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Journal of the Institution of Electronic and Radio Engineers

GUEST EDITORIAL

IERE NEWS AND COMMENTARY

The University of Surrey Satellite – UOSAT

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To promote the advancement of radio, electronics and kindred subjects by the exchange of information in these branches of engineering

Volume 52 Number 8/9 August/September 1982

The Radio and Electronic Engineer

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A University Satellite Project

THERE has been an increasing tendency in the past few years to draw attention to the shortcomings of UK engineering education. Universities in particular have been criticized for not producing graduates of the right quality, industry and universities have been criticized for not collaborating more closely, and university research claimed not to be relevant to the country's needs.

Without doubt there are elements of truth in all these statements, but in stressing the shortcomings there is a real danger of overlooking much that is not only good, but is actually better than in many other countries. The University of Surrey Satellite (UOSAT) project is just one of many examples of what is good in university engineering departments. It illustrates how a university department, with support from industry, has achieved a notable technical success in having built a relatively sophisticated spacecraft with an effort and at a cost representing only a small fraction of that hitherto normally associated with space projects. The project has also involved the closest collaboration between students and staff at all levels in the Department of Electronic and Electrical Engineering and, as such, has given both undergraduate and postgraduate students a taste of 'real' engineering at the same time as they have been learning the fundamentals of electronics. The project will also provide considerable opportunities for interested amateurs as well as professionals to receive, using relatively simple and inexpensive equipment, scientific data (including pictures) from space; it is very much to be hoped that the interest generated amongst school-children in particular will encourage more of our most able youth to take up electronic engineering as a challenging and worthwhile career.

It should be stressed, however, that *UOSAT* did not come about by chance. In the same way as invention favours the prepared mind, *UOSAT* came to pass in an environment which was prepared to accept the challenge when the opportunity arose. Thus the Department of Electronic and Electrical Engineering at Surrey had supported for many years a very active amateur radio society involving staff and students which had already carried out useful work for the amateur satellite community. One of its more notable achievements had been to take responsibility for the control of several early *OSCAR* satellites; it succeeded in prolonging the operational lifetime of one of these by several years.

Anticipating that a successful space shuttle could well transform the economics of spacecraft construction as a result of launches becoming very much cheaper and more readily available, members of the group were already convinced of the worthwhileness of attempting to build a satellite on an appropriately small budget to investigate the engineering, scientific, reliability and cost tradeoffs whilst achieving a technically useful mission.

The second way in which the Department was ready to take advantage of the opportunity was by virtue of its already close links with industry. Since its foundation, 17 years ago, a very close relationship with industry had been established, fostered to a significant extent by the fact that the Department's undergraduate course includes a mandatory year in industry (following completion of the second year of the course); moreover most of the students are unsponsored and the Department takes responsibility for finding a suitable industrial placement for each student.

Thus it was that when, in early 1978, the first hint of the possibility of a launch by NASA become apparent, the University's response was to provide 12 months' pump priming for the project during which the project manager (designate!) was to seek financial and technical assistance from industry; that was on the simple but unchallengeable basis that if industry considered the project worthwhile it would surely help; conversely if assistance was not forthcoming it would be a clear indication that the project was not worth undertaking. Within 12 months of seeking the views of industry approximately £200k was raised, roughly half in cash, and half in kind. The project thus began in earnest in January 1979.

The *UOSAT* project also highlights strengths of the university system that are all too readily overlooked. First, it is possible to work closely with experts from various disciplines in other university departments both within one's own university and elsewhere. Secondly, universities and their departments invariably have a management structure which allows such ambitious, even eccentric, proposals to gather momentum with a minimum of bureaucratic constraint. It is worth pointing out that the time intervals from its inception (when even the experiments which the satellite would carry were by no means decided) to the delivery of the spacecraft to the launch pad was only 30 months. I think it fair to say that few organizations—industrial or governmental—could have accommodated a project of this magnitude in so short a time scale.

And now for a few words about the spacecraft itself. It has been variously dubbed by some of the popular technical press as an 'amateur' or 'educational' satellite. From the papers in this Special Issue, however, it will be apparent that neither adjective correctly describes the spacecraft. The misnomers have arisen because the satellite has been so designed that the data which it generates can be easily received by simple and inexpensive groundstation equipment such as might be readily available to individual amateurs or educational establishments as well as to professional engineers and scientists. This is not to suggest, however, that the data are any less valuable to professionals than would have been the case had more sophisticated techniques been used. As an illustration, one might mention the on-board magnetometer instrument which, with a resolution of ± 2nT, is one of the most sensitive instruments of its kind currently circling the Earth; the data from this instrument will clearly be of considerable interest to professional geophysicists.

Many readers of this Journal will know that, following a successful launch and seven months of extremely encouraging results, *UOSAT* ran into difficulties in April 1982 which have prevented the completion of its stabilization programme. At the time of writing (July 1982) it is still hoped that the current difficulty will be overcome with the help of colleagues in other parts of the world who have more extensive facilities than those at Surrey. *UOSAT* has, however, already fulfilled most, though not all, of its major objectives. Most importantly, it has demonstrated that satellite activities need no longer be regarded as the exclusive province of the telecommunications and defence giants of the world. Relatively small but sophisticated satellites (acknowledging the cost/reliability/confidence trade-offs) *can* be built relatively quickly and inexpensively. With the Space Shuttle now a reality rather than an untested idea (as was the case when the *UOSAT* project began) there could well be a demand for such satellites dedicated to carry out one particular job (e.g. a specific Earth-resource survey). In this respect, *UOSAT* has played a pioneering role.

In conclusion I would like to add a personal note of gratitude to the Editor of the Institution's Journal and the members of its Papers Committee. For obvious reasons, the satellite group at Surrey were under great technical pressures over the period when the papers for this Special Issue were being written and the forbearance and assistance of the IERE are much appreciated.

J. D. E. Beynon

Head of the Department of Electronic and Electrical Engineering
University of Surrey

INSTITUTION OF ELECTRONIC AND RADIO ENGINEERS

Notice of Annual General Meeting

NOTICE IS HEREBY GIVEN that the twenty-first ANNUAL GENERAL MEETING of the Institution since Incorporation by Royal Charter will be held on THURSDAY, 28th OCTOBER 1982, at 6.00 p.m. in the Goldsmiths' Hall, London School of Hygiene and Tropical Medicine, Keppel Street, Gower Street, London W.C.1.

AGENDA

1 **To receive the Minutes of the twentieth Annual General Meeting of the Institution since Incorporation by Royal Charter, held on 29th October 1981.**
Reported on pages 9–12 of the January 1982 issue of *The Radio and Electronic Engineer*.

2 **To receive the Annual Report of the Council for the year ended 31st March 1982.**
To be published in the October 1982 issue of *The Radio and Electronic Engineer*.

3 **To receive the Auditors' Report, Accounts and Balance Sheet for the year ended 31st March 1982.**
To be published in the October 1982 issue of *The Radio and Electronic Engineer*.

4 **To confirm election of the Council for 1982–83.**
In accordance with Bye-law 49 the Council's nominations were sent to corporate members by a notice dated 17th May 1982 in the June 1982 issue of *The Radio and Electronic Engineer*. As no other nominations have been received under Bye-law 50 for the following offices, a ballot will not be necessary and the following members will be elected.

The President

Under Bye-law 45, the President retires each year but may be re-elected provided he does not serve thereby for more than two years in succession.

For re-election: H. E. Drew, C.B., Hon. C.G.I.A.

Vice-Presidents

Under Bye-law 46, all Vice-Presidents retire each year but may be re-elected provided they do not thereby serve for more than three years in succession.

For re-election: Colonel W. Barker; L. A. Bonvini; D. L. A. Smith, B.Sc.(Eng.); Group Capt. J. M. Walker, R.A.F.

For election: R. Larry; R. W. Wray, B.Sc.

Honorary Treasurer

For re-election: S. R. Wilkins

Ordinary Members of Council

Under Bye-law 48, Ordinary Members of Council are elected for three years and may not hold that office for more than three years in succession.

FELLOWS

The following must retire: R. Larry.
For election: Professor D. P. Howson, D.Sc., M.Sc.; G. A. Jackson, B.Sc.

MEMBERS

The following must retire: D. J. Kenner, B.Sc., M.Sc.; B. Mann, M.Sc.; K. R. Thrower.
For election: A. F. Dyson, Dip.El.; Professor P. A. Payne, Ph.D.; M. M. Zepler, M.A.

ASSOCIATE MEMBER

The following must retire: P. J. Hulse.
For election: Commander A. R. B. Norris, R.N. (Retd).

ASSOCIATE

The following must retire: M. V. Wright.
For election: D. R. Caunter.

The remaining Council members will continue to serve in accordance with periods of office laid down in Bye-law 48.

5 **To appoint Auditors and to determine their remuneration.**
Council recommends the re-appointment of Gladstone Jenkins & Co., 50 Bloomsbury Street, London, W.C.1.

6 **To appoint Solicitors.**
Council recommends the re-appointment of Bax, Gibb & Company, 14 Gray's Inn Square, London, W.C.1.

7 **Awards to Premium Winners.**

8 **Any other business.**
Notice of any other business must have reached the Secretary not less than forty-two days prior to the meeting.

2nd August 1982

By Order of the Council,
S. M. Davidson, *Secretary*

IERE BENEVOLENT FUND

The Annual General Meeting of Subscribers to the Benevolent Fund will be held immediately following the Institution's Annual General Meeting, starting at approximately 6.45 p.m.

Institution Premiums for 1981

The Council of the Institution announces that authors of the following papers are to receive Premiums for outstanding contributions published in the Journal during 1981.

MAIN PREMIUMS

CLERK MAXWELL PREMIUM (*Value £100*)

For the outstanding paper on the science of electronics or radio

'Experiments on the incendivity of radio-frequency breakflash discharges (1·8-21 MHz c.w.)'
Dr J. L. J. Rosenfeld, Dr D. C. Strachan, Dr P. S. Tromans and P. A. Searson (*Shell Research*) (April)

LORD MOUNTBATTEN PREMIUM (*Value £100*)

For the outstanding paper on the engineering aspects of electronics or radio

'Radio frequency ignition hazards'
D. J. Burstow, Dr R. J. Loveland, R. Tomlinson and D. W. Widginton (*Health & Safety Executive*) (April)

HEINRICH HERTZ PREMIUM (*Value £75*)

For the outstanding paper on the physical or mathematical aspects of electronics or radio

'Optical fibre transmission lines'
Professor W. A. Gambling, Dr A. H. Hartog and Dr C. M. Ragdale (*University of Southampton*) (July/August)

MARCONI PREMIUM (*Value £75*)

For the outstanding paper on the engineering of an electronic system, circuit or device

'A manpack satellite communications earth station'
C. H. Jones (*Royal Signals & Radar Establishment*) (June)

SPECIALIZED TECHNICAL PREMIUMS

PAUL ADORIAN PREMIUM (*Value £50*)

For the outstanding paper on communications or broadcasting engineering

'Optimum diversity separation for over-sea line-of-sight radio links'
A. J. Henk (*Brown & Root*) (November/December)

LORD BRABAZON PREMIUM (*Value £50*)

For the outstanding paper on aerospace, maritime and military systems

'Results from an experimental dual-band search radar'
P. D. L. Williams (*Racal-Decca Marine*) (November/December)

A. F. BULGIN PREMIUM (*Value £50*)

For the outstanding paper on the theory or practice of electronic components or circuits

'Semiconductor laser sources for optical communications'
Dr P. A. Kirkby (*Standard Telecommunications Laboratories*) (July/August)

ARTHUR GAY PREMIUM (*Value £50*)

For the outstanding paper on production techniques in the electronics industry

'Signature analysis for board testing'
J. R. Humphrey and K. Firooz (*Hewlett Packard*) (January)

J. LANGHAM THOMPSON PREMIUM (*Value £50*)

For the outstanding paper on the theory or practice of systems or control engineering

'A framework for systems engineering design'
Professor P. K. M'Pherson (*City University*) (February)

CHARLES WHEATSTONE PREMIUM (*Value £50*)

For the outstanding paper on electronic instrumentation or measurement

'Characterization of single-mode optical fibres'
Dr K. I. White, Dr S. Hornung, J. V. Wright, Dr B. P. Nelson and M. C. Brierley (*British Telecom Research Laboratories*) (July/August)

GENERAL PREMIUMS

ERIC ZEPLER PREMIUM (*Value £50*)

For the outstanding paper on the education of electronic or radio engineers

'Integrated optics: a tutorial review'
Dr P. J. R. Laybourn and Professor J. Lamb (*University of Glasgow*) (July/August)

HUGH BRENNAN PREMIUM (*Value £50*)

For the outstanding paper first read before any of the Local Sections and published in the Journal

'A review of magnetic bubble memories and their applications'
K. F. Baker (*Plessey Research (Caswell)*) (March)

Student Exchanges to be Encouraged

The concern felt at the falling-off of support by Britain for the International Association for the Exchange of Students for Technical Experience (IAESTE) which was commented upon in an editorial article in the May Journal under the heading 'Swapping Students', has been brought recently to the notice of the Government. Lord Mottistone, D.L. (Fellow) put down a Question for Written Answer in the House of Lords in which he asked what efforts have been and are being made to encourage student exchange under the auspices of IAESTE, since the use of the scheme by the UK has dropped from receiving 909 students and sending 883 students in 1960 to 88 'in' and 125 'out' in 1981. He further pointed out that this compared unfavourably with the Federal Republic of Germany's 1981 figures of 916 'in' and 726 'out'.

Replying on behalf of the Government, Lord Elton stated that to halt the decline in IAESTE's activities in this country the Government had encouraged the Central Bureau for Educational Visits and Exchanges in 1980 to assume responsibility for the day-to-day administration of the scheme in the United Kingdom. The United Kingdom national committee now has a permanent headquarters for the first time in its history, together with a full-time secretariat with support services. This has enabled it to canvass more than 2,000 firms in the past year seeking sponsorships. It has also introduced a new scheme of sponsorship. Lord Elton believed that these initiatives were modestly encouraging. Movements into and out of the United Kingdom this year are expected to grow to at least 100 and 130 respectively, with further expansion expected in the future.

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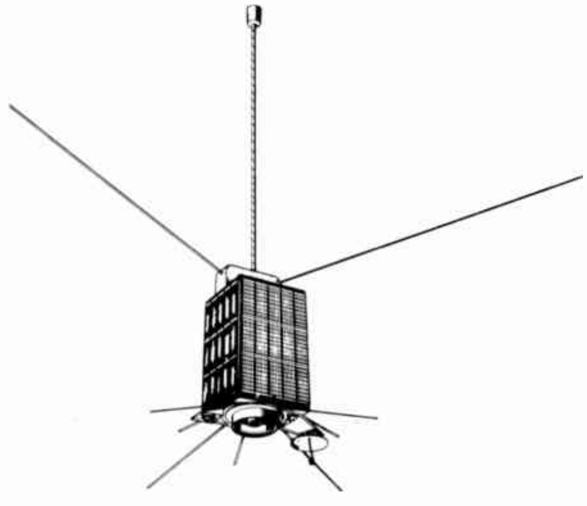
UOSAT—An investigation into cost-effective spacecraft engineering

M. N. SWEETING, B.Sc., Ph.D.*



SUMMARY

The *UOSAT* Project commenced at the University of Surrey in 1979 to investigate the feasibility of, and the engineering problems associated with, the design, construction, launch and orbital operation of a relatively small and inexpensive spacecraft capable of a worthwhile contribution to the scientific, engineering and educational communities.

The paper outlines the background to the project and the objectives of the *UOSAT* mission. After describing briefly the experiments on board the spacecraft, the paper deals with the design and fabrication of the satellite, the philosophies underlying these and the test and procurement procedures. Details of the project management and the resources necessary are given. The paper concludes with a summary of the present status of the spacecraft and its experiments.

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1 Introduction

The United Kingdom has actively pursued a national space programme since the early 1960s, initially supporting both launch vehicle and spacecraft development. The remarkably successful development of the *Black Arrow* launcher proved too costly and was axed, following the first and only successful all-British launch of *UK*-3, in 1971. Britain's national spacecraft programme continued with *UK*-4 and 5, however the operation of the last planned satellite *UK*-6, launched in June 1979, was terminated in February 1982.[1] A national space programme was simply too expensive and could not be justified based on accepted costings, whilst pooling resources with large and prestigious international projects appeared far more attractive.

Britain now has a flourishing and well respected space industry and, together with many research organizations, provides a substantial contribution to international projects within ESA and NASA. Large international scientific or commercial projects often, however, find it difficult to accommodate many relatively small but nevertheless interesting scientific and engineering experiments.

The Amateur Space Programme, through the construction and operation of a number of low-cost communications spacecraft, has demonstrated the basic feasibility and usefulness of a small spacecraft programme operated within a very modest budget.[2] Several British industrial and research organizations expressed a keen interest in the possibilities such a programme could offer and declared their support for a project to demonstrate its feasibility in this country.

The relevance of low-cost spacecraft is directly related to the availability of similarly priced launch opportunities. Conventional launches are prohibitively expensive, however, low-cost opportunities do exist in the form of secondary payloads, and whilst the Space Shuttle has yet to fulfil the promise of an inexpensive free-flying small payload facility in a useful orbit, future cost-effective alternatives may appear.[3,4] Given, albeit occasional, low-cost launch opportunities, it appears well worthwhile to investigate the feasibility and capability of a low-cost space programme.

The primary objective of the UOSAT Project was therefore to demonstrate the feasibility of and to investigate the engineering problems associated with the design, construction, test, launch and orbital operation of a relatively small and inexpensive spacecraft capable of significant contribution to the scientific, engineering and educational communities.

2 Background

The Department of Electronic and Electrical Engineering of the University of Surrey has been actively involved in low-cost space studies since 1974, when the student-based Electronic and Amateur Radio Society established a simple tracking station for receiving telemetry data from the Amateur Radio communications satellite *OSCAR*-6.

Sixteen Amateur Radio communications spacecraft have been launched since 1961, eight (in the *OSCAR*

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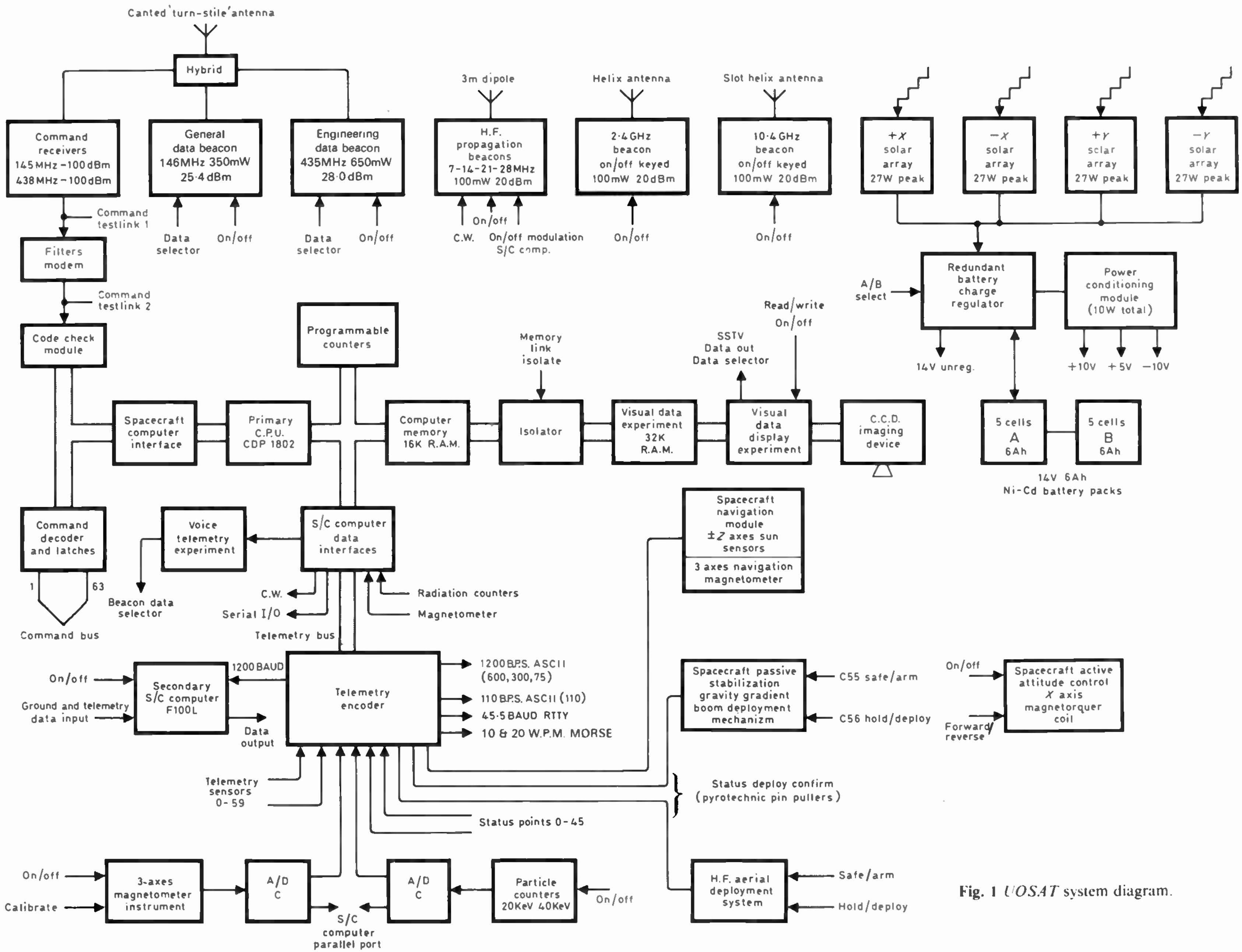


Fig. 1 *UOSAT* system diagram.

series) by the USA and eight (in the RS series), more recently, by the USSR. The Amateur Radio Satellite Corporation (AMSAT), with headquarters in Washington D.C., is responsible for the co-operation of the international Amateur Space Programme in the West, the spacecraft being constructed by radio amateurs who are, however, often professional aerospace engineers.

The success and enthusiasm shown in Surrey's early activities prompted AMSAT to request the Department's assistance, in 1975, with the global command network controlling the operation of *OSCAR*-6 with responsibility for maintaining the spacecraft's correct mode of operation over Europe.[5] *OSCAR*-6 operated within a negative power budget and without autonomous control so it was therefore essential to monitor the spacecraft closely with effective ground control in order to extend its useful life to a maximum.

A more sophisticated Telemetry and Telecommand groundstation was commissioned in late 1975 and operated by undergraduate and postgraduate students and staff who monitored and controlled *OSCAR*-6 until its demise in 1977—having extended the spacecraft's operational lifetime from the designed one year to over 4·5 years.

The groundstation grew in complexity and, following *OSCAR*-6, was intimately involved with the subsequent Amateur Radio communications spacecraft *OSCAR*-7 and 8. The Department further developed a facility for the reception and display of image data from the *Tiros*, *NOAA* and the Russian *Meteor* series of meteorological spacecraft on a daily basis. A very inexpensive television display for these image data was developed, which formed the basis of a later *UOSAT* experiment.

By 1978 a strong interest had grown within the Department in developing flight hardware based on a low-cost philosophy as demonstrated by the AMSAT programme. It was felt that this philosophy of cost-effective engineering could be adopted successfully in this country to support a worthwhile domestic space programme on a very modest budget, whilst presenting a demanding technical and managerial challenge.[6]

3 *UOSAT* Mission Objectives

In order to evaluate the broad project objectives outlined above, specific spacecraft mission objectives were set as follows:

To evaluate the operational performance and reliability of low-cost engineering techniques in a space environment.

To provide professional scientists with a readily available tool to carry out studies of the near-Earth electromagnetic environment and the relationship between solar and geomagnetic disturbances and their effect on radio wave propagation between h.f. and microwave frequencies, requiring minimal and inexpensive groundstation equipment.

To stimulate a greater degree of interest in space science in schools, colleges and universities through active participation, by ensuring that the experiment and housekeeping data are transmitted from the spacecraft in such a manner that they are easily accessible, not only to professional scientists, but also those with very simple, low-cost groundstations.

To broaden the scope of the Amateur Space Programme and to cater for the interests of the 'amateur scientist'.

To provide radio amateurs with direct access to a sophisticated tool enabling them to study the near-Earth electromagnetic environment and its relationship to radio wave propagation throughout the spectrum of amateur radio frequency allocations. The immense number and global distribution of radio amateurs provides a unique opportunity for observing little known and ephemeral phenomena.

4 *UOSAT* Mission Profile

The mission objectives dictate a low, polar Earth orbit—in particular to ensure availability of the spacecraft from all points on Earth and to keep the data downlink budgets healthy.

NASA were approached through AMSAT for launch support on educational grounds and upon approval, the Delta Project Office identified the Solar Mesosphere Explorer (*SME*) mission as having a suitable configuration and sufficient payload margin to carry a secondary spacecraft. *SME* is designed for a 3 p.m–3 a.m. Sun-synchronous, circular, polar Earth orbit at an altitude of 560 km. Having identified a specific launch opportunity, the mission profile was reworked to match the orbit characteristics and a spacecraft payload outline assembled. The orbital lifetime of a small spacecraft in this orbit would be around 3·5 years before decay.

The spacecraft payload was considered as two components—service modules and experiment modules. (See Fig. 1.) The service modules provided all the functions fundamental to the basic operation of the spacecraft, such as: power sources and conditioning, telemetry, telecommand, antenna systems and downlink transmitters. The experiment complement was defined as follows.

4.1 Electromagnetic and Propagation Studies Experiments

A series of experiments was designed to study the near-Earth electromagnetic environment and its relationship with radio wave propagation throughout the radio spectrum from h.f. to microwave frequencies.

4.1.1 *Magnetometer experiment*

A three-axis, wide-range, flux-gate magnetometer based on the highly successful instrument flown on the *Voyager* and *MAGSAT* spacecraft missions provides the facility to examine the fine structure of the Earth's magnetic field, any disturbances to it and their relationship to radio wave propagation. This instrument has a dynamic range of $\pm 64{,}000$ nT and a resolution of ± 2 nT compared to the *MAGSAT* instrument with a resolution of $\pm 0{\cdot}5$ nT.[7] Although the accuracy of the survey of the absolute value of the geomagnetic field will be restricted by the precision to which the attitude of the spacecraft can be determined, the *UOSAT* magnetometer will allow high-resolution measurements of the temporal variations in the geomagnetic field vectors over a considerable period of time. The magnetometer is used during the commissioning phase or orbital operation (prior to boom deployment) for precise navigational information in conjunction with, and also to calibrate, the low resolution navigation magnetometer described later.

4.1.2 *Particle radiation experiment*

Two particle radiation detectors, one detecting electrons of energies greater than 20 keV and another electrons of energies greater than 40 keV, provide real-time data on solar activity and auroral events. The variations in extent of the auroral oval and the effect of radiation events on radio wave propagation are of particular interest. The detectors are also sensitive to protons of approximately ten times the electron energies. Quick-look data from this instrument are also available in routine real-time telemetry to indicate the possibility of extended range v.h.f./u.h.f communications.

4.1.3 *H.f. radio beacons experiment*

A series of phase-referenced radio beacons transmitting on 7, 14, 21 and 29 MHz will support a wide range of trans-ionospheric radio wave propagation experiments and observations. It will be particularly valuable to be able to correlate propagation data with information from the magnetometer and particle instruments.

4.1.4 *V.h.f., u.h.f. and microwave radio beacons experiment*

Radio beacons on 146 MHz, 435 MHz, 2·4 GHz and 10·5 GHz allow the study of the effects of solar and geomagnetic disturbances, local weather conditions and other phenomena on propagation up to centimetre wavelengths, from a wide variety of 'look' angles. It is also hoped that the microwave beacons will encourage radio amateurs, with their history of very low-cost, innovative, design techniques, to develop inexpensive microwave satellite groundstations.

4.2 Spacecraft Engineering Systems Experiments

A major objective of the *UOSAT* mission is to evaluate the performance and reliability of low-cost techniques applied to spacecraft engineering. For this purpose the spacecraft carries a number of systems experiments:

4.2.1 *Stabilization and attitude control*

The camera and microwave beacons experiments require one facet of the spacecraft to remain Earth pointing, whilst the relatively limited power budget would not support continuous active attitude control. A study of various methods of passive stabilization resulted in the adoption of a technique using gravity gradient forces in conjunction with active damping control employing a magnetorquer which interacts with the local geomagnetic field. The magnetorquer is also used for initial attitude control before deployment of the gravity gradient stabilizing boom. The attitude and damping manoeuvres are carried out by the on-board computers in conjunction with navigation data from the spacecraft magnetometers and Sun sensors.

4.2.2 *Telemetry system*

The spacecraft housekeeping and experiment data telemetry system provides information on the real-time status of the spacecraft systems and is routed to the v.h.f, u.h.f. and s.h.f. downlink transmitters in a variety of formats. In keeping with the mission objectives, 60 analogue channels and 45 digital status points are encoded by the system and made available in the following formats to cater for a wide range of user groundstation facilities:

1200 baud ASCII†
600 baud ASCII† ⎫
300 baud ASCII† ⎬ available as an alternative to 1200 baud
75 baud ASCII† ⎭
110 baud ASCII
45·5 baud Baudot r.t.t.y.
10 or 20 w.p.m. morse code
Synthesized speech (computer controlled)

Any pair of the above formats (excluding amongst those marked †) are available independently via the v.h.f. and u.h.f. downlink transmitters simultaneously. The 1200 baud chain marked † has a channel dwell facility.

4.2.3 *Spacecraft computers*

Advances in integration level and power reduction of solid-state memory devices have made in-orbit data collection and computer control of the spacecraft increasingly attractive. In spite of the efforts of NASA and ESA, there remains some uncertainty concerning the long-term performance and failure mechanisms of these devices in a space radiation environment. There is also considerable interest in the performance of various types of languages and control algorithms for use with operational spacecraft.

Two powerful on-board computers based around the RCA 1802 and Ferranti F100L microprocessors and utilizing different types of c.m.o.s. memory devices have been incorporated into the spacecraft. They have access to the spacecraft experiments, telemetry and telecommand systems to enable the following functions to be carried out:

telemetry surveillance
command and status management
experiment data storage and processing
closed-loop attitude control using the magnetorquer
dissemination of orbital data, operating schedules and spacecraft 'news'

The primary spacecraft computer (1802) has direct high-speed data links with the magnetometer and radiation experiments to allow rapid data acquisition yielding fine time-resolution information whilst it also controls the operation of the speech synthesizer experiment. The primary computer uses 16 K bytes of Mostek 4116 dynamic c.m.o.s. memory devices and also has direct access to 32 K bytes of similar memory in the imaging experiment thus providing a facility of greater data storage and image processing.

The secondary spacecraft computer (F100L) represents a 'minimum' configuration system with only three serial ports to the command and telemetry systems. The F100L has, nevertheless, complete control over the spacecraft, by acting as a local ghost groundstation, and it can gather telemetry and experiment data at 1200 baud. The computer has access to 16 K bytes of static c.m.o.s. memory comprising Harris 6564 devices.

The computer software is resident in the random access c.m.o.s. memory loaded from the ground via the telecommand links. The software can be modified or replaced during flight by a Ground Command Station in order to execute specific tasks, accommodate changes in the mission profile and to allow for the rectification of possible hardware or software failures. Both computers did not have any operational software resident during launch, and may be loaded with either low or high level languages whilst in orbit. The primary and secondary computers may communicate with each other via a simple serial link.

4.2.4 *Power systems*

The provision of electrical power by solar panels often accounts for a major portion of the spacecraft construction budget. In keeping with the low-cost philosophy, four solar panels fabricated by Solarex Corporation (USA) provide around 30 W each when fully illuminated, at a procurement cost of £12,000. It is believed that these panels represent the lowest cost per watt ever orbited. The performance of these solar panels can be monitored via the telemetry system throughout the lifetime of the spacecraft. NiCd batteries were furnished by AMSAT-USA from flight spares.

4.2.5 *Navigation magnetometer*

It is essential that the attitude of the spacecraft, relative to the Earth, is known to within a few degrees in order that the spacecraft may be oriented appropriately for gravity-gradient stabilization and subsequently for nutation damping manoeuvres. A very simple three-axis, flux-gate magnetometer, in conjunction with two Sun-sensors, provides primary navigation information to within approx. $\pm 3^\circ$. This instrument is calibrated in flight against the primary magnetometer instrument before deployment of the gravity-gradient boom.

4.3 Educational Experiments

The aerospace industry requires a steady stream of engineers, scientists and technical support staff to maintain and enlarge its workforce. At present the overwhelming majority of recruits have very little, if any, experience or appreciation of space science or engineering and require considerable training before they become productive. It is therefore most important that every effort be made to encourage and familiarize students and school children with the basic concepts of space science and engineering and to capture their interest and imagination at the earliest possible age. A particularly effective way of doing this is to provide school children with an opportunity for active participation in spacecraft experiments.

4.3.1 *Earth imaging camera experiment*

It has been observed that, during visits to the University, young people are particularly fascinated by the land, sea and cloud cover images received from space via the *Tiros*, *NOAA* and *Meteor* meteorological spacecraft and that these images generate an interest in geography and also in space science in general. It was decided, therefore, that an Earth Imaging Experiment would be of particular educational interest. It would also provide a very considerable technical challenge!

An essential aim of this experiment was that the display of the received image data should be as easy and cheap as possible whilst maintaining reasonable visual quality. Electrolytic paper and wet photographic techniques are both expensive and inconvenient whilst polaroid oscillographs lack size and definition. Almost every home, on the other hand, has a television receiver and a very inexpensive television display of *Tiros* images had already been developed within the Department at a component cost of around £75.

The camera optics are kept pointing earthwards by the gravity gradient stabilization and the imaging device comprises a charge-coupled-device (c.c.d.) two dimensional array operating in a 'snap-shot' mode. The image is stored in a solid-state memory as 256×256 pixels with a 4-bit (16-level) grey scale. The primary spacecraft computer has access to the image memory area and thus may perform image processing or indeed generate images and graphs. The camera optics are arranged to cover a 500×500 km area of the Earth's surface yielding a maximum resolution of about 2 km. The response of the c.c.d. is more sensitive towards the red end of the spectrum and should show land features and land/sea boundaries clearly. The integration time (exposure) of the c.c.d. may be varied by the ground command station from 3 to 30 ms. Images may be gathered from regions of Earth out of range from the command station under instruction from the spacecraft computers and stored in memory for transmission later. Little is known about the performance of c.c.d.s of this type in a space environment.

4.3.2 *Synthesized speech experiment*

The *OSCAR* series of amateur radio spacecraft pioneered the use of simple transmission modes and formats to bring space science within reach of the classroom by transmitting telemetry data in morse code, thus removing the requirement for elaborate decoding equipment. However, even morse code can be a significant hurdle for schoolchildren and for the simplest of educational experiments the v.h.f./u.h.f. downlink transmitters can be modulated with synthesized voice. The National Semiconductor speech synthesizer is controlled by the spacecraft primary computer which may assemble letters, numbers and words from the synthesizer's vocabulary. The computer may be programmed from the ground to assemble fixed messages or to convert certain telemetry or experiment data into speech.

4.3.3 *Telemetry*

The variety of telemetry formats available are designed to cater for the widest possible spectrum of user facilities by providing morse code at 10 and 20 words per minute, 45 baud r.t.t.y. (Baudot) for the many amateur r.t.t.y. operators in existence, and 75 to 1200 baud ASCII for those with small or home computing facilities. The choice of data modulation was made compatible with home computer cassette storage standards. The ease with which the spacecraft data can be interfaced should allow individual experimenters to carry out quite sophisticated studies using small 'home' computer systems.

5 Spacecraft Design

The overall spacecraft design, and indeed the final experiment complement, was highly dependent on the characteristics of the launch vehicle configuration and programmed orbital elements of the primary payload (*SME*).

Discussions between University of Surrey, British Aerospace, Delta Project Office and McDonnell Douglas gradually resulted in the evolution of a spacecraft design.

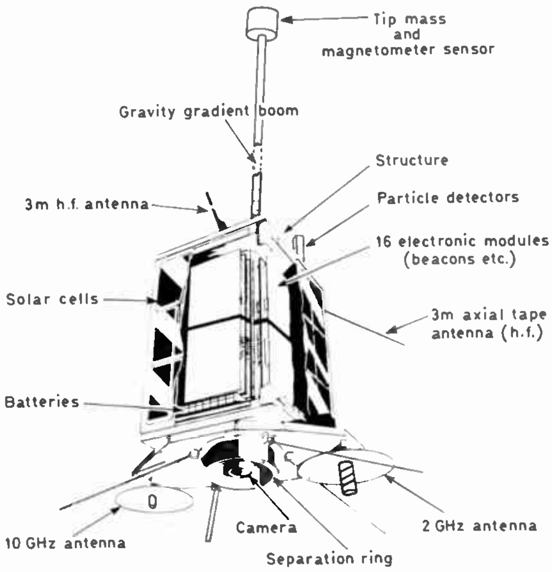


Fig. 2 Spacecraft structural diagram.

5.1 Structural Design (Fig. 2)

The spacecraft body is constructed as a rectangular box ($a \times b \times c$; 42·5 × 42·5 × 73·5 cm) supported by a central thrust column which supports the 'top' ($+z$) and 'bottom' ($-z$) panels. The four sides of the spacecraft body are covered with solar cells for the generation of primary electrical power. The electronic sub-systems are contained within 16 identical module boxes mounted in pairs on the central column. The two battery packs are supported on the 'bottom' floor panel and the h.f. antenna tuning network and particle detector's e.h.t. supply underneath the 'top' panel. The v.h.f, u.h.f and microwave antennas are attached under a 'wing' on the bottom of the spacecraft body, whilst the top of the wing supports the navigation magnetometer and the 10 GHz beacon. The spacecraft-launch vehicle attach fitting is mounted on the $-z$ facet panel and through to the central column. The 'top' panel supports the h.f. antennas and associated pyrotechnics, the particle detectors, the magnetometer and the Sun-sensor. Lifting points are provided in the corners of the $\pm z$ panels.

A deployable boom is stowed in the middle of the central column in the form of a roll of pre-stressed beryllium copper tape. When released slowly by means of a small electric motor, the copper tape forms a hollow tube which supports a 2·5 kg tip mass containing the magnetometer sensor out to a maximum distance of 13·7 m. The boom and tip mass also provide the gravity gradient stabilizing torque. The lower portion of the central column contains the Earth-pointing c.c.d. camera experiment.

A 300-turn magnetorquer is wound around the inside of two opposing solar array facets ($\pm x$) and the v.h.f. and u.h.f. antenna feed hybrid fabricated from semi-rigid coaxial cable is attached to the 'floor' panel inside the spacecraft.

The inclusion of magnetometers on *UOSAT* entailed careful control of magnetic materials used during fabrication. The entire spacecraft framework is fabricated from aluminium alloy and, where greater strength is essential, stainless steel. All the bolts and captive fasteners are similarly of non-magnetic stainless steel. The supporting side and floor members are formed by bending and jigs were made for the module box supports and rivet hole positions. The components of the structure are riveted resulting in a low-Q structure as each rivet is able to move slightly and thus absorb energy when experiencing vibration. The spacecraft body was anodized to prevent corrosion; however, electrical continuity of less than 2·5 mΩ was assured between any two points on the spacecraft body.

5.2 Thermal Finishes

Thermal analysis of the heat flow around the spacecraft body stepped through one orbit showed that the energy dissipated by the spacecraft electronics could be considered negligible compared to the energy falling on the solar arrays during the illuminated portion of the orbit. In order to maintain a reasonable operating temperature the top of the spacecraft is completely, and the 'bottom' partly, covered with silvered Teflon optical solar reflector (OSR). The reverse facet of each of the solar array panels is covered with Kapton film in order to radiate heat away from the array itself when illuminated. The spacecraft should maintain a slow residual spin around the z axis even when stabilized, in order to even out thermal gradients. The spacecraft has been designed to operate with a battery temperature between 0 to +20°C.

5.3 Electrical Harness

Inter-module electrical connections are made using standard 25 way 'D' connectors, with a maximum of three along both sides of each module box. The wiring harness assembly runs up the outside corners of the central column and around its 'waist'. The 'D' connectors are high temperature mouldings with recessed pins and all wiring is p.t.f.e. (Teflon) coated. Each connection to the 'D' connectors is sleeved with p.t.f.e. tubing, the connector secured with captive bolts and the joints supported by RTV potting compound.

The electronic module boxes are all mechanically identical to ease fabrication, assembly and integration with the spacecraft.

6 Design, Fabrication, Testing and Procurement Philosophies

6.1 Design

An important factor in keeping the overall cost of the spacecraft to a minimum whilst maintaining acceptable reliability was the set of ground rules adopted for the design of the service and experiment sub-systems. Adopting a 'gold-plated' philosophy throughout would cause the cost to escalate and might not necessarily improve reliability. The ground rules applied to the design were:

Keep it simple and examine thoroughly the task and environment of the sub-system and specify components/techniques that will accommodate the task with a suitable, realistic safety margin. Do not simply go for the highest rated /quality approach as this will generally increase costs dramatically, unless the function strictly requires it.

Service modules should use standard, well-known proven designs and hardware wherever possible. In view of the very short development time available, avoid indeterminate software development appearing in critical paths, even at the expense of reduced flexibility and greater hardware count.

Redundant paths and experiment sub-systems may use less well-known or proven designs or technologies providing they are not associated with potential single point failure nodes.

Use flexible design giving redundancy via alternative technologies.

Use easily defined, simple interfaces between sub-systems wherever possible. Assume sub-system to run independently of other systems even if it entails duplicate hardware (unless there is an obvious advantage to be gained).

Design around established, industrial, high-grade, volume production components and attempt to procure Hi-rel or MIL-SPEC screened versions. If a component has been in volume production it is highly likely that any 'bugs' have been removed.

Ensure that all components to be used in sub-systems can be procured within the development schedule (including flight hardware).

6.2 Procurement

Component and materials procurement posed a particular problem. First, the development time of the spacecraft was extremely short and 'shopping lists' of components had to be assembled as early as possible. This proved difficult as many sub-systems were slow in final definition. Long-lead items may take six months a arrive.

Hi-rel components were procured wherever possible and appropriate, however, when these were effectively unavailable, then MIL-SPEC and, failing that, high-grade industrial components were used. Any industrial components were subjected to a 1000 hours burn-in and parameter measurement. A group of reliable suppliers was identified for the procurement of the majority of components and materials, but where small quantities, long-lead or special items were required then contact was made on an individual basis within industrial and research establishments to assist by making these difficult-to-get items available rapidly. This proved to be an extremely effective and important mechanism and the co-operation of all those who assisted in any way is greatly appreciated.

6.3 Fabrication/Production

Except for those sub-systems contributed from AMSAT-USA and Germany, Microwave Modules and Plessey (magnetometer, power sub-system and the microwave beacons), the remainder of the spacecraft was designed and fabricated within the University of Surrey. This entailed assembling a number of facilities.

6.3.1 *Mechanical*

All the spacecraft structure and mechanics, with the exception of the sub-system boxes, were fabricated in the Department of Electronic and Electrical Engineering mechanical workshops employing the equivalent of one man for 15 months. The on-site facility allowed rapid mechanical development and easy fit-checks with any subsequent modification essential for the maintenance of the tight schedule. The sub-system boxes were milled from solid aluminium alloy using a numerically-controlled mill at the Royal Aircraft Establishment workshops at Farnborough. Sufficient mechanical hardware was fabricated for three spacecraft structures.

6.3.2 *Electrical*

An essential facility for the development of the electrical sub-systems was the computer-aided printed circuit layout machine—the Racal Cadet. This facility enabled us to design the p.c.b. layouts in-house, often by, or at least associated with, the sub-system designer, yielding fewer mistakes and far more rapid turn-around than could be achieved by sub-contract. Five members of the team were taught to use the machine and after around one week's experience it generally took about 1·5 days to lay out a single (double-sided) p.c.b. for the standard sub-system box containing around 45 i.c.s. More important still was the ability to have artwork back within one day, a prototype p.c.b. within three days and later, modifications to final flight p.c.b.s within a week!

The c.c.d. imaging experiment and primary spacecraft computer p.c.b.s were generated by a sub-contractor and CERN respectively.

6.4 Test

The electronic sub-system development followed the well-trodden path of bread-board, engineering model and flight model.

6.4.1 *Bread-board*

All the spacecraft sub-systems were constructed in a bread-board arrangement initially to assess overall performance, interface compatibilities and to uncover any unexpected problem areas. Provisional component procurement, interface and harness documentation was generated at this stage.

6.4.2 *Engineering model*

Engineering models of the spacecraft sub-systems were used to evaluate detailed system performance, interface and electromagnetic compatibilities, spurious emissions and responses and mechanical integration problems. Each sub-system was subjected to Flight Qualification Environmental Tests as follows (Fig. 3(a) and (b)).

Vibration—After initial screening test, 1·5 times the levels and duration of the Delta Restraints Document.

Thermal—100 hours thermal cycling between +50°C to −30°C.

Life test—1000 hours soak test at room temperature.

Antenna—a full-scale r.f. model of the spacecraft structure with antennas was evaluated.

Deployment—deployment of the gravity gradient boom and h.f. antennas were tested.

6.4.3 *Flight model and spacecraft integration*

Due to the extremely tight schedule (launch date had been provisionally brought forward six weeks), the flight-rated spacecraft structure that had been used for the flight acceptance vibration tests and the launch vehicle fit-check was cleaned to be used as the flight model. All flight hardware was assembled, and the spacecraft integration carried out in the clean area using clean procedures (Fig. 3(c)). The assembled sub-systems underwent through test and preliminary calibration before a screening environmental test sequence carried out at Guildford, followed, if satisfactory, by a sinusoidal sweep vibration test at flight acceptance levels using the R.A.E. facility nearby at Farnborough. The sub-systems were thermally cycled between +50°C to 30°C on a 12-hour cycle for three days. Wherever possible, an additional 1000 hours operation at room temperature was also completed.

Final flight acceptance tests of the integrated spacecraft were carried out at the environmental test facility at British Aerospace, Stevenage:

Spin balance—spacecraft structure underwent both static and dynamic balance to within ± 10 g.m.

Vibration—all axes to levels and duration specified in the Delta Spacecraft Design Restraints Document.

Thermal vacuum—five-day thermal cycling between +40°C and −20°C at 2×10^{-5} torr.

Solar array performance tests were also carried out at the R.A.E. facility and *VI* calibration curves obtained with reference to AM0 (Air Mass zero).

The structure was de-gaussed and magnetic cleanliness tests performed at the Goddard Space Flight Center Magnetic Test Facility (USA). The primary and navigation magnetometers were also calibrated using this facility (Fig. 3(d)).

Electromagnetic compatibility tests were carried out both at University of Surrey and the Western Test Range; however, it was not possible to perform these tests effectively in an anechoic chamber.

7 Launch Agency

Initial contact with NASA was established through AMSAT, whose headquarters are in Washington D.C., with a Spacecraft Laboratory within the Goddard Space Flight Center. AMSAT agreed to act as the *UOSAT* Project representative in the USA and a joint approach was made to the NASA Administrator for support of the mission. NASA had supported several previous AMSAT missions on educational grounds and agreed to support *UOSAT* on similar terms, broadly as follows:

That NASA would cover all costs relating to the launch providing every effort was shown to have been made to minimize additional cost to NASA.

That no additional orbit constraints be requested unless they represent a negligible overhead on existing mission requirements.

That *UOSAT* would meet whatever schedule was agreed by the Primary Mission.

That *UOSAT* represented a minimal safety hazard to the Primary Mission.

That *UOSAT* would cover all costs relating to the spacecraft and post-launch tracking, telemetry and telecommand activities.

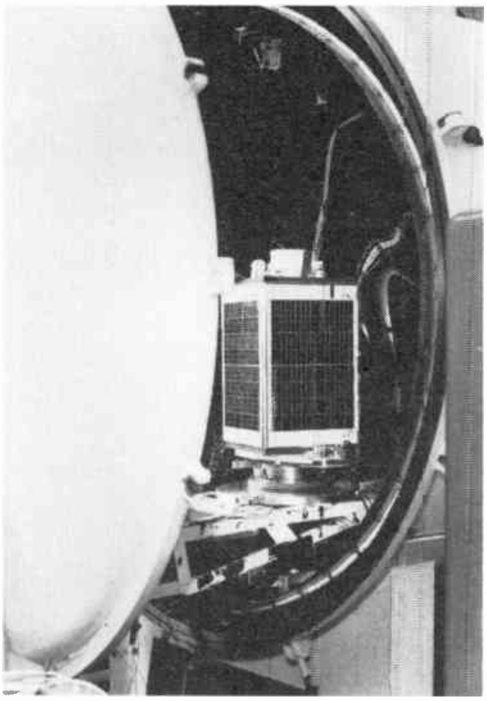
That *UOSAT* would satisfy the launch agency of the mechanical integrity and overall safety of the spacecraft and interfaces, and would provide supporting documentary evidence.

The NASA Delta Office was approached in January 1979 to identify possible launch opportunities, i.e. those scheduled launch slots for low polar orbits with spare payload capacity and with suitable configurations for accommodating a secondary payload. Several possible missions were identified, the most encouraging being *IRAS* and *SME*. *IRAS* was due for launch in 1982 which fitted the *UOSAT* timetable well; however, being a cryogenic astronomical spacecraft *IRAS* would impose stringent cleanliness constraints. Finally, the *SME* mission was chosen for ease of integration although the advanced launch date of autumn 1981 imposed a severe timetable on the development and integration of the *UOSAT* spacecraft.

Whilst there was direct contact between the Delta Project Office and the University of Surrey, AMSAT acted as a local *UOSAT* representative and dealt most effectively with the day-to-day matters thus minimizing travel (three *UOSAT* visits to Delta and one Delta visit to University of Surrey). AMSAT-USA were, of course, heavily involved in the *UOSAT* Project as they contributed the primary magnetometer instrument and thus were quite familiar with *UOSAT*.

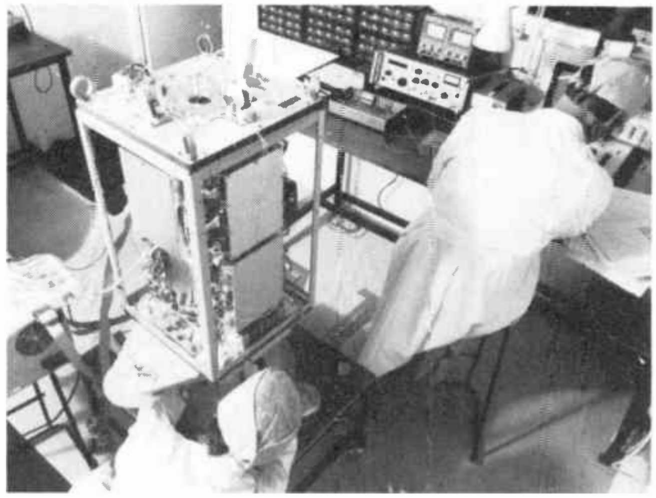
The paperwork normally required by NASA presented a severe problem to the small *UOSAT* team who had neither the manpower nor the experience to comply fully. Delta responded by agreeing to minimize the paperwork to that necessary to satisfy the mission specification and safety requirements, whilst AMSAT agreed to advise UOSAT closely on the preparation of the necessary documents, comprising:

Spacecraft questionnaire
Mission requirements
UOSAT spacecraft structural analysis
UOSAT launch procedures
UOSAT safety drawings and procedures.

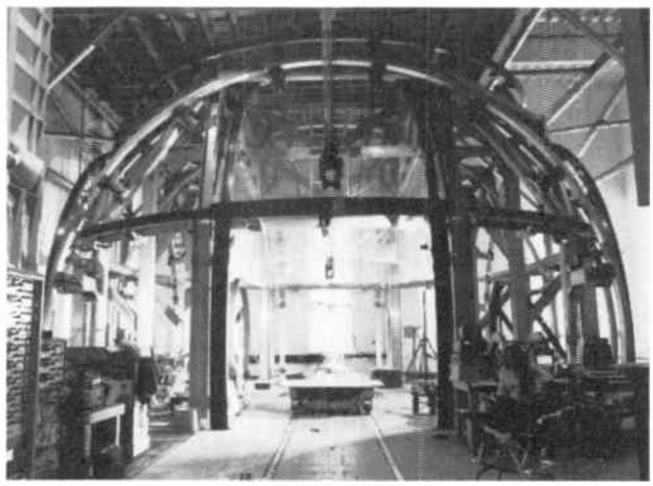


(a) Spacecraft during thermal vacuum at British Aerospace, Stevenage.

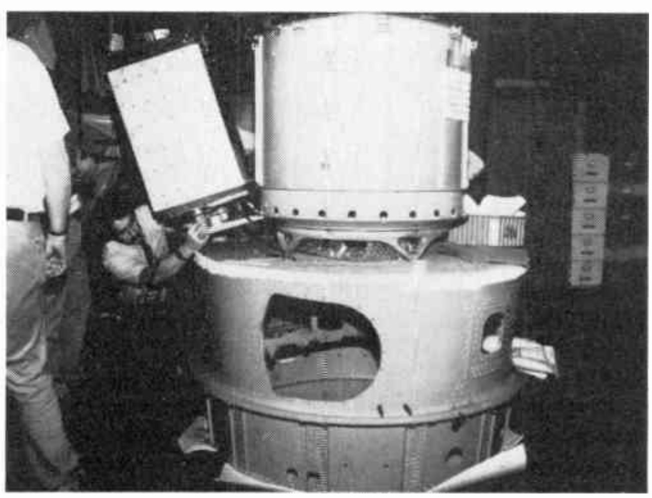
(b) Flight acceptance vibration tests at British Aerospace, Stevenage.



(c) Integration within the clean area at the University.



(d) Calibration of magnetometers at the magnetic test facility, Goddard Space Flight Center.



(e) Engineering model fit check with Delta 2310 launch vehicle during integration at McDonnell Douglas, Long Beach, California.

Fig. 3

A documentation schedule was agreed with Delta taking into account the *UOSAT* timetable, although, as usual, this timetable proved difficult to maintain due to the pressures of spacecraft development.

Two major reviews took place between UOSAT—AMSAT—Delta. The first was a 'fit-check' at the McDonnell Douglas Delta production facility at Long Beach, California where the flight *UOSAT* spacecraft structure was mated to the upper stage of the vehicle (Fig. 4). This proved to be particularly valuable as several potential integration problems were identified in time to be rectified and it was also possible to run through the detailed spacecraft mating procedures. A second major review was held at University of Surrey three months before launch where final launch campaign details were refined and final Delta integration and structural analysis examined.

UOSAT requested minimal support facilities from the launch site at Vandenberg Air Force Base, agreeing to cover all launch support requirements from either University of Surrey or AMSAT. Considerable launch support was provided by AMSAT members in California in terms of test equipment and necessary logistics. UOSAT requested a 400 sq ft clean area and a similar office area at Vandenberg with appropriate power sources only. In fact, the NASA-Delta-MDAC launch support staff were only too pleased to provide any additional support necessary during the campaign, and were most helpful.

The *UOSAT* launch campaign comprised:

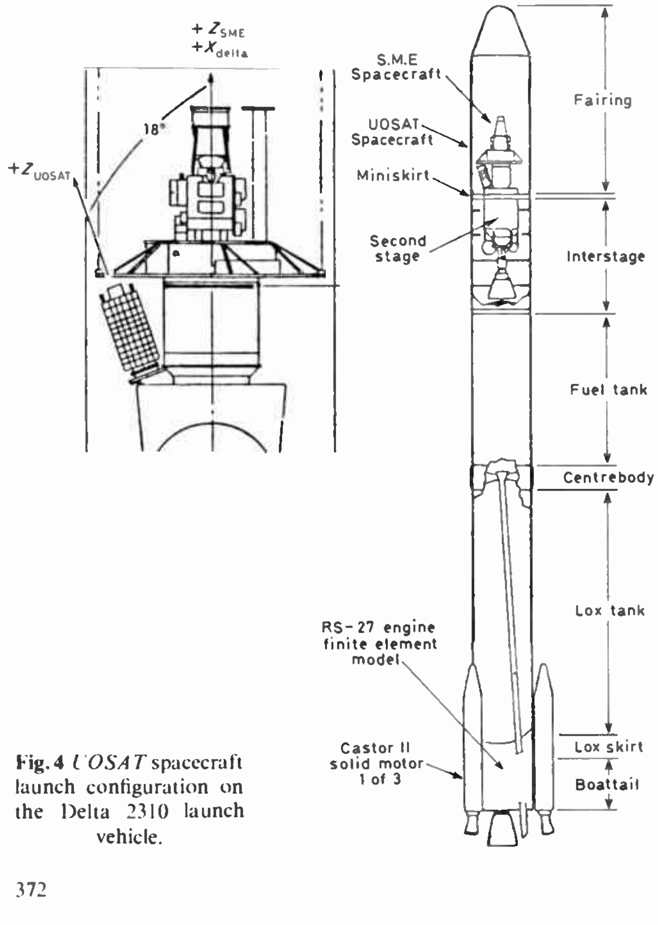


Fig. 4 *UOSAT* spacecraft launch configuration on the Delta 2310 launch vehicle.

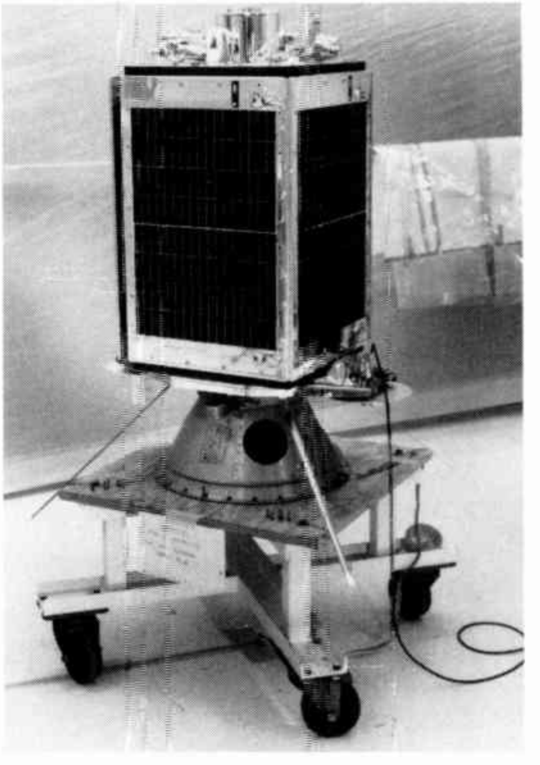


Fig. 5 *UOSAT-OSCAR 9*: the satellite ready for final integration in the launch vehicle.

2 days shipping by air.
3 days magnetometer calibration at GSFC.
2 days shipping by air.
4 days spacecraft final flight preparations
3 days final spacecraft functional and calibration checkout.
1 day spacecraft integration with Delta 2310 vehicle.
23 days spacecraft enclosed in nitrogen purge bag awaiting launch

8 Project Management

Effective management is crucial to the success of a project of this nature and the following ingredients are considered essential. It is only possible to complete a low-cost spacecraft programme if the project team exhibits:

high degree of motivation, determination and endurance
above average multi-disciplinary technical capability
geographic compactness
limited numbers to ensure effective communication

Considerable importance was attached to ensuring effective communication without the burden of an onerous paperwork structure. All the critical spacecraft services and designs were carried out within the University or supported by a very few previously well 'calibrated' outside teams. By maintaining the primary design, fabrication, test and integration facilities within a 200 metre radius the overhead on communication was minimized whilst retaining reliable flow of information and assuring compatible interfaces.

There is, however, a maximum team size beyond which a more elaborate and formal (and hence cumbersome and expensive) organizational structure has to be adopted. It would seem that the *UOSAT* project team approached that limit with around thirty members. Although not all the team were full-time that did not significantly alter the structural operation. It is just possible for one person to manage a team of that size with appropriate delegation of responsibilities. One of the most difficult aspects of the general management problem was that associated with the organization of a predominantly part-time and to some extent, volunteer team to ensure timely delivery of hardware and the speedy resolution of critical problems. A high degree of motivation and collective responsibility is therefore necessary within the team.

The nucleus of the *UOSAT* team comprised effectively three full-time staff over the duration of the project, with the following broad responsibilities:

Project Manager: (2·5 years) project schedules; spacecraft conceptual and system design; fund raising and budget control; distribution and organization of effort; interface with Project Sponsors, NASA, Environmental Test Facilities, Delta; design and construction of telecommand system hardware; spacecraft integration; UOSAT-NASA-Delta paperwork;

Project Engineer: (1·5 years) sub-assembly mechanical design and fabrication; module assembly; particle experiment electronic design and fabrication; environmental testing; spacecraft integration; launch close-out procedures;

Research Assistant: (1 year) electrical design and fabrication of Power Distribution Module, Safety Circuits; electrical harness design and fabrication;

Research Fellow: (1 year) design and prototype imaging experiment;

Research Fellow: (0·5 years) final design and fabrication of flight imaging experiment;

Mechanical Technician: (1 year) fabrication of spacecraft structure and sub-assemblies.

The overall organization structure of the Project is shown in Table 1; however, in hindsight, it is clear that a fourth full-time member was really necessary whose responsibilities would cover documentation and interface control, component procurement and harness design.

Table 1 Organizational Chart

Project Manager — Sponsors

University of Surrey Full-Time Staff | University of Surrey Part-Time Staff | External Groups

PROJECT MANAGER:
M. N. Sweeting

SPONSORS:
Ferranti, British Telecom, Racal, MEL, Government Communications HQ, Radio Society of Great Britain, University of Surrey, AMSAT-UK, Science and Engineering Research Council

UNIVERSITY OF SURREY FULL-TIME STAFF:
Project Engineer—I.C. Ferebee (Radiation Experiment)
Project Technician (Ministry of Defence)
Project Engineer—J. Z. Slowikowski (Power)
Project Technician (Ministry of Defence)
Research Student—S. K. Lee (Charge Coupled Devices)

UNIVERSITY OF SURREY PART-TIME STAFF:
Project Engineer—M. J. Blewett
Antenna—A. K. Brown K. Keen
Attitude Control—B. A. Packham
E. G. Lawrence
M. S. Hodgart
Charge Coupled Devices—C. P. Traynar
Mechanical—F. Keitch
A. J. Duffield
R. L. Finch
Primary Computer—T. G. Jeans
Secondary Computer—C. L. F. Haynes
M. Stubbs (Ferranti)
Telemetry—L. S. A. Mansi
R. S. Clarke
U.H.F./V.H.F. Transmitters—R. W. Haining
H.F. Beacons, C. R. Smithers
Finance—D. J. Pollard
Secretary—L. S. J. Crawford

EXTERNAL GROUPS:
AMSAT-Germany—K. Meinzer (Power)
AMSAT-USA—J. A. King (NASA/Delta/Procurement)
M. H. Acuna (Magnetometer Experiment)
University of Sheffield—B. Chambers (10 GHz Antenna)
Plessey—J. Arnold (10 GHz Beacon)
Microwave Modules plc—R. Porter (2GHz Beacon)
Rutherford Appleton Laboratories—D. R. Lepine (Particle Experiments)
Independent Consultants—R. Gape
J. Rabson
C. Trayner (1802 Software)
Navigation Magnetometers—C. Sweeting

9 Project Resources

The resources necessary for the successful completion of any space project, however modest are considerable. A primary objective of the *UOSAT* Project was to identify the essential resources necessary to build, launch and operate a small spacecraft. It is well known that such a programme can be undertaken by large organizations, e.g. NASA, ESA and the aerospace industries; however, it was questionable whether a small group could muster and utilize the necessary resources successfully.

The essential resources were identified and located as follows.

9.1 Finance

A cash budget covering the design, construction, testing and launch of a small spacecraft was drawn up assuming a three-year project and a launch provided at no cost by NASA, summarized under the following headings:

Salaries: £60,000
Capital Equipment: £5,000
Components: £20,000
Test Facilities: £10,000
Travel: £5,000

The total estimated budget in 1979 was £100,000, whilst the total cost to launch in 1981 was £118,500.

The funds were raised primarily from British Industry and Research Organizations on the premise that hard-

Fig. 6 The Ground Control Station for the spacecraft at the University of Surrey.

earned money from Industry would indicate that the project was worthwhile! Funds were provided from the following:

Ferranti Ltd. MEL Ltd.
British Telecom Racal Ltd.
Government Communications Headquarters
AMSAT-UK Radio Society of Great Britain
University of Surrey Research Committee
Science and Engineering Research Council
Private Donations

Environmental Test Facilities were contributed by British Aerospace (Stevenage) to a maximum value of £15,000.

9.2 Personnel .

Detailed Project Management has been discussed elsewhere in this paper, however, to summarize, it was necessary to assemble a small team of workers with sufficient experience and talent to cope with the wide range of engineering problems associated with a satellite project. A small full-time nucleus comprising:

Project Manager
Project Engineer
Research Assistant
Research Fellow
Project Technicians (2)
Project Mechanical Technician

were augmented by a number of academics and research staff at the University and in other institutions working part-time.

9.3 Development and Fabrication Laboratory Facilities

A clean area for the assembly of the flight modules and the integration of the spacecraft was not initially available at the University, so a small clean-room was constructed from wood with a polythene roof within an existing laboratory. The clean-room measured 12 ft × 12 ft × 8 ft (3·65 × 3·65 × 2·45 m) and was kept under positive pressure by a filtered-air pump to maintain a dust-free atmosphere. The inside of the clean-room had been painted four times at intervals of four days after pressurization to 'stick down' any dust. The cost of the clean-room was around £350 and proved to be considerably cleaner than most of the external test facilities. Gowns, gloves, over-shoes and hats were worn at all times in the clean area which was frequently vacuumed. By far the greatest amount of debris found was of human origin (hair, fluff).

A separate development laboratory area was established adjacent to the clean-room where the spacecraft modules were developed, tested and the ground support equipment assembled.

A 200 sq. ft. area was used for the Project Office and the assembly of the Ground Control Station adjacent to the main tracking antenna system.

10 Orbital System Performance and Experimental Results

Following the successful launch and orbital insertion of the *UOSAT* spacecraft on 6th October 1981, the 145 MHz downlink transmitter was activated on the first orbit from the command station at University of Surrey (Fig. 6). The downlink data selectors were initially set to monitor the data uplink and the next day the telemetry system was activated and 300 baud data transmission commenced. Initial telemetry data indicated that all the service sub-systems were performing correctly and that the spacecraft was stable and spinning around the z-axis at a rate of once every 27 seconds. The spacecraft batteries stabilized at +3°C and since launch have cycled between −5°C and +6°C on an approximately six-week cycle.

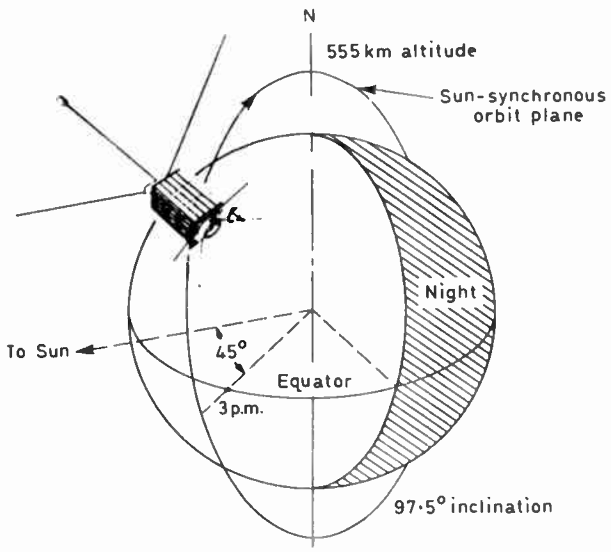
A checkout sequence was then initiated, progressively powering-up the engineering and science experiment sub-systems—all systems responded successfully. The importance must be stressed of a thorough and systematic checkout of the spacecraft computers, control algorithms and the calibration of the navigation sensors' initial spacecraft orientation, before any attitude manoeuvres are attempted.

11 Operational Status

11.1 Attitude

It has been widely thought that *UOSAT* is 'tumbling' in a somewhat uncontrolled manner—this is not the case. The spacecraft is, in fact, spin-stabilized with its z-axis fixed in inertial space, but, as the orbit plane is Sun-synchronous and not inertial, the observed attitude of the spacecraft from Earth changes from orbit to orbit and slowly with the seasons (Fig. 7).

UOSAT was ejected from the Delta launch vehicle with a considerable spin around the z-axis. The subsequent motion is complex, involving transverse spin around an inertial axis which results in a cone angle that oscillates as a function of the seasonal characteristics of



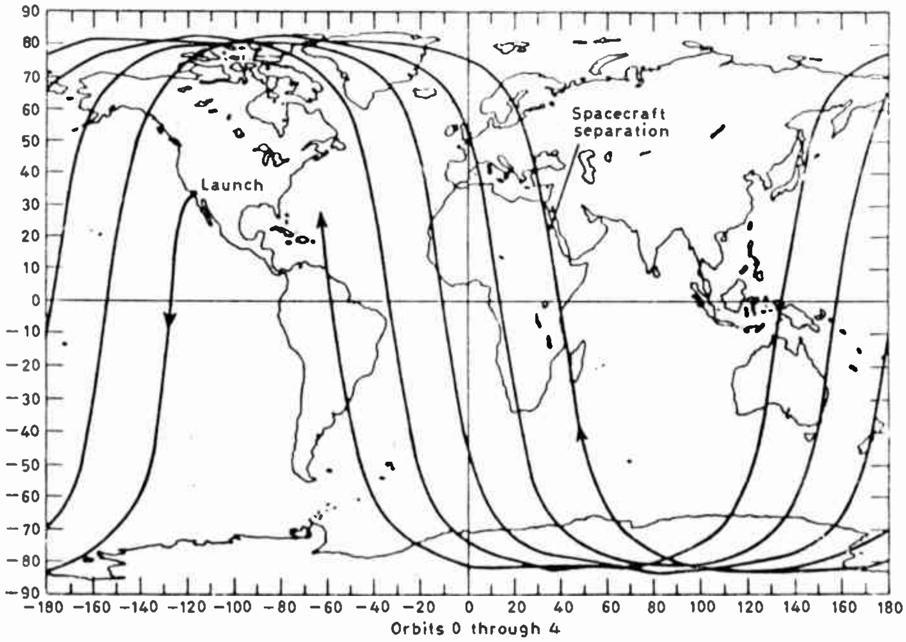


Fig. 7 *UOSAT* orbit geometry and typical ground tracks.

the non-inertial orbit plane. At certain times the spacecraft motion is also under the influence of small gravity-gradient forces—even without the boom extended. The effects of these long-term oscillations are most clearly observed from variations in the spacecraft battery temperature (Fig. 8) and, in a less obvious manner, the reception of the 435 MHz data beacon. The effect of air drag has been to slow the transverse spin rate from around 25 seconds (just after launch) to around 80 seconds (March 1982), whilst the body (z-axis) spin remains virtually unchanged at around 30 seconds.

11.2 Thermal Control

The spacecraft electronics have been designed to operate within the temperature range -30 to $+60$°C; however, the overall spacecraft thermal design aimed at an operational temperature of between 0°C and $+20$°C. The capacity of the NiCd batteries drops off below -10°C and above $+40$°C. The telemetry system monitors the temperature of a number of points around *UOSAT*; however, the battery pack and solar array temperatures provide a clear picture of the thermal behaviour of *UOSAT*. The thermal inertia of the batteries is such that they respond only to the long-term variations, whereas the solar arrays have a quite low thermal inertia and respond quickly to passing in and out of sunlight. The battery temperature monitored since launch exhibits a cyclic nature due to the changing average attitude of *UOSAT* to the Sun, whilst the arrays respond to the rapid spin and eclipse period within a single orbit (Fig. 8). The spacecraft temperatures have all settled to almost ideal values, however, these may change slightly with the different average attitude after gravity gradient stabilization has been achieved. *UOSAT* will be given a slow residual z-spin of around 0·1–0·2 rev/min after stabilization to even out thermal gradients.

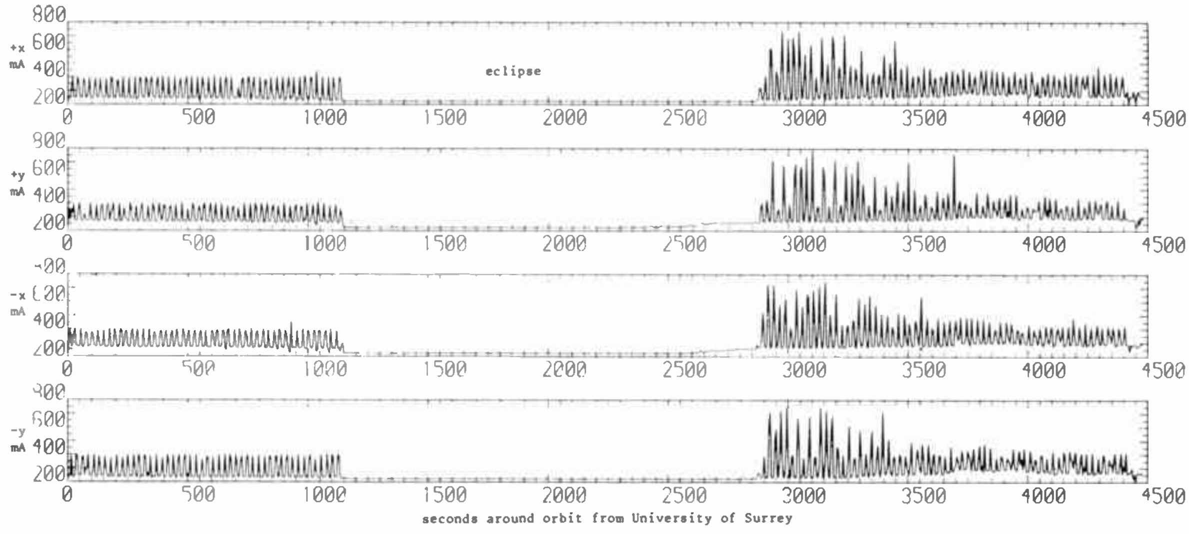


Fig. 8 Survey of *UOSAT* solar array currents sampled by secondary spacecraft computer from orbit 2472.

11.3 Power Systems

The on-board NiCd battery packs have remained fully charged since launch. The Solarex solar arrays have made more average power available than *UOSAT* needs at present. Peak power estimates indicate a maximum of 50 W from the arrays, up to 10% being contributed by Earth albedo at this low altitude! The battery charge regulator and power conditioning modules built by AMSAT-Germany are all performing nominally.

11.4 Data Beacons

Both 145 MHz and 435 MHz data beacons have been activated and are operating as expected, providing 350 mW and 650 mW output respectively. The 145 MHz general data beacon gives an extremely strong signal, easily received on a hand-held receiver. The modulation deviation on the 145 MHz beacon is lower than indicated before launch for synchronous data sources and nominal for asynchronous data sources. This difference came about because of a compromise between the levels available from the two sources that was not practicable to equalize before launch. The 435 MHz beacon has nominal deviation from synchronous sources and wider than nominal for asynchronous sources as it is anticipated that this beacon will carry mainly high-speed data.

11.5 Telemetry System

All modes of the telemetry system are operating perfectly and have provided 350 M bits of valuable data on the performance of the spacecraft systems and experiments.

11.6 Telecommand System

All modes of the telecommand system have been checked and are operating perfectly. An initial problem was experienced with the command and computer data audio phase from the groundstation, but this has now been resolved. The u.h.f. command receiver squelch status flag is often seen active, monitoring experiments have indicated the presence of high terrestrial noise levels and powerful radar signals (at u.h.f.). These noise levels are often sufficient to operate the receiver squelch but do not interfere with ground command.

11.7 Primary Spacecraft Computer

The primary spacecraft computer (RCA1802) has been successfully loaded with a number of software packages to exercise the experiment data ports, the voice synthesizer, the c.w. port, and the command port. The two most important functions provided by the computer at this stage have been the acquisition of high-speed magnetometer data (6 samples per second) for navigation purposes and the automatic operational control of *UOSAT* using the computer command port. The operation of the computer command port has proved to be unreliable in a single-shot mode (possibly due to a timing problem) so multiple commands are issued to ensure reliable on-board control. As each computer command only takes around 6 μs this problem represents only a small operational overhead. The most important task of the computer is attitude control and then, after stabilization, operations scheduling and nutation damping. *UOSAT* will shortly operate under IPS, a real-time, multi-tasking language.

11.8 Secondary Spacecraft Computer

The secondary spacecraft computer (F100L) has been carrying out regular data whole-orbit data collection and spacecraft status monitoring. It has enabled us to gather engineering and experiment data from entire orbits when out of range of the ground command station and dump them for analysis on later orbits. This facility has allowed us to study the operation of *UOSAT* very closely to ensure its correct behaviour and also to gather magnetometer and radiation data over the North and South Polar regions. The computer has also carried plain text bulletins containing spacecraft schedules and orbit data (1200 baud ASCII on 145·825 MHz). The computer is yielding useful information of the performance of 64 K r.a.m. memory devices in a space environment.

11.9 Magnetometer Instrument

The magnetometer has been providing real-time data on the Earth's magnetic field via the telemetry system since shortly after launch. High-speed data has been made available via the primary spacecraft computer and stored 'whole Earth' surveys available via the secondary computer. The instrument is operating perfectly. Software has recently been loaded into the primary computer to present the high speed magnetometer data in formatted hexadecimal and, more recently still, in plain decimal units

11.10 Radiation Experiment

The 20 keV particle detector has been providing real-time data on the radiation environment via the telemetry system and more recently 'whole Earth' surveys via the spacecraft secondary computer. Several electromagnetic 'storms' have been monitored and remote surveys of the auroral ovals over the North and South Poles have proved most interesting. The 40 keV detector appears to have failed during launch, the thin 'window' probably shattered. The 20 keV detector is functioning normally.

11.11 Navigation Magnetometer

The navigation magnetometer is a relatively simple instrument intended to provide attitude information on *UOSAT* and for calibrating the position of the main magnetometer instrument after boom deployment. Two axes of the navigation magnetometer are functioning normally (y and z); however, the x axis appears to exhibit an offset causing the channel to 'bottom'. The data are nonetheless still adequate to provide the necessary attitude information. The two Sun sensors on the top and bottom of the spacecraft are functioning correctly and provide data to resolve the 180° ambiguity along the geomagnetic field vector.

11.12 CCD Imaging Experiment

Several test transmissions from the c.c.d. experiment involving test patterns, grey scales and preliminary images have shown the experiment electronics and data selectors to be functioning normally. However, it is not possible to assess the image quality easily until stabilization is complete. Several random images of deep space and Earth have been received from *UOSAT* but these have not been suitable to assess optical image quality.

11.13 H.F. Beacons Experiment

This experiment cannot be activated until the spacecraft has been de-spun and is gravity-gradient-stabilized, so that the h.f. antennas may be deployed.

11.14 Speech Synthesizer Experiment

The speech synthesizer is operated by the primary spacecraft computer and several successful tests have been carried out using the 145 MHz data beacon. The deviation has been kept deliberately low in order to minimize transmission bandwidth. Once *UOSAT* has been stabilized it is anticipated that the synthesizer will be used primarily on early afternoon passes for school demonstrations.

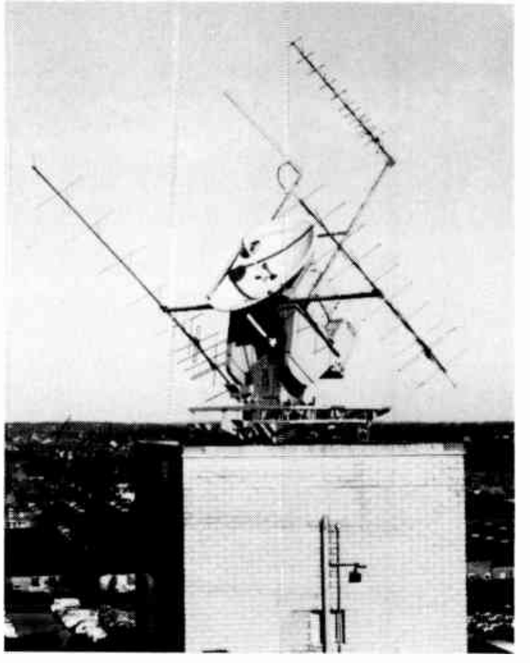


Fig. 9 The tracking antenna system above the Department of Electronic and Electrical Engineering, University of Surrey.

11.15 Microwave Beacons Experiment

The two microwave beacons at 2·4 and 10·4 GHz will not be activated until after *UOSAT* has been stabilized due to the high gain and directivity of the antennas.

12 Spacecraft Emergency

A spacecraft emergency occurred at 0320 G.m.t. on 4th April 1982, resulting in a temporary loss of spacecraft data. Both the v.h.f. and u.h.f. beacons were inadvertently activated simultaneously and this resulted in desensing of both command receivers due to insufficient isolation from the antenna feed hybrid. A 'watch-dog' program operating in the primary spacecraft computer was also inadvertently overwritten at the time. The problem has been thoroughly analysed using the engineering model spacecraft on the ground and shown to be resolvable by an increase in command transmitter radiated power.

13 Conclusions and Future Programs

The *UOSAT* Project has clearly demonstrated the feasibility and capability of a small, low-cost space programme within the UK, carried out between a University and British Industry and Research Organizations. Commencing in January 1979, the spacecraft was constructed, satisfactorily passed full functional and environmental tests and successfully launched by NASA two-and-a-half years later within a cash budget of £120,000 with additional facilities to the value of around £100,000. The *UOSAT* spacecraft has been operating correctly in orbit for over seven months and has returned large amounts of engineering and science data. The 40 keV particle detector appears to have been damaged during launch and some difficulty has been experienced using the primary computer to control the spacecraft day-to-day operations. One axis of the navigation magnetometer exhibits an offset and it was not possible before launch to achieve as great a degree of isolation from v.h.f./u.h.f. antenna hybrid as was desired. A problem associated with this latter constraint occurred in April 1982 temporarily halting data flow; however, the problem has been simulated on the ground and should be resolved shortly. All other spacecraft systems are performing normally.

Several lessons may be learnt from the Project, summarized as follows:

A small team can successfully generate and manage the resources necessary to design, build and operate a small spacecraft capable of worthwhile scientific and engineering contributions.

A project of this nature can be successfully completed within a tight budget of around £225,000 and within a very short time-scale of 2·5 years. It is only possible with a highly motivated, above average capability, multi-disciplinary team.

Geographic compactness of the primary team and in-house resources are essential.

Although several changes in approach would be taken for any future similar project, the basic approach is sound.

The post-launch operation and data collection from a low-orbiting spacecraft should not be under-estimated and requires similar resources to the design and construction phase.

A person dedicated to realistic documentation control, procurement and interface control is essential.

The importance of the *UOSAT* Project is in that it has demonstrated the potential for a continued national space programme, within a very reasonable budget, capable of significant science and engineering return. The relevance of low-cost spacecraft is directly related to the availability of inexpensive and useful launch opportunities. A number of sources exist and occasional opportunities do occur:

NASA secondary payloads—expendable and STS;
ESA *Ariane* secondary payloads; USAF;
Commercial launch agencies—Space Services Inc.;
OTRAG; India; Japan.

Providing cost-effective launches can be procured, the scientific and engineering communities now have a facility for carrying out relatively small but profitable scientific, technology or applications experiments within a realistic budget, as an alternative to large and costly programmes which favour exotic proposals and tend to preclude small science and industrial experiments.

14 Acknowledgments

It is impossible to acknowledge properly the very many individuals, groups, companies, research organizations and government departments that in some way or another supported the *UOSAT* Project and by their efforts ensured its success. It is equally impossible to isolate those who played an essential part as, in many instances, the provision of some small or relatively inexpensive part or service resolved a potential problem which would otherwise have halted the Project dead in its tracks and ensured that *UOSAT* would never have made the launch date! It was their personal determination in resolving the myriad of technical and logistical problems in the shortest possible time that made the Project a success on a short timescale.

It is appropriate to acknowledge with special appreciation the contribution of the primary sponsors of the Project as it was their 'faith' in the proposal to undertake a high risk project of this nature which, in hindsight, was well justified but, at the time, appeared well nigh impossible! I would similarly like to thank Dr. Frosch at NASA Headquarters for their support of the mission, Frank Lawrence of the NASA Delta Project Office for their advice, commitment and particular helpfulness, McDonnell Douglas and Gene Langenfeld at Western Test Range, Vandenberg for launch support.

I particularly thank Jan King (AMSAT-USA) for his very hard work, perseverence and deep commitment to ensuring the success of the Project in the face of great difficulties.

I would like to acknowledge and thank my colleagues at the University of Surrey, and within AMSAT for their faith, support, determination and endurance throughout a very taxing two years. None of this would have been at all possible if they had not given freely *many* hours of their own time.

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Attitude control and dynamics of *UOSAT* angular motion

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SUMMARY

The dynamics of *UOSAT* after launch are briefly reviewed and also the stages which are believed necessary to achieve a transition from an arbitrary state of nutation to one of damped gravity gradient and correct Earth-centred orientation. Novel algorithms have been developed as a form of adaptive commutation using the on-board magnetorquer.

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1 Introduction

An immediate objective in the post-launch phase of *UOSAT* is its orientation and stabilization. This work is likely to continue for some time, because of novel aspects of the *UOSAT* system which require original work, in particular the design of algorithms needed at various stages of the control, and the software to implement them. An important part of the *UOSAT* project is an attempt to realize an Earth-referenced, low-orbit satellite—one with a principal axis pointing towards the Earth centre—with a minimum of attitude sensors and the minimum of control hardware. Successful completion of the attitude sequence could justify the use of simpler systems compared to those currently used in Earth-referenced satellites, in some cases.

UOSAT carries some simple Sun sensors (which are non-directional and mounted on each end) and two 3-axis magnetometers for indirect sensing of its attitude via the geomagnetic field: a 'navigational' magnetometer attached rigidly to the main body, and a 'scientific' magnetometer mounted on a deployable boom. It carries no horizon sensors, calibrated Sun sensors, or any type of attitude-sensing gyroscopic system. For control of attitude *UOSAT* carries a simple magnetorquer consisting of two plane coils (see Fig. 1) aligned along its x-axis through which a current pulse can be applied under computer control, or directly through the up-link telemetry. The magnetorquer allows momentum change by interaction with the geomagnetic field. Torque components are created about the y axis and the z axis (but not the x axis) depending on the orientation of this field, and the current polarity.

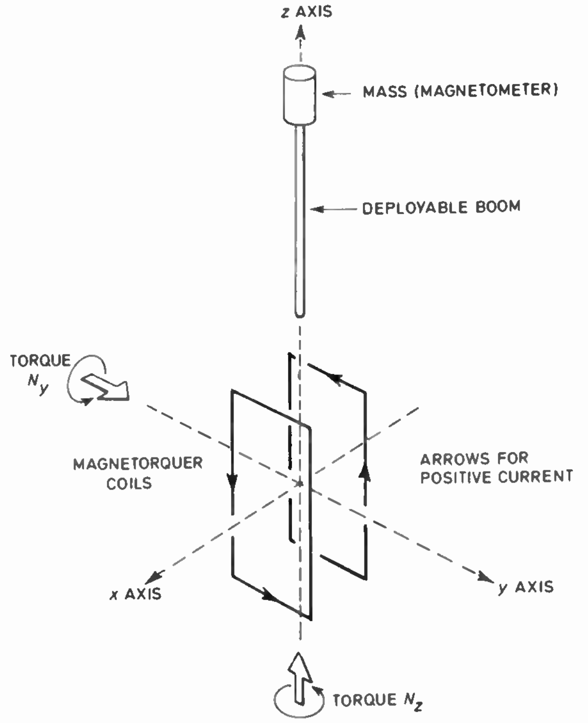

Fig. 1. Magnetorquer control and gravity gradient control from deployed boom

torque $N_y = -dH_z$, $N_z = dH_y$

where d is the magnetic dipole and H_x, H_y, and H_z are components of the geomagnetic field

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In addition, the spacecraft has an extendable boom carrying a mass (one of the magnetometers) on its extremity (see Fig. 1). The differential strengths of the gravitational field at the satellite's main body and at the end of the boom set up a restoring torque which tends to align *UOSAT* with its z-axis aligned on the centre of the Earth. The magnetorquer provides an active control; the gravity gradient on the boom provides a passive control.

There are few common procedures or systems employed in the control of the Earth-directed, low orbit satellites described in Ref. 1. However, there is one dominating feature in all the satellites described, and that is the spinning wheel—a momentum wheel which provides an essential control element. The angular velocity of this wheel is nominally directed along the orbit normal, at right angles to the orbit plane. The existence of this wheel confers a number of dynamic advantages not available to *UOSAT*. It is generally a vital element in direct control of the pitch of the satellite; it is also an important element in the control of both roll and yaw, in combination with other controlling devices such as gas jets, magnetorquer coils, or a gravity gradient boom, depending on the satellite. There is no space to summarize here the complex procedures involved. In any case they are irrelevant since *UOSAT* carries no momentum wheels of any type, or gas jets; nor does it carry an array of magnetorquers which would offer a choice of directions of the magnetic moment. There are no on-board passive damping systems.

The overall mission aim is to keep *UOSAT* permanently aligned on the Earth centre under the passive control of the extended boom, but to use the magnetorquer before and after deploying the boom as an active control.

2 Control Systems

2.1 Magnetorquer Control

The intention of using the magnetorquer is to modify the rotational state of *UOSAT* before deployment of the boom, and to control expected pitch and roll oscillations after this deployment. Details of these dynamical states is given in Section 3. It is appropriate here to describe one simple way using the magnetorquer; this is to apply a current pulse when $H_z \approx 0$. A torque then exists only along the z-axis. If $H_y \approx H$, the total magnetic field strength, then this torque would be a maximum. For a figure of $H = 20{,}000$ nT† the torque would be $0{\cdot}88 \times 10^{-3}$ N. This yields an angular acceleration or deceleration of $0{\cdot}5 \times 10^{-3}$ rad . s^{-2} assuming a moment of inertia about the z-axis of approximately 1·7 kg . m^2. This manoeuvre and several others are described in greater detail in Sections 4 and 5.

2.2 Gravity Gradient Control

The torque due to gravity gradient was briefly described in the Introduction. An orbiting body is weightless due to the cancellation of gravitational force by the centrifugal force, but it experiences a gravity gradient and a restoring torque if the body has different moments of inertia along its principal axes.

UOSAT has a moment of inertia I_x about the x axis and a moment of inertia I_y about the y axis which are nominally the same. Their common value will be referred to throughout as the transverse moment of inertia I_T. The value of I_T is about double that of the I_z, the moment of inertia about the z axis and becomes about two orders of magnitude greater than I_z with the boom fully deployed. The restoring torque T is given by the expression

$$T = -\tfrac{3}{2}\omega_0^2(I_T - I_z)\sin(2\theta) \qquad (1)$$

where $2\pi/\omega_0$ is the orbital period and θ is the inclination of the z axis away from the direction of the Earth's centre. The derivation of this formula is not simple, except when $I_z \ll I_T$ and the body can be approximated by two point masses at some fixed separation.

UOSAT was ejected from the launch vehicle with the boom retracted. Under command, although at the time of writing this has not yet been attempted, the boom will deploy along the z-axis pushing out the 2·2 kg magnetometer at its extremity to a maximum distance of 15 metres from the main body. The boom is a half-inch diameter beryllium copper tube which unwinds from a flat strip rolled-up on a drum. Prior to boom deployment the gravity gradient torque is insignificant because the transverse moment of inertia is not significantly greater than the z-axis moment of inertia (I_T is about 3·8 compared to I_z of about 1·7 kg . m^2), and this torque has little observable effect on the existing nutating spin states. But after boom deployment the increase in I_T to an estimated value of 425 kg . m^2 will be sufficient to control the satellite on the greatly increased gravity gradient torque. This control depends however on the spin rate about the z axis being substantially less than a certain critical value, otherwise this control will not be achieved (see eqn. (6) in Appendix 2).

3 *UOSAT* Dynamics

Before describing how the active and passive control systems may best be employed, the dynamics of *UOSAT* must first be explained. The equations of motion referred to the axes fixed in *UOSAT* are given in Appendix 1 together with a solution.

When ejected from the launch vehicle, *UOSAT* immediately commenced a state of nutation, which is a combination of two spins. *UOSAT* rotates at constant rate ω_z around its body z-axis, since $I_x = I_y = I_T$, and there is a transverse spin rate ω_T, also a constant, which slews this z-axis around in inertial space.

The net effect is that the z-axis describes a cone in space of an angle θ. Figures 2 illustrate the basic geometry and vector relationships of the different spins. The direction of the z-axis on average is the line of symmetry of the cone and defines an inertial direction which is called here the z_0-axis. The motion is in fact kinematically the same as the idealized frictionless spinning of a top in a gravity field, but the dynamics are different. In space this is a free body motion (torque free if one decides to ignore a small perturbation due to gravity gradient forces), whereas on earth such a motion is essentially supported by gravity.

† Over the equator at an altitude of 500 km.

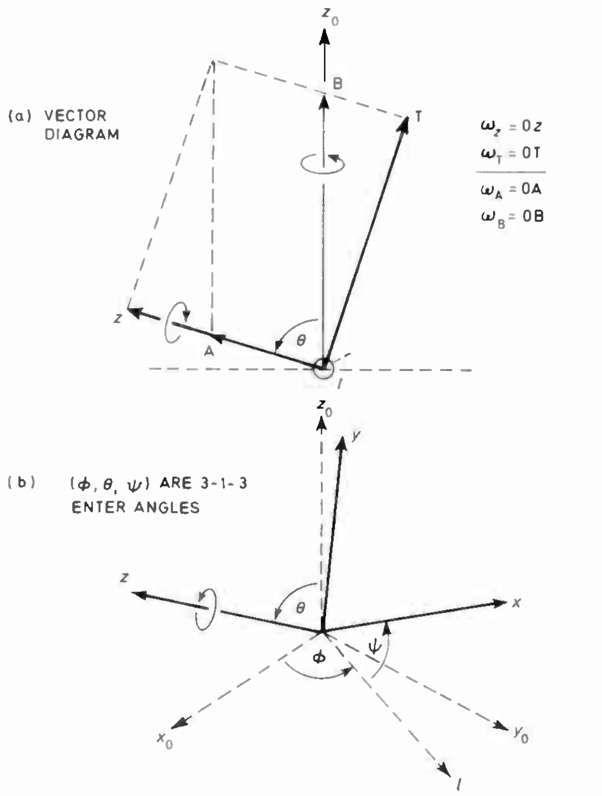

Fig. 2. Nutation geometry and vector relation of component spin velocities. (a) Vector diagram; (b) (ϕ, θ, ψ) are 3-1-3 Euler angles.

Figure 2(b) illustrates the attitude of *UOSAT* in an instant with respect to an assumed inertial set of axes; the z_0-axis has been identified with the direction of the angular momentum vector. The nutation angle θ exists therefore between this inertial z_0-axis and the body z-axis and is a constant. The other two angles shown are the remaining two Euler angles ϕ and ψ, which define the overall attitude. The angle ϕ is measured from the inertial reference axis x_0 to a line marked I in the Figure and then from this line by the angle ψ to the x body axis. The line I is defined by the intersection of two planes, one defined by the reference set x_0 and y_0, the other by the body axes x and y. Both angles increase at a constant rate. The angular velocities $\omega_A = \dot{\psi}$ and $\omega_B = \dot{\phi}$ constitute an alternative way of describing *UOSAT*'s nutation; these are the body nutation and the inertial nutation rates respectively. The relation between these quantities and the z-spin rate and the transverse spin rate are all shown in Appendix 1.

The parameters of the nutational state were not readily predictable before launch, but could be measured after launch from the magnetometer data. These data indicated that the inertial nutation period was about 25 s and the body nutation period about 50 s. Over the period October 1981 to February 1982 the former quantity has approximately doubled, due to dissipative losses.

After the boom is deployed the nutation state will be stopped and *UOSAT* will become Earth-referenced. It is required for the boom to be pointing away from the Earth, but the equation given for the gravity gradient torque shows the *UOSAT* could be passively controlled upside down equally well, with the boom pointing towards the Earth and the antennas and the camera pointing outwards. And irrespective of the average direction in which it could be pointing the satellite can oscillate both in pitch and roll with undamped oscillatory periods in the order of an orbital period (95 minutes). The defining equations are in Appendix 2. Adapted from Ref. 1, these equations show that in the case of *UOSAT*, with common transverse moments of inertia I_T the oscillations are effectively uncoupled. *UOSAT* may maintain a constant z-body spin, because the gravity gradient has no effect on this. The roll oscillation has a period $T/2$, where T is the orbital period while the pitch oscillation has period $T/\sqrt{3}$. It is to be noted that on average the roll is biased, as discussed later, so that even with the oscillation activity minimized, *UOSAT* would still have a residual roll angle. The equations assume small amplitude approximations.

4 Proposed Control Manoeuvres

Three distinct phases may be identified. In a first phase the boom is not yet deployed and the magnetorquer is used to impart the best possible parameters to the nutation state. In a second phase the boom is deployed from a best starting condition. If all goes well, *UOSAT* ends up with an ideal orientation on the centre of the Earth. In a third and continuing phase, errors accumulating in the first two phases will be reduced by again using the magnetorquer.

Phase 1. From the present state of arbitrary nutation it is proposed:

(i) To slow down the body z-spin rate to a period of about 10 minutes (the exact value is not important, but the slower this z-spin the less will be the residual roll after boom deployment).

(ii) To slow down the transverse spin rate to an optimum value of around 70 seconds (to be confirmed).

(iii) To re-orientate the angular momentum vector—the average z direction—from its present arbitrary direction to one which is aligned as near as possible along the orbit normal. This is technically a forced precession, in which the z_0 axis is slowly rotated, at a constant angle, around the current H vector.

All these manoeuvres can, in principle, be performed by correct interpretation of the magnetometer data and a form of adaptive commutation—correctly timed and signed pulses applied to the magnetorquer. The details are given in the next Section. *UOSAT* will still be nutating but with the best possible characteristics before commencing the next phase.

Phase 2. In this phase the boom is deployed starting from the most favourable orientation. After phase 1 *UOSAT* would be in a state of near 'flat spin' with a nutational angle approaching 90°, so that the z-axis itself is rotating almost in a flat plane. This state of motion may be visualized by placing a pencil flat on a table, then simultaneously both spinning it and rotating it. As the boom deploys the rotation rate slows down while the spin rate—about the z-axis—is unaffected. Numerical integration indicates that the boom rotates a further 970° before coming to a relative stop in space with its boom pointing outwards. Figure 3 gives a schematic of this manoeuvre. The boom will take about 30 minutes to deploy the full 15 metres in length. The desired initial attitude before boom deployment can be inferred from the H_z magnetometer data and from knowledge of the rotation of the H vector in the orbital plane at this point in the orbit. At the end of deployment, theory predicts that the orientation of *UOSAT* will be exactly right; it will be pointing the correct way towards the centre of the Earth, and will have no residual pitch or roll oscillation. This is, of course, the ideal. It is clear that errors accumulating in both phases (uncertainties about the true moments of inertia and the boom state of deployment) will result in a significant residual pitch and roll oscillation. The purpose of this control phase and the one preceding it can be said, therefore, to attempt to minimize these unwanted oscillations. It is important, of course, that errors do not accumulate to the extent that *UOSAT* lands 'upside-down'. Aerodynamic drag has been judged to be on the margin of observability and may make a small contribution to these errors.

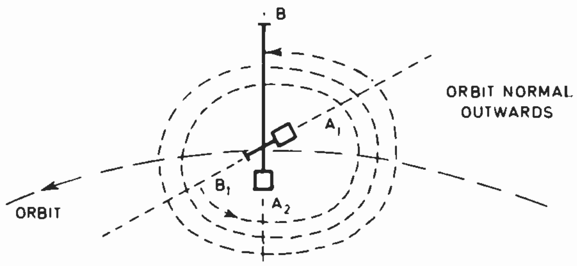

Fig. 3. Phased deployment of *UOSAT* boom. Line A_1B_1, start alignment of the z axis; line A_2B_2, finish alignment of the z axis. Rotation during deployment of approx. 970°.

Phase 3. Further active damping is possible, using the magnetorquer again to reduce the expected residual pitch and roll.[2] It is proposed to exploit the fact that, with the continuing z-body spin, it is possible to pulse the x-axis magnetorquer so as to interact with the geomagnetic field again to damp out actively these motions. Details of this procedure have not been fully worked out, mainly because it will probably be necessary to employ the other sensors, possibly the on-board camera. The difficulty in the final phase is that the craft is no longer spinning fairly rapidly with respect to the geomagnetic field, and there are consequent difficulties in using and interpreting the magnetometer data. This third phase will probably continue throughout the life of the satellite, since occasional corrections will always be necessary.

5 New Control Algorithm

This final Section describes in greater detail the proposed control algorithms for adjusting the attitude and spin states of *UOSAT* in its nutating state. These algorithms are applied in the first phase of the control manoeuvres. It is a truism of control theory that one cannot control what one has not first measured. In this context the only effective source of continual attitude data is the component geomagnetic field as measured by one of the two magnetometers. Some of the material presented in this Section appears to be new. It is assumed here that the spin states are relatively fast compared to the variations in the H field over an orbital pass.

5.1 Attitude Measurement

Section 3 described in some detail the state of nutation, defined by two spin states and an inertial angular momentum axis. The problem is how to measure these quantities from the component data H_x, H_y and H_z.

First, it may be remarked that the H_z data is relatively uncomplicated consisting of a simple sine wave variation (in the short term). In general, it is offset, having a non-zero average. It is easy to find the body z-spin on a zero crossing in H_z. It is readily shown that the value and sign of spin rate ω_z is obtained at that instant from the slope of H_x and the value of H_y, or the slope of H_y and the value of H_x, choosing whichever gives the greater accuracy (see Appendix 3). This spin rate is not harmonic and varies only slowly. There can be occasions—but not when *UOSAT* is aligned correctly before starting phase 2—when zero crossings do not happen and this information cannot be acquired.

The deduction of the transverse spin rate ω_T is more complicated. Essentially it can be updated only when the magnitude of H_z exceeds both the magnitude of H_x and the magnitude of H_y. Computer simulation shows that this occurs frequently but irregularly depending on the complicated nutational pattern. It should be noted that this information can be acquired with no knowledge of either the amplitude or direction of the H vector. The oscillations in H_z can be used every half cycle to provide further information. The period of this oscillation is neither the z-body rate nor the transverse spin rate but is the inertial nutation rate and can be used to check the nutation angle θ (see Sect. 3).

Of relevance to the immediate control problem is that the average H_d of H_z and its amplitude H_a can be updated every half cycle. Then a state variable can be obtained, which is a useful index of the satellite orientation. This normalized average field $h_d = H_d/H_a$ will be zero if the angular momentum vector is at right angles to the H vector.

It was stated that, as a rough approximation, the H vector lies in the plane of a polar orbit. Consequently by controlling the angular momentum vector, through a precessional manoeuvre to the smallest value of h_d, the best approximation to the desired satellite attitude prior to boom deployment will have been achieved. When the momentum vector is out of this required alignment, it can be shown that h_d will describe roughly a sine wave with two oscillations per orbit. It is necessary to observe the slope of h_d in order to infer in which part of the oscillatory orbit cycle *UOSAT* lies, and the required sign to the precession.

5.2 Change to z-spin

It was briefly mentioned in Section 2.1 that one simple procedure is to fire a pulse on a zero crossing in H_z, so minimizing an unwanted torque N_y. The pulse polarity clearly should be chosen depending on the requirement to speed up the spin or slow it down, the existing sign of this spin, and the sign of the H_y field at that moment. It can be shown that this reduces to the simple decision rule.

Set magnetorquer sign

$$S = S_0 \times \mathrm{sgn}\,(\dot{H}_x)\Big]_{H_z = 0}$$

where

$$S_0 = +1 \text{ to speed up}$$
$$S_0 = -1 \text{ to slow down.}$$

Computer simulation shows that the pulse duty factor may have a significant value (say up to 20%). Then within the duration of each pulse the H_z becomes non-zero. But on average the unwanted torque in the Y axis has no net effect. A more sophisticated mode is available which allows almost continual firing of the magnetorquer, although interrupts must be allowed in which to observe the magnetometer and update the state vector, with computed additional interrupts whenever the effect of the unwanted N_y torque exceed bounds. Computer simulation has validated this approach. Taking the figure already given for the peak acceleration (or deceleration), then in this adaptive mode the average acceleration available is in the region of half of this value (actual value depends on the inclination of the H field), i.e. $0{\cdot}25 \times 10^{-3}$ rad s^{-2}.

5.3 Change to Transverse Spin

In a similar way to the procedures described in Section 5.2, one method is to fire the magnetorquer pulse on a zero crossing in H_y, so minimizing the unwanted torque N_z. The pulse polarity should be chosen depending on the requirement to speed up or slow down, the present sign of the ω_y spin, and the sign of the H_z field at that moment. It can be shown that this reduces to a simple decision rule. Set magnetorquer sign

$$S = S_0 \times \mathrm{sgn}\,(\dot{H}_x)\Big]_{H_y = 0}$$

As in the case of changing the z spin a more sophisticated mode has been designed which allows almost continual firing of the magnetorquer subject to an analogous set of limitations. A calculation allowing for the greater moment of inertia about the transverse axis suggests a peak acceleration (or deceleration) as about $0{\cdot}2 \times 10^{-3}$ rad . s^{-2}. In the adaptive mode the average acceleration is in the region of half this value, i.e. $0{\cdot}1 \times 10^{-3}$ rad . s^{-2}.

5.4 Precession of *UOSAT*

The dynamics of an imposed precession on a torque-free nutation are particularly complicated to describe, and only an outline will be given here; experimental experience is necessary to test out and evaluate various possible modifications. The general aim is fairly easily explained. The word precession is reserved in spacecraft manoeuvres for deliberate changes to the direction of the angular momentum vector. This requires an external torque. Precession forces the angular momentum vector to slew around the direction of the H vector, but always keeping a constant angle to this H vector. If S_p is the desired sign of this precession, continual pulsing of the magnetorquer—always allowing gaps to look at the magnetometer—will cause precession according to this rule. Set magnetorquer sign

$$S = S_p \times \mathrm{sgn}\,(\dot{H}_y) \times \mathrm{sgn}\,(H_d - H_z)$$

or better

$$S = S_p \times \mathrm{sgn}\,(\dot{H}_y + \omega_z H_x) \times \mathrm{sgn}\,(H_d - H_z)$$

This has been extensively tested in simulation and shown to be consistently effective. The torques which are clearly created along the y and z axis can be shown on average to cancel out, so inducing no change to either the z-body spin or the transverse spin. To encourage this averaged cancellation one may impose occasional interrupts whenever the effect of the unwanted torques quasi-randomly exceeds certain bounds.

The effect of precession may be measured in terms of a slew angular velocity of the angular momentum axis about the H vector. This is simply given (maximum value)

$$\Omega = \frac{N}{\omega_B I_T} = \frac{0{\cdot}88 \times 10^{-3}}{\omega_B \times 3{\cdot}7} \text{ rad . s}^{-1}$$

where N is the maximum torque available (on the equator), and ω_B and I_T have been defined, and assuming $H = 20{,}000$ nT.

As was discussed earlier in this Section, it is desired to slew the angular momentum vector into a direction at right angles to the orbital plane. To a rough approximation, this is also at right angles to the H vector. It can be shown that the required information for the precessional sign is given by the simple formula

$$S_p = \mathrm{sgn}\,(\omega_z)\,\mathrm{sgn}\,(\dot{h}_d)$$

where $\dot{h}_d$ is the rate of change of the normalized averaged H_z field (see Sect. 5.1). Substitution of typical figures shows that a worst case attitude change—requiring a full 180°—could take around 30 minutes, for the inertial momentum period $2\pi/\omega_B$ in the order of one minute.

6 References

1 Wertz, J. R., 'Spacecraft Attitude Determination and Control' (D. Reidel, Dordrecht, Netherlands, 1978).

2 Svitek, T., 'An analysis of the attitude control by the gravity gradient technique for the UK amateur experimental scientific spacecraft project,' Liska Space Group, Prague (private communication).

7 Appendix 1: Angular velocity and nutational rates

The differential equations defining nutation are as follows

$$I_T \dot{\omega}_x = \omega_y \omega_z (I_T - I_z) \tag{1a}$$
$$I_T \dot{\omega}_y = -\omega_x \omega_z (I_T - I_z) \tag{1b}$$
$$\dot{\omega}_z = 0 \tag{1c}$$

The solution to this set of equations in the Eulerian 3-1-3 system of angles

$$\omega_x = \omega_T \sin(At) \quad = \dot{\phi}\sin(\theta)\sin(\psi) \tag{2a}$$
$$\omega_y = \omega_T \cos(At) \quad = \dot{\phi}\sin(\theta)\cos(\psi) \tag{2b}$$
$$\omega_z = \omega_A + \omega_B\cos(\theta) = \dot{\psi} + \dot{\phi}\cos(\theta) \tag{2c}$$

where ω_x, ω_y, ω_z are angular velocities about the x, y and z axes respectively and

ω_T = transverse angular velocity
$\dot{\psi} = \omega_A$ = body nutational rate
$\dot{\phi} = \omega_B$ = inertial nutational rate

Given ω_B, θ then

$$\omega_A = \left(\frac{I_T}{I_z} - 1\right)\omega_B\cos(\theta) \tag{3a}$$

$$\omega_B = \left(\frac{I_T}{I_z}\right)\omega_B\cos(\theta) \tag{3b}$$

$$\omega_T = \omega_B\sin(\theta) \tag{3c}$$

8 Appendix 2: Gravity gradient torque

Symbols

I_T = transverse moment of inertia
I_z = z axis moment of inertia
ξ_r, ξ_p, ξ_y = roll, pitch and yaw angles
ω_0 = orbital rate
ω_z = body spin rate

The differential equations defining the motion under control of gravity gradient are as follows:

In roll
$$I_T\ddot{\xi}_r + \omega_0^2(4I_T - 3I_z)\xi_r = \omega_0\omega_z I_z \tag{4a}$$
In pitch
$$I_T\ddot{\xi}_p + 3\omega_0^2(I_T - I_z)\xi_p = 0 \tag{4b}$$
In yaw
$$\dot{\xi}_y + \omega_0\xi_r = \omega_z \tag{4c}$$

These are small angle approximations. There are undamped oscillations in roll and pitch, with a constant velocity in yaw (constant spin about *UOSAT* z axis).

From equation (4) it is seen that there is an average roll angle

$$\bar{\xi}_r = \frac{\left(\dfrac{\omega_z}{\omega_0}\right)}{\left(\dfrac{4I_T}{I_z} - 3\right)} \tag{5a}$$

and it can be shown that the exact derivation corresponding to this (not a small angle approximation) is

$$\sin(\bar{\xi}_r) = \frac{\left(\dfrac{\omega_z}{\omega_0}\right)}{\left(\dfrac{4I_T}{I_z} - 3\right)} \tag{5b}$$

From equation (5) it is clear that there is a critical z spin rate above which capture by gravity gradient is not possible, given by

$$\omega_{z(\mathrm{crit})} = \omega_0\left(\frac{4I_T}{I_z} - 3\right) \tag{6}$$

9 Appendix 3: Spin rate equations

In the nutational state there are some simple parameters which may be deduced from the magnetometer data. These are approximations which assume that the spin rates are relatively fast compared to the variation in the geomagnetic field over the orbit.

The body spin rate may be derived from

$$\omega_z = \left.\frac{\dot{H}_x}{H_y}\right|_{H_x=0} \qquad \omega_z = -\left.\frac{\dot{H}_y}{H_x}\right|_{H_y=0} \tag{7}$$

and the transverse spin rate from

$$\omega_T = ((\dot{H}_y + \omega_z H_x)^2 + (-\dot{H}_x + \omega_z H_y)^2)^{\frac{1}{2}}/|H_z| \tag{8}$$

when

$$|H_z| > |H_x| \quad \text{and} \quad |H_z| > |H_y|$$

simultaneously.

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The primary *UOSAT* spacecraft computer

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SUMMARY

Two computer systems have been incorporated into the *UOSAT* spacecraft, the primary spacecraft computer being based on the RCA CDP 1802 microprocessor, and the secondary computer on the Ferranti F100 system.

This paper describes the system philosophy and design consideration of the primary computer, particularly as regards its operation in a space environment where low power and reliability are essential. An overview of the general system is given and the various interfaces to the other spacecraft systems and experiments are also described.

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1 Introduction

This paper describes the requirements and the general design of the *UOSAT* primary spacecraft computer. There are two independent computer systems on board the spacecraft. The primary computer system is based on the RCA CDP 1802 microprocessor linked to a 16 Kbyte error correcting memory. It also has access to a 32 Kbyte memory contained within the c.c.d. camera system experiment. It has a comprehensive set of interfaces to the various experiment packages and, with the ability to issue any spacecraft command, is capable of complete operational control of the spacecraft. It is possible, under certain well-controlled conditions, to pass information between the primary computer system and the secondary computer which is based on a 16-bit Ferranti F100 microprocessor.[1]

2 System Requirements

The spacecraft has on board 45 status points and 60 parameters such as temperature, current, etc. which must be monitored; in addition there are four experiments which generate data. There are at least two desirable modes of operation whereby data from each of these are sampled and transmitted over the telemetry link: first, it should be possible to sample each data point sequentially and transmit the information in some fixed format and secondly, it should be possible to sample continuously any single data point and exclude all others. Furthermore, it is desirable that the experiments can be switched on and off and the data acquired, stored, and read out when the satellite is out of reach of ground control.

It thus became clear at an early stage in the outline design of the spacecraft that the presence of at least one computer on board would be essential to the successful operation of the spacecraft systems. Since very little unclassified information was available on the reliability of processor systems in a space environment, however, it was decided that the computer should not occupy such a dominant position within the overall system that its failure would render most of the on-board experiments useless.

This led to the general philosophy that the computer system should have access to data produced by the various experimental packages, but that interfaces between the experiments and computer should be totally independent of the data routes from the experiments to the telemetry system. It was also decided to have a completely independent comprehensive telemetry system, based on hard-wired logic with access to most of the major experiments; this telemetry system would normally regularly sample the experimental data and other satellite data points and transmit this information quite independently of the computer system. This philosophy has created great flexibility within the satellite system since the computer and telemetry systems can, to a large extent, act as back-ups to each other.

The over-riding design consideration of the computer system, however, was that the power consumption should be low since, in order to meet requirements already outlined, the processor must be continuously powered. This meant that a c.m.o.s. processor system

0033–7722/82/080385+06 $1.50/0

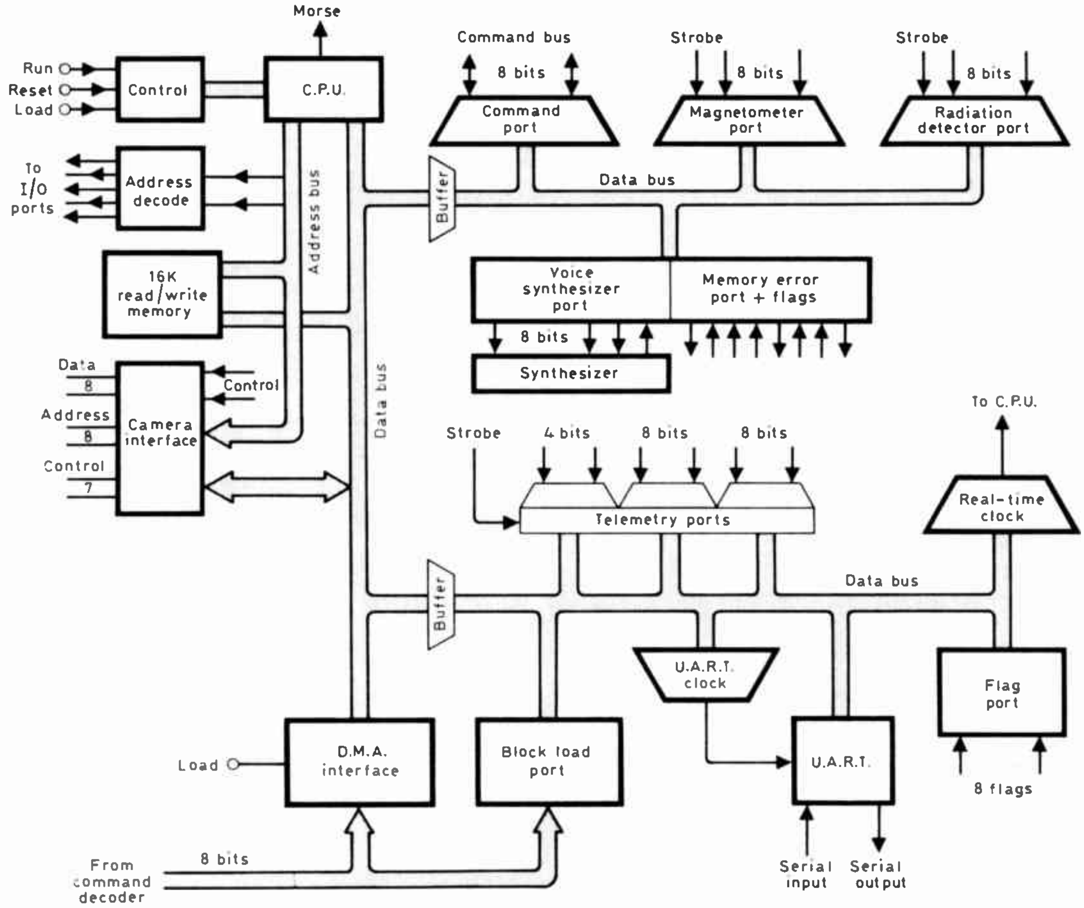

Fig. 1. General block diagram of the main spacecraft computer

was the only practicable choice and, in the interests of standardization with past and future AMSAT projects, the RCA CDP 1802 microprocessor was chosen. This microprocessor is supported by a wide range of peripheral integrated circuits and is available in military and extended temperature range versions.

Since reliable high voltage c.m.o.s. p.r.o.m.s were not available within the time scale of the project, a resident bootstrap/monitor program could not be provided; thus the computer system was designed so that such a program could be loaded from the ground station.

In order to reduce the transmission time and on-board storage necessary for utility programs, it was decided at an early stage to use the high level language IPS as the main programming language. IPS has been developed by Meinzer[2] for the AMSAT satellites and its use on the *UOSAT* will afford the first opportunity of evaluating its suitability for such applications. To enable programs written and transmitted in IPS to be run by the computer, an 8 K resident interpreter is first loaded into the memory in error corrected sections, using the previously loaded bootstrap program.

The usefulness of the computer is much increased if it has the capability of taking action as a result of monitoring the data from a particular experiment. In order to take worthwhile action, however, the computer system must be capable of issuing commands to the spacecraft. This brings problems owing to the possibility of the computer's failing and perhaps issuing a false command permanently, or of its issuing a command and blanking out a simultaneous ground command. Provision was made in the system design stage to protect against such eventualities.

The spacecraft c.c.d. camera experiment has associated with it a 32 Kbyte (8-bit bytes) frame store. It was decided that the computer should have access to this memory for three main reasons. First, the expected power consumption of the computer memory would allow continuous operation of only 16 Kbytes of memory; therefore, the ability to use a further 32 Kbytes when necessary would add major flexibility to the computer system. Secondly, the processor could be used to process the data in the camera memory (for instance, to enhance contrast, or eliminate blemishes in the c.c.d. camera image) prior to transmission to the ground. Thirdly, the computer may be used to put information in the c.c.d. camera memory for transmission on the independent c.c.d. system downlink.

In order that the computer system can act as a scheduler so that experiments may be turned on and off at fixed times, the system has access to a real-time clock.

To summarize, the requirements for the computer are:

Low power
Monitorless loading of programs from ground
Individual interfaces to the various experiments
Interface to c.c.d. camera system memory
Real-time clock.

3 The Primary Spacecraft Computer System

As has been indicated, the system is designed around an RCA CDP 1802 c.m.o.s. microprocessor. This c.p.u. is an 8-bit machine with an unusual architecture having 16 general-purpose 16-bit registers, which may be used as multiple program counters, data pointers, or data registers. It has a single accumulator, a single-bit on-chip output port and four single-bit input ports which may be directly tested by branch instructions. Since the c.p.u. circuitry is static, the clock may be stopped without loss of information from the internal registers. The 1802 also has a very useful on-chip direct memory access (d.m.a.) control that enables programs to be loaded by applying the program data to the data bus, with the c.p.u. supplying consecutive memory addresses and the relevant write control signal. This function forms the basis of the method of loading programs from the ground into the computer memory.

The c.p.u. has an 8-bit data bus, and a 16-bit address multiplexed on to an 8-bit bus, with the high order 8-bits presented first. With its associated peripherals the c.p.u. is capable of operating with power supply voltages ranging from 4 to 10·5 V. This tolerance to power supply voltage variations makes the 1802 very attractive for satellite applications. The master clock frequency may lie between d.c. and 5 MHz but, because there were no particular high speed requirements, to ease timing problems and to reduce power consumption, a clock frequency of 1·228 MHz was chosen from which the baud clock rate is derived. At this clock frequency the memory access time is approximately 3 μs, and the power consumption of the complete system—excluding memory—is less than 100 mW.

One of the major problems encountered was the interfacing of c.m.o.s. devices in a powered-up system and those in an unpowered system. Unpowered c.m.o.s. circuity may present a low impedance at its input or output and it is essential that current limiting resistors are placed in series with inputs and outputs where a powered-to-unpowered connection occurs.

An example of this was encountered during compatibility tests between the spacecraft computer systems and the telemetry system; it was found that when the computer was unpowered the current drawn from the telemetry system on the 20 line computer/telemetry bus roughly tripled the power consumption of the total telemetry system and caused the current foldback protection of its power supply to operate.

The block diagram of the system is shown in Fig. 1. The design is fairly conventional with a common 8-bit bidirectional data bus connected to all peripheral devices. In order to prevent a failure in a peripheral device causing a total system failure, two sets of data bus driver/isolators are used, effectively isolating the read/write memory (r.a.m.), the input/output ports, and the memory mapped devices from each other. Thus for example a failure on the magnetometer port will not prevent the telemetry port from being read, or cause a program memory failure.

The system can be divided into three main blocks; the c.p.u. and control, the r.a.m., and the input/output systems. A description of each follows.

3.1 CPU and Control

There are three main control lines from the command decoder to the computer system. These are the LOAD, RESET, and RUN lines and they are latched in the c.p.u. An active signal on the RESET line will cause a master reset of the whole system including all the peripheral devices. This command enables control of the computer to be established in the event of a software crash. The RUN command causes the computer to perform a momentary reset and then to execute the program from memory location zero. The LOAD command causes the system to enter a self-loading d.m.a. mode which is a useful feature of the 1802 microprocessor and works as follows. Information in the form of 8-bit bytes, together with a strobe pulse is presented on the data bus from the command decoder. With LOAD activated these data are strobed directly into the processor r.a.m. in consecutive memory locations beginning at location zero. The processor provides automatic incrementing of the memory address lines in this mode, which always loads the program from address zero, and is intended for loading low-level monitor or bootstrap programs into the system.

The c.p.u. has four input ports; one is used for a maskable 50 Hz real-time clock and another for a 25 Hz half-frequency version. The other two ports are used in conjunction with programmable u.a.r.t. (universal asynchronous receiver transmitter) to detect valid serial characters quickly. The c.p.u. has a direct single-bit output port which is intended to be used to tone modulate the satellite beacons in morse code, etc.

3.2 Input/Output

The 1802 microprocessor can handle up to seven input and output ports with the standard input/output instructions, but since this is inadequate for the number of interfaces required, some peripheral devices are memory-mapped, i.e. they are made to look like a memory location to the processor.

Input ports 1, 2 and 3 are used to interface the microprocessor to the telemetry system. Three 8-bit ports are used to connect to a 20-bit parallel bus on the telemetry system on which are simultaneously presented the telemetry channel number (0–79) and a BCD-coded 3-digit value (0–999). These data are presented every 60 ms together with a strobe and are latched into the input ports unless the previous latched data have not all been read, in which case the new data are rejected. This prevents any mixing of old and new data and is necessary since there is no synchronization between the telemetry clocks and the computer.

Input port 4 is used as a facility to read program data from the ground under program control when not in LOAD mode. The input data to this port and to the d.m.a. port is parallel. The port is known as the block load port.

Input port 5 is used as an 8-bit flag port. Various signals from other input ports indicating the availability of data are connected to this port so that, by means of an

Table 1

Digitalker vocabulary†

Word	8-Bit Binary Address SW8 SW1		8-Bit Binary Address SW8 SW1		8-Bit Binary Address SW8 SW1
THIS IS DIGITALKER	00000000	Q	00110000	IS	01100000
ONE	00000001	R	00110001	IT	01100001
TWO	00000010	S	00110010	KILO	01100010
THREE	00000011	T	00110011	LEFT	01100011
FOUR	00000100	U	00110100	LESS	01100100
FIVE	00000101	V	00110101	LESSER	01100101
SIX	00000110	W	00110110	LIMIT	01100110
SEVEN	00000111	X	00110111	LOW	01100111
EIGHT	00001000	Y	00111000	LOWER	01101000
NINE	00001001	Z	00111001	MARK	01101001
TEN	00001010	AGAIN	00111010	METER	01101010
ELEVEN	00001011	AMPERE	00111011	MILE	01101011
TWELVE	00001100	AND	00111100	MILLI	01101100
THIRTEEN	00001101	AT	00111101	MINUS	01101101
FOURTEEN	00001110	CANCEL	00111110	MINUTE	01101110
FIFTEEN	00001111	CASE	00111111	NEAR	01101111
SIXTEEN	00010000	CENT	01000000	NUMBER	01110000
SEVENTEEN	00010001	400HERTZ TONE	01000001	OF	01110001
EIGHTEEN	00010010	80HERTZ TONE	01000010	OFF	01110010
NINETEEN	00010011	20MS SILENCE	01000011	ON	01110011
TWENTY	00010100	40MS SILENCE	01000100	OUT	01110100
THIRTY	00010101	80MS SILENCE	01000101	OVER	01110101
FORTY	00010110	160MS SILENCE	01000110	PARENTHESIS	01110110
FIFTY	00010111	320MS SILENCE	01000111	PERCENT	01110111
SIXTY	00011000	CENTI	01001000	PLEASE	01111000
SEVENTY	00011001	CHECK	01001001	PLUS	01111001
EIGHTY	00011010	COMMA	01001010	POINT	01111010
NINETY	00011011	CONTROL	01001011	POUND	01111011
HUNDRED	00011100	DANGER	01001100	PULSES	01111100
THOUSAND	00011101	DEGREE	01001101	RATE	01111101
MILLION	00011110	DOLLAR	01001110	RE	01111110
ZERO	00011111	DOWN	01001111	READY	01111111
A	00100000	EQUAL	01010000	RIGHT	10000000
B	00100001	ERROR	01010001	SS (Note 1)	10000001
C	00100010	FEET	01010010	SECOND	10000010
D	00100011	FLOW	01010011	SET	10000011
E	00100100	FUEL	01010100	SPACE	10000100
F	00100101	GALLON	01010101	SPEED	10000101
G	00100110	GO	01010110	STAR	10000110
H	00100111	GRAM	01010111	START	10000111
I	00101000	GREAT	01011000	STOP	10001000
J	00101001	GREATER	01011001	THAN	10001001
K	00101010	HAVE	01011010	THE	10001010
L	00101011	HIGH	01011011	TIME	10001011
M	00101100	HIGHER	01011100	TRY	10001100
N	00101101	HOUR	01011101	UP	10001101
O	00101110	IN	01011110	VOLT	10001110
P	00101111	INCHES	01011111	WEIGHT	10001111

† 'Digitalker' is a trademark of National Semiconductor Corporation
Note 1. 'SS' makes any singular word plural.

instruction to read port 5, the processor may establish which of the other input/output ports require servicing.

Output port 4 is used as a programmable divider for the real-time clock generator. The divider ratios may be varied from 1 to 256 and a value of 16 gives a real-time clock rate of 50 Hz.

Output port 5 is also used as a programmable divider. This divider is part of a chain producing the baud rate clock for the computer system's serial input/output device, a programmable u.a.r.t. It has a range from 1 to 256 enabling the computer serial data link to work at baud rates between 1200 and 50 (divide ratio of 24). A divide ratio is available which will give a close approximation to the USA standard baud rate of 45·54 baud.

Input/output ports 6 and 7 are utilized for the computer system u.a.r.t. This enables the computer to communicate via the beacons in any of the standard serial asynchronous formats. Normally the system is set up to produce 1200 baud data with one start bit, eight data bits, no parity and one stop bit, but Baudot code may be programmed.

The serial input side of the u.a.r.t. is connected to monitor the up-link serial data reaching the command decoder thus enabling the computer system to monitor ground link performance. The serial output half of the u.a.r.t. is connected not only to the beacon source selector switch, but via some complex routing, to the command link serial input. This enables the computer under certain circumstances to issue serial commands to the spacecraft as though originating from the command receivers.

As well as using the standard input/output ports it was found necessary to memory map certain experimental interfaces. The memory space 7FF0H to 7FFFH was allocated to a number of input/output ports.

The command port was allocated the address 7FF0H. It consists of an 8-bit parallel bidirectional interface connected to the command bus using bidirectional tri-state drivers. The system is arranged so that if a command from the ground appears on the command bus while the processor is outputting to the bus, then there is a hardware over-ride to tri-state the command port output.

The processor is able to monitor the data on the command bus at any time by reading address location 7FF0H. To guard against failure of the tri-state outputs pulling the command bus lines to permanent levels, the command port is connected to the bus via resistors so that even in the event of a port short circuit, the command decoder can still drive signals along the command bus.

The command bus is actually only seven bits wide; the eighth bit of the command port is used to tri-state the command decoder output when the processor system issues a command, which may include the computer OFF command. All computer system-originated commands must be issued with a logic 1 in the most significant bit position in order to be accepted as valid commands.

The magnetometer experiment is interfaced to the computer by an 8-bit paralleled port assigned the address 7FF2H. An active high strobe from the magnetometer occurring every 20 ms latches data into the port which sets a flag bit on the flag port of the system. The data format is a frame of 8-bit bytes in a multiframe of four frames.[3]

The radiation detector interface is identical to that of the magnetometer, i.e. an 8-bit parallel bus with strobe. The format of the data is described elsewhere,[4] but the maximum data rate is one byte every 20 ms. The port address is 7FF3H.

The voice synthesizer experiment is interfaced to the computer system with an 8-bit output port and two handshake lines, which correspond to the command SPEAK from the computer system and the response FINISHED TALKING from the voice synthesizer. The voice synthesizer is based on the standard National Semiconductor 'Digitalker'. It has a preprogrammed vocabulary of 144 words which are selected with an 8-bit address and a signal on the SPEAK control line. When the synthesizer is ready to accept a word command, its FINISHED TALKING line is reset. The available vocabulary is technical along with the alphabet and other common words, and includes the digits required to reproduce any number. Table 1 shows the vocabulary available.

The c.c.d. camera interface enables the computer system to have full access to the 32 Kbyte memory resident in the camera experiment. The interface consists of the eight multiplexed address lines, the 8-bit data bus, the read and write control lines, two synchronizing clock signals, and the central processor master clock signal. Because of the need for timing accuracy, these signals are not isolated with independent bus drivers; instead analogue switches have been used in the data and address lines with their operation being under control of a ground command. To the computer system the c.c.d. system memory appears to be a 32 Kbyte r.a.m. occupying memory space 8000H to FFFFH. The precise mapping of memory locations to pixel locations on the received c.c.d. experiment television display is not straightforward, and will not be described here. The complication arises because, to the camera, the memory is configured as two parallel 16 K × 16-bit banks where each 16-bit word contains four picture elements. Further details of the c.c.d. experiment can be found in Ref. 5.

3.3 Computer System Memory

The initial design of the computer system assumed the use of c.m.o.s. r.a.m.s for low power consumption. However, during the course of the project, information was received from NASA and AMSAT that extended testing under space conditions had indicated that the then current c.m.o.s. memories had catastrophic failure modes leading to power supply short circuits. We were strongly recommended to use well-proved m.o.s. dynamic r.a.m.s. Since the chip area per bit is very small on these r.a.m.s, however, they are more sensitive to the effects of charged particles which produce errors in the stored bits. These errors are known as soft errors, since writing the correct bit value back to the location restores the original value. Hence, when using these r.a.m.s it is advantageous to use a memory design that is error correcting, especially in a space environment.

The memory design used is due to Meinzer.† Basically, each 8-bit byte is stored together with four extra bits produced by a Hamming coding system giving a 12-bit memory word. The system is capable on read-out of correcting one error in each 12-bit word. The probability of two errors in each 12-bit word is very small, especially as the estimated incidence of single bit errors is only four per day in the orbit of *UOSAT*. The effectiveness of the system was demonstrated in testing: for some time one of the memory bits failed because of a p.c.b. track fault but the memory consistently corrected for this missing bit. The memory has a 3-bit counter which counts the number of errors corrected. This count is constantly telemetered to ground on the status indicators of the main telemetry system. So far, with the small programs running on the computer system in the two months since launch (up to 65 bytes long) no memory errors have been detected or corrected. The computer system can also read the counter value via some spare bits on an input port.

† Private communication.

4 Implementation

The complete computer system was constructed on four standard-sized printed wiring fibreglass boards occupying two spacecraft standard boxes: the processor and input/output ports occupied two of the boards, the 16 Kbyte memory one board, and the c.c.d. camera interface the final board. Each board was double-sided and employed plated-through holes. All integrated circuits were soldered to the boards, since the use of i.c. holders would pose a significant reliability hazard. Components such as pot-cores and crystals were tied down to reduce vibration fatigue problems. For strain relief, wires to connectors incorporated a loop to allow flexure, and certain interboard wiring was potted at the wire-board junction to prevent movement. Altogether seven external 25-way connectors were required to the computer system.

The integrated circuits used were space-qualified versions where possible, but since these were difficult to obtain, most components were military grade high-reliability devices in ceramic packages. In order to reduce the effects of e.m. interference, a supply voltage of 10 V was chosen for the computer system even though the computer and memory peripherals work down to a supply voltage of 4 V.

It should be borne in mind that the design of this computer system began only in November 1980, with the satellite delivery taking place in August 1981. Consequently, features such as central-processing unit redundancy could not be incorporated to improve reliability within the total time-scale.

5 System Comparison

To summarize the primary computer, the c.p.u. is an 8-bit machine fabricated in c.m.o.s. technology using n.m.o.s. dynamic r.a.m.s. It was designed to be launched without a resident monitor program, and relies on a hardware program loading mechanism to load a bootstrap initially, which can then be used to load further programs.

It is intended to be used with a real-time operating system which has the capability of running foreground and background tasks. Utility programs are intended to be written in the high-level language IPS and run on the computer's 8 Kbyte interpreter which is loaded from the ground. The system has interfaces to the various experiments, to the command system, to the telemetry system and to the various beacons. All interfaces are 8 bits wide, except the telemetry interface which is 20 bits wide.

On the other hand the secondary computer c.p.u.,[1] fabricated in bipolar c.d.i. technology, has a 16-bit word length, and has a much faster instruction cycle time with correspondingly higher power consumption. At variance with NASA's recommendations for satellite memory systems it uses 64 K hybrid construction c.m.o.s. static r.a.m.s. with no error protection. Furthermore, in contrast to the primary computer it has a resident monitor program stored in c.m.o.s. p.r.o.m.s and only two bidirectional links to the spacecraft; both links are serial.

The existence of two computer systems on board is of considerable interest, enabling comparisons of the reliability, the ease of software development and the versatility of the two systems to be made when operating in a space environment with low data rate ground-station links which are prone to transmission-induced errors. It will be possible to gain valuable information on the use of the computers in real-time satellite control and monitoring using distributed processing techniques and multi-tasking operations. Experience should also be gained in the advantages of using a processor to perform enhancement and data reduction of the images from the c.c.d. camera experiment. Furthermore, an investigation of the most appropriate error correction procedure to apply to the bit-reduced image data to be transmitted to the ground-station could also be carried out.

Experiments on the optimum type of operating systems for the two dissimilar computer types is expected to be undertaken which should be of value to future satellite programs.

6 Post-launch Experience

Since launch the primary computer system has worked consistently well. Several machine code routines have been loaded from the ground to exercise various experiments. The computer has shown its ability to issue commands on board the spacecraft and has programmed the speech synthesizer to broadcast its vocabulary on a beacon for several orbits. With such a flexible system, the uses to which it may be put are legion; the restrictions are those of all microprocessing applications—the time taken to produce operational software.

7 Acknowledgments

We would like to acknowledge the work of Dr David Hicks who was responsible for the initial design of the voice synthesizer board, and Jerzy Slowikowski and Ian Ferebee who did much of the final assembly work. Thanks are also due to Mike Blewett who dealt with the problems of the memory board regulators. We would like to further acknowledge the debt owed to Chris Bard, Ron Deane and Peter Boreland who were final-year students at the time and who spent much time and effort building and testing the original system. Finally, we would like to thank Steven Greenland and Mark Pitts, MoD apprentices who performed much of the testing of the later versions, Joop den Herder (CERN) for design of p.c.b.s, and Christine Sweeting for breadboard and design checks.

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A low-power 16-bit computer for space applications on *UOSAT*

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SUMMARY

The *UOSAT* secondary computer provides a system for the evaluation of higher risk computing technologies, including a 16-bit bipolar microprocessor and large static c.m.o.s. memories. The system has both simple interfaces to the satellite and a simple loading command structure. A significant feature of the design is the provision for low-power operation in which the processor consumes a high supply current only when computing tasks are being undertaken.

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1 Introduction

The *UOSAT* satellite has been designed to provide, *inter alia*, a means for evaluating low-cost, high-performance satellite sub-systems. The general control policy for the satellite is that essential system functions should be provided by high reliability, hard-wired logic. Computers should be used to enhance these basic functions and should provide additional in-flight control and data processing capabilities. The primary computer[1] has been designed to undertake some of these functions using space-proven technology. The secondary computer, the subject of this paper, is designed to be a 'high risk' system in which there can be an assessment of technologies which are not space proven but which, if viable, would offer significant improvements in computer power and storage densities.

2 Design Aims

The principal aim of the secondary computer is to enhance the overall computing facilities on board the satellite. This enhancement is both one of quantity, in that added processor and memory resources are supplied, and one of degree, in that the secondary processor has a significantly different architecture from the primary computer. A secondary aim is to evaluate these differences so that the characteristics of the higher complexity 16-bit-word secondary computer can be compared under operational conditions with those of the 8-bit-word primary computer. These characteristics will include ease of programming, processing speed and the efficiency of use of the available memory space, and will be evaluated while undertaking operational tasks.

A further aim is to evaluate 'higher risk' computer technologies in space. These risk technologies comprise the F100-L collector diffusion isolation (c.d.i.) processor and the static complementary m.o.s. (c.m.o.s) memories used in this processor. The potential risk associated with the use of these space-unproven technologies is higher than that associated with the primary computer, but is acceptable because failure of the secondary computer will not impair normal operation of the rest of the satellite. The evaluation will be undertaken through the use of specially designed test programs which will monitor memory and processor error rates under space conditions and with the processor operating within the environment of a functional satellite. This evaluation will complement evaluations of the individual components under simulated space conditions on the ground.

One final aim of the secondary computer experiment is to investigate the use of both computers in multi-processor, multi-task operations. Of particular interest is whether the simple inter-processor communication links which have been provided can be used to achieve the synchronization and parameter-passing needed for such complex tasks as satellite stabilization and attitude control. As these links interact with the ground control links and other data flows in the satellite, the in-flight investigation will involve the real operating conditions which ground-based simulations can never fully represent.

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3 Design Constraints

Since the secondary computer is not flight-essential, certain design limitations have been applied to both the computer itself and to the units to which it was to interface. The computer is required to draw a peak power of less than $2\frac{1}{2}$ W from the satellite power system.[2] It is also required to be installed in a single standard machined box,[3] effectively limiting the computer to two circuit boards, each 150 by 220 mm. The interfaces between the secondary computer and other satellite units are as simple as possible and are such that had the secondary computer not finally been installed in the satellite, no changes would have had to have been made to the remaining units to enable them to function correctly.

A further requirement was that no failure mode of the secondary computer was to be capable of inhibiting or disrupting the operations of the satellite while in flight.

Authority to commence the design of the secondary computer, subject to these constraints, was received in January 1981.

4 System Interfaces

The interface between the secondary computer and the rest of the satellite has been made as simple as possible.

There are four serial data lines, as shown in Fig. 1. One accepts data from the command selector, the data having come from one of the two radio command links or from the primary computer. The second serial data input contains the 1200 baud output from the telemetry encoder, thus providing access to most of the available experiment and service data. The primary computer has additional parallel data ports which enable it to monitor experimental data at higher sampling rates than are available to the secondary computer via the 1200 baud telemetry. One of the secondary computer serial data outputs is supplied to the beacon multiplexers, where it may be selected for transmission to the ground. The second serial output feeds the command selecter where, subject to various protection gates, it may issue commands to the satellite or supply data to the primary computer.

The secondary computer power is applied from within the general satellite power system. One command line is used to control the solid-state switch which controls this power, and one analogue telemetry channel monitors the secondary computer supply current.

Two further command lines are used to control the operation of the computer, while three logic status telemetry outputs provide information on the internal state of the computer. Including duplicated power supply connections, there are a total of fifteen connections to the computer, for which a single 25-way connector is used.

5 System Design

The secondary computer has been designed to enable the majority of the facilities of the F100-L processor to be used with a minimum of supporting logic circuitry (Fig. 2). The system uses a conventional simple highway, providing access to 8192 words of read/write memory (r.w.m.), 512 words of read-only memory (r.o.m.) to the universal asynchronous receiver/transmitter (u.a.r.t.) devices implementing the serial data ports and to various subsidiary logic connected with the control of interrupts and with the serial communications baud rate.

The F100-L processor uses a data highway protocol which requires 'hand-shaking' from external memory control logic. Two memory control logic devices intended for use with the F100-L are commercially available, but unfortunately neither of these offer sufficient operating speed without excessive power consumption.

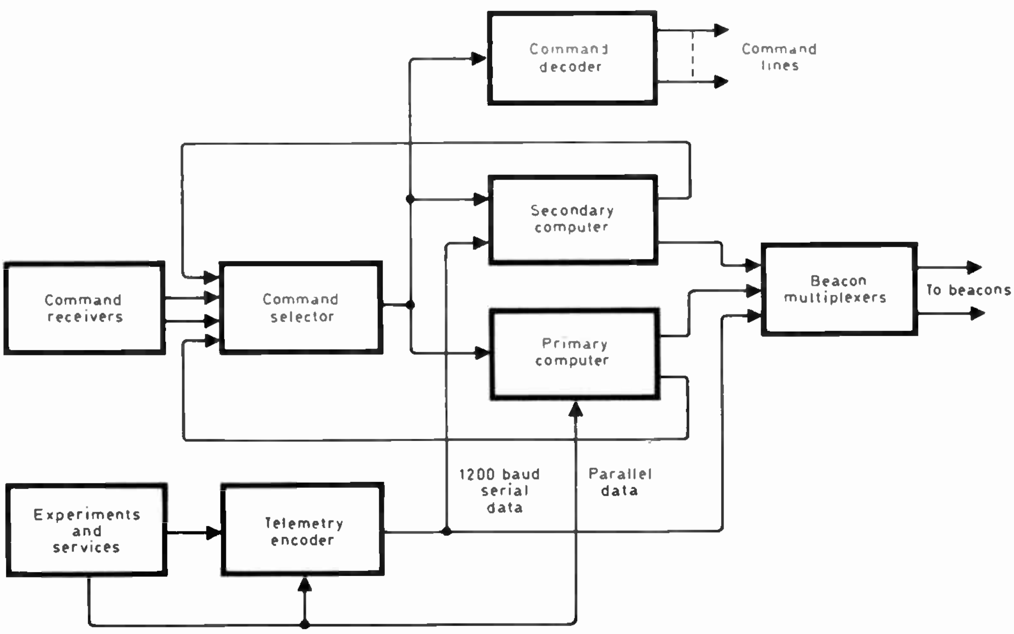

Fig. 1. Secondary computer interfaces.

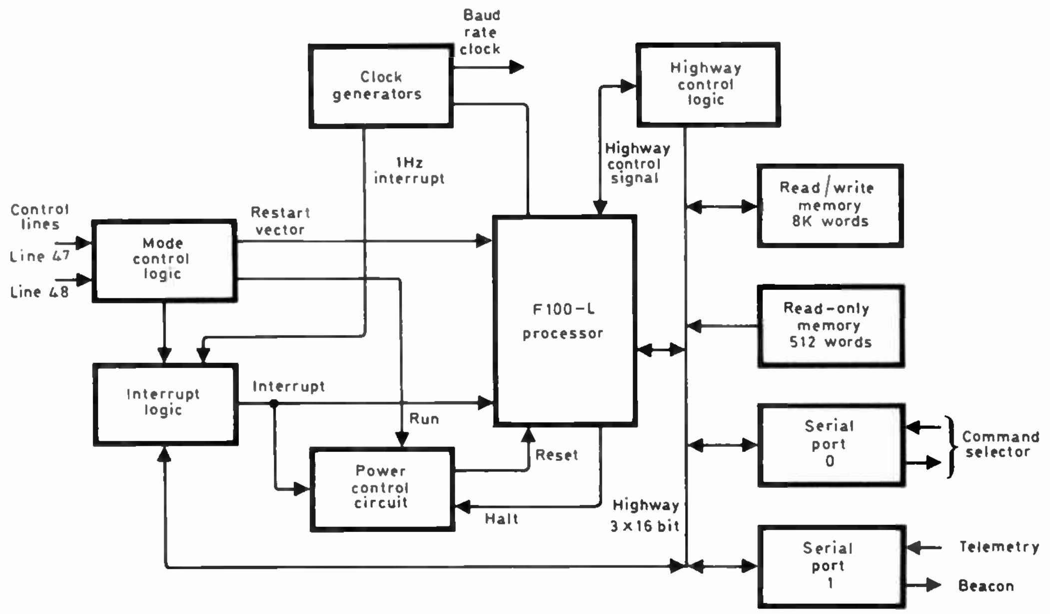

Fig. 2. Secondary computer block diagram.

A discrete logic c.m.o.s. highway control system was thus designed and permits full speed system operation with an acceptably low power consumption. The processor has a basic clock rate of 5 MHz. The processor is used with a simplified interrupt control system which allows an interrupt to be serviced by one of two processes. The loader process, stored in r.o.m., is activated by a restart vectored to address 4000 (a hexadecimal address notation is used throughout this paper). The alternative interrupt vector is to address 0800, which lies in r.w.m. and will normally initiate an interrupt-handling process which will interrogate the interrupt logic to determine the source of the interrupt and hence the required action.

Since the overall power consumption of the system has to be minimized, a power control circuit capable of providing significant power savings is employed. Its operation relies on the observation that the F100-L processor has two power supply rails. The main power input draws 290 mA (typical) at 1·2 V. A second input at 5 V consumes typically 15 mA and is used only in association with the processor output interfaces. It was found that removal and re-application of the 1·2 V supply, with suitable associated use of the 'reset' input, did not produce any transient highway control line changes so that the 1·2 V supply may be removed and re-applied without any danger of corrupting data in the r.w.m. or control registers. The power control circuit uses a bipolar transistor switch to remove the potential from the 1·2 V input at times when the processor is idle. The only data lost by the hardware during this process are those in the processor's accumulator, flag registers and programme counters. These data will normally be saved by a process which is initiated in response to a POWER DOWN interrupt. All other data in the system memory and interface circuits are preserved. The power is re-applied either by control from the ground or when an interrupt occurs.

6 Processor Control

The processor is controlled by two logic lines emanating from the main satellite control system. The mode control logic decodes the present and past states of these lines to control the application of power to the processor, the restart vector to be used and the ability of other events in the system to interrupt operation of the processor. There are five mode control states, MC0 to MC4, shown in Fig. 3. Whenever control line 47 is in the low state the processor enters the STOP mode, in mode MC0 or MC1, depending on the state of control input 48. The transition of control 47 to a logic high causes the power to be applied to the 1·2 V input of the processor, the appropriate restart vector (4000 for a cold start, 0800 for a warm start) to be applied and the processor reset sequence to be applied. Following a cold start to state MC2 (the LOAD mode) and possibly following the loading or inspection of data in the memory, the processor may be switched to the RUN mode by taking

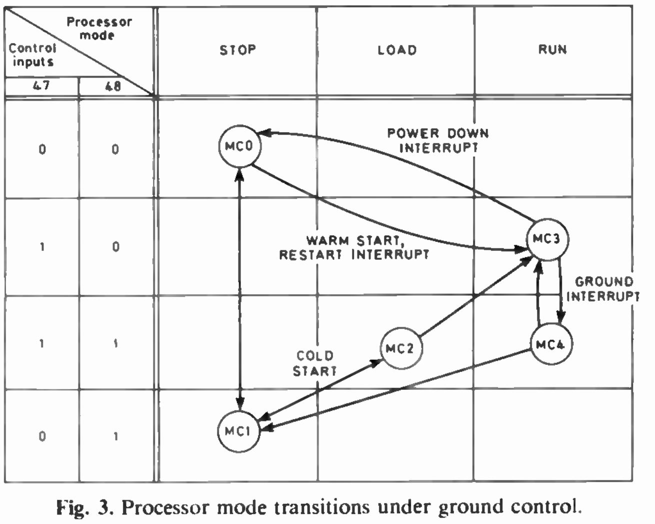

Fig. 3. Processor mode transitions under ground control.

control 48 low. This puts the processor into state MC3, which is also the state it enters following a warm start.

When in state MC3, control 48 may be taken HIGH. This will cause the interrupt logic to request a processor interrupt, giving GROUND INTERRUPT as the cause. A later return from state MC4 to MC3 by taking control 48 low again will have no effect on the system.

running mode P5 by changing to the RUN mode MC3. Note that this transition does not occasion of itself an interrupt request.

State P5 is also the stable consequence of requesting a WARM START by causing the RUN mode to be entered from the STOP mode. In this case the processor starts operation at location 0800, which normally contains the

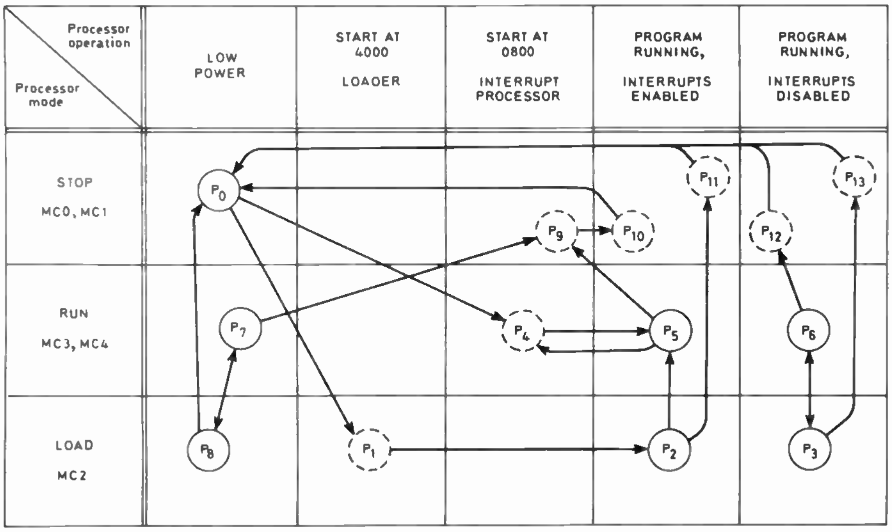

Fig. 4. Processor state transitions under ground control.

The effect of all of these Command Sequences may be seen by reference to the processor state transition map, (Fig. 4). The processor can occupy one of thirteen states, five of which (P9 to P13) are transient states associated with the power-down sequence and two of which (P1, P4) represent the transient restart or interrupt entry points.

The stopped, low-power state is P0 which is always the end result of the selection of the STOP modes MC0 or MC1. A cold start selection of the LOAD mode results in a restart at location 4000 (P1) followed by the execution in state P2 of the loader program or its nominated successor. This mode may be altered to the normal

start of the interrupt processor program. The warm start may be distinguished from interrupts through inspection of the interrupt logic status bits.

When in state P5 a GROUND INTERRUPT may be forced by moving from state MC3 to MC4 while still in the RUN mode.

In all stable operating modes the processor may be switched to the OFF state P0 by moving the STOP mode. From states P5 and P7 an interrupt, denoted POWER DOWN is generated, enabling rapid and brief consequential action to be undertaken during the transient state P10 before processor activity ceases. Although in principle it would be possible for entry to transient states P11, P12

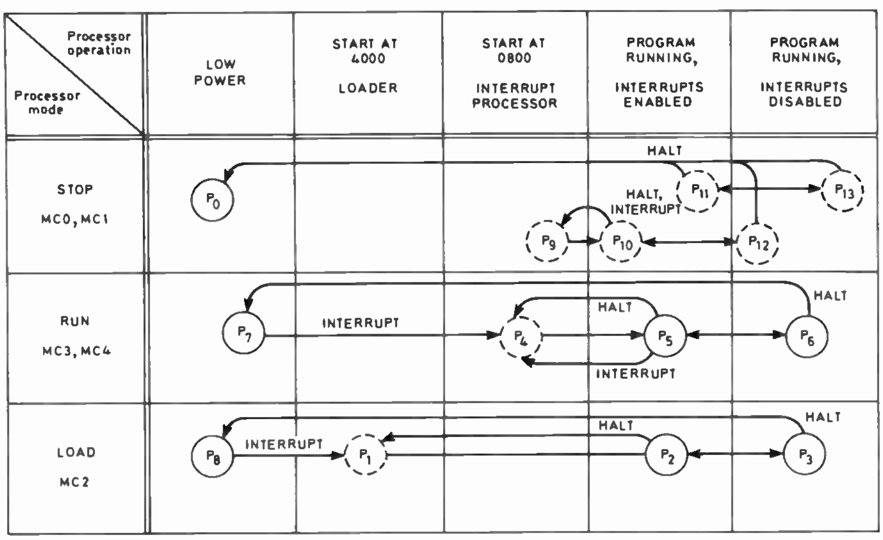

Fig. 5. Processor state transitions under program or interrupt control.

and P13 to be detected by processor, it is unlikely that this detection will ever be required and these states may be regarded as unrecognized states during the transition to P0.

There are three software commands which can affect the operation of the processor. The 'ENABLE INTERRUPTS' and 'DISABLE INTERRUPTS' commands control the internal ability of the processor to respond to external interrupt requests.

To ensure that the ground loading process has absolute priority, the external interrupt logic disables interrupt request when in the LOAD mode. It follows that the internal enabling of interrupts is generally only of significance when in state P5, as shown in Fig. 5. This is the only state in which a running program may be interrupted by an external event, resulting in a transition to state P4 before returning to state P5 (but presumably a different program segment, an interrupt handler). Notionally, an interrupt may also be processed while in the transient state P10, but this unlikely to occur.

The third software command which affects the processor is the HALT command. If the HALT command is issued with the internal interrupts enabled (states P2 or P5), the processor itself forces an interrupt to whichever start address is currently selected by the mode control logic, i.e. the processor enters state P1 or state P4.

If, however, the internal interrupts are disabled (states P5 or P6) and the HALT command is given, the processor enters a stopped condition and takes an external logic line low. In this system this line is used by the power control logic to remove the 1·2 V from the processor, allowing it to enter the LOW POWER mode P7. The power control circuit holds the processor in this low power mode until an interrupt is detected by the interrupt logic. When this occurs the 1·2 V supply is restored, the processor 'reset' sequence is re-applied and the processor restarts at location 0800 in state P4.

This sequence of states – P5 to P6 to P7 to P4 and back to P5 – forms the vitally important 'active low power' sequence which permits the mean power drawn by the processor to be minimized when undertaking tasks which require only brief operations in response to periodic interrupts.

Two further points should be noted concerning the HALT command. If issued when in state P3 (LOAD) mode with interrupts disabled) the processor will enter state P8 which will be left in response to an interrupt request. Of more significances is the use of the HALT command during the power-down process. A power-down interrupt results in an entry to P9 (Fig. 4), leading to appropriate actions being taken in state P10. When these actions have been completed it is desirable for the processor to be halted before incomplete and hence potentially data-corrupting instructions are executed as the 1·2 V input voltage drops and the reset signal is applied. If, however, the HALT command were to be issued while in state P10 (Fig. 5), the result would be another entry to the interrupt-handling process. The correct action in state P10 is the issue of the DISABLE INTERRUPTS command, to move to state P12, followed by issue of the HALT command to move directly to state P0 under program control rather than by default.

7 Hardware Design

The general *UOSAT* policy concerning the selection and procurement of components is described in Ref. 3. For this computer system, the hardware design policy was to provide reliability through simplicity and careful design to ensure that components were to operate well within tolerance boundaries. The provision of redundant components to enable continued operation after isolated failures was found to be totally impracticable in an equipment of this size. Since the secondary computer is not essential to the functional survival of the satellite as a whole, this simplistic design approach was felt to be appropriate.

The system uses the F100-L c.d.i processor and a crystal clock oscillator drive circuit using the same technology. The first divider circuit in each of the two clock divider chains and the highway control status latch use standard bipolar (TTL) logic, all remaining logic, memory, control and interfacing circuits use c.m.o.s. technology devices.

The read write memory of 8192 words of 16 bits uses two devices, each a 24 pin, 1 inch wide hybrid chip carrier holding 16 sub-memory chips each storing 4096 bits. Each carrier thus contains 64 K bits of static c.o.m.s. memory.

The read-only memory uses two fuse-programmed c.m.o.s. devices, each storing 512 bytes of data.

The system is assembled on two printed circuit boards. A two-layer board holds the 23 devices which comprise the processor, power control circuits, mode control, highway control and interrupt logic, clock generators and general interface circuitry. A four-layer board holds the highway buffers address decoders, memory and serial interface devices, amounting to 25

Table 1
Memory map

Allocation	Addresses from	to
fitted r.w.m.	0	1FFF
allocated to r.w.m. not fitted	2000	3FFF
read only memory	4000	41FF
serial ports	5000	5003
baud rate control	5004	—
interrupt ports	5008	—

Table 2
Interrupt coding

Interrupt	Bit	Disable code
power down	0	0
restart	1	1
ground interrupt	2	2
1 Hz clock interrupt	3	3
serial port 0	4	—
serial port 1	5	—

devices in all. The system was designed to accommodate a total of four r.w.m. devices, giving a total of 16 K words of 16 bits, but component delivery problems resulted in the system being released for flight with only 8 K words of r.w.m. fitted.

The memory map of the system is shown in Table 1. It should be noted that these are the prime addresses. Not all addresses are fully decoded to give unique responses within the memory space of 0 to 7FFF supported by the F100-L processor so some address ambiguity exists.

The interrupts are made available to the processor by reading a port at address 5008. The allocation of bits to interrupt causes is shown in Table 2. Also shown is the code number which, when written to address 5008, causes the corresponding interrupt request to be reset. The two serial port interrupts are reset by the process of reading their control registers. One special additional interrupt control facility has been added to the hardware – interrupts for serial port 0, which receives data from the command uplink, are only enabled when command 47 is high i.e. either in the LOAD mode (MC1) or the RUN mode following a ground interrupt (MC4). This prevents operation in the normal run mode MC3 from being interrupted by data at the command selector output which is not intended for reception by the secondary computer.

and, after the reset delay, the processor to start operating again.

8 Software Design

The read only memory, commencing at location 4000, contains a loader program which provides simple but comprehensive facilities for loading and examining data and programs and for executing programs. The program is designed to respond to simple commands issued from a normal ASCII-encoded keyboard and produces compatible output. All numeric data uses a conventional hexadecimal format.

A syntax diagram for the loader command format is shown in Fig. 7. A number consists of a sequence of any number of characters representing hexadecimal digits, i.e. characters in the range 0 to 9, A to F. A number containing no hexadecimal digits has a default value of zero. If more than four digits are present, the four right-most, least significant, are taken to represent the desired 16 bit datum. Numbers are normally terminated by a space character, a 'return' character or one of the command key letters. Any other character, printing or otherwise, will be taken as a terminator, but a question mark will be transmitted by the computer immediately following the receipt of any invalid character

The loader has an address pointer which is set to a

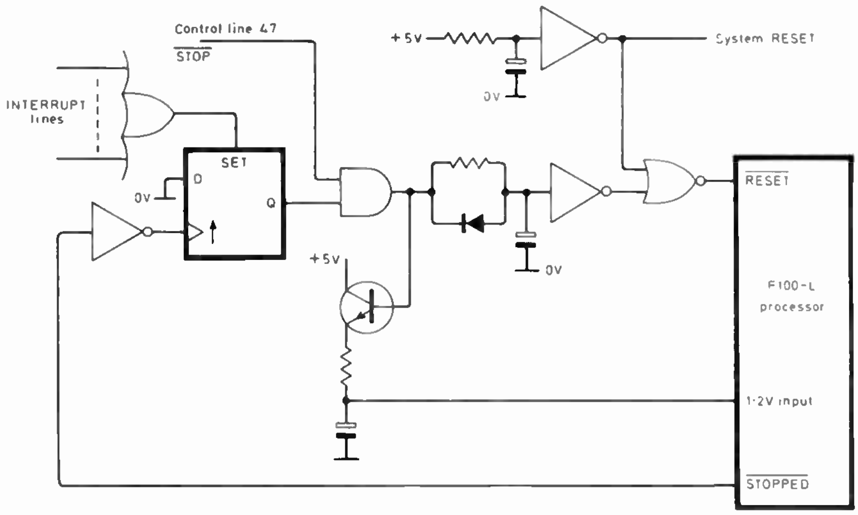

Fig. 6. Power control circuit

An outline of the power control circuit is shown in Fig. 6. Both a general SYSTEM RESET and a PROCESSOR RESET are issued when overall power is first applied to the secondary computer. Whenever the 1·2 V line is brought high, either as the result of a ground command or in response to an interrupt, a reset pulse is applied to the processor alone.

A halt command issued by the processor software when the internal interrupts are disabled results in the STOPPED line coming low. This causes the bistable circuit to move to a state in which the 1·2 V supply is removed from the processor.

The arrival of an interrupt signal causes the bistable to resume its former state, the 1·2 V supply to be re-applied

given value by means of a sequence such as @ 100. The sequence *Pn* results in *n* words, commencing at the current pointer location, being transmitted by the computer. The sequence *L n*1, *n*2, etc., results in the datum values *n*1, *n*2 .... being loaded into successive memory locations, starting at the currently designated location. The *J* command causes the processor to start execution at the currently designated location and thus terminates the command sequence and starts the execution of an application program. The format produced by the print command is compatible with the load command, thus one processor may copy its memory contents into another processor or may have its listed output stored for later reloading without any modifica-

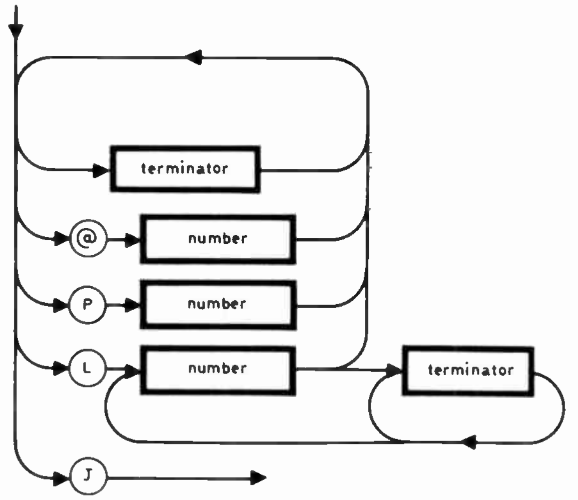

Fig. 7. Loader format.

tion whatsoever. This convention allows, for example, programs to be developed in a ground-based computer to be rapidly and efficiently transferred to the computer in the satellite.

Because the key control registers in the system are accessible within the available memory space of the computer, they may easily be modified by use of the loader. Thus key controls such as the serial port baud rate and the serial data word formats may readily be changed either by command from the ground or by programs.

The loader is written as a group of thirteen subroutines which are available for use by other programs. Key data, such as the current data input and output ports are held in read/write memory locations which are given default values by the loader start-up sequence. They may subsequently be changed so that, e.g., the routine which reads a number may be used for the reception of telemetry data on a different input port.

9 Evaluation of Design

The principal design aim of enhancing the computing facilities has been satisfied. The secondary computer provides significant extra processing and storage facilities in a form which has proved to be easy to programme and operate. Initial experience with the operational tasks so far undertaken has suggested that the 16-bit architecture provides a good match to the requirements for the simple storage of telemetry data.

The evaluation of the 'risk' technologies will take place in the months to come. Multi-processor communication links have been provided, but no synchronized multi-task activities have yet been undertaken. Sufficient monitor points and control facilities have been provided to enable the operation of the processor to be monitored so long as catastrophic failures do not occur. Thus it is planned to run extensive test programs which will run for several days and will detect and record any 'soft' memory errors and any interruption of the operation of the processor.

10 Results

The most critical parameter of the secondary computer is its power consumption. The design limit was $2\frac{1}{2}$ watts. The flight computer consumes 1·8 watts when running continuously, 312 mW mean when responding to the 1 Hz interrupt clock by incrementing a clock register and returning to a power down mode, and 300 mW when in the static power-down mode with all data and control registers preserved.

Design of the computer commenced in January 1981 and the flight model was integrated into the satellite in the second week of July 1981. Experience to date with the computer has shown that, with the exception of the detection of one minor design error, the system is functioning correctly. The secondary computer has been dedicated since launch to the collection of telemetry data and status indications to aid in the monitoring of the satellite systems and of the dynamics of the satellite. For this purpose a suit of general-purpose data collection and storage programmes has been developed and loaded into the computer r.w.m. store. No data or program errors unambiguously attributable to the secondary computer have so far been detected in operations involving the processing, in total, of at least one million bytes of data. Because the computer has been dedicated to servicing the overall satellite management activities it has not yet been possible to commence the programme of in-depth analysis of the reliability of the secondary computer.

11 Acknowledgments

Thanks are due to Dave Cruse, John Harding and Mike Stubbs of Ferranti Computer Systems for their design support, laying-out and manufacture of the printed circuit boards used in the computer, and for Ferranti's financial support of the *UOSAT* project.

Thanks are also due to Ian Ferebee for the construction of the secondary computer.

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(Paper No. 2040/IT 5)

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Indexing Terms: Television systems, Charge-coupled devices, Satellites

The development of a satellite-borne Earth imaging system for *UOSAT*

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and

T. G. JEANS, B.Sc.*

SUMMARY

Earth imaging systems have, in the past, involved the construction of fairly complex photographic and photo-electrical conversion techniques. Furthermore, the format of data transmission has restricted the widespread use of such data in educational establishments due to the fairly complex and costly receiving and display apparatus required. In this paper the development of a simple imager using a two-dimensional c.c.d. array is described. The design considerations were to reduce power consumption, weight and size, and to assemble the data in a format that could be received with very simple apparatus and displayed on a domestic television receiver. The resolution of the camera is 256 × 256 pixels with a 16-level grey scale; ground coverage is approximately 480 km square.

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1 Introduction

Because of the ready appreciation of visual information, images of the Earth from space have not only a high scientific value but also have considerable appeal for enthusiasts and educationalists. Unfortunately the reception of visual data transmitted from orbiting ground survey and weather satellites has, in the past, required complex receiving and display apparatus preventing the widespread use of this information and perpetuating the general air of mystery surrounding space-borne systems. Moreover the apparatus carried on-board a satellite for imaging and for subsequent transmission of images has been large, expensive, heavy and consumed relatively high power.

During the planning stages of the *UOSAT* project it was proposed that an Earth-pointing camera should be one of the experiments carried on board. Furthermore the format of the image data should be compatible with simple, inexpensive receiving and decoding apparatus and the display system should be a domestic television set. In this way images of the Earth would be available to anybody with a modest knowledge of electronics and with little (~£100) capital outlay.

2 System Design

The first step in designing such a system was to consider the format of the data transmission. To ensure compatibility with a wide range of amateur computing facilities the general satellite data link for telemetry has a maximum transmission rate of 1200 baud with a synchronous f.s.k. encoding format. Clearly the maximum image data transmission rate would have to be compatible with this. Further considerations of data formatting include the ease with which an image can be reconstructed and the amount of frame storage required; the latter is related to overall system cost, but the former includes not only decoding simplicity but the extent of immunity to interference-induced errors and temporary signal loss. To facilitate satisfactory image reconstruction at reasonable cost it was decided not to employ error correction on the image data but that each frame should start with a frame header code and each line with a line header code. The line header code is a 32-bit word comprising an 8-bit word and its complement repeated twice; the 8-bit word is $5B_{16}$ and was chosen on the intuitive basis that adjacent pixels would not demonstrate a large intensity variation.† This fact coupled with the complement and repetitive nature of the code should ensure low probability of detecting a false line code. The frame header is simply the line header repeated to fill an image line. The line and frame headers determine the registration of the image on the television screen and so it becomes a simple matter to combine several frames to eliminate transmission-induced errors in the non-error-corrected image data.

Having determined the transmission format, the allowable spatial and intensity resolution of the image becomes a function of the time available for trans-

† As will be seen later each pixel is coded as a 4-bit word and so the 8-bit word of the line header comprises two pixels.

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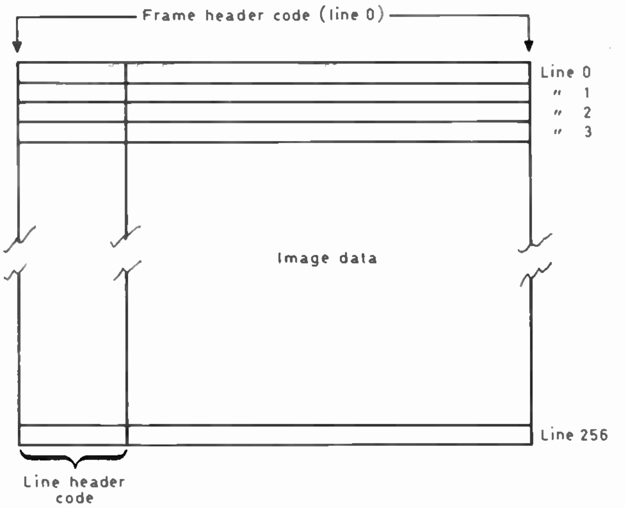


Fig. 1. The image format of the *UOSAT* image transmissions.

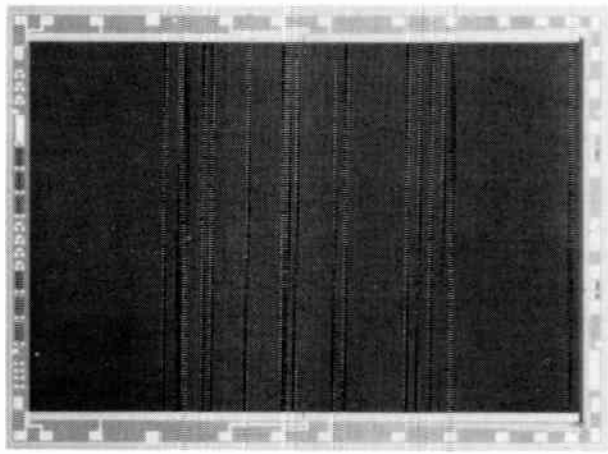


Fig. 2. The MA357 c.c.d. imager (Courtesy of GEC).

mission. During a good orbital pass of the satellite a transmission window of approximately 12 minutes exists during which several frames should be transmitted to ensure that at least one good frame is received along with any additional telemetry information from other experiments. These considerations lead to the definition of an image composed of 256 × 256 pixels each of which is coded into one of 16 intensity levels; the total transmission time of a complete frame is thus ~3·8 minutes. Figure 1 is a diagram of the image format.

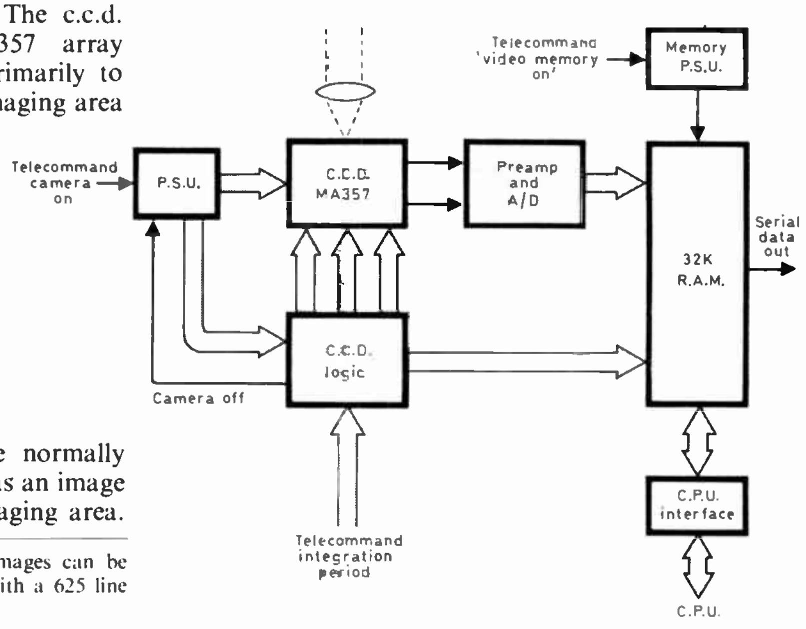
3 The Image Sensor

To fulfil the requirements of a small, compact, lightweight imager which requires little power, a charge-coupled device (c.c.d.) two-dimensional imaging array was chosen; in addition to those characteristics specified for the camera, c.c.d.s also demonstrate immunity from image burn at high exposure levels, low blooming characteristics, simple drive requirements and need relatively simple signal recovery circuits.[1] The c.c.d. chosen for this project was the MA357 array manufactured by GEC; it was designed primarily to provide television quality pictures with an imaging area of 288 × 385 pixels†[2] (Fig. 2). The device normally operates in a frame transfer mode[1] and so has an image storage area which is a duplicate of the imaging area.

Since the image transmission rate is limited to 1200 baud it would be convenient if the c.c.d. storage area could be used as a buffer which was accessed at a rate consistent with the bandwidth limitation. Unfortunately dark-current generation in the c.c.d. sets an upper limit on frame readout time at 0 C of approximately 100 ms, resulting in a minimum bandwidth of ~ 1·1 MHz.

Clearly an alternative buffer store is necessary to reduce the data rate and to maintain image integrity. This is implemented in the form of a 32 K byte memory using dynamic m.o.s. r.a.m.s. The addition of a digital buffer store with its attendant increase in power consumption and weight conflicts with the initial system requirements; however advantages accrue from the versatility that this approach offers. For instance since the buffer store is non-volatile as long as power is applied, a considerable period of time can elapse between acquisition of the image and subsequent transmission. Thus an image can be acquired under computer control at any point during an orbit and stored

Fig. 3. Block diagram of the imaging system.



† The device has been designed so that successive images can be interlaced, thus providing a video signal compatible with a 625 line television format.

type decoder which extracts the data and clock waveform from the synchronous f.s.k. signal. The detection of frame and line codes and the storage of image data uses staightforward logic design techniques; suitable circuitry is to be published elsewhere.[3]

Images obtained from the system demonstrate an acceptable degree of clarity and resolution; Figs 6 and 7 are monitor photographs of images obtained with the prototype system of some common laboratory items. Various types of blemish are observable; the vertical bands are due to c.c.d. defects and could be removed with some relatively simple interpolative signal processing. The single pixel defects are generally due to interference on the transmission link and could be eliminated with more sophisticated signal decoding or by combining several frames. Figure 8 demonstrates the facility whereby the on-board computer can write data into the video memory; in this case a grey scale test pattern has been generated.

The *UOSAT* project has provided an ideal opportunity to demonstrate the feasibility of providing Earth images of moderate resolution for reception on simple inexpensive receiving apparatus. The potential for such a system amongst amateurs and in education is large, where the simple reception of images could be supplemented with the development of signal processing algorithms to improve the image quality. In the present system spatial and intensity resolution have been drastically limited by the transmission format but if these constraints were removed the full capacity of the c.c.d. could be utilized. Figure 9 is an image from the 385×576 array demonstrating a dynamic range of > 40 dB. Thus it would be a simple matter to extend the system described in this paper to provide a relatively simple and reliable approach to earth imaging for future satellite systems.

7 Acknowledgments

The author acknowledges the work of S. K. Lee during the initial stages of this project and for the designs of the logic systems. The *UOSAT* team wishes to thank the General Electric Company for their generosity in donating the MA357 imager used in this project and particularly D. J. Burt, GEC Hirst Research Centre, for his extensive help.

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The *UOSAT* v.h.f. and u.h.f. data beacons and antenna system

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and

A. K. BROWN, B.Sc.*

SUMMARY

A single canted turnstile antenna is used for satellite transmission and command reception in the v.h.f. and u.h.f. bands. Details of the antenna hybrid and coupling filters are given, together with performance measurements. The v.h.f. and u.h.f. beacons use 5th overtone oscillators at 72 MHz, and a multistage varactor modulator provides sufficient phase modulation with overall frequency multiplication of only $\times 2$ at v.h.f. and $\times 6$ at u.h.f. Some unusual aspects of the design and construction are described.

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1 Introduction

Beacons operating at 145·825 and 435·025 MHz provide the main downlink transmissions from *UOSAT*, and a single canted turnstile antenna radiates both signals with circular polarization along the vertical axis of the spacecraft. The antenna hybrid also provides a separate isolated port for coupling to the spacecraft receivers, for reception of commands from groundstations.

Although both beacons can transmit either telemetry or data from any of the on-board experiments, the v.h.f. beacon generally transmits that information which is of interest to the majority of users, while the u.h.f. beacon will be used primarily for engineering purposes. At v.h.f. the output power and antenna polar diagrams are such that satisfactory reception is possible for a modest outlay on equipment, e.g. by using one of the readily available 2 m amateur band receivers and a crossed-dipole antenna.

A typical amateur 2 metre f.m. receiver requires an input signal of —116 dBm (0·25 μV into 50 Ω) for a recovered audio S/N ratio of 12 dB. With a satellite antenna gain of 0 dB, output of 25·5 dBm, and a receiving antenna gain of 2 dB, the basic transmission loss must not exceed 143·5 dB for satisfactory reception. At the horizon (2600 km) the transmission loss is 144 dB, so the simple crossed-dipole receiving antenna should provide a usable signal for most of a pass, and a steerable directional antenna of quite modest gain should provide horizon to horizon reception.

2 Beacon Design Considerations

The type of modulation was determined by the need for compatibility with the low cost amateur f.m. receivers which are commonly available. The bandwidth of most of these is wide enough to accommodate the ± 3 kHz Doppler shift at 145 MHz without undue distortion. The beacons were designed for phase modulation with a maximum deviation of 5 kHz for 3 kHz modulating frequency, but in fact the actual operating deviation of the v.h.f. beacon is lower, 1·6 kHz for a modulating frequency of 2400 Hz.

As the two command receivers are coupled to the same antenna system it was considered very desirable to minimize spurious emissions from the beacons in order to avoid intermodulation problems and possible receiver blocking. To this end an unusually high starting frequency is employed in each beacon. A 5th-overtone crystal oscillator generates a 72·9125 MHz signal which is doubled in the driver stage of the v.h.f. beacon. A similar starting frequency (75·5041 MHz) is used in the u.h.f. beacon, followed by tripling and doubling (Figs. 1 and 2). In order to obtain sufficient phase modulation a six-stage passive varactor diode phase modulator is used in the v.h.f. beacon (Fig. 3), and a three-stage version in the u.h.f. beacon.

In both beacons only three transistors, each of high f_t are used to generate, multiply and amplify the signal prior to the output stage, thus minimizing the number of active devices in the interests of reliability. The output stage of the v.h.f. beacon employs two T-pack TP394 stripline transistors in push-pull. The u.h.f. beacon uses a single PT8809 stud-mounted stripline output transistor,

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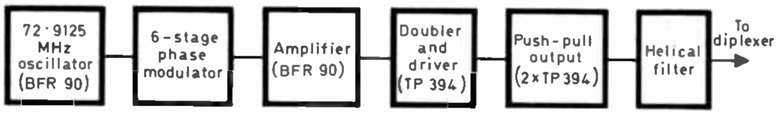

Fig. 1. Block diagram of the v.h.f. beacon.

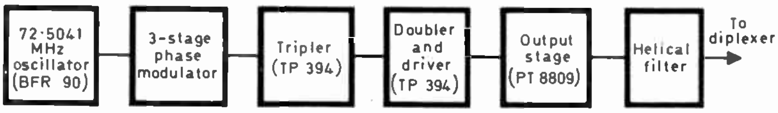

Fig. 2. Block diagram of the u.h.f. beacon.

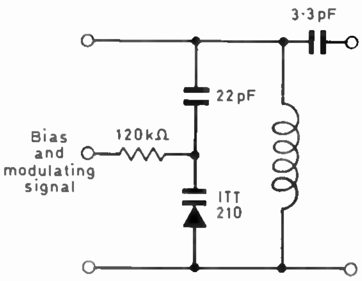

Fig. 3. Section of phase modulator.

and both beacons have helical bandpass output filters. These filters are potted in low density polyurethane foam in order to eliminate mechanical resonances. Harmonics and spurious outputs are at least 65 dB below carrier.

The impedance presented by the filters is very low outside their particular pass bands, thus permitting the use of a simple quarter-wave frequency diplexer. Both filter outputs are coupled to the single output socket via sections of coaxial cable one-quarter wavelength long at 145 MHz. This coupling arrangement ensures a high degree of isolation between the beacons regardless of antenna match.

3 Beacon Construction

The two beacons are mounted side by side in one of the standard *UOSAT* screened boxes. The 14 V power supply, modulating signals and temperature monitoring lines are connected via a multiway socket, and the combined r.f. output is coupled via a coaxial SMA connector (Fig. 4).

An unusual type of construction is used in the beacons. Instead of the normal etched and drilled circuit board, the circuit is engraved on single-sided 3 mm copper laminate board, and consists of small interconnecting areas isolated from the surrounding earth plane by 0·55 mm engraved lines. The smaller components are soldered between these areas, and holes are drilled to accommodate the larger and heavier components such as transistors and ferrite cored toroidal inductors, which are secured with epoxy resin. Power supply and modulation connections are made via wire links on the underside of the boards. The advantage of this form of construction is that all components and soldered joints are visible and accessible from one side of the board, a good r.f. earth plane exists across the whole board, and lead lengths are minimized.

For added mechanical stability the output stage of the u.h.f. beacon was potted in low density polyurethane foam. Access holes were incorporated, allowing final adjustment of trimmer capacitors after potting.

4 The UHF/VHF Antenna System

The u.h.f./v.h.f. antenna employs a canted turnstile system similar to those used on early (and therefore small) satellites (Fig. 5). It consists of four monopoles canted with respect to the spacecraft and fed via a hybrid in such a way that the monopoles are fed in phase progression, adjacent elements having $\pi/4$ difference. This gives circular polarization on boresight and linear polarization at 90° to boresight. The feed network provides a transmit and receive port which are isolated.

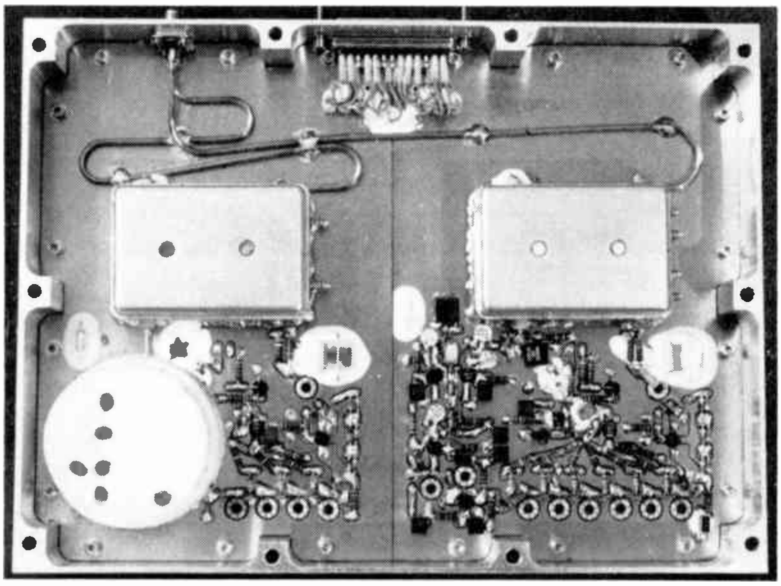

Fig. 4. Construction of beacon circuits.

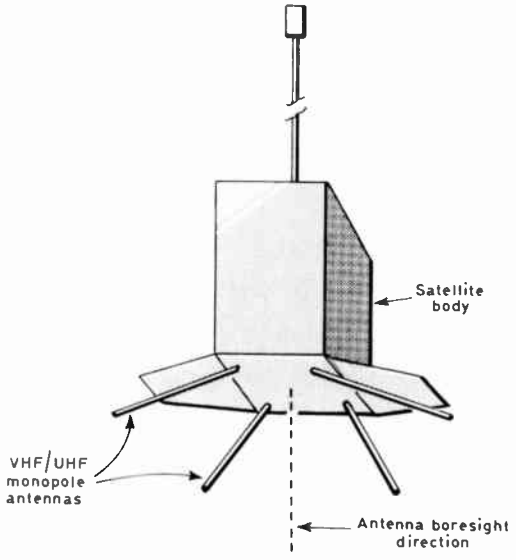

Fig. 5. Satellite outline showing canted turnstile array for v.h.f. and u.h.f. beacons.

The system is configured to give a left-hand circular polarized downlink, and a right-hand uplink. Simple cable frequency diplexers (as mentioned above) then split the transmit/receive frequencies into the u.h.f. and v.h.f. bands. It should be noted that the bandwidth required at the two bands is very small (approximately 1 MHz at 145 MHz, 3 MHz at 435 MHz) and the required u.h.f. frequencies are almost exactly the third harmonic of the v.h.f. frequencies. It is this odd-harmonic relationship which enables the same hybrid to be successfully employed for both frequencies. The use of a single antenna system was required for mechanical reasons, but it should be noted that, due to the inevitable proximity of antenna systems on the spacecraft, 'independent' u.h.f. and v.h.f. systems would couple strongly, making the advantages of this approach debatable. Initial coupling measurements between prototype multiple satellite antenna systems did not show any clear performance advantage over the dual band, single antenna scheme, and hence this was later adopted. The difficulties with the dual band approach are essentially:

(i) the impedance of the monopoles will be different at the two bands;

(ii) satellite body effects;

(iii) greater directivity at u.h.f. due to the effective use of three-quarter wave monopoles.

The monopole base impedance is important as this affects the performance of the feed network. Satellite body effects will cause differences in both impedance and overall radiation pattern from the values predicted by simple infinite ground plane techniques, and these can only be determined by measurement. To this end, a prototype antenna system was made and mounted on a full-scale model of the satellite. The cant angle and monopole length were design variables. The length and cant angles were adjusted so that a monopole real base impedance of approximately 100 Ω was achieved. A hybrid feed system was then constructed to complete the prototype. This was of cable construction with semi-rigid cables 0·085 inch diameter being used. This cable was chosen for mechanical and mass reasons. The feed system is shown in Fig. 6. It should be noted that in place of the three-quarter-wave hybrid arm a quarter-wave section may be used providing the inner and outer conductors are reversed at one end to provide an additional π phase change. This is common practice in satellites.[1,2] However, it forces high currents to flow on the outer conductor. These will cause r.f. interference particularly as the hybrid arms are not coiled. Hence interaction with nearby structures might be expected, and in practice it was found the overall isolation/input v.s.w.r. was extremely dependent on the nearby environment. This is clearly undesirable, and so a three-quarter-wave arm was employed, thus eliminating these difficulties.

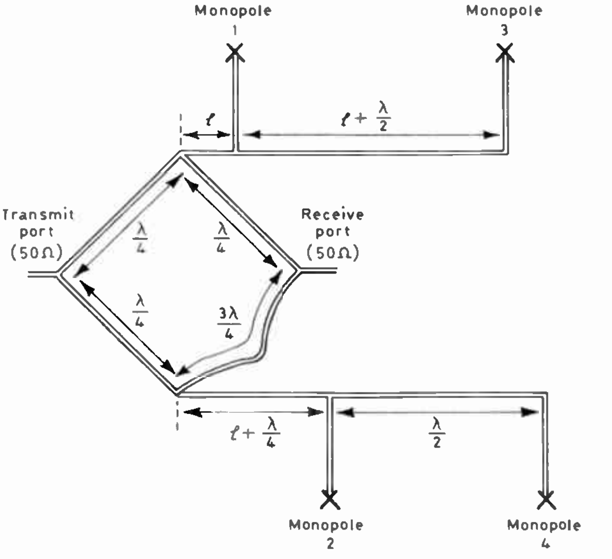

Fig. 6. Coaxial cable hybrid feed for turnstile antenna (λ = wavelength in the cable at 145 MHz. l is an arbitrary length).

Measurements of this system indicated that good isolation and input v.s.w.r. were achievable at both frequencies; radiation patterns were reasonably omnidirectional at 145 MHz (though with the anticipated polarization changes) whereas moderate directivity was observed at 435 MHz. Reasonable coverage was obtained, and as it was anticipated that the u.h.f. system would not be used until spacecraft stabilization had been achieved, the coverage degradations were acceptable. Typical radiation patterns are shown in Fig. 7, although significant differences were observed at 435 MHz for different planes of measurement.

Following these tests, a flight model was fabricated. Due to the severe temperature and vibration specification great care had to be taken in the manufacture. The monopoles were fabricated from shaped sprung steel (similar to that used in expanding steel rulers), which proved satisfactory and considerably more cost-effective than the alternative phosphor-bronze. The monopole mounts were fabricated from 'Delron' because of its known suitability for the space

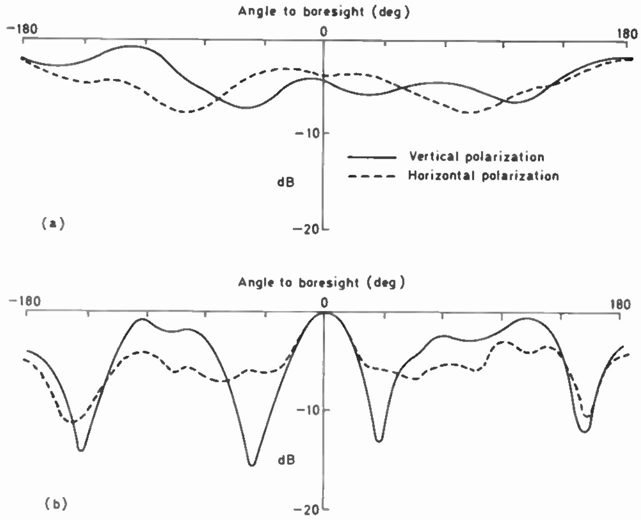

Fig. 7. Radiation patterns of turnstile antenna (a) at 145 MHz (b) at 435 MHz. Both measured using linearly polarized remote antenna, measurement plane 45° to boresight.

environment (particularly high-level ultra-violet radiation). The semi-rigid cable feeds to the monopoles were brought through the spacecraft base and secured by a suitable clamping arrangement. The feed through the base plate was sealed using epoxy resin, thus ensuring the feed cable was securely fixed. In order to allow some movement of the feed structure during vibration a small amount of cable 'slack' was incorporated.

The hybrid feed system was fabricated using the same semi-rigid cable as the prototype, with the necessary junctions constructed using specially designed mounts. These mounts supported the cable inner using PTFE inserts, and provided an extremely good earth connection to the outers. The inner conductors of the cables were then soldered and, before assembly of the junction, a thin layer of low-loss dielectric potting compound was applied to the PTFE insert to ensure proper support. This junction configuration proved extremely reliable even after severe vibration testing.

When completed and integrated into the satellite, this antenna system gave good results with isolation better than 25 dB and input v.s.w.r. of 1·3 : 1. No observable change in performance was noted after thermal cycling or vibration tests, and this has continued to be the case since launch.

5 Acknowledgments

The contribution of Mr K. M. Keen to the early antenna designs is gratefully acknowledged, as is the assistance of British Aerospace, Stevenage, and Standard Telecommunications Laboratories, Harlow, in the measurement of the antenna systems.

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The *UOSAT* h.f. beacons experiment

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SUMMARY

The University of Surrey amateur scientific satellite, *UOSAT* carries a set of phase related beacons on 7, 14, 21 and 29 MHz enabling experiments to be carried out into trans-ionospheric propagation at the four frequencies. The beacons, which deliver 100 mW each, are synthesized from the 7 MHz master oscillator and combined into one of two 2·5 m monopoles tuned against the satellite body and 15-metre boom. Modulation is by on/off carrier and can be routed from the telemetry system or the primary spacecraft computer.

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1 Introduction

It was decided at an early stage that *UOSAT* should carry at least one h.f. experiment. Initial ideas ranged from a simple h.f. beacon through to a remotely controlled communications receiver. Eventually it was decided to use four phase-related beacons within different amateur h.f. allocations.

The four beacon concept was considered to be the best compromise between a sophisticated system of doubtful reliability and a simple system of doubtful utility. In the final scheme it was decided that beacons should be located in the 7, 14, 21 and 29 MHz bands and be phase related. Furthermore they should radiate alternately continuous carrier and on/off modulation of slow Morse code sending identification and status information available from the telemetry unit. Alternatively modulation can be routed from one of the main computer output ports, therefore allowing any modulation scheme. Although the beacons can be selected individually, the modulation is simultaneous and common to all.

The beacons are primarily intended as a tool for propagation study and their use is directed towards two main groups:

(i) The professional scientist who can utilize the phase-related nature of the beacons to accurately remove the Doppler shift from relative phase measurements between the signals, thereby deriving the relative phase shifts for the different frequency paths which will be a function of atmospheric conditions.

(ii) Amateur scientific and radio enthusiasts who can use the difference between geometrically predicted and measured reception time for any particular orbit to give a relatively accurate measure of ionization density and therefore likely propagation performance. Due to the extensive worldwide network of amateur radio stations already equipped for these frequencies, it is hoped that some worthwhile correlation of these measurements with propagation conditions will be possible by the amateur community. In turn this may yield new information on h.f. propagation in unfavourable or unusual conditions.

This paper describes the design of the beacons and the associated antenna system; some ideas for receiver techniques suitable for making use of the beacon information are presented in the Appendix.

2 Designing the Beacons

The target specification to be met was as follows:

Outputs:	Four independently switchable outputs each of +20 dBm, to be capable of being phase related. Modulation by on/off keying.
Spurious outputs:	All harmonics and intermodulation products to be >50 dB below +20 dBm.
Total experiment d.c. power:	1 W.
Control format:	Compatible with 10 V c.m.o.s.
Construction:	To withstand launch and space environments.

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2.2 P.C.B. Layout and Construction

The p.c.b. which was designed on a Racal Cadet computer aided design facility, has a ground plane covering as much of the component side as possible. The prototype showed some unwanted interstage coupling, but fortunately the first p.c.b. displayed better performance in this respect and no layout difficulties were experienced.

Not all components used were space approved (mainly for cost/availability reasons); components such as coil formers and associated tuning slugs had to be protected and strengthened by the use of epoxy resin.

Soldering of the double-sided, plated-through-hole board was carried out with great care from both sides to ensure better reliability.

3 The H.F. Antenna and Multicoupler

For the h.f. beacons, two 2·5 metre self-unfurling tape antennas are mounted on the top of the satellite. These whips are disposed as a shallow 'V' on either side of the 15-metre gravity stabilizing boom which points vertically away from the earth. The 7 MHz and 21 MHz beacons are combined in a multicoupler which feeds one of the whips with the 14 MHz and 29 MHz beacons being combined in a second multicoupler feeding the second whip. The body of the satellite and the gravity boom act as a counterpoise for the whip antennas.

3.1 System Design

Originally, all four bands were to be fed to the tape elements connected as a 5-metre length dipole element (resonating in the 10-metre band), but the separation of the bands into two pairs has given a number of advantages; the multicoupler design is simplified, the efficiency and bandwidth on each band is improved, the isolation between the transmitters is greater, and the predicted radiation pattern is improved.

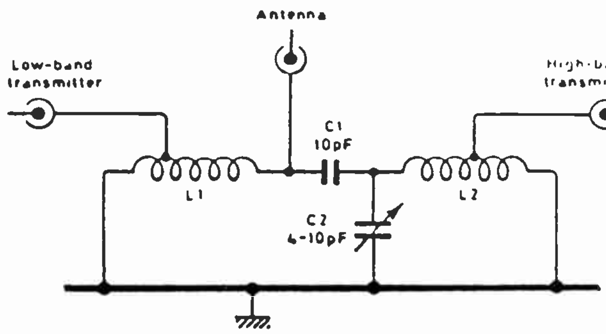
Because the 15-metre gravity boom is used as part of the counterpoise, it is strongly excited on the two lower frequency bands. This implies that at 7 MHz the radiation pattern should be that of an almost vertical antenna. At 21 MHz the gravity boom is near anti-resonance, and at 29 MHz the satellite body itself acts as a good counterpoise. On these two bands the radiation patterns correspond to those of almost horizontal dipoles. It was not possible to measure the satellite before launch with gravity boom deployed, and whether the validity of these predicted radiation patterns will hold true in practice will not be known until extensive further in-space measurements have been performed.

The two multicouplers, which are in fact in the same box, have been designed to be reasonably broadband to allow the inaccuracies in setting up. The predicted efficiency is, therefore, somewhat lower than might otherwise have been expected.

The design philosophy for the multicouplers was based on the fact that for an antenna fed at less than a quarter wave from one end, the reactance at the antenna terminals is very near that of the short section of antenna alone and is not much affected by the rest of the antenna. The resistive part as seen at the antenna terminals is, however, dominated by the length of the strongly excited remainder of the antenna system, i.e. the boom, which provides the primary contribution to the radiated field.

3.2 Design of the Networks and Results

Figure 4 shows the basic multicoupler network, which combines low and high band signals into a single antenna. Two of these networks are used in the final unit: one for 7 MHz and 21 MHz, and a second for 14 MHz and 29 MHz. The networks differ only in the coil details, and these are given in the table. Final coil values were obtained experimentally. The tapping points were also found by trial and error, and a compromise reached between the best response in order to preserve a reasonable bandwidth. The high band part of each network can be fine tuned by means of a trimmer capacitor, C2.



Freq. (MHz)	Coil	No. of turns	Tapping Point (turns from ground)	Inductance (calculated μH)
7	L1	33	6	9·8
14	L1	11	3	2·3
21	L2	18	3	4·5
29	L2	12	2	2·8

Fig. 4. The multicoupler network

For measurements made with representative dummy loads the multicoupler losses were estimated at −3·5 dB at 7 MHz, −2 dB at 14 MHz, −3 dB at 21 MHz and −1 dB at 29 MHz. However, because of uncertainties in the expected load impedances (dependent on final boom length), a further mismatch loss may degrade the figures by −4·5 dB at 7 MHz and −3 dB at 29 MHz.

4 Achieved System Performance

The only parameter which did not meet the target specification was the power consumption which, when all of the beacons and the synthesizers were on together, exceeded the 1 W limit by 20%.

5 Acknowledgments

The authors acknowledge with thanks the help of Derek A. Rye of Philips Research Laboratories in the realization, measurement and adjustment of the h.f. multicoupler hardware. Thanks are also due to Bernard Wright and others at PRL for manufacturing the housing and other component parts of the multicoupler unit. Encouragement and material assistance towards

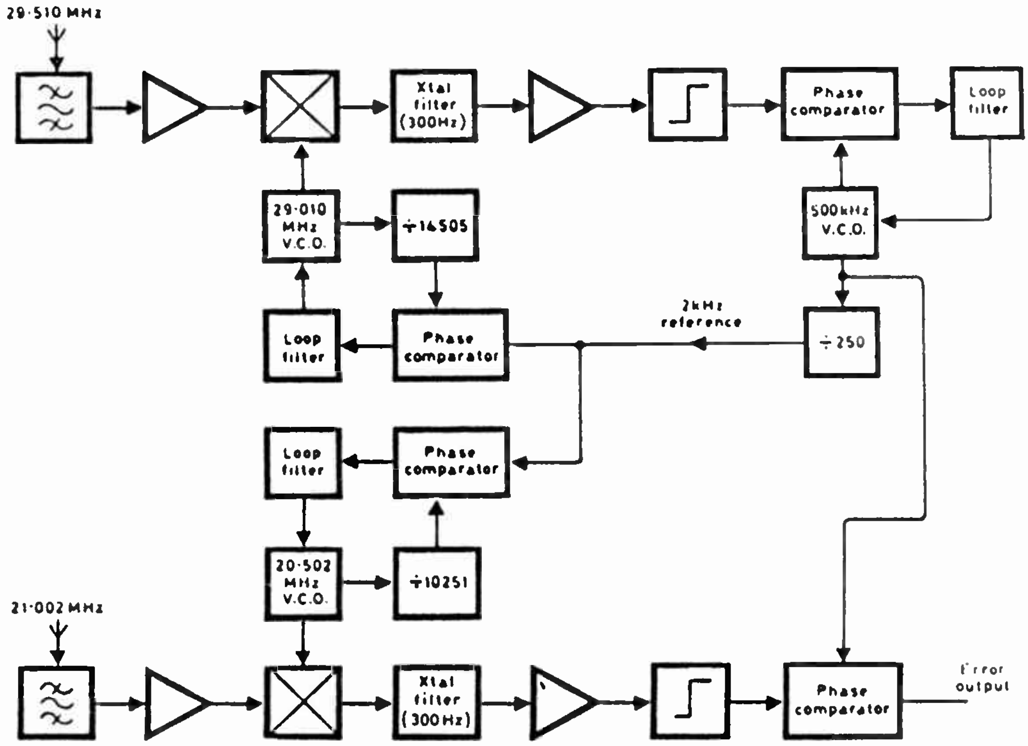


Fig. 5. Suggested beacon receiver system

the beacon board from Mullard Applications Laboratory, Mitcham is also acknowledged. The board was completed by one of the authors (C.R.S.) as the final-year project of the B.Sc. course in the Department of Electronic and Electrical Engineering at the University of Surrey, and during this period supervision was gratefully received from Mr M. S. Hodgart.

6 Appendix: Suggested Receiver Systems

The user of the h.f. phase-reference beacon system on *UOSAT* may wish to design purpose-built receiving equipment. To aid this, some possible receiver concepts are presented.

In order to make full use of the information from the beacon experiment, receivers must be capable of proportional frequency difference measurements on two or more signals in the presence of any Doppler shift effects (which affect all signals proportionally). To achieve this, an internal reference must be derived from one of the incoming signals and a frequency comparison technique employed. As the highest frequency beacon (29·510 MHz) is likely to be the least disturbed after trans-ionospheric propagation, the reference should be formed from this signal. A possible system is shown in Fig. 5 which, for simplicity, shows only two frequencies.

The incoming 29 MHz signal is down converted to 500 kHz using a 29·010 MHz local oscillator, which itself is synthesized from a 2-kHz reference. This reference is the divided output of a 500-kHz voltage-controlled oscillator (VCO). In this way, the 29·010 MHz is effectively controlled. This technique avoids any incremental error generated in the reference by the first frequency conversion.

Front-end image filtering is important as received signals are unlikely to be particularly strong. The i.f. filter should be a narrow crystal filter (300 Hz nominal) and chosen for minimum bandwidth consistent with required loop performance.

The second i.f. strip for the 21·002 MHz signal is controlled entirely by the reference frequencies from the first strip; therefore any continuously changing phase error at the output represents the changing path length difference between the two signals.

Another technique which might be suitable is that of direct frequency conversion. Here, a local oscillator frequency is generated which is locked to the incoming signal, thus avoiding difficulty with multiple conversion. Care would have to be taken, however, to avoid excessive local oscillator sideband noise limiting the receiver sensitivity.

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The UOSAT microwave beacons

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SUMMARY

The paper discusses the design and construction of the *UOSAT* microwave beacons. Several interesting design features are apparent, and these have been employed in an attempt to provide space-borne microwave transmissions for use with low-cost ground stations. In particular, difficulties due to excessive range loss have been partially overcome by the use of sophisticated design techniques, without the need for high power consumption from the satellite.

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1 Introduction

Two microwave beacons are available on the *UOSAT* satellite for use in propagation experiments. It is hoped their presence will encourage the use of microwave frequencies by educators and radio amateurs, in addition to providing research information on microwave propagation. The two beacons provided operate at 10·426 GHz and 2·401 GHz. Each beacon consists of an antenna and associated transmitter. At 10·426 GHz continuous wave emissions are provided, whereas the 2·401 GHz signal includes frequency-modulated telemetry information.

2 Design Philosophy

To encourage widespread usage of the *UOSAT* microwave beacons high performance, and hence costly, receiving equipment cannot be assumed. In order to provide a reasonable signal at a receiver, therefore, the satellite radiated power must be maximized. This is primarily limited by the available d.c. power on board the spacecraft. In practice, some 100 mW r.f. output power was achieved at 2·401 GHz and 127 mW at 10·426 GHz.

By assuming suitable ground station parameters, the required satellite antenna gain can be derived (Table 1). In this, clear weather conditions are considered. This is particularly important at 10·4 GHz as loss due to the atmosphere is strongly weather dependent,[1,2] and will hence affect the reception of signals at this frequency. The assumed ground station parameters are consistent with available radio amateur equipment, although it should be noted that a low-noise pre-amplifier must be employed at 10·4 GHz. To provide adequate ground station gain a steerable high gain antenna must be used. It is assumed a one-metre diameter parabolic dish antenna is available at 2·4 GHz, whereas a 0·4 m dish is used for 10·4 GHz. An aperture efficiency of 55% can be achieved with reasonable care without the use of sophisticated antenna feed designs. The -3 dB beamwidth of these antennas is of the order of 10° at 2·4 GHz and 5° at 10·4 GHz, which is approximately the minimum beamwidth compatible with moderate-cost antenna pointing mechanisms. The gain of these antennas is of the order of 25 dBi at 2·4 GHz and 30 dBi at 10·4 GHz.

In the calculations of Table 1, it should be noted that as the satellite moves along its orbital path, the distance to an observer on Earth will alter; thus as the satellite travels from the observer's horizon to overhead, the basic transmission path loss will decrease. In addition, the path loss due to the atmosphere will decrease, but assuming clear weather this is in any event very small. The transmission path loss is a function of distance from satellite to observer, which for any given orbit is directly related to the position of the observation point on the Earth's surface. The maximum variation occurs on an orbit which traverses directly over the observer and this is assumed in the calculations.

An ideal satellite antenna radiation pattern would compensate for the transmission path loss variation. The maximum angle subtended at the satellite antenna

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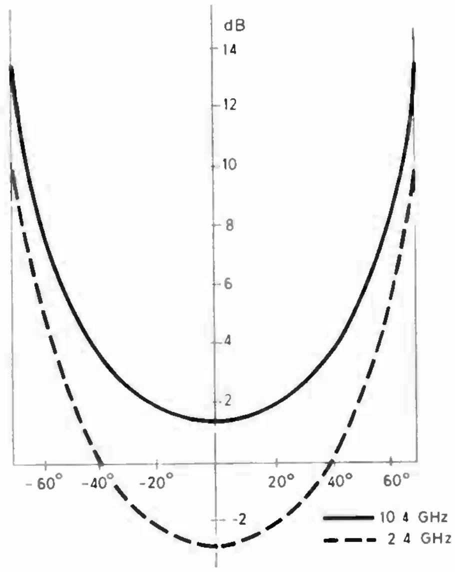

Fig. 1. Ideal satellite antenna radiation patterns

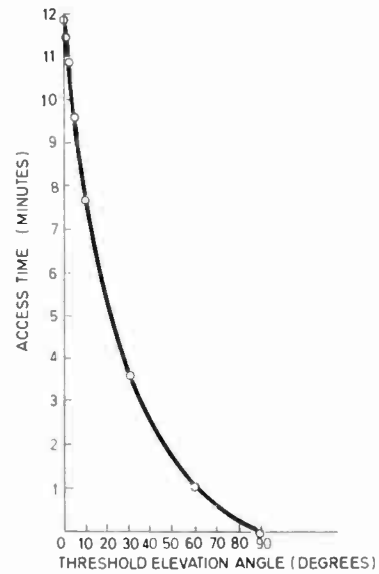

Fig. 2. Access time as a function of threshold elevation angle for an overhead pass

boresight by an observer is determined by the observer's radio horizon and is directly related to the satellite orbital parameters. In the case of *UOSAT*, this angle is approximately 70°. The ideal satellite antenna would hence provide low boresight gain (as the transmission path length is a minimum here), increasing with angle from the boresight until the 70° limit is reached. At wider angles zero gain is required. This type of pattern is therefore a 'saddle' shape and is illustrated in Fig. 1 for the two microwave beacons of *UOSAT*.

Unfortunately, for both theoretical and practical reasons these ideal radiation patterns are not realizable. Some range loss compensation by this means may be implemented, however.

The effect of a non-ideal satellite antenna radiation pattern is to increase the minimum elevation angle at which satellite signals can be received (for the given ground station parameters). In turn, for a given orbit, this limits the time for which the satellite can be accessed. Obviously, increasing ground station antenna gain and/or receiver sensitivity will improve the situation, but it is believed the specifications of Table 1 cannot be greatly improved without incurring large cost increases. The acquisition elevation angle as a function of access time for an overhead pass is shown in Fig. 2.

At microwave frequencies there is particular interest in the propagation of circularly polarized signals. Both antennas, therefore, generate this polarization, predominantly left-hand at 2·4 GHz, right-hand at 10·4 GHz (IEEE definition).

Table 1

Link power budget for microwave beacon showing required satellite antenna gain

	2·4 GHz		10·4 GHz	
	Overhead	Horizon	Overhead	Horizon
Output power of satellite transmitter (dBm)	20	20	21	21
Free-space transmission path loss (dB)	−154·8	−168	−167·5	−181
Atmospheric attenuation† (dB)			0·1	0·15
Ground station antenna gain (dBic)‖	25·5	25·5	30·1	30·1
Ground station v.s.w.r. loss, 1·25 : 1 (dB)	−0·05	−0·05	−0·05	−0·05
Receiver sensitivity (dBm)	−112‡	−112‡	−115§	−115§
Required satellite antenna gain (dBic)‖	−2·65	+10·55	+1·55	+14·80

† Assumes clear weather, standard atmosphere.[1,2]
‡ Assumes ±3 kHz bandwidth 20 dB recoverable signal noise ratio (frequency modulated beacon) and 4 dB receiver noise figure.
§ Assumes ±10 kHz bandwidth, 10 dB recoverable signal/noise ratio (c.w. beacon) and 5 dB receiver noise figure (low noise preamplifier required).
‖ The nomenclature dBic represents decibels with respect to a circularly polarized isotropic source.

3 2·4 GHz Beacon

3.1 Antenna Design

Implementation of the 'ideal' radiation pattern described above (Fig. 1) may be approached in a number of ways. Most common are choked wave-guide and inclined dipole configurations.[3] At 2·4 GHz, however, implementation of these techniques within the physical limitations of the spacecraft was not straightforward. As an alternative, the axial mode helix antenna was considered.[4] This antenna offers good circular polarization characteristics and simple, lightweight construction. However, a conventional radiation pattern results.

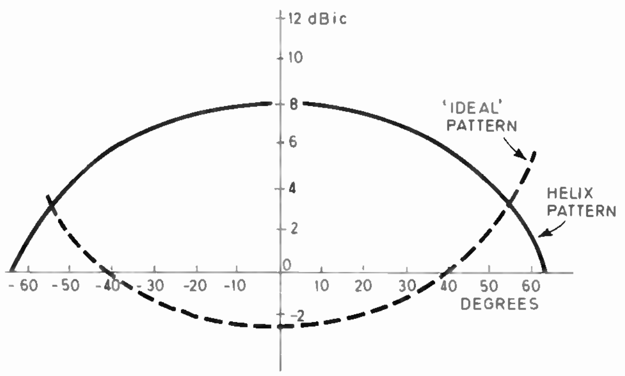

Fig. 3. *UOSAT* 2·4 GHz beacon antenna radiation pattern (3-turn helix)

Figure 3 illustrates the radiation pattern of a 3-turn helix, superimposed on the 'ideal' pattern of Fig. 1. The elevation acquisition angle will be approximately 20° which, for an overhead pass, gives a total access time of five minutes. Given the constraints of the small satellite environment, this was considered acceptable. The radiation pattern of Fig. 3 is for a helix on an infinite ground plan. However, due to the provision of adequate ground plane in practice and the antenna mounting position, the antenna pattern of the spacecraft will not greatly vary from this theoretical pattern, at least in the angular region of interest.

The antenna was constructed from a 'Delron' plastic former in a cross configuration to minimize weight. The helix itself was constructed from standard tinned copper wire 1·6 mm (1/16 in) in diameter. To achieve a good input match to 50 Ω a coaxial line quarter-wave transformer was constructed at the feed point, terminating in an SMA-type connector. An input v.s.w.r. of 1·15 : 1 was achieved. Particular care was taken in the construction of the antenna feed point to ensure good mechanical characteristics. Under thermal and vibration testing, no degradation of antenna performance was observed.

3.2 Transmitter

The S-band transmitter, provided for *UOSAT* by Microwave Modules, was designed to produce an output power of 100 mW at 2·401 GHz, with provision for frequency modulation. The crystal oscillator runs at a frequency of 80·0333 MHz, using a fifth overtone crystal, and drives a multiplier transistor stage, tripling to 240·1 MHz (Fig. 4). The following stage doubles to 480·2 MHz at a power level of 5 mW, and the signal is then filtered and amplified in two stages to a power level of 50 mW. This 480·2 MHz signal is finally amplified by a Mullard BGY 22 power amplifier module which can produce 2·5 W output. In order to reduce overall power consumption, however, the supply voltage is reduced to limit the output to 500 mW.

The all-important matching to the ×5 multiplier diode ensures that all the available power reaches the diode and that the diode sees a high impedance at the output frequency of 2·401 GHz. The output from the diode is transformed up to 50 Ω in a microstrip line, and fed to a multi-section half-wave side-coupled microstrip filter. The power output after the filter is 100 mW into 50 Ω resulting in an efficiency of 20% for the diode quintupler. This consumed the allowed maximum power of 3 W from the satellite's 15 V regulated line.

The frequency modulator consists of a varicap diode in series with the 80·033 MHz crystal. The audio input is connected to the varicap diode through a 100 kΩ resistor. For a deviation of ±10 kHz, the audio level required is 1 V peak-to-peak.

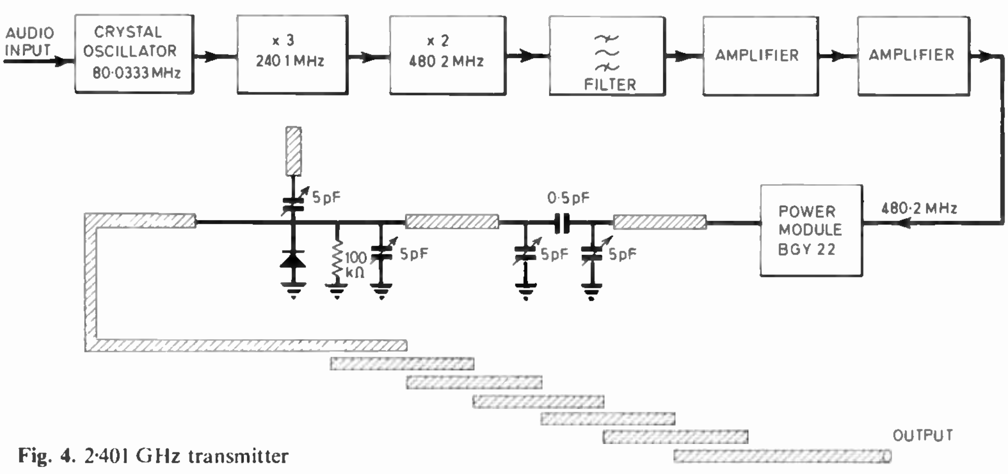

Fig. 4. 2·401 GHz transmitter

All circuitry was constructed on a teflon glass-fibre printed circuit board of 1/16 in thickness with continuous copper on one side acting as a ground plane for the microstrip circuitry. The printed circuit board measures approximately 10·2 cm × 20·3 cm (4 in × 8 in), and is fitted into one of the standard *UOSAT* module boxes. Critical components were encased in dielectric material to improve vibration resistance. During thermal and vibration testing the transmitter continued to meet its design aims. The 2·4 GHz beacon performance is summarized in Table 2.

Table 2

2·4 GHz beacon parameters

Operating frequency	2·401 GHz
Transmitter type	crystal-controlled frequency multiplier
Input power (d.c.)	3 W
Measured r.f. output power	100 mW
Expected elevation acquisition angle for overhead pass	22°
Polarization	left-hand circular
Expected maximum Doppler shift	±51·3 kHz

4 10·4 GHz Beacon

4.1 Antenna Design

The 10·4 GHz antenna, designed and built by the University of Sheffield, follows a similar philosophy to the 2·4 GHz antenna design. In this case, however, implementation of an effective 'saddle' pattern was more feasible due to the smaller antenna dimensions at 10·4 GHz. The solution finally adopted was effectively a coaxial cavity with the outer conductor slotted to produce a slot analogue of the inclined dipole antenna of Brown and Woodward,[3] as shown in Fig. 5. It is well known that a combination of a loop and a dipole fed in phase will, in principle, give ideal circular polarization over a dipole-like radiation pattern. The inclined slots effectively simulate this loop/dipole combination in a physically realizable form. In practice, the antenna is mounted on a ground plane which will affect its performance. By suitable choice of ground plane size and position, the radiation pattern of Fig. 6 was obtained (this being measured with respect to linear polarization); also shown is the 'ideal' pattern of Fig. 1. Although somewhat suboptimum, this performance was considered satisfactory. Polarization near the antenna peaks was somewhat elliptical, but predominantly right-handed.

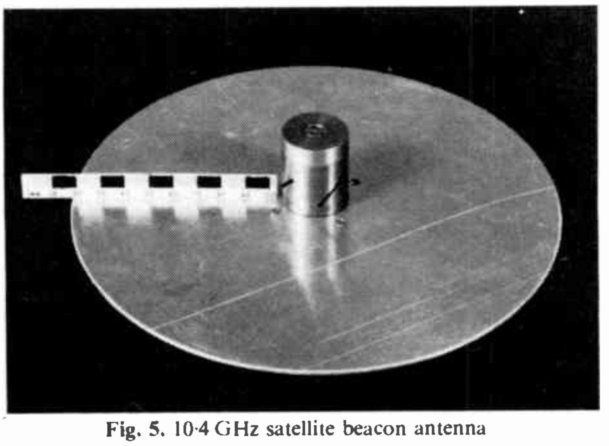

Fig. 5. 10·4 GHz satellite beacon antenna

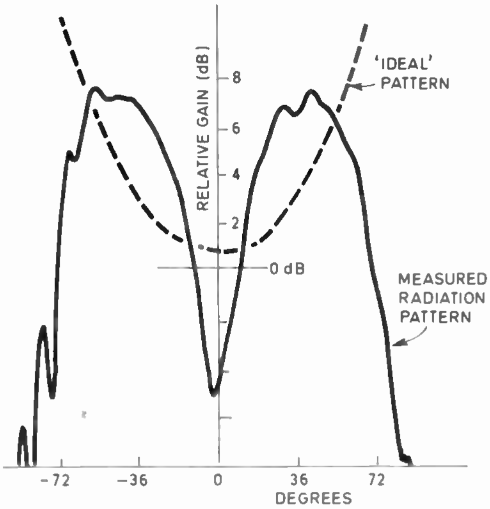

Fig. 6. 10·4 GHz beacon antenna: measured radiation pattern

The antenna was matched to 50 Ω impedance by use of variable short circuit at one end of the co-axial cavity and a PTFE quarter-wave transformer at the feed point, which was also supported by the inner conductor. An input v.s.w.r. of approximately 1·1 : 1 was achieved.

The adoption of this novel form of antenna should enable the reception of signals over the major part of satellite overhead passes. The reception of oblique passes should also be enhanced with this design compared to a conventional pattern shape.

4.2 Transmitter

For implementation of an efficient 10·4 GHz transmitter, the inefficiencies of frequency multiplication should be avoided. Furthermore, the complexities of phase-lock loop techniques are unattractive, particularly in the relatively severe temperature environment provided by the spacecraft.

For these reasons, a special transmitter unit was designed and built for *UOSAT* by the Allen Clark Research Centre of the Plessey Company. It utilizes a GaAs f.e.t. oscillator stabilized by a high *Q* resonator. This resonator uses a relatively new barium nanotitanate

Table 3

10·4 GHz beacon parameters

Operating frequency	10·426 GHz
Transmitter type	resonator stabilized GaAs f.e.t. oscillator
Input power (d.c.)	1·5 W
Measured r.f. output power	127 mW
Expected elevation acquisition angle for overhead pass	21
Polarization	right-hand circular
Expected maximum Doppler shift	±223 kHz

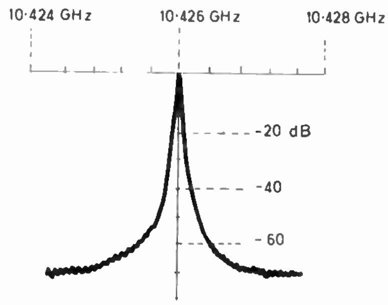

Fig. 7. Measured spectral output of 10·426 GHz beacon transmitter

compound which provides a high stability, high dielectric constant, high Q system. Typically, such an oscillator would provide 20–30 mW. Therefore, the oscillator is followed by a single power module using a Plessey PGAT 100 device. The complete transmitter is then capable of 127 mW output with 1·5 W power consumption. The measured spectral output of the unit is shown in Fig. 7. The performance of the 10·4 GHz beacon is summarized in Table 3.

5 Acknowledgments

The systems outlined in this paper could not have been achieved without the assistance of the following: Dr J. Arnold, Allen Clark Research Centre, Plessey Company; Mr R. Porter, Microwave Modules Limited; and Dr B. Chambers and Mr N. Totterhill of the University of Sheffield. Thanks are also due to Mr K. M. Keen, formerly of the University of Surrey, for his contribution to the 2·4 GHz satellite antenna.

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Indexing Terms: Telemetering, Satellites

The *UOSAT* telemetry system

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and

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SUMMARY

This paper describes the telemetry system for the satellite (*UOSAT*) designed and built in the Department of Electronic and Electrical Engineering of the University of Surrey. The function of the telemetry system is to encode the outputs of sensors, located around the spacecraft, into specified data formats prior to transmission via downlink transmitters. In addition, information relating to the status of sub-systems and data from on-board experiments require similar encoding. The basic philosophy of the telemetry system aims at providing a continuous and comprehensive surveillance of spacecraft systems for engineering purposes and a number of data formats to cater for different ground station facilities.

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1 Introduction

Over the years satellite telemetry systems have become more complex, requiring increasingly sophisticated ground station equipment. Amateur satellites, however, have concentrated on telemetry systems requiring simple ground station decoding facilities. These systems, in the main, have progressively offered a wider choice of data formats and a more extensive surveillance of spacecraft functions. The first five satellites of the *OSCAR* (Orbiting Satellite Carrying Amateur Radio) series had simple telemetry, ranging from the transmission of the character pair HI in Morse code at a keying rate dependent on temperature, to the transmission of a series of tones representing seven telemetered parameters. *OSCAR* 6 transmitted telemetry data in Morse code at ten and twenty words per minute. AMSAT-*OSCAR* 7 provided telemetry data in Morse code and radio-teletype (teleprinter) formats and also contained a codestore facility. The radio-teletype data, at 45·5 baud, consisted of sixty channels of analogue information and twenty channels of repeated command/status information. The telemetry system for the AMSAT-*OSCAR* 8 satellite, launched in 1978 and currently operational, measures six spacecraft parameters and converts the data to twenty words per minute Morse code. AMSAT Phase 111B, due to be launched in July 1982, provides telemetry formats, for educational purposes, in Morse code and radio-teletype and has a code store facility. It also provides a higher data rate format at 400 bits/second (p.s.k./p.c.m.) for engineering purposes.

Previous amateur satellites have functioned basically as radio relay stations, improving amateur radio coverage and providing valuable radio propagation information.[1,2] The University of Surrey satellite (*UOSAT*) is fundamentally different from these previous spacecraft having been designed as both an educational and a scientific satellite. While it is intended to stimulate a greater practical interest in space science in schools, colleges and universities, it also carries a number of experiments for scientific research and therefore requires a more comprehensive telemetry system than the *OSCAR* satellites.

2 Design Considerations

Knowledge of the status and performance of the spacecraft systems is essential for efficient mission management and to ensure longevity of the spacecraft's operation. Sensors located around the spacecraft give outputs representing temperature, battery voltage and current which are made available, in addition to processed data from the on-board experiments, to downlink transmitters via the telemetry encoder. The basic philosophy of the telemetry system is to provide a comprehensive surveillance of the on-board systems for engineering purposes and a wide selection of data formats to cater for the widest possible 'audience' having differing ground station facilities. To satisfy the educational objectives of the mission it is imperative that telemetry encoding formats should be in forms that can be readily received and decoded with a minimum of ground station equipment. Of equal importance is the

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AMSAT	10101	10000	00100	00000	00110	00101	10101	01000	01000
AMSAT	10101	10000	00100	00000	00110	00101	10101	01000	01000
00120	01060	02712	03001	04001	05685	06433	07681	08501	09447
10140	11080	12000	13610	14115	15509	16431	17477	18499	19450
20170	21200	22684	23152	24007	25420	26428	27255	28500	29453
30310	31080	32664	33217	34013	35287	36375	37433	38506	39449
40120	41240	42718	43009	44043	45000	46001	47490	48503	49513
50130	51100	52279	53178	54910	55421	56452	57488	58506	59477

Fig. 1. 1200 baud telemetry data frame.

provision of a high-speed data link in order to achieve the scientific and engineering aims of the mission.

The following four telemetry modes are all *simultaneously* available on board the spacecraft for transmission, as required, via down-link transmitters:

(i) Asynchronous 1200 baud ASCII7 with even parity. From ground command the data rate can be changed to 600, 300 or 75 baud. In addition, a channel 'dwell' facility is provided.
(ii) Asynchronous 110 baud ASCII7 with even parity.
(iii) Asynchronous 45·5 baud radio teletype; 5-bit Baudot (teleprinter) code.
(iv) Morse code at either 10 or 20 words per minute selected by ground command.

It was decided, at the outset, that the telemetry system should be autonomous in that although telemetry data would be transferable to the on-board computers, the system would not be controlled by the latter. Thus possible computer program loading faults would not lead to loss of telemetry. The only control access to the telemetry unit would be by direct telecommand input. It was further decided to implement the system in hardwired logic form. Although a microprocessor-based system offered certain advantages (e.g. encoding formats could be under program or ground control), the system would be liable to a possible single-point failure of a not totally known nature at the time. Also, from the point of view of completion by the specified launch date, the microprocessor-based approach suffered the disadvantage of it being difficult to estimate the time required for software development. The hardwired approach offered the benefit of redundancy since, as indicated above, the analogue-to-digital converter output could be transferred to the on-board computers. Thus, if the telemetry system were to fail down-line from the a/d.c., the computers could be programmed to transmit the data stream via the down-link transmitters.

3 Data Formats

A frame of telemetry data as displaced, for example, on a v.d.u. is shown in Fig. 1. It consists of two repeated lines each of which comprises an AMSAT header for identification followed by forty-five digital status/command indicators ordered in groups of five. The position of each digit corresponds to the status point or command function in question. These are followed by data from sixty analogue channels arranged as five-character words. The first two digits (00 to 59) of each word give the address of the analogue channel being sampled followed by the corresponding scaled datum in decimal

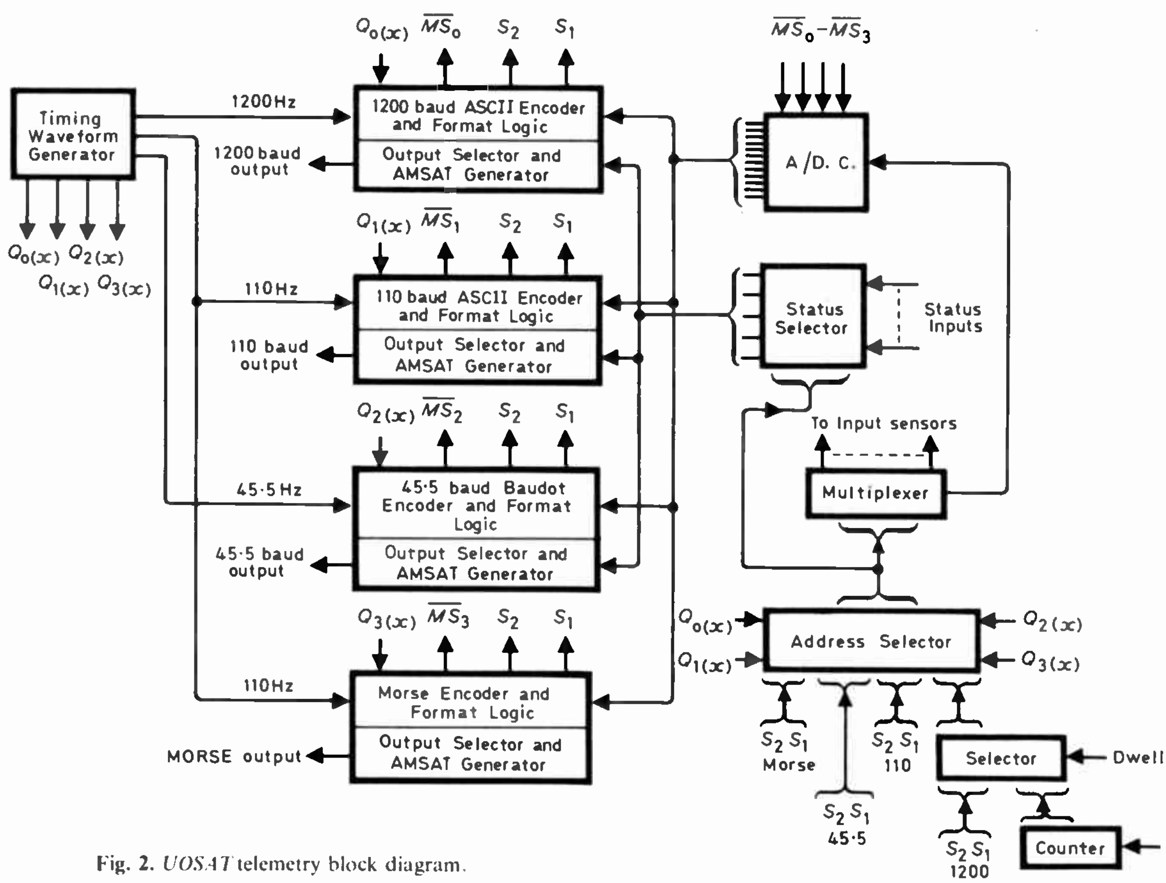

Fig. 2. *UOSAT* telemetry block diagram.

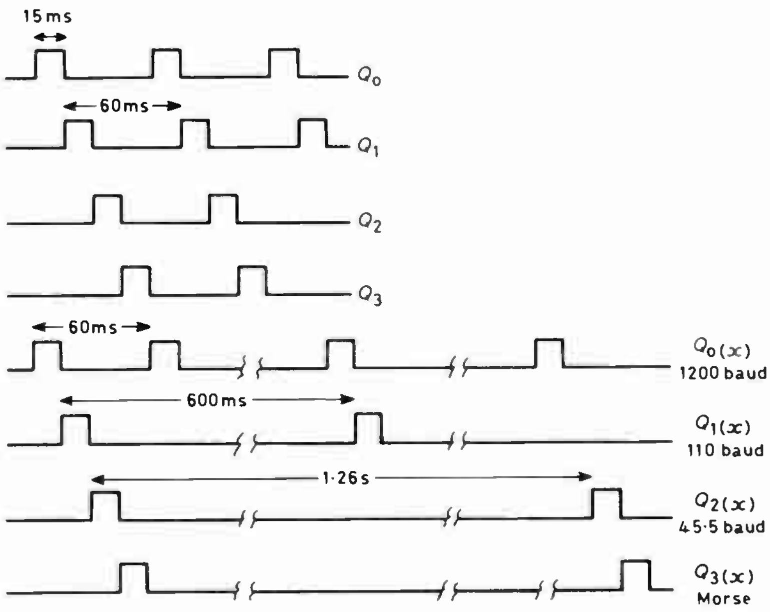

Fig. 3. System timing pulses.

form. Appendix 1 gives listings of the status points, command functions and analogue parameters being monitored. At the end of each frame additional line-feed characters are inserted in the format to achieve frame separation.

Due to the slow transmission rate for Morse code a limited set of data comprises the telemetry frame when Morse is used. This consists of the AMSAT header code followed by the first line of channel address (00 to 09) and analogue data shown in Fig. 1.

4 System Description

A basic block diagram of the telemetry encoder is shown in Fig. 2. A crystal-controlled timing waveform generator provides the bit rate for the 1200, 110 and 45·5 baud telemetry modes, and provides a clock frequency of 110 Hz for Morse. In addition, the waveform generator provides the main system timing pulses Q_0 to Q_3 from which are derived corresponding pulses $Q_{n(x)}$ ($n = 0, 1, 2, 3$) that commence each word in a given mode and allocate the time-sharing of common circuits (Fig. 3). $Q_{0(x)}$ occurs every 60 ms which is the time taken to transmit a word, consisting of five characters plus a space character, in ASCII7 code with parity at 1200 baud with one start bit and three stop bits per character. $Q_{1(x)}$ occurs every 600 ms and corresponds to the duration of a 110 baud ASCII7 word of six characters having one start and two stop bits. Similarly, $Q_{2(x)}$ repeats approximately every 1·232 seconds which is the transmission time of 5 characters plus space and shift characters in Baudot (teleprinter) code, each character comprising 5 bits plus one start and two stop bits. Although in normal teleprinter transmissions one-and-a-half stop bits are employed, it was considered worthwhile to avoid the additional hardware involved at the expense of a small reduction in data rate. Since 1·232 seconds is not an integral number of 60 ms periods, the system is designed to pause on completion of a word until the next available Q_2 pulse before generating $Q_{2(x)}$. Since Morse words are of variable duration, here also $Q_{3(x)}$ pulses occur in the first available Q_3 time after completion of a word. Coincident with the leading edge of *each* $Q_{n(x)}$ pulse a narrow pulse (15 µs) is produced, MS_n, which is used to initiate the analogue-to-digital converter.

Considering the 1200 baud encoder and associated logic, the 1200 Hz bit rate input generates character rate, word rate and line rate via a series of counters. The word and line rate outputs represent sequentially the switch address, S_1, S_2, for each sensor channel input. This address is fed to the counter selector circuit which is controlled by $Q_{n(x)}$ inputs in such a way that the 1200 baud address operates the multiplexer only during its allocated period of $Q_{0(x)}$.

The a/d.c., which is of the charge-balancing type, operates in a unipolar mode, giving a 12-bit BCD output in a full-scale range from 0 to 999. Its contents are transferred to buffer registers in each encoder on completion of the respective $Q_{n(x)}$ pulse. Each decimal digit is then converted to its appropriate code, after the corresponding switch address has been similarly encoded.

Status inputs are selected in groups of five digits and transferred to buffer registers in each output selector by the respective MS_n pulse. The output selector encodes these digits and in addition generates the alpha characters AMSAT.

For the 1200 baud channel a channel dwell facility is provided, where any one sensor may be continuously addressed. This is achieved by an address counter incremented to the required address by telecommand input, the latter also providing the dwell control for the selector circuit.

4.1 1200 baud ASCII7 Mode

Figure 4 shows a more detailed block diagram for the 1200 baud encoder and output selector. The 1200 Hz input operates the series of cascaded counters shown.

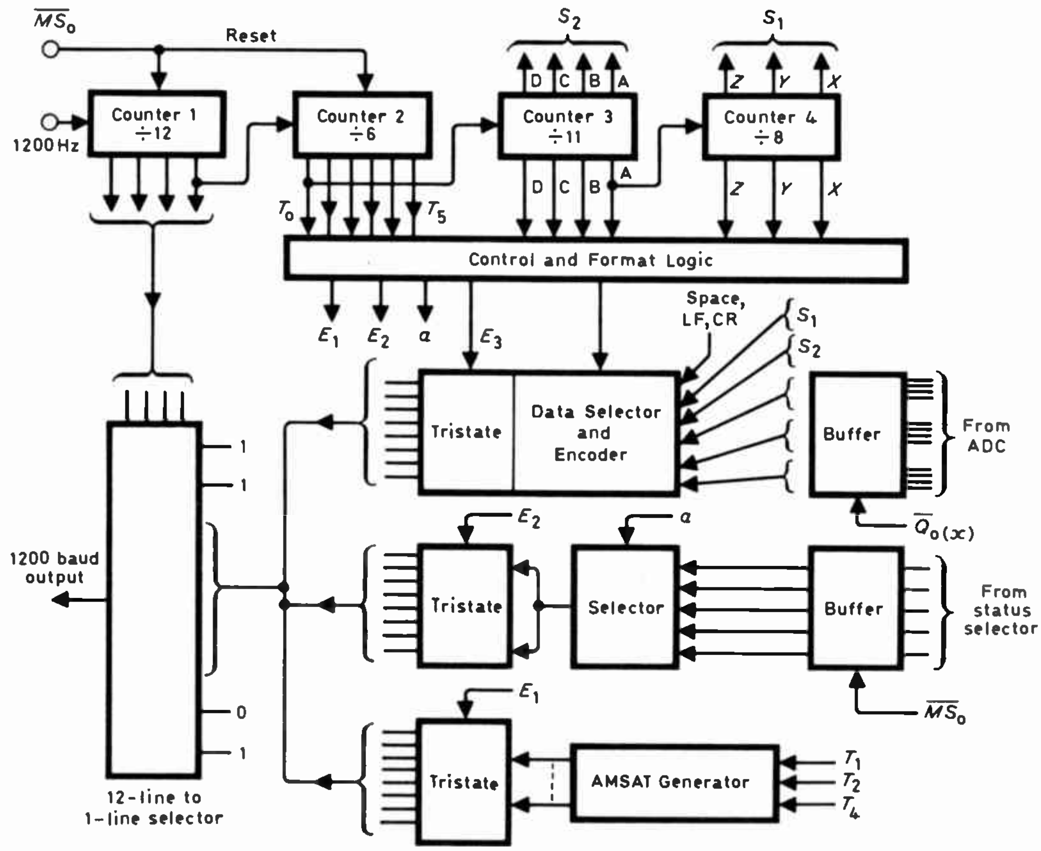

Fig. 4. 1200 baud ASCII encoder and output selector.

Counters 1, 2 and 3 are all synchronous reset types to ensure that false resetting does not occur. Counter 1 generates twelve states that sequentially select each input of the 12 line to 1 line output decoder; these inputs comprising a complete ASCII7 character with even parity, start and stop bits. To minimize subsequent logic hardware counter 2 has decoded outputs T_0–T_5, each allocated to one character in each word, T_5 being for the space character. Counter 3 is a divide-by-11 counter, each state allocated to one word in each line except for the eleventh state where line feed and carriage return characters are inserted. Each of the eight states of counter 4 is allocated to one line in a frame. The contents of counter 4 and counter 3 give the corresponding sensor switch address, S_1S_2, respectively, which connect to the selector shown in Fig. 2, S_1S_2 also connect to the data selector, together with the staticized a/d.c. output, via a BCD to ASCII7 encoder. This encoding is relatively simply achieved, requiring each BCD number to be preceeded by 011 and a parity bit generated. The data selector is also provided with inputs representing ASCII characters for space, line feed and carriage return. The tristate outputs of the data selector are enabled by E_3 for switch addresses and corresponding data from 00 to 59 and also for space, line feed and carriage return characters for addresses 60 to 79.

The second digit of the switch address, S_2, selects five digits of status information at a time, which are stored in the buffer register. Each digit is then selected and converted to ASCII code at the input of the tristate buffer. This is enabled during switch address times 61 to 79, by control signal E_2, but not when E_3 is present. It is noted that status information is made to appear at the start of a frame simply by inserting additional line feed characters at the end of the line following switch address 59.

The AMSAT word generator is shown in Fig. 5. In times T_0 to T_4 of counter 2 it sequentially generates the five alpha characters in ASCII code. The outputs are connected to a tristate buffer that is enabled during the period of switch addresses 60 and 70 (but not when E_3 is present).

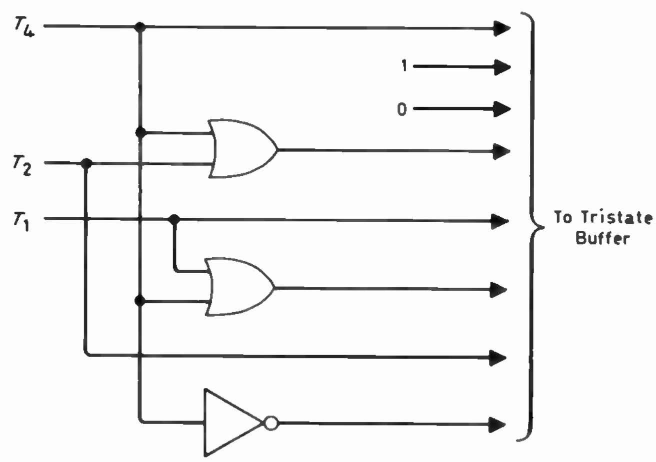

Fig. 5. AMSAT generator (ASCII7 with even parity).

The eight output lines from each tristate are combined and connected to the output 12-line-to-1-line decoder, which has fixed inputs of '0' and '1's for start and stop bits.

The a/d.c., which has a maximum conversion time of 12 ms, completes conversion within the time taken for the switch address characters to be available in serial form at the decoder output. Transfer of the a/d.c. output to the buffer register occurs at $Q_{0(x)}$, i.e. 15 ms after conversion has commenced.

4.2 110 baud ASCII7 Mode

The 110 baud encoder is very similar to that described above. It differs only in that it does not have a 'dwell' facility and each character has two stop bits. This latter feature means that the 110 and 1200 baud modes are synchronized at the expense, however, of a small reduction in data rate for the 1200 baud output.

Table 1

List of Morse characters

BCD	Morse	Binary Morse
0000	– – – – –	11111
0001	· – – – –	01111
0010	· · – – –	00111
0011	· · · – –	00011
0100	· · · · –	00001
0101	· · · · ·	00000
0110	– · · · ·	10000
0111	– – · · ·	11000
1000	– – – · ·	11100
1001	– – – – ·	11110
A	· –	
M	– –	
S	· · ·	
A	· –	
T	–	

4.3 45·5 baud Teleprinter Mode

The teleprinter (Baudot) characters to be generated comprise five coding elements plus one start and two stop bits. Each word consists of five characters, one hyphen character and a figure shift/letter shift character. The 45·5 baud encoder is similar in structure to the 1200 baud encoder but with necessary changes in timing, format and encoding logic due to the differing code requirements. It is a development of a method shown by Klein *et al.*,[3] incorporated into multiple mode operation with additional features. Further, since the duration of a word is not an integral number of 60 ms periods, the start of a new word is delayed until the next available Q_2 pulse (see Fig. 3).

BCD to Baudot code conversion may be carried out by a straightforward logic implementation. However, it is found more economical in circuit packages to employ a BCD-to-decimal decoder (CD4028) connected to subsequent OR logic gates (which are available in c.m.o.s. technology) as shown in Fig. 6.

4.4 Morse Mode

As stated previously, only a restricted data set (AMSAT header and one line of data) is transmitted in Morse due to the slow transmission speed involved. Morse elements are dots and dashes having unit durations of 1 and 3 respectively. Spaces between elements in a character are of 1 unit duration, spaces between characters are of 3 units and word spaces are of 7 units. A list of Morse characters for BCD numbers and for the alpha characters AMSAT are given in Table 1.

The Morse encoder is shown in Fig. 7, timing being achieved by a series of cascaded counters. Counter C_1 is set to divide by 12 or 6 by telecommand input setting the Morse speed at approximately 10 or 20 words per minute respectively.

The switch address output of counter C_5 connects via the input multiplexer the corresponding sensor output to the a/d.c., the output of which is stored in the buffer

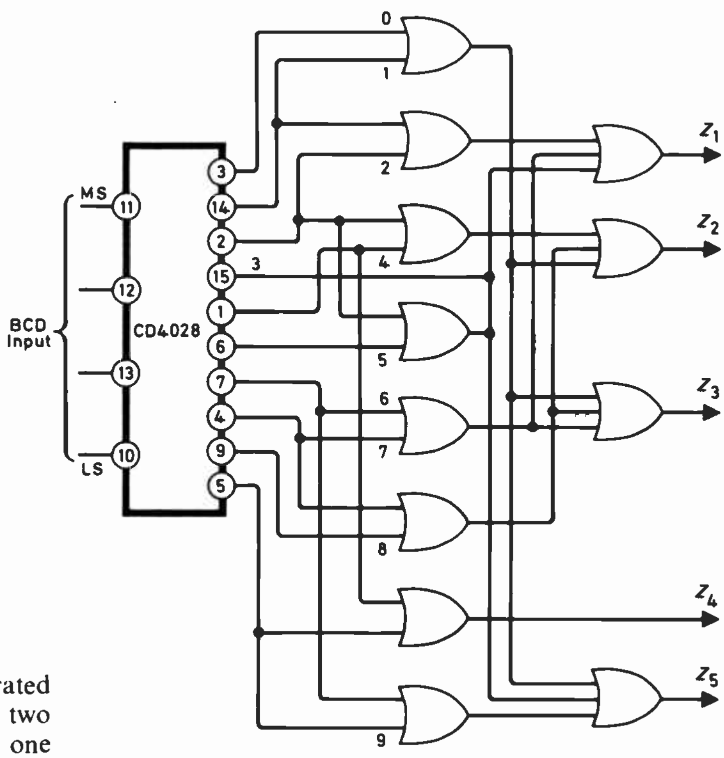

Fig. 6. BCD to Baudot code converter.

register. Counter 4 selects the switch address and corresponding datum digits in turn and encodes these to binary Morse, where a dash is represented by '1' and a dot by '0'. This encoder, too, is realized in minimal hardware by use of a BCD-to-decimal decoder feeding an arrangement of OR gates. Selector 2, operated by counter C_3, selects each Morse element in turn and controls counter C_2. For dots and dashes counter C_2 divides by 2 and 4 respectively. Output T_0 of C_2 is not used and introduces a 1-unit space. T_1 produces a 1-unit element, T_1, T_2, T_3 combined generate a 3-unit element. The number of elements (dots and dashes) are counted in C_3 and when five have occurred, in the case of numbers, the control logic inserts a 3-unit character space and

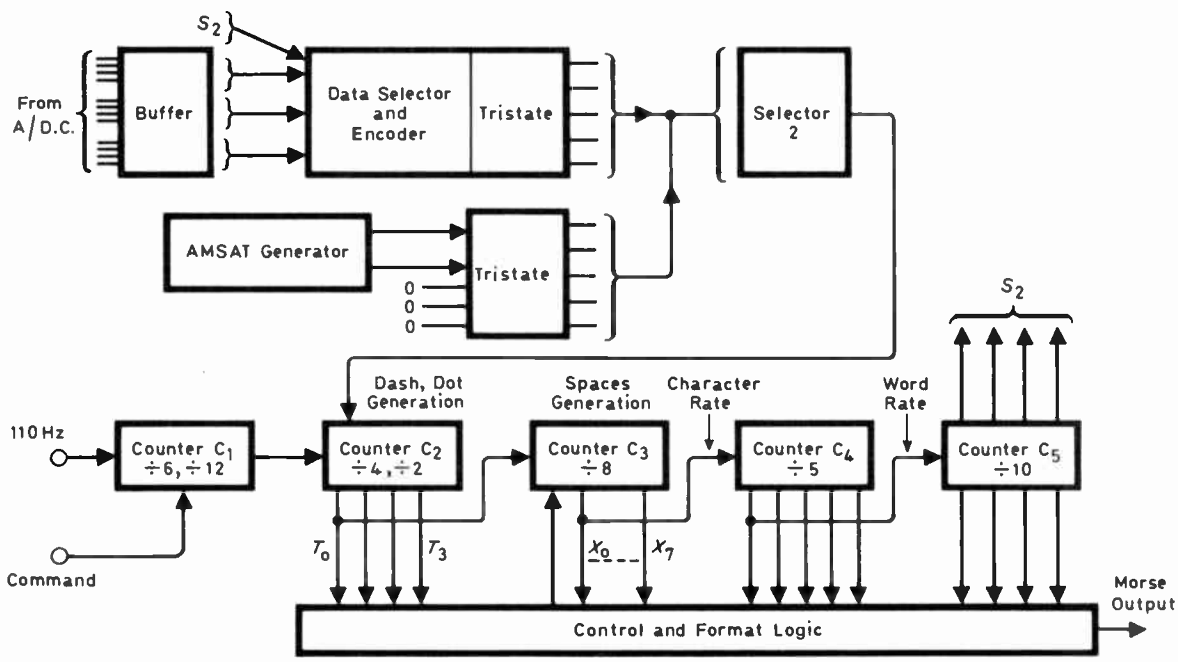

Fig. 7. Morse encoder block diagram.

resets C_3. At the end of each word the state of counter C_4 controls C_3 which generates a 7-unit space. The alpha characters also control counter C_2, as described above, the insertion of character spaces and re-setting of C_3 being derived from the control logic.

4.5 A/D.C., Input, Multiplexer and Status Selector.

The a/d.c., input multiplexer and status selector circuits are time shared and their outputs available for the various encoding formats at the required time given by $Q_{0(x)}$ to $Q_{3(x)}$.

Two-level input multiplexing is employed by arranging the multiplexers in ten groups of six. The most significant digit of the switch address selects one input in each group and the least significant digit selects one of the ten groups. The appropriate address digits are derived from the address selector as shown in Fig. 2. All analogue inputs lie in the range 0 V to 5 V corresponding to $0 \rightarrow 999$ (plus 1 digit) at the a/d.c. output. The multiplexer output is connected directly to the a/d.c. for all inputs except those connected to the temperature sensors as explained later.

The 45 status inputs are selected in groups of five (representing one word in the displayed format) by a series of digital multiplexers. Addressing is carried out by the least significant digit of the appropriate switch address derived from the address selector.

4.6 Sensors

The spacecraft parameters monitored by the telemetry system are voltage, current and temperature. They are converted to voltages in the range 0–5 V as required for the a/d.c. On-board experiments and beacons provide output voltages in the specified range and require no further processing.

4.6.1 *Voltage and current sensing*

Supply voltages being measured are brought into range by using potential dividers. Relatively low values of current are measured by monitoring the voltage produced across in-line series resistors. Each of these voltages is connected to the input of an operational amplifier having a closed-loop gain of approximately 30. The resistors were chosen so that the nominal current being measured produces a voltage, at the output of each operational amplifier, between one-third and one-half of full scale. In all analogue sections of the system, the gain setting resistors employed were experimentally selected for low temperature coefficient and closely matching characteristics. In this way gain variation due to resistance variation with temperature is maintained at better than 10 parts/10^6 deg^{-1}C. The operational amplifiers employed (LM124A quad amplifier) were chosen for their packing density, low power consumption and low temperature coefficients of offset voltage and offset current. Amplifiers were selected having roughly typical, and not the maximum, values specified by the manufacturer for these latter characteristics. Thus the low-current sensing circuits, after calibration, exhibited temperature related errors of less than two least significant digits in the output display over the anticipated temperature range of -20°C to $+50^\circ$C.

Twelve high-current measurements in the ranges 0–1 A and 0–3 A are carried out by a method developed by Meinzer[4] (Fig. 8). A 6·6 kHz square wave provides a triangular current drive via inductor L through sequentially operated f.e.t. analogue switches to twelve sense-coils. Each coil is wound on a core through which a line-current conductor is passing. For zero current in one of these conductors the a.c. drive saturates the corresponding core equally for both positive and negative excursions giving equally spaced consecutive

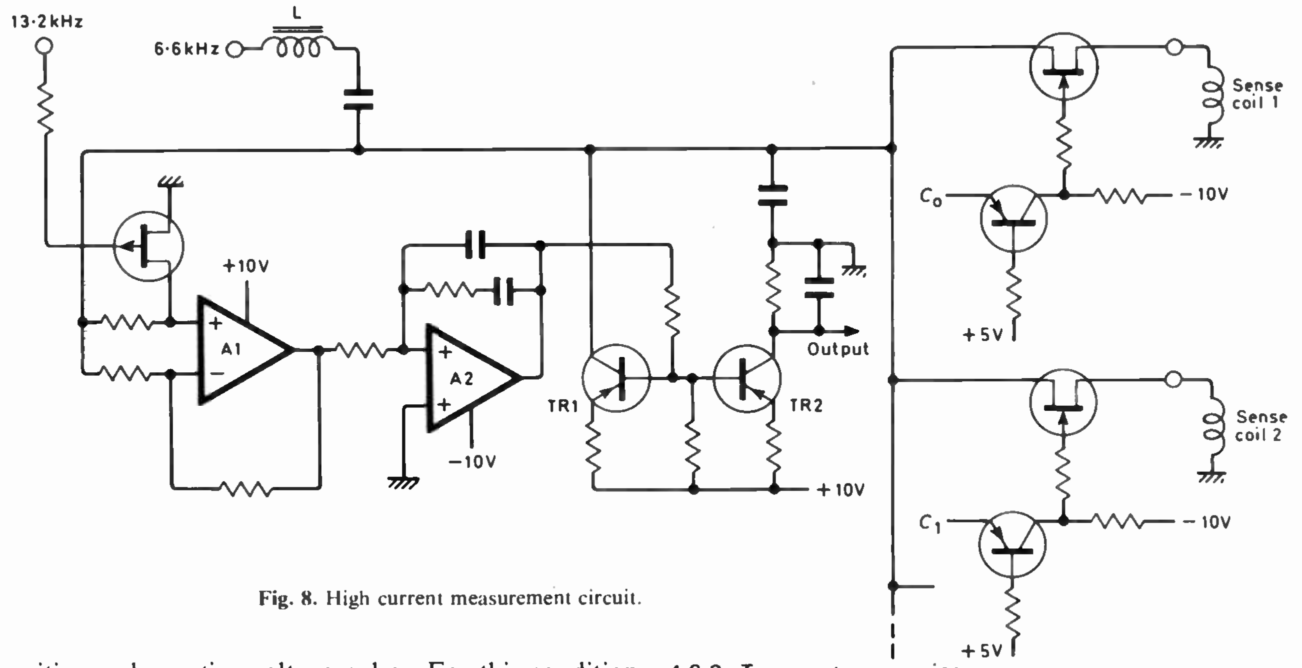

Fig. 8. High current measurement circuit.

positive and negative voltage pulses. For this condition the output from amplier A1, which incorporates a phase detector, gives a net zero output. The integrator output in steady state is approximately +7 V and transistor TR1 is off. For a line current of, say, 1 A core saturation occurs at a higher point in the slope of the triangular current wave, resulting in consecutive pairs of voltage pulses to converge. The mean voltage at the output of A1 is positive and the integrator output decreases, turning on transistor TR1. The integrator output reaches a steady value where the resulting collector current of TR1 flowing in the coil drive windings cancels the field produced by the line current. Thus with a sense coil having 100 turns the collector current of TR1 will be 10 mA for a line current of 1 A. The method operates basically as a control system, the minimum settling time being approximately 70 ms, determined by the integrator gain, the value of L and the number of turns for the drive coil. Thus each channel is sampled approximately every second.

The full-scale current ranges employed are 1 A and 3 A having corresponding sense-coil windings of 100 and 300 turns respectively. The base of TR1 is corrected to the base of transistor TR2 which, due to its different value of emitter resistor, produces a full-scale collector current of 1 mA. The collector output of TR2 is a time-division-multiplexed signal containing twelve channels of current data.

High-current measurement is carried out remotely from the telemetry unit: the incoming multiplexed signal is demultiplexed and each sample stored on a capacitor connected to the main input multiplexer. For a hold time of one second and a capacitor value of 0·22 μF the change of voltage due to analogue switch leakage current at the highest expected temperature results in an error of between 0·1 and 1% of full scale. For this reason the least significant digit in the output display is set to zero for this measurement (i.e. channels 0, 10, 20, 30, 40, 50, 11, 21, 31, 41, 51).

4.6.2 *Temperature sensing*

Thermilinear thermistor networks (YSI 44203) are employed for the temperature sensors. These are composite devices consisting of resistors and precise thermistors which produce an output voltage that varies linearly with temperature in the range −30°C to 50°C; the devices are accurate to ±0·15°C. The stable reference voltage required for these sensors is derived from a reference diode connected to a unity-gain non-inverting amplifier having very low offset temperature coefficients. To meet the specification above, the reference voltage must not exceed 3 V; this results in a somewhat low output voltage at 0°C. For this reason, but more importantly that the least significant digit should represent a convenient value, it is arranged that when these sensors are sampled by the input multiplexer their outputs are connected to an amplifier having a gain of 1·4 that feeds the a/d.c. The resulting calibration law for temperature is

$$T = \frac{(474 - N)}{5}(1{\cdot}01) \quad \text{degC}$$

where N is the displayed reading. Thus 0°C is represented by a reading of 474 and the least significant digit represents a nominal temperature of 0·2°C. These sensors and their associated circuits were found to operate extremely satisfactorily.

5 Concluding Remarks

Designed to a tight power and monetary budget, the telemetry system described achieves the objectives of operating in several modes simultaneously and gives a comprehensive surveillance of spacecraft performance. The power consumption of the system, excluding the current-sensing circuits, is less than 100 mW, this being achieved by using c.m.o.s. digital integrated circuits and low power analogue circuits throughout. The current-sensing circuits have a total power consumption of approximately 150 mW.

Space-qualified components were employed as far as possible; where these were not available military specification devices were used. The complete system underwent the mandatory vibration and thermal-vacuum tests laid down by NASA for all spacecraft launched by the Delta 2310 vehicle.

Since launch (6th October 1981), the telemetry system has operated continuously with all modes functioning and responding to command inputs. Figure 1 is a copy of an actual frame of 1200 baud ASCII7 telemetry received via one of the down-link transmitters. The system has shown no sign of any failure and it seems reasonable to expect that it will survive the full mission.

6 Acknowledgments

The authors thank J. A. King of AMSAT–USA, and K. Meinzer of the University of Marburg for their advice and assistance during this work.

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(Paper No. 2045/MI 25)

8 Appendix 1: Telemetry Sensor Allocation

Ch.	*Parameter*	*Calibration Law*
00	Secondary S/C Computer (F100L)	$I = 1{\cdot}2N$ mA ($0{\cdot}125$ A $< I <$ 1 A)
01	Solar Array Current $+X$	$I = 200 + 1{\cdot}12N$ mA
02	Battery Half Voltage	$V = 0{\cdot}01N(1{\cdot}01)$ V
03	Radiation Detector A O/P	Count $= 40N(1{\cdot}04)$
04	Radiation Detector B O/P	Count $= 40N(1{\cdot}04)$
05	Magnetometer Expt: HX-coarse	$B_x = (129Nxc - 64324) - 18{\cdot}05(NxF - 511)$ nT
06	Magnetometer Expt: HY-coarse	$B_y = (129Nyc - 64433) - 17{\cdot}97(NyF - 510)$ nT
07	Magnetometer Expt: HZ-coarse	$B_z = (129Nzc - 64072) - 17{\cdot}76(NzF - 510)$ nT
08	Battery Pack-A Temperature	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
09	Spacecraft Facet Temperature $+X$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
10	Visual Display Expt. & CCD Current	$I = 1{\cdot}2(N - 30)$ mA ($0{\cdot}15$ A $< I <$ 1 A)
11	Solar Array Current $+Y$	$I = 200 + 1{\cdot}12N$ mA
12	2·4 GHz Beacon Expt: Power O/P	$P = 0{\cdot}45(N - 145)$ mW
13	Radiation Detectors Expt: EHT volts	$V = N$ volts
14	Radiation Detectors Expt: Current	$I = 0{\cdot}125(N - 20)(0{\cdot}983)$ mA
15	Magnetometer Expt: HX-fine	see channel 05
16	Magnetometer Expt: HY-fine	see channel 06
17	Magnetometer Expt: HZ-fine	see channel 07
18	Battery Pack-B Temperature	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
19	Spacecraft Facet Temperature $-X$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
20	Spacecraft Computer Current	$I = 1{\cdot}2(N - 25)$ mA ($0{\cdot}125$ A $< I <$ 1 A)
21	Solar Array Current $-X$	$I = 200 + 1{\cdot}12N$ mA
22	Battery/BCR +14 V Bus	$V = 0{\cdot}02N(1{\cdot}056)$ V
23	Sun Sensor $+Z$ Axis	$V = 1{\cdot}01N/200$ V
24	10·4 GHz Beacon Expt: Current	$I = 0{\cdot}25(N - 40)(0{\cdot}97)$ mA
25	Magnetometer Expt: Temperature	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
26	Magnetometer Expt: Current	$I = 0{\cdot}125N(0{\cdot}99)$
27	Telecommand Receiver Current	$I = 0{\cdot}125(N - 16)(0{\cdot}952)$ mA
28	Radiation Expt: Temperature $+X1$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
29	Spacecraft Facet Temperature $+Y$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
30	Battery Charge Current	$I = 2{\cdot}9N$ mA
31	Solar Array Current $-Y$	$I = 200 + 1{\cdot}12N$ mA
32	Power Conditioning Module +10 V	$V = 0{\cdot}93N/60$ V
33	Telemetry System Current	$I = 1{\cdot}084(N - 16)/30$ mA
34	2·4 GHz Beacon Expt: Current	$I = 0{\cdot}4(N - 11)(1{\cdot}072)$ mA
35	145 MHz Data Beacon: Power O/P	$P = 1{\cdot}67(N - 82)$ mW
36	145 MHz Data Beacon: Current	$I = 0{\cdot}25(N - 11)(1{\cdot}072)$ mA
37	145 MHz Data Beacon: Temperature	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
38	Pri. S/C Computer Temperature $-X1$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
39	Spacecraft Facet Temperature $-Y$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
40	+14 V Line Current	$I = 2{\cdot}86N$ mA
41	+5 V Line Current	$I = 1{\cdot}28(N - 50)$ mA 75 mA $< I <$ 1 A
42	Power Conditioning Module +5 V	$V = 2{\cdot}24N/300$ V
43	Sun Sensor $-Z$ Axis	$V = 1{\cdot}01N/200$ V
44	H.F. Beacons Expt: Current	$I = 1{\cdot}038\ (N - 36)/3$ mA
45	435 MHz Data Beacon: Power O/P	$P = 1{\cdot}792(N - 102)$ mW
46	435 MHz Data Beacon: Current	$I = 1{\cdot}053(N - 34)/3$ mA
47	435 MHz Beacon: Temperature	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
48	Sec S/C Computer temperature $+Y1$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
49	Spacecraft Facet Temperature $+Z$	Temp $= 0{\cdot}2(474 - N)(1{\cdot}01)$ degC
50	+10 V Line Current	$I = 3N$ mA
51	−10 V Line Current	$I = 1{\cdot}3(N - 60)$ mA
52	Power Conditioning Module −10 V	$V = 0{\cdot}015N - 0{\cdot}022N'$ (N' of +10 V line)
53	Navigation Magnetometer Y-Axis	$B_y = 183{\cdot}49(N - 663)$ nT
54	Navigation Magnetometer Z-axis	$B_z = -189{\cdot}54(N - 337)$ nT
55	Navigation Magnetometer Z-Axis	$B_x = -194{\cdot}55(N - 496)$ nT
56	Speech Synthesizer Current	$I = 0{\cdot}1(N - 16)(1{\cdot}009)$ mA
57	C.C.D. Imager Temperature	Temp $= 0{\cdot}2(474 - N)(0{\cdot}01)$ degC
58	Telemetry System Temperature $-Y1$	Temp $= 0{\cdot}2(474 - N)(0{\cdot}01)$ degC
59	Spacecraft Facet Temperature $-Z$	Temp $= 0{\cdot}2(474 - N)(0{\cdot}01)$ degC

9 Appendix 2: Telemetry Status/Command Function Allocation

		'0' '1'
01	145 MHz General Data Beacon	OFF/ON
02	435 MHz Engineering Data Beacon	OFF/ON
03	Primary Spacecraft Computer	OFF/ON
04	C.C.D. Camera Module	OFF/ON
05	Radiation Detector—A	OFF/ON
06	Magnetometer Expt	OFF/ON
07	7 MHz Beacon Expt	OFF/ON
08	14 MHz Beacon Expt	OFF/ON
09	28 MHz Beacon Expt	OFF/ON
10	21 MHz Beacon Expt	OFF/ON
11	2·4 GHz Beacon Expt	OFF/ON
12	10·47 GHz Beacon Expt	OFF/ON
13	145 GHz Command RX	SQUELCH O = signal present
14	435 MHz Command RX	SQUELCH O = signal present
15	Status Calibrate	
16	BCR Status	A/B
17	H.F. Beacons Expt Synthesizers	OFF/ON
18	Telecommand Decoder Status	Ground/Primary Computer
19	Magnetorquer	OFF/ON
20	Primary S/C Computer Block Load Port	DISABLE/ENABLE
21	Secondary S/C Computer Data O/P	ACTIVE/INACTIVE
22	Secondary S/C Computer Clock	ACTIVE/INTERRUPT FAILURE
23	Secondary S/C Computer Processor	OFF/RUNNING
24	Secondary S/C Computer Power-Down	ON/OFF
25	14 MHz H.F. Beacon Synthesizer Lock	OUT/IN
26	28 MHz H.F. Beacon Synthesizer Lock	OUT/IN
27	21 MHz H.F. Beacon Synthesizer Lock	OUT/IN
28	Radiation Detector—B	OFF/ON
29	Tip Mass Uncaging Confirmation	NO/YES
30	Speech Synthesizer Power	OFF/ON
31	Visual Data Display Memory	OFF/ON
32	Gravity Gradient Boom Motor Power	OFF/ON
33	Secondary S/C Computer Power	OFF/ON
34	H.F. Beacons Expt Power	OFF/ON
35	Navigation Magnetometer Power	OFF/ON
36	S/C Computer Memory Error Bit—1	
37	S/C Computer Memory Error Bit—2	
38	S/C Computer Memory Error Bit—3	
39	Status Calibrate	
40	Primary S/C Computer Data UART O/P	INACTIVE/ACTIVE
41	Gravity Gradient Boom Motor	FORWARD/REVERSE
42	Magnetorquer Power	FORWARD/REVERSE
43	Magnetometer Expt	MEASURE/CALIBRATE
44	Navigation Magnetorquer	SAFE/ARM
45	Gravity Gradient Boom Motor	SAFE/ARM

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Indexing Terms: Radio equipment, power supplies, Solar cells, arrays, Satellites

Power supplies, conditioning and distribution on *UOSAT*

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SUMMARY

The paper describes the solar arrays, the batteries and their characteristics, and how the control of power is managed on board *UOSAT*.

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1 Introduction

Primary power is provided by solar arrays on four faces of the spacecraft. One of the battery control regulators is used to ensure that maximum power can be transferred from these to the rest of the spacecraft. The battery charge regulator also ensures that the battery is not overcharged. In order to perform these functions accurately it is necessary to make allowance for the temperature of both the solar arrays and the battery. The second battery control regulator is redundant and used to improve system reliability. It may be selected by command or automatically on failure of the other regulator. A d.c.-to-d.c. convertor in the power conditioning module provides alternative voltages to the battery voltage. A switching and distribution module supplies power to selected parts of the spacecraft on instruction from the command module. Over-current protection is provided to switch off malfunctioning modules. A block diagram of the complete power system on *UOSAT* is shown in Fig. 1.

2 Solar Cells

The *UOSAT* spacecraft is powered by solar cells manufactured by the Solarex Corporation;[1,2] they are mounted on four panels attached to the satellite mainframe, each panel supporting 408 high-efficiency solar cells. The panels themselves comprise an aluminium honeycomb sandwich 6·35 mm × 380 mm × 628 mm. The outer face of each panel is covered with a thin layer of Tedlar dielectric material which electrically isolates the cells from the panel (see Fig. 2(a)). The panels have built-in metal inserts to enable the completed assemblies to be bolted directly onto the four faces of the satellite. All wiring and electrical connections are duplicated for redundancy.

The cells are electrically connected with silver mesh in groups of three in parallel, then further in 17 rows of 3 in series, and finally in four columns of 17 × 3 in series, (see Fig. 2(b)). This layout was decided upon so as to maximize the number of solar cells within the physical dimensions of *UOSAT*. To ensure redundancy, two of these banks of solar cells are arranged on each satellite face.

A thin layer of adhesive-backed kapton is attached to the back of each panel in order to increase thermal emissivity and thus reduce the temperature excursions of the solar cells. A bead thermistor capable of measuring temperatures from $-30^\circ C$ to $+50^\circ C$ is attached to the honeycomb panel by means of epoxy resin. All sun-facing areas not covered by solar cells are covered in an optical solar reflecting layer to reflect heat radiation incident on the satellite. The four panels are routed in parallel to the battery charge regulator (BCR) via blocking-diode pairs. These diodes prevent current flow back into the non-illuminated solar cells.

The complete solar panel assemblies were designed to withstand temperatures of at least $-120^\circ C$ to $+90^\circ C$, severe vibrations and shock, since these conditions can be experienced in a space environment and during launch.

The individual solar cells have an active area of

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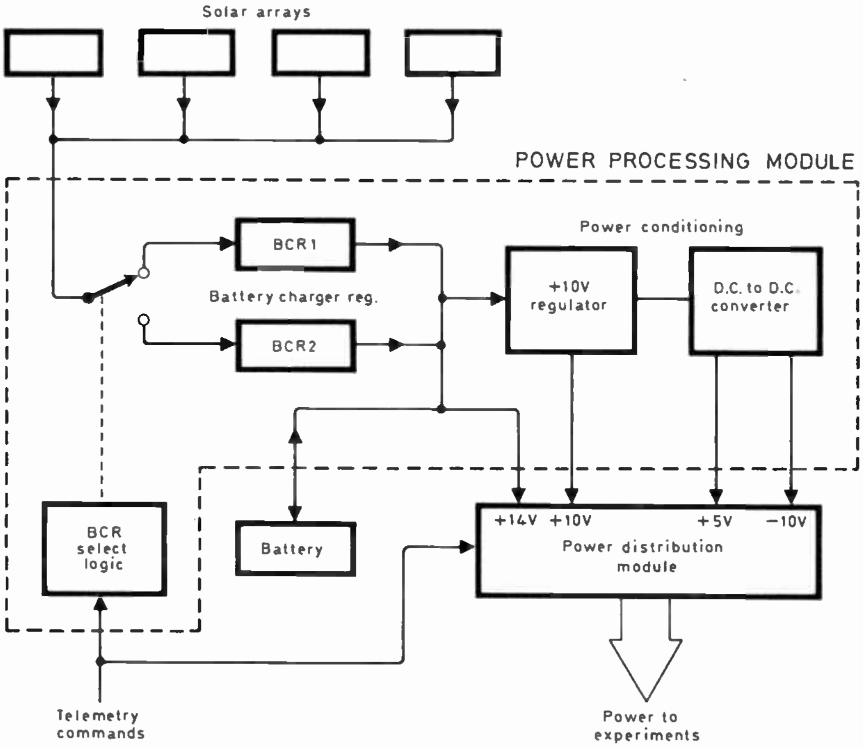

Fig. 1. Block diagram of the *UOSAT* power system.

2 cm × 2 cm and are of an n-on-p shallow-diffused, gridded structure. The silicon cells are covered with an optically-transmissive cover-slide to protect the cell from hard particles and radiation. The cover-slide also increases the thermal emissivity of the cells.

The conversion efficiency of the solar cells is approximately 12%, resulting in an average power from each solar panel of 29 W when illuminated at AM0 (Air Mass Zero) at 25°C. The maximum power available from the solar cells varies considerably with temperature, a typical cell characteristic being shown in Fig. 3. It can be seen that the maximum power decreases from 0·11 W at 5°C to 0·08 W at 85°C. As with all silicon diodes, the solar cell terminal voltage decreases with increasing temperature. To obtain maximum power, the BCR input voltage must therefore track with the solar array voltage. In orbit, the power system is designed to provide 18 W peak power and 8 W (orbit) average to charge the 14 V NiCd battery.

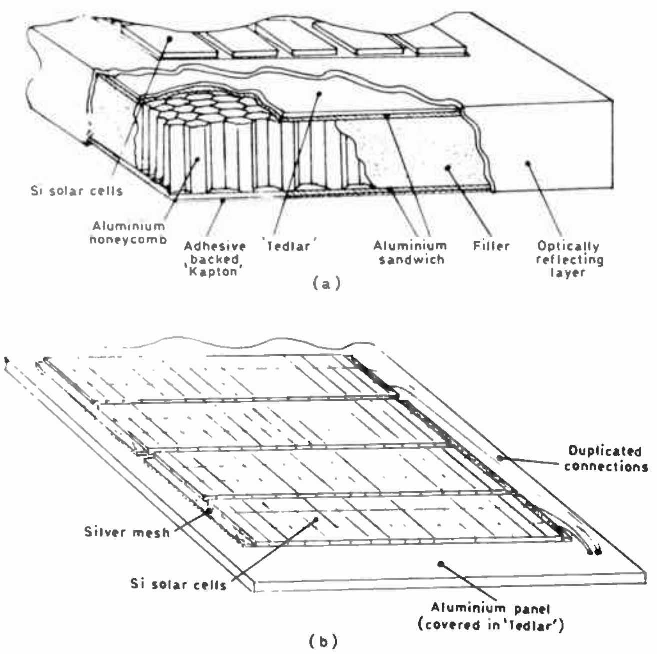

Fig. 2. (a) Construction of solar cell mounting panels (b) Connection of solar cells.

3 The Power Processing Module

The power processing module was designed and built by Meinzer and Hass at the University of Marburg, and was based on a design used in previous *OSCAR* series satellites. Simple circuitry ensures high reliability. Despite this, reliability was further improved by operating two identical battery charge regulators in a redundant arrangement.

3.1 Battery Charge Regulator Design

A circuit diagram of the battery charge regulator is shown in Fig. 4. This was derived from a standard series switch mode regulator. In order to provide an optimum load for the solar arrays it is necessary to make the regulator input characteristic track that of the solar arrays. This may be achieved using the emitter-base voltage of TR1 (a p-n junction similar to the solar cells) as the input reference voltage.

TR1 is physically remote from the solar arrays and

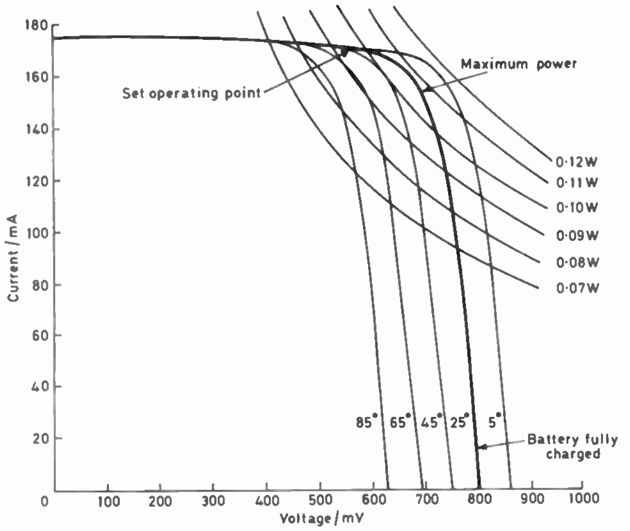

Fig. 3. A typical cell characteristic.

thermal analysis of the satellite suggested it could be at a significantly lower temperature than them. The regulator input voltage was therefore set at a lower than optimum value, as shown in Fig. 3. This avoids the possibility of a considerable reduction in input power should the regulator input voltage become too high.

The output characteristic of the regulator is equivalent to a current source which dumps power into the battery regardless of its voltage. Under these conditions the battery voltage indicates its state of charge. As the fully charged voltage is approached, the input conditions to the regulator are modified thus spoiling the match to the solar arrays. This results in the operating point moving higher in voltage along the solar array characteristic (Fig. 3) so that only a fraction of the available power is used. When the battery is fully charged its charging current is reduced to about 100 mA.

The battery voltage sensing circuit also contains a temperature compensating reference voltage in the form of diodes D2 and D3 which are situated one in each of the two battery packs. These diodes alone produce a higher temperature coefficient than the battery. For this reason a constant voltage, produced by Zener diode Z1, is added to the reference voltage. By adjusting the Zener voltage, nearly exact tracking of the battery voltage may be achieved, thus ensuring that maximum charge may be transferred to the battery at all temperatures without the risk of overcharging. High overall efficiency of the BCR is achieved by keeping the voltage drop across the series switch TR4, TR5 and the catching diode D1 small compared with the input and output voltages respectively. Also their switching times are small compared with the switching frequency of 30 kHz. A Schottky diode was therefore chosen for D1.

One of the two BCRs is selected by a latching relay. Change-over may be achieved by ground command or automatically if the selected BCR fails to oscillate. Automatic change-over is inhibited if the solar array voltage is less than the battery voltage, thus preventing hunting when the spacecraft is in the shadow of the earth.

3.2 Power Conditioning

In addition to the 14 V battery supply it was necessary to provide voltages of +10 V, −10 V and +5 V. The +10 V supply is provided by a series switching regulator. This produces in excess of 10 W with 5% regulation. The simple inverter power supply shown in

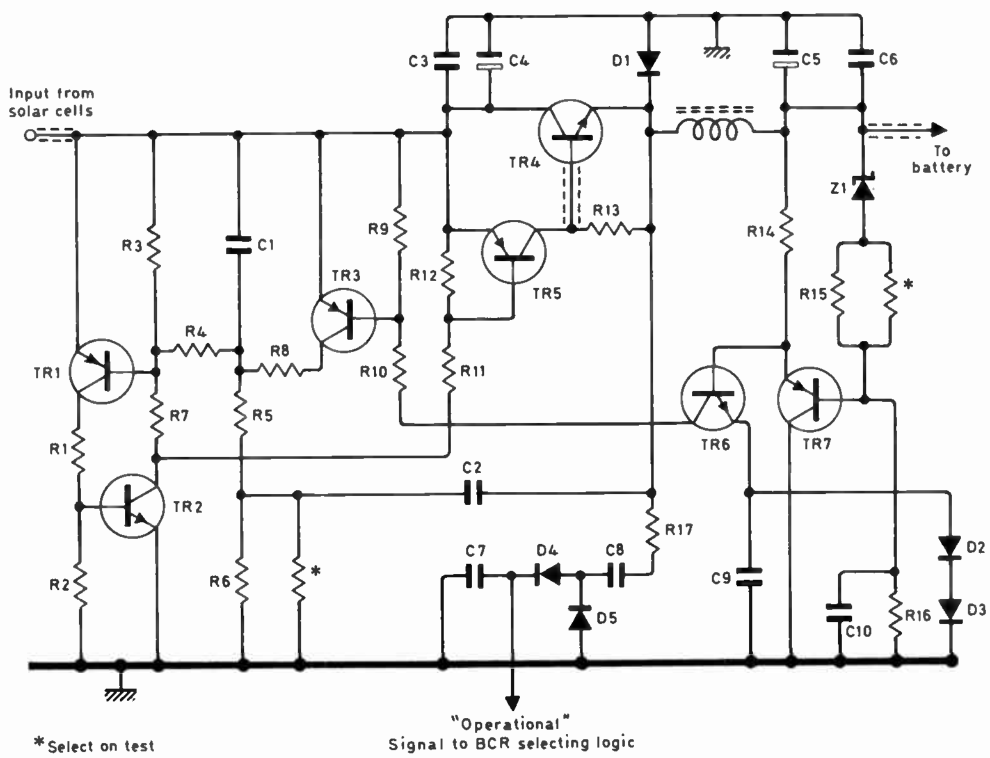

Fig. 4. The battery charge regulator.

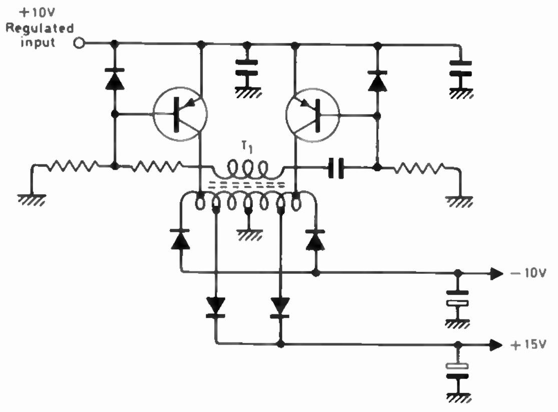

Fig. 5. Invertor power supply.

Fig. 5 is used to provide the +5 V and −10 V supplies. A combination of careful design of transformer T1 for minimum leakage inductance and use of low forward voltage drop rectifier diodes produces supplies with acceptable regulation. The overall efficiency of these power supplies is about 88%. The +5 V and −10 V converter is supplied from the +10 V regulator; the total output power of all three supplies is therefore limited to 10 W.

4 Battery Test and Conditioning

The nickel-cadmium batteries used in *UOSAT* were provided by NASA. They are arranged as two packs of five cells all connected in series with a nominal battery voltage of 14 V and a capacity of 6 Ah. It is necessary to use cells with very similar capacities since, during deep discharge, a low capacity cell in the battery may become reverse charged and suffer permanent damage. Fourteen cells were available and when first received, their condition was unknown. An initial conditioning charge and discharge was carried out, enabling the cells to be put into matched groups of four or five for the remaining tests. The reconditioning and test procedure is shown in Table 1.

The electrolyte leakage test was performed by spraying each cell with a cresol red–chalk–acetone–water-solution. Any electrolyte leakage is indicated by a violet colour change. The final capacities were found to lie in the range 7·2 to 8·3 Ah. From these it was possible to select a set of ten with capacities between 7·3 and 7·8 Ah for the spacecraft battery.

5 The Power Distribution Module

It was decided to use a central switching unit to supply power to all those experiments which may need to be switched off to conserve power. The unit acts as an isolation point between the power supplies and the individual experiments and includes over-current fold-back in order not to drain the battery if an experiment malfunctions. In all, there are 26 power requirements to be switched (often more than one to each experiment). Only the receivers and command module are not routed via the power distribution module (PDM). Typical current consumptions for each module are shown in Table 2.

The 14 V power is taken directly from the battery whilst the ±10 V and 5 V power is drawn from the power conditioning module. Since the 5 V rail from the

Table 1
Battery reconditioning and test procedure

	Action	Rate† (amperes)	Period (hours)	NOTES
Test 1: Conditioning	charge	C/20	48	Condition charge
	discharge	C/2	~2	Charge recorded down to 1V per cell using a capacity meter.
	discharge		16	Connect 1Ω across each cell
	short circuit		2	
Test 2: Capacity check	charge	C/10	24	Charge recorded down to 1V per cell using a capacity meter
	discharge	C/2	~2	
	discharge		16	Connect 1Ω across each cell
Test 3: For short circuit	open circuit		24	Terminal voltage should recover to 1·15 V per cell
Test 4: Electrolyte leakage (see text)				
	charge	C/3	1·5	Charge to prevent deep discharge during tests 5 to 15
Test 5: Simulated operation	charge	C/3	1·5	
	discharge	C/3	1·5	
Test 6 to 15	Repeat Test 5			

† C is the capacity of the cells in ampere-hour.

Table 2
Maximum current consumption (mA) of each module.

Power Supply	14 V	+10 V	−10 V	+5 V
Camera	61	20	50	466
Telecommand	—	25	—	—
F100L computer	—	—	—	250
CDP1802 computer	—	175	—	—
Receivers	—	30	—	100
Voice synthesizer	—	50	—	100
10 GHz Tx	—	100	5	—
U.H.F. Tx	150	—	—	—
V.H.F. Tx	95	—	—	—
H.F. Tx	—	150	—	—
Radiation expt.	—	100	3	—
Boom	—	—	—	250
2·4 GHz Tx	200	—	—	—
Magnetorquer	1100	—	—	—
Navigation magnetometer	—	50	—	—
Telemetry	—	7	10	5
Magnetometer	—	55	—	—

power conditioning module is unregulated, a separate regulated 5 V supply is generated and routed to the telemetry. It was felt that semiconductor switching would be most suitable because of the high power drain of conventional relays or the relatively high cost of latching relays. All power supply wires were duplicated for redundancy which used up all the 150 connections allowable from any one module (6×25 way D connectors). All earth wires were routed to a single point on the satellite mainframe.

6 Current Sensing

The PDM also includes the telemetry current monitoring circuitry. For all low current experiments (up to 200 mA), a current sensing resistor is connected in the appropriate power line, the voltage across which is measured (typically 50 mV) as an indication of the current being passed. High current experiments have their power lines routed through toroidal magnetic cores used by the telemetry system for current measurement.[3]

7 Current Foldback

Current foldback was designed into all power switches, which automatically switches off any experiment which may be drawing excess current. The general form of the

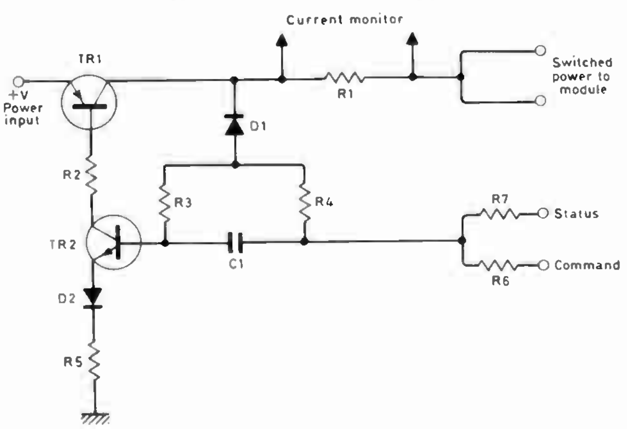

Fig. 6. Positive rail power switch.

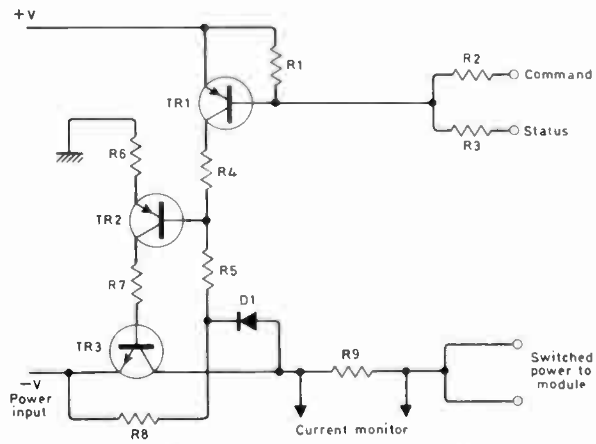

Fig. 7. Negative rail power switch.

positive supply switching is shown in Fig. 6. TR1 was chosen to have the smallest possible V_{CE} drop at the required current such that minimal power is dissipated in this transistor. R1 is the current sensing resistor.

When a 10 V level c.m.o.s. command is applied to R6, the status flag corresponding to this command is shown on the first line of the telemetry frame (1 for ON, 0 for OFF). The command switches on TR2 via C1 and is held on by R3 and R4. TR2 is biased so that, when the current through TR1 increases (and therefore its V_{CE} increases), the diode D1 forces it out of saturation. If the current passing is approximately double the nominal working current of the module being switched, then TR2 will be switched off completely and therefore TR1 will be switched off. If the fault is only short-lived, then the module will switch on again when the fault has cleared, as long as the command is still high. The general form of the negative supply switch is shown in Fig. 7.

When the command is low, TR1 is put into saturation and TR2 is biased OFF by R4, R5 and R8. When the command on R2 goes high, TR1 is switched OFF. The potential divider chain R4, R5 and R8 biases TR2 ON and therefore TR3 switches ON, supplying power to the module. If excess current is drawn by a faulty module, then the voltage across TR3 increases. The diode D1 increases the voltage on the base of TR2, which switches off TR2 and TR3 when the current through TR3 is approximately doubled. All biasing resistors were chosen to have as high a value as possible to decrease energy loss in the circuitry.

8 Construction

The circuit boards were laid out using a Racal-Cadet c.a.d. facility. All low power transistors are metal-can devices to enable greater heat radiation, and all high power transistors are bolted to the aluminium boxes for good heat dissipation.

All high-standing components (greater than $\frac{1}{4}$ in high) were coated in 'Ecosil' to minimize vibration, and the complete board was thoroughly cleaned and then conformally coated to keep it free from contamination.

9 Post-launch Operations

To date, the complete power system on board *UOSAT* is functioning correctly. The battery voltage is stable at approximately 14·4 V, and the solar cells have supplied a current in excess of 0·75 A. All modules switched on so far have consumed the power specified. The temperature tracking of the solar cells is working satisfactorily and an excess of power is available.

10 Acknowledgments

The authors would like to acknowledge the help of Drs K. Meinzer, V. Hass and K. Gladisch from the University of Marburg for designing and building the battery charge regulators and power conditioning modules.

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The *UOSAT* magnetometer experiment

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SUMMARY

Much still remains to be learned about the Earth's magnetic field and, in particular, the variations that occur in it as a result of solar–terrestrial interaction. The magnetometer on board *UOSAT* has a resolution of ±2 nT which is within a factor of two of the most sensitive magnetometer ever put into space. This, coupled with the polar orbit of the satellite and its expected life of several years, should result in a substantial increase in our knowledge of the Earth's field. A detailed description of the instrument and its associated electronics is given.

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1 Objectives

A detailed understanding of the Earth's magnetic field is of fundamental importance in fields such as navigation, guidance, crustal resources exploration and radio propagation studies. The geomagnetic field is generally assumed to be constant and as being generated by a giant dynamo imbedded in the Earth with its 'coil' axis tilted about 11° with respect to the rotational axis and located slightly away from the Earth's centre. To a large degree this simplified concept is accurate, but it is known that the field is not steady and significant disturbances take place due to complex systems of currents flowing in the ionosphere and magnetosphere which surround the Earth. Some of these disturbances, called 'geomagnetic storms' can induce large currents at high latitudes in power transmission lines, telephone circuits, and even the Alaskan pipeline where enhanced corrosion effects are observed due to the currents flowing in the pipe (of the order of 100–1000 A).

The source of these disturbances lies in the complex interaction between the solar wind and the Earth's magnetic field which creates currents that flow in the ionosphere and along magnetic field lines in the auroral regions. These currents can reach values of millions of amperes, although the current density itself is very small. The characteristic time of these disturbances varies from seconds to the period of the solar cycle (11 years), and major magnetic storms occur every few years. Minor 'substorms' occur every few weeks.[1]

The magnetometer experiment aboard *UOSAT*, in addition to providing an effective medium to illustrate and measure the effects of these gigantic currents upon the geomagnetic field, will provide data which can help us understand how solar wind disturbances couple into the magnetosphere and give rise to magnetic storms and the associated high-frequency propagation effects with which radio amateurs are quite familiar. Since these effects are particularly strong over the auroral regions, special emphasis will be placed on the acquisition of real time and memory data over the poles. Additional contributions will be made to the study and mapping of the main geomagnetic field, although due to limited knowledge of absolute spacecraft attitude, position and sensor orientation, these data will tend to be more difficult to analyse. Efforts will be made to correlate the data acquired with that obtained from *MAGSAT* since the resolution of *UOSAT*'s magnetometer is within a factor of two of the very sensitive instrument on board *MAGSAT*.[2] This spacecraft was launched in October 1979 to obtain the most accurate magnetic field data to date, both in magnitude and orientation.

If successful, the analysis effort could yield additional information on the secular variation of the geomagnetic field and the rate of decay of the dipole term which, if the present trend continues, will decay to zero and reverse polarity in approximately 1200 years. Additional correlative studies with spacecraft such as DE and ISEE† will be carried out whenever possible.

The completeness of coverage, data acquisition and reduction will be the determining factors in making these important scientific studies possible. The *UOSAT*

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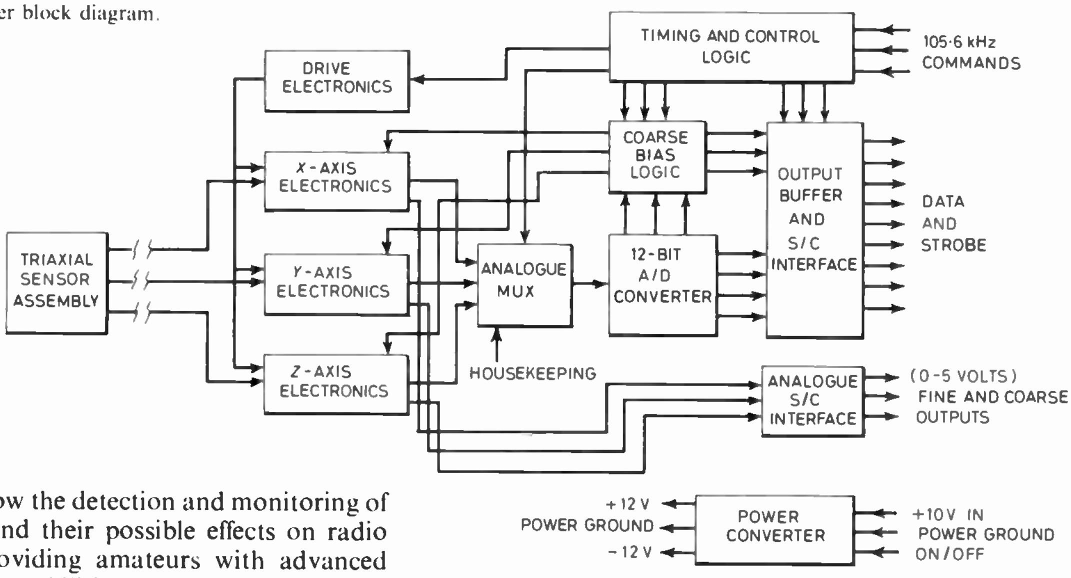
Fig. 1. Vector magnetometer block diagram.



magnetometer will allow the detection and monitoring of geomagnetic storms and their possible effects on radio propagation, thus providing amateurs with advanced diagnostic and study capabilities.

2 Instrumentation

A block diagram of the 3-axis magnetometer is shown in Fig. 1. The heart of the instrument is a ±8000 nT (1·0 nT = 10^{-5} gauss) fluxgate magnetometer whose outputs are digitized by a 12-bit, successive approximation a–d converter. The strength of the geomagnetic field is approximately 30 000 nT at the equator and 60 000 nT at the poles, thus this basic range of 8000 nT must be extended to 64 000 nT.

The schematic diagram for the basic fluxgate magnetometer is shown in Fig. 2. A 105·6 kHz clock from the spacecraft is divided down by 8 to 13·2 kHz and the signal is used as the excitation source for the fluxgate sensors. These sensors are constructed with three orthogonally-mounted 1·2 cm diameter 4·79 Mo Permalloy tape cores around which a diametrically-wound flat coil has been located. (See Fig. 3.) The 13·2 kHz signal is used to drive the cores to saturation with a non-linear resonant system, and the presence of an external field causes a signal at the second harmonic of the excitation frequency to appear at the terminals of the diametrically-wound coil.[3]

This can be intuitively understood if we consider that the voltage induced in the sense coil is always zero unless there is an unbalance in the halves of the toroidal core. This unbalance is introduced by the presence of the external magnetic field. The second harmonic signal amplitude is proportional to the magnitude of the external field component parallel to the coil axis, and its phase with respect to the excitation signal (0° or 180°) depends on the direction of the external field. Thus the fluxgate sensor is analogous to an a.c. Wheatstone bridge except that its output is at the second harmonic of the excitation frequency.[4]

This signal is amplified and applied to a synchronous detector and an operational integrator whose output is used to drive a voltage-controlled current source or transconductance amplifier. The output current is applied to the sensor in such a way as to cancel the applied field. Thus a measure of the current required to cancel the external field as given by E_{out} is a measure of the magnitude and direction of the field itself (parallel to the coil axis). The scale factor from gauss to volts is then

† DE refers to the twin *Dynamic Explorer* spacecraft launched in July 1981 to study the coupling between the solar wind, ionosphere and magnetosphere, while ISEE stands for *International Sun–Earth Explorers*, a complex of three spacecraft launched in 1977 to study solar-terrestrial coupling effects.

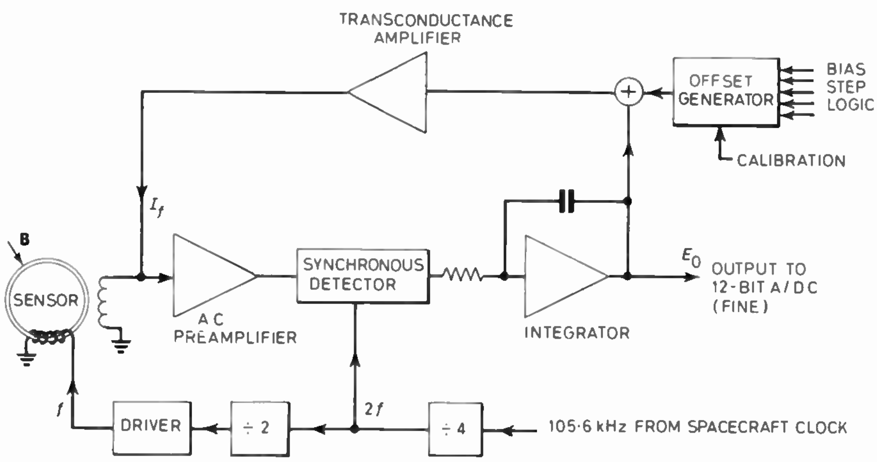


Fig. 2. Basic circuit of sensor for one axis.

given by the characteristics of the sense coil and the transconductance amplifier or voltage controlled current source. The latter has, in addition, a summing input into which we can add positive or negative signals, which will result in the outputs being biased at a particular level. In this case, the dynamic range of the magnetometer remains at ± 8000 nT but the zero level can be biased to any desired point to achieve a differential measurement. This technique is used to extend the basic range of ± 8000 nT to $\pm 64\,000$ nT.

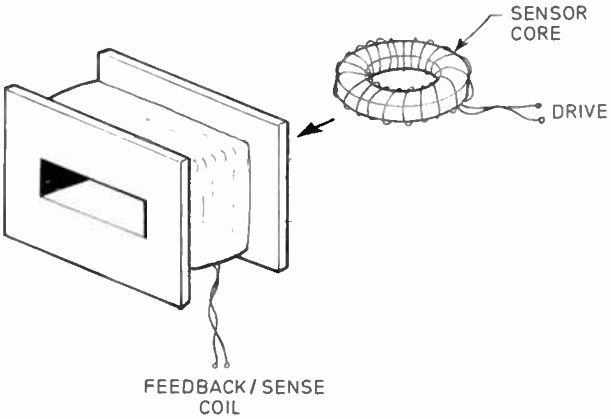


Fig. 3. The ring-core fluxgate sensor.

Three 4-bit digital-to-analogue converters, one for each axis, are used to generate 16 bias steps of 8000 nT each. The digital input to the d–a converters is derived from an up–down counter which is allowed to count up, down or remains in a given state depending on the value of the 12-bit digital output of the a–d converter. This mode of operation results in the composite response characteristics for each axis of the magnetometer shown in Fig. 4. Note that two combinations of 'coarse' and 'fine' counts are possible for each value of the external field, depending on whether the field was increasing or decreasing at the time.

Table 1

UOSAT vector magnetometer—summary of technical characteristics

Basic fluxgate magnetometer	
Dynamic range:	± 8000 nT
Resolution (12-bit a–d):	± 2 nT
Noise (3·13 Hz bandwidth):	<1 nT
Zero level stability:	Sensor (-60 C to $+60$ C): ± 5 nT Electronics (-20 C to $+50$ C): ± 2 nT
Drive frequency:	13·2 kHz
Linearity error:	$<2\times10^{-5}$
Bias field generator	
Dynamic range:	$\pm 64\,000$ nT
Quantization step:	8000 nT
Temperature coefficient:	2 parts/10^6 deg^{-1} C
Sensor assembly	
Mass:	120 g
Dimensions:	$7{\cdot}5\times4\times4$ cm
Electronics	
Mass:	1 kg
Power consumption:	500 mW

The 16 resulting bits are placed in a tri-state elastic buffer and then placed into two bytes for loading onto the data bus. Each time a vector sample is placed in the buffer, the instrument signals the microprocessor that data are ready for collection with a data strobe signal. At 1200 bit/s spacecraft bit rate the effective sampling rate for the instrument is 6·25 vectors per second. A data and strobe timing diagram is shown in Fig. 5. Housekeeping data (temperature and voltages) are multiplexed into the first and second bytes of a measurement sequence in addition to a sync bit to allow decoding on the ground. Table 1 summarizes the principal characteristics of the magnetometer. The power converter accepts $+10$ V from the spacecraft and converts it to $+12$ and -12 V for use in the magnetometer.

In addition to the digital interface, an analogue interface (0–5 V) is provided for the telemetry encoder.

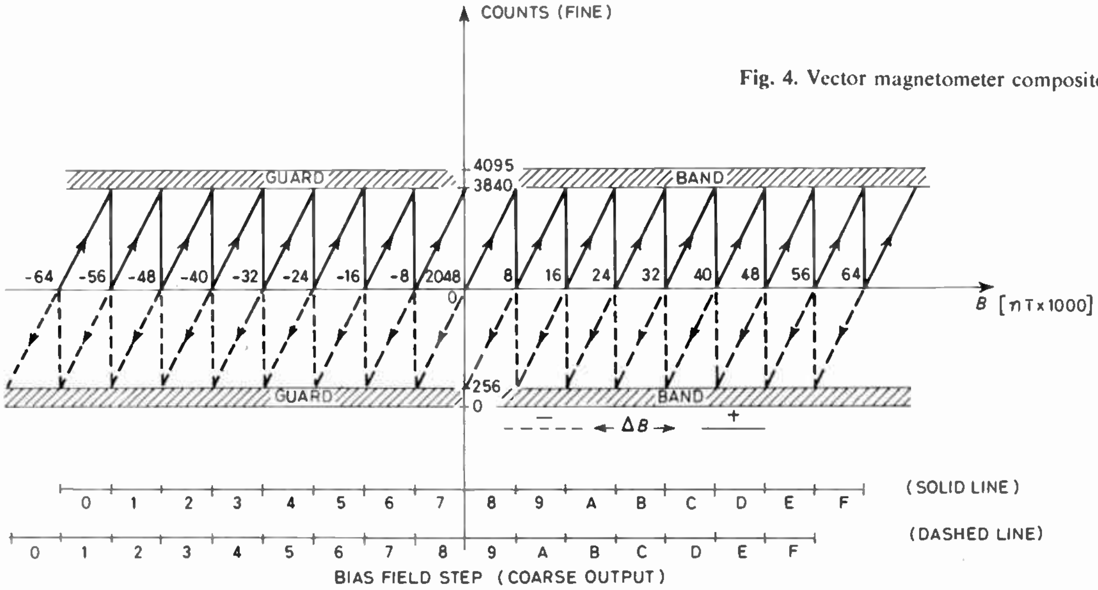


Fig. 4. Vector magnetometer composite response.

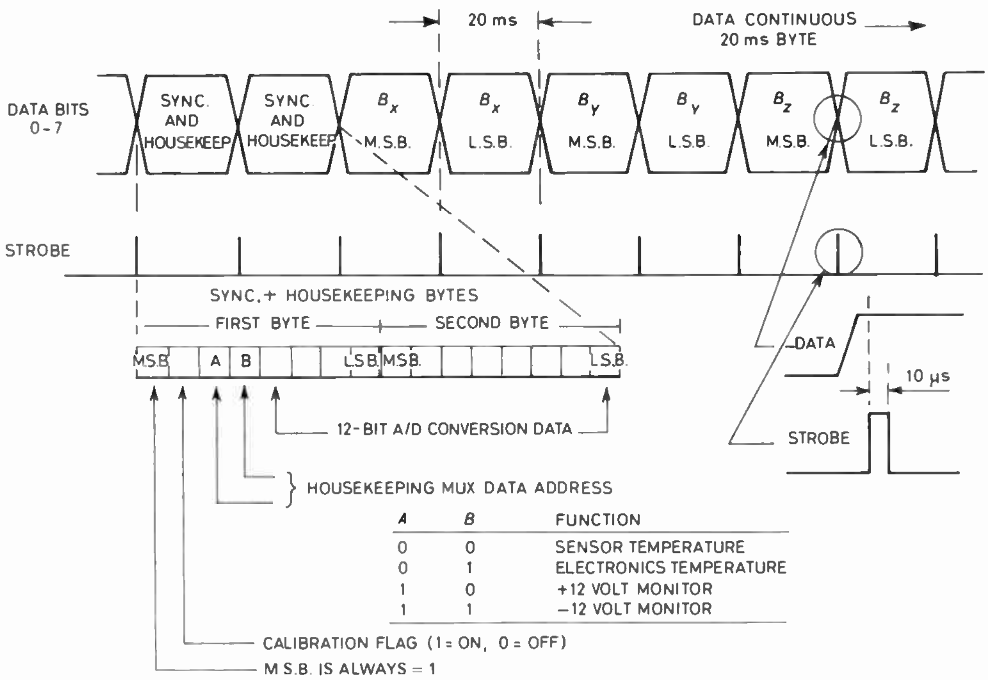


Fig. 5. Vector magnetometer timing diagram.

In this case, the 'coarse' and 'fine' signals are digitized to 10-bit resolution by the spacecraft and the magnetometer maximum resolution is then reduced to ± 8 nT. The sensor and the instrument electronics are shown in Figs 6 and 7.

3 Calibration and Alignment

The magnetometer was calibrated utilizing a triaxial Helmholtz coil system which is used to (a) null the Earth's magnetic field and (b) superimpose known, accurate fields along three orthogonal directions represented by the axes of the coils. The magnetic fields generated by the coils are calibrated with respect to a proton precession magnetometer which is used as a standard. In addition to determining the 'fine' and 'coarse' calibration constants, the angular alignment of each sensor with respect to a reference coordinate system was determined. Since the sensors are not mounted perfectly orthogonal with respect to each other, it is necessary to introduce a small correction to the raw data in the form of a matrix multiplication

$$B_{\text{orth}} = \{M\} B_{\text{meas}}$$

where B_{orth} is the measured field B_{meas} corrected to an orthogonal reference frame by the alignment matrix $\{M\}$.[5]

Since the deviations from orthogonality in the sensors are generally less than 1°, the quick-look data represented by B_{meas} can be used without the correction for coarse attitude measurements and similar applications where knowledge of the field orientation with 1° accuracy is sufficient. When more accurate measurements are required and the magnitude of the magnetic field vector must be known to the limit of absolute accuracy of the instrument, the orthogonality correction should be applied. The values of the elements m_{ij} of the alignment matrix $\{M\}$ are given in Table 2.

The measured magnetic field values for each component can be obtained from the corresponding 'coarse' and 'fine' counts telemetered to ground as follows:

$$B_i = \left(\sum_{1}^{4} A_{ij} CB_j\right) - A_{0i} - K_{fi}(FC_i - Z_i), \quad i = x, y, z$$

where CB_j denotes the 'coarse' bit values (0 or 1) for $j = 1, 2, 3, 4$ and A_{ij} represents the 'weighting coefficients' given in the Table for the ith axis and jth coarse bit. A_{0i} is a constant coefficient for each axis, while K_{fi} denotes the 'fine' scale factor for the fine counts associated with the ith axis, FC_i. The zero field count is denoted by Z_i. The following example clarifies the use of the above expression. Assume that we are reading the X-axis and it shows a coarse reading of D (hex) or 1101

Table 2

Calibration constants

Coefficient	X	Y	Z
A_0	64824	64433	64072
A_1 (m.s.b.)	64822	64432	64072
A_2	32411	32220	32039
A_3	16213	16114	16022
A_4 (l.s.b.)	8131·1	8062·5	8011·4
K_{fi}	4·478627	4·4825	4·43368
Z	2054	2051·2	2042

Alignment Matrix

$$[M] = \begin{matrix} 1{\cdot}000 & 0{\cdot}008685 & -0{\cdot}014653 \\ 0{\cdot}001201 & 1{\cdot}000 & 0{\cdot}000398 \\ -0{\cdot}00090 & -0{\cdot}015134 & 1{\cdot}000 \end{matrix}$$

(binary) and a fine reading of 2856 counts. Looking at the values for the coefficients in the Table we have

$$B_x = (1\times 64\,822)+(1\times 32\,411)+(0\times 16\,213)+ \\ +(1\times 8131{\cdot}1)-64\,824-4{\cdot}478627(2856-2054) \\ = 36\,948{\cdot}2\ \text{nT}.$$

The same formula applies for the analogue outputs from the experiment. The coarse analogue output voltages corresponding to the 16 discrete steps in each axis are given in Table 3. Also given in the Table are the constants for the fine analogue outputs which are digitized by the spacecraft.

The instrument fine calibration can be checked by means of the 'calibration' command which injects an accurately known current into each sensor corresponding to approximately 900 counts (see Table 3). In addition, two temperature monitoring circuits are included and their outputs digitized by the experiment 12-bit a–d converter. To obtain the temperature readings from the appropriate housekeeping multiplexer data the following expression must be used:

$$T\,(^\circ\text{C}) = K_T(T_C-2048)+b$$

where T_C denotes the reading in digital counts and K_T and b are constants given in Table 3. The power converter output voltages of $+12$ V and -12 V are also monitored through the experiment a–d converter and these values can be derived from the corresponding digital counts as

$$V = -7{\cdot}324\times 10^{-3}\ (\text{reading } -2048).$$

The outputs from the magnetometer are bandwidth limited by a 3·13 Hz low-pass filter, which is included to minimize aliasing errors since the maximum sampling rate is 6·25 samples/s. When analysing high-frequency magnetic field fluctuations the dynamic response characteristics of the instrument must be taken into account. This is easily accomplished since the overall transfer function is accurately represented by that of a single pole low-pass filter with a cut-off frequency of 3·13 Hz

$$A(f) = 1/(1+j\omega\tau_1)$$

with $\tau_1 = 5{\cdot}08\times 10^{-2}$ s.

Table 3

Analogue calibration constants

Coarse step	X	Y	Z
F(1111)	939	941	935
E(1110)	875	877	871
D(1101)	812	814	808
C(1100)	749	750	745
B(1011)	686	689	682
A(1010)	623	624	619
9(1001)	560	560	556
8(1000)	496	497	493
7(0111)	433	434	430
6(0110)	370	370	367
5(0101)	307	307	304
4(0100)	244	243	241
3(0011)	181	180	178
2(0010)	117	115	115
1(0001)	54	53	52
0(0000)	0	0	0
K_F	18·671	18·5872	18·3688
Z	494	493	493

Temperature circuit constants

	T_1 (electronics)	T_2 (sensor)
K_T	$4{\cdot}0209\times 10^{-2}$	$4{\cdot}0209\times 10^{-2}$
b	$-5{\cdot}9$	$+9{\cdot}82$

Fine 'calibration' command

	X	Y	Z
Counts	+908	+902	+904

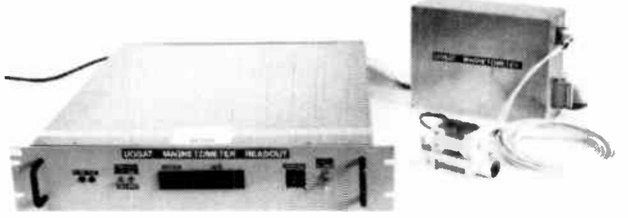


Fig. 6. *UOSAT* magnetometer sensor with ground support check-out equipment.

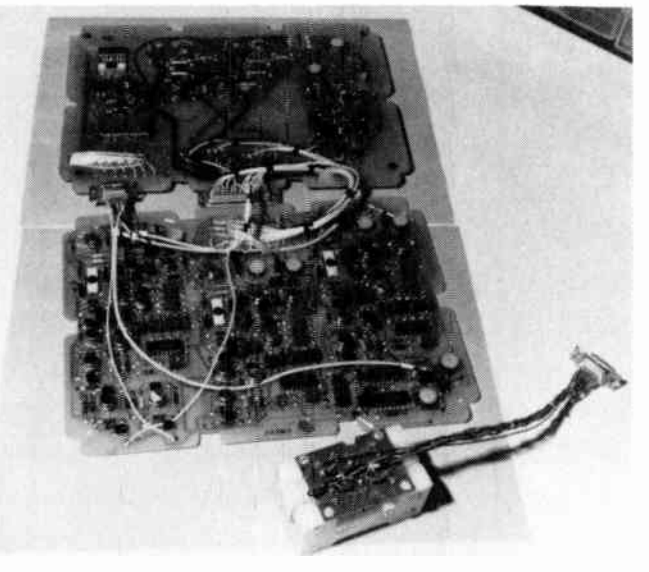


Fig. 7. Magnetometer and instrument electronics.

4 Acknowledgments

The development, implementation and calibration of the *UOSAT* magnetometer would not have been possible without the contributions of Andy Deskur, John Scheifele, Everett Worley and Clell Scearce to whom the author is deeply indebted. The assistance of Jan King and Gordon Hardman was essential to the successful completion of this project and is hereby acknowledged.

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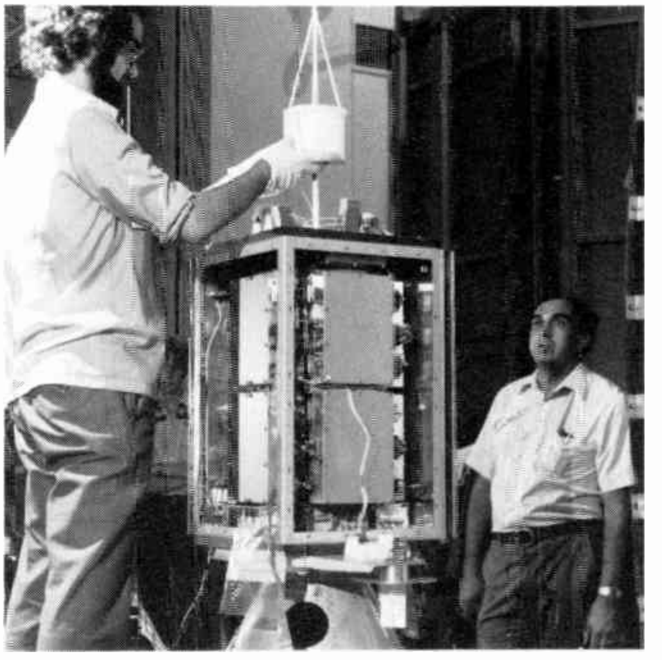

Fig. 8. *UOSAT* spacecraft during integration showing tip mass containing magnetometer sensor during boom deployment test.

UDC 551.510.5: 621.387.424: 629.19

Indexing Terms: Ionosphere measurements, Geiger counters, Satellites

The *UOSAT* radiation-monitor experiment

I. C. FEREBEE

and

D. R. LEPINE, B.Sc.†

SUMMARY

The radiation-monitor experiment for the *UOSAT* spacecraft is described. This instrument employs two G–M tubes with energy thresholds of 20 and 40 keV. The tubes are sampled 5 times each second and the count rate telemetered to ground directly or stored for later transmission. Examples of observations during geomagnetically quiet and disturbed conditions are presented.

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1 Introduction

The aim of this paper is to introduce the *UOSAT* radiation-monitor experiment and to provide some background information regarding the nature of the particle fluxes it is designed to measure. Observations during orbit 1799, 2 February 1982, representing geomagnetically quiet conditions, and orbits 220 and 221, 21 October 1981, representing disturbed conditions, are discussed in order to illustrate the type of results available from this experiment. Because the geometry of the geomagnetic field and the distribution of particles in space and energy are crucial in determining what particles *UOSAT* is expected to measure, it is appropriate to start with a discussion of these.

The fact that the Sun has an expanding, highly-conducting outer atmosphere, known as the solar wind, causes the dipole-like magnetic field generated by currents within the Earth to be considerably distorted at geocentric distances greater than about 6 R_e (R_e = Earth radii), and also leads to the formation of a magnetosphere.[1] In elementary terms the magnetosphere can be thought of as a comet-shaped cavity, which is devoid of solar wind particles, and carved out of the solar wind by the geomagnetic field. At the boundary of the magnetosphere is a region known as the magnetopause;[2] within the magnetopause currents flow which prevent the geomagnetic field from penetrating further into the solar wind. Inside the magnetosphere the original dipole-like field is distorted. The effects is to compress field lines together on the dayside and stretch them out to large distances to form a tail on the nightside.

Although there is no simple direct entry for solar wind particles, inside the magnetosphere exist several particle populations, very largely collisionless and often far from thermodynamic equilibrium. Nearest the Earth, in a region known as the plasmasphere, are found co-rotating particles of typical energies less than 1 eV, which at lower altitudes are in equilibrium with the ionosphere. The ionosphere is distinguished from the other plasmas on the grounds that collisions are important. As well as the low-energy co-rotating population, the magnetosphere contains energetic particles with energies ranging from a keV to 100's of MeV. In most regions the Larmor orbit of these energetic particles about the geomagnetic field is much smaller than the gross features of the field. These particles are therefore trapped, and they gyrate, bounce, and drift in well defined orbits. It is the trapped particles that form the doughnut-shaped van Allen radiation belts[3] and the plasma sheet,[4] a flattened extension of the belts which extends into the nightside magnetosphere.

At geomagnetically active times, plasma sheet particles become detrapped and enter the atmosphere to produce the aurora[5]. Entry occurs most frequently at geomagnetic latitudes of 67° in a region called the auroral oval.[6,7] During active times, electric fields are set up and plasma is accelerated at the expense of the magnetic-energy stored in the geomagnetic tail. Two degrees of activity are recognized: during substorms[8] currents flow in the auroral zone ionosphere which

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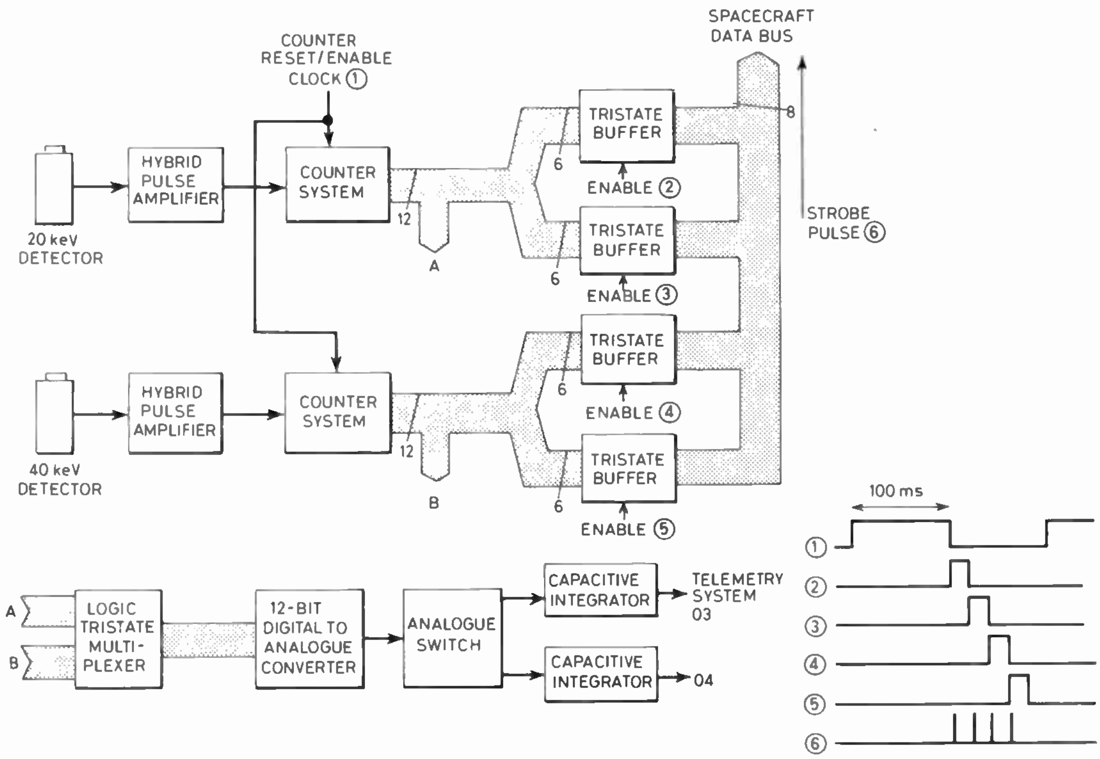

Fig. 1. Block diagram of the radiation-monitor experiment on-board data processing. When the experiment is on, outputs from the capacitive integrators are telemetered continuously using channels 03 and 04. Stored data are telemetered to ground on request.

produce significant perturbations in the geomagnetic field observed at auroral latitudes; during storms[9] the low-latitude magnetic field is also depressed, indicating the growth of a ring current that encircles the Earth. The growth and decay of the ring current is associated with precipitation in the high-latitude ionosphere.

2 Instrumentation

The Geiger–Müller tubes measure fluxes of electrons above threshold energies of 20 and 40 keV. The tubes are mounted outside the spacecraft on the top honeycomb-plate and are inclined at angles of 13° and 18° to the spacecraft $+z$ axis. The tubes have thin mica end-windows for particle entry and contain neon plus a small quantity of halogen to provide quenching. A collimator consisting of two circular apertures separated by an 8 mm spacer is used to define the effective collecting area and angular response of each of the tubes. In addition to counting electrons the tubes detect protons of approximately twenty times the electron threshold energy. Further details of the detectors are given in Table 1.

A block diagram of the on-board data processing is shown in Fig. 1. Pulses from each detector are counted in a 12-bit ripple counter which feeds both a capacitive integrator and the on-board computer memory. Integrator output is telemetered to ground directly using channels 03 and 04, (detectors A and B respectively); stored data are telemetered to ground on request.

Table 1 G–M detector characteristics

Detector	A	B
Approximate energy threshold	20 keV	40 keV
Telemetry channel (direct data)	03	04
Factory used to obtain intensity ($m^{-2} s^{-1} sr^{-1}$) from telemetry value N	$5{\cdot}2 \times 10^8$	$1{\cdot}2 \times 10^8$
Tube type	LND705	LND710
Window thickness	$0{\cdot}34 \pm 0{\cdot}05$ mg cm^{-2}	$1{\cdot}75 \pm 0{\cdot}25$ mg cm^{-2}
Geometric factor	0·08 mm^2 sr	0·35 mm^2 sr
Inclination to spacecraft $+z$ axis	13	18
Anode voltage	640V	640V
Identifier for computer interface	00 (l.s.b.) 01 (m.s.b.)	10 (l.s.b.) 11 (m.s.b.)

3 Results

3.1 Geomagnetically Quiet Conditions

Intensities measured by detector A during part of orbit 1799, 2 February 1982, are shown in Fig. 2. The observations are an example of afternoon sector fluxes measured during geomagnetically quiet conditions. It is clear that there is a gradual increase in intensities as the satellite moves from lower (30°) to higher (65°) geographic latitudes. Although Fig. 2 alone cannot indicate if the observed increase is spatial or temporal,

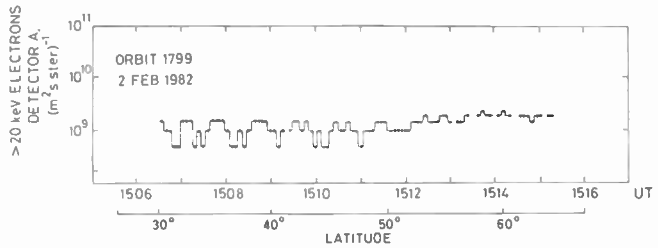

Fig. 2. Intensities measured by detector A (>20 keV electrons) during part of orbit 1799 provide an example of afternoon sector fluxes observed during geomagnetically quiet conditions. The modulation of period ~52 s at latitudes less than 50 is produced by beating (see text).

evidence from similar orbits confirms that the increase is a spatial effect. There is also a suggestion in Fig. 2 that prior to 1512 UT when the satellite was at latitudes of less than 50°, the intensities were modulated with a 52 s period. A closer inspection shows that this modulation represents beating, produced by the anisotropy of the particle distribution and the changing inclination of the detector relative to the geomagnetic field.

3.2 Disturbed Conditions

The change in the horizontal component of the geomagnetic field measured at low latitudes during the interval 19 to 26 October 1981 is shown in Fig. 3. The large decreases beginning at midday on 20 October and early morning on 22 October are associated with magnetic storms, and signify a build-up of the ring current. Taking this current to be located in the equatorial plane at a geocentric distance of 4·5 R_e, a current of ~9 MA is required to produce a perturbation of −200 nT at the equator. Electron precipitation associated with the 20 October storm and measured in the morning sector during orbits 220 and 221 on 21 October 1981 is shown in Fig. 4. Peak fluxes during the first of these orbits exceed the quiet time fluxes of orbit 1799 by nearly two orders of magnitude. From observations with this satellite alone we cannot tell if the variations are spatial or temporal. However, the fact that a systematic variation with latitude is seen on two successive orbits serves to indicate that the changes are spatial.

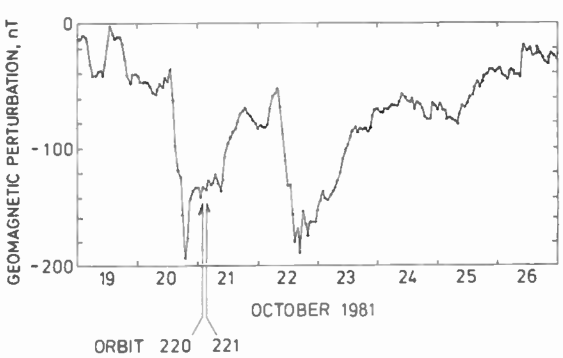

Fig. 3. Perturbations in the average magnetic field observed at the equator, and showing the geomagnetic storms on 20 and 22 October 1981. The weakening of the field (negative perturbation) is associated with the growth of an equatorial ring current.

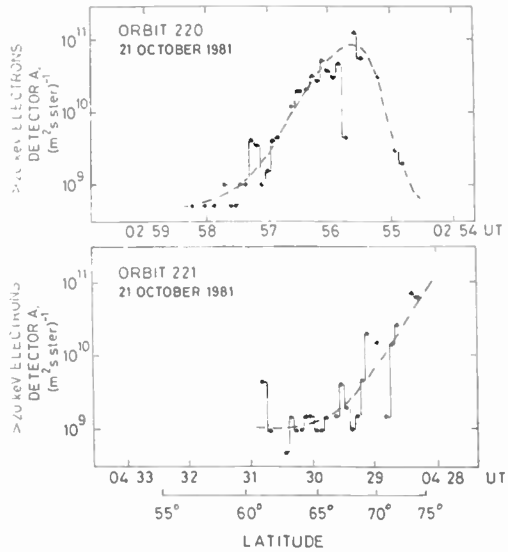

Fig. 4. High intensities measured by detector A during orbits 220 and 221 on 21 October 1981 (morning sector) illustrate the precipitation associated with the geomagnetic storm of 20 October (see Fig. 3).

4 Conclusions

In this paper we have introduced the *UOSAT* radiation-monitor experiment, provided some background information regarding the nature of the particle fluxes it is expected to measure, and presented some results demonstrating geomagnetically quiet and disturbed conditions. While the data presented here cover too small a time interval to permit any general statement regarding the nature of the particle fluxes, they nevertheless illustrate some characteristics of the fluxes which can be investigated by this experiment.

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(Paper No. 2048/MI 27)

Contributors to this issue

Mario Acuna received his undergraduate degree from the University of Cordoba, Argentina, the M.S.E.E. degree from the University of Tucuman, Argentina, in 1967, and the Ph.D. in space science from The Catholic University of America in 1974. In 1969 he joined the Goddard Space Flight Center, where his main research interests have been instrumentation for geophysical and space research and studies of magnetic fields and plasmas in interplanetary space and in magnetospheres. Dr Acuna has been involved with *Explorers 47* and *50*, *Mariner 10*, *Pioner 11*, *Voyagers 1* and *2*, *MAGSAT*, the International Solar Polar Mission, *Project Firewheel*, and *AMPTE* as instrument engineer, principal investigator or engineer, or project scientist. He has received many awards in recognition of his contributions to NASA programmes.

Michael Blewett received a diploma in electrical engineering from Brighton Polytechnic in 1964. He then joined the English Electric Valve Company as a test gear engineer, producing test equipment for microwave valves. In 1973 he moved to the University of Surrey, joining an electronic services group producing a wide range of equipment for teaching and research. He has recently received an honours degree from the Open University.

Anthony Brown graduated from the University of East Anglia in 1974 and joined Marconi Research Laboratories' Antenna Group where he was employed on a number of projects, mainly in the satellite communications field. In 1976 he went to Standard Telecommunications Laboratories, where he was primarily involved in the design of high performance array antennas for radar use; as a Senior Research Engineer his interests covered the v.h.f. to millimetric wave frequency range. In December 1980 Mr Brown became the Racal Research Fellow at the University of Surrey and his major work has been on microstrip and adaptive antennas. He became involved with the *UOSAT* project shortly after joining the University, and has contributed both to its design and operation.

Richard Clarke received an honours degree in 1977 from Queen Mary College, London, since when he has worked at the University of Surrey on a variety of designs including picoampere instrumentation, a.m. transmitters, e.p.r.o.m. programmers, a teletext receiver, a magnetic tape data store, Z80 and 6800 development systems and microprocessor controllers for dishwashers, ion-implanters and, currently, antenna positioning.

Ian Ferebee obtained an HND in applied physics at Kingston Polytechnic in 1975. He worked as an electronics technician at the Polytechnic for five years and joined the *UOSAT* team at the University of Surrey two years ago as the spacecraft technician.

Bob Haining is a graduate of London University, and after a graduate apprenticeship with EMI Electronics he joined the Weapons Division Study Group; subsequently he designed oceanographic telemetry equipment with Elliott Bros. In 1970 he went to the University of Surrey where his current interests lie in unusual applications of radio techniques, and in particular the design of communications equipment for use in tunnels and mines.

Christopher Haynes obtained his B.Sc. degree from the University of Birmingham in 1970 while being sponsored by EMI Electronics as a student apprentice; in 1971 he was selected as EMI's Graduate Apprentice of the Year and he subsequently returned to Birmingham to undertake the Bosworth M.Sc. course in Radio Communications and Radar Technology. During his time with EMI Mr Haynes was responsible for various systems projects in the fields of radar, infra-red and associated technologies. From 1976 to 1979 he was rapporteur of a NATO AGARD working group on the processing of airborne reconnaissance data. In 1979 he was appointed to a lectureship at the University of Surrey, where his research interests include multi-processor computing systems and robotics.

A graduate from Pembroke College, Cambridge, **Michael Hodgart** joined the General Electric Company at Stanmore, Middlesex in 1964 (now Marconi Space and Defence Systems), and worked in microwaves and systems design. He went to the Department of Electrical and Electronic Engineering of the University of Surrey in 1970, and has worked on opto-electronic communication, error correcting codes, system design for improved measurement in ion beam location and in deep-level transient spectroscopy.

Tony Jeans graduated from Cambridge University in 1967, and took an M.Sc. in electronics at Southampton University in 1968. For the next three years he was with the Guided Weapons division of Marconi Space and Defence Systems at Stanmore. In 1971 he joined the Department of Electronic and Electrical Engineering at the University of Surrey as a lecturer in Telecommunications. He has been involved in work on optical communications and microprocessor applications in recent years.

Luigi Mansi (Member 1964) joined the Fairey Aviation Company's Weapons Division in 1955. Following part-time study at the Northern Polytechnic, he obtained the degree of

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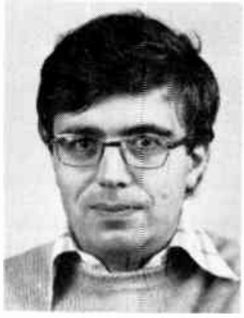

M. S. HODGART

T. J. JEANS

L. S. A. MANSI

B.Sc. (Eng.) of the University of London in 1961, and in 1962 he was appointed as a lecturer at the (now) University of Surrey. He carried out research in hybrid methods of root locus computation and was awarded a Ph.D. in 1971. Since then Dr Mansi has been involved in several industrial research projects and has worked on many aspects of signal conditioning.

Jerzy Slowikowski graduated from the University of Sheffield in 1979 with a B.Eng. degree. He spent 18 months in postgraduate training at Sperry Gyroscope before joining the *UOSAT* project team.

Colin Smithers became a member of the Radio Society of Great Britain in 1970 and gained his amateur transmitting licence in 1974 at the age of 15. In 1981 he graduated from the University of Surrey with a B.Sc. (Hons) degree, and was awarded the Alan Oxford Prize established by the Plessey Company for the final year student with the best performance in Telecommunications. During his industrial year he was sponsored by and worked with Mullard Mitcham, working on miniature power supplies for image intensifiers and frequency synthesisers. The beacon transmitter work was begun in the Application Laboratories at Mitcham and completed as project work during his final year. Mr Smithers is currently studying for a Ph.D. at the University of Surrey in the field of r.f. amplifier linearity, sponsored by MEL.

Martin Sweeting received a B.Sc. (Hons) degree in electronic and electrical engineering from the University of Surrey in 1974 and a Ph.D. for the study of communications efficiency of electrically short h.f. aerials in 1979. He worked for a time with Marconi Space and Defence Systems at Stanmore on underwater weapons and later on the *MARECS* spacecraft. He became interested in the application of low-cost engineering techniques to space communications in 1975 and established a tracking, telemetry and telecommand station at the University of Surrey for the Amateur Radio Communications spacecraft *OSCAR 6, 7* and *8*. This station was expanded to receive and display meteorological image data from the American *TIROS* and *NOAA* series and Soviet *Meteor* spacecraft. Dr Sweeting conceived the *UOSAT* project in 1978/79 and spent a year gathering support, resources and preparing preliminary proposals. Following the successful launch of *UOSAT*, he has been involved in the post-launch orbital operation and performance analysis of the spacecraft whilst studying the relevance of proposals for future low cost missions. He has been a radio amateur since 1969 and an active member of AMSAT since 1974. He has recently been appointed to a Lectureship.

Paul Traynar studied electronics at the University of Southampton and received his B.Sc. degree in 1974. He remained at Southampton to do research in the Microelectronics Group on the applications of charge-coupled devices to radar signal processing. He was awarded the degree of Ph.D. in 1978 and continued research into c.c.d. techniques and processing technology as a Research Fellow. At the end of 1978 Dr Traynar joined Plessey Semiconductors, Swindon to design m.o.s. l.s.i. circuits for television digital timing systems. In 1981 he took up his present post as Research Fellow at the University of Surrey to develop a very low light level c.c.d. imaging system for astronomical applications. Recently he was appointed to a Lectureship.

Mike Underhill (Fellow 1982, Graduate 1963) graduated in physics at Oxford in 1960 and joined the then Mullard Research Laboratories. He started work first on ultrasonics, then moved on to magnetic logic, control of focused electron beams for microcircuit manufacture, and radio techniques and systems. In 1972 he obtained his Ph.D. as a collaborative student at the University of Surrey for the work on control of electron beams. From 1968 Dr Underhill has been a visiting lecturer on the Systems Engineering M.Sc. course in the Department of Electronic and Electrical Engineering at the University of Surrey, and until now has been lecturing on sampled data systems. In March 1982 Dr Underhill became Head of the Systems Division in the Philips Research Laboratories.

J. Z. SLOWIKOWSKI

C. R. SMITHERS

M. N. SWEETING

C. P. TRAYNER

M. J. UNDERHILL

Members' Appointments

BIRTHDAY HONOURS

The President and Council have congratulated the following members of the Institution whose names appear in Her Majesty's 1982 Birthday Honours List.

MOST EXCELLENT ORDER OF THE BRITISH EMPIRE

To be an Ordinary Commander of the Order (C.B.E.)

Francis Kenneth Chorley (Fellow 1979, Member 1956, Graduate 1951) Deputy Chairman and Managing Director, Plessey Electronic Systems Ltd.

To be an Ordinary Officer of the Order (O.B.E.)

John Elfed Jones, B.Sc., (Fellow 1964, Member 1950, Graduate 1944) Principal, Tameside College of Technology, Ashton-under-Lyne 1964 to 1981; formerly a member of the IERE Examinations Committee, an examiner and a member of the South Wales Section Committee.

To be an Ordinary Member of the Order (M.B.E.)

Frederick John Stone, M.Sc., (Member 1973, Associate 1955) Chief Safety Officer, Hatfield Division, Dynamics Group, British Aerospace plc.

CORPORATE MEMBERS

E. C. Buckland (Fellow 1971, Associate 1952) is now Managing Director of B S Instruments, Rustington, West Sussex. He was previously Engineering Director of KDG Instruments of Crawley. Mr Buckland was a member of the Organizing Committee for the Conference on The Influence of Microelectronics on Measurements and Instruments.

P. J. Gallagher, M.Sc., Ph.D., (Fellow 1980, Member 1969, Graduate 1966) is to take up the appointment of Principal of Bradford and Ilkley College in September. Since 1976 he has held the post of Head of the School of Technology and Design at the College which he joined in 1964 as a Senior Technician when it was Bradford Technical College. He is a member of the Education and Training Committee.

J. R. Halsall (Fellow 1979, Member 1954, Graduate 1949) has been appointed a Senior Research Fellow in the Department of Computer Science at the University of Warwick and is Deputy to the Head of the Robot Research Group. Mr Halsall retired from ICI last year after some 30 years with the company, latterly as Group Manager (Cybernetics) at the Corporate Laboratory. He has served on the Council and the Papers Committee and is at present a member of the Professional Activities Committee.

H. Silbermann (Fellow 1962, Member 1947) has recently been awarded a Ph.D. degree by the University of Witwatersrand, Johannesburg, for research work in the field of ferro-resonant devices. Chief Engineer for Philips in South Africa from 1946 to 1948, Dr Silbermann was Chairman of Associated Rectifier and Electronics Pty Ltd, Johannesburg up to his retirement in 1976.

Flt Lt P. M. Eckert, B.Sc., RAF (Member 1968, Graduate 1972) has completed an M.Sc. of the University of Loughborough in Aerosystems Engineering following two years study at the Royal Air Force College at Cranwell, and has been posted to RAF Aberporth.

D. J. Glyde (Member 1975, Graduate 1964) who has been with Westinghouse Signals since 1960, latterly as Chief Engineer of the Rapid Transit and Systems Department, has been appointed Technical Director to the Board.

R. G. W. Maltby (Member 1973, Graduate 1969) has taken up the post of Technical Manager, Racal Training Services, Reading. He had previously held the post of Senior Customer Training Engineer with Racal-Tacticom since 1976 when he retired from REME.

Standard Frequency and Time Service

Communication from the National Physical Laboratory

Relative Phase Readings in Microseconds NPL—Station

(Readings at 1500 UTC)

MAY 1982	MSF 60 kHz	GBR 16 kHz	Droitwich 200 kHz	JUNE 1982	MSF 60 kHz	GBR 16 kHz	Droitwich 200 kHz
1	−10·6	33·8	51·7	1	−11·8	33·5	43·1
2	−10·8	33·8	51·3	2	−11·8	33·8	42·5
3	−10·7	33·7	51·1	3	−11·6	34·0	42·2
4	−10·7	34·4	50·9	4	−11·9	34·7	42·0
5	−10·8	33·9	50·9	5	−12·1	34·5	41·7
6	−10·8	33·8	50·9	6	−12·0	34·3	41·3
7	−10·8	33·9	50·7	7	−12·1	33·0	40·7
8	−10·8	33·8	50·5	8	−12·1	33·1	40·0
9	−10·9	33·7	50·1	9	−12·1	33·5	39·4
10	−10·9	33·7	49·9	10	−12·0	33·3	38·8
11	−10·9	33·4	49·6	11	−12·0	34·0	38·4
12	−10·9	33·5	49·3	12	−11·7	33·8	38·2
13	−10·9	33·8	49·0	13	−11·7	33·8	38·0
14	−11·1	33·7	48·7	14	−11·9	35·4	37·8
15	−11·2	34·0	48·2	15	−11·9	35·0	37·5
16	−11·2	33·5	47·7	16	−12·0	33·5	37·2
17	−11·1	33·3	47·4	17	−11·9	33·6	37·0
18	−11·1	34·0	46·8	18	−11·7	33·8	36·7
19	−11·0	33·9	46·5	19	−11·7	33·8	36·4
20	−11·1	35·5	46·2	20	−11·9	33·8	36·2
21	−11·1	35·3	46·0	21	−11·7	33·9	36·1
22	−11·1	33·5	45·7	22	−11·9	35·7	35·9
23	−11·0	33·7	45·5	23	−11·9	33·6	35·6
24	−11·1	33·7	45·2	24	−11·9	33·5	35·4
25	−11·2	33·5	45·0	25	−11·9	33·4	35·2
26	−11·5	34·0	44·8	26	−11·8	33·6	35·0
27	−11·5	33·9	44·5	27	−11·8	33·7	34·8
28	−11·5	35·3	44·3	28	−11·9	33·9	34·6
29	−11·6	34·2	44·0	29	−11·9	33·5	34·4
30	−11·7	34·2	43.8	30	−12·1	33·4	34·1
31	−11·8	34·0	43·5				

Notes: (a) Relative to UTC scale (UTC_{NPL}–Station) = +10 at 1500 UT, 1st January 1977.
(b) The convention followed is that a decrease in phase reading represents an increase in frequency.
(c) 1 μs represents a frequency change of 1 part in 10^{11} per day.
(d) It may be assumed that the satellite stations on 200 kHz at Westerglen and Burghead will follow the day to day changes in these phase values.

Conferences, Courses and Exhibitions, 1982-83

The date and page references in italics at the end of an item are to issues of *The Radio and Electronic Engineer (REE)* or *The Electronics Engineer (EE)* in which fuller notices have been published.
The symbol ★ indicates that the IERE has organized the event.
The symbol ● indicates that the IERE is a participating body.
An asterisk * indicates a new item or information which has been amended since the previous issue.

Further information should be obtained from the addresses given.

SEPTEMBER

***Aviation and Electronics** *6th to 12th September FARNBOROUGH*
International Exhibition and Flying Display, organized by the Society of British Aerospace Companies, at Farnborough, Hampshire. Information: SBAC, 29 King Street, St James's, London SW1Y 6RD. (Tel. 01-839 3231)

Microwaves *6th to 10th September HELSINKI*
Twelfth European Microwave Conference organized by the IEEE in association with URSI to be held in Helsinki. Information: IEEE, Conference Co-ordination, 345 East 47th Street, New York, NY 10017.

BA '82 *6th to 10th September LIVERPOOL*
Annual Meeting of the British Association for the Advancement of Science, will be held at the University of Liverpool. Information: British Association, Fortress House, 23 Savile Row, London W1X 1AB.

ICCC '82 *7th to 10th September LONDON*
Sixth International Conference on Computer Communication, sponsored by the International Council for Computer Communication, to be held at the Barbican Centre, London. ICCC '82 PO Box 23, Northwood Hills HA6 1TT, Middlesex.

Personal Computer *9th to 11th September LONDON*
Personal Computer World Show, to be held at the Cunard Hotel, Hammersmith, London W6. Information: Interbuild Exhibitions Ltd, 11 Manchester Square, London W1M 5AB. (Tel. 01-486 1951).

Remotely Piloted Vehicles *13th to 15th September BRISTOL*
Third Bristol International Conference on Remotely Piloted Vehicles jointly sponsored by The Royal Aeronautical Society and the University of Bristol. To be held at the University of Bristol. Information: Dr R. T. Moses, Organizing Secretary, RPV Conference, Faculty of Engineering, Queen's Building, The University, Bristol BS8 1TR. (Tel. (0272) 24161, ext 846)

***Safety** *14th to 16th September MANCHESTER*
A three-day course on Safety of Electrical Instrumentation in Potentially Explosive Atmospheres, organized by SIRA Institute, will be held at UMIST, Manchester. Information: Conferences and Courses Unit, SIRA Institute Ltd, South Hill, Chislehurst, Kent BR7 5EH (Tel. 01-467 2636)

Transcon '82 *15th to 16th September*
A Symposium and Exhibition, organized by the Teesside Section of the Institute of Measurement and Control. Information: Dr P. R. Bunn, Department of Electrical Instrumentation and Control Engineering, Teesside Polytechnic, Borough Road, Middlesbrough, Cleveland.

***Fibre Optics** *15th to 17th September MANCHESTER*
Conference on Fibre Optics for Instrumentation, organized by the SIRA Institute, will be held at UMIST, Manchester. Information: Conference and Courses Unit, Sira Institute Ltd, South Hill, Chislehurst, Kent

●**Broadcasting** *18th to 21st September BRIGHTON*
The ninth International Broadcasting Convention, IBC '82, organized by the IEE, and EEA with the association of IERE, IEEE, RTS and SMPTE, will be held at the Metropole Conference and Exhibition Centre, Brighton. Information: IEE, 2 Savoy Place, London WC2R 0BL (Tel. 01-240 1871).

> *Throughout the International Broadcasting Convention, the IERE Publications Sales Department will have a stand in the Exhibition area at which Institution publications can be examined and orders placed.*

Non-Destructive Testing *20th to 22nd September YORK*
National Non-Destructive Testing Conference, organized by the British Institute of Non-Destructive Testing, to be held in York. Information: BInst NDT, 1 Spencer Parade, Northampton NN1 5AA. (Tel. (0604) 30124/5).

★ **Electromagnetic Compatibility** *21st to 23rd September GUILDFORD*
Third conference on Electromagnetic Compatibility, organized by the IERE with the association of the IEE, IEEE, IQA and RAeS, to be held at the University of Surrey, Guildford. Information: Conference Secretariat, IERE, 99 Gower Street, London WC1E 6AZ (Tel. 01-388 3071)

***Automatic Testing** *21st to 23rd September PARIS*
Exhibition and Conference on ATE and Test Systems will be held at C.I.P., Paris. Information: Network, Printers Mews, Market Hill, Buckingham. Tel. (02802) 5226/5227

Telecommunications and Fibre Optics *21st to 24th September CANNES*
Eighth European conference on Telecommunication and Fibre Optics organized by the Electronics Industries Group (GIEL), to be held in Cannes. Information: GIEL 11 rue Hamelin, 75783 Paris Cedex 16

Automated Assembly *23rd September LONDON*
One day Seminar on 'Automated Assembling: Starting a Project and making it work' organized by the Institution of Production Engineers to be held at the Bowater Conference Centre, London. Information: The Manager, Conferences & Exhibitions, Institution of Production Engineers, Rochester House, 66 Little Ealing Lane, London W5 4XX. Tel. 01-579 9411

Man-Machine Systems *27th to 29th September BADEN-BADEN*
Conference on Analysis, Design and Evaluation of Man-Machine Systems sponsored by IFAC in association with the IFIP/IFORS/IEA, to be held in Baden-Baden, Federal Republic of Germany. Information: VDI/VDE-Gesellschaft, Mess-und Regelungstechnik, Postfach 1139, D-4000 Dusseldorf 1. (Tel. (0211) 6214215)

***Reliability Engineering** *27th September to 1st October ROTTERDAM*
International CBO Seminar on Reliability Engineering—Advanced Technology and Industrial Application will be held in Rotterdam. Information: CBO—Management and Technology Systems Centre, P.O. Box 30042—Exchange Building, 3001 DA Rotterdam, Netherlands. (Tel. (010) 13 90 20).

Instrumentation in Flammable Atmospheres *30th September LUTON*
A short course on Instrumentation in Flammable Atmospheres, organized by Measurement Technology to be held in Luton Information: Customer Training Department, Measurement Technology Ltd, Power Court, Luton LU1 3JJ. (Tel. (0582) 23633).

Non-Ionising Radiations *30th September LONDON*
A one-day conference on Current Views on the Hazards of Non-ionising Radiations, organized by the University of London, will be held at the University of London Senate House. Information: The Safety Adviser, London University, 15 Woburn Square, London WC1H 0NS. (Tel. 01-580 1752).

OCTOBER

Electrical *5th October LONDON*
One-day Seminar on Electrical Equipment for Use in Zone 2 Areas: Design, Selection and Installation, organized by Sira Institute, will be held at the Wembley Conference Centre. Information: Conference Unit, Sira Institute, South Hill, Chislehurst, Kent BR7 5EH. (Tel. 01-467 2636).

Electronic Displays *5th to 7th October LONDON*
Electronics Displays Exhibition and Conference, to be held at the Kensington Exhibition Centre. Information: Network, Printers Mews, Market Hill, Buckingham. MK18 1JX. (Tel: (0282) 5226).

Defendory Expo '82 *11th to 15th October ATHENS*
The 4th Exhibition for Defence Systems and Equipment for Land, Sea & Air, organized by the Institute of Industrial Exhibitions in association with the Defence Industries Directorate of The Hellenic Ministry of National Defence to be held in Athens, Greece. Information: Mrs Duda Carr, Westbourne Marketing Services, Crown House, Morden, Surrey SM4 5EB (Tel. 01-540 1101)

Internepcon *12th to 14th October BRIGHTON*
Internepcon Conference and Exhibition, organized by Cahners Exposition Group, to be held at the Metropole Exhibition Hall, Brighton. Information: Cahners Exposition Group, Cavridy House, Ladyhead, Guildford, Surrey, GU1 1BZ. (Tel. (0483) 38083).

***Optoelectronics** *12th to 14th October LAUSANNE*
Conference on Optoelectronics in Telecommunication and Measurements Systems to be held in Lausanne, Switzerland. Information: Secretariat, Journees d'Electronique, EPFL, Ch. de Bellerive 16, CH-1007 Lausanne, Switzerland

CAMPRO '82 *13th and 14th October LONDON*
Conference on Computer Aided Manufacturing and Productivity organized by Institution of Production Engineers and the Society of Manufacturing Engineers USA, will be held at the Mount Royal Hotel, Marble Arch, London. Information: The Manager, Conferences & Exhibitions, Institution of Production Engineers, Rochester House, 66 Little Ealing Lane, London W5 4XX. Tel. 01-579 9411

● **RADAR '82** *18th to 20th October LONDON*
International Conference on Radar, organized by the IEE in association with the IEEE EUREL, IERE, IMA, RAeS and RIN, to be held at the Royal Borough of Kensington and Chelsea Town Hall, Hornton Street, London W8. Information: IEE Conference Department, Savoy Place, London WC2R 0BL. (Tel. 01-240 1871).

● **Military Microwaves '82** *19th to 22nd October LONDON*
Third International Conference and Exhibition organized by Microwave Exhibitions and Publishers, to be held at The Cunard International Hotel. Information: Military Microwaves '82 Conference, Temple House, 36 High Street, Sevenoaks, Kent TN13 1JG

Electronics *20th to 22nd October HONG KONG*
Second Hong Kong Electronics Fair, sponsored by the Hong Kong Exporters' Association and the Hong Kong Electronics Association, will be held at the Miramar Hotel, Kowloon. Information: Secretary, Hong Kong Electronics Fair Committee, c/o Hong Kong Exporters Association, 1625 Star House, 3 Salisbury Road, Kowloon, Hong Kong.

Multivariable Systems *26th to 28th October PLYMOUTH*
Symposium on the Application of Multivariable Systems Theory, organized by the Institute of Measurement and Control to be held at the Royal Naval Engineering College, Manadon. Information: The Institute of Measurement and Control, 20 Peel Street, London W8 7PD. (Tel. 01-727 0083).

Instrumentation *26th to 28th October LONDON*
Electronic Test & Measuring Instrumentation Exhibition and Conference, to be held at the Wembley Conference Centre. Information: Trident International Exhibitions Ltd, 21 Plymouth Road, Tavistock, Devon PL19 8AU. (Tel. (0822) 4671).

Instrumentation in Flammable Atmospheres *28th October LUTON*
(See item for 30th September)

Pattern Recognition *19th to 22nd October MUNICH*
Sixth International Conference on Pattern Recognition, sponsored by the IEEE in association with the IAPR and DAGM, to be held at the Technical University of Munich. Information: Harry Hayman, P.O. Box 369, Silver Spring, MD 20901 (Tel. (301) 589-3386).

Broadcasting *19th to 21st October SAARBRUCKEN*
Conference on Broadcasting Satellite Systems organized by the VDE(NTG) with the association of the specialized groups of the DGLR and the IRT. Information: Herrn Dipl. Ing. Walter Stosser, AEG-Telefunken, Gerberstrasse 33, 7150 Backnang

Manufacturing Technology *26th to 28th October GAITHERSBURG*
Fourth IFAC/IFIP Symposium on Information Control Problems in Manufacturing Technology organized by the National Bureau of Standards, US Department of Commerce, in association with IFAC/IFIP will be held in Gaithersburg, Maryland. Information: Mr J. L. Nevins, Vice Chairman, National Organizing Committee, 4th IFAC/IFIP Symposium Charles Stark Draper Labs, Inc. 555 Technology Square Cambridge, MA 02139 USA. (Tel. (617) 258 1347)

NOVEMBER

***Robotics** *2nd to 4th November LONDON*
International Conference on Robot Vision and Sensory Control—'Intelligent Robot Systems for the Mid Eighties'. Information: Conference Director, IFS (Conferences) Ltd 35-39 High Street, Kempston, Bedford

Computers *16th to 19th November LONDON*
Compec Exhibition, to be held at the Olympia Exhibition Centre, London. Information: IPC Exhibitions Ltd, Surrey House, 1 Throwley Way, Sutton, Surrey SM1 4QQ. (Tel. 01-643 8040).

Safety *23rd to 25th November SEVENOAKS*
A three-day course on Safety of Electrical Instrumentation in Potentially Explosive Atmospheres, organized by SIRA Institute, will be held at Cudham Hall, Sevenoaks, Kent. Information: Conferences and Courses Unit, SIRA Institute Ltd, South Hill, Chislehurst, Kent BR7 5EH (Tel. 01-467 2636)

Instrumentation in Flammable Atmospheres *25th November LUTON*
(See item for 30th September)

DECEMBER

***Electrical Safety** *1st to 3rd December LONDON*
Conference on Electrical Safety in Hazardous Environments, organized by the IEE. Information: Conference Department, Institution of Electrical Engineers, Savoy Place, London WC2R 0BL. (Tel 01-240 1871)

ONLINE *7th to 9th December LONDON*
The Sixth International On-line Information Meeting, organized by *Online Review*, will be held at the Cunard Hotel, London. Information: Organizing Secretary, Online Information Meetings, *Online Review*, Learned Information, Besselsleigh Road, Abingdon, Oxford OX13 6LG. (Tel. 0865-730275)

***INDEX-EL '82** *10th to 15th December ATHENS*
Second International Electrical & Electronics Engineering Exhibition, will be held at The Zappio Palace, Athens. British exhibits sponsored by EEA. Information: EEA, 8 Leicester Street, London WC2H 7BN. Tel. 01-437 0678.

Computers *14th December BIRMINGHAM*
A one-day Seminar on Using Personal Computers in Industry, organized by Sira Institute, will be held at the University of Aston in Birmingham. Information: Conference Unit, Sira Institute, South Hill, Chislehurst, Kent.

1983

JANUARY

***Computer Simulation** *27th to 29th January SAN DIEGO*
Multiconference on Modelling and Simulation on Microcomputers organized by The Society for Computer Simulation, will be held at the Holiday Inn, Embarcadero, San Diego. Information: SCS, P.O. Box 2228, La Jolla, California 92038, U.S.A.

FEBRUARY

MECOM '83 *7th to 10th February BAHRAIN*
Third Middle East Electronic Communications Show and Conference, organized by Arabian Exhibition Management, to be held at the Bahrain Exhibition Centre. Information: Dennis Casson, MECOM '83, 49/50 Calthorpe Road, Edgbaston, Birmingham B15 1TH. (Tel. (021) 454 4416).

MARCH

Component Assembly *March BRIGHTON*
Brighton Electronics Exhibition on matching components with insertion, connection and assembly aids and techniques, to be held in Brighton. Information: The Press Officer, Trident International Exhibitions Ltd, 21 Plymouth Road, Tavistock, Devon PL19 8AU. (Tel. (0822) 4671).

● Telecommunications Networks *21st to 25th March BRIGHTON*
Second International Network Planning Symposium (Networks '83), organized by the Institution of Electrical Engineers with the association of the IERE, to be held at the University of Sussex, Brighton. Information: IEE Conference Department, Savoy Place, London WC2R 0BL. (Tel. 01-240 1871).

Inspex '83 *21st to 25th March BIRMINGHAM*
Tenth International Measurement and Inspection Technology Exhibition, sponsored by *Measurement and Inspection Technology* in association with IQA and Gauge and Tool Makers' Association, to be held at the National Exhibition Centre, Birmingham. Information: Exhibition Manager, Inspex '83, IPC Exhibitions Ltd, Surrey House, 1 Throwley Way, Sutton, Surrey SM1 4QQ. (Tel. 01-643 8040).

APRIL

Engineering Education *6th to 8th April PARIS*
Second World Conference on Continuing Engineering Education, organized by the European Society for Engineering Education, to be held at UNESCO Headquarters in Paris. Information: Mr N. Krebs Ovesen, Danish Engineering Academy, Building 373, DK 2800, Lyngby, Denmark.

● ICAP '83 *12th to 15th April NORWICH*
Third International Conference on Antennas and Propagation organized by the IEE in association with the URSI, IEEE, IMA, IoP and the IERE, will be held at the University of East Anglia, Norwich. Information: IEE Conference Department, Savoy Place, London WC2R 0BL. (Tel. 01-240 1871, ext. 222)

MAY

Test and Measurement *2nd to 5th May SAN JOSE*
The Second Annual Test and Measurement World Expo will be held at the San Jose Convention Center. Information: Meg Bowen, Test and Measurement World Expo, 215 Brighton Avenue, Boston, MA 02134 U.S.A.

Noise *17th to 20th May MONTPELLIER, FRANCE*
Seventh International Conference on Noise in Physical Systems/3rd International Conference on 1/*f* Noise, will be held in Montpellier, France. Information: Dr B. Jones, Department of Physics, University of Lancaster, (Tel. Lancaster 65201), or Professor H. Sutcliffe, Department of Electronic & Electrical Engineering, University of Salford.

Electron Tubes *18th to 20th May GARMISCH-PARTENKIRCHEN, F.R.G.*
Conference on Electron Tubes organized by VDE (NTG) in association with the German Section of the IEEE, will be held in Garmisch-Partenkirchen, Bavaria. Information: Conference Chairman, Dr H. Heynisch, Siemens AG, Werk fur Rohren und Sondergebiete, St Martinstrasse 76, D-8000 Munchen 80.

JUNE

IOOC '83 *27th to 30th June TOKYO*
The Fourth International Conference on Integrated Optics and Optical Fibre Communication, sponsored jointly by the Institute of Electronics and Communication Engineers of Japan and the Institute of Electrical Engineers of Japan, will be held at Keio Plaza Hotel, Tokyo. Information: Y. Suematsu, Department Elec. Phys. Tokyo Institute of Technology, 2-12-1, O-okayama, Meguro-ku, Tokyo, 152 Japan.

JULY

***Reliability '83** *6th to 8th July BIRMINGHAM*
Fourth National Reliability Conference organized by the National Centre of Systems Reliability in association with the Institute of Quality Assurance, will be held at the National Exhibition Centre, Birmingham. Information: Mr A. Cross, National Centre of Systems Reliability, UKAEA, Wigshaw Lane, Culcheth, Warrington WA3 4NE.

SEPTEMBER

***C.A.S.T. '83** *13th to 15th September BIRMINGHAM*
First International Conference and Exhibition on Cable and Satellite Television organized by Cable and Satellite Television Exhibitions, will be held at the Birmingham Metropole Hotel. Information: Exhibition, Michael Hyams, Managing Director, Cable & Satellite TV Exhibitions Ltd. 5 Barratt Way, Tudor Road, Harrow HA3 5QG. (Tel. 01-863 7726) Conference, The Economist Conference Unit, 25 St James's Street, London SW1 1HG. (Tel. 01-839 7000)

Weightech '83 *13th to 15th September LONDON*
Third International Industrial and Process Weighing and Force Measurement Exhibition and Conference, organized by Specialist Exhibitions in association with the Institute of Measurement and Control, to be held at the Wembley Conference Centre. Information: Specialist Exhibitions Ltd, Green Dragon House, 64/70 High Street, Croydon, CR9 2UH. (Tel. 01-686 5741) Conference Information: IMC, 20 Peel Street, London W8 7PD. (Tel. 01-727 0083).

Simulators *26th to 30th September BRIGHTON*
International Conference on Simulators, organized by the IEE, will be held at the University of Sussex. Information: IEE Conference Department, Savoy Place, London WC2R 0BL (Tel. 01-240 1871) *(Synopses by 4th October.)*

***★NEOS '83** *26th to 30th September READING*
International Conference on Networks and Electronic Office Systems, organized by the IERE, will be held at the University of Reading. Information: Professional Activities Department, IERE, 99 Gower Street, London WC1E 6AZ. (Tel. 01-388 3071) *(Synopses by 1st February 1983, Papers by 1st May 1983).*

OCTOBER

***Computer Graphics '83** *4th to 6th October LONDON*
Conference and Exhibition on Computer Graphics organized by Online Conferences. Information: Online Conferences Ltd, Argyle House, Northwood Hills, Middlesex HA6 1TS. (Tel. (09274) 28211)

Security Technology *4th to 6th October ZURICH*
17th Carnahan Conference on Security Technology, organized by the Institute for Communication Technology at the Eidg. Technische Hochschule Zurich, in association with the College of Engineering, University of Kentucky, will be held in Zurich. Information: P. de Bruyne, ETH Zentrum-KT, CH-8092 Zurich, Switzerland. (Tel. 411-2562792)

***Viewdata '83** *18th to 20th October LONDON*
Conference and Exhibition organized by Online Conferences. Information: Online Conferences Ltd, Argyle House, Northwood Hills, Middlesex HA6 1TS. (Tel. (09274) 28211).

Telecom '83 *26th October to 1st November GENEVA*
Second World Telecommunication Exhibition, organized by the International Telecommunications Union, to be held at the New Exhibition Conference Centre in Geneva. Information: Telecom '83, ITU, Place des Nations, CH-1211 Genève 20, Switzerland. (Tel. (022) 99 51 11).

Organizers of appropriate events are invited to submit details to the Editor for inclusion in this calendar.

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EDITORIAL

THE INSTITUTION

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Microscopic Non-destructive Testing

Recent developments in scanning acoustic microscopy **479**

D.A. SINCLAIR, I.R. SMITH and H.K. WICKRAMASINGHE
(University College London)

The scanning acoustic microscope, operating at a frequency of 1GHz or more, gives resolutions significantly greater than are possible with the optical microscope because of the lower velocity of sound waves. The paper describes transmission and reflection scanning types and the techniques of using them. Performance is illustrated with a number of examples of photographs of image displays, both of biological and physical specimens.