideband tracking-communication feed had been completed and was the basis of the hardware design. The requirements of the feed (radiating source, Cassegrainian subreflector, and supporting frames) were as follows:

1) Optimum performance-to-cost ratio for the antenna complex (performance is characterized by ratios of antenna gain to noise temperature);
2) Minimum of microwave component types;
3) Large flexibility in operational modes, and coverage of a wide frequency band to accommodate possible changes of frequency requirements;
4) Low circuit loss and difference-mode minimum depth by using highly symmetrical waveguide cross-sections;
5) Good aperture field distribution for the main reflector in the receiver frequency band, while maintaining at least —3-dB antenna efficiency in the transmitter frequency band;
6) Freedom for field alignment of the feed relative to the main reflector; and
7) Both circular and orientable linear polarization.

The first requirement virtually dictates the use of some form of Cassegrainian system, in which the radiating source first illuminates a primary reflector, reflecting the wave toward a secondary reflector. In the simplest form of Cassegrainian system, the radiating source aperture is small and the subreflector is in its far field. Then the subreflector is a symmetrical hyperboloid and the secondary reflector is a symmetrical paraboloid. Mechanical symmetry results in the simplest construction and minimum weight. An 85-foot-diameter solid paraboloid with a focal distance of 36 feet and RMS surface tolerance (half-path-length error) of less than 0.040 inch was selected for the main reflector. Such a paraboloid is easily fabricated and has known mechanical characteristics; it represents a good compromise among weight/aperture-area ratio, rigidity, and weight of feed-support structure.

To illuminate a relatively flat paraboloid with low spillover, requires an illuminating aperture of ten or more wavelengths. Therefore, an 8.5-foot-diameter hyperboloid subreflector with a focal distance of 18 feet was used.

The radiating source system can be operated in a linearly polarized mode and may be oriented in any plane. In principle, this system is relatively simple and, with the exception of the mode filter, does not contain any radically new elements.

The transmitter (Tx) terminal (WR-159) accepts power up to 10 kilowatts CW. Power is divided into two equal parts; each half travels through a cutoff section and a bandpass filter that provide more than 125-dB attenuation below 4200 MHz, and exhibit less than 0.13-dB attenuation for the Tx frequencies. In addition to these filters, the arm connected to the side output of the source system has an adjustable phase shifter using a short-slot hybrid with non-contacting plungers. The outputs of the Tx bandpass filters are connected to the receiver (Rx) bandpass filters at the junctions (S) and (C); these form the duplexing points, with negligible junction effects. Beyond these junction points, both the center and side-arm transmitter power is divided into four equal parts by identical wideband hybrids. (The four-way power division is necessary to introduce monopulse tracking capability for the receiving band.) After these manipulations, the center power and side-arm power are recombined in a four-port orthogonal coupler that has four square-waveguide outputs. Each of these cross-sections support the $TE_{10}$ and $TE_{01}$ modes corresponding to the center and side outputs of the system. Each of these modes is decomposed in the following symmetrical diamond transformers and converted into a left and right circularly polarized wave by the 90° differential phase shifter that terminates the four-port circuit.

The four-port polarizer launches four individual $TE_{10}$ modes into the mode filter, an oversized square-waveguide section having two basic purposes: 1) match the four-port circuit to the multimode horn, and 2) produce a symmetrically tapered field distribution in the aperture of this horn.

An oversized square-waveguide cross-section capable of supporting the $TE_{10}$, $TE_{11}$, $TE_{20}$, $TE_{21} + TM_{21}$, and $TE_{30}$ modes was used. This waveguide, approximately 2 λ long, has a series of crosses at the input end for matching the sum and difference modes and a series of posts at the output end for pattern shaping. The mode filter is followed by a square horn that has a 5 λ aperture size in the transmitting band.

The operation of the system for the received frequencies is basically identical to that for Tx operation, although the direction of propagation is opposite

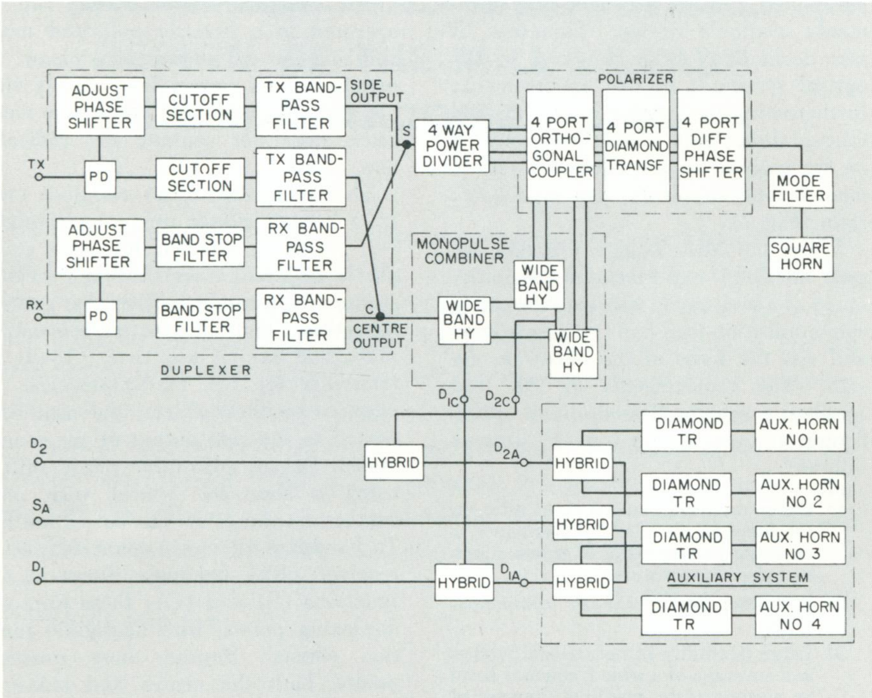


**Fig. 1—Block diagram of the feed system.**

and band-stop filters are used in the receive arm of the duplexer instead of the cutoff waveguide sections of the transmit arm.

The difference-mode operation of the system is somewhat unconventional, mainly because it combines two independent monopulse systems. One monopulse system is realized in the center multimode system and is obtained through the three wideband hybrids of the monopulse combiner. This system is quite effective, even though the difference-mode aperture is only ½ the optimum value. This limitation is removed by the addition of an auxiliary four-horn monopulse system consisting of four circularly-polarized rectangular horns located symmetrically around the multimode square center horn. The horns are diagonally fed, and circular polarization is achieved by the selection of horn length and aperture aspect ratio.

Conversion of the feasibility design study into operational field hardware took approximately 1½ years. Most of the electrical tests were done in RCA Victor's anechoic chamber, and consisted of amplitude and phase pattern recordings of the radiating source, VSWR and isolation tests at the various terminals, and loss measurements through various channels of the system. All measurements had to be conducted in wide-frequency band, and for various polarization modes and polarization positions. Since no scale model of the antenna was ever built, great effort was maintained to predict the performance of the antenna system after the installation of the feed.

The efficiency, noise temperature, pointing, and tracking accuracy of the overall system were calculated using computer programs. An improvement in state of art was achieved in predicting the final $G/T_s$ to within a few tenths of a decibel.

The most important feed system patterns and overall antenna system patterns are indicated in Figs. 2 and 3 respectively. Table I shows the achieved antenna gain. Table II exhibits the noise temperature versus elevation angles for dry weather conditions. It may be noted that the completed system not only achieved the specified gain and noise temperature, but its performance is still unique as compared to 85-ft diameter antennas in the same frequency band.

### MECHANICAL CONSIDERATIONS

The antenna and pedestal combination was designed to meet the specific pointing accuracy requirements and natural frequencies, provide means for data takeoff from each axis, and provide for mounting electronic equipment behind the antenna assembly.

The antenna pedestal assembly consists of a stationary base azimuth- and elevation-rotating assemblies, and the antenna assembly. The stationary base provides for levelling. The azimuth rotating assembly mounted on the stationary base provides trunnion supports for

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**PETER FOLDES** received his diploma of electrical engineering from the Technical University of Budapest in 1950. From 1950 to 1956 he was a research engineer at the Hungarian Telecommunication Research Institute, where he did his postgraduate work. During the same period he lectured at the Technical University. In 1957 he joined RCA Victor Company, Ltd., Montreal. His work covered a wide range of radar and communication systems, microwave transmitters, transmission lines, filters, antennas and propagation. In recent years he has made extensive studies on Cassegrainian feed systems and ground antennas for satellite communications. He is a member of the IEEE.

the elevation rotating assembly which supports the microwave optics.

#### Natural Frequencies

The natural frequencies of a system can be defined as those frequencies where the system tends to oscillate or vibrate when acted on by some external force.

To determine the significant frequencies consider the system as two lumped masses with a connecting spring. A force applied to one mass results in two resonant frequencies: the locked-rotor resonant frequency and the free-free resonant frequency. The locked-rotor frequency is one at which the amplitude of motion at the system input is minimum. The motion at the output does not reflect this decrease in amplitude. At the free-free frequency, however, there is a peaking of the amplitude of motion both at the input and the output of the spring-mass system. It is noted that the free-free frequency is always larger than the locked-rotor frequency. Also, at both frequencies, the decrease in amplitude at the locked-rotor frequency or the increase at the free-free frequency is dependent upon the amount of damping in the system.

These frequencies are discussed here since they are very important considerations in designing an antenna pedestal assembly and drive system. There are many other resonance frequencies for the structure, but they are higher and usually do not affect system performance.

#### Pointing Accuracy

The pointing accuracy for the antenna system may be defined as the angular difference between the direction of propagation of the electro-magnetic wave at the maximum RF power density in the vicinity of the satellite and the angular direction indicated by the position-data output shafts. Pointing accuracy for the antenna and pedestal assembly only may be defined as the peak, mechanical angular-error summation for the azimuth and elevation axes from the theoretical RF beam axis to the output shaft on each axis to which the shaft-position sensors are attached.

A pointing accuracy of 0.17° for the antenna and pedestal assembly per axis was achieved, limited by practical limitations in manufacturing, assembly, erection and measurement accuracies. These limitations in accuracies or errors can be divided into bias or DC errors and random errors. Bias errors such as azimuth and/or elevation axis inclination and feed-horn deflection due to gravity are constant and may be programmed out. Random errors such as effects of wind and/or solar radiation are not constant and cannot be programmed out.

The antenna is driven in azimuth by two 5-hp servo motors counter-torqued to eliminate backlash in the gear trains; the antenna is driven in elevation by one 5-hp servo motor. Backlash was eliminated by providing sufficient unbalance about the elevation axis on the counterweight side so that the teeth of the drive pinion were always in contact with those of the elevation bull gear.

The parabolic reflector is 85 feet in diameter and has a focal length-to-diameter ratio of 0.42. The reflector surface consists of solid, adjustable, aluminum panels. Electrical continuity is provided mechanically along all edges inside the reflector periphery. During erection of the antenna, the reflector was measured, using optical techniques, at 0, 45, 90 (zenith) and 180 degrees in elevation. A computer program was written expressly for the measurement and analysis of data; a best-fit parabola of 0.37-inch RMS was attained.

The hyperbolic reflector is 8.5 feet in diameter, has an RMS surface accuracy of 0.012 in., and is supported by a quadrupod.

This hyperbolic reflector can be positioned axially by remote control of a motorized drive mechanism, and laterally, vertically and angularly by manual means. The surface of the reflector was measured mechanically; those parameters necessary to define its surface were varied within certain limits; and another computer program was employed to determine the best-fit hyperbolic surface.

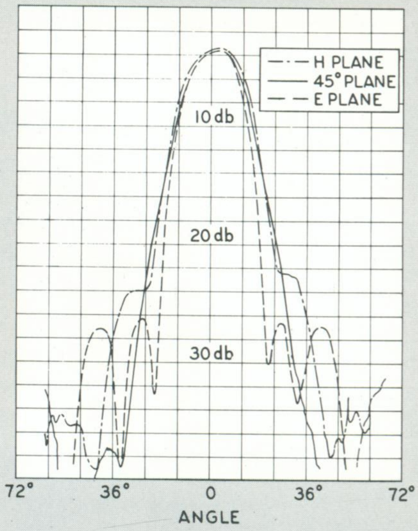

Fig. 2—Measured source pattern at 4.1 GHz.

The quadrupod was designed to reduce blockage under transmitting and receiving conditions and limit relative translational and rotational movement of the hyperboloid relative to the paraboloid. The feed cone was designed to house the feed system, two parametric amplifiers, tracking downconverters and front ends, stable local oscillators, noise measuring equipment and other test instruments.

Electrical energy was made available to the rotating portions of the antenna pedestal assembly from the stationary base by means of a maypole-type cable windup device. Cables were designed and fabricated to withstand torsion as well as bending stresses. From the azimuth to the elevation rotating structure, electrical cables were supported on cylindrical sections mounted at the elevation axis to accommodate an elevation rotation of 180 degrees.

The parametric amplifier mounted in the antenna assembly required cooling to 4.2 degrees Kelvin. This was accomplished through the use of a state-of-the-art closed-cycle cryogenic (helium) system. To provide sufficient on-the-air time for the earth station, two closed-cycle systems were installed in parallel and provided with automatic switchover from one to the other in case of malfunction. These were installed on the azimuth rotating platform.

A data take-off device is attached to each axis of the antenna pedestal through a bellows-type flexible coupling. The mechanical characteristics of the antenna pedestal are summarized in Table III. The antenna pedestal was designed for a six-weeks duty cycle as follows:

**TABLE III—Antenna Pedestal Characteristics**

| *Characteristics* | *Azimuth* | *Elevation* |
|---|---|---|
| Number of DC Servo motors | 2 | 1 |
| Rotational travel (degrees) | +300 and −300 | −2 to +182 |
| Velocity (max) $\Theta$/sec. | 0 to 3 | 0 to 3 |
| Acceleration (max) $\Theta$/sec.$^2$ | 0 to 3 | 0 to 3 |
| Locked rotor frequency (Hz) | 2 | 2 |
| Max. breakaway friction (ft-lbf) | 20,000 | 14,000 |
| Max. running friction (ft-lbf) | 16,000 | 11,200 |
| Weight of pedestal (lb) | 590,000 | |
| Weight on azimuth bearings (lbs) | 500,000 | |
| Elevation load inertia referred to elevation axis and excluding gear train (slug-ft$^2$) | 2,000,000 | |
| Azimuth load inertia, referred to azimuth axis and excluding gear trains (slug-ft$^2$) | 2,000,000 | |

**5 weeks:**

The entire system to be 'on the air' 24 Hours/Day (regardless of standby time).

| | |
|---|---|
| Hours/Day | 24 |
| Days/Week | 7 |
| Targets/Day | 24 |

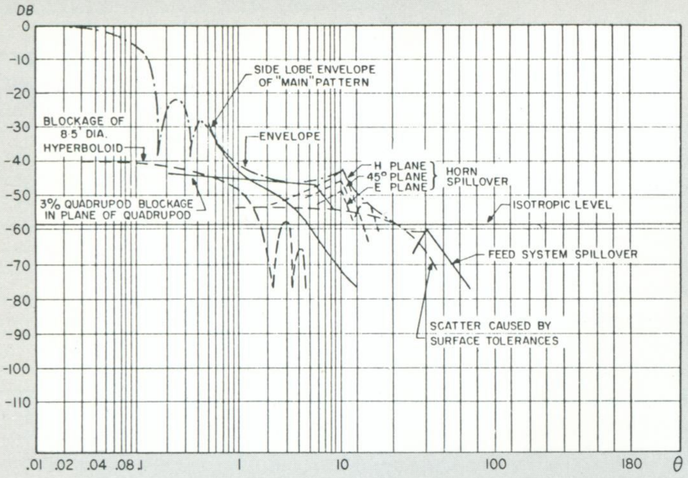


Fig. 3a—Antenna system pattern calculated from the source pattern at 4.1 GHz.

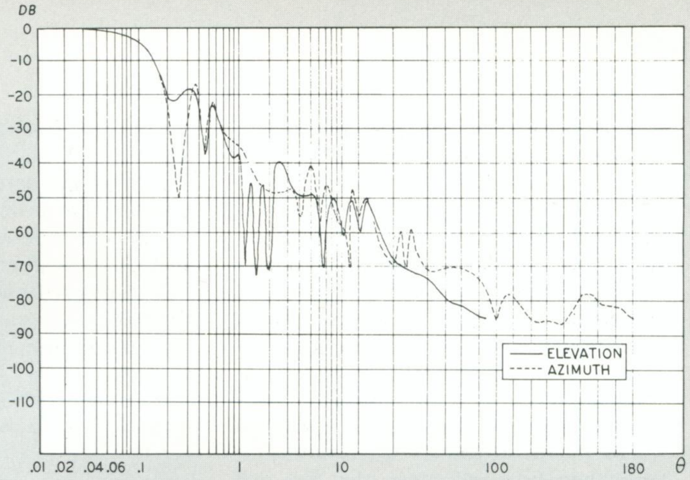


Fig. 3b—Directly measured antenna system pattern at 4.1 GHz.

| | |
|---|---|
| Minutes/Target | 40 |
| Programmed steering (acquisition) | 1 min. |
| Automatic tracking (after acquisition) | 35 min. (minimum) |
| Return slew (180°) | 4 min. |
| Standby (after each target) | 20 min. |

**1 week:**

Standby of the entire system with complete shutdown permissible not to exceed 1 week/yr plus 4 hrs/mo for planned maintenance.

The reliability required was 97% minimum; the downtime for the antenna pedestal was not to exceed 3% or 262 hours of any given one year period during the ten year life span. The downtime was to include all planned maintenance time.

## RADOME DESIGN

### Electrical Considerations

If cost considerations are discarded, heated radome-protected systems have better performance characteristics than exposed systems, with the exception of heavy rain conditions. The most obvious advantage of the radome-protected system is the elimination of wind fading.

On the other hand, radome-protected systems require special attention for the following cases:

1) Attenuation and reflection caused by the radome under ideal weather conditions,
2) Pointing and tracking inaccuracies associated with radome inhomogeneity,
3) Attenuation and reflection caused by precipitation,
4) Temperature stratification around the antenna, caused by the presence of the closed environment, and
5) Maintenance required for the skin of the radome.

Radomes can be divided into self-supported and air-supported categories. Although the self-supported types (space frame, rigid-dielectric) have better reliability and less maintenance requirements, they also have larger attenuation and scattering and higher investment cost. To reduce dry-weather noise temperature contribution, and inflated type of radome was selected. For the given environment and antenna size, a 0.056-inch wall thickness, and a dielectric constant of maximum 3.5, resulted in minimum reflection coefficient and dry-weather noise temperature, with a flexible-membrane dacron fabric. The water-film formation for heavy-rain conditions was minimized by a hypalon coating, which, at least for a number of years, has reasonable, although not ideal, non-watering characteristics.

The pointing and tracking errors were minimized by uniformity of the radome fabric and minimizing the overlap at joints. No effect on this account was measurable on the completed system.

### Mechanical Considerations

The air-supported hypalon-coated dacron radome is approximately spherical in shape and has a diameter at its equator of 120 feet. The height from the 90-foot base diameter to the pinnacle is 102 feet. The radome is attached and sealed by means of a steel base ring atop a circular concrete structure, which is actually part of the antenna pedestal building.

Air is ducted to the radome from a weather protected circular opening approximately 600 feet from the center of the radome. This air is supplied to the radome at pressure levels of 1.5, 3.0 and 5.5 inches of water, depending upon external wind conditions. Two anemometers mounted atop a 30-foot mast control the operating pressure level automatically with a time delay in the control system to minimize cycling under gust conditions. The radome was designed to withstand steady winds of 65 mi/h and gusts of 105 mi/h and to deflect less than one foot in any direction.

The three-foot diameter pinnacle of the radome incorporates aircraft warning lights, a lightning arrestor, a pneumatically operated air exhaust, an exit hatch to the outside of the radome, a 4,000 pound capacity sky-hook, and a folding ladder. Heavy-duty copper cables are mounted on six-inch standoffs from the lightning arrestor downward along the radome wall to a grounding installation. Thermocouples are attached to several points inside the radome to detect temperatures at various elevations.

The circular building on which the radome is mounted incorporates one 14-foot high by 14-foot wide by 55-foot long vehicle airlock and two personnel airlocks.

Multiple pressurization equipment, including self contained power units independent of the station power, are provided to assure fail-safe operation.

## CONCLUSIONS

Canada's Mill Village earth station was probably the last of the large stations built for the purpose of experimental microwave communication. In fact, the station is presently used for mostly commercial satellite communication on a time-shared basis with ComSat's Andover facility.

The system met all the original design objectives, and its communication performance in the bottom slot is characterized by an NPR (noise-power ratio) of about 35dB, with a conventional receiver for 240 voice channels, while working with the Early Bird satellite. This performance approaches the NPR characteristics of other stations with similar antenna aperture, but equipped with an FM feedback receiver.

The $G/T_s$ ratio of the antenna exceeded by approximately 1dB the original design objective. Although the station has been in operation for only about 1½ years, it is already obvious that the antenna-radome complex is its most reliable subsystem, and probably its most important single feature; no downtime ever was reported on the account of the antenna system during this period of operation.

# COST CONSIDERATIONS IN DESIGNING EARTH STATIONS

The cost of early ground stations varied considerably—partly because of their experimental nature and partly because substantial developmental efforts were included in their construction. The construction of ground stations is now entering a more mature phase: requirements are more precisely defined and standardized; characteristics of the available equipment and techniques are better known; and future technological developments, both in the spacecraft and in the ground terminals, are more closely anticipated as a consequence of the accumulated studies. Also, data for statistical variables caused by satellite behavior, weather, and site characteristics is beginning to accumulate. All these developments now make it possible to design a ground station more on a technical-business basis than during the era of experimental ground stations.

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Although a satellite communication ground station is quite complex, the main performance requirements can be summarized briefly (Table I).

**TABLE I—Main Requirements for Satellite Communication Ground Stations**

| | |
|---|---|
| $G/T_s$: | 40.7dB (400 MHz, $\theta = 5°$ elevation angle, clear weather) |
| ERP: | 95dBW (HS-303A); 89dBW (Global System) |
| Receive Frequency: | 3700 to 4200MHz |
| Transmit Frequency: | 5925 to 6425 MHz |
| No. of RF carriers: | 1-20 |
| No. of channels: | 1-600 voice or 1 TV per carrier |
| Quality of channels: | CCIR standards |
| Time of full operation capability: | 99.7% |
| Down time: | 0.1% |
| Tracking range: | Full hemisphere |
| Polarization: | circular or orientable linear |

$G$ = gain of the antenna
$T_s$ = noise temperature of the system

Within the limits specified in Table I, the designer still has considerable freedom in determining his station. Will he provide the full frequency band initially, when only part of this band is required for early operation? Will he meet the $G/T_s$ requirement at 5° or at 1° or 2° higher? Should he select a large transmitter on the ground, or a smaller one rotated with the antenna? Should the down-time budget be utilized to take care of weather effects only, or be utilized for less reliable equipment? Should the tracking be based on either program steering, or autotrack, or both?

The selection of the *features* of the station will affect not only the cost effectiveness but also some other characteristics sought by the customers, such as flexibility, easy operation, maintenance, and low rate of obsolescence. Table II summarizes the features selected in the present study. The indicated choices are only the most important ones, and the selection is indicated by italics.

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## COST SENSITIVE SYSTEM PARAMETERS

The total cost of a ground station for a *given* set of requirements is determined by a very large number of system parameters. Fortunately, only a relatively small number of these parameters have substantial effect on cost and (or) are related to each other. When a subsystem parameter is independent of the rest of the system parameters, it can be optimized by itself, and will not enter into the overall studies.

**TABLE II—Main Features of the Ground Station**

| | | |
|---|---|---|
| Compliance with requirements: | *Full* | Partial |
| Redundancy: | Full | *Gracefully failing* |
| Investment cost: | Rigid | *Flexible* |
| Operation method: | *Centralized* | Decentralized |
| Operation cost: | Low | Medium |
| System concept: | Fully developed | *Partially developed with growth capacity* |
| Subsystem concept: | *State of art* | *Well proven* |
| Component concept: | *Modular, solid state* | |
| Delivery: | 12 months | 14 months |

The sensitive system parameters are:

1) Use of radome;
2) Diameter ($D$) of paraboloid reflector.
3) RMS surface accuracy ($\Delta$) of paraboloid for ideal weather;
4) Lock-rotor frequency $f_L$ of antenna structure;
5) Weight $W_s$ of subreflector support (a Cassegrain antenna is assumed);
6) Diameter ($d$) of hyperboloid reflector;
7) Aperature size ($a$) of feed horn;
8) Average antenna efficiency, $\eta_{AV} = 0.5 (\eta_{Rx} + \eta_{Tx})$;
9) receive-to-transmit antenna efficiency ratio ($\tau = \eta_{Rx}/\eta_{Tx}$);
10) Tracking accuracy ($\delta$) of antenna;
11) Noise temperature ($T_{Rx}1$) of main parametric amplifier;
12) Down time ($R_{Rx}1$) of main parametric amplifier;
13) Noise temperature ($T_{Rx}1$) of back-up parametric amplifier;
14) Down time ($R_{Rx}1$) of back-up parametric amplifier;
15) Output power ($P_{Tx}1$) of main transmitter;

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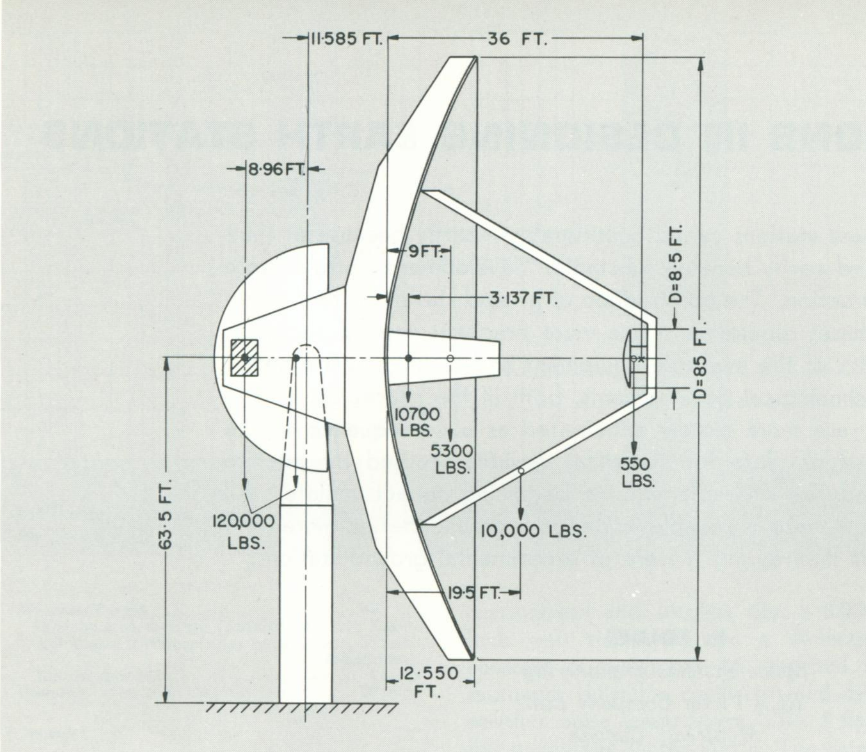



Fig. 1—Mechanical layout of a typical 85-ft diameter antenna.

16) Down time ($R_{Tx}1$) of main transmitter;
17) Output power ($P_{Tx}2$) of back-up transmitter;
18) Down time ($R_{Tx}2$) of back-up transmitter;
19) Down time ($R_{As}$) of antenna for syncronous satellite;
20) Down time ($R_{Am}$) of antenna for medium orbit satellite;
21) Threshold extension ($L$);
22) Distance between antennas ($K_A$) when more than one antennas is used; and
23) Distance between first antenna and control building ($K_C$).

These parameters depend on at least one parameter in the set; the value for each must be determined by varying all the other parameters which have influence, and also by varying those requirements which are not constant (Table I). A computer program can then be used to determine the minimum of the total cost function.

#### UPLINK AND DOWNLINK FADING WITH AND WITHOUT RADOME

Two possible cases are of interest:

1) The customer prefers to use a radome because of its operational advantages. In this case, there is no choice. Nevertheless, the fading characteristics must be known, to obtain the most economical system.
2) The customer wants a system with the highest possible cost effectiveness. In this case, the choice between radome and no radome is open; for most sites, the systems without radomes are more economical.

Generally, weather causes a $G/T_s$ fading in the receive frequency band and an ERP (effective radiated power) fading in the transmit frequency band. The $G/T_s$ fading is usually more important, since it is more costly to compensate for. This fading may be caused by rain, wind, snow, ice, and solar radiation (differential temperature).

The greatest fading shows up for the smallest antenna size, lowest receiver noise temperature, and highest elevation angle; also fading is greater in the radome-protected system. For a given $G/T_s$ ratio, the fading decreases with increasing antenna size (a little less rapidly for systems with radome than for systems without, but the absolute value of signal fade is less for the system with no radome).

In the fading calculations of the system without radome, it was assumed that the total time per year when the ice on the antenna is more than 1.5 inches thick, or the snow is more than 5 inches deep, is restricted to 8 hours. For stations with two antennas, this would not present a serious problem, particularly if the other (standby) antenna is under a radome. For stations with one antenna, the care of medium orbit satellites does not present a problem, since system interruptions for satellite changeover already exceed 8 hours. For stations with a single antenna and a syncronous satellite (a more practical case) virtually no interruption is tolerated.

Assuming that the paraboloid has $F/D$ of 0.42, the lower edge of the reflector is approximately horizontal for elevation angles of 30° and is the most sensitive for snow accumulation. However, the total area of the reflector where the tangent to the surface encloses less than 45° with the horizontal, is less than 831 ft², when the aperture area is 6318 ft². For most practical cases, snow accumulation for angles above 45° is negligible. Then the snow or ice removal from the 831 ft². surface, by electrical heating, is quite inexpensive. For instance, when the snowfall rate is 2 in/h, and the density factor is 0.1, then the total melting power is 9.9 cal/sec (41W/ft²).

The efficiency with which this melting power can be provided depends on thermal radiation losses, distribution of heating elements, wind velocity, etc. For ideal heat distribution and no wind, the efficiency is about 80%, which drops to 66% for 20 mi/h wind. Assuming 41% efficiency, the heating power requirement is 100 W/ft.² Such a heater can be provided for a cost of approximately 10 to 20$/ft². This is a fraction of the cost of a radome installation.

Another weather hazard is icing at temperatures between 26 and 32°F and high relative humidity. For most sites, the icing rate very rarely exceeds 0.1 in/h for prolonged periods. Therefore, 50 W/ft heating power is adequate for this purpose. If the heating facilities for de-icing are restricted for the inner part of the reflector, defined by level lines within which half of the total RF power is

**TABLE III—Mechanical Data of a Typical 85-ft Diameter Antenna**

| Component | Weight ($10^3$ lbs.) | Distance from pivot (ft) | Moment ($10^3$ ft. lbf) | Reflector area (ft²) | Weight/Area (lbs/ft²) |
|---|---|---|---|---|---|
| Feed Cone | 5.3 | 20.58 | 109.074 | | |
| Quadrupod | 10.0 | 31.08 | 310.080 | | |
| Hyp. | 0.55 | 46.58 | 25.619 | | |
| Basic panels | 10.7 | 15.76 | 168.632 | 6140 | 1.74 |
| Extra panels | — | — | — | — | — |
| Basic trusses | 54.3 | 13.0 | 705.9 | 6140 | 8.84 |
| Extra trusses | — | — | — | — | — |
| Subtotal | | | 1319.305 | | 10.58 |
| Shell | 120.000 | —1.97 | —60.695 | | |
| | | —10.48 | —1258.610 | | |
| Counterweight | 120 000 | | | | |
| El. structure | 320,850 | | | | |
| Az. structure | 212.150 | | | | |
| Ant. structure | 533.000 | | | | |
| Cost ($) | 740.000 | | | | |
| $/lb | 1.39 | | | | |

NOTES:
$D = 85$ ft.
$\Delta = 0.037$ in. RMS
$f_L = 2.25$Hz
Hyp. Support: Steel Quadrupod

**TABLE IV—Calculated Lock Rotor Frequencies for Various Practical Cases**

| | | | | | | $f_L$ | | | | | | | | |
|---|---|---|---|---|---|---|---|---|---|---|---|---|---|---|
| | | | | | | Case | 1 | | | 2 | 3 | 4 | | |
| | | | | | | kHz | | 2.47 | | 2.47 | 2.47 | 2.25 | 2.14 | 1.91 |
| | | | | | | m | | 1.6 | | 1.0 | 0.1 | 0.1 | 0.1 | 0.1 |
| D(ft) | d | lgd. | $d^{1.6}$ | $d^{0.7}$ | $d^{0.1}$ | Δ (in.) | 0.037 | 0.040 | 0.050 | Variable | | 0.037 | 0.040 | 0.050 |
| 85 | 1 | 0 | | | | | 2.47 | 2.35 | 2.10 | 2.47 | 2.47 | 2.25 | 2.14 | 1.91 |
| 90 | 1.0588 | .02472 | 1.0953 | 1.0407 | 1.0055 | | 2.25 | 2.14 | 1.91 | 2.37 | 2.45 | 2.23 | 2.12 | 1.89 |
| 95 | 1.1176 | .04828 | 1.1945 | 1.0810 | 1.0110 | | 2.06 | 1.97 | 1.75 | 2.28 | 2.44 | 2.22 | 2.11 | 1.88 |
| 100 | 1.1764 | .07056 | 1.2967 | 1.1205 | 1.0163 | | 1.90 | 1.81 | 1.62 | 2.20 | 2.43 | 2.21 | 2.10 | 1.87 |

concentrated, then an additional 26% of the paraboloid reflector (in the upper half) has to be equipped with (half-density) heaters.

The operational cost of such a heater facility is only a fraction of the operational cost of the radome. This example is typical of problems related to weather protection of large antennas without radomes.

### ANTENNA SYSTEM COST CONSIDERATIONS

Before any cost analysis of the system can be made, the cost of the antenna system must be determined as a function of its diameter, surface accuracy, lock-rotor frequency, tracking accuracy, and design of the subreflector support. The situation can be analysed on a typical existing 85-ft. diameter design; Fig. 1 shows the basic layout of the antenna. The breakdown of the weight and overturning moments are indicated in Table III.

One of the factors influencing the cost of the antenna is the type of support for the subreflector. Fig. 2 indicates the relative cost variation of the above antenna with a monopod support in an open environment for no-wind conditions. Three different design conditions were assumed for the accuracy of the reflector:

*Case 1:* Practical design, in which the surface accuracy, Δ, varies according to the curve given in Fig. 2, as the diameter varies. This surface accuracy variation is obtained without a reinforcement of the back-up structure. The "practicality" in this design means that only the counterweight, hyperboloid support, and surface panels beyond the 85-ft. diameter, are changed relative to Table III. The limit for this design is about 95 ft.

*Case 2:* "Ideal" design, with varying Δ. This case is similar to case 1, but the structure is redesigned to compensate for varying weight conditions. Below 95.5 ft., the structure is lighter; above that limit it is heavier than the "practical" design.

*Case 3:* Ideal design with constant Δ. The selected values are: 37, 40 and 50 $\tau$).

In each case calm weather was assumed. Fig. 3 shows the relative variation of cost of an inflated radome versus antenna diameter, if this protection is added to the system.

From Table IV, it can be seen that the monopod results in a superior lock-rotor frequency. Generally, the lock-rotor frequency decreases with the diameter, because of the increase of the structural weight, but with a light hyperboloid support the lock-rotor frequency can be kept above 1.8 Hz. This frequency is adequate for tracking medium altitude and syncronous satellites.

### MINIMUM COST FOR A GIVEN $G/T_s$ RATIO

The two most important components in the cost, to obtain a given $G/T_s$ ratio, are the antenna structure and the low-noise preamplifier.

Assuming 500-MHz receiver bandwidth, the parametric amplifiers are the most practical preamplifiers to achieve low noise temperatures. Generally, the cost of a parametric amplifier increases rapidly by decreasing noise temperatures; the necessary cooling machine becomes more complex, less reliable, and more expensive to maintain. There are at least four different types of amplifiers for the various temperature ranges. Their characteristics and price range are exhibited in Fig. 4.

The indicated cost figures refer to the currently required small quantities. A cost improvement of an order of two is expected in the coming five years as larger quantities of standardized types of these amplifiers will be required. The basic character of the curve on Fig. 4, however, is not expected to change during the next 5 year period.

Using Figs. 2, 3, and 4, the cost variation of the antenna structure, plus parametric amplifiers, versus antenna diameter can be calculated (Fig. 5). The same figure indicates the variation of $G/T_s$ and lock rotor frequency.

Fig. 6 indicates the cost of the structure plus parametric amplifiers for constant $G/T_s$ ratios. It can be seen that for each $G/T_s$ ratio, there is an antenna diameter and receiver noise temperature, which results in minimum cost. With increasing $G/T_s$ requirement, this minimum shifts toward larger antennas and smaller noise temperatures. For the range of 40.7dB $< G/T_s <$ 42.7dB, the optimum antenna diameter is 91.5 ft. $< D <$ 99.5 ft., and the receiver temperature is $45°K > T_{Rx1} > 22°K$. The "optimum" receiver noise temperature falls within the range of helium gas type cooling machines.

On the basis of previous results, it is possible to predict the cost of a given increase of the $G/T_s$ ratio under optimum conditions. The approximate differential increases in cost needed to obtain 1dB improvement in $G/T_s$ are indicated in Table V.

**TABLE V—Cost of 1-dB $G/T_s$ Increase for Optimum Conditions (Rohr dish, aluminum quadrupod)**

| *Increase from* | *to* | *Cost ($10^3$\$)* |
|---|---|---|
| 40dB | 41dB | 38 |
| 41dB | 42dB | 74 |
| 42dB | 43dB | 139 |
| | Ave. | 84 |

### DETERMINATION OF OPTIMUM FEED SYSTEM

Excluding the possibility of focal-point feeds as operationally inconvenient, and horn reflectors as costly, the only remaining type of feeds is the Cassegrainian. In this category, three main types of feeds are developed, with increasingly better characteristics. These are:

1) One horn, single-mode feeds;
2) One horn multimode feeds; and
3) Multihorn multimode feeds.

The most important single characteristic related to the feed is the overall average antenna efficiency $\eta_{AV} = 0.5\ (\eta_R + \eta_{Tx})$, where $\eta_R$ and $\eta_{Tx}$ are the average antenna efficiencies for the receive and transmit band respectively. Fig. 6 shows the cost of these feeds versus their antenna efficiency. The generalized curve between these discrete points is somewhat academic, but it predicts the general tendency for the cost of antenna efficiency. Using the previous results, the differential cost variations versus $\eta_{AV}$ can be calculated. Fig. 7 shows the change of cost for the 100-ft dish, aluminum quadrupod, and $T_{Rx1} = T_{Rx2} = 22°K$. The curve predicts, for this case, an optimum value of —2.80dB for $\eta_{AV}$. The closest existing feed system is the 1-horn multimode feed, with 2.64dB.

### COST OF THE EFFECTIVE RADIATED POWER

For the requirements given in Table I, the cost of the effective radiated power is determined by the antenna gain (diameter of antenna), the reliability of the transmitter system, and the location of power amplifiers relative to the feed.

Fig. 8 shows the variation of cost for antenna diameter, $D = 105$ ft. and for ground located transmitters, assuming 2.5-dB transmission line loss between the transmitter tube output flange and the antenna feed input flange. The cost of transmitted power was calculated for the condition that output power of the tube is sufficiently below saturation, to keep third order products below —20dB

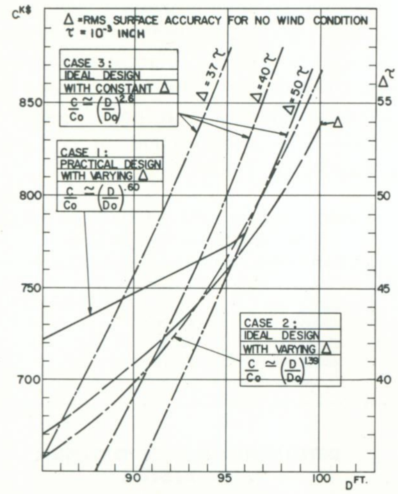



Fig. 2—Cost and RMS surface accuracy of antenna structure versus diameter (with monopod support).

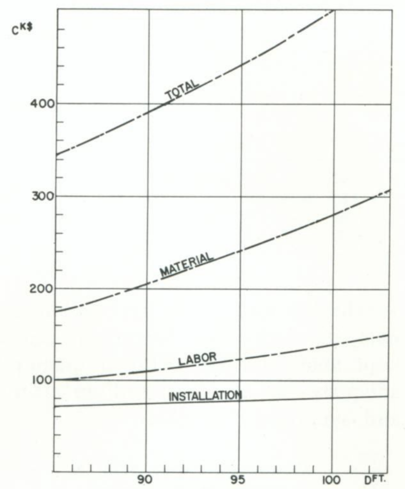



Fig. 3—Cost of inflated type radome versus antenna diameter.

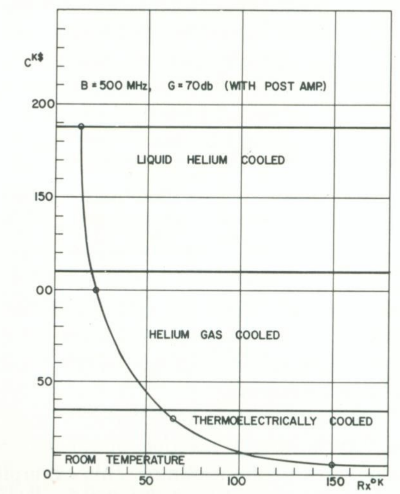



Fig. 4—Cost of parametric amplifier versus receiver noise temp. (instantaneous bandwidth: 3.7 to 4.2 GHz, post amp. included).

for 2-carrier operation. Two types of transmitters were assumed:

1) "Modular" type of transmitter, in which 1.88kW, 3.53kW, and 6.63kW useful power can be obtained by the paralleling of two, four, and eight 1kW tube modules, respectively, with a combination efficiency of 94% (such a 1kW tube is not available at the moment).
2) Single tube approach for the 1kW, 2.8kW (Siemens), 5kW (Hughes) case

Fig. 8 shows that the cost of transmitter power increases more than linearly with the level of the required ERP and the variation increases by decreasing the antenna diameter.

## OPTIMUM ANTENNA EFFICIENCY RATIO FOR THE RECEIVER AND TRANSMITTER BAND

From the cost variation of $G/T_s$ and cost of ERP (Fig. 8) the optimum value of the $\tau = \eta_{Rx}/\eta_{Tx}$ ratio can be calculated.

Fig. 9 indicates the differential cost variation as a function of $\tau$ for ERP = 95dBw, $D = 100$ ft., and $\eta_{AV} = -2.65$dB. (1-horn multimode feed).

## COST OF THRESHOLD EXTENSION

To provide adequate fading margin in the receiving system, at an economical cost, the threshold level of the demodulator must be extended relative to the level (10dB) achievable by a standard FM demodudator. Various methods have been recommended for this purpose (FM feedback, phase lock, IF bandwidth reduction), which, however, have one common characteristic: the cost of such a demodulator increases with the number of required RF carriers, with voice channel capacity, and with the required threshold improvement ($\Delta$ $L$).

Fig. 10 shows the variation of cost for 1-dB threshold improvement versus the number of voice channels ($n$) for the number of RF carriers of $N = 2$, 6, 10 and 20. The cost estimate is based on the development and production costs of the equipment. Fig. 10 indicates also the theoretically expected maximum possible threshold extension, $\triangle$ $L$. The curves are prepared for the ideal case, when the complexity (and cost) of the threshold-extending device is adjusted according to the required amount of extension and channel capacity. Again this is somewhat hypothetical, but as more different modules will be developed for the various requirements, the practice will be closer to this "optimum" provision of extension. It can be seen that for large numbers of carriers, and for large channel capacities, the cost of threshold extension is quite large. For instance, for an $N = 6$, $n = 600$ system, with a total improvement of $\triangle$ $L = 3$dB, the cost of the extension system is \$259,000, or \$86,500/dB. This is about the same as the cost of 1dB through $G/T_s$ improvement.

Fig. 5—Cost, $G/T_s$ and $f_L$ versus antenna diameter ($\theta = 5°$, $T_A = 36°$K)

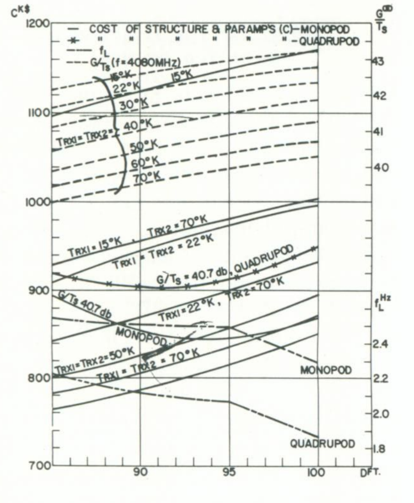



Fig. 6—Cost of feed system versus average antenna efficiency.

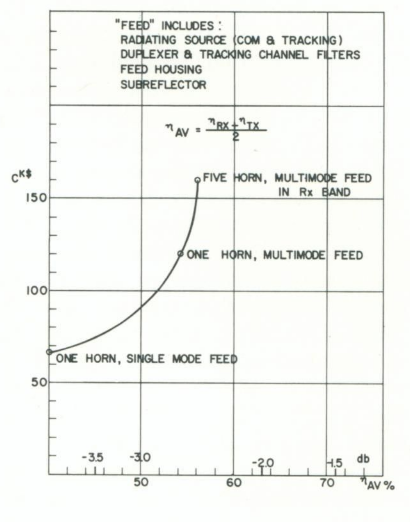
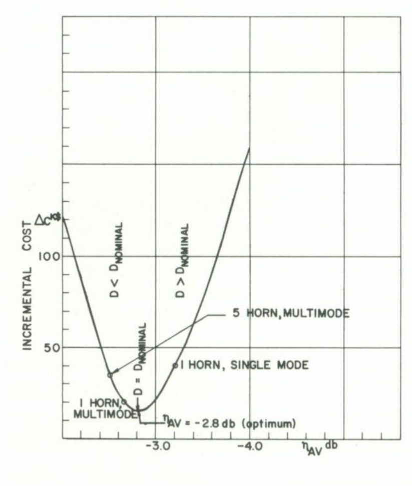



Fig. 7—Incremental cost variation of antenna structure and feed versus $\eta_{AV}$ for $T_{RX1} = T_{RX2} = 22°$K and $D$ (nominal) = 100 ft.



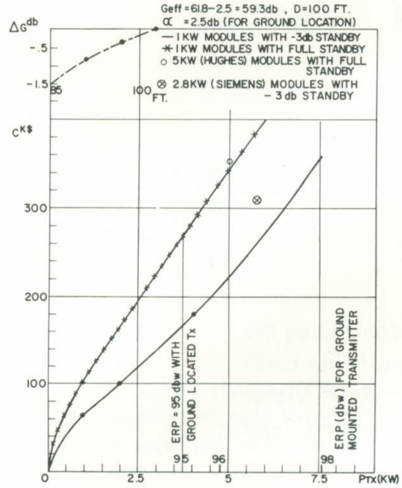



Fig. 8—Cost of transmitter power.

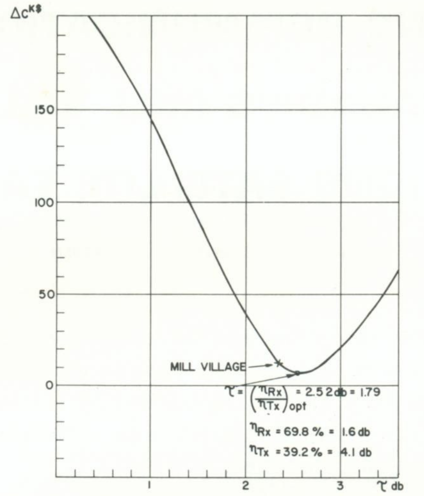



Fig. 9—Differential cost variation of antenna structure paramps and transmitter versus $\tau = \eta_{Rx}/\eta_{Tx}$ (ERP = 95 dBw, D = 100 ft, $\eta_{AV}$ = 54.596)

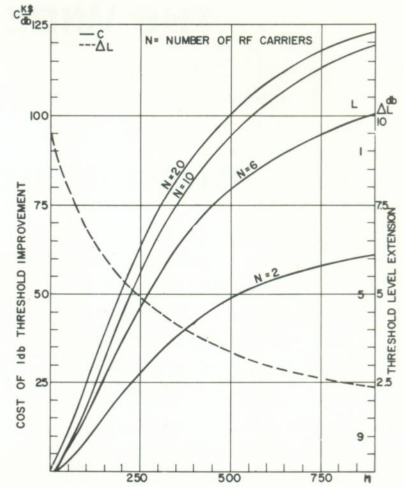



Fig. 10—Cost of FM threshold improvement and maximum possible improvement versus number of voice channel, $\eta$.

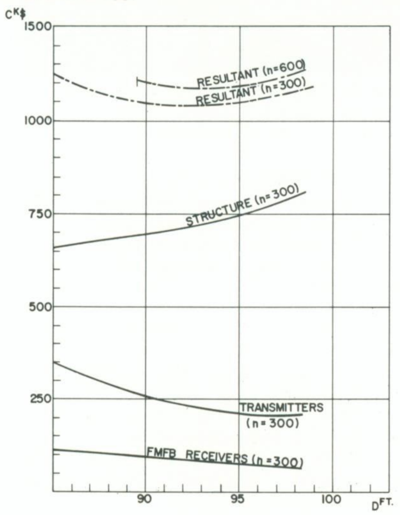
To illustrate the sensitivity of the system cost to the cost of the antenna strucure, cost of power amplifier, and cost of FMFB receiver, the variation of the cost of these equipments versus antenna diameter is plotted in Fig. 11. The conditions for this figure are indicated in Table VI.

**TABLE VI—Typical Conditions for a Practical RF System**

| |
|---|
| $T_{Rx_1} = 22°K$ |
| $T_{Rx_2} = 60°K$ |
| $P_{Tx_1} = 95$dBm (in more than 90% of time) |
| $P_{Tx_2} = 92$dBm (in more than 10% of time) |
| $N = 2$ |
| n = 300 and 600 |
| Hyperboloid support: Monopod |
| No radome |
| $\theta = 5°$ |

It can be seen from Fig. 11 that by ideal (proportional) design of the three subsystems which influence the selection of antenna diameter, the total cost can be minimized around $D = 93.5$ ft.

Fig. 11—Cost of antenna structure, transmitters, and FMFB versus antenna diameter for $n$ = 300 and 600.




By the use of the curves given previously, a series of cost tables can now be prepared for various combinations of the parameters in Table VI. These combinations have to be restricted to an acceptable standard of reliability (down time) consistent with the 99.7% overall operational reliability specified in Table I.

## CALCULATION OF COST FOR OPTIMUM CONFIGURATIONS

For the small user ($V = 48$ channels), the cost per channel is very large (minimum $60,541). Nevertheless, this cost may still be attractive, if there is no other way for reliable communication. For the medium user ($V = 240$ channels), the installation cost per channel is very economical ($11,681, minimum, with feed cone mounted transmitter, without reflector heater, and with two simultaneous carriers). Finally, for a very large user, $V = 1200$, the cheapest system costs $2606/channel. A more refined system (with program steering included), costs $2741/channel, and a deluxe version (with radome), $3100/channel.

## CONCLUSIONS

On the basis of the previous analysis, the following conclusions can be made:

1) With the exception of very severe climatic conditions, the use of radome is not economical for commercial satellite communications. For identical performance, a system with radome cost about 13.5% more than a system with antenna heaters.
2) The limit of the present antenna designs can be expanded to 100-ft. diameter.
3) A pre-stressed steel hyperboloid support (monopod) offers a 0.2dB improvement in the $G/T_s$ ratio, and an associated saving in the structure. Hyperboloid positioning mechanism virtually disappears in this structure.
4) For channel capacities/carrier of 300, or more, the optimum antenna diameter is above 90 ft. To provide future growth and room for lower cost receiver front ends, an antenna diameter of 95 ft. should be selected.
5) The optimum type of feed system is the single horn, multimode design. The system economy does not tolerate either a simpler or a more complicated feed concept.
6) The optimum ratio for the receive-to-transmit frequency-band antenna efficiency is 2.52dB.
7) A modular type of transmitter is required to make an optimally adaptable system to the various requirements. For this purpose, the development of a TWT with 1400-W saturated output is necessary with an operational level of about 1000 to 1100W.
8) It is more economical to locate the transmitter in the feed cone than on the ground, but very high reliability must be achieved before customers will accept such a location.
9) The back-up transmitter can be —3dB, relative to the main transmitter, without affecting overall loop performance, if it is operated only in a maximum 10% of the time.
10) The main receiver front end has to be a helium-gas-cooled paramp for stations with 40.7-dB $G/T_s$ ratios.
11) The cost of FMFB receivers represents a very substantial part of the station for a large number of channels. The development of FMFB receivers, in which cost can be made proportional to the number of voice channels and required threshold improvement, could result in a substantial saving in the cost of the total station.
12) There is about one-million-dollar difference between the price of the cheapest and most expensive stations, among the analyzed cases.
13) The cost per channel varies between very wide limits (from about $60,000 for a 48 channel equipment, to about $3,000 for a 1200 channel equipment).

# THE WIDEBAND COMMUNICATIONS SYSTEM OF CANADA'S MILL VILLAGE COMMUNICATIONS SATELLITE EARTH STATION

This paper describes the transmitting and the receiving equipment comprising the Wideband Communications facilities of Canada's communications-satellite Earth Station at Mill Village built for the Department of Transport by RCA Victor Company, Ltd. Various systems aspects, noise, interface requirements, and equipment locations are also discussed.

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CANADA and the United States agreed in 1963 to participate in a space communications experiment using a proposed NASA satellite to be known as the Advanced Technological Satellite (ATS). General characteristics of this experimental space link for telephony signals would be SSB transmission in the up-path (ground to satellite), and PM transmission in the down-path (satellite to ground), while for television signals they would be FM for both up-path and down-path.

Canada's contribution to the experiment was the development, construction, and operation of a communications-satellite earth station in the eastern part of her territory, forming one terminal of the experimental system. Canada's Department of Transport was authorized to proceed with the station.

**EARLY PLANNING BY RCA VICTOR, LTD.**

RCA Victor Company, Ltd., in Montreal, Quebec, was engaged in a consulting capacity for the preparation of system parameters and specifications, and assistance in site selection. A contract was awarded for the project management, system engineering, integration, and contracting responsibilities for the antenna, radome, servo and tracking system, transmitters, receivers, and low-noise amplifier. The Department of Transport acted as the design authority and retained responsibility for building, services, multiplex, and boresight.

The location chosen for Canada's earth station was near Mill Village, Nova Scotia, some 9 miles from the Atlantic ocean. This site is relatively free from interference from radio relay systems and aircraft, and is reasonably close to a major communications terminal.

While the Mill Village station was under construction, the launch date for ATS was delayed due to the increasing diversity of the experiments.

Meanwhile, the Communications Satellite Corporation, (ComSat) obtained permission to own and operate commercial satellites, placed the "Early Bird" satellite into orbit, and leased Andover (Maine) earth station for the American terminal of its operation. Attention focused on the possible use of Mill Village as a second North American commercial terminal sharing the duties with Andover. Expediency and time would not permit the construction of a second earth station for this purpose, therefore the experimental station would have to be used until it could be relieved by

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**TABLE I—Initial Operating Frequency Capability for Mill Village**

| Satellite | Transmit Frequencies (MHz) | | Receive Frequencies (MHz) | | Beacon Frequencies (MHz) |
|---|---|---|---|---|---|
| | Telephony, TV Video | TV Audio | Telephony, TV Video | TV Audio | |
| ATS (NASA) | 1) 6212.094 | Subcarrier | 4119.599 | Subcarrier | 4135.946 |
| | 2) 6301.050 | Subcarrier | 4178.591 | Subcarrier | 4195.172 |
| Early Bird (ComSat) | 1) * 6389.97 | 6400.97 | 4081.00 | 4092.00 | 4104.14 |
| | 2) ** 6301.02 | 6290.02 | 4160.75 | 4149.75 | 4137.86 |

* Used in North America
** Used in Europe

another Canadian station. The capability for handling signals through Early Bird were therefore added to Mill Village requirements while the station was under construction. Thus, Mill Village had a dual requirement from the outset, and this situation is reflected in many of the system design details and equipment complements.

### BASIC STATION CONCEPT

Conceived with an experimental viewpoint, the communications capabilities of Mill Village were to be kept flexible. Primary services would be for Early Bird and ATS, as reflected in Table I. However, the earth station design, through the addition of relatively inexpensive and simple hardware, must be capable of operation at other frequencies. Requirements for the initial equipment complement were as follows:

1) Transmit and receive up to 1200 telephony channels with a performance meeting CCIR recommended standards;
2) Transmit and receive 525 line monochrome television signals with a performance meeting CCIR recommended standards;
3) Transmit both FM and SSB signals with powers up to 10 kilowatts average anywhere in the 5.925- to 6.425-GHz common-carrier band;
4) Receive two FM or PM carriers simultaneously, at power levels as low as —100 dBm anywhere in the 3.7- to 4.2-GHz common-carrier band;
5) Achieve a dry weather system noise temperature of 65°K at an antenna elevation angle of 7.5° above the horizon; and
6) Track any present or future satellite in an orbit appropriate for communications.

For convenience in the distribution of baseband signals throughout the station, the following baseband signal standards were selected:

*telephony:* —20 dBm test tone across 75 ohms, unbalanced;

*video:* 1 volt peak-to-peak, positive going, across 75 ohms unbalanced; and

*audio:* +9 dBm (maximum signal) across 600 ohms, balanced.

### NOISE CONSIDERATIONS

Noise is a basic limitation in all systems. In communications systems, noise is generally classified into two types:

*Thermal noise* which is generally produced by the physical temperature of amplifiers and lossy elements such as waveguides, cables, etc;

*Intermodulation noise* which is produced after demodulation by non-linear circuit elements and uncorrectable group delays in transmission elements.

Most thermal noise originates in the signal processing stages of the receiving system, while intermodulation noise may result from deviations in the equipment parameters almost anywhere; however, the latter is usually more prominent in the frequency limiting circuitry such as filters, amplifiers, and the transmitter power tube.

Since the performance specifications for communications links are given as noise aggregates, more thermal noise can be tolerated for economic reasons than intermodulation noise. It is very expensive to provide additional ERP capability in satellites. Thus, ¾ or more of the permissible 10,000 picowatts of psophometrically weighted noise allowed by the CCIR standards for telephony is usually allowed for the thermal-noise contribution; the balance is for intermodulation noise and is shared by both the satellite and earth-based equipment non-linearities.

Signal power radiated earthwards by present generation satellites is quite modest; at the earth station, the power received by the antenna in the order of −90 to −100 dBm. Since wide-index FM is the rule, power is spread out over a broad-signal spectrum. The useful recovery of these signals is a difficult task and requires special low-noise techniques and equipment.

To prevent thermal noise contributions from exceeding the allowance, system noise temperature must be kept below 65°K under dry weather conditions. Table II lists the contributors to the system noise temperature for the minimum antenna elevation angle of 7½° above the horizon. This very low system temperature was achieved by:

1) Reducing the antenna side-lobe pickup by careful design;
2) Reducing the feed system attenuation and mismatch losses to very low levels, and
3) Physically cooling the first RF amplifier in the receive path.

The first two of the above techniques are outside the scope of this discussion; the third is appropriate. The first RF amplifier, a parametric amplifier, was cooled to 4.2°K in a closed-cycle cryogenic refrigerator. A parametric amplifier, rather than a maser, was chosen because of its relative simplicity and its wide instantaneous bandwidth. When cold, the amplifier produced about 13°K of the measured 65°K noise temperature.

For intermodulation noise the design goal was set at 500 picowatts or less so that future requirements would not be compromised. Table III summarizes the designed upper level of performance of the station with respect to intermodulation noise contributions from the various sources of non-linearities.

**TABLE II—System Thermal Noise-Temperature Budget (Antenna Elevation Angles 7½° Above Horizon)**

| Contributor | Noise Temp. (°K) |
|---|---|
| Atmosphere (dry weather) | 17 |
| Dry Radome (absorption and scattering) | 9 |
| Antenna Sidelobes | 9 |
| Feed System and Waveguide | 17 |
| Parametric Amplifier | 13 |
| Other Amplifiers | 1 |
| *Total System Temperature* | 65 |

**TABLE III—Design Objective for the Intermodulation Noise Budget (300 channel)**

| Contributor | Noise (picowatts) |
|---|---|
| FM Exciter | 200 |
| Receiver (FM Path) | 330 |
| Power Amplifier | 80 |
| Parametric Amplifier | 50 |
| Feed (4 GHz path) | 25 |
| Feed (6 GHz path) | 25 |
| Baseband Link | 50 |
| *Total Noise* | 500 |

Equivalent Noise Power Ratio: 44.1 dB

### EQUIPMENT LOCATIONS

The station consists of a central-control building and an antenna structure protected by a radome and separated from the control building by about 1200 feet. Thus the antenna beam clears the top of the control building when the antenna elevation angle is as low as 3° above the horizon. More antennas may be added to the complex as long as the spacing between structures provides the necessary beam clearances; a total of six antenna structures could be accommodated and served by the one control building.

The antenna uses an 85-foot diameter reflector with a Cassegrainian feed system and is mounted on an AZ-EL type of pedestal mount. A feed cone located at the center of the reflector houses the low-noise amplifier and the receiver front ends. Below the reflector is a 19-foot diameter cage in which the transmitter RF assemblies are placed. The antenna system, antenna pedestal, antenna radome, reflectors, and feed systems are described in another paper.[1]

The high-voltage transmitter power supplies are located on the radome floor at the base of the antenna; heat exchangers are located outside the radome structure. Adjacent to the radome, an equipment annex houses the main portions of the receiver, exciters, transmitter control, and parametric amplifier controls. Signal interface between the annex and the antenna-mounted equipment is at the IF frequency and passes through a "maypole" cable wrap through the azimuth axis and a cable "loop" past the elevation axis.

Baseband, multiplex, and test equipments are located at the control building. Therefore, baseband transmission links connect the control and radome buildings; telephony, television audio, and video signals are passed through the buildings in both transmit and receive directions.

### COMMUNICATIONS SYSTEM

A block diagram of the communications system equipment is shown in Fig. 1. The components may be divided into 1) *transmitting* equipment, 2) *receiving* equipment, and 3) *baseband* equipment.

#### Transmitting Equipment

The transmitting equipment comprises 1) an SSB exciter, transmitter, heat exchanger, power supply; and 2) an FM exciter, transmitter, heat exchanger, power supply.

The use of SSB requires very tight control of absolute frequency: frequency shifts more than a few hertz are not tolerable, since considerable signal distortion will result from the loss of harmonic relationships between the frequency components.

Because the transmit frequencies are in the 6-GHz region, stabilities of the order of 1 part in $10^8$ (long term) are necessary. Therefore all IF, RF, and reference frequencies are derived from a single, highly stable 5.0 MHz crystal

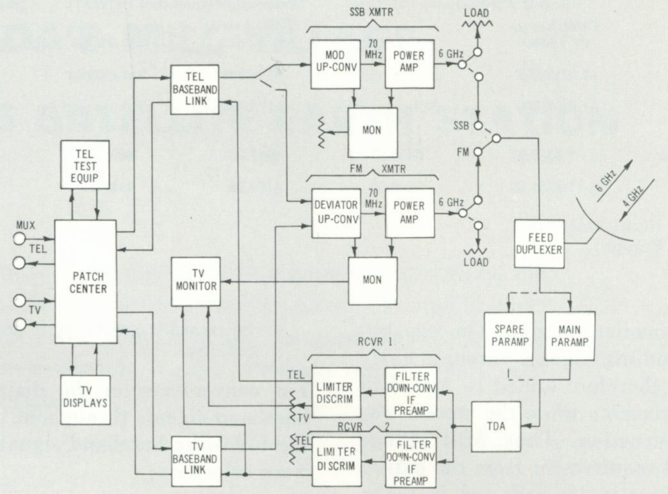



Fig. 1—Block diagram of the communication system of Mill Village.

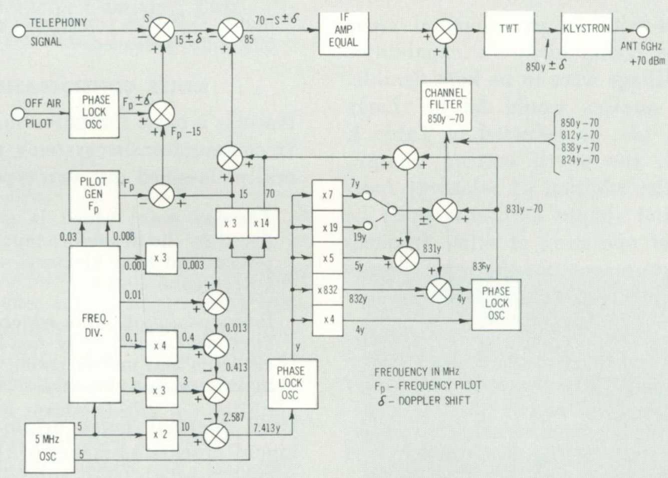



Fig. 2—Block diagram of the SSB transmitting system.

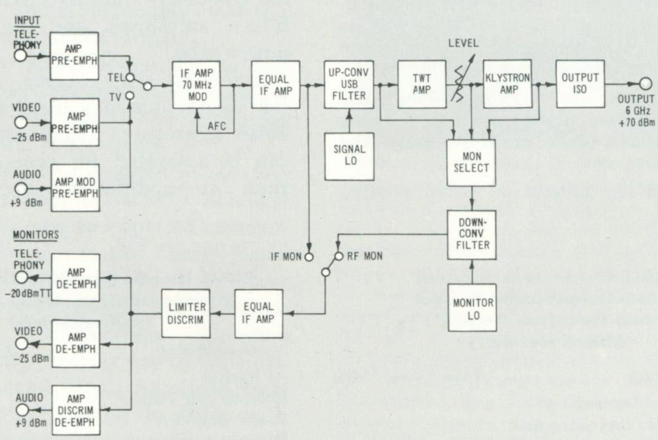



Fig. 3—Block diagram of the FM transmitting system.

oscillator. Fig. 2 is a block diagram of the SSB System.

The FM transmitting system is simpler in concept than the SSB since frequency amplitude correction feedback circuits are not required. However, FM must handle telephony and television, video and audio, therefore the flexibility and complexity of its baseband and its circuit parameter adjustments must be greater. Fig. 3 is a block diagram of the FM transmitting system; the waveguide transmission line has a switch permitting selection of either FM or SSB power as the radiating source.

Telephony and television video signals are handled conventionally. The audio signals may be handled in two ways: the subcarrier method, as adopted for the ATS, and a separate carrier method.

**Receiving Equipment**

Two separate RF signals may be simultaneously received. One signal may be the desired signal originated by another distant earth station while the second may be the "off-air" monitor for the station's own transmissions. To provide this capability, the following receiving equipment is used:

1) Two liquid helium cooled, two-stage parametric amplifier systems: a main wideband system and a standby narrowband system.

2) A two channel receiving system consisting of one low-noise two-stage room temperature tunnel diode amplifier followed by two sets of IF mixer-preamplifiers, IF main amplifiers, limiters, demodulators, and baseband amplifiers and de-emphasis circuits for telephony and television.

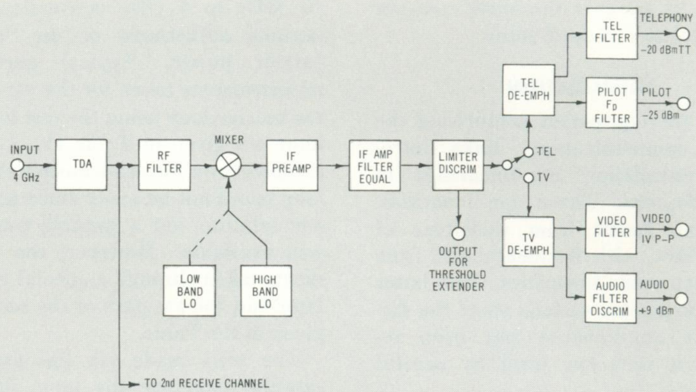
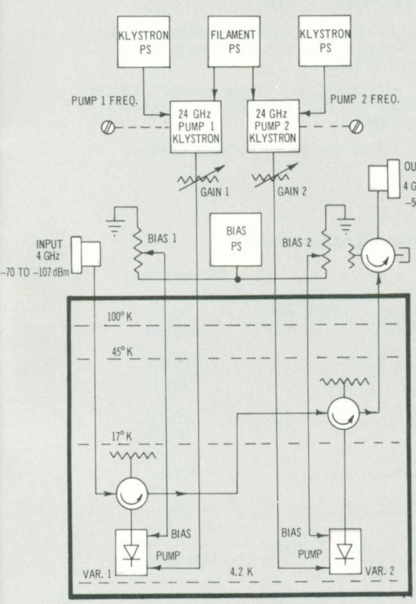



Fig. 5—Block diagram of a receiver system.

The parametric amplifiers, tunnel diode amplifier, and IF mixer preamplifiers are located in the feed cone to avoid the introduction of significant noise caused by waveguide losses. The remaining equipments are located in the annex. Fig. 4 is a block diagram of the parametric amplifier system; Fig. 5 is a block diagram of one channel of the receiver system.

Fig. 4—Block diagram of the parametric amplifier.




**Baseband System Equipment**

The baseband system includes the transmission links between the buildings, and the monitoring equipment necessary for control and analysis of the signals being handled. Such activities begin in the control building and center around control consoles and patching bays. At the consoles, the operators have access to system status indication, power level meters showing the transmitter power output, AGC indication of received signal level, and television switching and monitor equipment. Fig. 6 is a block diagram of the baseband system.

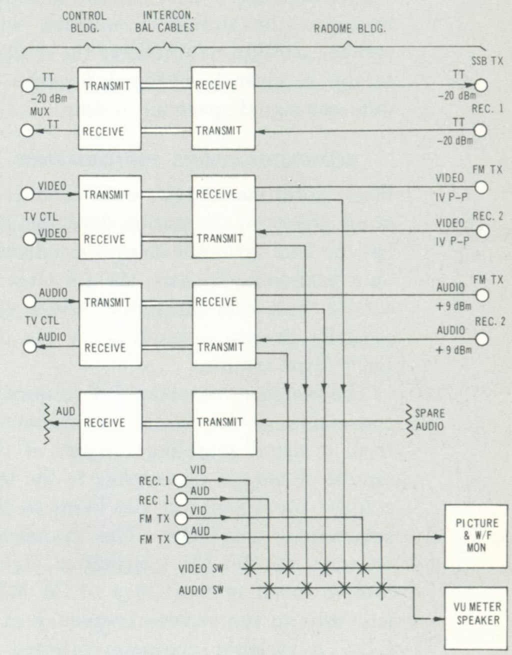
## AUDIO FOR EARLY BIRD

Facilities for receiving audio as a separate carrier were not provided until well into the station development after the plans and characteristics for a communications system using Early Bird were established by ComSat. These facilities were therefore added onto the existing scheme in the following manner.

Audio signals at standard level are used to drive an 81-MHz FM deviator, separate from the existing exciter but physically located within the exciter enclosure. The audio IF at 81 MHz is added to the video IF at 70 MHz, and the combined signal is upconverted by the exciter in the usual manner and thereby generating the audio carrier at 6400.97 MHz, the North American Early Bird frequency. The up-conversion process actually generates a second carrier at 6378.97 MHz (below the main carrier) but this is a spurious carrier and cannot be used for signal transmission. The ratio of useable video to audio carrier power at RF is adjusted to the desired value of 13 dB by adjusting the power level of the audio carrier at IF.

In the receive direction, use is made of the existing facilities for receiving two carriers simultaneously. A new local oscillator frequency of 4002.00 MHz was provided for one of the receivers to convert the audio signal for the North American receive frequency of 4092.00 MHz to an IF of 90 MHz. A high quality FM tuner was then tuned to 90 MHz and the audio signal was recovered in a conventional manner. An audio amplifier

Fig. 6—Block diagram of the baseband system.



was used to increase the tuner's output signal level to standard value.

### EQUALIZATION

Much of the equipment comprising the satellite communications links introduces transmission non-linearities in varying degrees. These non-linearities, in general, result from variations of group delays, with frequency and from discriminator non-linearities. The latter problem is not too serious since the discriminator non-linearity has been reduced to a very low level by careful design.

Group delay non-linearity on the other hand, is quite significant and special action is taken to overcome it. Significant contributors are the IF amplifiers, side-band filters, and klystron power tube in the transmitting system, the satellite electronics, the IF amplifiers, RF filters and IF filters in the receiving chain.

If the group delay follows a linear or parabolic law across the frequency band, this may be cancelled out by adding passive elements, or "equalizers" which have the opposite characteristic to the circuit. Higher order non-linearities cannot yet be equalized but fortunately they are of quite small magnitudes.

The transmit system group delay contribution is cancelled out first by adding equalizers in the exciter IF path so that the group delay from the deviator to the power amplifier output is essentially a constant.

The satellite and receiver group delay contributions are then cancelled by adding equalizers in the receiving IF path. These equalizers have to be changed whenever the station is working with another satellite, or whenever the IF filter in the receiver is changed because of different signal spectrum widths.

### COMMUNICATIONS PERFORMANCE

Many performance tests were conducted at all phases of the station development, but the final system tests were conducted on a back-to-back basis, via a 6 GHz to 4 GHz "test loop translator" which can partially imitate a satellite, or via the Early Bird satellite.

The test loop translator is a frequency conversion device which is connected from a signal sampling coupler at the output of the FM transmitter to the test coupler input point at the input to the parametric amplifier. The translator converts the 6-GHz transmitter signal sample to an IF frequency of 70 MHz and then to the receive frequency of 4 GHz. A variable attenuator inserted in the translator's local oscillator for the 70 MHz to 4 GHz conversion process permits adjustment of the "received" carrier power. Typical performance measurements taken for the FM transmit-FM receive loop using the test loop translator are given in Table IV. Loop measurements on the SSB-transmit-PM-receive loop could not be made since neither the ATS satellite, nor a suitable transponder was available. However, the measurements taken through a special PM modulator and the PM path of the receiver are given in the Table.

The tests made via the Early Bird satellite loop are the most interesting and useful for immediate purposes. Typical telephony, television video and television audio measurement will therefore be given. Noise power ratio measurements using 240 channel noise loading and pre-emphasis were performed on a back-to-back basis using a 2.5-MHz RMS multichannel frequency deviation and an 18-MHz noise bandwidth IF filter in the receiving system. The following readings were obtained:

| *Slot Frequency (kHz)* | *NPR (dB)* | *Weighted S/N (dB)* |
|---|---|---|
| 70 | 35.0 | 53.8 |
| 534 | 33.0 | 51.8 |
| 1002 | 33.0 | 51.8 |

These measurements were taken under dry weather conditions and are 1.8 dB better than the minimum CCIR weighted S/N ratio of 50.0 dB. For television, the weighted peak signal-to-RMS noise ratio was measured to be about 50 dB, using a peak-to-peak frequency deviation of 5.5 MHz and CCIR pre-emphasis. This is somewhat below the CCIR recommended minimum but it is equivalent to the performance requirements established by ComSat for television and is quite satisfactory. Other television performance deviations, such as line time linearity, were within recommended tolerances.

The audio maximum signal-to-RMS noise ratio was measured to be about 49.9 dB. This is slightly lower than that recommended by the CCITT for type-A broadcast lines but it is still quite reasonable and is in accordance with performance established by ComSat. This noise is mostly thermal noise, the video-to-audio crosstalk noise being about 7-dB lower. Audio distortion across the 50-Hz to 10-kHz band was less than 2.4% while the amplitude response was within ±1.1 dB over the same band.

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**TABLE IV—Communications Loop Performance Measurements**

| *Parameter* | *Measurement* |
|---|---|
| Noise power ratio for 300-channel noise loading, pre-emphasis, and 1-MHz RMS test-tone deviation for high C/N ratios (compare this with Table III) | 48.8 dB |
| Baseband frequency response from 60 kHz to 1300 kHz (300 channel) | +0 dB<br>−3 dB |
| Video baseband response from 10 kHz to 4 MHz (525 line television) | +0 dB<br>−5 dB |
| Line-time distortion (linear) | 0.5% |
| Field-time distortion (linear) | 2% |
| Audio frequency response from 30 Hz to 15 kHz for maximum signal on the 7.5 MHz subcarrier and a deviation of 140 kHz. 75 μsec pre-emphasis used. | ±2.2 dB |
| Audio distortion over the frequency range | 1% |
| Noise power ratio for 1200-channel noise loading and 0.25-radian RMS test-tone deviation. (PM loop only), high C/N ratio. | 40.4 dB |

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