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Application of Advanced Technology to Space Automation

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FOREWORD

At the request of the Office of Advanced Space Technology, of the National Aeronautics and Space Administration, Martin Marietta Corporation undertook a study directed toward the assessment of automation technology requirements and benefits relating to future NASA programs.

This report presents the results of the "Application of Advanced Technology to Space Automation Study," which was performed under Contract NASW-3106. A synopsis of specific technology areas is provided along with an analysis of the role of automation in the space program. It also summarizes the benefits to be derived from automation and provides a discussion on the required technology efforts that should be directed toward obtaining these benefits. Furthermore, presentation of a logical approach directed toward achievement of the desired automation goals is provided for future NASA space missions.

It is intended that this report be updated and expanded at periodic intervals so that a current summary of the respective technology areas be available as required for new initiatives or technology prediction exercises. More concise summary techniques are being developed and will be incorporated in the future. However, in the interest of time and available funds, this issue has been published as a first step toward achievement of a technology summary.

The primary contributors and their respective areas of research are as follows:

Study program manager - Roger T. Schappell;

Electronics - John T. Polhemus;

Information gathering - John T. Polhemus;

Image data processing - Chieng-Y Chang;

Ground support systems - Catherine A. Hughes;

Rendezvous, stationkeeping, and docking - Roger T. Schappell;

Fault tolerant processors - James R. Stephens;

Autonomous navigation - James W. Lowrie;

Attitude determination - James W. Lowrie;

Spacecraft performance management and monitoring - James R. Stephens.

Philip Carney also provided significant contributions in the areas of onboard processors and electronics in general.

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I INTRODUCTION

Automated operations in space provide the key to optimized mission design and data acquisition at minimum cost for the future. The results of this study strongly accentuate this statement and should provide further incentive for immediate development of specific automation technology as defined herein. Furthermore, essential automation technology requirements have been identified for future programs such as Space Shuttle, Large Space Structures Missions, Advanced Teleoperator Retrieval Systems, Advanced Planetary Programs, and Future Earth Resources Missions.

Therefore, this study was undertaken to address the future role of automation in the space program, the potential benefits to be derived, and the technology efforts that should be directed toward obtaining these benefits. In support of this objective, we proceeded as follows:

1) We investigated and summarized future automation requirements based on available NASA and contractor information relating to future space programs.

2) We investigated and summarized the applicable technologies, both existing and developmental, within NASA, DOD, and industry in general.

3) We evaluated these applicable automation-related technologies with respect to future space mission objectives and summarized the potential benefits.

Martin Marietta was particularly qualified to perform this study because of our involvement in previous programs such as the Viking Mars Lander, Skylab, Titan Launch Vehicles; current programs such as the Space Shuttle and Shuttle Experiments; and future programs such as the Advanced Teleoperator Retrieval System, Large Space Structures, Mars Sample Return, Manned Maneuvering Units, Manipulators, Free Flying Satellites, and the Global Positioning System. Our involvement in these latter programs emphasizes the need for automation technology and, in many cases, dramatizes the lack of timely technology projections. A case in point is automated rendezvous for the remote teleoperator retrieval systems or, in retrospect, a planetary landing site selection system for Viking. There are many examples one could quote for past and current programs where *a measure of automation could have provided an immeasurable amount of useful science data and/or resulted in significant savings to the project.*

II SUMMARY & RECOMMENDATIONS

This chapter is a summary of the study results and includes our recommendations relating to specific automation technology requirements.

The results of this study indicate that, although future NASA programs can benefit significantly from specific automation technologies, attention must also be given to the development of tools to enhance automation technology requirements predictions, and methodologies for implementation. Therefore, before summarizing the automation technology results, tools and implementation methodologies will be discussed since they will influence the acquisition of the required technology.

TOOLS AND IMPLEMENTATION METHODOLOGIES

Paramount to development of space technology forecasting is a concise common data base for each technology. This is a prerequisite to technology requirements prediction and currently exists in fragmented reports and papers throughout NASA, DOD, and industry. With the amount of research being funded each year, it is extremely difficult for individuals to review the available documentation associated with a specific technology. As a result, it has become hard to anticipate future technology needs.

The use of data-base management systems to record an abstract or detailed document outline that is accessible by terminal and selected using "key words" or "identifiers" that are continually updated, would provide NASA with the means to access current technology efforts as required. This would require the assembly of a simple but "universal" format to summarize the relevant technology.

To date, there are at least 21 primary data bases available to the engineering community. SCISEARCH, a multi-disciplinary index to world science and technology literature, and SSIE, a data base which indexes government and privately funded scientific research projects, are examples of existing data bases. It appears, however, that it would be advantageous for NASA to develop a low-cost data base, recording only those documents and research activities of interest for a given technology area. This would provide NASA with a concise and up-to-date summation of technology with minimum facility requirements. For example, a PDP 11/70 minicomputer could be used to service NASA Headquarters personnel via accessible terminals and printers at minimal cost.

NASA, DOD and industry personnel would willingly respond to a request to format the results of their specific research in a common format for input into this data-base system. Future NASA space automation workshops could be instrumental in bringing this about. Depending on access time and mass storage requirements, it is conceivable that a microcomputer with memory discs and printer could perform the job at minimum user cost.

A third alternative is to tie into a NASA data system such as the Applications Data Service (ADS). ADS, a concept under study at GSFC, is a decentralized but integrated data and information network developed to help NASA solve applications data problems. By working with GSFC it may be possible to build and store the automation technology data base on the ADS and, using an "off-the-shelf" terminal, dial via a public data communication network and use the ADS.

The advantages of a data-base system are obvious: All data is up to date and concisely organized. Multiple users can access the same data for different purposes and the access time is significantly faster than a manual search.

In addition to the establishment of a common data base, it is recommended that the following suggestions receive serious consideration by NASA:

- Encourage the use of automation technology by NASA selected contractors.
- Implement a more efficient means for tracking and influencing industrial IR&D.
- Pursue more strongly the low-cost "protoflight" experiment philosophy via the shuttle test bed.
- Relate to industrial automation needs via "seed money" and NASA's technology utilization program.

Implementation in New Projects - The first suggestion relates to new projects and missions where automation technology could be readily used if OAST coordinated with the respective internal project office before release of the RFP. This would be done by providing data-base assessments of the applicable technology areas to bidders. If the automation technology is not considered at this point, where potential cost savings could be shown, it is likely that traditional approaches would be taken with little consideration given to the advanced automation technology available. Those of us frequently receiving NASA RFPs will generally respond on the basis of our knowledge and experience. If time permits, we conduct a literature search and contact the respective NASA and/or university specialists, but we do not often have convenient access to this data. As a result, the system design is often frozen upon contract award and the potential cost-savings via automation are not considered.

Influence Industrial IR&D - The annual investment in advanced technology by large industrial concerns ($5M to $15M per corporation) could more directly benefit NASA. It is not unusual to see duplication of effort under IR&D nor is it unusual to see technology worked for the sake of technology. This occurs because the IR&D Principal Investigator is either not aware; i.e., does not have the necessary data base, or because he is so involved in the technology that he is unaware of its potential value due to limited perspective. Therefore, it is suggested that NASA assign personnel to more closely monitor tasks relating to their areas of expertise. The key is interaction with industry during the IR&D definition phase from August through November of each year rather than only during the review and grading periods of the respective tasks.

Propagate "Protoflight" Experiment Philosophy - The third suggestion is self evident; however, it must be emphasized that there is often a cross-over point where a concept should be committed to "protoflight" hardware and flown. Space shuttle provides a unique test bed for concept evaluation. Therefore, it is recommended that a "standard" pallet configuration such as the JSC OSTA-1 pallet be used, the high-risk and low-cost experiments philosophy be propagated, and, like the OSTA-1 payload, integration and documentation costs should be kept to a minimum. This would encourage the development and use of automation technology via feasibility demonstrations.

Propagate Technology Utilization Via "Seed Money" - The last suggestion is made to indicate the gap in this country between automation technology research and the eventual user--industry. NASA could play a key role in bridging this gap via their technology utilization program personnel.

AUTOMATION TECHNOLOGY RESULTS

It is of little value to allege that NASA is behind the state-of-the-art in any given technological area unless a future need for that technology can be shown. A primary goal of this study was to identify these needs and to summarize automation technology requirements. A brief summary follows for each technology.

Electronics

Electronics technology is treated as a separate section because it pervades every mission model and constitutes a major cost factor in nearly every system on the ground or in space. Digital electronics and computer technology are especially important because of the improvements in performance and the potential of achieving the long sought goal of hardware commonality across a spectrum of missions. Electronics technology will play a key role in reducing costs while converting 1000 times more space data into useful information by 1990, a NASA goal stated by Kurzhals (1977). Much of the desired improvement will occur because of normal technology trends and the inertia of the massive semiconductor industry, but directed effort is needed to accelerate technology developments toward NASA's automated operations. The greatest potential for technology automation leverage lies in computer-aided design, development, and production of LSI and VSLI electronic hardware. Maturation of this technology base is the key to array processing, special purpose logic chips, transparent software, and a host of other advances that will be embodied in data management and computer systems of the 1980s.

Digital electronics embodied in semiconductors, including microcomputers and other computer hardware, have demonstrated roughly an order of magnitude improvement in performance every three years for about a decade. This trend will continue into the 1980s when, according to a surprise-free forecast, some fundamental limitations will disrupt it. For instance, by 1985 switching element dimensions will approach molecular size. Already the low-number of doping atoms per gate require new design strategies to counter resultant gate-swtching threshold uncertainties. Improved models of circuit elements are essential to continued progress as element sizes recede from our accustomed scale, and circuit electrical properties become harder to characterize.

In spite of the challenges, most of the desired technology will occur without a direct NASA involvement. Space hardware users should cooperatively concentrate on aspects necessary to them but not profitable to the electronics industry such as parts qualification and radiation resistance. The development of testable and reliable Large Scale Integrated (LSI) circuits should be encouraged. This can be accomplished in part at the device design level. Continued pressure for a standard family of space microcomputers is recommended.

Of importance to NASA planners is the obsolescence of the traditional cost function relating to digital hardware. The cost of communication between switching elements already exceeds the cost of elements, and the gap will continue to widen. It is now practical to use as little as 15-20% of the capability of a piece part in order to achieve commonality, testability, and better reliability. Redundancy and block reconfiguration can replace expensive voting circuits on complex software in achieving fault tolerance.

Very high data rates from multiple sensors and synthetic aperture radar will overrun both capability and justification for onboard spacecraft storage and will tax space-to-ground link capacities. Onboard data processing is needed using high performance onboard computers pushing toward gigabit rates. Onboard processing is also needed for landmark acquisition and tracking, fine pointing, and processing and/or selective elimination of sensor data. Advances in device technology are needed to support these requirements. Candidates discussed are silicon on sapphire, Gallium Arsenide devices, and Josephson junction circuits.

Advances in software generation have been slower than hardware improvements--roughly a factor of two every three years. So, where a choice exists, functions will be cheaper to implement in hardware than in software. Modern programming practices and a high-order language approach should be mandated and standardized throughout NASA. Strong emphasis needs to be placed on problem analysis and statement generation in a way that permits syntactic or formal checking; e.g., expression in Backus Nauer Form. This is perhaps the most cost-effective advance to be made in the computer systems arena.

Storage technology has made remarkable progress in recent years in most areas. Semiconductor memories have increased capacity (bits/chip) in quantum jumps of a factor of four every two years, with a lower cost per bit for each advance. One megabit dynamic memories are predicted by 1980, and by 1982 large ground-based semiconductor main memories are likely to contain a quarter-billion bits at 0.04 cents/bit.

In the middle ground of the memory hierarchy, between the small fast memories and larger slower archival memories, there is a decided gap. The best candidates to fill this gap are metal oxide semiconductors (MOS), charge coupled devices (CCD), and magnetic bubble memories. The latter has the unique property of quiescent storage without power, and continued development support by NASA and DOD is advisable.

Mass memory capabilities are increasing at about an order of magnitude every three years, but the tend is destined to slow without user involvement. Very large memories are needed to facilitate the development of large shared data bases, a key to future earth observation missions.

Our basic recommendations concerning electronics are to track the technology and obtain sound long-range forecasts, taking advantage of the best hardware available for a given mission. Software development should anticipate hardware development.

Many recommendations are contained in the main text of the electronics chapter and are summarized below:

1) Recognize the obsolescence of the traditional cost function; Communication between digital hardware elements costs more than the elements themselves.

2) Recognize that performance of digital hardware is improving tenfold every three years while software production improves by twofold every three years.

3) Prepare to bear the cost of production test and burn-in of Very Large Scale Integrated Circuits (VSLI). These circuits will be produced in low quantities, not profitable for semiconductor manufacturers.

4) Foster design practices at all levels that use regular arrays of digital circuits and components but use only a fraction of each circuit capability.

5) Use redundant digital systems and block reconfiguration to obtain fault tolerance.

6) Choose (or develop) a standardized family of space microcomputers.

7) Encourage development of testable and reliable large scale integrated circuits (LSI).

8) Encourage and support the development of computer-aided design, development, and production of electronic hardware.

9) Insist on a problem statement compatible with syntactic or formal checking as a first step in computer program design.

10) Standardize and mandate modern programming practices.

11) Adopt and mandate a single standard NASA high-order language.

12) Use hardware instead of software for highly repetitive computer functions.

13) Support space qualification of digital hardware and microcomputers.

14) Support development of onboard data management systems for a broad spectrum of tasks, including development of gigabit-rate computers.

15) Support development of radiation-hard devices

16) Support development of mass memories for large data bases.

INFORMATION GATHERING

Our recommendations come from system aspects and applications area requirements. In the space shuttle era, the primary challenges shift from data collection to data analysis, and from single problem solution to complex multi-disciplinary systems often involving several platforms. Work on sensors, platforms, launch facilities, data management and electronics should continue, but modeling and data base integration deserve increased emphasis. Specific detailed recommendations concerned with individual application areas will be found in the discussion for those areas; e.g., high rate onboard data handling systems for synthetic aperture radar data, multifrequency radiometric measurements at 19.3 GHz and 1.42 Hz for soil moisture, Summary recommendations for these areas should not be provided out of context, so the reader is urged to refer to the text of this section. We recommend emphasis on the following primary technology automation needs in order of priority:

1) Improved models
2) Integration of Large Data Bases
3) Rapid communication of remote sensing data to user
4) Substantial improvement in pointing and tracking capability
5) Automated landmark acquisition and tracking capability
6) Onboard processing and pre-processing of data
7) Automated atmospheric effects correction
8) Variable resolution sensor systems
9) Improved algorithms and techniques for scene analysis
10) Small intelligent user terminals with display capability

More specific recommendations are as follows:

1) Insure NASA representation on working groups peripheral but important to key technology issues.

   Modeling is an example of a "difficult" technology. Modeling groups exist at the various centers but are not able to bring together all the inputs needed for effective modeling; e.g., users, modelers, computer scientists, researchers, system architects.

   A working group called ARSENIC (Applications of Remote Sensing to Insect Control) provides a framework for dealing with one of the more difficult modeling problems (and other problems), but this group has not been effective because of the lack of NASA representation.

2) Retain one or more interdisciplinary consultants to forecast technology, relate diverse technologies to earth observation missions--especially users--and recommend specific implementation plans on a continuing basis.

3) Continue to have peer groups review programs on a continuing basis. An example is the LACIE peer group headed by Dr. Paarlberg of Purdue University. Peer reviews are very valuable in that political pressures and the realities of budget constraints are not (or should not be) part of their fabric.

4) Establish direct liaison with user communities. An example is the recently established link between NASA and USDA; e.g, P. Thome and R. Whitman from NASA and W. Kiblen and C. Candill from USDA.

IMAGE DATA PROCESSING

All forecasts and the future space mission scenarios have pointed to massive increases in image data return from the spaceborne sensor platforms designed to provide global monitoring of agriculture, minerals, forest, and water. To increase use of large-scale data banks, multiplicity of peripheral equipment and services, and sophisticated analytical and computational capabilities, a highly efficient end-to-end data management program will be needed more than ever. To ensure widespread and timely dissemination of resource monitoring results to user communities, one of the ultimate solutions to the image data processing systems would lie in adaptive real-time onboard processors capable of transmitting processed results rather than just raw data.

To acquire different generic type surface categories for specific applications, to exclude many unwanted data sets from further processing, and to facilitate large area and/or long durations surveys, adaptive capabilities are desirable in the following areas: a) searching and tracking observation coverages, b) optimally selecting measurement sensors and channels, c) screening and editing image data, d) signature extension preprocessing, and e) adaptive modeling of classification schemes.

GROUND SUPPORT SYSTEMS

Presently, man exercises direct control over nearly all decisions required by spacecraft, including those decisions associated with data handling and processing. To perform these multitude of tasks, extensive communications, ground equipment, and many support personnel are required. This is extremely expensive when extended missions are considered or for extensive data collection and processing. However, with the anticipated use of one class of launch vehicle, space shuttle, the development of the TDRSS and related payload operations control center, POCCNET, and the coming generation of standardized spacecraft, it will become feasible to provide automated ground support systems.

During the early years of the space program, differences in philosophy existed regarding the best way to operate a spacecraft and launch vehicle. As a result, many new operations control centers were developed. Now, with the advent of a new era in space activity comes the time to reorganize the development philosophy of ground operations. The recent development of the Launch Processing System, the Satellite and Tracking Data Network, and the concept for the Payload Operations Control Center Network--three areas of the total ground support system discussed herein--will provide NASA with a much more capable ground support system. Though these support systems will not be totally automated, the continued development of more sophisticated semi-autonomous support systems in conjunction with the increased autonomy of onboard spacecraft operations (autonomous navigation, attitude control, onboard data evaluation and reduction) will result in a significant reduction of manhours required to perform these tasks. With manpower costs rising tremendously, this reduction will yield a considerable cost savings to NASA as well as helping NASA achieve its goal of increasing capability 1000-fold in the near future.

Dr. McReynolds of JPL has estimated that about one-third of the overall $1.5 billion savings available through the extensive use of automation is related to ground support operations; $480 million per year by the year 2000 is the anticipated savings. Dr. McReynolds has also estimated that the efficiency of ground support operations will increase by a factor of four through increased system capabilities and productivity.

The overall objective of the new, autonomous support systems philosophy is to provide an evolutionary hardware/software computing system which can reliably operate the ground support systems at a significantly lower cost than current systems. The autonomous system development should be driven by the following factors:

- The nature of computing systems is changing. The trend is to provide more computing power with hardware rather than software. Software costs, relative to hardware costs, continue to climb. The software/hardware cost ratio has been projected at 10:1 by 1985 (Myers, 1978). Software productivity is improving only by a factor of two every three years compared with hardware improvements of an order of magnitude every three years.

- Cost is a major consideration in systems development due to NASA budgetary restrictions. It is well known that direct personnel costs of implementing and operating a system for an extended period of time far exceeds equipment costs. Large cost savings can be achieved by higher efficiency software and a concommitant decrease in manpower.

- The use of standardized equipment throughout ground support systems will provide a specific set of requirements and interfaces. The user can then organize to take advantage of these requirements. Though the user will encounter an initial cost, the standardized systems can be used repeatedly with only minor modifications rather than implementing an entirely new and expensive system if properly designed.

- As ground support systems become more standardized, they will need to be more flexible and responsible. With a majority of the NASA missions lasting from a few months to five years, ground equipment will have to interface with a number of different mission spacecraft. Spacecraft will continue to become more complex as NASA's mission objectives are developed. Flexibility is essential to allow software programs to be modified due to changes in requirements from one mission to the next and to allow easy implementation of new technology developments, which are continually evolving for both hardware and software. Distributed computing systems give this needed flexibility.

The three support systems discussed herein have incorporated these factors into their operating philosophy enabling their continued growth into a more efficient, more autonomous support system. In order to insure the continued autonomous development of these systems and all others associated with the total ground support system, NASA needs to establish specific guidelines to direct future research and development. These guidelines should include these general categories:

- What functions of the ground support system should be totally automated, semi-automated or left as a manned responsibility? Defining these functions will direct the development of hardware/software equipment.
- Establish hardware/software requirements, and from these, develop the necessary equipment to perform the tasks.
- Develop the total support system concept that will incorporate the necessary hardware/software and that will perform all required tasks.

Establishment of these guidelines will help to direct the research of private industries and, in turn, result in the increased efficiency and capability of the support systems NASA is looking for.

The Defense Department has done just this--sponsoring development of advanced technology for military application. In mid-November, 1978, a program was begun to advance semiconductor microcircuit technology for military applications with the emphasis on achieving significant increases in fundamental complexity and device operating speed (Klass, 1978). Over the next six years, 200 million dollars will be spend on the development of the Department's near-term goals with primary emphasis on increasing functional complexity and device operating speed. The Defense Advanced Research Projects Agency (DARPA) will also fund a more ambitious long-term goals program to take a fresh look at basic digital computer architecture. In discussing the Defense Department's development of these programs, Klass points out that the Pentagon was, in the past, content simply to ride the coattails of the commercial market. Viewed in retrospect, says Klass, this decision is seen as a mistake because it led the military into a dependence on a wide variety of commerical microprocessors that had not demonstrated their ability to meet the rugged military environment.

As the present time, NASA also seems to be riding the coattails of the commercial market. Rather than spending money on what is available, NASA should decide what it wants and needs and then direct the necessary research and development. By doing this, NASA will most likely spend less on the implementation of a program, and at the same time, have equipment meeting its unique requirements.

As discussed in detail in this report, the LPS, STDN, and the Payload Operations Control Center Network, represent the state-of-the-art in ground support systems. Each system could be considered semi-autonomous at this time. However, as NASA's missions increase in number and complexity, these systems and all other systems of the total ground support operation will need to be more efficient and flexible. The use of hardware and software advances will provide the desired increased efficiency and capability.

Development of automated software tools to be used extensively during the design and test phase of a program could reduce the life-cycle costs of a program by reducing the number of manhours required during the design and test phase and by correct system development. This would be enhanced by detailed specifications which define constraints, record decisions, and evaluate designs made during the early phases of the software life cycle.

Various theories and systems are presently being developed by commercial contractors to support the development and evaluation of specifications. To date, there are many design aids available that apply to various phases of the software development. A few are listed below:

1. SSL - Software Specification Language
   (High level design oriented)

2. PSL/PSA - Problem Solving Language/Problem Solving Analyzer
   (Requirements and design)

3. PDL - Program Design Language
   (Detailed design)

4. RatFor - Rational Fortran
   (Structured Fortran)

In addition, Martin Marietta's research program is producing a very high-level requirements identification and analysis system for total life-cycle requirements, traceability, and management.

The software design aids are characterized by a data base which is produced from a specification language. The languages have their own unique vocabulary and syntax that language processors use to automatically develop the data base. Further analytic tools are then applied to produce design feedback information.

Software productivity is improving slowly, by roughly a factor of two every three years, and its cost is steadily increasing. Implementation of these modern software tools within NASA programs will help to reduce cost and improve efficiency and their use, therefore, is strongly recommended.

Development of a standardized family of higher-order languages is an important technology goal. In a recent survey, Ambler (1978) found that most system languages do not have the capability of linking to other languages by methods other than cleverly coded assembly language routines. Uniformity within a ground support system will be dependent upon the use of a standardized family of languages--system uniformity will be difficult without using standardized languages.

The space shuttle system has incorporated this idea using GOAL throughout the shuttle system. However, there are problems associated with its use (shuttle system computers are not capable of handling the entire language capabilities due to their size). With more thorough planning, this problem would have been avoidable.

Developing a standardized family of higher-order languages will require significant effort and funding. To help minimize this cost, we should adopt a well-known language used extensively throughout private industry, NASA, and DOD, such as Fortran, and direct efforts toward expanding the capabilities of this language. Developing a family of higher-order languages around one which is well known and accepted will help reduce manpower costs.

With software improvements occuring slowly and cost increasing steadily, a trend is developing to provide more computing power with hardware rather than software. Since the early 1970s, when the era of microprocessors began, their capabilities have been increasing and their cost decreasing. Today there are many semiconductor vendors and a vast number of microprocessors on the market. If NASA is to fully realize the benefits of microprocessor technology, guidelines for selecting microprocessors for NASA should be established.

The microprocessors currently on the market are packaged in one of three forms: Single-chip microcomputers, single-chip microprocessors, and bit-slice processors. Single-chip microcomputers, which contain the processor, program and data memories, and input/output data parts, are at the low end of the performance spectrum and are normally used in dedicated low-performance applications. In NASA programs, such devices could be imbedded in instruments for control and data management purposes.

Single-chip microprocessors, the most common microprocessor, contain only the processor with additional chips required for memory and input/output ports. This design is hardware flexible while the single-chip microcomputer is very expansion limited unless complex multiplexing logic is added externally. The majority of these single-chip microcomputers and microprocessors use an 8-bit-wide data path. Early next year, Intel and Zilog Corporations will have 16-bit single-chip microprocessors available and NASA should investigate incorporating these chips into ground support hardware.

Bipolar bit-slice processors are at the top end of the microprocessor performance spectrum. Such systems typically perform five to ten times faster than do equivalent MOS units. Bit-slice processors also have much greater hardware and software flexibility. The designer has the ability to define the processor instruction set as well as define an architecture to achieve special capabilities or perform a specific application with the highest level of efficiency. A third advantage of bi-polar bit slice is expandability--word lengths can be expanded by cascading units.

The previous paragraphs describe the present status of commercial microprocessors and, as previously stated, NASA should define the microprocessor products it requires if maximum benefits from this technology are to be realized. It is unlikely semiconductor vendors will produce the devices which NASA needs without direction or financial encouragement from NASA.

A second step NASA should take to derive maximum benefits from LSI technology is specification of fabrication techniques. Silicon on Sapphire (SOS), a conventional technology, appears to have a very good speed-power ratio. This ratio is necessary to handle computing rates that are ever increasing as high performance on-board computing systems are developed and used. Use of SOS will avoid a large investment in a new fabrication technology within the next few years, but unless NASA takes the initiative, it is unlikely that SOS will be used in the microprocessor products NASA requires due to the relatively high cost.

A unique difficulty associated with LSI devices is testing. Efforts to date in this area have shown that the ability to test LSI devices is very limited and requires time. As the functional complexity of chips increase, the difficulty of testing will also increase. Therefore, test strategies need to be developed now. One of the near-term goals of the Defense Department's semiconductor microcircuit technology program, discussed early in this report, is the inclusion of built-in self-test provisions on a chip and looking at functional partitioning that can facilitate self-test (Klass, 1978). It is recommended that NASA research this area.

Increased functional complexity and capability of LSI devices will necessitate faster, more dense memory. Semiconductor memory is of three forms: RAM, random access memory or read-write memory; ROM, read only memories; and read-mostly memory. RAM is the most flexible and fast but is a volatile device. ROM is a nonvolatile, but it is not possible to change memory contents once they have been generated. Read-mostly memory exists in two forms: Ultraviolet erasable, electrically programmable ROMS (EPROMs); and electrically alterable ROMs (EAROMs).

General Electric is currently doing research, funded by DARPA, aimed at achieving an extremely large-capacity archival-type of memory with a capacity of 10^{15} bits by using an electron beam both to write and read data stored in memory.

In current mass memory technology development, serial-type memories are under intensive study. Charge-coupled devices and bubble memories appear to be some distance away from taking over mass memory technology. In these areas, much effort is still being devoted to improving system-device interaction.

It appears that a memory hierarchy may be evolving for future processors. The fastest devices will have low densities and be fabricated by bipolar or I^2L. Access times will be 10 to 100 ns. Main memory will have access times of 400 to 1000 ns and use SOS, core, or CCD technology. Cost will be approximately an order of magnitude less than the fastest devices. Auxiliary memory will have the greatest package densities but will have speeds in the 10 μs to 500 μs range. CCD, bubble, or beam access technology could result in costs half that of main memory.

The problem areas to be settled include the need for new system components, the proliferation of device organizations and technologies, and the need for common-usage components. Both NASA and Air Force are already investigating the use of spaceborne bubble memory and CCD systems. It would appear appropriate that investigations of these devices should continue; but similar studies must be conducted on other memory technologies, especially EAROM, if appropriate benefits are to be obtained.

The ability to quickly produce LSI circuits is being demonstrated by a number of semiconductor vendors. Custom masked ROM are now available within two or three weeks. When properly packaged, LSI support devices can result in considerable savings in both hardware and software development efforts. For example, it is now very uncommon to implement a serial interface using either SSI/MSI devices or techniques. Universal synchronous and asynchronous receiver/transmitters are available in LSI and fulfill 90% of computer and peripheral serial interface.

LSI support devices can be of special importance to NASA. First, they would help reduce component count and would, therefore, simplify design efforts. Second, they could improve computer performance by permitting parallel operations where appropriate. Third, they could conveniently implement some of the more standard functions such as telemetry formatting, deformatting, and control of multiplexed busses. The important aspect to support circuits is their interfacing architecture. A poor interface to an LSI support device may necessitate a large SSI/MSI interface circuit that degraded the desirability of the support device.

Fiber optics is another technology area NASA should investigate. Fiber optics systems, which transmit information by means of encoded light beams traveling through thin glass fibers, have significant advantages over all electronic systems in that they are free from electromagnetic interference and pulse effects, they provide a high degree of immunity from intelligence probing and jamming, they are lighter weight, and perhaps, most important, provide a substantially greater data-handling capability (Elson, 1978). As NASA missions become more complex, high-speed data transmission will be essential between support stations and within the ground station. Fiber optics technology will be able to provide the necessary high-speed data transmission.

The Boeing Company is currently involved in a wide-ranging series of development programs exploring fiber optics technology. The investigations include basic measuring instruments (measuring liquid level, liquid flow rate, linear displacement, strain, pressure and temperature), avionics data busses operating at 10-megabit/second and kilometer-scale data links designed for large intra- and inter-plant computer networks (Elson, 1978). By incorporating fiber optic measuring instruments, data buses, and data links into ground support systems, primarily the LPS, the system's efficiency and checkout capability would increase.

It is recommended that NASA invest in fiber optics development for checkout applications and high-speed data transmission, which will be necessary to handle the reams of data and imagery data resulting from the missions NASA plans to fly.

As NASA develops guidelines for uniformity within ground support systems, it will also need to develop a flexible and responsive system. With a majority of the NASA missions lasting from a few months to five years, ground support equipment will be forced to interface with a number of different mission spacecraft. Spacecraft will continue to become more complex as NASA's mission objectives develop. Flexibility will be essential to allow software programs to be updated due to requirement changes from one mission to the next and to allow for easy implementation of new, evolving hardware and software development. Distributed computing systems give this needed flexibility.

The concept of distributed processing lends itself easily to ground support operations. Many of the tasks are highly specialized, involving tedious, continuous, step-by-step control of many remote sensors and stations. The distributed processing system is designed so multiple processors work independently on a specific task. Additional processors can be added and tailored to fit any type of processing situation, thus offering the best opportunity for obtaining maximum computing power.

Some disadvantages exist in a fully distributed system. A distributed network is a complex technique in computer science. Since the system reaches its maximum performance through asynchronous parallel execution and processor count expansion, good system management must be provided in both hardware and software. More research and experience is needed to develop a system management concept capable of handling a complex ground support system, so it is recommended that NASA invest in this research area. Once the problem of system management is resolved, the distributed processing system for ground support operations will be able to expand and automate more and more of the ground operations support tasks.

RENDEZVOUS AND DOCKING

Having investigated numerous mission models ranging from Mars Sample Return to Large Space Structures, it is apparent that a requirement exists for automated rendezvous, stationkeeping, and docking. This is due to limited man interaction, fuel and trajectory optimization requirements, safety, communications limitations, and a need for real-time operation.

The feasibility of developing an accurate ranging and tracking system for rendezvous, stationkeeping, and/or close-in proximity operations is feasible but has not been accomplished. Feasibility investigations have been conducted, breadboard hardware for systems such as scanning laser radars has been built and tested, but system limitations, potential mission constraints, cost, etc., have prevented the development and demonstration of an acceptable concept(s) for many future mission models. Furthermore, the lack of required technology was apparent during the early phases of the Teleoperator Retrieval System development and during the DOD large space structures studies. We recommend that NASA institute a simple low-cost logical approach whereby after having established the design criteria for representative mission models, a feasibility demonstration be undertaken to develop, as a minimum, a close-in proximity sensor capable of providing accurate ranging and tracking information for close-in rendezvous, stationkeeping, and docking. This will require an investigation of RF, laser, and video concepts.

FAULT TOLERANCE AND REDUNDANCY MANAGEMENT

Space missions that require long unattended spacecraft service will require extensive implementation of automation technology. Planetary missions require autonomous spacecraft capability for navigation, attitude control, and payload data management. Orbital missions where spacecraft are in ground contact for only a small portion of the mission also require a high level of automation.

Autonomous operations on long-duration missions involve heavy computing loads for spacecraft onboard operations and, therefore, put a greater emphasis on the contribution of onboard processing to the overall system failure rate. This fact leads us to look for improved computer reliability through fault detection and recovery. Hence, development of the fault tolerance computer is indicated.

Fault tolerance, by definition, is the ability to render at least the essential level of service (housekeeping functions) after the occurrence of a fault. This ability implies a measure of redundancy because there had to have been parts or circuits not used for the basic level of service.

The state-of-the-art in redundant systems is best characterized by the shuttle avionics system, discussed in detail in the Spacecraft Performance Management and Monitoring chapter. This system has two major requirements: Fail operational, which requires that no single failure will produce the inability to achieve mission objectives, and fail-safe, meaning a second system failure will not result in the loss of the crew or vehicle. For these requirements, the shuttle avionics system provides automatic reconfiguration for time-critical failure modes.

Redundancy management involves fault detection, identification, and reconfiguration including determination of how redundant sensor outputs are used and the manner in which faults are neutralized. Failure detection and identification are provided by one or more of built-in-test-equipment (BITE) indications, data transmission checks, comparison test, or crew observation for non-time critical functions. Reconfiguration is either automatic or manual, depending on time.

Redundancy itself is not sufficient for fault tolerance. Fault tolerance requires that recovery be accomplished onboard automatically. If any error detection, reconfiguration, or recovery depends on diagnosis by ground control, then the system is not fault tolerant.

To date, fault tolerance has gained only limited acceptance by both NASA and DOD; the primary obstacles being large development costs and the onboard resources required for such a computer. These obstacles have been reduced due to progress in semiconductor technology. Now, logic functions can be realized at a lower weight, lower cost, and lower power consumption. Semiconductor technology is discussed in detail in the Electronics chapter.

A greater need exists today for the fault tolerant computer due to longer mission durations involved with planetary exploration, and more demanding mission objectives. To date, only one fault tolerant computer has flown--the Primary Processor and Data Storage (PPDS) computer, designed in the early 1960s, which flew on NASA's Orbital Astronomical Observatory. The Self-Test and Repair Computer (STAR), developed by JPL, was a system using extensive dynamic fault tolerant techniques. Though this concept also originated in the early sixties, the majority of the work was done a decade later when a lot of interest existed in planetary missions involving spacecraft operation for up to ten years with limited earth communication. Due to budget cuts, however, the program was cancelled. Three important concepts developed during the STAR program were the modular concept and bus-oriented architecture, both used extensively today, and the test and repair processor, a restarting function.

The Fault Tolerant Spaceborne Computer, FTSC, is being developed by the Air Force to support long-duration missions and represents the most recent concept in fault tolerance computers. The program objective is to provide a five-year on-orbit capability to perform computational tasks. Though the present FTSL configuration developed by Raytheon provides 95% reliability over five years, problems with the design have necessitated more time be spent studying alternate configurations.

Much work needs to be done in the areas of fault tolerance and redundancy management to understand the problems associated with these concepts; specifically, the hardware/software aspects.

The example provided by the extensive use of multiply redundant components and redundancy management in space shuttle should be used as a guideline in the design of automated spacecraft systems for long-duration missions. In the interest of saving weight and power, and in reducing software complexity and computer loads, dual redundant elements operating in prime/standby mode may be necessary. To support this, emphasis should be placed on improvement in individual component, reliability, internal component fault tolerance, and self-test and BITE capabilities.

There are several areas in which research and advanced development are needed to exploit the capabilities of fault tolerant computers and to lay groundwork for more extensive applications of fault-tolerant data processing for future spacecraft. These areas include:

(a) Reliability calculations for the FTSC and other computers have been carried out using an exponential reliability model at the part levels. Although the validity of this model has been questioned, deviations were felt to be tolerable in view of the simple calculations resulting from exponential assumption and extensive modeling capabilities that existed specifically for the exponential failure law. Recent investigations of on-orbit failure rates have shown drastic deviations from the exponential failure law. This suggests that its use for predicting failure rates for long-duration missions should be avoided.

The existing data point to high failure rates during very early mission stages followed by successively lower failure rates for each six-month interval up to three years. Although these observations lead to optimism regarding the success of long-duration missions, they point up the need for new analysis tools. A failure law applicable to computer architecture rather than the part level is needed, as well as agreement on a form of a reliability function for predicting on-orbit reliability for long-duration missions. Considerable research and organization effort in this area will be required to fully exploit the capabilities expected from components currently under development (see Electronics Chapter).

(b) The software to be executed on a fault-tolerant computer for spacecraft is another important issue. Although the reliability of the hardware can be demonstrated at a level that will allow computer control of critical spacecraft functions, the full advantages of the fault tolerant computer can be realized only if the software has equivalent fault tolerance. Neither testing nor formal verification can assure the total correctness of the software. Fault-tolerant software techniques are available but are costly to implement because of the memory required. However, with the advent of magnetic bubble memories, it may be possible to keep alternate programs in a backup store and this may facilitate the use of fault-tolerant software techniques for essential program elements. The effort necessary to define appropriate software and memory architecture that will permit realization of the potential of the fault-tolerant computer seems fully justifiable.

(c) Some tasks needing to be carried out at high speeds impose burdens on the fault-tolerant computer or are completely impossible to secure in this manner. Examples are frequency control of communications equipment and sensor data compression. In both, a sample instruction repertoire is sufficient and a minimum of local storage is required. Microprocessors seem well suited to these tasks and can be made fault tolerant by being tested periodically and, if necessary, replaced under the control of a program residing in the fault-tolerant central computers to a wider range of spacecraft applications is an area requiring further research and development.

Autonomous Navigation

Historically, navigation of satellites has been performed using tracking stations and ground processing. Therefore, to automate this procedure, sensors must be developed that can replace tracking stations in determining ephemeris of the vehicle. The Department of Defense has developed many such sensors to fulfill their own needs for autonomous navigation and NASA can take advantage of these developments.

Autonomous navigation systems can be separated into three groups--position-sensitive angular measurements to celestial objects, Earth-based target reference measurements, and range measurements to known beacons as indicated on the folloiwng text:

- Position-sensitive angular measurements to celestial objects
  - LES 8/9 — Sun - Local vertical
  - SS-ANARS — Moon - Star
  - AGN — Planet - Star
- Earth-based target reference measurements
  - Natural landmark identification
    1) area correlator
    2) linear feature detection
  - Artificial landmark identification
    1) Systems using optical emitters (lasers, search lamps)
    2) ILT using microwave emitters (radars)
- Range Measurements to known beacons
  - GPS — Earth orbital beacons

It is beneficial to evaluate autonomous navigation systems in terms of the mission for which they will be used. The three basic types of missions considered here are interplanetary, earth observational, and earth orbital without observation capabilities.

The JPL Autonomous Guidance and Navigation (AGN) system, consisting of a CCD imaging sensor/processor, appears to be the best suited for this type of mission. The only other system which could be adapted would be the space sextant.

However, the sextant is not a primary imaging sensor as is the AGN system. Furthermore, the accuracy achievable with the sextant relies on precise knowledge of the lunar ephemeris and a model of the lunar terrain. For interplanetary missions, this data is not as precise, so the accuracy of the system will be degraded. Development of the AGN system should continue for it will enhance capability of future missions such as:

- Rendezvous with a small moon or asteroid
- Higher accuracy orbit injections and course corrections
- Real-time orbital maneuvers

The AGN program is also advancing the state-of-the-art in areas of CCD imaging devices and onboard processors.

The primary goal of earth observation missions is to acquire data pertaining to the earth's surface. The amount of data being acquired has become enormous and led to the end-to-end data management problem. There have been several concepts proposed to deal with the problem including:

- Selective data acquisition through sensor pointing and cloud discrimination
- Onboard registration of sensor data
- Onboard geometric correction of sensor data
- Real-time data link between users and satellite

Each of these concepts would benefit from an autonomous navigation and attitude determination system. There have been two basic approaches to such a system. The first is to navigate the vehicle with extreme accuracy and then derive the ground position of the sensor's FOV through appropriate transformation. Although this method would appear desirable due to the development of GPS, the system is subject to the following errors:

- Misalignment between the sensor axis and the vehicle axis
- Atmospheric effects on the line-of-sight of the sensor (atmospheric bending)

Therefore, even though the position and attitude of the vehicle are known precisely, large errors can exist in knowledge of the sensor's field-of-view. This is particularly important in the areas of data registration and rectification where users often require sub-pixel (<100m) accuracy.

The second approach to a navigation system used for this purpose involves the detection of landmarks or ground control points from space. These systems have the advantage of determining the position of the sensor's field-of-view directly. Therefore, the effects of the atmosphere and misalignment are minimized and the efficiency of the data management system increased.

Of the landmark navigation systems studied, the area correlation approach is best suited to autonomous operation for the following reasons:

1) Linear feature detectors acquire only a single component of position information for each sighting. This implies a loss of accuracy in position information while requiring extensive filtering of measurements.

2) Artificial landmark trackers require that the landmarks be maintained leading to increased support costs. Also, landmark sightings are limited to areas where equipment exists.

3) It is possible to implement an autonomous pointing system in the area registration. This will allow selective acquisition of data rather than non-deterministic acquisition. This is not possible with the other systems.

4) The use of area landmarks is extensive in the rectification of imagery, and the development of an area landmark navigation system will make autonomous rectification feasible.

On earth orbital missions not viewing earth or requiring an extensive data management system, a landmark tracker is of little value. The three primary systems applicable to such missions are GPS, Space Sextant, and LES 8/9 concept. The LES 8/9 system has limited accuracy but is a simple and inexpensive system. For missions requiring automation but not high accuracy, this system would be beneficial. For missions requiring higher accuracy, Space Sextant and GPS are applicable. The Space Sextant provides not only navigation but attitude determination as well and would be best suited for high altitude missions such as large space structures. GPS, on the other hand is designed for low earth orbit (about ½ of GPS orbit or 5,000 nautical miles) and is extremely accurate in this application. There are studies being performed to determine the feasibility of receiving the navigation signal above the GPS orbit. A vehicle above the GPS altitude would receive messages from satellites on the opposite side of the earth. This study should be pursued to establish the accuracy obtainable through this configuration.

Benefits of autonomous navigation systems can be separated into three areas:

- Reduction in mission support costs
- Increase in the scope of future missions
- Reduction of the data management problem

The largest benefit will be realized in the area of data management where the amount of data and its processing can be cut tremendously while bringing several advantages to users.

ATTITUDE DETERMINATION

The sensors used for attitude determination have characteristically been carried onboard the satellite and signals from these sensors telemetered to ground for processing. Therefore, major development of sensors is not required for the automation of the attitude determination system. However, certain sensors are being developed which promise unique benefits and are mentioned below.

Many attitude sensors have been standardized by NASA and most of these are best suited for future needs. Such components include the NASA standard IMU and sun sensor. The NASA standard star tracker, however, is not the best suited sensor for future missions. The tracker suffers from errors due to variations in the local magnetic field, temperature, and stray electronics. CCD star trackers eliminate these error sources while allowing an increased accuracy (.25 arc sec vs. 10 arc sec) and the capability of tracking several stars or objects simultaneously. Development of CCD star tracker should be continued to replace the current NASA standard star tracker.

Two other sensors under development are worthy of mentioning because they provide not only autonomous attitude determination but navigation as well. The first of these is the Space Sextant, mentioned in the Navigation Section of this report. While the sextant would not be used for attitude determination alone, the combination of this capability with autonomous navigation makes the system attractive for many missions. NASA should take advantage of this DOD developed technology for future needs.

The second system uses a set of phased antennas in an interferometer configuration in conjunction with a GPS receiver to allow both attitude determination and navigation. The system is being developed under internal IR&D at Lockheed.

The complete system would provide all necessary inputs to the satellite's guidance and control system.

Onboard Processors require extensive study and development. Two attitude determination processor studies were recently performed for Goddard Space Flight Center and for the Air Force, and both found the NSSC-1 to have insufficient capabilities. Rather than eliminating standardized processors, it would be better to standardize components of a processing system and architecture for expanding the capabilities of each component. An example of this approach was found in the GSFC/OADS contract where a central processor was used to control the flow of data to several external arithmetic processors. Other approaches to expanding processing capabilities are through parallel data processing and preprocessing. To achieve the approach to standardized computer architectures and processing components, a NASA standard data bus and standard interfaces must be developed. This will allow for the expansion of onboard processing capabilities while retaining the benefits of NASA-standardized hardware.

CRITICAL TECHNOLOGY REQUIREMENTS

This study makes it apparent that specific technologies should be supported and funded to insure availability of specific capabilities for future missions. Table II-1 summarizes major automation/technology requirements, potential applications, and benefits to be realized.

It should be noted that robotics and manipulators were included with emphasis on industrial automation applications. This should be accomplished by NASA OAST via the technology utilization organization. It has been stated frequency that a dire need exists for automated operations in U. S. Aerospace Manufacturing for increased productivity. This can be accomplished via the following:

- Robotics for tooling, handling, and maintenance
- Solid-state imagers and real-time image processing for pattern recognition
- Computer numerical control to handle "N" tools simultaneously
- Computer-aided design, computer graphics, etc.

A prime example of industrial automation in Japan is the Fanuc Robot. It is used for loading, unloading and stocking. Its capabilities include 24-hour daily operation, a 5-foot reach, 5-foot vertical movement, 45 lb. lift capability, fully servoed, and costing $13,000.

Conversely, an excellent example of the application of robotics and computer technology in the U. S. can be found at the F-16 aircraft production facility at General Dynamics in Fort Worth, Texas. Since September 1978, a programmable Cincinnati Milacron Model T^3 industrial arm robot has been in use to drill and rout composite F-16 vertical fin skins on the production floor. It incorporates multiple part fixturing compliant tooling with sensing, automatic tool changing, and rapid program data changing. Propagation of this technology is important because in this country we still treat robots as space unique tools, not as versatile precision laborers.

TABLE II-1 SPACE AUTOMATION STUDY RESULTS

TECHNOLOGY REQUIREMENT	APPLICATIONS	BENEFITS
Rendezvous, Stationkeeping And Docking Sensor System	• Large Space Structures • Advanced Teleoperator Retrieval System • MARS Sample Return • Free Flyers	• Vehicle And Personnel Safety • Optimized Fuel Consumption And Time To Docking • Reduced Cost
Adaptive Science Sensors	• Earth Resources Missions Such As Advanced Landsat • Planetary Observation	• Reduced Data Storage And Reduction • Optimized Science Data Selection • Reduced Cost
Robotics Including Manipulators	• Space Manufacturing • Industrial Processes • Remote Handling, Refurbishment, And Retrieval Of Spacecraft • MARS Sample Return	• Planetary Surface Mobility • Extended Payload Life • Reduced Astronaut Workload • Hazardous Materials Applications • Greater Precision And Repetitive Reliability • Reduced Cost
Automated Pointing Mount Smart Sensor(s)	• Science Sensor Pointing • Stellar Observation	• Reduced Science Sensor Requirements, i.e., FOV, Speed, Complexity, Etc. • Fine Pointing Capability • Reduced Cost
Radiation Hardened MicroProcessors	• All Planetary And Earth Orbital Vehicles	• Utilization Of Commercial Hardware • Reduced Cost
Automated Ground Support Systems	• All Shuttle And Launch Vehicle Missions	• More Frequent Launches • Increase Percentage Of Successful Launch • Less Ground Support Personnel • Reduced Cost

TABLE II-1 SPACE AUTOMATION STUDY RESULTS (Continued)

Pattern Recognition	• Planetary Programs Such As MARS Sample Return • Industrial Manufacturing • Remote Handling, Refurbishment, And Retrieval Of Spacecraft • Earth Observation • Space Manufacturing	• Identification Of Science Observables For Planetary And Earth Observation Missions • Identification Of Parts For Tactile Sensors And For Machine Tool Accuracy Control • Reduced Cost
Computer-Aided Design Tools	• All NASA Programs • Industrial Manufacturing	• Design Of "0"g Structures • Optimized Design Of PC Boards, Etc. • Less Documentation • Reduced Cost
Mass Memory	• On-Board Data Storage	• High Reliability Compared To Tape • High Density And Speed • Low Power • Reduced Cost
Precision Pointing And Attitude Control	• Large Space Structures	• Reduced Ground Support • Reduced Cost
Autonomous Landmark Navigation	• Global Monitoring Missions	• Automatic Registration Of Data • Reduced Ground Support • Pointing Of Science Sensors • Periodic Resampling Of Same Surface Area • Reduced Cost
Autonomous Navigation	• Earth Orbital And Inter-Planetary Missions	• Improved Accuracy • Reduction In Mission Support Costs
Fault Tolerance, Redundancy Management	• All Launch Vehicles, And Shuttle • Planetary And Earth Orbital Vehicles • Ground Support Equipment	• Allow For Extended Missions Without Earth Communications • Development Of More Reliable, Complex Onboard Computing Systems • Increase Percentage For Successful Mission • Reduce Ground Support Personnel • Reduced Costs

III ELECTRONICS

Introduction

NASA has a long-range space technology goal of converting 1000 times more space data into useful information while reducing costs (Kurzhals, 1977). Normal technology trends will bring about much of the desired improvement, but directed effort is needed to accelerate technology developments toward automated operations. The following discussion relies on a surprise-free technology forecast as a basis for discussing trends and technology needs in electronics. It is worth noting that technologists and technology forecasters tend to over estimate performance improvements for the near future and under estimate them for the far future.

The justification for a separate section on electronics rests on a general applicability of electronics technology, especially digital electronics and computer hardware and software, to all of the scenarios and mission models that we considered. This report is intended to complement the works already available by Kurzhals (1977), McReynolds (1978) and others, and NASA documents on technology automation; e.g., NASA, 1976.

The emphasis here is on digital electronics and computers because these have by far the greatest cost implications for future high technology systems. The startling improvements in cost per function noted for these areas are applicable to a lesser degree to analog electronics, and as a consequence, there is considerable pressure to implement all parts of a system in digital hardware or convert analog signals to digital form as soon as possible within the system string.

Digital Electronics

Digital electronics embodied in semiconductors, including microcomputers and other computer hardware, have demonstrated about an order of magnitude improvement in performance, size and cost every three years for at least a decade. More recently, increases in reliability of electronic components have followed a similar curve as illustrated in Figure III-1 (Mayo, 1977).

In 1964, Gordon Moore noted a trend for the number of components per circuit for the most advanced integrated circuits to double every year; he predicted that the trend, started in 1959, would continue and it has (Figure III-2; adapted from Noyce, 1977). Like trends are seen in the reduction in cost of switching elements (Figure III-3; adapted from Mayo, 1977) and the cost per bit of computer memory (Figure III-4; adapted from Noyce, 1977).

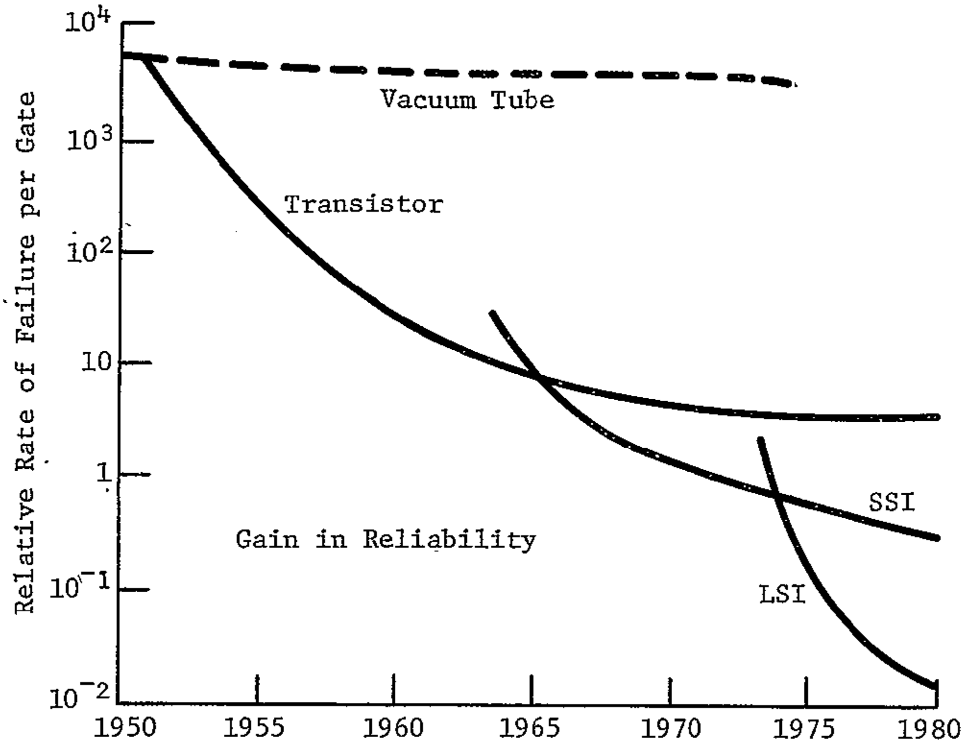

Figure III-1 Switching Element Reliability (Adapted from Mayo, 1977)

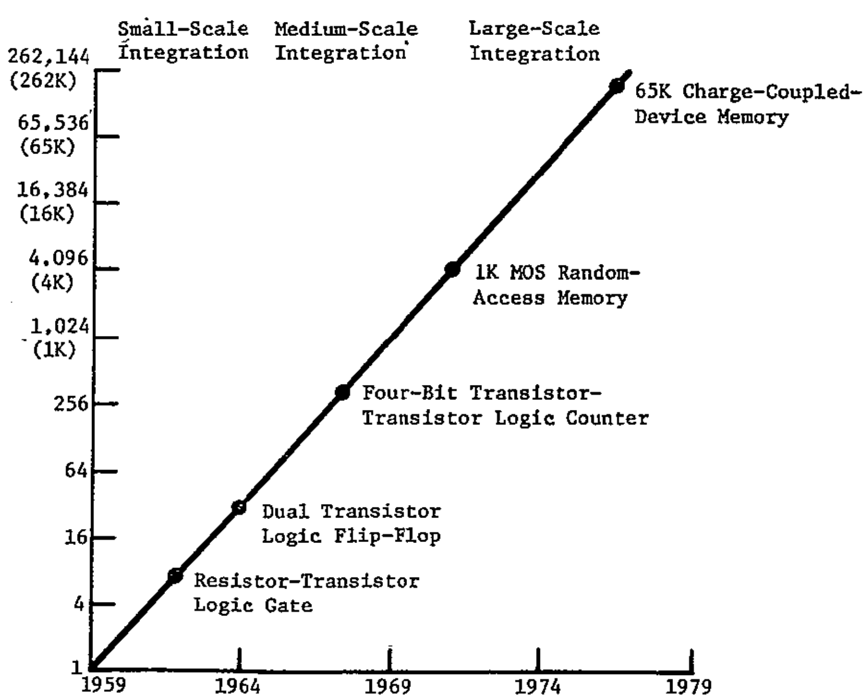

Figure III-2 Number of Components per Integrated Circuit (Adapted from Noyce, 1977)

FOLDOUT FRAME

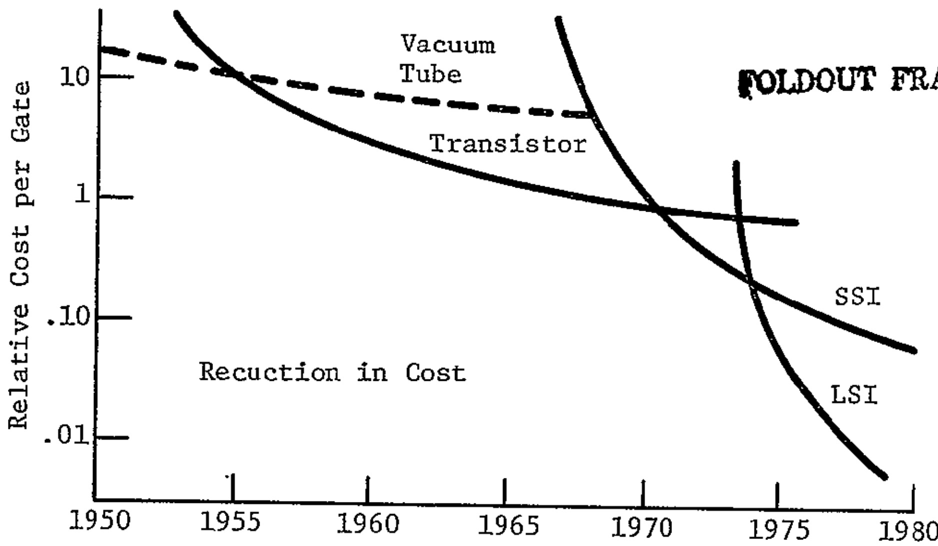

Figure III-3 Switching Element Cost (Adapted from Mayo, 1977)

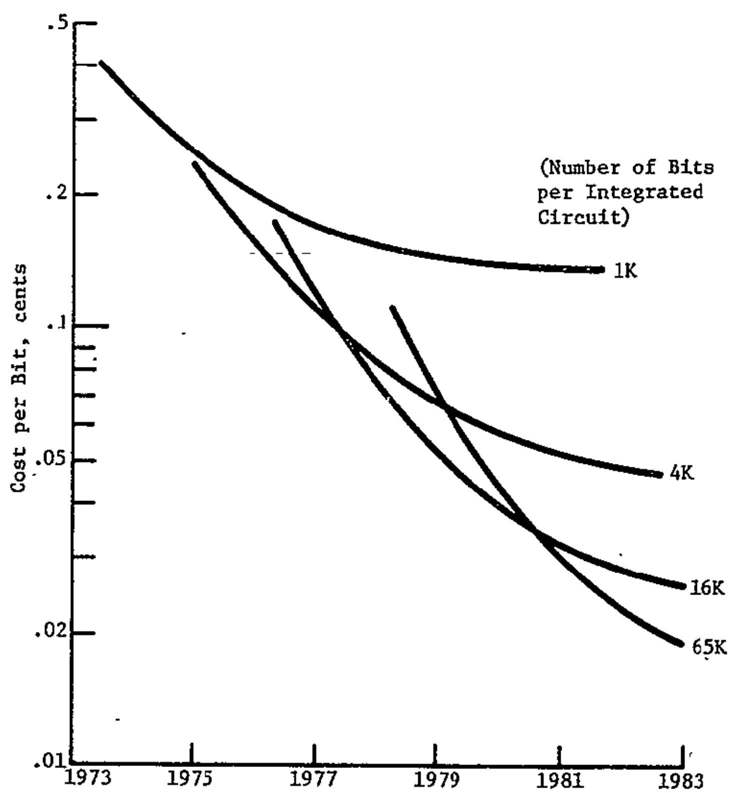

Figure III-4 Cost per bit of Semiconductor Memory (Adapted from Noyce, 1977)

The performance measure usually applied to microelectronic devices is the speed-power product. One by one, new semiconductor technologies; TTL, n-MOS, CMOS, I^2L; have been introduced that lower the speed-power product, with the curve plotted against time following roughly the same pattern as the other figures of merit mentioned above (Meindl, 1977).

These improvements refer to the best available technology at any given time. Because of considerations such as the lag time between system design and flight and the time necessary to qualify parts, the hardware actually used in NASA systems, especially aboard spacecraft, is typically a number of years behind the state of the art. This problem may, in part, be obviated by good technology forecasts, and a design philosophy that permits technology advancements to be anticipated in the design phase. Increased emphasis on timely qualification of key hardware items will also help.

The trend in microelectronics described above should continue into at least the mid 1980s when, according to a surprise-free forecast, some fundamental limitations will disrupt them. For instance, the cell size for switching element will, of necessity, approach molecular size, and connecting lines between elements should, by scale, be even thinner. X-ray and electron-beam patterning presently hold the promise of soon reducing by more than 1000 times the area occupied by a transistor, permitting Very Large Scale Integrated (VSLI) circuits to be fabricated. The resulting smaller geometries pose a technological challenge because with only twenty or thirty doping atoms per gate, the switching threshold uncertainty will increase to perhaps twenty percent, and other switching characteristics will become similarly uncertain. These and other physical mechanisms that are inconsequential in larger devices will become important or dominant as the dimensions of the elements shrink below one micrometer.

Improved models of the circuit elements are a requirement for continued progress (Meindl, 1977) but the electrical properties of the circuit elements become ever more difficult to characterize as the element sizes recede from our accustomed scale.

In spite of these challenges, we concur with Kurzhals (1977) that much of the desired technology advancement in electronics will occur with or without a direct NASA involvement because of the momentum of the electronics industry correlated with sales of a massive amount of bit processing, logic and storage hardware. NASA and other users of hardware for space use should concentrate on specific areas not in the mainstream of the electronics industry such as parts qualification and improving radiation resistance.

Another fruitful area lies in encouraging the development of LSI circuits having regular arrays of similar cells instead of dedicated logic cells. This will work to improve both testability and reliability which are much needed attributes for space hardware.

An important consequence of the very rapid improvements in electronics hardware is the obsolescence of the traditional cost function. The area of a circuit devoted to communication between elements usually far exceeds the area devoted to switching elements, and communication delays are much longer than logic delays (Mayo, 1977).

Packaging, testing and algorithm generation costs are much greater than parts costs. This means that emphasis no longer belongs on parts cost or minimizing piece parts in a circuit or system, or minimizing logical functions required for a given task. Emphasis should instead be placed on obtaining regular, testable arrays, avoiding digital race problems and coping with switching uncertainties and reducing communications costs. Some of the ways to accomplish this include implementing system functions in hardware instead of software if practical, and using identical piece parts where possible throughout the system even if only 15-20% of the capability is utilized. Redundant subsystems, self-checking and block reconfiguration should be used to obtain high reliability. This trend was noted by NASA planners in the 1976 OAST workshop.

We believe the greatest single potential for technology automation leverage lies in computer-aided design, development and production of electronic hardware. The maturation of this technology will be multiplicative, computing power generating computing power. This technology base is the key to the long sought goals of inexpensive special purpose logic chips, array processing on a large scale, transparent software and a host of other advances that will characterize the powerful systems of the 1980s.

Computers

Computers are treated in addition to digital electronics because of the importance of the subset to systems for every sort of mission. Additionally, computers embody both hardware and software, inviting discussions of the relationship between the latter two from the standpoint of technology automation.

Computers have traditionally been considered as systems, not system elements. But the advent of microcomputers and Large Scale Integrated circuits (LSI) has changed the picture and computers may now be either.

Hardware - Computer hardware has seen a revolution in the last twenty years that is seemingly without parallel. World computing power has increased from 500,000 operations per second to 8 billion operations per second. Both large and small machines continue to increase capability at a rate that even good planners and technology forecasters underestimate in their predictions (Turn, 1974; 1978). These trends in performance have been described under digital electronics.

Synthetic Aperture Radar (SAR) systems such as flown on SEASAT gather data at an extremely high rate that overruns both capability and reasonableness for onboard storage and taxes real time communication and ground processing facilities. This sensor technology exemplifies the need for onboard processing, hence high performance onboard computing systems pushing toward gigabit computing rates. Silicon on Sapphire (SOS), a conventional technology, is a good contender for this application as it takes advantage of the vast body of MOS technology and has a respectable speed-power product and good radiation hardness. The use of SOS will avoid a heavy investment in new technology and yields are already predictable and controllable. The "unconventional" technology is Gallium Arsenide (GaAs). This material has much better bulk mobility than silicon, and the peak electron velocity is higher in lower electron fields. Its gate density is greater than bipolar, at orders of magnitude less in power dissipation. In more traditional parameters, the unity current gain frequency ranges from 10 to 30 times MOS and 3-10 times bipolar. Microwave devices are presently made in GaAs, and 10 to 16 gHz digital microcircuits are forecast by 1982-1985 (Anonymous, 1977). There are processing anomalies, however, GaAs "does not rust" so that oxide-deposition is a problem. The supporting technology for GaAs is microwave devices, which do not equal the investment in MOS or bipolar, however at least four U. S. companies are working with digital GaAs circuits. Much of this effort is being supported by the Air Force Avionics Laboratory. It is significant to note that Japan has made a positive commitment to both GaAs LSI devices and gigabit computing architectures.

By the mid-1980s, it is likely that logic circuits employing cryogenic cooling will be in use. IBM has launched a major development effort on one such family of devices, the Josephson junction circuits. The circuit density is presently 400,000 bits/square inch in memory, about the same as the best integrated circuits of 1975. The attractive feature is the low power dissipation of the switching elements. In a computer with 100,000 logic circuits and a cycle time of 500 ps (assuming 10 circuits switch serially at 50 ps each in one machine cycle), silicon circuits would generage thousands of watts but Josephson circuits would generate a few milliwatts. The alloy used in Josephson circuits (lead-indium-gold) must be operated at liquid helium temperature, -273°C. The most likely early usage of cryogenically cooled electronics is in ground-based computers where the tradeoff is between a large air-conditioning unit to cool conventional circuits or a small cryogenic refrigerator for circuits such as Josephson devices. Cryogenically cooled sensors are now flown, however, and there is no fundamental barrier to space use of cryogenic circuits of other kinds. It is recommended that NASA track developments in this area and consider supporting development efforts.

As explained in the next section, improvements in software have not been as dramatic as in hardware. Where a clear tradeoff exists, functions should be implemented in hardware rather than software. This approach has been used to provide intercommunication between computers in a JPL designed spacecraft modular computer network; software requirements are minimized (Rennels, 1978). In the same system, microprocessors are utilized at only a fraction of their capacity to allow simplification of software and simulation, and the overall design approach embodies modularity. We recommend that NASA's computer design philosophy for the 1980s include these elements.

In 1972, when the era of the microprocessor began, it was heralded as the end of custom and one-function LSI devices; the microprocessor, it was said, is programmable hence one device would be broadly applicable. However, this promise was not fulfilled; improvements and desired changes have led to a proliferation of microprocessors, culminating in a bewildering array of over 100 types in 1977 (Figure III-5). Verhofstadt (1978), characterizes the following more or less distinctive classes of microprocessors.

1. "Low-end" microprocessors for relatively simple control applications in industrial and consumer equipment as replacements for electro-mechanical devices as well as for brand new applications.

2. Intermediate microprocessors for more complex industrial

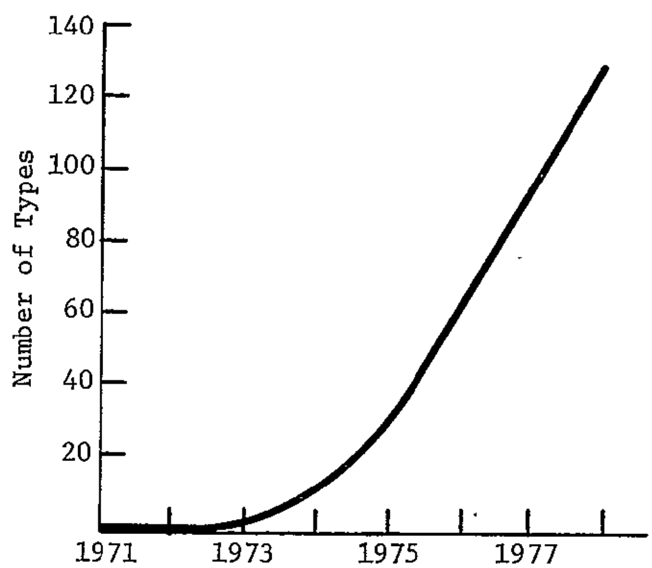

Figure III-5 Proliferation of Microprocessor Types
(from Verhofstadt, 1978)

controllers, peripheral equipment, military, communications and instrumentation as well as terminal applications. A large percentage of these will replace hard wired logic or custom LSI as well as going into new applications.

3. "High-end" minicomputer-like microprocessors for data processing, business, complex real time control, advanced communications, stand-alone terminals, and other similar applications as well as for use in distributed processing networks.

4. Bit-slice type "microprocessors" for very high-performance applications requiring considerable architectural flexibility. Significant areas of interest will be the emulation of existing computers, very high performance controllers and usage as building blocks for larger computers.

Initially, microprocessor manufacturers stressed how much could be done with clever programming. Recently the trend has been to de-emphasize the need for experienced programmers and provide more power in hardware, a healthy trend. Systems of the 1980s will continue this trend; their building blocks will be special purpose chips, plug-in read-only memories, and firmware.

It is unlikely that NASA will have a requirement to design microprocessors to meet its needs, because microprocessors, along with memories, benefit so greatly from LSI technology (Torrero, 1978), but guidelines for selecting microprocessors need to be established. This constitutes a major project in itself which is beyond the scope of this report, however, one major architectural feature is worth discussing in general terms. It is the concept of "bundled" versus "orthogonal" architecture. In orthogonal architecture mutual independence is maintained in operating levels and major subsystems (Klingman, 1977). Major functions are separated into eigen-vector-like functions that can be scaled with no effect on neighboring functions. On the other hand, in a bundled architecture, extension of any particular resource entails the automatic extension of unrelated resources. We recommend, in general, selection of microcomputers with orthogonal architectures.

Radiation hardness is a characteristic that the semiconductory industry is not likely to improve without support from NASA and DOD. Special geometries are necessary to achieve radiation resistance. NMOS microprocessors, for which the most software and application data are available, are radiation soft ($\approx 10^4$ rads(Si)) (Myers, 1977) with moderate power dissipation. Bipolar circuits are fast and radiation hard (in excess of 10^6 rads (Si)) but have heavy power consumption.

Two low power technologies appear attractive. CMOS has moderate radiation resistance (5×10^5 rads (Si) for specially processed parts) (King and Martin, 1977) and the Air Force Materials Laboratory at Wright Patterson Air Force Base is working with RCA to develop a standard high reliability CMOS microprocessor, with production quantities available by 1980 (GSFC, 1977). I^2L devices, in spite of being a relatively new technology, show promise of being relatively radiation hard (10^6 rads (Si)) (JPL, 1977; Stanley, <u>et al.</u>, 1977) and relatively fast with low power. Cooperative efforts involving NASA and DOD are suggested to develop radiation resistent families of microprocessors and other integrated circuits.

During system development, microprocessor technology should be carefully tracked, and in the system design phase, hardware implementation should be delayed if possible while software development proceeds. In this way, the final system can take advantage of hardware advances. It is quite likely that NASA will find it necessary to invest in reliability, packaging and integration (into subsystems) of advanced technology computer hardware for flight.

<u>Software</u> - At JPL, about one-fifth of the budget and one-sixth of the manpower is committed to some aspect of computing, and presumably the percentage is similar at other NASA facilities. Within NASA, the software/hardware cost ratio five years ago was 2:1, and a projection for the Air Force is 10:1 by 1985 (Myers, 1978). Thus, software is and will continue to be a key target for cost reduction through technology automation.

Software productivity is improving only slowly, roughly a factor of two every three years, while hardware effectiveness and costs improve by an order-of-magnitude every three years. Thus, software cost relative to that of hardware is increasing steadily. One way of countering this trend is by using hardware to avoid software requirements where possible (see previous section). For instance, hardware can emulate existing macro-instruction sets, thus saving existing software. Improvements in software engineering can be effected by implementation of modern programming practices, a complex of practices and organization including (from Myers, 1978):

1. Chief Programmer Teams. . . . . . M
2. Development Support Librarian . . M
3. Top-down Development. . . . . . . T
4. Modular Decomposition . . . . . . T

5. Structured Design. . . . . T
6. Program Design Language. . . . . . . T
7. Project Workbook . . . . M
8. Hierarchy/Input - Process - Output . T
9. Structured Programming . . . . T
10. Structured Walk-through. . . . . . . M

M = Management or Process Control Technique

T = Technique or Method

Both NASA and DOD have taken steps to encourage more effective software engineering. Within DOD a fifteen volume series was generated intended to document everything known about structured programming technology (IBM, 1975). NASA has generated at least twelve documents since 1974 (see Myers, 1978, p. 22) dealing with modern programming techniques, and has been successful in having these practices adopted in the different centers. A Boeing study (Black, 1977) showed that on three large programs modern programming practices reduced actual costs over forecast cost by 73 percent. Stronger implementation of these modern methods within NASA is recommended.

The above techniques are effective for large systems, and have been used on large systems to produce software products that run on very small systems. However, software costs and effectiveness for small systems usually depend more on individuals and architecture. Presently, small system operating software is inadequate for software generation. Assembly language debugging aids or user invisible file structures represent better payoff areas for technology development than language design at present.

The availability of 64K and larger memory parts (see Storage Technology section) will affect the small systems profoundly. Low level instruction sets that save 2 or 3 bits of control store but require long routines for moderate data structure will no longer be attractive. Timesharing system usage with the exception of shared data bases will decrease when small, local terminals with 64K bytes of memory are available. But each new design or iteration represents a large software investment in software utility packages, high-level languages, and applications programs. The problem is not with hardware architecture or cost reduction, but software generation, support and reliability. A compiler that provides a good debugging environment for applications programs is generally better than a highly efficient one.

Transportability and commonality of software is an important technology goal, but difficult to achieve. Machine dependence is often dictated by architectural idiosyncracies. Nonetheless, some emphasis should be placed on developing machine independent languages that permit transportation from one machine to another with minimal changes; a notably successful attenpt is embodied in BCPL (Richards, 1969). In a recent survey, Ambler (1978) found that most systems languages surveyed have no facility for linking to other languages other than by cleverly coded assembly language routines designed to accommodate the various linkage conventions.

There is a strong move toward using high order languages (HOL) on microprocessor machines. PASCAL is available on the Intel 8080 and others; BASIC is offered on all of the "hobby" computers. Often the translator is embodied in read-only-memory and run interpretively, leaving the random-access-memory for high level (thus more powerful) instruction storage. NASA should take advantage of and encourage this trend. Additionally, NASA should resolve its HOL standardization approach, settling on one of the following:

1. Continue and expand the use of HAL/S
2. Follow the DOD standard HOL
3. Revert to FORTRAN (1977 or beyond).

Emphasis needs to be placed on problem analysis. For instance, problem statements can be made in such a way that analysis of proposed solutions is possible; e.g., expression in Backus Nauer Form, that provides syntatic, or formal checking. Another example is an extra step in the system design process, called the Algorithm State Machine, successfully used by Hewlett-Packard designers (Clare, 1973). These analytical techniques affect both software and hardware, and are useful in avoiding an unworkable systems design in both areas. They are somewhat unrelated in their approach, but both provide a formal way of expressing problems and solutions. This is perhaps the single most cost effective advance that could be accomplished in the computer systems arena.

Large Scale Integrated Circuits (LSI)

LSI refers to the aggregation of 100's to 10,000's of transistor equivalent circuits on one chip. Very Large Scale Integrated circuit (VSLI) technology is now making its debut. The revolution in microelectronics is in a large part due to these developments. LSI is viewed as the salvation of programs too costly to implement with single elements or medium scale integrated circuits. The promise is great, but so are some of the problems.

Perhaps the most significant technical problem associated with LSI has been characterized by Tudor Finch of Bell Labs as the "tyranny of numbers." Exhaustive testing of a single part of relatively simple architecture requires 45 minutes at a rate of one test per fifteen nanoseconds; if a few peripheral registers are added, the testing would require 91 years (Stieglitz 1978). Complex systems of the same kind defy exhaustive testing.

Test strategies are being developed to counter this problem to a degree and deserve more attention. Suggestions to decrease testing costs include a more universal simulation language and better models of devices supplied by the manufacturer.

The testing problem has prompted new design approaches to LSI circuits which also usually improve yields. In one such approach, all internal storage elements were designed to operate as shift registers; sequential logic was transformed into combinational logic during test generation, the latter easier to test (Eichelberger and Williams, 1978). Another promising technique employs the replication of identical cells throughout the chip; logic functions are determined by the selected pattern of interconnects in a manner similar to setting the pattern in a read-only-memory. Programmed logic arrays and allied technologies typify this approach to testable architecture, which may utilize only 10 to 15% of the capability of an individual cell. This is no longer unthinkable because of plummeting cell cost.

LSI will reduce systems costs, but it will also significantly shift the cost distribution of a system. Consider a large, real-time computing environment such as mission control or communications networks. The processor characteristics would include high processing rates, uninterrupted around-the-clock service, long lifetimes and only a few systems per year. The architecture might include battery backup, expensive power switching networks and redundant computing elements. As the number (and relative cost) of devices decreases, the overhead increases, so that there becomes a point of diminishing returns, whereby further device cost reductions do not reduce system costs. The relationship between LSI components and overall system reliability shifts the repairability tradeoff towards more thorough in-house testing. LSI will substantially increase functions per board, but also cost per board and complexity (cost) of test and repair. The cost elements of the repair process include the "fixed" spare parts inventory, and the "variable" diagnostic time and lost revenue costs. LSI increases the ratio between these costs; the total repair cost might be 50-200 times the failed component cost. From the expected failure rate, the system repair cost can be calculated. Since device failure rates are initially high and decreasing in time, the total repair cost can be lowered by longer burn-in times, better pre-assembly screening, etc.

The use of LSI involves other subtle tradeoffs. In reducing the number of silicon components, each increasingly unique, the total volume drops, an anathema to semiconductor manufacturing techniques. Each mainframer has proprietary designs of a relatively small number of systems. The total market for a given component may be less than 1000 pieces. VLSI will climax this problem. The semiconductor manufacturer may not develop these components. Memory requirements will be met by suppliers, but NASA may find it necessary to bear the cost of developing, producing, burning-in and testing of LSI logic parts. Computer aided design will undoubtedly be employed in the 1980s in producing special purpose logic chips, alleviating part of the problem.

In summary, some remedies available to counter problems associated with LSI and VSLI technology are as follows:

1. Architecture utilizing similar parts or cells in both systems and cells, and using perhaps only 10 to 15% of the capability of each unit. The result is improvement in reliability, testability, applicability and lower front-end costs.

2. Improved test strategies, including better simulation software and parts models.

3. Increased emphasis on in-house or contractor burn-in and testing.

4. Use of spares for redundancy and implementation of automatic failure detection.

5. Development tools for VSLI circuit design.

Storage Technology

Storage technology has seen notable advancements in both electronic memories and mechanically accessed memories in recent years. Notable within the realm of electronic memories are semiconductor memories that have enjoyed remarkable technological advancements and market success. These memories have increased capacity (bits/chip) in quantum jumps by a factor of four about every two years, with a cost per bit starting higher but going lower for each advance as shown in Figure 4.

Predictions through 1980 include a 256K part and a 1 megabit dynamic memory before the end of that year. The technology that will support this development is based on three factors:

first, optimum die size is increasing, as yields increase; second, the percentage of total chip area devoted to support circuitry is decreasing; and third, the minimum cell size is decreasing. The first two items are synergetic; as memory size doubles, decode circuitry increases by only one bit; increases in die size then go directly to memory cell area.

By 1982, semiconductor main memories for large ground-based computers are likely to contain a quarter-billion bits at a cost of 0.04 cents per bit. Spacecraft computer memories will typically be scaled down from those of ground based computers due to smaller data bases and processing loads.

In the hierarchy of storage systems needed to span the space from very fast Random Access Memory (RAM) to archival stores there is a decided gap in the middle ground (Figure III-6). Rajchman (1977) thinks that semiconductor RAMs may well bridge this gap by simple extension of capacity. Other technologies are contenders (Figure III-4); of these, Charge Coupled Devices (CCD) and bubble memories show the most promise. Rajchman (1977) has given a technology review and comparison of the competing technologies.

CCDs operate serially (like delay lines) and have access times in the order of 100 times slower than RAMs. These disadvantages are offset by high bit densities (four times that of RAM) and lower cost. Adding inertia to their continued development, CCD arrays operate very well as solid state TV cameras. Many semiconductor houses are producing and improving CCDs, which take advantage of semiconductor production technology.

Bubble memories operate in a manner similar to CCDs, but do not enjoy the benefits of semiconductor production technology. For this reason, their development and acceptance has been much slower than that of CCDs in spite of their earlier invention. Bubble memories are very attractive for space use due to their ability to store without holding power and potential high reliability. Commercial products with 92K bits per chip are available from Texas Instruments, and Rockwell International has produced bubble memories intended for space use for both NASA and the Air Force. At least eleven companies have been working on bubble memories, three of them Japanese. In a review of this technology, Hu (1978) provided the following comparison with competing technologies:

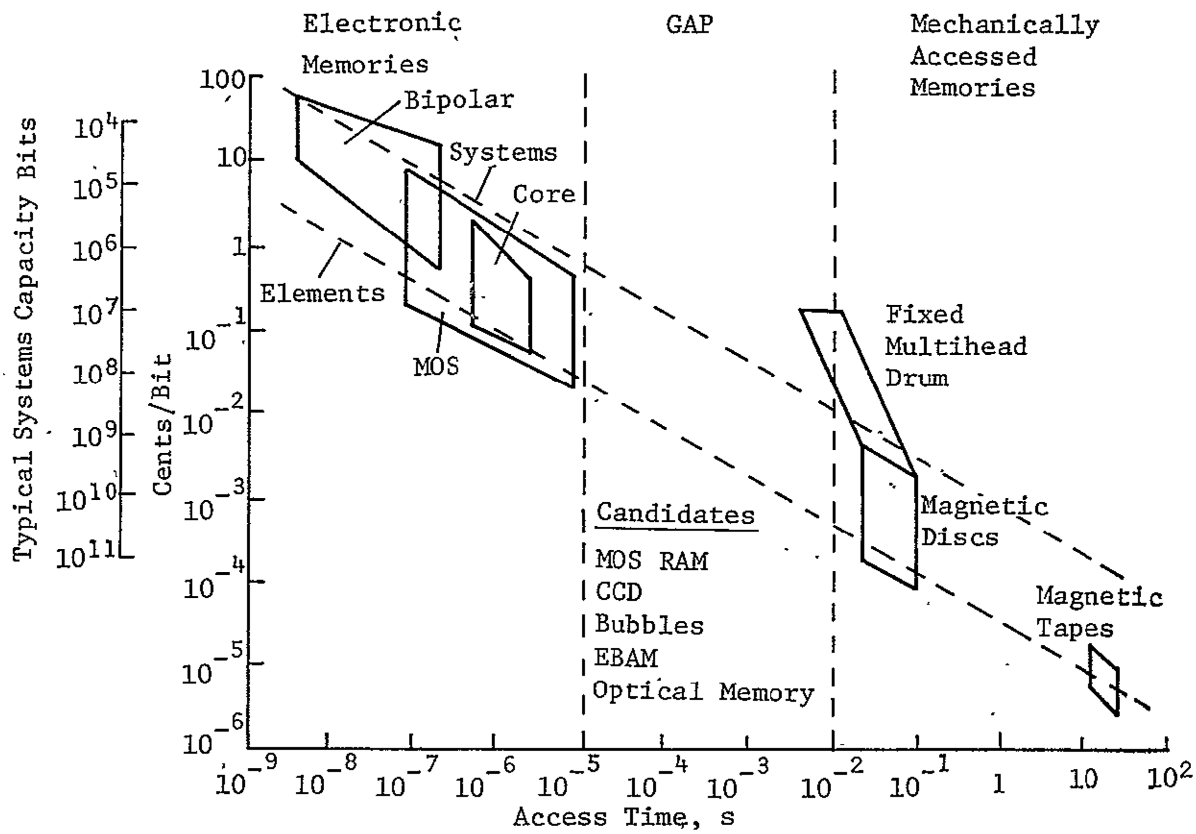

Figure III-6 Memory Cost-per-Bit and Capacity Plotted Against Time in 1977 (from Rajchman, 1977)

	Semiconductor (RAM & CCD)	Magnetic Bubbles	Disk
Cost/Bit	High	Medium	Low
Entry Price	Low	Low	High
Access Time	Fast	Medium	Slow
Transfer Rate	Fast	Slow	Medium
Non-Volatility	No	Yes	Yes
Reliability	Good	Better	Good
Media Removability	No	No	Yes
Physical Size	Small	Small	Large
Interfacing (Electronic Support Circuits)	Simple	Medium	Complex
Other	---	Asynchronous clock rate and stoppable system performance optimization possible	---
Environment Sensitivity	Medium	Good	Medium

Work continues on Electron Beam Addressable Memory (EBAM) (Rajchman, 1977 and Smith, 1978), however, this technology does not appear as promising as the solid-state memories. This is due to its cathode ray tube-like configuration and attendant difficulties with multiple power supplies, size and high voltage requirements, to say nothing of technical difficulties in obtaining a workable system.

For fast mass memories, an attractive technology in principle is the optical memory, also called the holographic store. The storage capacities are estimated at 10^{10} to 10^{12} bits in a moderate space. The great advantage of a holographic store is that it provides an entire mass memory system operating in a true random access fashion (see Rajchman, 1977). The primary problems are the requirement for a laser and materials technology. As yet, there is simply not a storage medium known that is sensitive enough to work at reasonable speeds with reasonable laser power. In spite of this, some NASA support of optical memory is recommended.

Advances in mass storage systems technology will be required to support the large data bases needed in future NASA systems. Mass storage technology advancements are on a healthy growth curve as illustrated in Figure III-7, and will continue because of the inertia of the entire computing industry (Gilmore, 1977), however, some NASA involvement will be required to catalyze the

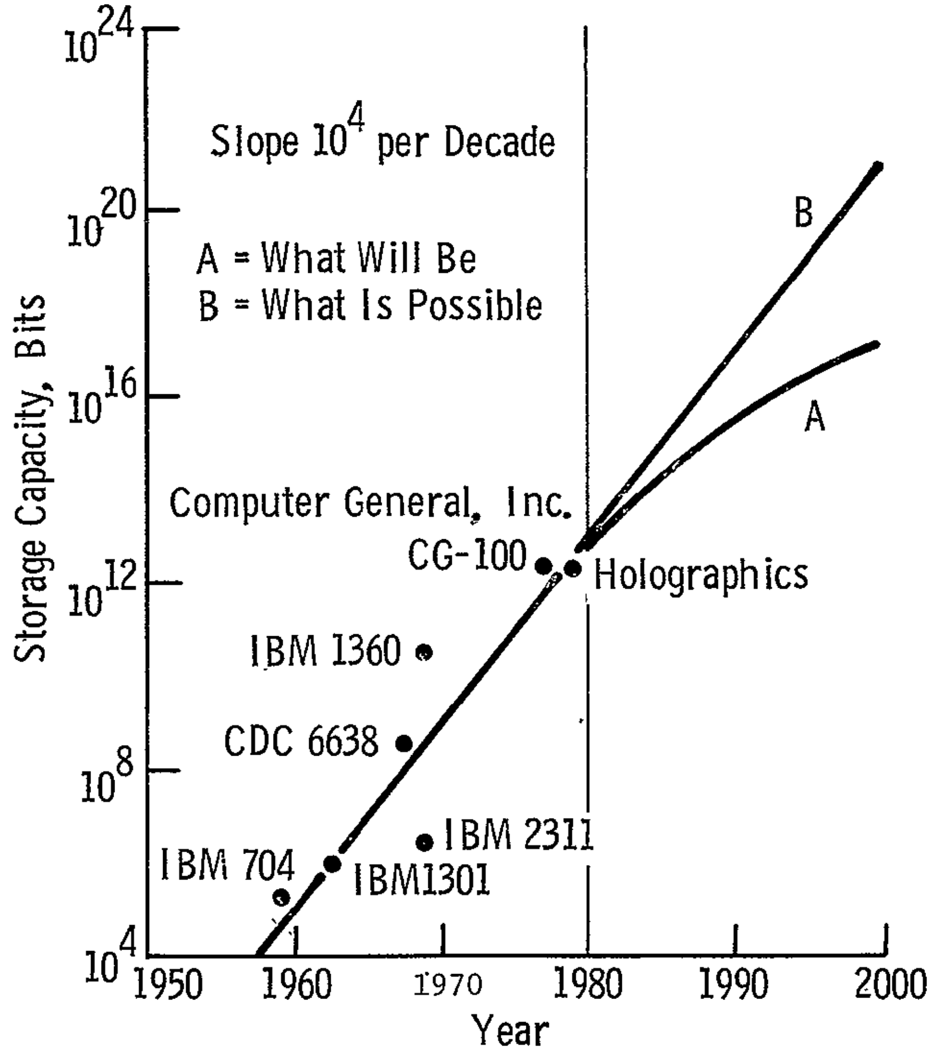

Figure III-7 Capacity of Mass Storage Systems (from Gilmore, 1977) technology advancements necessary to meet projected program needs (see Gilmore, 1977 and Polhemus, 1978).

REFERENCES

Ambler, A. L., "Directions in System Language Design", Digest of Papers, COMPCON, Spring 1978, pp. 40-43.

Black, R. K. E.,"Effects of Modern Programming Practices on Software Development Costs", Digest of Papers, COMPCON, Fall, 1977, pp. 250-253.

Clare, C. R., "Designing Logic Systems Using State Machines, McGraw-Hill, Inc., New York, 1973.

Eichelberger, E. B., and Williams, T. W., "A Logic Design Structure for LSI Testability", Digest of Papers, COMPCON, Spring 1978, pp. 120-129.

Gilmore, C. D., "A User-Oriented Interactive Information Extraction System", Proc. MSFC/UAH Data Management Symposium, October 18-19, 1977, Huntsville, Alabama, pp. 289-305.

GSFC Product Assurance Briefs, Goddard Space Flight Center, Greenbelt, Maryland, No. 77-2, February/March, 1977

Hu, H. L., "Bubbles - A New Magnetic Solid-State Technology" Digest of Papers, COMPCON, Spring 1978, pp. 165-166.

IBM, "Structured Programming Series", IBM Corporation, Guithersburg, Maryland (Contract No. F30602-74-C-0186), 1975.

King, E. E., and Martin, R. L, "Effects of Total Dose Ionizing Radiation on the 1802 Microprocessor", IEEE Trans. Nuclear Sci., NS-24:2172-2176, 1977.

Klingman, E. E., "Comparisons and Trends in Microprocessor Architecture", Computer Design, 16:83-91, September, 1977.

Kurzhals, P. R., "New Directions in Space Electronics", Astronautics and Aeronautics, February, 1977, pp. 32-41.

JPL, "Electronic Parts Engineering Newsletter", Jet Propulsion Laboratory, Pasadena, California, Issue 84, p. 5, May, 1977.

McReynolds, S. R., "A Benefit and Role Assessment of Advanced Automation Needs for NASA", Jet Propulsion Laboratory, Pasadena, California, p. 74, April 14, 1978.

REFERENCES (Continued)

Mayo, J. S., "The Role of Microelectronics in Communication", Scientific American 237:192-209, September, 1977.

"Microcircuit Speed Increase Sought" (Anonymous), Aviation Week and Space Tech., p. 59, October 24, 1977.

Myers, D. K., "Ionizing Radiation Effects on Various Commercial NMOS Microprocessors", IEEE Trans. Nuclear Sci., NS.24: 2169-2171

Myers, W., "The Need for Software Engineering", Computer 11:12-26, February, 1978.

NASA, "A Forecast of Space Technology 1980-2000", NASA SP-387, Washington, D. C., January, 1976.

Noyce, R. N., "Microelectronics", Scientific American, 237:62-69, September, 1977

Polhemus, J. T., "Earth Observation Missions - An Overview of Current Capabilities and an Assessment of Technology Automation Needs", Martin Marietta Aerospace, Denver, Colorado, (Prepared under NASA Contract NASW-3106), 1978.

Rajchman, J. A., "New Memory Technologies", Science, 195:1223-1229, March 18, 1977.

Rennels, D. A., "Reconfigurable Modular Computer Networks for Spacecraft Onboard Processing", Computer, 11:49-59, 1978.

Richards, M., "BCPL: A Tool for Compiler Writing and System Programming", Proc. AFIPS SJCC 34:557-566, 1969.

Smith, D. O., "Electron Beam Addressed Memory", Digest of Papers, COMPCON, pp. 167-169, Spring, 1978.

Stanley, A. G., Mallen, W., and Springer, P., "Ionizing Radiation Effects on SBP 9900 Microprocessor", IEEE Trans. Nuclear Sci., NS-24: 1977-1978.

Stieglitz, C. B., "An LSI Test Overview", Digest of Papers, COMPCON, pp. 97-105, Spring, 1978.

REFERENCES (Continued)

Torrero, E. A., "Solid State Devices", IEEE Spectrum 14:48-54.

Torrero, E. A., "Solid State Devices", Spectrum 15:36-40, January, 1978.

Turn, R., "Computers in the 1980s and Beyond", IEEE Computer Society, Digest of Papers, Spring COMPCON, 1978, San Francisco, California, pp. 297-300.

Turn, R., "Computers in the 1980s", Columbia University Press, New York and London, p. 257, 1974.

Verhofstadt, P. W. J., "VLSI and Microcomputers", Digest of Papers, COMPCON, pp. 10-12, Spring 1978.

IV INFORMATION GATHERING

This assessment of technology automation needs arising from Earth observation missions, addresses, in part, a preliminary mission model derived from the Post LANDSAT-D Advanced Concept Evaluation (PLACE, 1978) and a number of other sources.

It is divided into sections including system considerations applications, and capabilities of platforms and sensors; summary recommendations are given in the first of these.

System Considerations

Primary Technology Automation Needs and Recommendations - As we move into the shuttle era, there will be a revolutionary change in earth observation missions. The reasons for this are three-fold: 1) the valuable experience base derived from programs to date, improving our insight concerning mission requirements and technology needs, 2) the recent or imminent availability of a new and versatile array of sensor and platform systems (e.g., LANDSAT-D, SEASAT, microwave sensors in space) coupled with steadily improving data management capabilities; 3) the frequent flight capabilities offered by Space Shuttle.

As we move into this new era, the primary challenges shift from data collection to data analysis, and from single problem solution to structuring complex multi-disciplinary systems. As an example, early image processing development emphasized geometric corrections and radiometric enhancement; the next major thrust concerned pattern recognition or classification. Now substantial efforts are needed on data base integration and modeling in order to provide truly effective earth observation systems to the intended users. (See Figure IV-1; adapted from Gilmore, 1977.)

Undeniably, additional development work is needed on sensors, platforms, improved launch facilities, data management, and electronics, but these needs are relatively well understood and the evolutionary progress in these areas is satisfactory. For example, reducing image processing time is a worthwhile goal justifying continued effort, and progress in this area is gratifying as shown in Figure IV-2 (Gilmore, 1977). The development of large data bases and their integration into earth observation systems can be accomplished with modest effort on the part of the system architect due to the evolution of mass storage systems and interfacing hardware. The development of adequate models will run a different course, however, because modeling is a relatively neglected discipline as it relates to earth observation missions. In the following paragraphs, these and other needs will be related to a

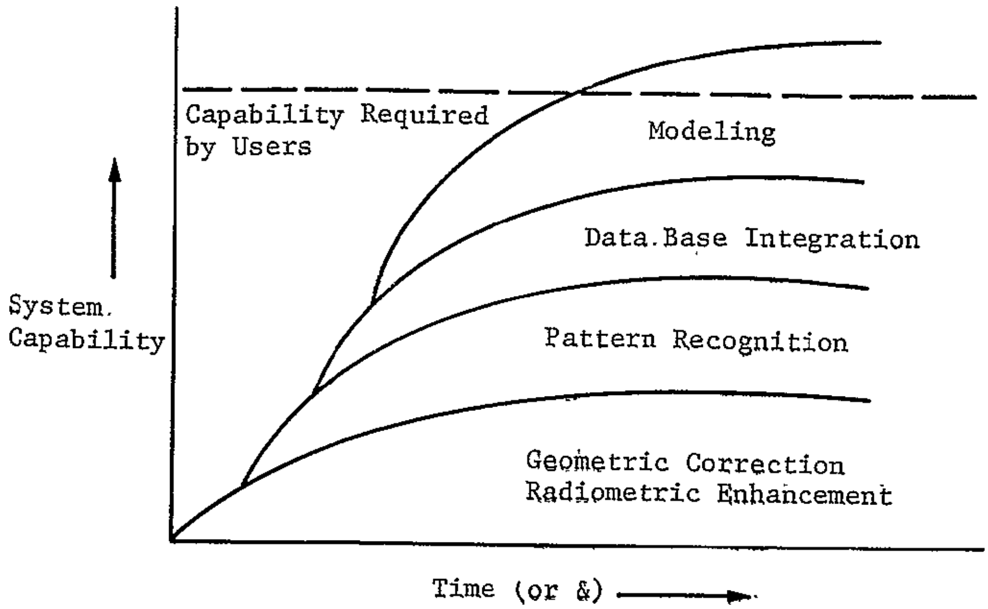

Figure IV-1 Image Processing Development

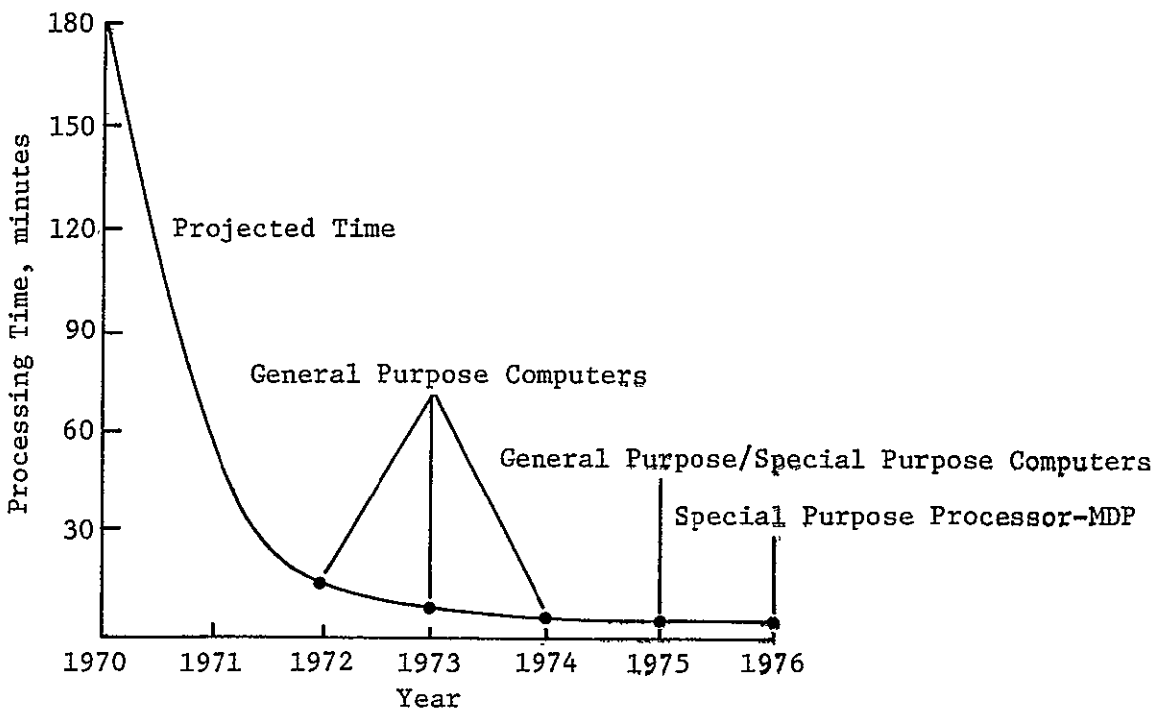

Figure IV-2 Reduction in Image Processing Times for Landsat Data generalized complex earth observation mission to support the recommendations given in Table IV-1.

Table IV-1 Primary Technology Automation Needs for Earth Observation Missions

1.	Improved Models
2.	Integration of Large Data Bases
3.	Rapid Transfer of Remote Sensing Results to User
4.	Substantial Improvements in Pointing and Tracking Capability
5.	Automated Landmark Acquisition and Tracking Capability
6.	On-board Processing and Pre-processing of Data
7.	Automated Atmospheric Effects Correction
8.	Variable Resolution Sensor Systems
9.	Improved Algorithms and Techniques for Scene Analysis
10.	Small Intelligent User Terminals with Display Capability

The Earth Observation System - A generalized Earth Observation System (EOS) is shown in Figure IV-3. The role of such a system is to aid in better performing some act or activity, whether it be crop prediction, weather forecasting, geological mapping or pollution monitoring. The system shown implies an impact of the action on what is being measured and is, therefore, the generalized complex system. An EOS used solely for mapping would be archival and somewhat simpler than that shown, however, the most challenging EOS applications will have all of the elements shown.

Early in the history of Earth Observation Missions, the emphasis was on understanding how to make meaningful measurements, hence how to make sensors and platforms measure what we thought was needed, and later the emphasis was on analysis leading to prediction. Models, data bases, and the analyst have either been given low priority or not included until large interactive systems such as the Large Crop Inventory Experiment (LACIE) pushed toward operational systems and user acceptance. The Screwworm Eradication Data System (SEDS) never had an analyst in the loop, hence, the loop was never closed and the system never became operational, although very valuable results were obtained in the areas of measurement, analysis and data base construction, particularly regarding temperature mapping.

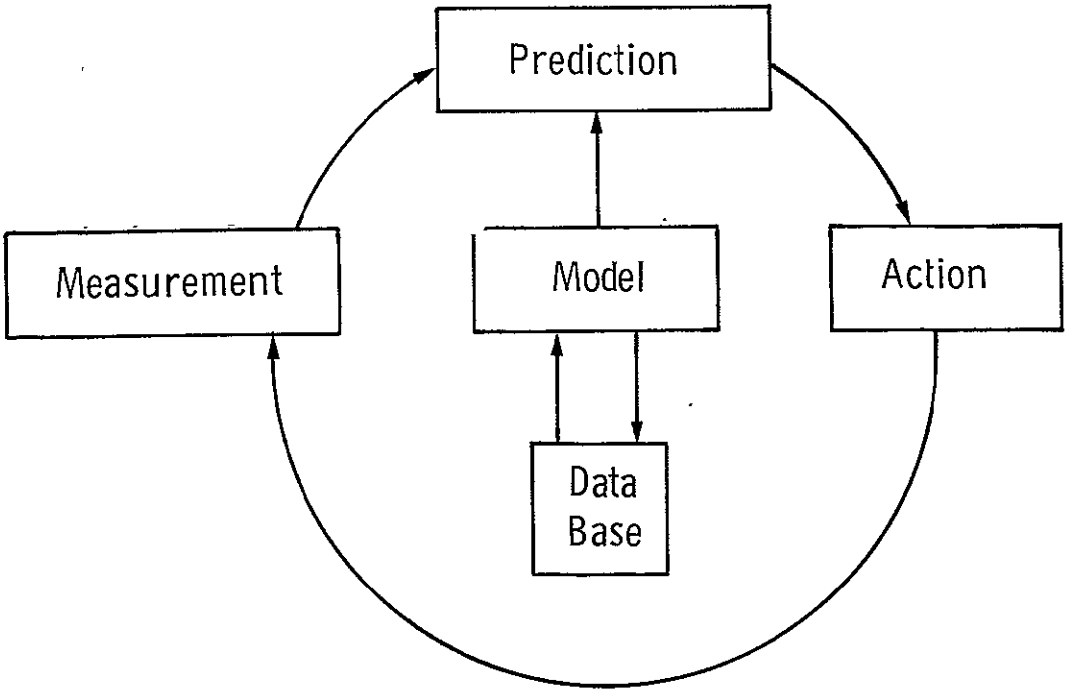

Figure IV-3 Generalized Earth-Observation System

A technology development needed for effective analyst participation is a small inexpensive user terminal along with the communications system to support it. This should be an intelligent terminal with display capability. The need for and evaluation of such a terminal has been described by Kurzhals (1977). A key element pacing this development is the technology of low-power flat displays; a rather optimistic review of the state of the art has been provided by Torrero (1978).

The role of the analyst in an EOS is shown in Figure IV-4. Analyst implies one or more people to perform all of the functions shown plus possibly restructuring the entire system. The ultimate technology automation goal is to eliminate the analyst, and this is the thrust of this section. Each of the elements of the generalized Earth Observation System (except action) will be examined as to technology automation needs. The block labeled "prediction" will be discussed as analysis.

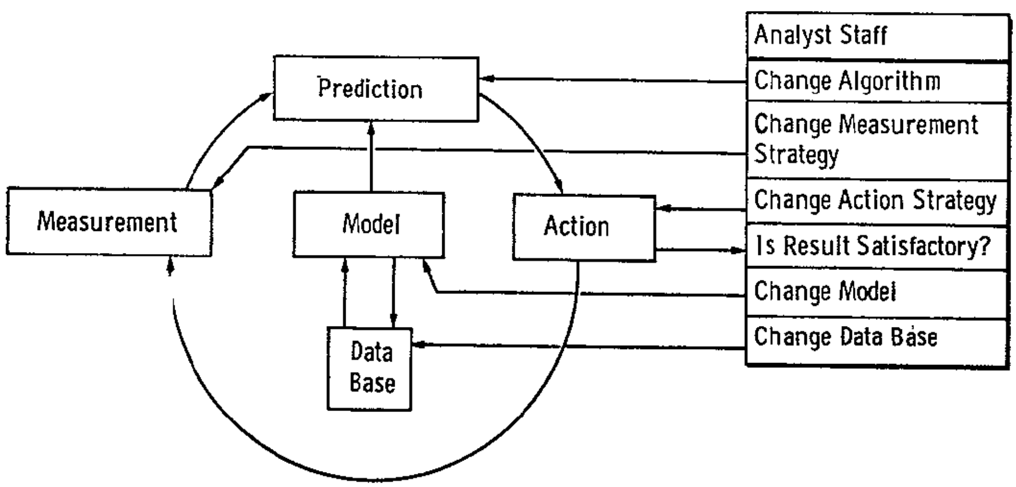

Figure IV-4 Analyst in an Earth-Observation System

Models - Virtually every in-depth study dealing with earth observation from space contains a recommendation that improved models be developed. The pleas for better models concern virtually every facet of Earth Observation Missions, including such diverse things as atmospheric parameters, soil characteristics, crop growth, insect life cycles, viewing and solar parameters, satellites and sensors, data processors, and data analysis algorithms (Idso et al., 1975; LaRocca, 1975; Chism and Hughes, 1976; Gilmore, 1977; Maxwell, 1976; McReynolds, 1978). Modeling has so far seen comparatively little emphasis in earth observation programs, perhaps partly because successful modeling requires the combined efforts of the system architect and the user, is a multidisciplinary effort, and is technically difficult.

The most challenging earth observation remote sensing systems will typically involve complex models. In some instances, simple models will suffice, but it is evident that multi-parameter, multi-dimensional models will be required for the successful management of complex phenomena. Development of these complex models and associated large data bases have been identified as major future thrusts in the remote sensing program (Gilmore, 1977). User interaction with the model and entire system for that matter, is essential for the successful application of remote sensing to complex problems during development. In fact, any such system that does not require user iteraction on a continuing basis during early stages is predisposed to eventual failure.

No matter what the intended end use of the model, it is essential that during design, the purpose, goals, and acceptability criteria be defined so that the right kind of model is developed. There are many different sorts of models for different usages and successful model building is not a haphazard endeavor (Maki and Thompson, 1978; Draper and Smith, 1966).

Ecosystem models are perhaps the ultimate challenge to the model builder, and they are very complex. Generalized descriptive models of biological processes (e.g., Verhulst's logistic equation, Lotka-Volterra predator-prey model) theoretically relate to the complex phenomena of organism population dynamics, but in practice fall far short of describing actuality. Mimic models have been developed in recent years that permit many more variables to be accommodated (see Huffaker, et al., 1977) but often these were not constructed in a way that would initiate interaction between experiment or measurement and model building (Conway, 1977). Biological models with an open structure, relatively easy to comprehend and modify, have relied heavily on the Leslie matrix (Leslie, 1945); this model permits incorporation of several independent parameters and breaks up the phenomena into time segments, but assumes a single start time for all time-dependent functions. The fundamental two-dimensional Leslie matrix can be replicated, with each matrix having a different start time, and additional parameters incorporated in each matrix; one such n-dimensional model has been embodied in a computer program written in FORTRAN and applied to pest control (Watson, 1973; Butler and Watson, 1974). While the Leslie matrix model has a neat and conceptually clear orthogonal structure, some workers have not found it as effective as desired in stimulating feedback between model building, experimentation and field work. Birley (1977) believes that the answer lies in transfer function models. On the brighter side, several good books on ecosystem modeling have recently appeared (Hall and Day, 1977; Gold, 1977).

The most effective model for complex remote sensing applications will have the following characteristics:

1. Orthogonal structure
2. Modular architecture
3. Conceptually clear
4. Mimic of the phenomena
5. Intensively user interactive, at least in development
6. Data base coupled

A model having these features will be heuristic, easily modified to incorporate new parameters or data, and easy to program and embody in computer systems. At the same time, those parameters whose relationships to the phenomena have been satisfactorily characterized can be updated through the data base. The system will, therefore, continually evolve toward the ideal of a data base driven simulation of the phenomena being managed, with the result being an automated system rather than a strongly human interactive system.

<u>Data Bases</u> - The uses of data bases are severalfold: they can supplant, augment, or hold remotely sensed data, and interact with models in systems.

None of the sensors except microwave can get ground coverage where clouds block the line of sight. Data bases can supplant the hoped-for remotely sensed data for some measurements. For instance, this problem was encountered with ITOS-VHRR during the Screwworm Eradication Data System (SEDS) program; daily ground meteorological station data and a so-called ΔT field model of temperature variations due to altitude away from weather stations) were held in a data base and used to fill in cloud cover areas. The success of this data base usage suggests that a model and data base could supplant remotely sensed temperature data for extended periods, perhaps months, with remote sensing used to periodically check and recalibrate the system (Giddings, 1976). Other augmentation data must or should be held in data bases; e.g., daily maximum/minimum temperatures, ground slope, elevation, and soil types.

Cloud-free composites of remotely sensed data can be entered into a base to delineate ecological zones over a broad area. There is presently no operational remote sensing system furnishing soil moisture measurements with acceptable accuracy, therefore, soil moisture must be mapped into a data base if it is needed in a system. Using, say, 25 $mi.^2$ pixels, the entire surface of the earth can be stored on one 9-track computer tape. Remotely sensed data needs to be entered into those data bases that interact with models, or serve as landmark recognition maps. In the interactive data base-model system, a number of parameters will need to be stored, perhaps multi-dimensionally.

Gilmore (1977) has identified the development of very large, flexible data bases as a major thrust. Management of data bases will require technology advancements (Scheuermann, 1978). The data processing industry will provide part of the needed storage capability through their inherent inertia (Figure IV-5); however, some directed effort will be necessary to structure these for the Earth Observation Missions, and perhaps to augment their capabilities.

Measurement - A summary of the recent, current and planned earth observation sensors and platforms is given in Table IV-7 at the end of the chapter. Included are most of the multispectral scanners, radiometers, scatterometers and microwave sensors; excluded are most camera systems using visible range firlm, although these have considerable utility for some applications.

The environmental sciences constitute a very important user area for earth observation data as evidenced by the emphasis on agricultural usage for LANDSAT-D. The principal forms of data available for the environmental sciences are summarized in Table IV-2 (Giddings, unpublished). Several important capabilities (e.g., microwave sensors on satellites, geosynchronous platforms, thematic mapper) are just becoming available at the present time, and their adequacy for critically needed measurements (e.g., soil moisture) is not yet known. Some recommendations offered concerning sensor and satellite system development must be considered

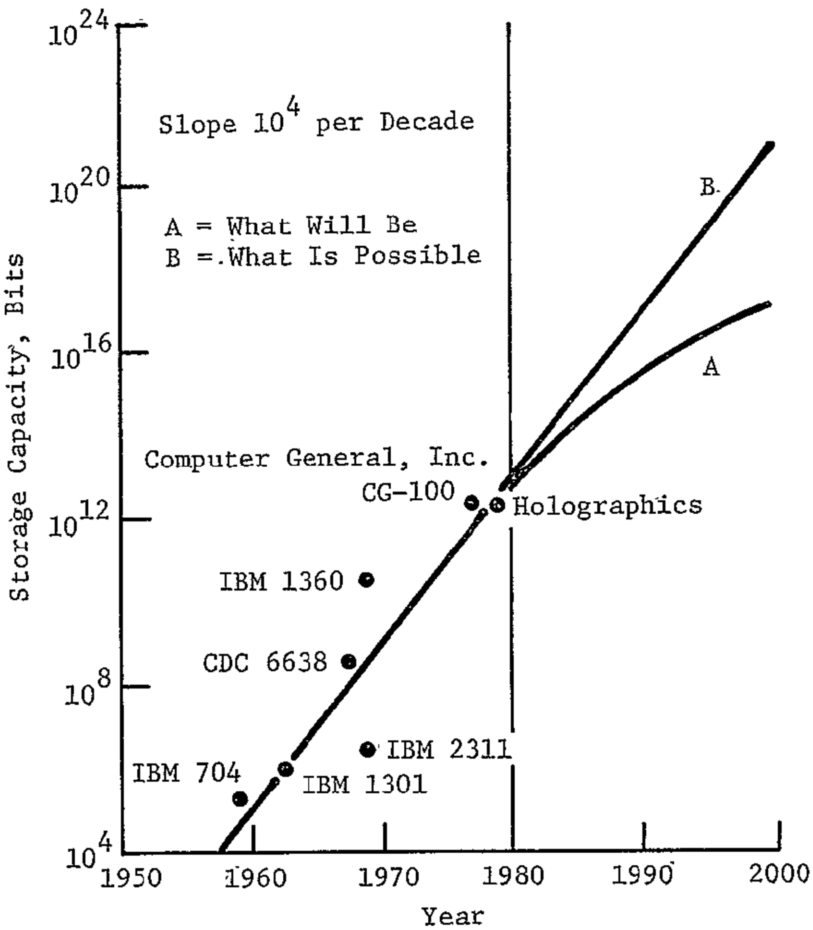

Figure IV-5 Capacity of Mass Storage Systems

Table IV-2 Forms of Data Available for Environmental Sciences

General Type of Data	Satellite Sources	Date	Best Spatial Resolution	Form Furnished to User	Conversion by User	Some Applications
Single-image Photography	Gemini	1965-66		Color and black and white photography	None	Examine surface features
	Apollo					
	Skylab	1973-74	11 m			
	Apollo-Soyuz	1975				
	(Shuttle)	1979-future				
Multispectral Photography	Apollo-9 Skylab	1973-74	24m	Color and black and white photography	None, but black and white images can be combined	Examine surface features
Single Channel panchromatic visible and near-infrared images from scanning radiometers	ESSA series	1966-1975				Clouds
	TIROS-1 to TIROS-10	1960-1965				
	TIROS-M to ITOS/ NOAA-5	1970-1978	0.9km/4km	Photographic images	None	Regional surface features
	ATS-1 and ATS-3 (geosynchronous)	1969-1975		Computer tapes	Flexible conversion for display on screens or as photographs	
	SMS-GOES series (geosynchronous)	1974-future	1 km			
	Nimbus series	1964-future				
	DMDP		0.6 km			
	LANDSAT-C vidicon	1978-future	40 m			
Single channel thermal infrared IMAGES	ESSA series	1966-1975				Identify clouds
	Early TIRON series	1960-1965		Photographic images	None	Classify clouds
	TIROS-M to ITOS-H/ NOAA-5	1970-1978	7.5 km			Distinquish topography
	TIROS-N and beyond	1978-future	4 km	Computer tapes	User-supplied algorithm	Monitor radiometric temperature Approximate surface air temperature Extend surface temperatures
	SMS-GOES series	1974-future	7 km			
	Nimbus series	1964-future	8 km			
	DMSP		0.6 km			
Single channel humidity images	Nimbus -4, -5, -6 and Nimbus G	1970-future	22 km	Photographic images Photographic mosaics Computer tapes	Various	Monitor troposphere humidity; extent moisture analysis (400 mb); extend wind field analysis
Multispectral visible, near-infrared, and thermal infrared images from scanning radiometers	LANDSAT-1, -2	1972-future	80 m	Three-channel false color composite photographic images	None	Surface features
	LANDSAT-C	1980s	57 m visible 171 thermal			
	LANDSAT-D	1980s	30 m visible 120 m thermal IR		Merging algorithms; Classifying algorithms; clustering algorithms; others	Wide variety of analysis of surface features
	SEOS	1981	100 m	Computer tapes		
	Skylab	1973-1974	80 m			
	TIROS-N and beyond	1978-future	1 km	Various		Analyses of surface features
	Nimbus-G	1978-future	0.8 km	Various		
Microwave Images	ESMR from Nimbus-5 and -6	1972-future	25 km	Photographic images and Computer tapes	Various	Ice and snow detection Heavy precipitation Soil moisture
	Other Nimbus-6	1975-future	150 km			
	Nimbus-G	1978-future	--			
	SEASAT	1978-future	--			
Special Applications of several images, not limited to one satellite source	Any one or several sources of taped imagery data	--	--	Three-channel color composite photographs	None	Zone discrimination
				Computer tapes	Registration to common map, unser-supplied algorithms; Clustering algorithms; Classification algorithms, etc.	Zone discrimination Cloud-free composit
Multiband spectrometers infrared, microwave	ITOS-NOAA	1970-1978	55 km	Raw data or converted data	If raw, apply algorithm	Vertical temperature profiles Vertical humidity profiles; Other profiles Stratispheric aerosole and ozone
	DMSP	Current	22 km, etc.			
	(TIROS-N and series)	1978	22 km, etc.			
	Nimbus					
	AEM-B					
Tabular Data	Nimbus ITOS-NOAA, others	--	--	Tabular data	User-furnished algorithms	Solar protons, X-rays, Ozone, tropical winds Radiation budget, other
Delayed Data	LANDSAT SMS-GOES TIROS-N, etc. others	Current	--	Various	User-furnished conversion	Any environmental parameter that can be measured and telemetered automatically

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as tentative pending analysis of the data from these new systems.

In general, the measurement capabilities flown today exceed our ability to analyze and utilize them, yet finer bands and resolutions are called for (e.g., Kibler, 1978). Some sensor requirements not yet met will be identified in the applications areas discussed in next section. Measurement and modeling of atmospheric effects on spectral responses is a technology need area; this is one of the primary problems in signature extension and automated classification. Ratioing techniques are also helpful.

The advent of the Synthetic Aperture Radar (SAR) forces the issue of selective data gathering. The bit rates from the SAR range from 10 to 100 mbs, far too high for on-board storage of large scenes. Already the need for selective data gathering was apparent with LANDSAT, where data is offered in 10,000 mi^2 blocks and a user interested in a few hundred to a few thousand acres could not justify the costs of the tapes and processing (Maxwell, 1976). Environmental sciences programs such as identification of desert locust outbreak areas, mosquito breeding sites, timber survey and damage assessment, rangeland classification, coastal pollution and subsidence monitoring, etc., all need selective data coverage and variable resolution. The technology requirements generated are precise pointing and tracking, landmark identification and tracking, zoom capability, and onboard data processing, including data compression, coding, and scene analysis. The pointing accuracies required for EOS sensors is shown in Figure IV-6 (from GE Space Division, 1976).

Mundie, et al. (1975), in a comprehensive study on design considerations for advanced scanners, concluded that the key to increased angular resolution is the employment of large numbers of elements in the detector arrays. The resulting increased data load increases the data processing requirements, a matter treated above.

Analysis - Much of the analysis that is done now on the ground should be done onboard. A rather obvious need is for scene rejection or selection on the basis of cloud cover. Probably 40% of hand-held photography from space and 50% or more of LANDSAT imagery is useless because of cloud cover. Onboard preprocessing of imagery could obviate data gathering when cloud cover is above a threshold percentage; cloud cover is not difficult to identify with presently flown sensors.

Onboard processing should be provided to compensate for altitude and off-axis effects on scene pixels. Presently such

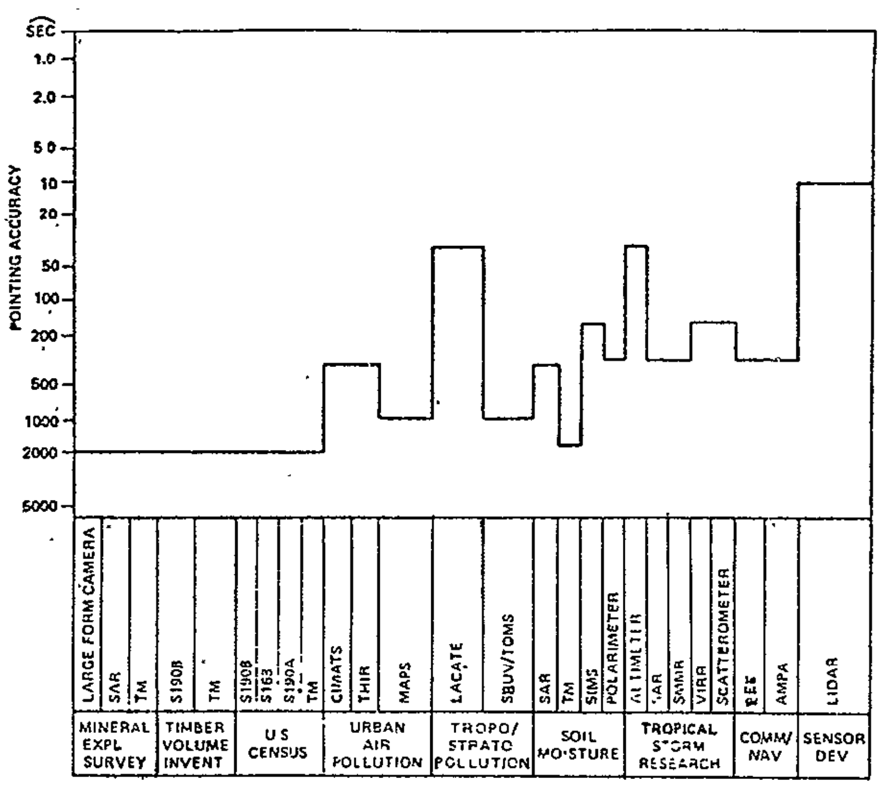

Figure IV-6 EOS Sensor's Pointing Accuracy effects cause hybrid pixel edges that are difficult to deal with in subsequent scene processing and analysis. Scene data position tags would also be useful.

A distributed onboard computer network has been developed at JPL (Rennels, 1978) that incorporates many of the desirable features discussed in the section on electronics, including trading hardware for software where possible, operating dedicated computers below capacity and providing redundant hardware for fault tolerance.

Improved algorithms and/or transforms along with accurate atmospheric models are needed before signature extension will be successful. It would be helpful to the analyst if the transforms were conceptually clear, perhaps amenable to factor analysis, at least in the early program stages. The transforms most commonly used now are the principal components, Kauth (tasselled cap rotation matrix) and cannonical.

Applications

In this section, application areas are examined. Table IV-3 shows the application areas and related sensors and satellites. Much of the data in this section was based on Chism and Hughes (1976); however, many other sources were used in addition, and the latter will be referenced when appropriate.

Agriculture - The major directions in agricultural applications relate to crop production estimates and the assessment of threats to food and fiber production. Accurate and timely knowledge of world crop production is important for global management of resources, a need partly brought about by burgeoning populations in under-developed nations and evidence of climatic change.

Two basic factors affect crop production; acreage and yield. In principle, both of these can be measured by remote sensing techniques involving emission or reflection of radiation from the biomass. The radiation characteristics depend on canopy cover, leaf angle or projected area, and color and vigor of the plants. Atmospheric effects modify the radiation before it reaches the sensors, however, principally due to aerosol, water vapor and ozone; solar and viewing angles also modify the radiation.

Workers at Colorado State University have had reasonable success in estimating biomass of rangelands and separating as many as nine classes within the scene (Maxwell, 1976; Tucker and Maxwell, 1976). Of the sampled variables, leaf water provided the best indicator of alive, green and photosynthetically active biomass.

Table IV-3 Earth Resources Applications Area and Related Sensors/Satellite Systems

Application Area	Multispectral Scanners			Microwave			Photographic Systems
	Visible	Reflective IR	Thermal IR	Radiometers	Scatterometers	SAR	
Agriculture	Landsat-1, 2, C&D SEOS	Landsat-1, 2, C&D SEOS	Landsat C&D SEOS	Seasat-A	Seasat-A*		
Cartography						Seasat-A Shuttle SAR	S-190B Equiv. Or Better
Wetlands Or Coastal Studies	Landsat-1, 2, C&D SMS/BOES		Landsat C&D SMS/GOES		Seasat-A*	Seasat-A Sarsat Shuttle SAR	
Damage Assess.	SEOS	SEOS	SEOS		Sarsat	Seasat-A Sarsat Shuttle SAR	S-190B Equiv. Or Better
Forestry	Landsat C&D	Landsat C&D	Landsat C&D		Seasat-A*	Seasat-A Sarsat Shuttle SAR	S-190B Equiv. Or Better
Geology	Landsat-1, 2, C&D	Landsat C&D SEOS		Seasat-A		Seasat-A Sarsat Shuttle SAR	S-190B Equiv. Or Better
Land-Use	Landsat-1, 2, C&D	Landsat-1, 2, C&D					S-190B Equiv. Or Better
Oceanography & Hydrology	Landsat C&D Seasat-A SMS/GOES		Landsat C&D Seasat-A SMS/GOES	Seasat-A	Seasat-A	Seasat-A Sarsat Shuttle SAR	
Soil Moisture			Landsat C&D	Seasat-A*			

*Potential Or Limited Utility

Threat assessment has been largely limited to insect pests (desert locusts, screwworm flies, mosquitos) through the sensing of habitat, temperature, and moisture conditions. These programs have met with limited success, primarily because of inadequate models and/or lack of biological data. The desert locust program has as a primary input soil moisture estimation through analysis of rainfall; it is expected that the new microwave sensors may help with this.

Langrebe (1976) recommends improvements in remote sensing systems for crop estimating, consisting of: 1) significantly increasing knowledge of the variability of the scene, 2) more sophisticated sensors incorporating improvements balancing spectral, spatial and temporal aspects of the scene, 3) more complex data processing algorithms, 4) increased use of ancillary data, 5) more knowledgeable use of man in the analysis process, and 6) a more suitable array of output products to match user needs.

Recommendations for improvements in remote sensing systems for agriculture involve: 1) improved models of the process or scene, 2) more sophisticated sensors systems (finer resolution, more and finer spectral bands, and more frequent coverage), 3) more rapid communication of raw remote sensing data to users, and 4) more elaborate data processing and analysis techniques and algorithms, including spatial and temporal aspects, ratioing, greater use of ancillary data, and increased analyst participation. The sensor and spacecraft improvements are at least in part embodied in systems scheduled to fly by the early 1980s.

Atmospheric Corrections - The atmosphere generally modifies the radiation between source and sensor. In small areas where ground truth is available, atmospheric effects can often be calcluated and visible and near infrared radiation normalized to a large degree. Signature extension is difficult to impractical, however, unless atmospheric data are known. Water vapor primarily affects bands in the 0.8 - 1.1 μm region, and seriously degraded the classification of LANDSAT-1 data at medium to high horizontal gradients. Relatively small changes in sun angle and haze level substantially reduce classification accuracies.

Haze and thin clouds can both be corrected for in roughly the same way, and both will be referred to as haze. Methods of determining haze include 1) solar radiometers, 2) minimum value method (dark targets such as deep lakes or wooded areas have a minimum value in each line of LANDSAT channel 1; haze increases brightness), 3) channel correlation method (LANDSAT channels 1 and 2 (MSS3 and MSS5) are highly correlated; haze affects MSS4 more than MSS5, so a scatter plot and regression indicates haze level), 4) LIDAR (laser) scanner. Once the haze level has been determined, an atmospheric model is necessary to relate target reflectance to observed radiance. At least one algorithm has been proposed to correct haze and sun angle using only LANDSAT MSS data (Lambeck, et al., 1978). Algorithms involving transforms, clustering, and maximum likelihood classification have been used for classifying agricultural scenes without determining haze levels, with varying degrees of success.

Atmospheric corrections needed in the mid and far infrared are not as well defined as the visible and near infrared, however, the primary corrections needed are 1) eliminating pixels (picture elements) containing visible cloud cover and relatively invisible high cirrus, and 2) a final correction for water vapor and emissivity in the 10 μm to 12 μm atmospheric window, perhaps through ratioing. Correction techniques have been discussed in a summary article by LaRocca (1975), who states the need for a better atmospheric model.

Cartography - Despite the importance of maps, a substantial portion of the world is inadequately mapped. A recent United Nations' study estimated the worldwide availability of topographic maps as follows:

Map Scale	Percent of the World	Percent of the U. S.
1:1,250 - 1:31,680	6.0	40.5
1:40,000 - 1:75,000	24.5	44.7
1:100,000 - 1:126,720	30.2	20.8
1:140,000 - 1:253,440	92.0	100.0

The rate of obsolescence of maps is an even greater problem. Map production cycles of 3-4 years render many maps obsolete at the time of publication.

Satellite mapping is obviously a great interest because of the large area covered by each frame; however, geometric fidelity becomes a problem. The following characteristics recommended for a satellite-borne camera to obtain the necessary resolution are 1) calibrated focal length and principal point, 2) radial and decentering lens distortion, 3) film flatness, and 4) internal reseau system. The recommended space mapping system to support construction of photomosaics at scales of 1:24,000 to 1:1,000,000 would consist of 1) a 12-inch or 18-inch focal length camera with a 9-inch x 18-inch metric frame format, and 2) a 24-inch focal length panoramic camera with a 4.5-inch x 45.25-inch format.

Mapping cameras are slated to be flown on Space Shuttle and should provide excellent results.

Side looking radar (SLAR) has an all-weather mapping capability with resolution dependent primarily on pulse length and is expected to provide a powerful new tool for space mapping.

Coastal Studies - The coastal zone is the most varied physiographic unit on earth, spanning almost all biotic and abiotic conditions. The importance of estuaries has been established as breeding grounds for a major part of the human food derived from the ocean. Pollution monitoring, subsidence, vegetation and edaphic changes, and coastline shifts are only examples of remotely sensed characteristics important for coastal studies.

LANDSAT 1 and 2 coastal zone studies using visible and near infrared bands demonstrated acceptable environmental detection and marginally acceptable identification capabilities. Low altitude aircraft studies provided thermometric studies useful in environmental detection and identification. Aircraft studies with active microwave sensors have proven the potential of these sensors in environment discrimination.

Coastal vegetation is ordinarily composed of mixture classes and pure stands grading into one another. Automatic classification of coastal wetlands is thought to be possible; however, an integrated approach involving simultaneous spacecraft and aircraft remote sensing with resolutions on the order of 5 m to 25 m has been suggested. In the near infrared recommended bands are 0.70 - 0.80 μm, 0.80 - 1.10 μm, 1.10 - 1.60 μm and 1.60 - 2.25 μm; in the thermal infrared, the 8.00 - 14.0 μm band is recommended.

The technology needs for coastal studies primarily concern high resolution imaging systems, with the concommitment high data load.

Forestry and Range - The productivity of forests and rangelands is generally low in spite of good capacity to produce. Remote sensing has a definite place in managing these resources. The Bureau of Land Management (BLM), in cooperation with NASA, is engaged in a survey program slated to encompass all BLM lands. Maxwell (1975) developed a remote rangeland analysis system. The Forest Service (USDA) has used remote sensing to estimate timber biomass. Table IV-4 gives an overview of information needed for forest management (from Chism and Hughes, 1976); many of these parameters can or have been measured by remote sensing.

Table IV-4 Information Required and Parameters to Measure for Forest Management

Information Required	Parameters
Forest Inventory	Acreage Of Individual Trees Species (Or Species Groups) Location Number Of Trees Tree Size Phenological Data
Timber Yield	Tree Sizes Tree Density Growth Rate Acreage Of Crop Trees Tree Mortality Rate
Forest Stress	Disease Infestation Wildlife Drought Index Air Pollution Competitive Species Flood, Landslides Icing
Understory Inventory	Species Distribution Plant Density Plant Vigor
Soil Capability	Mineral Content Organic Material Content Moisture Content Soil Depth And Horizon Mechanical Properties Natural Drainage
Precipitation	Form Amount Rate Extent Depth Of Snow Fields Water Equivalency Of Snow
Surface Relief And Drainage	Topography Vegetative Cover Location Of Intermittent Streams

Information Required	Parameters
Forest Fire Assessment	Dynamics Fire Size Fire Temperature Wind Speed And Direction Rain Topography Past Fire Assessment Location Of Burn Acreage Of Burn Degree Of Damage
Flammatory Conditions	Fuel Moisture Wind Direction & Intensity Air Temperature Humidity Precipitation Abundance Of Dead Fuels Density Of Brush/Slash Moisture Content Of Organic Debris Topography Prevailing Causative Factors
Grasslands Inventory	Species Acreage Location Plant Vigor Phenological Data
Grasslands Stress	Plant Diseases Infestation Soil Moisture Drought Index Humidity Insolation Wind Air Pollution Animal Grazing Fire
Grasslands Fire Potential	Condition Of Live Vegetation Abundance Of Dead Grass Wind Humidity Precipitation Topography Prevailing Causative Factors

Future remote sensing systems for forestry should include:

Camera	Film - CIR
	Format - 9 x 9
	Resolution - 5 m
Multispectral Scanner:	Bands - .55 to .60 μm
	.66 to .70 μm
	.75 to 1.00 μm
	1.50 to 1.80 μm
	2.10 to 2.50 μm
	9.30 to 11.0 μm
	Resolution - 30 meters
Microwave:	Bands - unknown
	Polarization - unknown
	Resolution - unknown

Recommendations affecting technology needs for forestry applications are:

1. Determine how the data from all sensors can be used together to derive the most useful forest information data. Decide from where the user and decision models will come.

2. Develop models for prediction of biomass, forest composition, fire potential, etc., which make use of visible, near infrared, and microwave data.

3. Conclude what the microwave region can do for forestry, and what are the optimal frequencies, resolution, and polarization.

4. Evaluate the information content of the thermal region by study of the smaller bandwidths, importance of thermal region for classification accuracy, and resolution versus accuracy.

5. Determine what resolutions are required for automatic classification of forest features in the visible, thermal, and microwave regions.

Geology - Remote sensing has been used for geological mapping since the 1920s, in the form of aerial photography. Photogeology reached a peak in the mid 1950s. LANDSAT and Skylab data opened a new dimension in thie field, and radar mapping adds still another dimension.

A recommended sensor package for future geologic mapping from space would have a 250-mile polar drift orbit and carry:

1. A multispectral camera system consisting of at least four cameras with S-190B resolution (12 meters), 9 inch format, and loaded with color, color infrared, black and white panchromatic, and black and white infrared film. This system should be operated to obtain worldwide cloud-free steroscopic coverage at lease once and also on a demand basis.

2. A high-resolution multispectral scanner to obtain data in the near, middle, and thermal infrared region and to be operated only over specific areas of interest.

3. An imaging radar system to be operated only over areas covered with dense vegetation or clouds. The areas which are constantly covered by clouds and have a vegetative cover should be imaged by a radar system at least once.

Oceanography - Oceanography encompasses physical, biological and geological features and phenomena. Of course, the most emphasis is on the physical parameters, with the recent SEASAT being the most powerful system intended for remote sensing. Biological data are available from visible and near infrared sensors on existing platforms. Geological data are obtained largely by inference, not direct measurements.

SEASAT-A was intended to measure and monitor the following data:

1. Ocean wave statistics, heights, lengths, and energy spectra
2. Ocean currents, tides, surges, and tsunamis.
3. Surface winds.
4. Ocean temperature, including the effects of polar ice.
5. Surface topography and surface roughness.
6. General ocean geoid.
7. Ice extent, age, etc.

The sensor requirements for oceanographic measurements are given in Table IV-5 and the SEASAT sensor techniques in Table IV-6 (from Chism and Hughes, 1976).

Technology needs arising from oceanographic missions include data handling methods to cope with the extremely high (10 Mbs to 100 Mbs) data rates from the active radar (SAR); further sensor requirements cannot be adequately estimated until SEASAT results have been analyzed.

Table IV-5

Table IV-5 General Oceanographic Sensor Requirements

Sensor Type	Infrared	Laser	Microwave	Multispectral	Photographic	Radar
Physical Parameters:						
Sea State		X			X	X
Currents	X			X	X	
Surface Temperature	X					
Sea Ice	X			X	X	X
Salinity			X			
Tides	X			X	X	
Sea-Air Interface	X		X			
Biological Parameters:						
Coastal Vegetation	X			X	X	X
Sea Life Areas	X		X	X	X	
Geological Parameters:						
Coastal Changes				X	X	X
Coastal Sedimentation				X	X	
Bottom Mapping		X			X	
Geoid		X				X

Table IV-6 SEASAT Sensor Techniques

Table IV-6

Frequency Range	Active Or Passive	Differentiating Technique	Measurement Types	Seasat - A Payload
Visible And Infrared	Active	Range Processing	Altimetry Ocean Bottom Sounding	Range Antenna
	Passive	Spatial Variation	Images Thermal Maps	Scanning Radiometer
		Multifrequency	Atmospheric Profiles (Water, Thermal)	None
		Spectrometry	Atmospheric And Surface Constituents	None
Microwave	Active	Range Processing	Altimetry Surface Roughness-Wave Height	Radar Altimeter
		Doppler Processing	Backscatter (Winds, Surface Roughness)	Far Beam Scatterometer
		Range And Doppler	Images Wave Spectra Ice Maps	Synthetic Aperture Radar (SAR)
	Passive	Spatial Variation	Images Thermal Maps	SMMR
		Multifrequency	Atmosphere Path Corrections	SMMR

Soil Moisture - The measurement of soil moisture over sizeable areas is a high priority need for diverse earth resources applications such as agriculature, civil engineering, and meterology. Idso, et al., (1975) have discussed a number of these needs, and a variety of measurement techniques.

Three general methods have been used for estimating soil moisture, using radiometric data from the 0.40 to 14.0 μm region: 1) measurement of spectral reflectance or albedo, 2) measurement of visible polarization, and 3) measurement of surface radiometric temperature. The reflectance method detects a decrease in reflectance as soil moisture increases. The polarization method utilizes the sensitivity of polarization of reflected light to soil moisture. The soil temperature method depends on the decrease in day to night soil temperature with increasing soil moisture.

Factors complicating remote sensing of soil moisture are vegetation cover, great differences in the observable soil moisture characteristics for different soil types, and variations in moisture content of soils over a given area. The reflectance and polarization methods share drawbacks: 1) the wide variation in soil albedos make a universal relation (model) difficult to develop unless soil type mapping is included, and 2) a water budget model is needed to relate subsurface water to reflectance or polarization. The temperature method also requires ground truth soil information.

Experimental work is underway at present that will hopefully answer questions as to the practicality of remotely sensing soil moisture. NASA's Heat Capacity Mapping Mission (HCMM; 1978 launch) carries a high resolution two-channel radiometer (0.5 to 1.0 μm and 10.5 - 12.5 μm) for measuring reflectance and surface radiometric temperature. A joint soil moisture experiment involving NASA-JSC, the University of Kansas and Texas A&M University has been conducted using various sensor suites flown on aircraft (see Table IV-7). SEASAT, launched in mid-1978 was expected to return data pertinent to soil moisture measurements.

The General Electric Space Division (1976) has proposed the following as soil moisture mission sensor requirements:

	Polarimeter	Thematic Mapper	SIMS	SAR
Instantaneous Field of View	1°	30 rad (15 meters)	1.1 to 17° depending on freq.	40-100 Km
Total Field of View	Spot Scan within $\pm 60^{\circ}$ nadir	60 Km	$\pm 60^{\circ}$ of nadir	TBD

Microwave sensing of soil moisture is thought to be a viable technique but a number of problems remain. Both passive and active microwave sensors possess the potential of effectively mapping and estimating soil moisture. Their advantages over visible and infrared sensors are: 1) nearly all weather capability due to penetration of non-raining clouds, 2) time of day independence, 3) penetration of light vegetative cover, and 4) soil moisture in deeper soil layers can be sensed.

The problems lie in the areas of 1) microwave sensor systems, and 2) microwave terrain surface interactions and modeling. Studies and measurements with both radiometers and active microwave sensors are needed in the following areas:

1. Simultaneous radiometric measurements with dual polarizations at wavelengths other than 2.8 cm. An optimal wavelength may be found so that the effects of surface roughness, vegetal cover and moisture content in the soil can be clearly separated.

2. Further study on the correlation between ground truth and spacecraft radiometric measurements at 19.3 GHz and 1.42 GHz as well as other frequencies.

3. Establishment of the relationship between sensor response at a given wavelength with the moisture and temperature profiles.

4. Measurements at both aircraft and satellite altitudes with scatterometers. Most of the scatterometer experiments were done on the ground with a truck-mounted system. Measurements at high altitudes are clearly necessary.

5. Measurements at both aircraft and satellite altitudes with imaging radar systems. So far, the measurements performed with an aircraft synthetic aperture radar system at both X and L bands have not given any positive and convincing results. Extensive work is needed in this area.

The terrain and modeling problems are associated with 1) the water content of soil, 2) the type of soil, 3) the vertical profile of water content, 4) the surface roughness and row direction (for furrowed fields), 5) the temperature at the surface and its variation with depth and 6) the state of vegetal cover.

Aside from modeling, technology needs concerning soil moisture measurements cannot be properly assessed until results of three experimental programs (discussed above) are available.

Capabilities of Platforms and Sensors -- Recent, Current and Planned

This summary provides background information for the preceding sections on technology automation needs. While it is intended to be reasonably complete, film camera systems (aircraft, Space Shuttle), some proposed platforms, and some proposed Space Shuttle experiments have been omitted. Admittedly, the selection process was somewhat subjective, however, the included spectrum of capabilities is sufficiently representative to support the conclusions reached.

A large number of sources were consulted in an effort to offer current information, and to cite all of these would be burdensome. Some references are given in the short synopses offered for each platform series. Overview documents that we found quite useful are the Advanced Sensors and Applications Study (Lockheed Electronics Company, 1976) and Satellites for Health Applications and the Life Sciences (Giddings, 1974).

Aircraft - At the Johnson Space Flight Center (JSC), NASA maintains a large inventory of sensors that can be flown on aircraft based at Ellington Field near Houston, Texas. The inventory, aircraft and capabilities are summarized in the NASA Earth Resources Program - JSC Earth Resources Aircraft Plan, revised November, 1975. Selected sensors have been included in the summary table at the end of this section. Not included are references to sensors carried only on the Bell 206B helicopter; most camera systems have also been omitted, as well as a number of obsolete sensors.

Application Explorer Mission (AEM) - These missions concentrate on specific applications and are relatively low in cost. The first, AEM-A, is scheduled for launch in 1978, carrying the Heat Capacity Mapping Radiometer (HCMR). This mission has been included in the summary because the half kilometer resolution is of considerable interest for future earth observation programs. Later, AEM-B (1979) is slated to carry a stratospheric aerosol and gas probe, and later yet a terrestrial magnetism mission (probably) will be flown on AEM-C.

Since these are low-cost missions, the satellites will not carry tape recorders. Coverage will be limited to line of sight of NASA receiving stations. This implies that data will be available for the United States, along with Alaska and Northern Mexico, but not Hawaii, most of Europe, and about half of Australia. Other stations may also collect these data, if they have appropriate facilities, but they must be considerably more sophisticated than ordinary APT receivers.

Applications Technology Satellite (ATS) - Satellites of the ATS series are designed to develop technology for a variety of applications and provide an orbital test bed for advanced concepts. Viable sensor systems developed through the use of ATS satellites have been later flown on operational platforms and likely will continue to do so, consequently, the ATS series was not included in the summary.

Defense Meteorological Satellite Program (DMSP) - The U.S. Air Force, through its Space and Missile Systems Organization (SAMSO), conducts an ongoing program of surveillance of weather by satellite. Its Defense Meteorological Satellite Program (DMSP) (formerly called DAPP) keeps two polar-orbiting satellites in operation to permit six-hourly coverage of any portion of the Earth. The next generation Block-5D satellites will contain only high-resolution sensors (0.3 n.mi) that will read out digitally, as do the TIROS-N and ITOS-H through ITOS-J. However, Block-5 satellites will have onboard processing facilities that will result in degradation of high-resolution images. This will allow storage of complete swaths and will produce low-resolution data at the receiving station. This capability is not mentioned in current descriptions of TIROS-N and ITOS-H through ITOS-J, suggesting that they may have tape recording capacities for an entire swath of high-resolution data.

The DMSP is furnishing data to civil users through NOAA, so the available information on their systems were included in the summary table. A considerable amount of information has been published on Block-5C and -5D satellites (Bull. Amer. Meteorol. Soc. 55:9-15, 1974 and Aviation Week and Space Technology; pp. 52-55, December 3, 1973; pp. 16-17, May 13, 1974; pp. 22-23, June 24, 1974; pp. 40-47, July 15, 1974).

Geostationary Orbiting Earth Satellites (GOES) - The early satellites in the GOES series were called Synchronous Meteorological Satellites (SMS). These earth-synchronous satellites orbit at 34,781 km and provide coverage of the earth disc every 20 minutes, thus have a greatly different capability than the low earth orbital satellites.

The quantity of data transmitted by GOES is large compared to previous series, and required doubling the staff at the Wallops Island, Virginia receiving station to handle the increased data load.

ITOS, TIROS - This series of operational weather satellites has had various acronyms during its sustained development program.

The series began with the Television and Infrared Observation Satellite (TIROS) with 10 satellites launched from 1960 to 1965. This program was followed by the Tiros Operational Satellite (TIROS) with 10 satellites launched from 1960 to 1965. This program was followed by the Tiros Operational Satellites (TOS), named ESSA 2 through 9 (Environmental Science Services·Administration), operational as late as 1974 or later.

The second generation operational satellites are called the Improved TIROS Operational Satellite (ITOS) series. They are in sun-synchronous polar orbit at 1464 km.

A later generation yet launched in 1978 to a polar orbit of 833 km is dubbed TIROS-N (see Summary Table).

LANDSAT - The early LANDSAT satellites were originally called Earth Resources Technology Satellites (ERTS), but all eventually came to be known as LANDSAT vehicles. This series continues to provide the primary source of Earth resources data for a wide variety of users, available from the central source (EROS Data Center, Sioux Falls, South Dakota, 57198).

The U. S. Government publishes maps showing the best available coverage over any land surface in the world. Unfortunately, the criterion used is cloud cover; these "best" images may be ideal for geography, but others may be preferable for environmental uses. In any case, upon written inquiry to EROS a list of entire coverage of an area in form of a computer printout may be obtained.

With the advent of the thematic mapper on LANDSAT-D, a quantum jump in the utility of LANDSAT data is expected, especially in vegetation classification and geology.

NIMBUS - The Nimbus series serve as testbeds for research and development of systems and sensors primarily for meteorological programs. Resulting technological advances have been incorporated in operational satellites such as ITOS.

A fairly extensive listing of the sensor systems flown on various Nimbus missions has been included in the summary table, as data gathered by Nimbus sensors is available for various projects. The sensors have been described in users guide, and non-meteorological uses in other publications, all avilable from Goddard Space Flight Center.

SEASAT - The SEASAT-A satellite launched in June 1978 represented a great step forward in remote sensing, as it was the first satellite to carry a Synthetic Aperture Radar (SAR) for earth resources programs. The primary mission was to study and monitor ocean characteristics including surface wind, water temperature and sea state. The initial coverage was global at 36 hour intervals, expected to reduce as additional satellites are added to form a network. The system additionally provided a limited amount of data to land applications investigators, hopefully providing an early insight concerning the utility of microwave sensors for these applications, and valuable experience for those planning to use the Shuttle Imaging Radar. An excellent review of SEASAT is available (Committee on Science and Technology, 1977). Unfortunately, this system failed in October 1978.

Skylab - The three Skylab missions provided a mass of imagery useful to Earth resources programs, including photographs from hand-held cameras, multispectral photographs, high resolution metric photographs, 13-channel multispectral scanner images or tapes, and other experimental scanner data.

The multispectral bands on the S192 experiment were chosen in much the same way as the Bendix MSDS Aircraft Multispectral Scanner (MSS); i.e., they covered the visible, near infrared and thermal infrared windows.

The multispectral data taken by the S192 experiment have hardly been used. Still, for areas where they exist, they are perhaps the most complete images ever taken from space. Thirteen separate images exist for each scene, in a continuous swath. The research potential for these images has scarcely been touched -- most principal investigators have only examined photographic reproductions of a few channels. The S192 multispectral scanner tapes are nowhere satisfactorily catalogued.

Space Shuttle - The Space Shuttle will provide for the first time the exceedingly valuable capability of sequential development of earth observation systems through signature research, sensor development and applications development (Schappell and Tietz, 1978). The relatively low cost and possibility of scheduling flights to meet system development needs should catalyze a remarkable expansion of user oriented earth resources programs.

Many of the experiments to be carried on the Shuttle flights are still in the definition phase or speculative. Those included in the table seem firm for early flights. An exhaustive review of the role of Space Shuttle in earth resources programs is given in an eight-volume series entitled "Definition of the Total Earth Resources System for the Space Shuttle Era," published for NASA by the General Electric Company Space Division (see

Synchronous Earth Observatory Satellite (SEOS) - While some earth resources data has been available from geosynchronous meteorological satellites (GOES), the SEOS (scheduled 1981) will be the first geosynchronous earth resources satellite. The sensors will be pointable, affording a target of opportunity ability to gather data of a transient nature, such as weather occurrences, disasters, and transient growth phenomena valuable to agricultural programs. The ability to "retake" scenes missed by LANDSAT due to cloud cover is of enormous value.

An interesting study is available (Lowe, et al., 1973) defining earth resources applications that require the unique temporal coverage to be provided by SEOS.

Synthetic Aperture Radar Satellite (SARSAT) - SARSAT is being developed by the European Space Research Organization (ESRO) for a multiplicity of studies including snow and ice monitoring, mapping, geology, and sea surface characteristics including oil slicks. It will be flown either on a THOR Delta 2910 or Space Shuttle. For more information, see the Proceedings of the Ninth International Symposium on Remote Sensing of Environment, April 15-19, 1974, pp. 1517-1540.

Onboard processing and preprocessing techniques have been considered due to the maximum anticipated data rate of 100 megabits per second.

National Oceanic Satellite System (NOSS) - Originally called SEASAT-B, NOSS is slated to carry a sensor suite derived from the SEASAT-A and NIMBUS programs. Its objective is to extend the SEASAT-A "proof of concept" and hopefully demonstrate a set of capabilities that would justify an operational SEASAT system. The primary applications of NOSS involve the measurement, estimation or mapping of ocean circulation, wind stress, sea surface temperature, waves, ice, ocean color and biological resources.

The synthetic aperture radar has apparently been deleted from the program, thus eliminating one of the major data sources planned for wave and ice studies.

Upper Atmosphere Research Satellite (UARS) - The UARS has objectives primarily concerned with the investigation of the chemistry and dynamics of the stratosphere and mesosphere on a global basis, and ozone depletion studies. A few of the sensors such as the Nadir Emission Radiometer, intended to measure cloud coverage and cloud top temperature, have possible application to earth observation missions.

The mission concept calls for the use of the Space Transportation System (STS) and Tracking and Data Relay Satellite System

(TDRSS) to launch, support and refurbish the UARS. After one and one-half years in orbit, the UARS is scheduled to move to a 297 km orbit for retrieval or refurbishment by the STS

STEREOSAT - The purpose of STEREOSAT is to acquire high resolution stereoscopic images of the earth's land masses to latitudes $\pm$ 80°. Three telescopes are planned with 600 nm glass lenses using four DET assemblies; these telescopes will point 30° forward, at the nadir, and 30° aft. The design lifetime is one and one-half years.

Table IV-7 Summary of Earth Observation Remote Sensing Systems

System/Agency	Sensor Subsystems	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
Aircraft Multispectral Scanners (NASA–JSC)	Michigan 12-Band M-7	Objective Plane Scanner	ERIM	C-47 (ERIM Aircraft)	10,000 ft	Swath = 2.0 h*	NA	2.0–3.3	$2.0–3.3 \times 10^{-3}$ h*	Any 12 at One Time: 0.32–0.38 0.40–0.44 0.44–0.46 0.46–0.48 0.48–0.50 0.50–0.52 0.52–0.55 0.55–0.58 0.58–0.62 0.62–0.66 0.66–0.72 0.72–0.82 0.82–0.96 1.0–1.4 1.5–1.8 2.0–2.6 8.0–13.5	Not Operational for NASA	*h = altitude of aircraft. No longer listed as operational in JSC Earth Resources Aircraft Plan.
	Bendix 24-Band (MSDS)	Objective Plane Scanner	Bendix	NC-130B	30,000 ft	Swath = 1.68 h	NA	2.0	2.0×10^{-3} h	0.34–0.40 0.40–0.44 0.46–0.50 0.53–0.57 0.57–0.63 0.64–0.68 0.71–0.75 0.77–0.81 0.82–0.87 0.97–1.06 1.06–1.095 1.13–1.17 1.18–1.3 1.52–1.73 2.1–2.4 3.54–4.0 6.0–7.0 8.3–8.8 8.8–9.3 9.3–9.8 10.1–11.0 11.0–12.0 12.0–13.0	Operational	––
	Modular Multispectral Scanner (M^2S)	45° Mirror Objective Plane Scanner	Bendix	NC-130B	30,000 ft	Swath = 2.38 h	NA	2.5	2.5×10^{-3} h	0.38–0.44 0.44–0.49 0.49–0.54 0.54–0.58 0.58–0.62 0.62–0.66 0.66–0.70 0.70–0.74 0.76–0.86 0.97–1.05 8.0–12.0	Operational	––
	RS-14	Four-Sided Mirror Objective Scan	Texas Instruments	––	––	Swath = 1.68 h	NA	1.0 or 3.0	1.0 or 3.0×10^{-3} h	Any Two Modes at One Time: 0.3–0.55 0.7–0.9 1.0–1.5 1.5–1.8 2.0–2.5 3.0–3.5 8.0–14.0	Stored; Probably To Be Surplused	This instrument is listed in the JSC Earth Resources Aircraft Plan as being replaced by the M^2S.

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Table IV-7 (cont)

System/Agency	Sensor Subsystems	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, µm (Unless Otherwise Noted)	Status	Comments
Aircraft Multispectral Scanners (NASA–JSC) (cont)	RS-18 Multispectral Scanner	45° Single Mirror	Texas Instruments	WB-57F	60,000 ft	Swath = 1.68 h	NA	1.0	1.0×10^{-3} h	10.0–12.0 Ch 1 0.5–0.6 Ch 2 0.6–0.7 Ch 3 0.7–0.8 Ch 4 0.8–1.1 Ch 5 HgCdTe–Thermal Detector Si (Blue Enhanced) Visible–NIR Detectors)	Operational (Aircraft Modification Completed 1977)	—
	RS-18B Thermal Scanner	45° Single-Mirror Cassegrain Reflector	Texas Instruments	WB-57F	60,000 ft	2.38 h	NA	1.0	1.0×10^{-3} h	10.0–12.0 HgCdTe	Operational	—
	NS001	45° Single-Mirror Newtonian Reflector	JSC	NC-130B	30,000 ft	2.38 h	NA	2.5	2.5×10^{-3} h	0.45–0.52 0.52–0.60 0.63–0.69 0.76–0.90 1.00–1.30 1.55–1.75 2.08–2.35 10.4–12.5	Nov 1976 One Channel Operational; Others under Construction	Thematic mapper simulator.
	Active–Passive Multispectral Scanner	Boresighted Laser and Passive Scanner	ERIM	C-47 (ERIM Aircraft)	10,000 ft	2.0 h	NA	2.0	2.0×10^{-3} h	Active (Laser) 1.06 (Image) 0.4–0.64 (Profiler, Tunable) Passive (Band Center) 0.45 0.48 0.50 0.52 0.54 0.57 0.61 0.65 0.75 1.7 or 2.3 or 11.0	Operational for ERIM	One demonstration flight under Advanced Application Flight Experiment, early 1977.
	Solid-State Array Spectroradiometer	Pushbroom CCD Detectors	JSC	NC-130B RB-57	30,000 ft 60,000 ft	0.263 h	NA	0.76	0.76×10^{-3} h	20 Bands 0.4 to 0.8 Equally Distributed	Operational Mar 1977	—
	Airborne Multispectral Photographic System (AMPS)	Six-Camera Assembly	ITEK	NC-130B WB-57F	30,000 ft 60,000 ft	—	NA	21°	—	0.4–0.9	Operational	See S190A under Skylab and Space Shuttle.

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System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, µm (Unless Otherwise Noted)	Status	Comments
Aircraft Radar Systems (NASA—JSC)	Passive Microwave Imaging System (PMIS)	Passive, Dual Polarized, 50° Constant Angle of Incidence Microwave Imager	Aerojet	NC-130B	30,000 ft	Beamwidth 1.8–2.7°; 44 Beam Positions ± 35° Scan Angle	NA	—	—	10.69 GHz ± 5.0 MHz	Operational	—
	Multifrequency Microwave Radiometer (MFMR)	Four-Frequency Passive Microwave Radiometer Plus C-Band	Aerojet	NC-130B	30,000 ft	Beamwidths Ch 1—16° Ch 2,3,4,5—5°	NA	—	Ch 1—560 ft at 2000 ft Alt Ch 2,3,4,5—175 ft at 2000 ft Alt	Ch 1—1.42 GHz Ch 2—18.0 GHz Ch 3—22.05 GHz CH 4—37.0 GHz CH 5—5.0 GHz	Operational	—
	16.5-GHz Side-Looking Radar	Radar Mapping System	Philco—Ford	NC-130B	30,000 ft	Horizontal Beamwidth—0.8° Vertical Beam-width—43.0°	NA	—	Slant Range—40 ft Ground Range—45 to 87 ft Azimuth—65 to 185 ft	16.5 GHz ± 5.0 MHz	Marginal Data Quality; Placed in Storage Feb 1973	—
	13.3-GHz Single Polarized Scatterometer	Active Backscatter Radar	Ryan (Modified at JSC)	NC-130B	30,000 ft	Beamwidth Along Track ± 50° from Nadir; Beamwidth Across Track—15°	NA	—	—	13.3 GHz ± 50.0 MHz	Operational	Reactivated for joint soil moisture experiments.
	1.6-GHz Scatterometer	Active Dual-Polarized Fan Beam Scatterometer	Ryan	NC-130B	30,000 ft	Beamwidth Along Track—60° from Nadir; Beamwidth Across Track—9°	NA	—	—	1.6 GHz ± 1.6 MHz	Operational	Reactivated for joint soil moisture experiments.
	400-MHz	Active Dual-Polarized Interrupted CW Scatterometer	Emerson Electric	NC-130B	30,000 ft	Beamwidth Along Track 36° Pointed 42° Aft of Nadir; Beamwidth Across Track—16°	NA	—	—	400.85 MHz ± 1.0 MHz	Operational	Reactivated for joint soil moisture experiment.
	APQ-102 Side-Looking Radar	Active Single-Polarized SAR	Goodyear	WB-57F	60,000 ft	36 km (Swath)	NA	—	15x15 m	9600 MHz ± 5.0 MHz	Scheduled Operational in 1978	Installed in aircraft in early 1978.
	4.5-GHz Scatterometer	Active, Dual-Polarized from Beam Scatterometer	Ryan & New Mexico State University (PSL)	NC-130B	30,000 ft	Beamwidth —60° from Nadir Along Track; Beamwidth Across Track—12°	NA	—	—	4.75 GHz ± 5.0 MHz	Operational in 1978	—
Improved TIROS Operational Satellite, ITOS (NOAA)	Scanning Radiometer	Line Scanning Visible IR Radiometer	Santa Barbara Research Center	NOAA-2, Oct 1972 NOAA-3, Nov 1973 NOAA-4, Nov 1974 NOAA-5, Jul 1976	1464 ± 45 km Polar, Sun Synchro-nous	Horizon-to-Horizon	12 hr	5.3	7.5 km at Nadir	0.5–0.7 10.5–12.5	Operational	Si photovoltaic (visible); thermistor bolometer (IR).
	Very High Resolution Radiometer (VHRR)	Line Scanning Radiometer	RCA			± 75.0°		0.6	0.9 km at Nadir	0.6–0.7 10.5–12.5		Si photodiode (visible); HgCdTe (IR).
	Vertical Temperature Profile Radiometer (VTPR)	Visible—IR Spectroradiometer	Barnes Engineering			± 31.45°		38.0	55.6 km	6 Bands in CO_2 Channel ≃ 15; 1 Channel in H_2O Band = 18.7 1 Channel in "Window" = 8–12		

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Table IV-7 (cont)

System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
LANDSAT 1 & 2 (NASA)	Multispectral Scanner (MSS)	Objective Mirror Scanner	Hughes	LANDSAT 1, Jul 1972 LANDSAT 2, Jan 1975	907 km Near-Polar, Sun Synchronous	Swath = 185 km	18 days	0.086	79 m	0.5–0.6 0.6–0.7 0.7–0.8 0.8–1.1	Operational	—
	Return Beam Vidicon (RBV)	3 RBV Camera	RCA			185x185 km		—	80 m	0.475–0.575 0.580–0.680 0.698–0.830		
LANDSAT 3 (NASA)	Multispectral Scanner (MSS)	Objective Mirror Scanner	Hughes	1977	911.8-km Circular, Near-Polar, Synchronous	Swath = 185 km		0.086	Visible–NIR = 79 m Thermal IR = 240 m	0.5–0.6 0.6–0.7 0.7–0.8 0.8–1.1 10.4–12.6	Operational	1–3 Bands PMT 4th Si Photodiode 5th HgCdTe
	Return Beam Vidicon (RBV)	2 RBV Camera (Side-by-Side Coverage)	RCA			93x93 km		—	40 m	0.50–0.75 (Both RBVs)		4500 TV lines, 90 lines/mm.
LANDSAT D (NASA)	Multispectral Scanner (MSS)	Same as on LANDSAT 3		1980							Scheduled for 1980	
	Thematic Mapper (TM)	6-Band MSS	TBD		704.2-km Circular, Near-Polar, Sun Synchronous	Swath = 185 km	9 days (Two Satellites)	Visible–Near IR = 0.043 Thermal IR = 0.250	30 m 120 m	0.45–0.52 0.52–0.60 0.63–0.69 0.76–0.90 1.55–1.75 2.08–2.35 10.4–12.5		Alternative two-band set for LANDSAT D(1) and D(2) proposed by Chism (see Chism 1976 and text).
NIMBUS 5 (NASA–GSFC)	Electrically Scanning Microwave Radiometer (ESMR)	Imaging Microwave System	—	December 1972	1222-km Circular, Sun Synchronous	Swath = 3200 km	Every 12 hr, Local Noon and Midnight (Applicable to Imagers Only)	—	25x25 km at Nadir	19.225–19.475 GHz	Operational	—
	Temperature–Humidity Infrared Radiometer (THIR)	Thermal Mapper	—			Global		—	22x8 km at Nadir	6.5–7.0 10.5–12.5		—
	Surface Composition Mapping Radiometer (SCMR)	Thermal and Near-Infrared Mapper	—			Swath = 800 km		—	660x660 m at Nadir	0.8–1.1 8.3–9.3 10.2–11.2		—
	Infrared Temperature Profile Radiometer (ITPR)	Atmospheric Sounder	—			Swath = 1050 km		—	31.8 km at Nadir	3.8 11.0 13–15.0 (4 Bands) 19.8		—
	Selective Chopper Radiometer (SCR)	Atmospheric Sounder	—			Nadir Only		—	42 km (15 μm) 29 km (All Others)	2–100 (16 Bands)		
	Nimbus E Microwave Spectrometer (NEMS)	Atmospheric Sounder	—			Continuously Along Nadir		—	204 km	27.23 GHz 31.40 GHz 53.65 GHz 54.90 GHz 58.80 GHz		—

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Table IV-7 (cont)

System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
NIMBUS 6 (NASA–GSFC)	Temperature–Humidity Infrared Radiometer (THIR)	Thermal Imager	—	June 1975	1100-km Polar, Sun Synchro-nous	Swath = 3000 km	Every 12 hr, Local Noon/ Midnight (Applicable to Imagers Only)	—	22.5 km (6.5–7.1 μm) 8.2 km (10.3–12.5 μm)	6.5–7.1 10.3–12.5	Operational	—
	Earth Radiation Budget (ERB)	Spectrometer	—			500x500 km		—	—	0.2–50.0 (22 Channels)		—
	Scanning Microwave Spectrometer (SCAMS)	Atmospheric Sounder	—			2060 km		—	145 km at Nadir	22.235 GHz 31.650 GHz 52.850 GHz 53.850 GHz 55.450 GHz		
	High-Resolution Infrared Radiation Sounder (HIRS)	Atmospheric Sounder	—			Swath = 1650 km		—	25 km	0.69 3.71 4.24–4.57 (5 Bands) 6.7 8.2 11.0 13.4–15 (7 Bands)		
	Electrically Scanning Microwave Radiometer (ESMR)	Atmospheric Sounder (Conical Scan)	—			1200 km		—	18–22 km Crosstrack 35–54 km Downtrack	37.0 GHz		
	Limb Radiance Inversion Radiometer (LRIR)	IR MS Scanning Radiometer	—			NA (Scans Earth Limb)		—	2x11.7 km 2x20.7 km 2x20.7 km 2.5x25.4 km	14.9–15.5 (Narrow CO_2 Band) 14.4–16.9 (Broad CO_2 Band) 8.6–10.2 (O_3) 23.0–27.0 (H_2O)		
	Pressure Modulator Radiometer (PMR)	Atmospheric Sounder	—			Steerable to ± 15° Forward & Aft		—	77 km Along Track 383 km Across Track	CO_2 Pressure Comparison in Several Bands Around 15		

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Table IV-7 (cont)

System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, µm (Unless Otherwise Noted)	Status	Comments
NIMBUS G (NASA–GSFC)	Stratospheric and Mesospheric Sounder (SAMS)	Atmospheric Sounder	—	1978	955-km Polar, Sun Synchronous	—	Every 12 hr (Applicable to Imagers Only)	28x2.8	—	15 (CO_2) 4.1–5.4 (CO_2, CO, NO) 2.7, 2.5–100 (H_2O) 7.6–7.8 (N_2O, CH_4)	Operational in 1978	Detectors: 4 TGS 1 InSb 1 PbS
	Temperature Humidity Infrared Radiometer (THIR)	Two-Channel Scanning Radiometer	—			—		21 (6.5–7.0 µm) 7 (12.5 µm)	—	6.5–7.0 12.5		—
	Coastal Zone Color Scanner (CZCS)	Scanning Radiometer	—			±0.7-rad Scan ±0.35-rad View Angle		0.865 sq		0.433–0.453 0.510–0.530 0.540–0.560 0.660–0.680 0.700–0.800 10.5–12.5		Detectors: Thermal IR = HgCdTe Others = SiPD
	Solar Backscatter Ultraviolet Spectrometer and Total Ozone Mapping (SBUV/TOM)	—	—			Unknown		0.052 rad sq		160–400		Continuous Scan (PMT)
	Limb Infrared Monitoring of Stratosphere (LIMS)	Infrared Radiometer Modified Nimbus 6 LRIR	—			—		0.5 sq (1x8 — H_2O, NO_2)	—	6.08–6.39 (NO_2) 6.41–7.25 (N_2O) 8.64–10.64 (O_3) 10.87–11.76 (HNO_3) 13.16–17.24 [CO_2 (wb)] 14.71–15.75		Detectors: HgCdTe
	Earth Radiation Budget	22-Channel Radiometer	—			—		4.4x89.4 (Scan), 2.32 rad Cone (Earth), 0.46 rad (Solar)	—	Solar Channels 0.2–4.0 (2 Ch) 0.4–0.5 0.35–0.45 0.2–0.5 0.53–3.0 0.7–3.0 0.3–0.4 0.28–0.35 0.25–0.30 Earth Channels 0.2–0.5 (2 Ch) 0.2–4.0 0.7–3.0 Scanning Channels 0.2–5.0 (4 Ch) 4.5–5.0 (4 Ch)		—
	Stratospheric Aerosol Measurement II	—	—			Altitude Range 10 km to Space		0.145	—	0.06 Band at 1.0		—
	Scanning Multichannel Microwave Radiometer (SMMR)	Five-Wavelength Dual-Polarized	—			—		—	—	37 GHz (0.8 cm) 21 GHz (1.4 cm) 18 GHz (1.7 cm) 10.7 GHz (2.8 cm) 6.6 GHz (4.0 cm)		

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Table IV-7 (cont)

System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
Skylab Earth Resources Experiment Package, EREP (NASA)	S190-A	Six Multiband Cameras	ITEK	Operated May 1973–Feb 1974	435 km	163x163 km	NA	FOV = 21.23° sq	78–223 ft† (24–68 m)	0.7–0.8 B&W IR 0.8–0.9 B&W IR 0.5–0.88 Color IR 0.4–0.7 Color 0.6–0.7 B&W Red 0.5–0.6 B&W Green	Uncertain – Not Operational from 1974 to 1978	†78 ft for SO-356 color film, 223 ft for EK 2424 B&W IR film. Six bands 0.4–0.9; bands selected with filters. Grid resau, 0–30 frames/minute.
	S190-B	Earth Terrain Camera	ACTRON			109x109 km	—	FOV = 14.24° sq	35–99 ft§ (11–30 m)	0.40–0.90 Film Selected from Following: Aerial Color 0.4–0.7 Aerial B&W 0.5–0.7 Color IR 0.5–0.88		§35 ft for SO-242 aerial color film, 99 ft for EK 3443 color IR film. Varying overlap between frames, 0–25 frames/minute.
	S191	Pointable Filter Wheel Spectrometer	Block Engineering/ Martin Marietta			Nonimaging		1.0	0.435-k m Diameter	Continuous 0.39–2.5 (Si and PbS) 5.82–15.99 (HgCdTe)		—
	S192	13-Band Conical Scanner	Honeywell			Swath = 68.5 km		0.182° sq	79.2 m	0.41–0.46 0.46–0.51 0.52–0.56 0.56–0.61 0.62–0.67 0.68–0.76 0.78–0.88 0.98–1.08 1.09–1.19 1.20–1.30 1.55–1.75 2.10–2.35 10.2–12.5		—
	S193	Ku-Band 3 Modes: 1) Radiometer 2) Scatterometer 3) Altimeter	General Electric			Total Scan of 22.7°		1.6° Half-Power Beamwidth	11.0 km-Diameter Circle at Nadir	13.9 GHz		—
	S194	Nonimaging L-Band Radiometer	AIL			—		—	Half-Power Points: 124-km Diameter	1.414 GHz		—

Table IV-7 (cont)

System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
TIROS N, Second Generation of ITOS/NOAA (NASA/NOAA)	Basic Sounding Unit (BSU)	Rotating Mirror, Cross-Track Scan	Ball Brothers Research Corporation	1978	833 ± 80-km Polar	± 1127 km (± 49.5°)	NA	—	21.8-km Diameter at Nadir, 75.2x37.3 km at Scan End	3.70 4.26 9.71 11.12 13.33 13.61 13.99 14.29 14.49 14.75 14.95 18.80 23.15 29.41	Launched 1978	No. steps/angle per step = 56/1.8°
	Stratospheric Sounding Unit (SSU)	Selective Absorption Pressure Cells	Provided by United Kingdom			± 737 km (± 40°)	NA	—	147-km Diameter at Nadir, 244x186 km at Scan End	Three Channels Each at 668 cm^{-1} Central Wave Number Cell Pressures: 100 mb 35 mb 10 mb		No. steps/angle per step = 8/10°.
	Microwave Sounding Unit (MSU)	Four-Channel Dicke Radiometer	JPL			± 47.35°	NA	—	7.5°	50.3 GHz 53.74 GHz 54.96 GHz 57.95 GHz		No. steps/angle per step = 11/9, 47°
	Advanced Very High Resolution Radiometer (AVHRR)	Four-Channel Visible—IR Radiometer	ITT			—	12 hr	—	1.1 and 4.0 km	0.55—0.9 0.725—1.0¶ 3.55—3.93 10.5—11.5 (11.5—12.5)¶		¶Not on first, but considered on later TIROS N/NOAA satellites as AVHRR—II.
	Space Environment Monitor (SEM)	Four Detector Arrays	NOAA/ERL, Boulder, CO			—	NA	—	—	Detector Arrays: 1) High-Energy Proton-Alpha Telescope (HEPAT) Protons = 100, 400, 600 MeV; Alpha = 2400, 4000 MeV 2) Low-Energy Proton-Alpha Telescope (LEPAT) Protons = 150 KeV—40 MeV; Alpha = 0.6—100 3) Proton Omnidirectional Detector (POD) Protons = 0.75, 10, 30, 60 MeV; Alpha = 75, 40, 120, 140 MeV 4) Total Energy Detector (TED)		

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System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
SEASAT A (NASA–JPL)	Synthetic Aperture Radar (SAR)	Imaging System, Single Polarization	Hughes/JPL	Jun 1978	790-km Circular, Not Sun Synchronous, Near-Polar Orbit	100 km on One Side from 17–23°	Within Receiving Station Range, Real-Time Only	—	25 m	1.275 GHz	Scheduled Operational in 1978	Polarization: H.
	Radar Scatterometer	Fan Beam	GE			750 km Each Side, ±25–65° Around Nadir ±70 km	NA	—	50 km ±5%	14.6 GHz		Polarization; sequenced HH, VV.
	Scanning Multifrequency Microwave Radiometer (SMMR)	Bidirectional Scan	JPL			638 ± 70 km	36 hr	—	See Comments	6.6 GHz 10.69 GHz 18.0 GHz 21.0 GHz 37.0 GHz		Footprint sizes in order due to integration: 121x79 km 74x49 km 44x29 km 38x25 km 21x14 km. Polarization: dual liner.
	Radar Altimeter	Precision Pulse	APL, Johns Hopkins			12-km Circle**	NA	—	1.6 km	13.5 GHz		**Due to integration.
	Visible-Infrared Radiometer (VIRR)	Scanning Two-Band	Santa Barbara Research Center			2127 km	36 hr	Visible, 2.8; IR 5.3	Visible, 2.2 km; IR 4.2 km	0.47–0.94 10.5–12.5		—
Applications Explorer Mission, AEM–A (NASA–GSFC)	Heat Capacity Mapping Radiometer (HCMR)	Visible–Infrared Imager	ITT	Mid–1978	600-km Polar-Orbiting, Sun Synchronous	693 km	Every 12 hr 2:00 am, 2:00 pm Local Time	0.83	0.518 km	0.55–1.1 10.5–12.5	Scheduled Operational in 1978	—
Synchronous Meteorological Satellite/ Geostationary Operational Environmental Satellite (SMS, NASA–GSFC) (GOES, NOAA)	Visible-Infrared Spin Scan Radiometer (VISSR)	Visible–Infrared Imager	Hughes	SMS-1, May 1974 SMS-2, Feb 1975 GOES-1, Oct 1975 GOES-B, May 1977 GOES-C, Jan 1978	34,781-km Earth Synchronous	Earth Disc	Earth Disc Every 20 min	0.026 (Visible) 0.20 (IR)	0.9 km (Visible); 7 km (IR)	0.55 to 0.75 10.5–12.5 (Advanced Versions Also Include Bands Centered at 3.94, 4.44, 4.52, and 13.3)	Operational	—
Defense Meteorological Satellite Program, DMSP 1 & 2; Block B/C (Air Force)	Scanning Radiometer††	Visible–Infrared Imager	—	Unknown – Now in Orbit	450-n-mi, Polar, Sun Synchronous	Swath = 1600 n mi	DMSP-1 Sunset, Sunrise; DMSP 2 Midnight, Noon	—	Visible VHR–0.34 n mi HR–2.0 n mi; IR WHR–0.36 n mi IR–2.4 n mi	0.4–1.1, 8–13	Operational	††Includes very high resolution (VHR) and high resolution (HR) for visible; very high resolution (WHR) and high resolution (IR) for Infrared. Detectors: Visible–VHR–back-biased Si diode; HR–photoconductive mode–Si diode; Infrared–WHR–HgCdTe; IR–thermistor bolometer.
	Special Meteorological Sensor	Atmospheric Sounder	—			—	—	Unknown	Unknown	Five CO_2 Bands Around 15; Atmospheric Window at 12; One H_2O Band at 20		—

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System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
Defense Meteorological Satellite Program, DMSP Block D (Air Force)	Operational Line Scan System	Imager	Westinghouse	To Be Launched	450-n-mi Polar, Sun Synchronous	1655 n mi	12 hr	—	0.3 n mi 1.5 n mi	0.41–1.1 8–13	To Be Launched	Detectors: PMT Si photodiode HgCdTe.
	Special Meteorological Sensor H (SSH)	Atmospheric Sounder	Barnes Engineering			±551.1 n mi 25 Steps Across Track	NA	2.7°	—	9.8 – O_3 12.0 – Window 13.4 CO_2 13.8 14.1 14.4 14.8 15.0 18.7 H_2O 22.7 23.9 24.5 25.2 26.7 28.2 28.3		—
Space Shuttle Earth Resources Sensors, Definition Stage (NASA)	Shuttle Imaging Radar (SIR)	Synthetic Aperture Radar	JPL/ Hughes§§	Scheduled 1982	~225 km to 450 km, Circular	Swath = 100 km Swath = 70 km	Unknown	1–17	25x25 m 12.5x12.5 m	1.4 GHz 9.8 GHz	— —	§§Under study.
	Mark II Interferometer	Nonimaging	JPL	Unknown		—	NA	Fixed-Nadir FOV	—	1–9	—	—
	Spaceborne Meteorological Radar	Synthetic Aperture Radar; Afocal	Hughes§§	Unknown		—	NA	—	—	~10 GHz	—	§§Under study.
	Shuttle Imaging Microwave System (SIMS)	Passive Scanning, Imaging Radar	JPL	Unknown		FOV = 60° Across Track 17° Along-Track	Unknown	17° 7.3° 3.8° 1.6° 1.0° 0.5° 0.47° 0.3° 0.2° 0.11° 0.09°	—	0.61 GHz 1.4 2.7 6.6 10.7 20 22.2 37 53 94 118.7 GHz	—	—
	Feature Identification and Location Experiment (FILE)¶¶	Two CCD Cameras	Martin Marietta	Scheduled 1979 (OFT-2 or OFT-3)	278 km	113x85 km	NA	23x18°	1.1x0.85 km	0.640–0.660 0.840–0.860	—	¶¶Two-band classification system and visible-IR camera.
		Hasselblad Camera				200x200 km	NA	40x40°	~120 m	0.5–0.9***		***Kodak Aerochrome infrared film 2443.
	Thematic Mapper	Six-Band MSS	(Similar or same as on LANDSAT D; characteristics given under LANDSAT D)									
	S190-A	Multispectral Camera	(Characteristics given under SKYLAB)									
	Monitoring Air Pollution from Satellites (MAPS)	Thermal Infrared Analyzer	—	Unknown	—	FOV = 7°	—	IFOV = 7°	—	—	—	—
	Correlation Interferometer for Measurement of Atmospheric Trace Species (CLIMATS)	Two-Channel Interferometer	—	Unknown		FOV = 7°	—	IFOV = 7°	—	2.0–2.4 4.0–9.0	—	—
	THIR	(See Nimbus G for characteristics)										

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System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
Space Shuttle Earth Resources Sensors, Definition Stage (NASA)	LIDAR†††	Laser Gas Analyzer	—	Unknown	—	Swath = 0.2–0.6	—	57 or Less	0.2–0.6 km	Infrared	—	†††Conceptual, Geological mapping experiment.
	Shuttle Multispectral Infrared Radiometer	10-Channel Near-Infrared Radiometer	—	Unknown		—	—	—	—	10 Bands 0.6–2.5	—	
	Ocean Color Experiment	10-Channel Scanning Radiometer	—	Unknown		—	—	—	—	10 Bands 0.44–0.75	—	Same as flown aboard U-2. Oceanic phytoplankton mapping.
Synthetic Aperture Radar Satellite, SARSAT (ESRO)	Active SAR	—	TBD	Unknown	Thor Delta 2910 or Space Shuttle	Swath = 50–100 km; Depression Angle of Sight to Horizon, 20 to 80°	NA	—	50x50 m	10 GHz (Also C-Band ?)	—	System in developmental stage. Polarization: two simultaneous modes, HH and HV.
SEOS A (NASA–GSFC)	Large Earth Survey Telescope (LEST)	Multispectral Scanner with Pushbroom Scan	TBD	Scheduled 1981	~35,700 km Geo-synchro-ous, Equatorial Orbit	Pointable to ±1 km FOV = 0.6x1.2°	Nearly Continuous on Earth Disc	Imaging: Visible 2.8; IR 22.4	Imaging: Visible 100 m, IR 800 m; IR Sounding: 18–30 km	0.45–1.1§§§ 2.3 3.5 4.3 6.5–7.0 10.5–12.5§§§ 14–15 23–24	—	A series of bands will fall in the spectral regions given in the table. §§§Imaging bands.
	Advanced Atmospheric Sounder and Imaging Radiometer (AASIR)¶¶¶	Imaging Radiometer	Santa Barbara Research Center				NA			0.55–1.1§§§ 3.7 4.24–4.57 (5 Channels) 6.71 7.25 11.11§§§ 12.66–15 (7 Channels)		¶¶¶Potential candidates; see Walter, L. S., EASCON 1974, pp 631–636. §§§Imaging bands.
	Microwave Sounder¶¶¶	—	—				NA		200 km at 50 GHz; 50 km at 220 GHz	5 Bands in the 50 to 220 GHz		
	Framing Camera	Television	—			1000x1000 km 200x200 km	Frame Every 16 s of Earth Disc		216 m for 1000 km; 45 m for 200 km	0.4–0.9		—

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Table IV-7 (cont)

System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
National Oceanic Satellite System; NOSS; Formerly SESAT B (Navy, JPL–NASA, NOAA)	Ocean Topography Altimeter	Precision Pulse	APL/ Johns Hopkins	1983–1984	700-km Circular, Polar, Non-synchronous	12-km Circle	NA	—	1.6 km (?)	Ku-Band; 13.9 GHz	In Planning – NASA Inhouse Studies Completed	
	Scanning Multi-frequency Multibeam Radiometer (SMRR)	Bidirectional Scan	JPL			638 ± 70 km	36 hr (?)	0.8	6.6 km	12 Channels, 6.6–94 GHz		—
	Synthetic Aperture Radar (SAR)	—	—			100 km	—	—	25 m	L-Band		Apparently deleted.
	SASS Radar Scatterometer	Fan Beam	GE			750 km Each Side ± 25–65°; Around Nadir ± 70 km	NA	—	50 km ± 5%	14.6 GHz		
	Advanced Very High Resolution Radiometer	Four-Channel (?) Visible–IR Radiometer	ITT				12 hr (?)	—	1.1x4.0 km	0.55–0.9 0.725–1.0+ 3.55–3.93 10.5–11.5 11.5–12.5+ (?)		
	Coastal Zone Color Scanner (CZCS)	Scanning Radiometer	—			± 0.7-rad Scan ± 0.35-rad View Angle	—	0.865 sq	—	0.433–0.453 0.510–0.530 0.540–0.560 0.660–0.680 0.700–0.800 10.5–12.5		
Upper Atmosphere Research Satellite, UARS (NASA–GSFC)	Solar Ultraviolet Spectrometer	—	—	1983	297 km	—	—	—	—	—	NASA Inhouse Studies Completed	—
	Doppler Visible	—	—			—	—	—	—	—		—
	Pressure-Modulated Radiometer	—	—			—	—	—	—	—		—
	Filter Radiometer	—	—			—	—	—	—	—		—
	Nadir Emission Radiometer	—	—			—	—	—	—	—		Cloud coverage. Cloud top temperature.
	Emission Radiometer	—	—			—	—	—	—	—		—
	Occultation Radiometer	—	—			—	—	—	—	—		—
	Ultraviolet Airglow Spectrometer	—	—			—	—	—	—	—		—
	1.27-micrometer Emission Spectrometer	—	—			—	—	—	—	—		—
	Far-Infrared Spectrometer	—	—			—	—	—	—	—		—
	Microwave Limb Sounder	—	—			—	—	—	—	—		—
	Electric Field Sensor	—	—			—	—	—	—	—		—

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System/Agency	Sensor Subsystem	Type	Manufacturer	Launch Date or Aircraft	Altitude or Orbit	Ground Coverage	Repeat Coverage	IFOV, mrad (Unless Otherwise Noted)	Resolution	Spectral Bands, μm (Unless Otherwise Noted)	Status	Comments
Upper Atmosphere Research Satellite, UARS (NASA–GSFC) (cont)	Particle Flux Detectors	–	–	1983	297 km	—	—	—	—	—	NASA Inhouse Studies Completed	—
	Magnetometer	—	—			—	—	—	—	—		—
	Laser Heterodyne Radiometer	—	—			—	—	—	—	—		—
	Cryogenic Limb Radiometer	—	—			—	—	—	—	—		—
STEREOSAT (NASA–JPL	Stereoscopic Telescopes	Three Telescopes with Vidicon Imagers	—	1983	705 km, 98.2° Inclination, Sun Synchronous 9:30 am; Opposite Side of Earth from LANDSAT D	30° Forward, Nadir, 30° Aft	18 days	—	—	0.6 Glass Lenses (Bandwidth ?)	FY 1980 New–Start Candidate	—

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V IMAGE DATA PROCESSING

An upsurge of interest in earth resources remote sensing programs over the last decade has prompted a series of developments in the image data processing systems technology. Indeed, lively image processing activities in remote sensing for earth resources can be readily attested by a wide spectrum of literature ranging from the Journal of Remote Sensing of Environment, the Journal of Pattern Recognition, Photogrammetry and Remote Sensing, Proceedings of the International Symposium on Remote Sensing of Environment, Proceedings of the IEEE, IEEE Transactions on Computers, and IEEE Transactions on Systems, Man, and Cybernetics, to mention just a few examples.

Of course, the applications of image data processing are not confined to the field of remote sensing. It encompasses such areas as optical character recognition, finger print recognition, EKG waveforms classification, chromosome identification, bubble and cloud chamber photographs analysis, and three-dimensional scene analysis. All these areas have been contributing greatly to the advancements in the image data processing technology. Owing to the scope of this report, however, only the image processing of spaceborne remotely sensed data will be reviewed.

Since it is anticipated that voluminous imaging data on the order of 10^{13} to 10^{15} bits per day (NASA Forecast of Space Technology) will be generated from various earth applications satellites by the year 2000, and since the rate of transmission of information over a spacecraft-to-Earth data link alarmingly shrinks as a function of inverse square of distance, (Darling and Joseph; 1968), it becomes clear that adaptive and highly automated image data processing technologies should be developed and incorporated into the spaceborne image processing systems. Specifically, it would be a giant step forward in the future space automation endeavors if the preprocessing functions such as data screening and editing for some generic categories, automatic field boundaries delineation, spectral band selection, adaptive statistical parameter estimation, together with some sort of automated pattern recognition processing, could be performed onboard the spacecraft prior to transmission to Earth. Potential cost-effective benefits resulting from such onboard processing would be undoubtedly tremendous.

Basically, an image data processing system can be briefly depicted in Figure V-1.

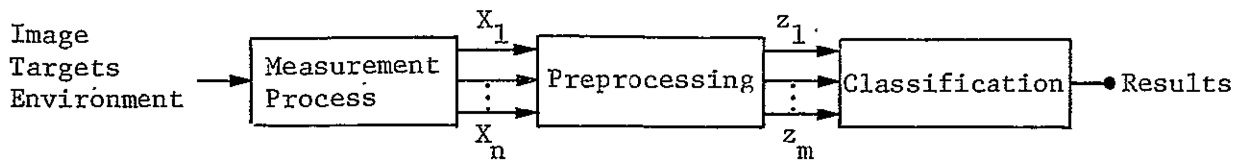

Figure V-1 Image Data Processing System

In the context of this report, an area of surface on the Earth or other planetary objects corresponds to the image target environment. The measurement process is undertaken onboard a spacecraft (e.g., LANDSAT or SEASAT) via specially designed image sensors such as a multispectral scanner. The dimensionality of the data vector is usually determined by the number of spectral bands in the multispectral scanner. The reflectance and emissivity of the imaging targets constitute an input to the measurement block. The preprocessing is a catch-all functional block that must be performed prior to classification processing. After all preliminaries are completed, the classification proceeds to produce the final results.

The purpose of this chapter is to summarize the state-of-the-art in the image data processing systems technology, with emphasis on the spaceborne image processing applications. Radar and the passive microwave image processing applications are omitted because of the limited scope of the report. In the following section, some remote sensing image-processing facilities are first briefly reviewed. A number of important preprocessing functions are then described. Finally, an overview of various classification schemes applicable to remote sensing is presented. The last section attempts to highlight relevant remarks on the subject matter with recommendations.

<u>Image Processing Systems</u>

The major recent spaceborne imaging systems are embedded in the Skylab EREP (Earth Resources Experimental Package) and the LANDSAT I, II, and III (formerly called the Earth Resources Technology Satellite or ERTS). The Skylab was launched in 1973 with one 14-band multispectral scanner and six 70-mm cartographic cameras onboard. The multispectral scanner covers the spectral ranges from 0.42 - 2.35 μm to the thermal infrared region 10.01 - 12.63 μm. It was designed to scan the object plane with conical lines. The cartographic camera systems are equipped with film-filter combinations for four spectral bands ranging from 0.4 μm to 0.9 μm.

LANDSAT I and II were launched in 1972 and 1975, respectively. Both carry the following two subsystems (ERTS Reference Manual): a) Return Beam Vidicon (RBV) camera subsystem contains three individual cameras that operate in different nominal spectral bands within the total band range from 0.475 to 0.83 μm. Camera 1 is sensitive to blue-green, Camera 2 yellow-red, and Camera 3 red-near infrared. When the cameras are shuttered, images are stored in photosensitive surfaces within each vidicon camera tubes which are then scanned to produce video outputs. It requires 3.5 seconds to read out each of three images. The cameras are reshuttered every 25 seconds; b) Multispectral scanner subsystem consists of a 4-band scanner, 6 detectors per band, operating in the spectral region from 0.5 to 1.1 μm. The object plane is scanned by these 24 detectors simultaneously through an oscillating-mirror cross-track mechanism. The instantaneous field of view of each detector subtends an earth-area square of 79 meters on a side from the nominal LANDSAT orbital altitude.

LANDSAT III was just launched last March, 1978. In addition to the imaging sensors onboard LANDSAT I and II, the latest LANDSAT III carries a fifth multispectral band operating in the thermal infrared region of 10-12 μm. By means of cubic convolution preprocessing at GSFC, the effective non-overlapping pixel size will be a 59 x 59 meter square ground area.

In general, the spectral resolution, which is defined as $\lambda/\Delta\lambda$ (λ being the nominal wavelength, and $\Delta\lambda$ the spectral coverage of a multispectral band), is higher for the Skylab multispectral scanner than that for the LANDSAT multispectral scanner.

To date, no onboard image processing capabilities have been developed in space missions, let alone the implementation of any spaceborne adaptive automatic pattern recognition processor. Some feasibility studies were conducted such as Darling and Joseph (1968) on the possibility of performing pattern recognition processing onboard the spacecraft. However, the evaluation results based on the classification of lunar topographic features and earth's cloud patterns were not conclusive. Perhaps the closest endeavor attempted so far along this line is the FILE experiment (Feature Identification and Location Experiment) which will be flown on Space Shuttle early 1980 (Schappell and Tietz, 1978).

The FILE experiment will be primarily composed of a) a sun sensor capable of sensing the sunrise in order to control the timing of the experiment, b) two CCD solid-state cameras having a bandwidth of 0.02 μm, with one being centered at 0.65 μm and the other 0.85 μm, and c) a film camera for concurrently taking photographs. The major thrust of this experiment would be to test the onboard image data classification capabilities utilizing the spectral ratio information.

Currently, most of image processing activities (Nagy, 1972; O'Handley and Green, 1972; Landgrebe, et al., Chien and Snyder, 1975), are performed at the ground facilities. Typical examples of these facilities are briefly summarized as follows:

a. Purdue University: Laboratory for Applications of Remote Sensing (LARS) consists of an IBM 360/67 system, 577 by 768 elements 16-level digital TV display, lightpen, continuous image scroll, selective Polaroid or negative hardcopy, FM tape conversion.

b. Environmental Research Institute of Michigan (ERIM): CDC 3600, CRT display, FM tape conversion, analog film record, drum scanner.

c. NASA/JSC: The Earth Resources Interactive Processing System (ERIPS) consists of an IBM 360/75 system, a CRT display of standard LACIE segments with 117 lines by 196 pixels per line, and optional Polaroid or hardcopy outputs. In addition, there are tow image processing and pattern analysis systems at JSC. One is the Purdue Terminal connected to the Purdue's LARS system, and the other is the GE Image-100 system which is capable of interactive processing of image data as well as pattern analysis and classification.

d. JPL: IBM 360/75 with film-scanner, CRT display, FM conversion, facsimile hardcopy. The resolution of its imaging devices is getting 729 pixels per line. This high resolution is needed for exploration of the surface of another planet. The Charge Coupled Device (CCD) technology is being extended for two-dimensional, monolithic imaging arrays in order to increase the imaging sensitivity while reducing its cost, weight, and power.

e. NOAA/National Environmental Satellite Center at Suitland, Maryland: three CDC 6600s, CDC 160A, two ERM 1630s and 6050s, CDC 924, three Muirhead recorders.

f. IBM Yorktown Heights Research Center: film scanner, CRT output, image dissector, digital color TV display, IBM 360/67, 360/91, and graphic tablets.

g. University of Kansas: The Kansas Digital Image Data System (KANDIDATS) and the Image Discrimination, Enhancement, and Combination System (IDECS) consist of three flying spot scanners for transparencies and a vidicon camera controlled by a PDP 15/20, linear processor and level selector, 24-channel digital disk storage, and monochrome and color displays with built-in crosshatch generator and film output.

Since none of the above facilities are involved in real-time image processing, the time required for processing the image data is relatively insignificant. However, there are image processing centers like the Artificial Intelligence Lab at Stanford University, Stanford Research Institute (SRI), and the Coordinated Science Lab (CSL) at the University of Illinois, Urbana, each involved in various aspects of computer vision research which do require very high speed image data input capabilities. The strong interactions among speed, dynamic range, and resolution are such that the high speed requirements on the imaging systems often necessitate a sacrifice in the lower resolution and lower dynamic ranges (Chien and Snyder, 1975).

Sophisticated programming efforts are invariably an integral part of an image processing system. Examples of such programming systems are: NASA/JSC's ERIPS, Purdue University's LARSYS, University of Maryland's PAX, University of Kansas' KANDIDATS, and Rome Air Development Center's OLPARS (Sammon, 1968 and Kanal, 1972). In addition to the final classification functions, these programming systems also possess a number of important preprocessing capabilities. Among them are data editing, greyscale histograms and normalization, registration, spectral plot, training fields selection, statistics computation, separability evaluation, feature selection, and clustering. Normally, these preliminary preprocessings have to be completed before the classification can proceed, and the final decision can be made on whether the pixel will be assigned into the categories of interest or into the threshold group. In the next section, a brief account of some preprocessing functions will be given.

Preprocessing Functions

The preprocessing is a catch-all functional group in the image data processing system that must be performed before the final classification. Generally, the needs for preprocessing arise from various operational requirements.

Registration of different images is required for temporal processing such as for LACIE, sensor-to-sensor comparison such as found in Chang (1974), and for color composite preparations.

Gain and bias corrections are designed to account for different dynamic ranges in individual detectors.

Geometric distortions due to nonlinearities of scanning system and changes in the attitude and altitude of the sensor need to be removed to standardize the processing results (O'Handley and Green, 1972).

Atmospheric corrections for compensating scattering and diffraction effects are crucial in the signature extension processing in large areas and/or long duration surveys application. ERIM is one of the image processing institutions extensively engaged in the atmospheric haze and sun angle corrections research Two preprocessing algorithms, XSTAR and XBAR, have been recently developed by ERIM (Lambeck, 1977). The XSTAR algorithm is intended to provide significant compensation for the effects of atmospheric haze and sun illumination angle in LANDSAT MSS data.

The XBAR algorithm is a sophisticated version of XSTAR designed to compensate for not only the atmospheric and sun illumination effects, but also the scan angle and background albedo factors. A piecewise linear approximation of the signal X recorded by a multispectral scanner can be expressed in terms of an input radiance L, a gain of the scanner G, and an additive signal offset δ by: $X = GL + \delta$ (1)

The quantities X, G, L, and δ are all functions of the spectral wavelength of an individual scanner channel. Furthermore, the input radiance L, at a given wavelength, observed by a satellite, is a complicated function of the target reflectance the optical thickness of the atmosphere, the sun angle and the scan angle. However, an appropriate approximation enables one to obtain a standardized signal X' which is related to the original signal value X by the multiplicative and additive factors A and B, respectively:

$$X' = Ax + B \tag{2}$$

The XSTAR and XBAR algorithms are derived using the above relation.

Although some positive experimental results have been demonstrated with the aforementioned atmospheric correction algorithms, an additional development of similar algorithms for the sensor other than LANDSAT multispectral scanner should be pursued, and further evaluation of the algorithms in non-agricultural imageries should be conducted in order to assure a step closer to the eventual adaptive automated processing of large scale spatial and temporal imageries.

A desire always exists to edit the image data before input to a classifier. For example, identification and removal of clouds, cloud shadows, water, snow, dense haze, and other wild points from image data would clearly enhance the classification performance. Moreover inhibition of a part or all of these data from transmitting to the ground stations constitutes a form of data compression which would definitely ease the bandwidth constraints in the image data downlink transmission. Recently, a set of algorithms called SCREEN has been devised by ERIM based on studies of 32 LACIE acquisitions (Lambeck, 1977). Using SCREEN, an accurate separation of water, clouds, and cloud shadows from other categories is reported to be possible without the aid of ground truth information. To apply the SCREEN algorithms, the LANDSAT data is first standardized through a simple sun angle correction, and then rotated according to the Tasselled Cap Transformation (Kauth and Thomas, 1976). The linear threshold used in the screen in terms of these standardized data vector z are respectively as follows. (Note that z is a vector of 4-dimensions as in the case of the LANDSAT data vector):

(a) A pixel is labeled as garbled data if it satisfies the following inequalities:

$$z_4 > 16 \text{ or} \tag{3a}$$

$$z_4 < -12 \text{ or} \tag{3b}$$

$$z_3 - 0.1\, z_1 > -4 \text{ or} \tag{3c}$$

$$z_3 + 0.2\, z_1 < -14 \text{ or} \tag{3d}$$

$$z_2 + 0.1\, z_1 < -20 \text{ or} \tag{3e}$$

$$z_2 + z_1/1.8 > 156 \text{ or} \tag{3f}$$

$$z_2 - z_1/1.2 > -8 \tag{3g}$$

(b) A pixel is labeled as cloud (or snow) if not labeled garbled, and if the following conditions are satisfied:

$$z_1 > 100 \tag{4a}$$

$$\text{and } z_3 + 0.1\, z_1 < -7.5 \tag{4b}$$

(c) A pixel is labeled as dense haze if it does not fit into either mentioned above and if the following inequalities are satisfied:

$$z_1 > 69 \tag{5a}$$

$$\text{and } z_3 + z_1/7 < -3.25 \tag{5b}$$

(d) A pixel is labeled as water if it has not yet been categorized and the following inequalities are satisfied:

$$z_1 < 75 \text{ and} \tag{6a}$$

$$z_2 + z_1/16 < -5 \tag{6b}$$

$$z_4 < 1.5 \text{ and} \tag{6c}$$

$$z_2 + z_4 < -4.5 \text{ and} \tag{6d}$$

$$z_2 + 0.5\, z_1 + z_3 + 5z_4 < 10 \tag{6e}$$

(e) A pixel is labeled as cloud shadow if none of the above category requirements have been met and the following inequalities are true.

$$z_2 - 0.4\, z_1 - 0.6\, z_3 - 0.6\, z_4 > -9 \text{ and} \tag{7a}$$

$$z_1 - 0.4\, z_2 < 38 \tag{7b}$$

Any pixel left unlabeled is then suitable for subsequent classification processing. It is clear that the major advantage of this algorithm is its simplicity and ease in implementation. However, it would be much more interesting if an adaptive SCREEN-like algorithm be developed for other than LANDSAT multispectral scanner data.

Feature selection is another important preprocessing function which attempts to preserve the maximum separability of the data set while reducing the dimensionality to ease the classification processing. Experimental results seem to indicate that four to six channels would be an optimal dimension in data vectors. Essentially two fundamental problems are inherent in the feature selection: feature selection criteria and feature subset search procedures. The first problem is concerned with the criteria on which the evaluation of a feature subset's effectiveness is to be based. Usually, the criteria which are most capable of revealing the classification error are the ones that should be used for feature selection. Among those noteworthy criteria are the probability of misclassification (PMC), the Mahalanobis distance, the Divergence measure, and the Bhattacharyya distance. Their definitions for two-class cases are briefly given below (Duda and Hart, 1974; Kanal, 1974; Chang, 1978; and Anderberg, 1973).

(a) $$PMC = q_1 \int_{R2} P_1(x)\, dx + q_2 \int_{R1} P_2(x)\, dx \tag{8}$$

where: q_I is the a priori probability for class i

P_i (x) is the conditional probability density function for class i

R_i is the decision region for class i

x is the data vector consisting of the feature subset to be evaluated

(b) Mahalanobis distance

$$\rho_M = (\mu_1 - \mu_2)^T \ \Sigma^{-1} \ (\mu_1 - \mu_2) \tag{9}$$

μ_i is the mean vector of class i

Σ is the covariance matrix of the data set

μ^T indicate the transpose of a vector

Σ^{-1} is the inverse matrix of Σ

(c) Divergence measure

$$\rho_D = \int \left[P_1(x) - P_2(x)\right] \ \ln \left[\frac{P_1(x)}{P_2(x)}\right] \ dx \tag{10}$$

(d) Bhattacharyya distance

$$\rho_B = -\ln \left[\sqrt{P_1(x) \quad P_2(x)} \ dx\right] \tag{11}$$

The probability of misclassification is the best criterion, but is difficult to implement. The Mahalanobis distance is simpler to evaluate. Between the last two criteria, the Bhatt Bhattacharyya distance is more effective in revealing the probability of misclassification than the divergence measure. For the feature subset search procedures, there are a number of approaches reported in the literature, among them: a) exhaustive search procedure, b) without-replacement procedure, c) dynamic programming procedure, d) linear combination, and e) branch and bound procedure. It would be intuitively clear that, except linear combination, only the exhaustive procedure is the optimal one in finding the best feature subset. Nevertheless, the number of subsets required for evaluation is often exceedingly large. An example of selecting two features out of eight requires evaluation of $C^8_2 = 28$ subsets. On the other hand, the suboptimal procedures usually require far less subsets for evaluation. Indeed, the without-replacement procedure, one of the easiest to implement, needs to evaluate only 15 subsets in this case. Briefly, a without-replacement procedure begins with the selection of the best single feature in accordance with a given criterion. Then, the remaining features are scanned for the next best single feature, resulting in the best pair of features when combined with the previously chosen best single feature, etc. Usually, a considerable improvement in throughput can be achieved by using one of the suboptimal search procedures in cases of high dimensionality in data vectors.

Ground truth information or the collection of representative samples in each class of interest, is often used in conjunction with various preprocessing activities. Like statistical parameter estimations, spectral signature identifications, classification performance predictions, and feature selections, all of these rely heavily on the premise of correctness of ground truth information. The need for the quality ground truth information has been a serious bottleneck to automated image data processing. This is especially true for the large scale spatial temporal multi-category image processing. One way to get around this problem is to use the unsupervised clustering analysis to partition the spectral spatial (or even temporal) pattern space.

Clustering is a process of finding groups of data points or clusters such that the degree of similarities is strong within members of the same group and weak between members of different groups (Chang, 1978 and Andenberg, 1973). Clustering analysis has long been applied to the homogeneity detection and subclass selection at Purdue LARS and NASA/JSC. The algorithm used is based on the k-means clustering procedure which is essentially an iterative clustering. The basic k-means clustering has an inherent convergence property. However, its clustering results are usually not unique in that different starting points lead to different final clustering configurations. The LACIE clustering algorithm is a modification of k-means clustering with additional options of allowing splitting and combining clusters. With these options, the number of final clusters may not be necessarily the same as the number of initial means. Because of splitting and combining capabilities, the clustering process is no longer convergent. The final clustering results are largely dependent on various parameters used in the program. Nevertheless, the LACIE clustering is still being used as a diagnostic means in detecting the homogeneity of data set.

At ERIM the clustering analysis has been utilized in recent years as a part of training sample selection procedures (Kauth et al., 1977). In their clustering schemes, not only spectral information is involved in the partitioning process, the spatial (coordinates of image data), temporal (biophases of image data), and even the ancillary data like meteorological information are fed into the clustering process. Thus, such homogeneous fragments (called blobs at ERIM) of image data extracted by spatial-spectral-temporal clustering are used as training samples, to model a classifier in order to perform a large area and/or long duration image data processing.

An alternative approach to alleviate the ground truth acquisition prior to processing was suggested by Nagy et al. (1971). The suggestion is to apply a single-pass clustering algorithm to the raw image data. An interpretation of clustering results is then rendered after the processing based on judiciously sampled ground truth information. With incorporation of geographical proximity and spectral similarity into their single-pass clustering algorithm, it was shown that such an approach based on clustering the image data without prior information about the crop categories would yield useable results on unpreprocessed noisy data. Clearly, much work still needs to be done in this respect in order to lay the foundation for developing onboard image data processing systems.

Classification Schemes

Almost without exception, the classification schemes employed in the image data processing systems within the scope of this report fall into the realm of statistical pattern recognition which is in contrast with another basic approach called syntactic pattern recognition. In essence, the statistical pattern recognition is devoted to the interpretation of statistical relationships among scalar measurements. This approach usually assumes the existence of underlying multivariate distributions or discriminant funtions for each class of interest. The classification problem is then transformed into that of developing a decision rule from those distribution functions and classifying each new pixel with this rule. Whereas the syntactic pattern recognition focuses on preceptually higher level structural elements than scalar measurements for interpretation. The grammars are inferred from primitive structural elements, and the pattern is then constructed and classified in accordance with the grammars.

At Purdue LARS, Michigan ERIM, and NASA/JSC, the classification schemes invariably center upon the maximum likelihood approach. Being a special case of Bayesian classification, their schemes involve the use of equal a priori probabilities and the multivariate Gaussian assumptions for the class-conditional probability density functions. The quadratic decision boundaries are derived from the exponential term of the Gaussian density function

$$g_i(X) = (X - \mu_i)^T \Sigma_i^{-1} (X - \mu_i) \qquad (12)$$

where X is the multispectral data vector, μ_i is the mean vector for ith class, and Σ_i is the covariance for ith class. The maximum likelihood decision rule is then to assign the pixel having the value X into class i if $g_i(X)$ exhibits the smallest value.

This Bayesian approach to classification emphasizes the minimization of the cost of misclassification. As such, it takes into account the consequences of decisions during decision processes. In fact, theoretically if the a priori probabilities and the class-conditioned probability distribution functions are known, the Bayesian classification always leads to the minimum probability of misclassification (Anderson, 1958). In practice, however, both the a priori probabilities and the statistical parameters of distribution functions have to be estimated from a group of representative training samples. The quality of training samples thus directly affect the classification accuracy. As mentioned in the preceding section, ERIM has been looking into the blob-clustering techniques to group training samples. The LACIE at NASA/JSC recently employed random selection of systematic grid points for training sample generation. Although their experiments have resulted in slight improvement in the classification accuracy, the problem of training sample selection is far from resolved.

From the computational point of view, the evaluation of the quadratic term, Equation (12), for each pixel for all classes of concern presents a burden to the classification processing. Cases of high dimensionality and large number of generic classes drastically reduce the throughput of data processing. Attempts have been made to process multispectral data through hybrid parallel processing (Marshall and Kriegler, 1970 and Chang and Hayden, 1972). In recent years, a special purpose processor configured with digital parallel processing has been implemented at JSC for LACIE applications. Another entirely different approach called Table Look-Up has been suggested to increase the speed of classification (Eppler <u>et al.</u>, 1971 and Eppler, 1974). Their experiments exhibit promising results. It is reported that the Table Look-Up Approach in one instance has reduced processing time by a factor of 30 compared with the conventional series processing. Another advantage is that the Table Look-Up Approach provides a flexible means of implementing not only parametric Gaussian classifiers but also any non-parametric classifiers tailored to one's specific applications.

Perhaps one of the most important examples in the non-parametric approach is the linear classifier which is generally characterized by a set of linear discriminant functions:

$$g_i(X) = w_1x_1 + w_2x_2 + \ldots + w_dx_d + w_{d+1}, \quad i = 1, \ldots, n \quad (13)$$

where w_i's are the constant weights to be determined, d is the dimensionality of data vector, and n is the number of classes. Historically speaking, the first successful synthesis of the linear machine as the learning element of a neuron-like system (called perceptron) is credited to Rosenblatt (1961). A variety of names such as ADALINES (Adaptive Linear Devices) and TLU (Threshold Logic Unit) have appeared in the literature since then. Many error-correction procedures have also been developed in connection with training linear classifiers (Duda and Hart, 1973). Among those algorithms tested using remotely sensed data are as described by Darling and Joseph (1968): a) the fixed-increment error correction rule; b) the mean square error correction method; c) forced adaptive learning; d) Madaline--another piecewise linear method, e) Bayes Weights technique--yielding optimal solution if the spectral bands are statistically independent; and f) minimum loss approach. Each algorithm has its unique advantages and limitations. Several of them even possess some forms of adaptive learning capabilities. However, the experimental results revealed no one single scheme notably superior to the others. In fact, the convergence of some of these iterative algorithms is assured only by the linear separability of the data set, rendering these algorithms unworkable in linearly non-separable cases. A procedure developed by Ho and Kashyap (1965) is a useful one in that it optimizes the search process with the convergence guaranteed when the data set is linearly separable, but it will stop the search process in finite steps when the data set is linearly non-separable.

In the context of agricultural spectral signatures, a couple of simple non-parametric classification schemes have been proposed in recent years. They are: a) the ratio processing, and b) the delta classification.

The ratio techniques for monitoring vegetation biomass and processing agricultural image data have increasingly become popular in the remote sensing community (Tucker, 1977; Killer, 1976; Siegal and Goetz, 1977; and Shappell and Tietz, 1978). The use of a radiance ratio of 0.8/0.675 μm for determining the leaf area index for forest canopies was first reported by Jordon (1969). Thereafter, the investigations of the ratio techniques have spread over a variety of applications. In the field of LANDSAT image processing, a ratio of channel 4/channel 2 was found to relatively effective in quantifying the greenness or dryness condition of the grassland. Some recent studies,

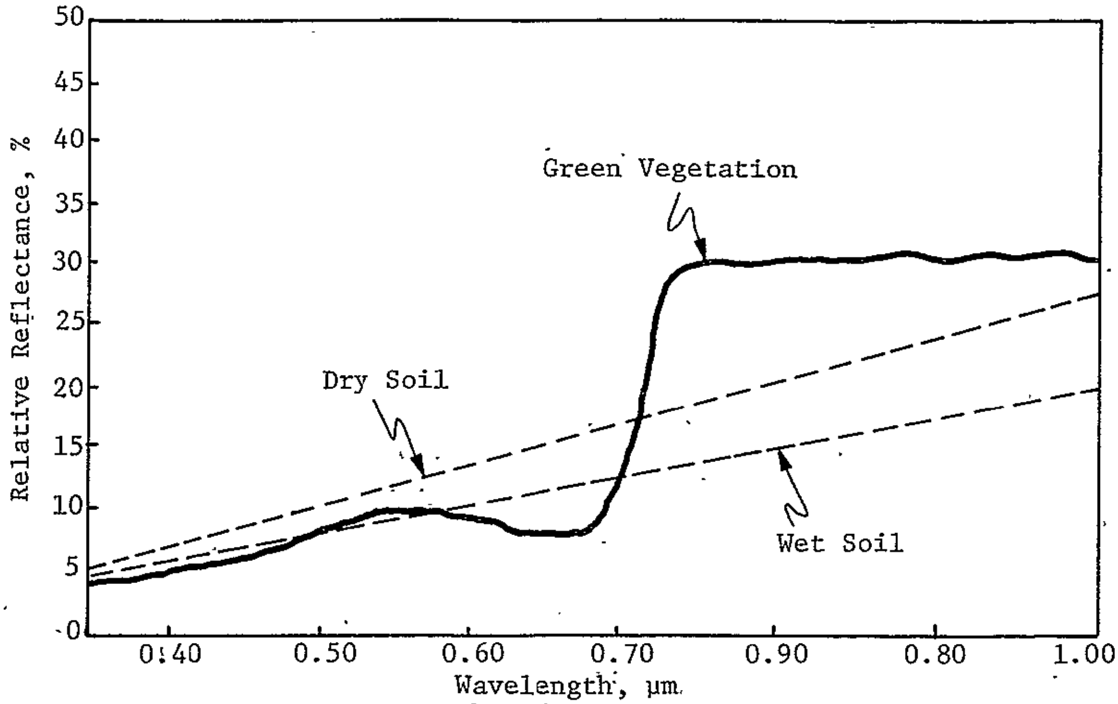

Figure V-2 Spectral Reflectances for Dry Soil, Wet Soil, and Green Vegetation (after Figure 2 of Reference 28).

however, indicated that the ratio of channel 3/channel 2 appeared to contain more agricultural information and was more statistically significant than channel 4/channel 2. The strong spectral absorption of incident radiation by chlorophyll molecules over the range 0.63 - 0.69 µm as illustrated in Figure V-2 is in part attributable to the effectiveness of channel 3/channel 2 processing.

In view of equations (1) and (2), it is conjectured that some of the multiplicative factors such as the haze and atmospheric effects could also be partially compensated by the ratio techniques. As to the spectral bandwidths used in the ratioing channels, limited studies showed that no substantial differences were found in regression significance among the infrared bandwidths of 0.75 - 0.80 µm, 0.80 to 0.90 µm, 0.80 to 1.00 µm, and 0.75 to 0.90 µm.

Along a slightly different direction, the channel difference techniques have been investigated for LANDSAT image processing (Killer, 1976; Rouse et al., 1973; Rouse et al., 1974 and Richardson and Wiegand, 1976). It was found that the Vegetation Index (VI) and the Transformed Vegetation Index (TVI), defined respectively as:

$$VI = \frac{\text{channel 4 - channel 2}}{\text{channel 4 + channel 2}} \tag{14}$$

$$TVI = \sqrt{VI + 0.5} \tag{15}$$

might be able to compensate the signature errors due to the geographical location and growth cycle deviations that would otherwise be introduced in the ratio of channel 4/channel 2 or channel 3/channel 2. Another criterion, Perpendicular Vegetation Index (PVI), was devised to account for the distance from the data point to the so-called soil background line in the feature space:

$$PVI = \frac{1}{2.6} \; (2.4 \text{ (channel 4) - channel 2)} \tag{16}$$

The experimental results suggested that PVI = 0 indicates bare soil, a negative PVI indicates water, and a positive PVI indicates vegetation. The soil background line is a very useful indicator in the LANDSAT agricultural imageries. It was first developed by Kauthand Thomas (1976) at ERIM. Subsequent work by Thompson and Wehmanen (1977) utilized the soil line information to generate what they referred to as the green number. The procedure of relating the green numbers to drought conditions appears to be reliable. It has become a routine process at JSC to screen the LANDSAT imageries by means of the green number information for drought conditions.

The delta classification technique is another non-parametric scheme designed to classify remotely sensed LANDSAT data without ground truth information (Tubbs and Engrall, 1975). This technique first transforms the original LANDSAT data vector X into a three-dimensional vector according to the formula:

$$\begin{aligned} y_1 &= x_1 - x_2 + 32 \\ y_2 - x_2 &= x_3 + 32 \\ y_3 - x_3 &= x_4 + 32 \end{aligned} \tag{17}$$

These values are then normalized:

$$z_1 = y_1/(y_1 + y_2 + y_3)$$
$$z_2 = y_2/\ (y_1 + y_2 + y_3) \tag{18}$$
$$z_3 = y_3/\ (y_1 + y_2 + y_3)$$

The resultant vector Z' is finally plotted on a two-dimensional triangular shaped domain. If the LANDSAT data acquisitions corresponding to the four biophases of crops (i.e., when it is in the emerging state, greening state and harvesting state) are available, then a unique trend can be observed by plotting pixels of a specific crop type on the triangular domain. It is reported that the trend an individual crop makes is fairly consistent regardless of geographical location.

Concluding Remarks

All of the forecasts and the future space mission scenarios have pointed to massive increases in image data return from the space-borne sensor platforms designed to provide global monitoring of agriculture, minerals, forest, land, marine, and water resources (NASA Space Technology Forecast, 1976 and Total Earth Resources Prediction by General Electric, 1974). In order to increase use of large-scale data banks, multiplicity of peripheral equipment and services, and sophisticated analytical and computational capabilities, a highly efficient end-to-end data management program will be needed more than ever. To ensure widespread and timely dissemination of resource monitoring results to the user communities, one of the ultimate solutions to the image data processing systems would lie in the realization of adaptive real-time onboard processors capable of transmitting processed results rather than just raw data.

To acquire different generic type surface categories for specific applications, to exclude many unwanted data sets from further processing, and to facilitate large area and/or long duration surveys, adaptive capabilities would seem to be desirable in the following areas: a) searching and tracking observation coverages; b) optimally selecting measurement sensors and channels; c) screening and editing image data, d) signature extension preprocessing, and e) adaptive modeling of classification schemes.

It is clear from the preceding sections that very limited onboard processing has been developed for space missions.

To achieve this goal, the following areas should receive immediate attention: a) the spectral analysis of categories of interest from many different sensors responses; b) adaptive training sample selections; and c) signature extension for remote sensing applications over a wide geographic area and long temporal duration. The point to be stressed here is the need for adaptive machine processing that would utlimately maximize quantitative system returns. It is hoped that the drudgery of human operations in photointerpretation and other preprocessing activities could eventually be relieved by highly automated spaceborne image processing systems.

REFERENCES

"A Forecast of Space Technology 1980-2000," NASA SP-387, Washington, D. C., January, 1976

Anderson, T. W., An Introduction to Multivariate Statistical Analysis, Wiley, 1958.

Anderberg, M. R., Cluster Analysis for Applications, Academic Press, 1973.

Chang, C. Y., "Skylab S192 Data Evaluation: Comparisons with ERTS-1 Results," LEC-1711, Lockheed Electronics Company, Houston, Texas, January, 1974.

Chang, C. Y., and Hayden, D. R., "A Prototype Hybrid Multispectral Processing System," LEC/HASD No. 640-TR-143, Lockheed Electronics Company, Houston, Texas, September, 1972.

Chang, C. Y., "Introduction to Statistical Pattern Recognition with Applications in LACIE," LEC-11701, Lockheed Electronics Company, Houston, Texas, January, 1978.

Chien, R. T., and Snyder, W. E., "Hardware for Visual Image Processing." IEEE Trans. Circuits and Systems, Vol. CAS-22, No. 6, June 1975

Darling, E. M., Jr., and Joseph, R. D., "Pattern Recognition from Satellite Altitude," IEEE Trans. Systems Science and Cybernetics, Vol. SSC-4, pp. 38-47, March, 1968.

Duda, R. O., and Hart, P. E., Pattern Classification and Scene Analysis, Wiley, 1973.

Eppler, W. G., Helmke, C. A., and Evans, R. H., "Table Look-Up Approach to Pattern Recognition," Proc. 7th International Symposium Remote Sensing Environment, Ann Arbor, Michigan, April, 1974.

Eppler, W. G., "An Improved Version of the Table Look-Up Algorithm for Pattern Recognition," Proc. 9th International Symposium Remote Sensing Environment, Ann Arbor, Michigan, April, 1974.

REFERENCES (Continued)

"ERTS Reference Manual," General Electric, Valley Forge Space Center, Philadelphia, Pennsylvania.

Ho, Y. C., and Kashyap, R. L., "An Algorithm for Linear Inequalities and Its Applications," IEEE Trans. Electronic Computers, EC-14, 1965.

Jordon, C. F., "Derivation of Leaf Area Index from Quality of Light on the Forest Floor," Ecology, Vol. 50, No. 4, April, 1969.

Kanal, L. N., "Interactive Pattern Analysis and Classification Systems: A Survey and Commentary," Proc. IEEE, Vol. 60, No. 10, October, 1972.

Kanal, L. N., "Patterns in Pattern Recognition: 1968-1974," IEEE Trans. Information Theory, Vol. IT-20, No. 6, Nov., 1974.

Kauth, R. J., and Thomas, G. S., "The Tasselled Cap--A Graphic Description of the Spectral-Temporal Development of Agricultural Crops as Seen by LANDSAT," Symp. Proc. Machine Processing of Remotely Sensed Data, Purdue University, W. Lafayette, Indiana, June 29 - July 1, 1976.

Kauth, R. J., Pentland, A, and Thomas, G. S., "Blob, An Unsupervised Clustering Approach to Spatial Preprocessing of MMS Imagery," Eleventh International Symposium on Remote Sensing of Environment, ERIM, Ann Arbor, Michigan, April 25-29, 1977, pp. 186-187.

Killer, R. M., "Channel Ratio Processing," LEC-7780, Lockheed Electronics Company, Houston, Texas, March, 1976.

Lambeck, P. F., "Signature Extension Preprocessing for LANDSAT MSS Data," ERIM 122700-32-F, Environmental Research Institute of Michigan, Ann Arbor, Michigan, November, 1977.

Lambeck, P. F., Thomas, G. S., and Kauth, R. J., "SCREEN, A Procedure for Automatically Detecting Garbled Data, Clouds, Snow, Cloud Shadows, and Water in LANDSAT MSS Data," NASA New Technology Report, ERIM, Ann Arbor, Michigan, November 10, 1977.

Landgrebe, D. A., et al., "LARSYSAA, A Processing System for Airborne Earth Data," LARS, Purdue University, Info. Note 091968.

REFERENCES (Continued)

Marshall, R. E., and Kriegler, F. J., "Multispectral Recognition System," Proc. 9th IEEE Symposium Adaptive Processes, Decision and Control, Austin, Texas, December, 1970.

Nagy, G., "Digital Image-Processing Activities in Remote Sensing for Earth Resources," Proc. of IEEE, Vol. 60, No. 10, October, 1972.

Nagy, G., Shelton, G, and Tolaka, J. "Procedural Questions in Signature Analysis," IBM Research RC 3517, Yorktown Heights, New York, August, 1971

O'Handley, D. A. and Green, W. B., "Recent Developments in Digital Image Processing at the Image Processing Laboratory at the Jet Propulsion Laboratory," Proc. of IEEE, Vol. 60, No. 7, July, 1972.

Richardson, A. J., and Wiegand, C. L., "Criteria for Distinguishing Vegetation from Soil Background Information and Their Use in Processing LANDSAT MSS Data," Agricultural Research Service, USDA, Weslaco, Texas, 1976.

Rosenblatt, F., Principles of Neurodynamics: Perceptron and the Theory of Brain Mechanisms. Spartan Books, Washington, D. C., 1961.

Rouse, J. W., Haas, R. H., Schell, J. A., and Deering, D. W., "Monitoring Vegetation Systems in the Great Plains with ERTS," Third ERTS Symposium, NASA SP-351, December, 1973.

Rouse, J. W., Maas, R. H., J. A. Schell, Deering D. W., and Harlan, J. C., "Monitoring the Vernal Advancement and Retrogradation (Greenwave Effect) of Natural Vegetation," NASA/GSFC Type III Final Report, Greenbelt, Maryland, 1974.

Sammon, J. W., Jr., "On-Line Pattern Analysis and Recognition System (OLPARS)", Technical Report No. RADC-TR-68-263, August 1968.

Schappell, R. T., and Tietz, J. C., Video Landmark Identification, Acquisition and Tracking, Paper No. 78-1702, Presented at the AIAA/NASA Conference on "Smart" Sensors, November 14, 1978.

REFERENCES (Continued)

Siegal, B. S., and Goetz, A. F. H., "Effects of Vegetation on Rock and Soil Type Discrimination," Photogrammetric Engineering and Remote Sensing, Vol. 43, No. 2, February, 1977.

Thompson, D. R., and Wehmanen, O. A., "The Use of LANDSAT Digital Data to Detect and Monitor Vegetation Water Deficiencies," Eleventh International Symposium Sensing of Environment, Ann Arbor, Michigan, April, 1977.

"Total Earth Resources System for the Shuttle Era," Vol. 3, DRL No. T-880, General Electric, Valley Forge Space Center, Philadelphia, Pennsylvania, November, 1974.

Tubbs, J., and Engvall, J., "A Non-Parametric Approach to Classifying Remotely Sensed Data," Tenth International Symposium Remote Sensing of Environment , Ann Arbor, Michigan, October, 1975.

Tucker, C. J., "Use of Near Infrared/Red Radiance Ratios for Estimating Vegetation Biomass and Physiological Status," X-923-77-183, Goddard Space Flight Center, Greenbelt, Maryland, July, 1977.

VI GROUND SUPPORT SYSTEMS

Introduction

This chapter provides a synopsis of the current status of advanced automation technology in ground support systems. Recommendations are made that will further enhance the efficiency of ground support systems while continuing to reduce the cost of such complex systems.

A cost analysis task presented to NASA in April of this year has indicated that substantial benefits can be obtained through automation of ground operations (mission operation, design and test of hardware), increased spacecraft automation, data analysis and orbital operation. These benefits include direct cost savings due to reduced man hours and indirect benefits due to more efficient use of facilities and hardware investment. It has been estimated that without automation, NASA's yearly costs in the 1980s to support the currently proposed space activities will be 3.57 billion dollars (calculated in 78 dollars). Table VI-1 is a summary of the main results of the cost analysis task entitled "A Benefit and Role Assessment of Advanced Automation for NASA" published in April, 1978. (McReynolds, 1978)

Table VI-1 Estimated Yearly Cost in 1980s for NASA Without Automation (Millions of 78$)

Budget Category	Ground Operations	Orbital Operations	Data Analysis	Design	Test	Other	Total
1. Mission Items	121.7	--	15.4	89.6	116.0	90.9	433.6
2. Multimission Operation Support	73.8	--	--	--	--	73.8	73.8
3. Postmission Data Analysis	--	--	103.4	--	--	--	103.4
4. Network Operations	43.7	--	--	56.5	7.6	107.5	215.5
5. Shuttle/Skylab Payloads	--	13.6	20.0	80.0	100.0	--	213.6
6. Space Transportation	182.0	--	--	--	908.0	840.0	1930.0
7. Space Industrialization	224.0	24.0	--	72.0	120.0	160.0	600.0
Total	645.2	37.6	138.8	298.1	1251.6	1098.4	3570.0

The analysis results state that direct cost savings of 1.5 billion dollars (78 dollars) are possible per year by the year 2000. However, it should be noted that this amount does not include estimated development costs required to achieve these savings. Table VI-2 is a summary of the predicted cost savings per year in millions of 78 dollars that can be achieved through the introduction of advanced automation technology. (McReynolds, 1978)

Table VI-2 Estimated Potential Yearly Cost Savings by Year 2000 (Millions of 78$)

Budget Category	Ground Operations	Orbital Operations	Data Analysis	Design	Test	Total
1. Mission Items	90	--	10	45	80	225
2. Multimission Operations Support	55	--	--	--	--	55
3. Postmission Data Analysis	--	--	70	--	--	70
4. Network Operations	30	--	--	30	5	65
5. Shuttle/Skylab Payloads	--	10	10	40	65	120
6. Space Transportation	135	--	--	--	605	740
7. Space Industrialization	170	10	--	--	80	300
Total	480	20	90	150	835	1575

Note: Rounded to the nearest $5 million.

Table VI-2 figures are based on the following assumptions:

1. Budget categories 1 through 4 will continue to be funded at the same level.

2. The space transportation area is based upon the 1976 Shuttle traffic model and includes reimbursed costs.

3. Areas 6 and 7 are based upon predictions.

Quite evidently, from Table VI-2, further automation of ground support systems will contribute significantly to the estimated yearly cost savings; therefore, efforts to apply advanced automation technology to ground support systems should continue. Development and utilization will require NASA support.

To summarize the state-of-the-art in ground support systems, two main areas of support were investigated, the launch vehicle checkout systems and the ground tracking systems. A discussion of both support areas follows, with conclusions and recommendations presented that will serve as guidelines to effectively increase the efficiency of the overall ground support system through the use of specific advanced automation technologies.

Discussion

Launch Vehicle Checkout

Since the Apollo years, the use of automation technology has been increasing in ground support systems. The Viking program of the mid 1970s utilized two ground support systems--the System Test Equipment, STE, for the Viking spacecraft checkout, and the Vehicle Checkout Set, VECOS, for the Titan III launch vehicle. The STE provided a real-time spacecraft checkout system utilizing a single computer, the Honeywell 632, which supplied the first command uplink/downlink telemetry system. The VECOS system was used in conjunction with the Control Monitor Group (CMG) and the Data Recording Set (DRS) to control and monitor prelaunch checkout tests of the Titan III launch vehicle and to control countdown sequences up to engine ignition and liftoff. Though this support system was at the time the most sophisticated system developed, the hardware involved was old and rapidly becoming obsolete while replacement parts were becoming harder to locate.

To replace this checkout system, a more advanced launch vehicle checkout system has been designed within the last three years. The Programmable Aerospace Control Equipment, PACE, was developed to replace the three systems--VECOS, CMG, and DRS. Building on automated testing and control technology developed for the Viking STE, PACE integrates three MOD COMP 11/45 mini computers and a microprocessor into one flexible computer system. The functional requirements of PACE are listed below:

1. To perform and evaluate launch vehicle subsystem and system tests

2. To perform and evaluate the launch countdown sequence, which includes the following:

   a. Failsafe provision - The design of the PACE system precludes damage to the space system launch vehicle circuitry or setup of hazardous test conditions as a result of a single failure in the PACE system operating in the triple redundant launch countdown mode.

   b. A single malfunction of the PACE system in the automatic launch countdown shall not allow the launch of a malfunctioning vehicle but may hold an acceptable vehicle.

   c. A single failure in the PACE system shall not issue an erroneous launch critical command.

3. To record all discrete monitor changes for troubleshooting and post-test evaluation as required.

A summary of the PACE system capabilities, which are presented in the PACE Design Criteria, are listed below:

1. Concurrent single-ended testing at two locations
2. Triple redundant launch countdown control and monitoring
3. Monitor and process 1280 discrete inputs and record changes of state
4. Monitor up to 208 analog inputs (12 at a time) (300 samples per second)
5. Automatic on-line evaluation based on test sequence criteria
6. Automatic abort on critical parameters
7. Test sequences written in vehicle test language by test engineers
8. Data driven system with vehicle and aerospace ground equipment (AGE) interface data contained in data file
9. Process up to 16 discretes in 3.33 milliseconds
10. Display criteria violations on CRT and/or printer
11. Display any monitored channel on operator request
12. Record test data and operator actions on operations log tape for post-test evaluation as required
13. Issue discrete and analog stimulus as required by test sequence

14. Drive countdown readout displays

The PACE has recently been installed at the ETR launch facility and will soon be installed at the WTR launch facility to provide the checkout and monitoring of future Titan III launches.

The Launch Processing System (LPS), developed to provide the checkout and control system for the Space Shuttle Vehicle (SSV), best represents the state-of-the-art in automated ground support checkout systems. LPS consists of a network of computers, data links, displays, controls, hardware interface devices, and computer software used to prepare the Shuttle for launching. Modern automation techniques, off-the-shelf components and modular design are being used to achieve the goals of LPS. Prior to launch, the LPS monitors and controls a myriad of SSV and ground support equipment (GSE) currents, voltages, pressures, temperatures and data exchanges. Should problems occur anywhere in the complex launching equipment, the LPS alerts a responsible person so that the problem can be corrected, thus assuring a successful launch.

The Space Shuttle Vehicle will serve as the major launch vehicle throughout the 1980s. With estimates of Shuttle vehicle flights ranging from a 1974 traffic model of 573 flights by the year 1992 to a 1977 traffic model of 1,091 flights (McReynolds, 1978) within the same time frame, it becomes increasingly important that ground operations be made as uncomplicated and modernized as possible in order to facilitate ground turn-around and in-orbit support with minimum crews. These two factors, the number of anticipated Shuttle launches and ground turn-around of 160 hours processing time, proved to be the most demanding factors in the design of Ground Support Equipment for the Space Shuttle checkout.

The LPS development guidelines, when evaluated with the Shuttle operations philosophy, have resulted in a system design with the following characteristics (Byrne, et al., 1976).

1. High degree of automation;
2. Standard and modular hardware and software;
3. General purpose and multiple use, high density, non-dedicated consoles;
4. Test-engineer-oriented language for application programming;
5. Readily available planning and engineering information.

The LPS comprises three subsystems: The Checkout, Control, and Monitor Subsystem (CCMS), the Central Data Subsystem (CDS), and the Record and Playback Subsystem (RPS). The CCMS provides real-time checkout and launch control of the SSV. The CDS provides centralized large-scale computer support to the CCMS for real-time data storage and recall, engineering/management operations, software program library storage, and simulation for software validation and operator training. Recording and playback of raw data for post-test processing and analysis is provided by the RPS.

The LPS integrates all data processing for the Shuttle vehicle into a network of computers performing parallel checkout and monitoring functions. A distributive processing concept is utilized. Figure VI-2 depicts a typical block diagram of the LPS (CCMS Sets, Volume 1, 1978). The LPS consists of a variety of sets of hardware, similar to that shown in the block diagram. These LPS sets are modular in structure, a concept which allows many different configurations to be generated from one common set of hardware. It is the modularity of the LPS hardware and software which provides an extremely flexible system. Each assembly is specifically configured to perform certain tasks unique to that set.

The CCMS is the control portion of the LPS. It consists of computers, displays, controls, data transmission devices, and electronic interface devices that provide a flexible, reliable, cost-effective approach to performing systems testing, launch operations control and status monitoring of the Shuttle vehicle and Ground Support Equipment throughout the ground operations at the launch site. A functional representation of the CCMS launch support configuration is included in Figure VI-1 (Bryne, et al.)

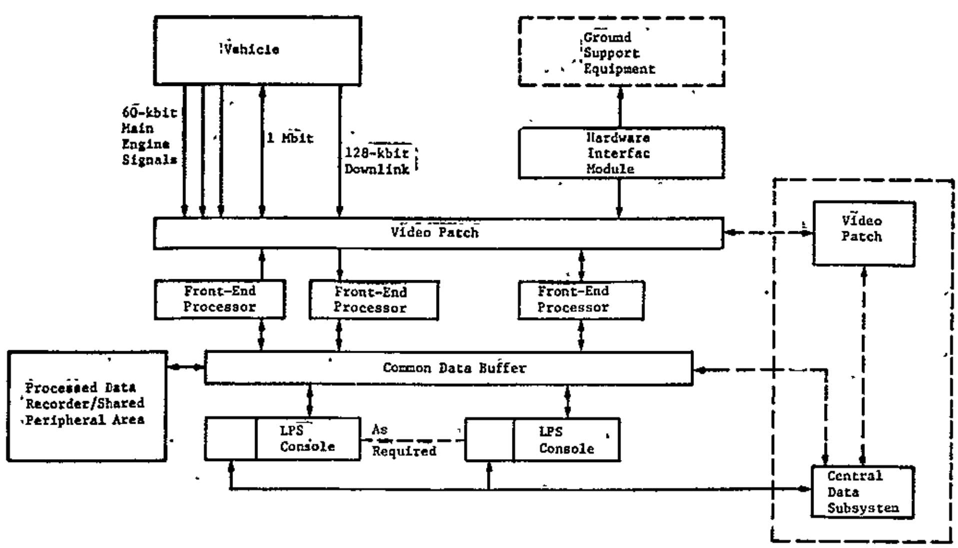

Figure VI-1 Checkout, Control and Monitor Subsystem

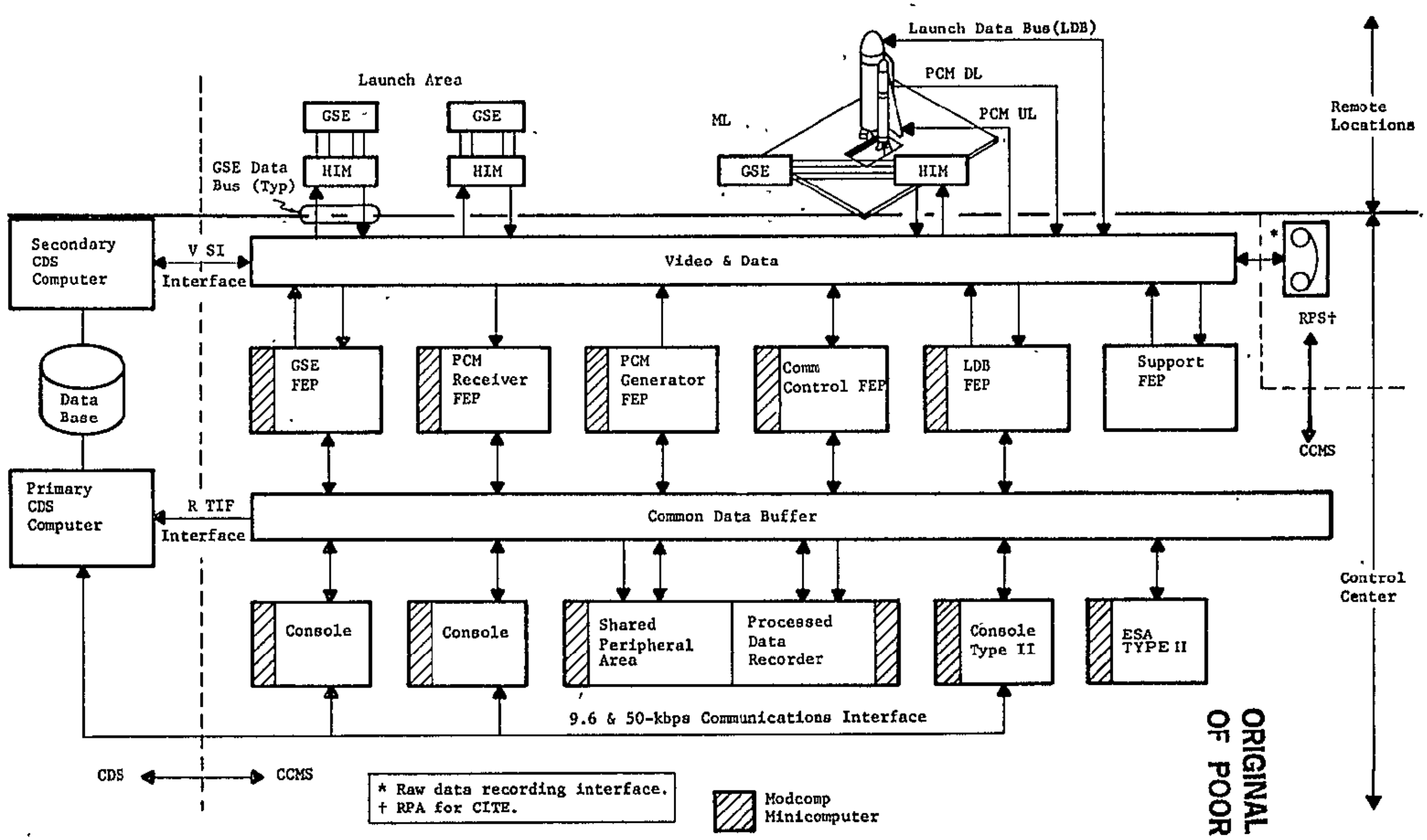

Figure VI-2 LPS Simplified Block Diagram

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1976). Its system design uses state-of-the-art digital computer system architecture schemes and implementation technology. The CCMS systems architecture, which is based on a real-time distributive processing system hierarchy of standardized minicomputers (Mod Comp 11/45) and microprocessors (Motorola 6800), is the first ground support system to utilize the distributive processing concept for checkout and monitoring of a launch vehicle. Large Scale Integration (LSI) logic and memory circuits and microcode/firmware technology is used extensively. The application of these technologies has provided significant benefits which include easy expansion/contraction and redundancy implementation, an increase in data processing/system throughput efficiency and universal use of standardized, modular hardware and software components.

The six basic hardware assemblies illustrated in Figure VI-2 are used to implement the various CCMS set configurations to be installed at Kennedy Space Center and Vandenberg Air Force Base to support Space Shuttle turnaround operations. The six assemblies are (CCMS Sets, Volume 1, 1978):

1. common data buffer (CDBFR)
2. console
3. front-end processor (FEP)
4. processed data recorder/shared peripheral area (PDR/SPA)
5. video and data assembly (V&DA)
6. hardware interface module (HIM)

The common data buffer is key to implementing the CCMS distributive processing concept in that it uniquely accomplishes the very critical CPU-CPU intercommunications task which allows for performance in parallel. The buffer consists of a 64-port, high-speed, solid state, externally shared memory consisting of 64K-32 bit words. The capability exists for multiple common data buffer configurations to exist in any one Launch Control Center area. In addition, first-in, first-out buffer interfaces are attached directly to the address and data bus lines within the common data buffer to provide a direct data-read interface to a number of external devices in parallel. One of 64 buffer Access Cards provides the standard interface from the common data buffer to the rest of the LPS subsystem. Each of the cards is scanned in rotation by the scanner controller and transfered between CPUs on a priority basis. The common data buffer is designed to be transient error tolerant by utilizing error check and correction encoding on both address and data lines.

The primary man/machine interface for operator control and monitoring of all test functions is provided by the minicomputer based CRTS and a four-channel color graphics Display Generator. The consoles provide automatic test sequence execution and formatting/display of appropriate test data in real time. The universal design of the consoles permit any of the connected consoles to be used for control and monitoring of any Shuttle Vehicle or ground support equipment subsystem. Since the system operates on a time-sharing arrangement, each console is able to use the system and communicate with appropriate front-end processors via the common data buffer high speed external shared memory. The time sharing is transparent so that each console appears to be operating independently within the entire system.

Test procedures are written in the Ground Operations Aerospace Language, GOAL, by test engineers. The language allows GOAL test program compilation directly from procedures.

The GOAL application programs are initially loaded in each console computer and remain resident throughout a Shuttle processing cycle, so that each operator has all required procedures on the console disk.

During the execution of any procedure, the console operator may start or stop at any point, branch to any logical step within the procedure, single-step through the procedure or choose to exception-monitor test data where only data which exceeds specific limits are displayed.

The front-end processor is a minicomputer (Mod Comp 11/45) which provides all communication with the ground support equipment and Shuttle vehicle. The front-end processor pre-processes the Shuttle vehicle and ground support equipment uplink and downlink data, performs limit and validity checks, and converts data to an acceptable format for the consoles and common data buffer. Computer controlled commands issued from the various consoles are transmitted via the common data buffer to the appropriate front-end processor for proper formatting and routing to the ground support equipment or Shuttle vehicle. Resulting test data from Shuttle or ground support equipment are received by the front-end processor, decoded and limit-checked for out-of-tolerance conditions. Anomalies detected by the front-end processor are routed to the appropriate console for operator display and evaluation.

The functions of the front-end processor are subdivided into three major areas: Pulse Code Modulation (PCM) Generation and PCM Receiving, Vehicle Launch Data Bus Control, and Ground Support Equipment Hardware Interface Module Control.

The PCM front-end processors provide for issuing uplink commands to the Shuttle avionic subsystems and processing downlink telemetry data from the Shuttle.

The Launch Data Bus provides test control and monitoring of the Shuttle vehicle. Interface with the Shuttle avionics subsystem is provided by a standardized one megahertz Launch Data Bus.

Ground Support Equipment front-end processors, interconnected via a one megahertz Ground Support Equipment Data Bus to remotely located hardware interface modules, are used to control and monitor Shuttle ground support equipment. These front-end processors acquire and process measurement data from Hardware Interface Modules. Measurement processing includes significant data change detection and associated Common Data Buffer status updates. Data are also checked for tolerance conditions and the front-end processor issues output commands to the Hardware Interface Modules.

"The Ground Support Equipment front-end processors throughput requirements limit data bus utilization. Current estimates of processing requirements indicate that 70 to 100 microseconds per measurement sample is the maximum required rate. One-hundred microseconds per measurement sample is equivalent to 10,000 transmissions per second or an approximately twenty-eight percent data bus utilization. In the event of an increase in measurement numbers or sample rate requirement, a front-end processor throughput limitation may occur. In that case, additional data buses and front-end processors can be added. This precludes any subsystem overloading and takes advantage of the baseline front-end processor modular design philosophy." (Byrne, et.al., 1976)

The HIM provides a versatile, general-purpose discrete, analog and parallel digital stimulus and monitoring interface with GSE. A variety of standard input/output cards are available to accommodate specific user requirements. Up to 30 input/output cards can be mixed in any combination for a single HIM. System design permits multiple GSE FEP/Data Bus compliments to be accommodated. Quantities are theoretically limited only by the number of available CDBFR parts.

The PDR/SPA provides a means to record and store common data buffer transactions and provides a line printer and printer/plotter stripchart recorder hardcopy printouts. The primary responsibility of the PDR is recording all data on magnetic tape, with a secondary responsibility of maintaining thirty minutes of test data on the bulk disk for direct access. SPA functions include general peripheral processing for each of the computers and near real-time stripchart recording and plotting.

The video and data assembly subsystem provides signal conditioning, impedance matching and switching of signals between the front end processors located in control center areas and remotely located hardware interface modules, ground support equipment and the Shuttle vehicle. It also includes a micro-processor controlled Video Switch which is used to automatically connect selected ground support equipment and Shuttle vehicle signal lines to the front end processors used to generate CCMS commands and preprocess test data.

In summary, the CCMS enables the Shuttle checkout to be completed within a one-hundred-sixty hour time period with a minimum of operation personnel interaction. Personnel interface with specific subsystems is through the use of one of fifteen consoles, each of which can command and monitor particular subsystem tests and support fully operational integrated system tests.

Spaceflight Tracking and Data Network Support

A second major area of ground support systems is the telemetry and tracking support. The Space Transportation System of the 1980s will use the Spaceflight Tracking and Data Network (STDN) system to provide command, telemetry, and tracking support.

The STDN consists of two principal elements--the ground-based network (GSTDN), consisting of five orbital support and two launch support sites (Eastern Test Range and Western Test Range), and the Tracking and Data Relay Satellite System (TDRSS) which consists of three satellites, two of which will be operational with the third satellite serving in a backup capacity. This section presents a brief description of the ground-based network mission support capabilities of the STDN, and the TDRSS.

The configuration of the STDN, as described above, is shown in Figure VI-3 (Godfrey, Stelter, 1975). The STDN will have th capability to provide at least 85% coverage to all mission spacecraft above 200 kilometers orbital altitude and complete coverage to those mission spacecraft above 1200 kilometers orbital altitude. With the ground-based network providing support to spacecraft with orbital altitudes greater than approximately 5,000 kilometers, the TDRSS will provide support to those spacecraft with lower orbital altitudes.

The STDN is presently evolving from the totally ground-based network used today, shown in Figure VI-4 (Godfrey, Stelter, 1975) to the combined configuration of the ground-based/TDRSS network

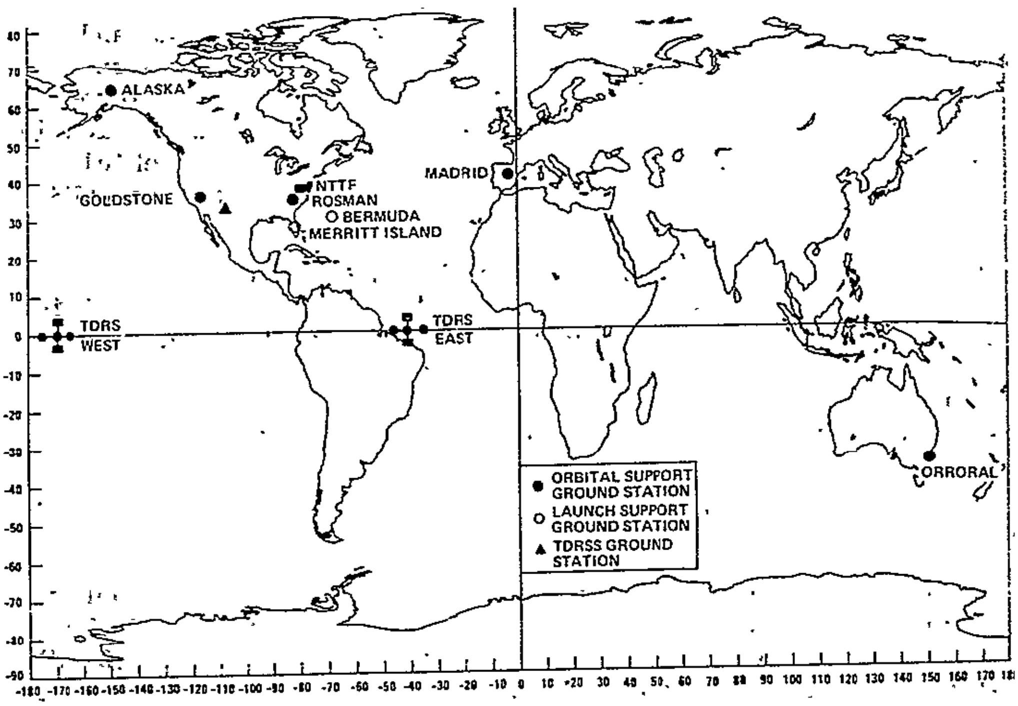


Figure VI-3 1980s STDN Configuration

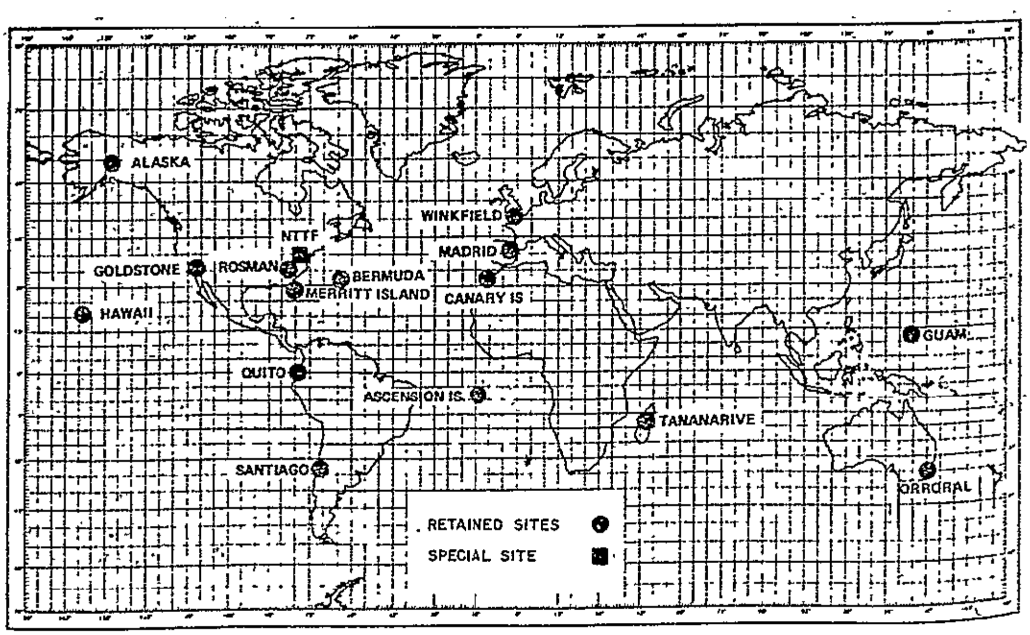


Figure VI-4 Current Spaceflight Tracking and Data Network

The basic ground-based elements of STDN will achieve the desired configuration by July, 1980. Modifications required to support specific user requirements will continue to be implemented as needed. January, 1980, is the estimated operational date for the TDRSS elements (Godfrey, Stelter, 1975).

Communication between the mission support sites and control centers of STDN is provided by the NASA communications system (NASCOM). This system distributes site operational instructions and mission spacecraft command data to the sites and site status information, mission spacecraft telemetry and tracking data from the support sites to the appropriate destination. As the STDN evolves from its current configuration, as shown in Figure VI-4, "the NASCOM system will change from its current narrow band transmission system, which is capable of transmitting only a part of the data received on site, to a wide band system which can transmit, in real time, most of the data received by each ground-based site and all of the data received by the TDRSS" (Godfrey, Stelter, 1975). A minimum of one 56 Kb/second circuit will be provided between each site and Goddard Space Flight Center, but when a spacecraft's data rate exceeds the NASCOM circuit capability, the data will be recorded and as the capacity becomes available, will be transmitted.

As previously mentioned, the basic ground-based network elements of the STDN will remain much the same as currently exists. Not all existing ground sites will be retained; however, the on-site systems of those sites which will be utilized will be upgraded. The operational philosophy will not change from that of the current STDN except for the handling of mission spacecraft data. POCCNET, Payload Operations Control Center Network, is being developed for this purpose.

POCCNET is Goddard Spaceflight Center's approach to organizing operations control centers in the 1980s to respond to the system requirements, interfaces and opportunities which will be generated by the Space Shuttle and STDN. The POCC is the focal point for payload in-orbit operations. It serves as the master control point for coordinating and controlling the activities of the ground support complex for the payload, including ground stations, communications links, and support computers. The POCC also serves as the interface between the experimenter and his experiment instruments (des Jardins, Hahn, 1976).

The primary mission of the POCC is to ensure accomplishment of payload objectives while maintaining the health and safety of the payload. Monitoring the spacecraft's status, coordinating experiment evaluation and operations planning, processing control commands and directing data acquisition operations ensure that the primary mission is fulfilled.

Two basic POCC functions, command and telemetry processing, are shown in Figure VI-5 (des Jardin, Hahn, 1978).

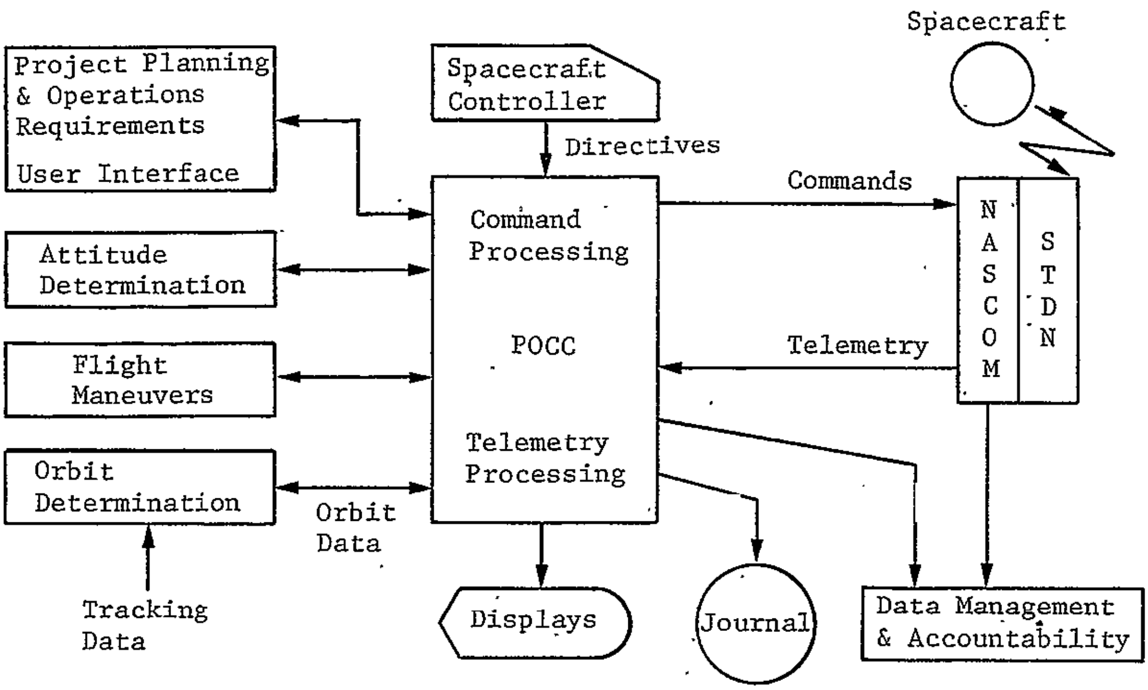


Figure VI-5 Typical Payload Operations Computing System Functions

Operations of the payload and its experiments are controlled from the ground by commands sent to the payload through the STDN. Telemetry processing, performed in real time, allows for monitoring and evaluation of the payload operations.

The overall POCCNET system approach is to adapt the use of standards compatible with the widest possible segment of national and international distributed systems which will allow for uniformity within the total system. POCCNET will seek to impose only enough uniformity to allow resource sharing and reuse of standard subsystems. This concept will allow POCCNET to be compatible by design with future GSFC-wide and NASA-wide computing systems (des Jardins, Hahn, 1976).

Since POCCNET must ensure compatibility of resources within the entire network and coordinate the conventions by which work is performed, it is designed as a distributed computing network. This system will be implemented over a period of several years, gradually acquiring operational control of missions.

A typical STDN ground-site configuration for the mid 1980s is shown in Figure VI-6 (Godfrey, Stelter, 1975). The major systems

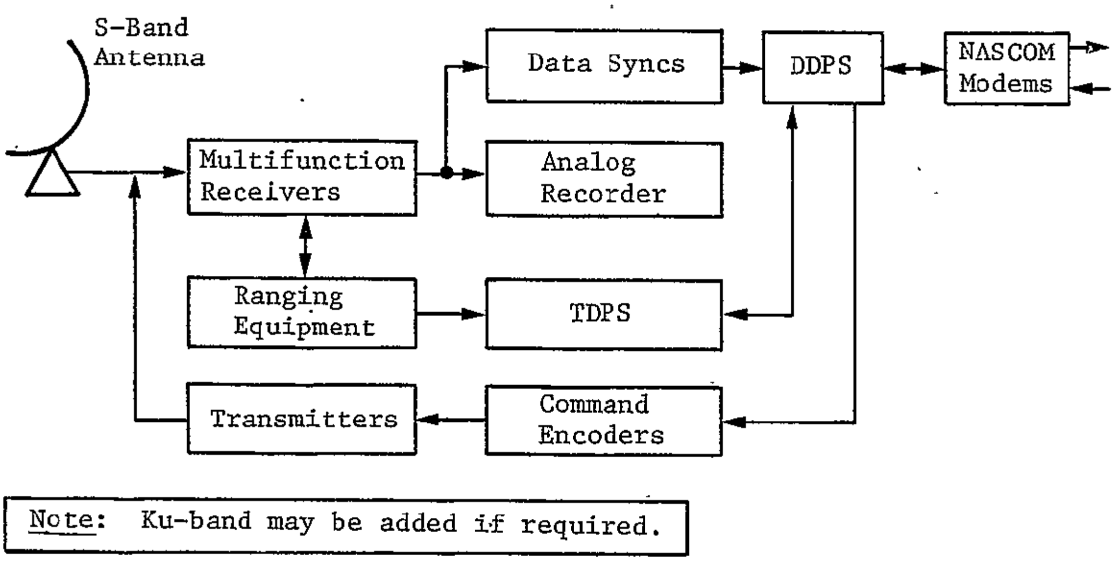


Figure VI-6 Typical STDN Ground Site Configuration for Mid 1980s

systems contained within this site, as presented in Godfrey's and Stelter's paper, are summarized below:

1. Antenna Systems

   Each orbital site will be equipped with a 26-meter (diameter) antenna system. Some sites will also use a 9-meter antenna system including all launch support only sites.

2. Operating Frequencies

   These sites will be capable of providing support at S-band (2025 to 2120 MHz command link and 2200- to 2300-MHz telemetry link). Ku-band (13.4 to 14.05 GHz command and 14.6 to 15.25 GHz telemetry) capability will be provided when required.

3. Receiving System

   These sites will be equipped with the STDN multifunction receivers (MFR), which have a maximum receive bandwidth of 20 MHz at S-band (2200 to 2300 MHz).

4. Transmitter System

   These sites will be equipped with an SCE transmitter system which has a maximum bandwidth of 20 MHz. The maximum transmitter power will be 20 kW.

5. Range and Range Rate System - The ranging system will be the standard Goddard Range and Range Rate (GRARR) system, which is a coherent system utilizing ranging sidetones and a carrier doppler range rate system.

6. Telemetry System

   The telemetry system at each site will be configured around the Digital Data Processing System (DDPS) and will be capable of supporting any data signal compatible with the GSFC aerospace data standards.

7. Command System

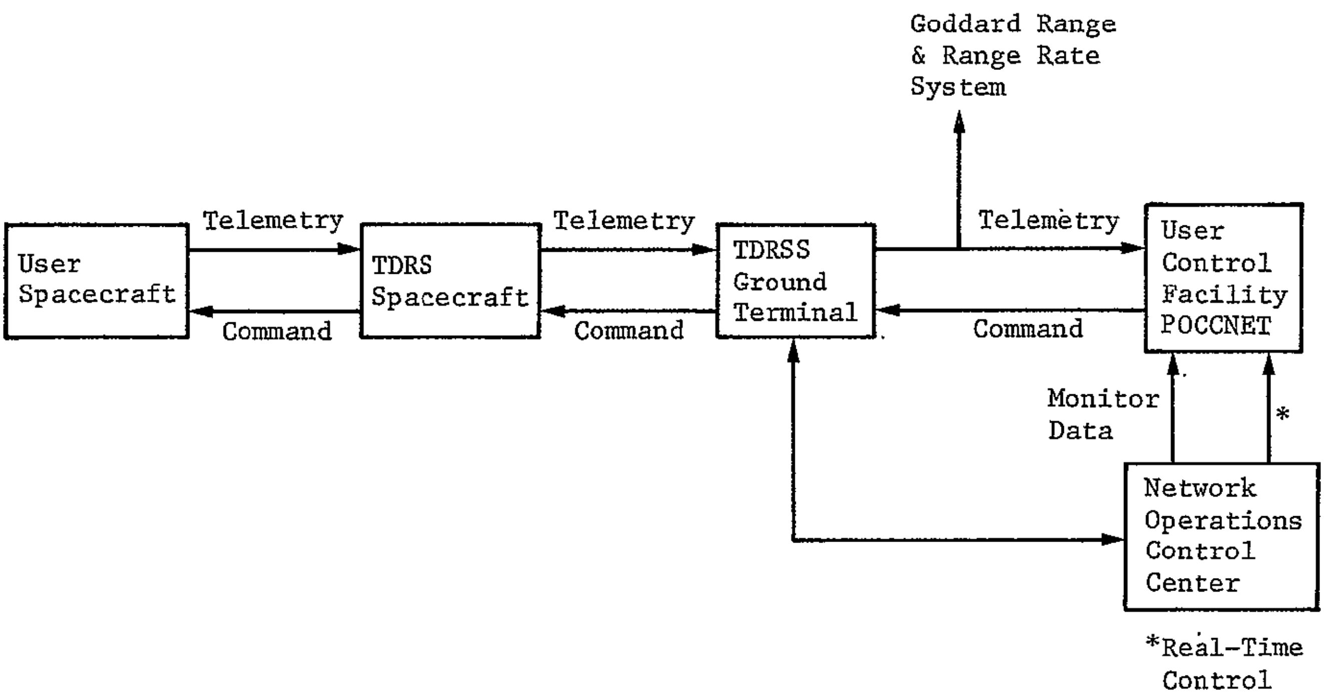
   This system will be configured around the spacecraft command encoders and be capable of supporting any signal consistent with the GSFC aerospace data standards.

The purpose of the TDRSS, as briefly mentioned before, is to provide telecommunications services which relay communications signals between earth-orbiting spacecraft below 5,000 kilometers and the control and/or data processing facilities. A real-time, bent-pipe concept is utilized in the operation of the TDRSS telecommunications services. Two geosynchronous relay satellites, approximately 130 degrees apart in longitude, a ground terminal and two spare satellites, one in orbit, the other ready for a rapid replacement launch, make up the TDRSS.

There are two types of telecommunication systems available: a multiple-access and single-access system. The multiple access communication service system, which operates at S-band, is designed to provide simultaneous real-time and dedicated return link service up to twenty spacecraft with real-time data rates up to 50 kb/sec. The single-access communications service system, which operates at both S-band and Ku-band, provides a high data rate return link to spacecraft. This system operates on a priority basis only and will not normally be available for dedicated support of a specific mission, with the exception of the Shuttle.

The major operational features of TDRSS are the availability of extended user spacecraft contact time and user spacecraft-to-user data facility telecommunications bandwidths in excess of user spacecraft data rates, thus providing real-time data transfer.

An overview of the TDRSS user operation interfaces is shown below (Godfrey, Stelter, 1975).



Conclusions

During the next twenty years, in contrast with the past, weekly space launches, on-orbit maintenance and continuous manned ground operation will become routine and what is now regarded as extremely ambitious technology will become everyday engineering practice.

The space transportation system will support numerous proposed NASA space activities with the Space Shuttle Vehicle serving as the primary launch vehicle. Through the early 1990s, it has been estimated that the Shuttle vehicle alone will fly approximately 1091 missions. For those missions currently planned in the late 1980s and beyond which have destination orbits beyond the Shuttle's operational orbit, a space transportation system in the form of an orbital transfer vehicle will be required. Many of these orbit-to-orbit vehicles will be supported by an initial launch from Shuttle.

Presently, man exercises direct control over nearly all decisions required by spacecraft, including those decisions associated with data handling and processing. To perform these multitude of tasks, extensive communications, ground equipment, and many support personnel are required. Each of these items in themselves become extremely expensive when extended missions are considered or for extensive data collection and processing. However, with the anticipated use of one class of launch vehicle, the Space Shuttle, the development of the TDRSS and related payload operations control center, POCCNET, and the coming generation of standardized spacecraft, it will become quite feasible to provide automated ground support systems.

During the early years of the space business, differences in philosophy existed as how to best operate a spacecraft and launch vehicle. As a result, many new operations control centers were developed as new series of spacecraft or launch vehicles were developed. Now, with the advent of a new era in space activities, comes the time to reorganize the development philosophy of ground operations. The recent development of the Launch Processing System, the Satellite and Tracking Data Network, and the concept for the Payload Operations Control Center Network, three areas of the total ground support system discussed herein, will provide NASA with a much more capable and effective ground support system. Though these support systems will probably not be totally automated, the continued development of more sophisticated semi-autonomous support systems, in conjunction with the increased autonomy of onboard spacecraft operations; e.g., autonomous navigation, attitude control, onboard data evaluation and reduction will result in a significant reduction of manhours required to to perform these tasks. With the manpower costs rising tremendously, this reduction will yield a considerable cost savings to NASA as well as helping NASA to achieve its goal of increasing capability by a 1000-fold in the near future.

The overall objective of the new, autonomous support systems philosophy is to provide an evolutionary hardware/software computing system which can reliably operate the ground support systems at a significantly lower cost than current systems. The autonomous system development should be driven by a number of factors which include the following:

- The nature of computing systems is changing. The trend is to provide more computing power with hardware rather than software. Software costs, relative to hardware costs, continue to climb; the software/hardware cost ratio has been projected at 10:1 by 1985 (Myers, 1978). Software productivity is improving only by a factor of two every three years compared with hardware improvements of an order-of-magnitude every three years.

- Cost is a major consideration in systems development due to NASA budgetary restrictions. It is well known that direct personnel costs of implementing and operating a system for an extended period of time far exceeds the requirement costs. Large cost savings can be achieved by higher efficiency software and a concommitant decrease in manpower.

- The use of standardized equipment throughout the ground support systems will provide a specific set of requirements and interfaces. The user can then properly organize to take advantage of these specific requirements. Though the user will encounter an initial cost, if the standardized systems have been properly designed and implemented, they can be used repeatedly with only minor modifications rather than implementing an entirely new and expensive system.

- As ground support systems become more standardized, they will need to be more flexible and responsible. With a majority of the NASA missions lasting from a few months to five years, ground equipment will have to interface with a number of different mission spacecraft. Spacecraft will continue to become more complex as NASA's mission objectives are developed. Flexibility is essential to allow software programs to be modified due to changes in requirements from one mission to the next and to allow for easy implementation of new technology developments which are continually evolving for both hardware and software. Distributed computing systems give this needed flexibility.

The three support systems discussed within have incorporated these factors into their operating philosophy enabling their continued growth into a more efficient, more autonomous support system. In order to ensure the continued autonomous development of these systems and all others associated with the total ground support system, NASA needs to establish specific guidelines to direct future research and development in this area. These guidelines should include the general categories:

- What functions of the ground support system should be totally automated, semi-automated or left as a manned responsibility? Defining these functions will direct the development of hardware/software equipment.
- Establish hardware/software requirements and from these develop the necessary equipment to perform the tasks.
- Develop the total support system concept which will incorporate the necessary hardware/software and which will perform all required tasks.

Establishment of these guidelines will help to direct the research of private industries and, in turn, will result in the increased efficiency and capability of support systems which NASA is looking for.

The Defense Department has done just this--sponsoring development of advanced technology for military application. In mid-November, 1978, a program will be initiated with the purpose of advancing semiconductor microcircuit technology for military applications with the emphasis on achieving significant increases in fundamental complexity and device operating speed (Klass, 1978). Over the next six years, two-hundred million dollars will be spent on the development of the Department's near-term goals with primary emphasis on significantly increasing functional complexity and device operating speed. The Defense Advanced Research Projects Agency, DARPA, will also fund a more ambitious far-term goals program to take a fresh look at basic digital computer architecture. In discussing the Defense Department's development of these programs, Klass points out that the Pentagon was, in the past, content simply to ride the coattails of the commercial market. Viewed in retrospect, says Klass, this decision is seen as a mistake because it led the military into a dependence on a wide variety of commercial microprocessors that had not demonstrated their ability to meet the rugged military environment.

At the present time, NASA also seems to be riding the coattails of the commercial market and rather than spend money on what is available, NASA should decide what it wants and needs and then direct the necessary research and development. By doing this, NASA will most likely spend less overall on the implementation of a program and at the same time, have equipment meeting its unique requirements.

Recommendations

NASA's future mission plans, which include increased planetary exploration, intensified earth observation studies, the development and utility of permanent space stations, and the industrialization of space, will only become a reality if NASA's mission costs are substantially reduced. Estimates indicate that NASA's yearly costs in the 1980s will be 3.57 billion dollars but if automation technology is used extensively, this cost can be reduced by approximately 1.5 billion. Automation of ground support systems will contribute significantly to this cost savings as indicated in Table VI-2 since approximately one-third of the over all estimated cost savings of 1.5 billion is related to ground support systems, it is strongly recommended that NASA initiate advanced automation studies in this area.

As discussed previously, the LPS and STDN represent the state-of-the-art in ground support systems. Each system could be considered semi-autonomous at this time. However, as NASA's missions increase in number and complexity, these systems and all other systems of the total ground support operation will need to become more efficient and flexible. The use of hardware and software advances will provide the desired increased efficiency and capability.

Development of automated software tools to be used extensively during the design and test phase of a program could significantly reduce the life cycle costs of a program by reducing the number of manhours required during the design and test phase and by correct system development, which is enhanced by detailed specifications which define constraints, record decisions, and evaluate design made during the early phases of the software life cycle.

Various theories and systems are presently being developed by commercial contractors to support the development and evaluation of specifications. To date, there are many design aids available that apply to various phases of the software development. A few are listed below:

1. SSL - Software Specification Language
   (High level design oriented)
2. PSL/PSA - Problem Solving Language/Problem Solving Analyzer
   (Requirements and design)
3. PDL - Program Design Language
   (detailed design)
4. RatFor - Rational Fortran
   (structured Fortran)

In addition, Martin Marietta's research program is producing a very high level requirements identification and analysis system for total life cycle requirements, traceability and management.

The software design aids are characterized by a data base which is produced from a specification language. The languages have their own unique vocabulary and syntax which language processors use to automatically develop the data base. Further analytic tools are then applied to produce design feedback information.

Software productivity is improving slowly, by roughly a factor of two every three years, and its cost is steadily increasing. Implementation of these modern software tools within NASA programs will help to reduce cost and improve efficiency and their use, therefore, is strongly recommended.

Development of a standardized family of higher order languages is an important technology goal. In a recent survey, Ambler (1978) found that most system languages investigated do not have the capability of linking to other languages by any other method than cleverly coded assembly language routines. Uniformity within a ground support system will be heavily dependent upon the use of a standardized family of languages; system uniformity will be difficult to achieve without the use of the standardized languages.

With software improvements occuring slowly and its cost increasing steadily, a trend is developing to provide more computing power with hardware rather than software. Since the early 1970s when the era of microprocessors began, their capabilities have been constantly increasing and their cost decreasing. Today there are many semiconductor vendors and a vast number of microprocessors on the market. If NASA is to fully realize the benefits to be gained from microprocessor technology, then guidelines for selecting microprocessors for NASA use should be established.

The microprocessors currently on the market are packaged in one of three forms: single chip microcomputers, single chip microprocessors, and bit slice processors. Single chip microcomputers, which contain the processor, program and data memories, and input/output data parts, are at the low end of the performance spectrum and are normally used in dedicated low-performance applications. In NASA programs, such devices could be imbedded in instruments for control and data management purposes.

Single chip microprocessors, the most common microprocessor device available, contain only the processor with additional chips required for memory and input/output ports. This design is hardware flexible while the single chip microcomputer is very expansion limited unless complex multiplexing logic is added externally. The majority of these single-chip microcomputers and microprocessors use an 8-bit wide data path. Early next year, the Intel and Zilog Corporations will have 16-bit single-chip microprocessors available, and NASA should investigate incorporating these chips into ground support hardware.

Bipolar bit slice processors are at the top end of the microprocessor performance spectrum. Such systems typically perform five to ten times faster than do equivalent MOS units. Bit slice processors also have much greater hardware and software flexibility. The designer has the ability to define the processor instruction set as well as define an architecture which achieves special capabilities or performs a specific application with the highest level of efficiency. A third advantage of bipolar bit slice is expandability; word lengths can be expanded by cascading units.

The previous paragraphs describe the present status of commercial microprocessors and as previously stated, NASA should define the microprocessor products it requires if maximum benefits from this technology are to be realized. It is unlikely the semiconductor vendors will produce the devices which NASA needs without direction or financial encouragement from NASA.

A second step NASA should take to derive maximum benefits from LSI technology is specification of fabrication techniques. Silicon on Sapphire (SOS), a conventional technology, appears to have a very good speed-power ratio. This ratio is necessary to handle computing rates which are ever increasing as high performance on-board computing systems are developed and used. Use of SOS will avoid a large investment in a new fabrication technology within the next few years but unless NASA takes the initiative, it is unlikely that SOS will be used in the microprocessor products NASA requires due to the relatively high cost involved.

A unique difficulty associated with LSI devices is testing. Efforts to date in this area have shown that the ability to test LSI devices is very limited requiring quite a bit of time. As the functional complexity of chips increase, the difficulty of testing will also increase; therefore, test strategies need to be developed now. One of the near-term goals of the Defense Department's semiconductor microcircuit technology program, discussed early in this report, is the inclusion of built-in self-test provisions on a chip and looking at functional partitioning that can facilitate self-test (Klass, 1978). It is recommended that NASA also begin research in this area.

Increased functional complexity and capability of LSI devices will necessitate faster, more dense memory. Semiconductor memory is of three forms: RAM, random access memory or read-write memory; ROM, read only memories; and read-mostly memory. RAM is the most flexible and fast but is a volatile device. ROM is nonvolatile, but it is not possible to change memory contents once they have been generated. Read-mostly memory exists in two forms: ultraviolet erasable, electrically programmable ROMS (EPROMs); and electrically alterable ROMs (EAROMs).

General Electric is currently doing research, funded by DARPA, aimed at achieving an extremely large-capacity archival-type of memory with a capacity of 10^{15} bits by using an electron beam both to write and read out data stored in memory.

In current mass memory techology development, serial-type memories are under intensive study. Charge coupled devices and bubble memories appear to be some distance away from taking over mass memory technology. In these areas, much effort is still being devoted to optimizing system-device interaction.

It appears that a memory hierarchy may be evolving for future processors. The fastest devices will have low densities and be fabricated in bipolar or I^2L. Access times will be 10 to 100 ns. Main memory will have access times of 400 to 1000 ns and use SOS, core or CCD technology. Cost will be approximately an order of magnitude cheaper than the fastest devices. Auxiliary memory will have the greatest package densities but speeds in the 10 μs to 500 μs range. CCE, bubble or beam access technology could result in costs half that of main memory.

The problem areas to be settled include the need for new system components, the proliferation of device organizations and technologies and the need for common usage components. Both NASA and Air Force are already investigating the use of spaceborne bubble memory and CCD systems. It would appear appropriate that investigations of these devices should continue; however, similar studies must be conducted on other memory technologies especially EAROM if appropriate benefits are to be obtained.

The ability to quickly produce LSI circuits is being demonstrated by a number of semiconductor vendors. Custom masked ROM are now available within two to three weeks. When properly packaged, LSI support devices can result in considerable savings in both hardware and software development efforts. For example, it is now very uncommon to implement a serial interface using either SSI/MSI devices or techniques. Universal synchronous and asynchronous receiver/transmitters are available in LSI and fulfill 90% of computer and peripheral serial interface.

LSI support devices can be of special importance to NASA. First of all, they would help reduce component count and would, therefore, simplify design efforts. Secondly, they could be used to improve computer performance by permitting parallel operations where appropriate. Third, they could be used to conveniently implement some of the more standard functions such as telemetry formatting, deformatting, and control of multiplexed busses. The important aspect to support circuits is their interfacing architecture. A poor interface to an LSI support device may necessitate a large SSI/MSI interface circuit which degraded the desirability of the support device.

Fiber optics is another technology area NASA should investigate thoroughly. Fiber optics systems, which transmit information by means of encoded light beams traveling through thin glass fibers, have significant advantages over all electronic systems in that they are free from electromagnetic interference and pulse effects, they provide a high degree of immunity from intelligence probing and jamming, they are lighter weight, and perhaps most important, provide a substantially greater data-handling capability (Elson, 1978). As NASA missions become more complex, gathering enormous amounts of data and relaying this data to ground stations, high speed data transmission will be essential between support stations and within the ground station equipment. Fiber optics technology will be able to provide the necessary high speed data transmission.

The Boeing Company is currently involved in a wide-ranging series of development programs exploring fiber optics technology.

The programs investigations include basic measuring instruments (measuring liquid level, liquid flow rate, linear displacement, strain, pressure and temperature), avionics data busses operating at 10-megabit/second and kilometer-scale data links designed for large intraplant and interplant computer networks (Elson, 1978). By incorporating fiber optic measuring instruments, data buses, and data links into ground support systems, primarily the LPS, the system's efficiency and checkout capability would increase.

It is recommended that NASA invest in fiber optics development for checkout applications and high speed data transmission which will be necessary to handle the reams of data and imagery data which will result from the missions NASA plans to fly.

As NASA develops guidelines for uniformity within ground support systems, it will also need to develop a flexible and responsive system. With a majority of the NASA missions lasting from a few months to five years, ground support equipment will be forced to interface with a number of different mission spacecraft. Spacecraft will continue to become more complex as NASA's mission objectives are developed. Flexibility will be essential to allow software programs to be updated due to changes in requirements from one mission to the next and to allow for easy implementation of new hardware and software development which is constantly evolving. Distributed computing systems give this needed flexibility.

The concept of distributed processing lends itself easily to ground support operations; many of the tasks are highly specialized, involving tedious, continuous step-by-step control of many remote sensors and stations. The distributed processing system is designed such that multiple processors independently work on a specific task. Additional processors can be added and tailored to fit any type of processing situation, thus offering the best opportunity for obtaining maximum computing power.

Some disadvantages do exist in a fully distributed system. A distributed network is a complex technique in computer science. Since the system reaches its maximum performance through asynchronous parallel execution and processor count expansion, significant system management must be provided in both hardware and software. More research and experience is needed to develop a system management concept capable of handling a complex ground support system; therefore, it is recommended that NASA invest in this research area. Once the problem of system management is resolved, the distributed processing system for ground support operations will be able to expand, becoming larger and more efficient and able to automate more and more of the ground operations support tasks.

REFERENCES

Ambler, A. L., "Directions in System Language Design", Digest of Papers, COMPCON, Spring, 1978.

Bryne, F., Doolittle, G., Hockenberger, R., "Launch Processing System", IBM Journal of Research and Development, Volume 20, January, 1976.

CCMS Sets, MCR-76-56, Data Item Number 037, "Systems Operation and Maintenance Manual - Checkout, Control, and Monitor Subsystem", Volume I, Systems Overview, Martin Marietta Aerospace, December, 1977.

des Jardins, R., Hahn, J., "A Concept for a Payload Operations Control Center Network (POCCNET)", GSFC Report No. X-510-76-250, November, 1976.

Elson, Benjamin M., "Fiber Optics Technology Investigated", Aviation Week and Space Technology, Volume 109, Number 16, 16 October 1978.

Godfrey, R., Stelter, L., "Spacecraft Tracking and Data Network Support in the Space Tug Era", Paper Number ASS 75-214, Presented at 21st Annual Meeting of the AAS, Space Shuttle Missions of 80s, Volume 32, Part 2, August, 1976.

Klass, Philip J., "Microcircuit Complexity, Speed Pushed", Aviation Week and Space Technology, Volume 109, Number 14, 2 October 1978.

McReynolds, Stephen R., "A Benefit and Role Assessment for NASA", 14 April 1978.

Myers, D.K., "Ionizing Radiation Effects on Various Commercial NMOS Microprocessors", IEEE Transactions, Nuclear Science, NS.24: 2169-2171, 1977.

PACE, MMC Report Number 8080F040005, Design Criteria - Programmable Aerospace Control Equipment, Revision A, Martin Marietta Aerospace, July, 1976.

VII EXPERIMENT POINTING MOUNTS

Introduction

A review of the experiment requirements for the Space Shuttle payloads reveals the need for pointing systems to orient and stabilize experiments to their desired orientation in space while yet allowing the Space Shuttle orbiter with sufficient attitude freedom to perform normal astronaut and vehicle housekeeping functions. From selected representative payloads, the absolute pointing accuracy requirements range from 5 arc seconds to 2 degrees and pointing stabilities from 0.2 arc seconds to a few degrees.

The absolute pointing accuracy is not a parameter that enters into the selection of pointing mounts since this parameter is dependent totally on the sensing instruments (star sensors, gyros, etc.) used which can be mounted on any pointing mount. The pointing stability parameter plays a large part in the choice of pointing mount.

There have been a large number of pointing mount concepts that have been proposed to provide pointing stability in the presence of Space Shuttle motion. Five of the more seriously considered systems are shown in Table VII-1.

Only one pointing mount (SIPS) falls into the category of a c.g. mount; i.e., the center of rotation of the payload passes through the c.g. of the payload mass.

This arrangement has the advantage of experiencing minimal disturbance from Space Shuttle motion but is more difficult to operate since the center of rotation must be linearly translated far enough that the aft end of the payload clears the Space Shuttle cargo bay. Furthermore, care must be exercised to properly balance the payload or the advantage of c.g. mounting is last. Also, will ball-bearing pivots, this system's limiting stability is about one arc second.

The remainder of the pointing systems incorporate a payload end mount configuration whereby the center of rotation of the gimbals is at or near the aft end of the payload. This configuration has the advantage of ease of operation and payloads stowage. It has the disadvantage of induced disturbances from other Space Shuttle rotations and translations acting through the displaced payload c.g. For the ball-bearing types of systems (IPS, MPM, and ASPS-AGS), the limiting angular stability is about 1 arc second.

Table VII-1 Summary of Pointing Mount Systems

Pointing Mount System	Pointing* Stability Limit	Type of Payload Interface	Type of Bearing	Weight
Instrument Pointing System (IPS)	1 arc sec	End Mount	Ball	750 Kg
Small Instrument Pointing System (SIPS)	1 arc sec	c.g. mount	Ball	
Miniature Pointing Mount (MPM)	1 arc sec	End Mount	Ball	
Anular Suspension Pointing System (ASPS) Coarse System AGS	1 arc sec	End Mount	Ball	
ASPS Magnetic AVS	0.001 arc sec	End Mount	Magnetic Suspension	
Gimbalflex	0.001 arc sec	End Mount	Flex Pivots	

*Not including control system sensor noise

Pointing stabilities below one arc second require non-conventional pivot designs such as magnetic suspension (ASPS-AVS) or the flex points (Gimbalflex). These types of suspensions allow both rotational and linear degrees of freedom which allow them to both act as motion isolators as well as low noise pivots. Thus, even though the payloads are end mounted, these pointing mounts can filter the Space Shuttle motion to reduce the resulting disturbance from the offset payload c.g. These types of suspension systems, however, do not have wide field of view or coverage and generally need a coarse pointing system to provide this wider range.

Conclusions and Recommendations

The pointing requirements of the known payloads indicate pointing stabilities of 0.2 arc second are necessary. Furthermore, the trend of scientific investigation has been toward more stringent requirements. Stabilities on the order of 0.01 arc second appear to be coming feasible. These trends suggest that several experiment pointing mounts may be required to accommodate all of the requirements.

Therefore, for pointing stabilities less severe than one arc second, the IPS system can be used for virtually all payload sizes up to 3000 Kg. While it can be used for small payloads less than 200 Kg, its weight (750 Kg) is a heavy penalty to pay. Thus, a smaller ball-bearing mount would be advantageous such as the ASPS-AGS (coarse mount). Thus, the recommended approach is to develop both the IPS and ASPS-AGS.

For stabilities better than one arc second, either the ASPS-AVS or the Gimbalflex vernier system could be used in conjunction with either the IPS or ASPS-AGS. The Gimbalflex is the lower risk of the two systems since it uses proven hardware. It further can provide for more motion, both angularly and linearly, to provide greater capability in motion isolation. The ASPS-AVS has the potential of better pointing stability about the less demanding roll and about the line of sight.

Hence, the recommended approach would be to develop the Gimbalflex for the near-term fine pointing mount and the ASPS-AVS as a more accurate system for later, more stringent payloads and allow for a more lenient schedule for its development.

VIII RENDEZVOUS, STATIONKEEPING, AND DOCKING

Introduction

It has been established that remote rendezvous and docking, with man in real time control, will not always be feasible. This is due to the relative orbital positions, round-trip delay times associated with the command signals and accuracy constraints for some missions. Also, man-in-the-loop is more likely to misjudge the ideal rates of closure of two vehicles than is a pre-programmed, automated system. Therefore, automation of the process of convergence and docking is imperative to the attainment of long-range mission goals. The anticipated requirements and applications of automatic rendezvous technology are substantial and include the following type missions and programs.

- Shuttle upper stage retrieval, inspection and refurbishment missions
- Comet rendezvous and docking
- Inspection of vehicles with either a maneuverable TV or a manned maneuvering unit
- Asteroid rendezvous and docking
- Mars sample return
- Unmanned lunar missions
- Large Space Structure assembly and repair

Before proceeding with the discussion of the basic requirements of an automated system, the general problem of in-space rendezvous will be discussed and the applicable terminology defined.

The process of <u>rendezvous</u> encompasses the insertion of two or more objects in space, the convergence of these objects, and possibly but not necessarily the mechanical engagement; i.e., <u>docking</u> of their hulls. In certain cases several of these operations may not be performed, such as when one of the vehicles has been in orbit for some time or when the mission is that of observation, in which case, docking would not occur. In the latter example a formation flight; i.e., a <u>stationkeeping</u> operation might be used instead.

It is, therefore, obvious that at least two vehicles participate in the rendezvous operation. During this convergence process, one of the vehicles is "active" and is therefore a maneuvering space vehicle or <u>interceptor</u>. The non-maneuvering or "passive" vehicle is called the <u>target</u> vehicle. Furthermore, the <u>target</u> vehicles are subdivided into "friendly," or known if the essential information about their equipment is available, and "unfriendly," or unknown, if such information is completely or partially lacking. "Friendly" targets are further subdivided into cooperative and noncooperating vehicles and are generally equipped with various devices which facilitate or enable rendezvous and docking respectively, such as radio transponders, retroreflectors, lights and possibly docking ports. Noncooperating vehicles are generally not equipped with rendezvous aids or may be cooperating vehicles by design, but for some reason not functional. Examples of the latter might be an out of fuel condition or damaged transponder. The target vehicle can be "active" or "passive" as a function of whether it contains an active transponder or not. The interceptors can be further subdivided into those with a retrieval capability and those which are expendable; i.e., they do not return to the parent vehicle or earth. In addition to this classification, the interceptor and target vehicles can be subdivided into manned and unmanned.

A typical mission-dependent sequence of events for a rendezvous and docking operation could be as follows:

1) launch and injection of target
2) launch and injection of interceptor
3) orbital and/or interplanetary navigation of each vehicle
4) search and acquisition
5) long-range rendezvous 1000 mi.
6) short-range rendezvous < 10 mi.
7) close-range rendezvous 100 ft.
8) stationkeeping
9) docking

A earth rendezvous mission sequence is shown in Figure VIII-1.

This study is concerned with the automatic rendezvous, stationkeeping and docking sensor technology which would be used during the latter six phases. The technology could be used with manned vehicles, with overriding provisions, but the discussions will assume the operations are autonomous and automatic with only overall mission control by man.

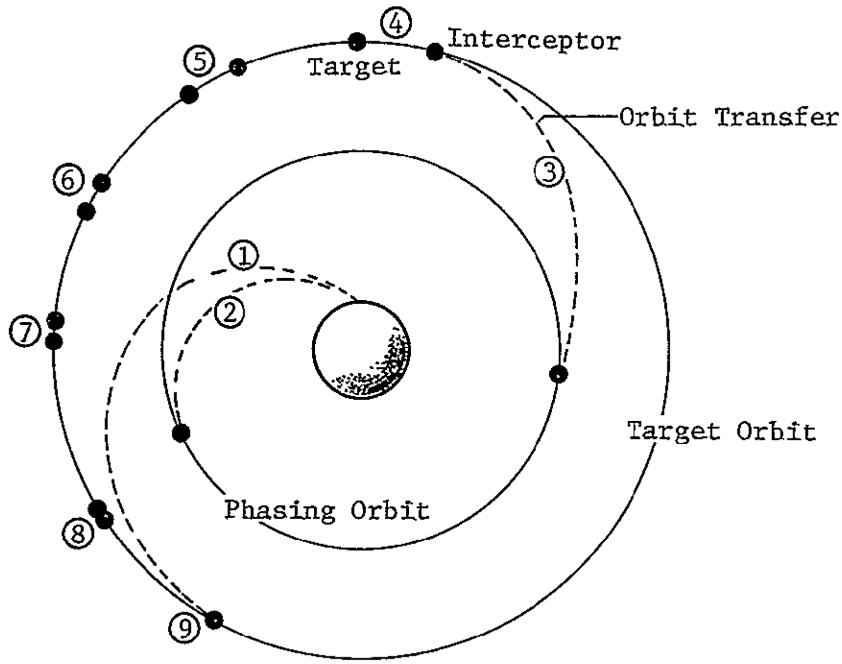


Figure VIII-1 Typical Rendezvous and Docking Mission

For physical docking to take place, the axes of the two spacecraft involved must be made coincident. In order for this to occur, the following variables must be determined and controlled (see Figure VIII-2).

1) azimuth of the target relative to the interceptor
2) elevation of the target relative to the interceptor
3) range of the target relative to the interceptor
4) pitch of the interceptor relative to the target
5) roll of theinterceptor relative to the target
6) yaw of the interceptor relative to the target
7) rates of the above quantities

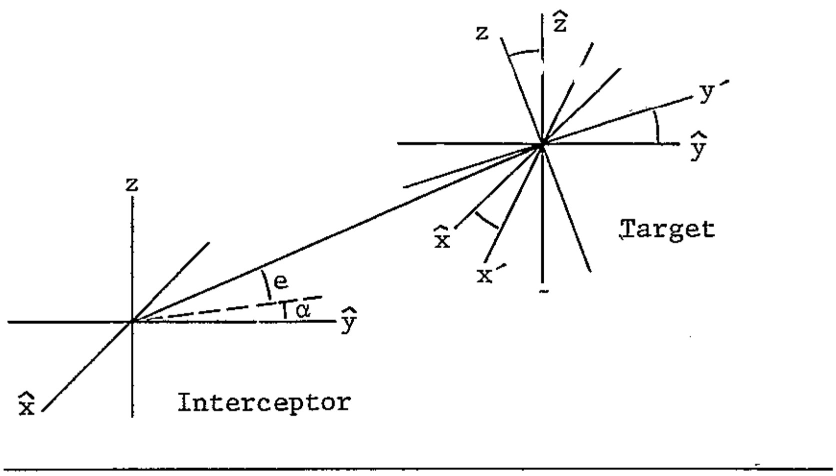


Figure VIII-2 Target Misalignment

In previous rendezvous missions, the first three requirements were performed by the sensor, however, the relative attitudes of the vehicles were eyeballed by astronauts. In an autonomous system, this function must be performed by the onboard system which will undoubtedly need some sort of processor. This system can be implemented in one of two ways. Either the interceptor rendezvous system can process data from one sensor, as will be shown later, or each of the spacecraft can contain an Inertial Reference Unit (IRU) and the target vehicle telemeters its attitude information to the interceptor. Although both concepts would be adequate for performing the task, the idea of performing all calculations onboard the interceptor has several advantages.

1) Weight of the target vehicle could be reduced due to the elimination of the transmitter and IRU

2) Lifetime of the target could be increased; i.e., vulnerability to the failure of the IRU and transmitter

3) Complexity of target vehicle would be reduced (timing of T/M, etc.)

4) Reduced development cost since a high accuracy IRU is not required and the communication link between vehicles is eliminated.

5) Applicability of system is extended in that disabled vehicles can be considered feasible targets.

Before proceeding, an important consideration relating to several planned missions must be discussed. That is the ultimate need for autonomous operation. The subsequent sections in this chapter will show that although the technology is available for accomplishing automatic real-time rendezvous and docking, a hardware system that can perform this operation has not been flown by the United States. Conversely, Russia has successfully accomplished automatic rendezvous and docking when in 1967 Kosmos-186 and Kosmos-188 were joined in space. A brief account of the Russian rendezvous experiment follows:

Immediately after insertion of the second satellite into orbit the appropriate rendezvous equipment on both satellites was switched on, mutual search was accomplished, and the active satellite carried out the operations of approach, rendezvous, and docking, while the passive satellite oriented itself with its docking unit toward the active one. Rendezvous was actually initiated when the distance between the two satellites was on the order of 24 Km and their relative velocity approached 25 m/s. Russia has noted that besides having automated all of these processes, the actual rendezvous and docking experiment occurred outside the region of visibility from the territory of the Soviet Union so that the scientists controlling the flight were unable to observe the rendezvous or even intervene in its course. The entire course of the process of automatic docking was studied later in telemetry data which was first directed to memory devices installed on each satellite and transmitted to ground control points after the satellites appeared over the territory of the Soviet Union. The docking phase was also later televised to Earth. The process of rendezvous was concluded when the distance between the satellites was on the order of 300 meters. At this point docking began. The moment at which the docking units of the satellites touched, their speed of approach was within the limits of 0.1 m/s to 0.5 m/s. The satellites then flew in the docked configuration for 3.5 hours. At the end of that time, the satellites were separated and the transition to independent flight began. A block diagram of the rendezvous, guidance and control system of the active satellite is shown in Figure VIII-3.

This chapter briefly summarizes various RF radar, optical radar, and video processing systems which could be implemented in an autonomous rendezvous configuration. Some of these systems such as the Apollo and Gemini rendezvous radars, are flight-proven operational systems which have seen extensive service in past years. Other systems, consisting of modified versions of these rendezvous and docking radars, have been proposed but were never tested or flown. These systems are mainly extensions of current radars to increase their maximum range or to permit the rendezvous system to operate in conjunction with non-cooperative target vehicles. Modifications have also been proposed to the NASA Unified S band tracking system to include the rendezvous and docking function. Some of these systems are mere concepts at this time, while others have been laboratory or field tested. Still others have been carried to the feasibility model stage although they have not been tested as flight hardware. VHF Range and Range Rate Systems have been considered insofar as they represent existing Apollo hardware or modifications to the Apollo ranging system.

The above airborne rendezvous and docking systems can be subdivided into eight general groups, as follows:

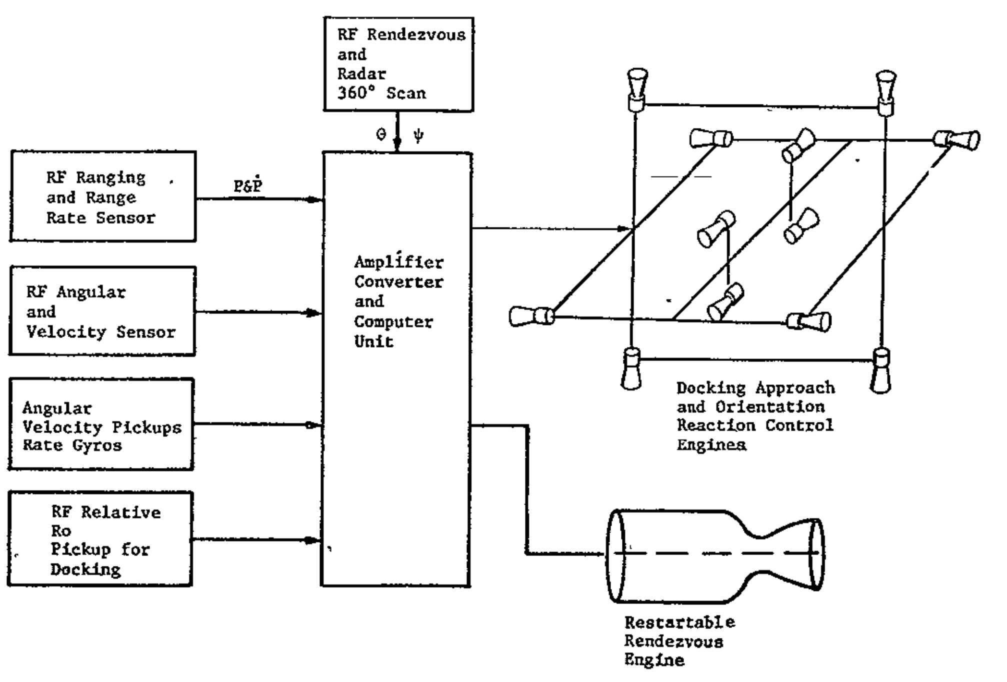


Figure VIII-3 Block Diagram of Russian Rendezvous G&C System

RF Systems

Several of the systems discussed in this section are summarized in the following groups:

Group A: X-Band Systems

1) Apollo Lunar Module Rendezvous Radar and Transponder
2) Modified Apollo Lunar Module Rendezvous Radar (non-cooperative)
3) Modified Apollo Lunar Module Rendezvous Radar and Transponder
4) Gemini Rendezvous Radar and Transponder

Group B: S-Band Systems

1) Apollo S-Band Rendezvous System - Extension of NASA Unified S-Band System (cooperative)
2) USCANS Rendezvous System - Modification of NASA Unified Communication and Navigation System (cooperative)

Group C: Ku-Band Systems

1) Modified AN/APQ-148 Rendezvous Radar (non-cooperative)
2) Modified AN/APQ-144 Rendezvous Radar (non-cooperative)
3) Motorola Missile Seeking Radar (non-cooperative)
4) Raytheon Shuttle Rendezvous Radar (non-cooperative)
5) Emerson Dual-Mode Rendezvous and Docking Radar (non-cooperative)

Group D: VHF Systems

1) Apollo VHF Ranging System (cooperative)
2) Modified Apollo VHF Range and Range Rate System (cooperative)
   - RCA Modification to Apollo VHF Ranging System
3) Modified Apollo VHF Range and Range Rate System (cooperative)
   - Motorola Modification to Apollo VHF Ranging System

Group E: C-Band Systems

1) AN/SPS-105 Tracking Radar (cooperative & non-cooperative)

2) AN/MPS-36 Tracking Radar (cooperative & non-cooperative).

3) Autonomous Navigation Technology System (cooperative)

4) One-Way Doppler Navigation System (cooperative)

5) Motorola AROD System (cooperative)

6) Tenative/GPS Navigation System (cooperative)

Optical Systems

Group F: Laser Rangers

1) ITT Scanning Laser Radar

2) Lockheed Laser Ranger

Group G: Stereo Optical Rangers

1) RCA Optical docking sensor

2) Martin Marietta Stereo Rangefinder

Group H: Advanced Concepts

1) FM Doppler Radar

2) FM/CW Harmonic Docking Radar

3) CR100/ELF III Rendezvous System - combination of CR-100 and ELF III systems

4) Advanced TRS Rendezvous System

5) Feature Identification Docking Operation

6) Optical Cetroid Tracker

7) Simulations

This chapter will briefly summarize the major systems in each of these groups and from this information, recommendations will be made on the basis of applicability to mission models. In addition, an overall technical summary is given in the attached matrices.

RF Systems Summaries

Group A: X-Band Systems

1. Apollo Lunar Module Rendezvous Radar (RCA) - The basic sensor in this group is the Apollo LM Rendezvous Radar which has been successfully employed in most Apollo flights. This sensor consists of a lightweight, solid-state X-band radar tracking system operating in conjunction with a cooperative transponder. The radar provides range, range rate, and angle data and has an acquisition time of 1.8 seconds with a 98% probability using a -122 dbm signal at a range of 400 n. miles. Range is determined by measuring the phase shift between the received and transmitted signal multitone phase modulation waveform while range rate is determined by measuring the two-way doppler frequency shift. Angle tracking is provided by the amplitude-comparison monopulse technique which yields maximum angular sensitivity and accuracy. The normal acquisition sequence for the radar and the transponder is automatic and includes the following sequential steps:

a) Radar interrogator antenna is designated in angle by the computer to point in the direction of the coherent transponder.

b) Transponder stops frequency sweep and phase locks to the received radar signal.

c) In turn, the radar interrogator receiver stops its frequency sweep and phase locks to the received transponder signal. The completion time of steps (b) and (c) is 4.5 seconds.

d) Computer transmits "autotrack enable" when antenna LOS is within 1^{o} of transponder LOS to prevent acquisition on sidelobe. The completion of steps (c) and (d) closes radar angle tracking loop and nulls the angle error.

e) Radar interrogator initiates ranging modulation and the range tracking error is nulled within a maximum of 7 seconds after completion of step (c). The coherent tracking loop is now closed.

Upon completion of the above acquisition sequence, angle, range, and range rate data are available to the computer and the Astronaut Display Panel. Angle rate is also available to the display panel.

2. Apollo Lunar Module Rendezvous Radar Modification No. 1 and No. 2 - In order to permit operation of the above rendezvous system in the "uncooperative" or skin-track mode RCA has proposed the following modifications:

a) Increased transmitter power

b) Increased antenna gain

c) Improved receiver noise figure

d) Interrupted CW mode instead of CW mode of operation

e) Improved acquisition technique to reduce acquisition time

If tracking down to a minimum range of 30 meters is required (Option 2), the following additional modifications should be made:

f) Reduced power for close-in operation

g) Modified range tracker for short-pulse operation

h) Increased antenna beam width in stationkeeping mode

In this system, the guidance computer designates the target at a range of 60 n.m. to within a 1σ error ellipsoid of 4 n.m. in diameter and 5 n.m. long. The radar scans this volume in azimuth, elevation, and range until the target is acquired. Subsequently, azimuth, elevation, range and range rate data are supplied to the computer for optimization of the approach path. The process is fully automatic. The system does not have frequency agility and thus is subject to wide variations in the target radar cross-section.

In order to provide extremely long acquisition ranges on cooperative targets, the above system can be operated with a transponder located in the target vehicle. To achieve a maximum range of 2680 km the transponder antenna gain is increased from 0 to 11 db. The resulting reduction in antenna bandwidth will reduce the angular coverage and will require pointing of the transponder antenna in the approximate direction of the interrogator radar. An addition modification is required to make the skin-tracking interrogator radar suitable for transponder operation. This concists of providing an additional local oscillator to accommodate the frequency shifted reply from the transponder.

One of the problems with these modified Apollo rendezvous radars is the large range error encountered at long ranges. This is illustrated in Figure VIII-4 which show both the bias and random errors for a non-cooperative rendezvous radar as a function of range. A similar behavior can be expected for a long-range cooperative radar.

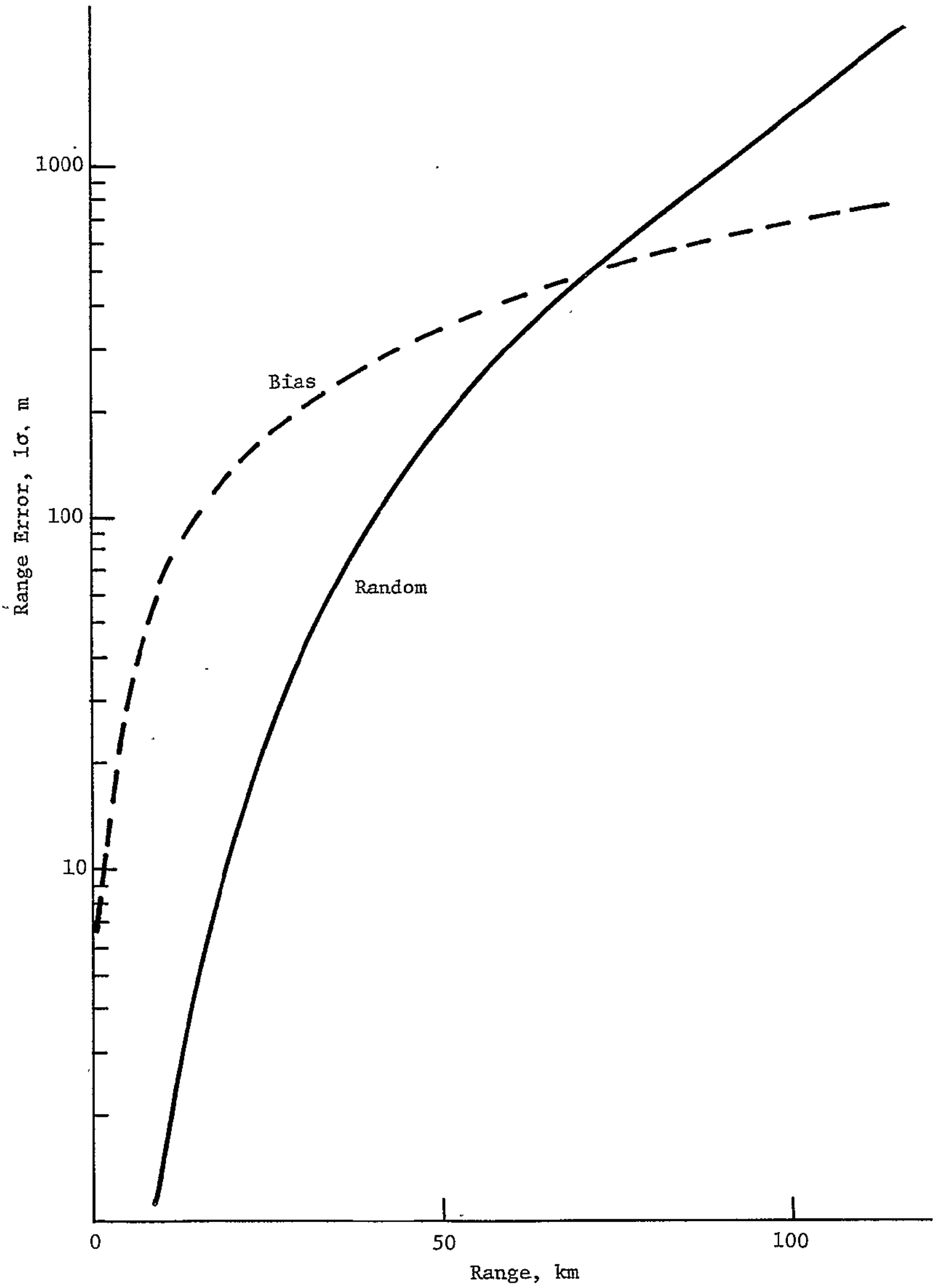


Figure VIII-4 Modified Apollo Rendezvous Radar - (Noncooperative Rendezvous) Range Error vs. Range

It can be seen that the random range error will increase almost linearly at ranges in excess of 60 km thus providing a 1σ error of 2.1 km at a range of 111 km (uncooperative mode) and 10.7 km at a range of 2680 km (cooperative mode). This may be excessively high for some applications and needs to be seriously considered in the selection of future long-range rendezvous and docking systems. The sensitivity of the basic Apollo rendezvous and docking radar has been measured for a variety of production models and found to exceed the specifications: Considering a signal to noise ratio of 10 db as adequate for acquisition at a maximum range of 400 n.m., a typical system with a radiated power of 318 mw will lose the acquired target at -129.5 dbm and reacquire it at -127.5 dbm. The two modified systems have been scaled to provide the same sensitivity performance but at different maximum ranges. Despite the large range errors at increased ranges these systems offer a series of important advantages which are listed below:

1) High-reliability performance with ICW mode

2) Accurate range rate data

3) Operation with either cooperative or uncooperative targets

4) Flight-proven system design

3. Gemini Rendezvous Radar - The Gemini rendezvous radar represents an early development in rendezvous and docking radar systems which also has been extensive flight tests. The radar is a pulse radar which again operates in conjunction with a transponder to provide increased range performance with a small, lightweight, low power consuming system. A pulse-doppler system mechanization was considered, but rejected because of the following reasons:

1) It is unable to effect velocity lockup and tracking if the relative velocity of the target with respect to the chaser is low.

2) No ground clutter rejection is possible with broad-beam, wide angle antennas.

3) System is more complex, heavier, and more costly

An FM/CW system mechanization was also considered, but it, too, was rejected because of the following reasons:

1) Spurious returns from Earth and ionsphere layers provide false range information.

2) Requires complex mechanization to separate the range and velocity information.

The pulse radar selected provides range, range rate, and angle information during terminal and docking guidance. To eliminate the need for the development of two radars, every effort was made to use common techniques and hardware in both the chaser and target systems.

Four wide beam, spiral antennas are employed for both the search and track functions. Three of these antennas are used for measuring range and bearing angle, while the fourth is used for transmission only thus eliminating the need for a duplexer. The interrogator radar transmits 1 sec pulses at a PRF of 250 pps and the transponder replies with a 6 sec pulse at the same PRF. The 6 μsec pulsewidth permits measuring range at the leading edge of the pulse as well as elevation and azimuth angle over the remaining portion of the pulse. The angle measurement is performed with a phase interferometer as shown in Figure VIII-5. The interferometer technique determines the angle off boresight by measuring the phase difference of the r.f. energy received by separated receiving antennas ($\Delta \phi = \frac{2\pi D}{\lambda}$ sin)θ. To avoid ambiguities over a $\pm$ 45 degree field of view, the spacing between the flat spiral antennas is made half a wavelength. The two horizontal antennas are used to obtain azimuth angle data, while the two vertical antennas are employed to obtain elevation angel information. Range is measured by standard methods; i.e., initiating the linear range sweep with the transmitted pulse and terminating it with the leading edge of the return pulse. The amplitude reached by the sweep is thus a measure of range. Range rate is obtained by differentiating the range data. The system has been successfully flown in the Gemini missions and has performed well. Since a klystron transmitter is employed, the long-term reliability is rather poor, and a fair amount of redesign would be required to adapt this sensor for future space missions. Similar modifications can be performed on this rendezvous sensor as have been proposed for the Apollo rendezvous radar (System #2 and #3) to achieve non-cooperative rendezvous and longer acquisition ranges.

Group B: S-Band Systems

1. CSM-LM Unified S-Band System - Basically this group contains systems which are extensions or modifications of the NASA Unified S-Band system to provide cooperative rendezvous functions at minimum cost for equipment modifications. Thus, if a particular application specifies the use of the NASA Unified S-Band system, these systems must be given serious consideration. The first system considered is a system proposed by Motorola in 1963. Motorola has designed and built a feasibility model consisting of a ranging unit, a CSM rendezvous transponder, a LM rendezvous transponder, and an auto track antenna/ pedestal. The transponders are modified Apollo units. The GSM transponder was modified to operate on two frequency ratios, 240/221 for earth mode operation and 220/239 for rendezvous. A turn-around ranging filter was also added. The LM transponder was modified to operate on slightly different frequencies for rendezvous and normally in the Earth mode. Angle track

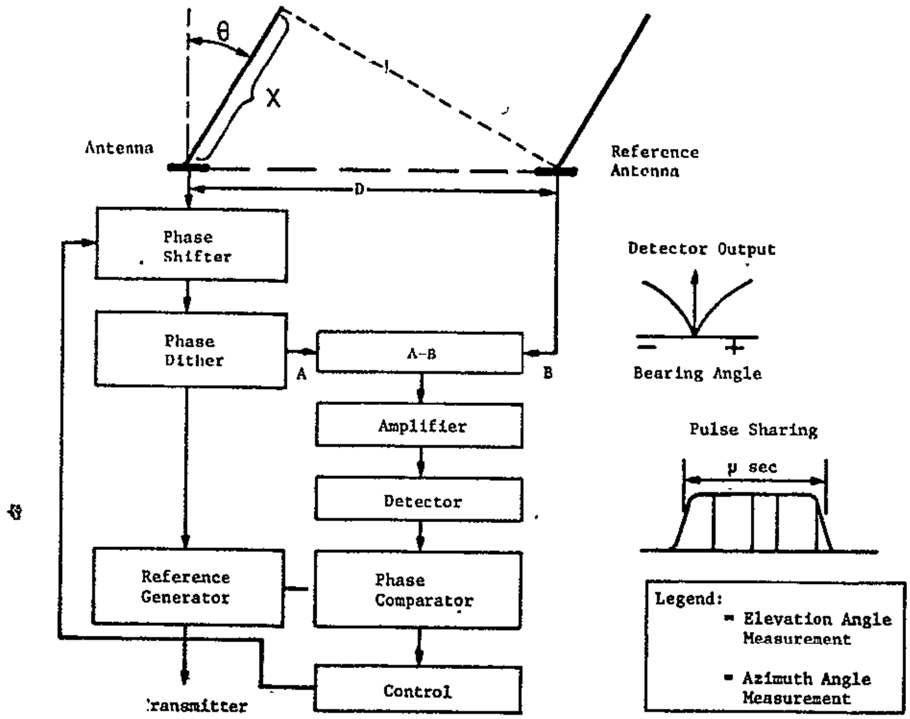


Figure VIII-5 Gemini Rendezvous System, Angle Measurement Block Diagram

receiver channels were added to the LM transponder. The ranging modulation consists of a 500 khz square wave subcarrier which is biphase modulated by a PN code sequence at a rate of 62.5 kilo-bits/sec. The PN sequence is 511 bits long and provides an unambiguous range of 800 n.m. The range modulation signal is phase modulated onto the S-band carrier. The transponder coherently removes the modulation from the carrier, filters it, and remodulates the transponder transmitter. The interrogator receiver then coherently removes the range subcarrier and transfers it to the ranging unit. The ranging unit recovers the ranging subcarrier and PN code, extracts the range data and converts it to a 17 bit binary number. Range rate is obtained by measuring the change in the received frequency at the interrogator due to the doppler effect. Angle measurements are performed at the interrogator which has a steerable antenna that contains five helices and an amplitude-comparison front end. Azimuth and elevation data are provided but the high-gain antenna must be pointed to within $\pm$ 16° of the target transponder in order to lock onto the proper null. Two-way r.f. cable losses of 10 db in the transponder appear unduly high for this design, and need to be reduced for an operational system. Acquisition is performed in two stages:

1) Rf acquisition

2) Ranging Signal acquisition

Since acquisition time of the ranging signal is directly proportional to range, this system is not the ideal solution for very long-range rendezvous missions. Typical acquisition times are 1 sec for R = 13.5 n.m. and 29.6 sec for R=400 n.m. If the maximum range is increased to 1200 n.m. the time for code acquisition increases to 88.8 sec. and the overall maximum acquisition time is 110.8 sec. If the system were to be used also in conjunction with ground transponders, these long acquisition times would make sequential interrogation for trilateration impractical. Another disadvantage is the mechanical problems associated with the installation of a 4 foot steerable antenna in the chaser vehicle. However, despite these disadvantages, the system has been checked out and does provide a relatively simple modification to the current NASA Unified S-band system. A block diagram of this system is shown in Figure VIII-6. The sensitivity of the system, although reduced by the high r.f. cable losses, yields a S/N = 23.4 db at 400 n.m.

2. USCAN Unified S-Band System - Another modified S-band rendezvous system is the USCAN (Unified S-Band Communication and Navigation System) proposed by TRW Systems. The major difference between this system and the just-described Motorola system is listed below:

1) USCAN provides rendezvous and ranging to ground transponders while the Motorola system provides the rendezvous function only.

2) USCAN does not measure angles while the Motorola system provides accurate angular data for cooperative rendezvous.

3) USCAN uses a sequential BINOR Code instead of the PN code employed by Motorola.

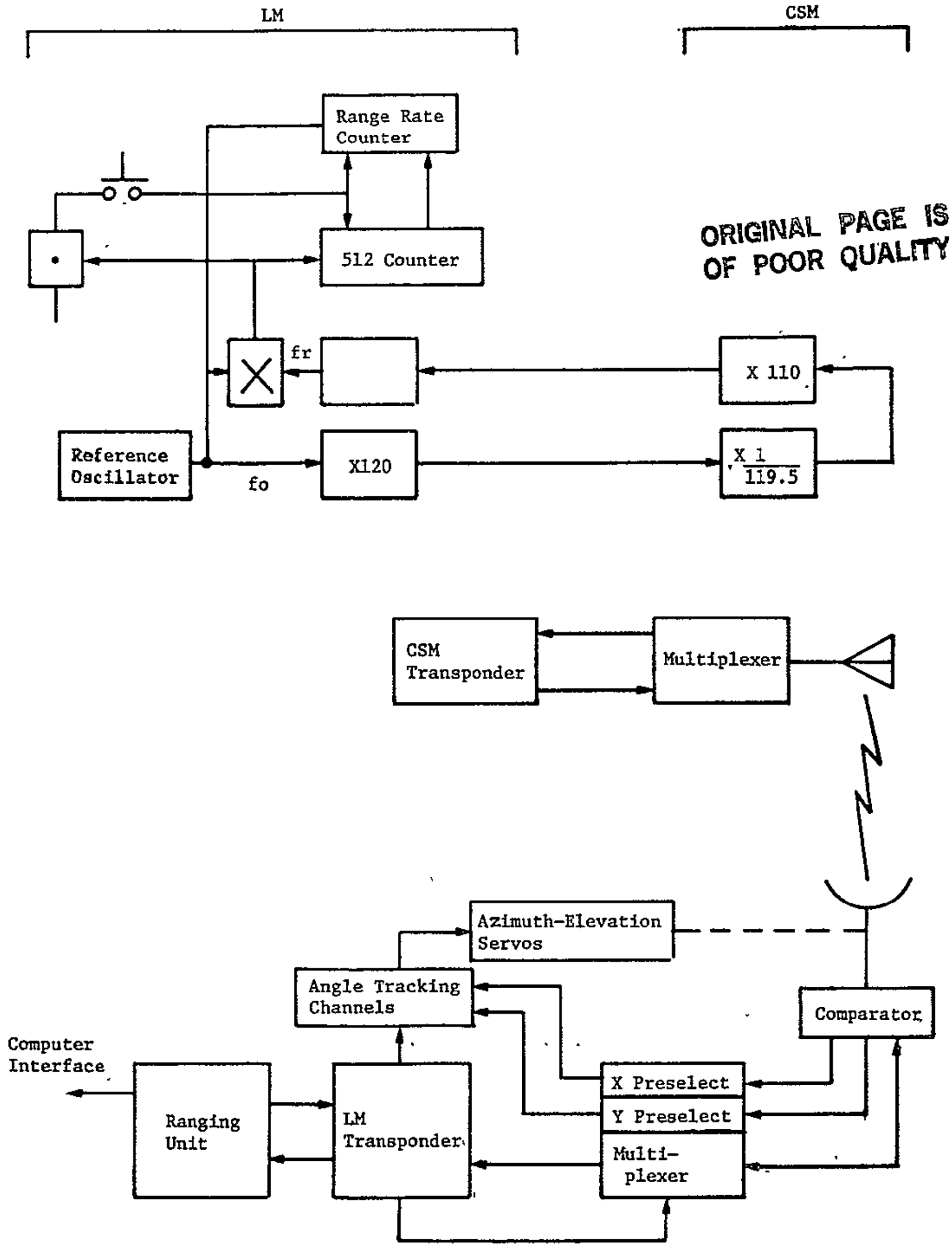


Figure VIII-6 Motorola Unified S-Band Rendezvous System, Feasibility Model

Furthermore, with the USCAN system, voice, data transmission and ranging can be performed using one S-band carrier. Multilateration using ground transponders is employed for navigation updating in the terminal region. All ground transponders as well as the rendezvous transponder are unmodified. USB transponders with individual identification codes. For multilateration, the following sequence takes place:

1) Airborne interrogator is energized (f = 2106.4 mHz) and all ground receivers within line of sight will acquire and lock onto this signal.

2) Airborne interrogator is modulated by the 70 kHz FSK coded command signal to turn on ground transponder #1.

3) On receipt of command, transmitter #1 will radiate at a frequency 240/221 times the received carrier frequency *f = 2272.5 mHz) and the interrogator receiver will acquire this signal.

4) Airborne BINOR code generator transmits the code sequence which is trun-around transponded and the round trip time is measured by the BINOR code processor.

5) Range rate is measured by counting carrier doppler cycles.

6) Transmitter #1 is commanded OFF and the ranging sequence is repeated for the other ground beacons.

The sequential BINOR code provides an acquisition time of 0.3 seconds, which is a tremendous improvement over the Motorola system. A block diagram of the USCAN system is shown in Figure VIII-7, while a block diagram of the transponder is shown in Figure VIII-8. Voice and data are transmitted both ways while range and range rate is extracted from the received signal. The data link has a S/N ratio greater than 10 db. up to ranges of 1000 n.m. The voice threshold of 16 db is reached at approximately 200 n.m. The principal disadvantage of the USCAN concept for rendezvous and docking is that the system does not provide angle data. On the other hand, the fast acquisition time and the communication and navigation functions provided by the system represent a distinct advantage in many applications.

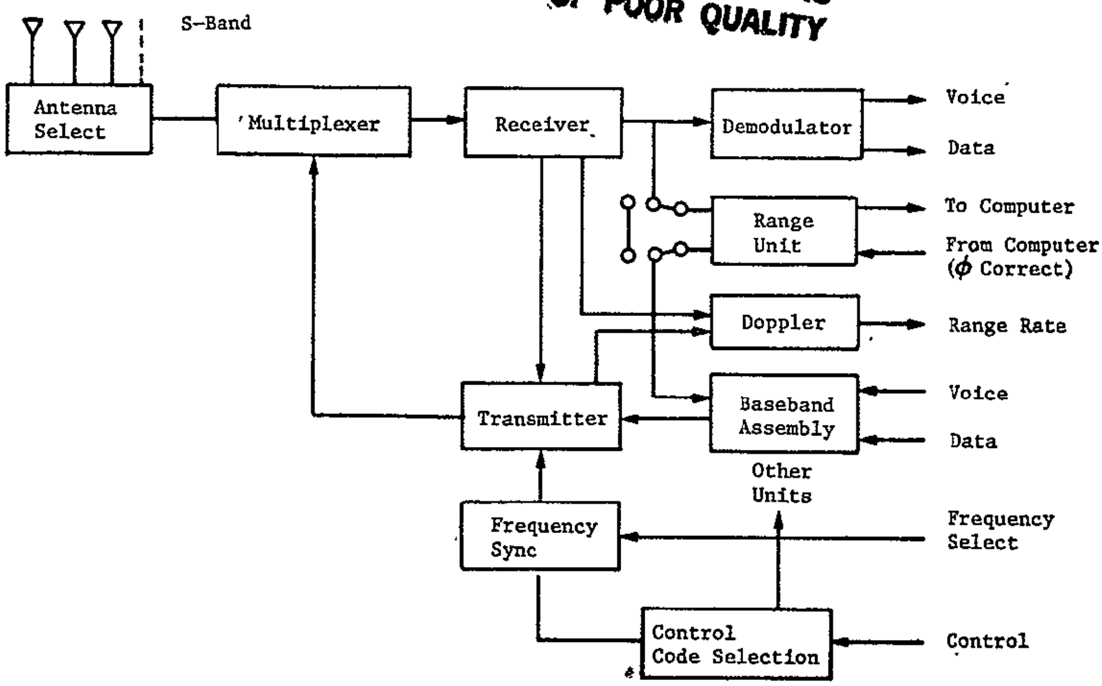


Figure VIII-7 USCANS System Vehicle Equipment

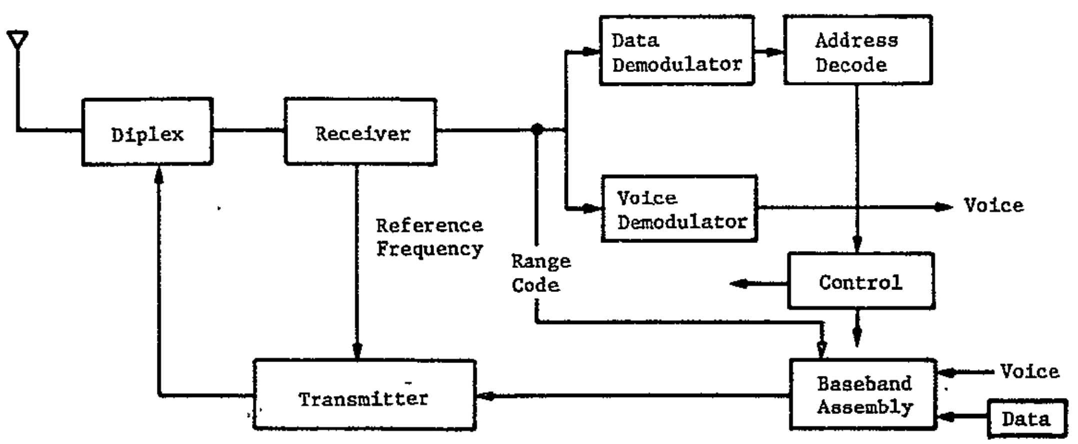


Figure VIII-8 USCANS System, Landing Site Transponder Block Diagram

Group C: Ku Band Systems

This group contains special rendezvous radar systems for the non-cooperative rendezvous mission. The system described in this section will be the Norden Ku-band radar, the MMC solid state docking radar, and the Raytheon shuttle rendezvous radar. The last two are proposals for implementing a long-range rendezvous radar and a short-range docking radar in non-cooperative rendezvous missions, and incorporate the latest advances in rendezvous radar design.

Modified military radars such as the AN/APQ-148 and the AN/APQ-144 are special, complex radar systems which must be modified by removing unneeded circuitry in order to adapt them to the relatively simple rendezvous task. These systems, although providing extremely high accuracy, generally have a lower reliability than the systems described in Group A, but have the advantage of having been thoroughly tested for the specific military applications.

1. AN/APQ-148 - The AN/APQ-148 is a highly sophisticated, military "attack multimode" radar which provides phase interferometry, azimuth beam splitting, and range tracking techniques for pinpoint accuracy in three dimensions on fixed and moving targets. It is the only system analyzed which provides frequency agility over the 16-17 gc band thus effectively smoothing out the drastic variations in the radar cross-section of uncooperative targets. Unfortunately, the system is much more complex than what is required for the rendezvous and docking mission, and modifications would be required to eliminate a lot of unnecessary circuitry in the radar.

The system's maximum range of 102 km is based on an "average target radar cross-section" of $10m^2$ which appears marginal for some applications. Most "uncooperative" space vehicle targets are expected to have "average" radar cross-sections in the $1\text{-}3m^2$ range unless passive augmentation devices such as corner reflectors or retrodirective lenses are provided to increase the target radar cross-section. The "uncooperative" target is designated by the computer to a 1σ ellipsoid of 4 n.m. diameter and 5 n.m. long at a range of 100 kms. Radar scan in range, azimuth, and elevation is initiated until target acquisition occurs. Range, azimuth, and elevation data are then supplied to the guidance computer for optimization of the approach path. Unfortunately, no range rate data are provided. This limits the system's usefulness for rendezvous missions. Other disadvantages are the short maximum range, high power dissipation, and the relatively low system MTBF. As stated before, some of the major advantages provided by this system are its high accuracy and capability to provide frequency agility. Also, the basic system has been extensively tested and qualified, and as such could be readily modified for a particular application.

2. Raytheon Rendezvous Radar - The Raytheon rendezvous radar is shown in Figure VIII-9. The radar provides range and angle tracking of a one square meter, non-cooperative, Swerling I target from 30 n.m., and uses a low risk, low weight and low cost, all solid-state approach. This design has been proposed for the Space Shuttle Orbiter. The assigned radar portion of L-Band, centered at 1325 MHz, was selected to exploit the solid-state transmitter/receiver module technology developed by Raytheon for other radar applications, and also to take advantage of the efficiencies to be gained at the frequency band. An attractive combination of electronic beam steering for elevation and mechanical steering for azimuth is proposed for a 36" x 36" planar array that can provide hemispherical coverage, if necessary. Sixteen solid-state modules make up the array and provide 0.99 probability of acquisition at 30 n.m. with only 40 watts of peak power per module. Acquisition is accomplished in a 40^{o} x 40^{o} sector in 12 seconds. Design efficiencies are reflected in the relatively low maximum power input required of 732 watts (which decreases rapidly with target range) and low weights of 41.6 lbs. for the antenna and 40.6 lbs for the separate electronics package.

A three-foot L-Band array is proposed which has 13-degree beamwidth. The 3 tracking accuracy requirement is 10 mr (for random errors) equivalent to a beam splitting accuracy (3) of approximately one part in 20. With the high signal-to-noise noise ratios obtainable at L-Band with relatively simple signal processing, this beam splitting accuracy is readily achieved.

The target, is representative of a one square meter, Swerling 1 target, and probability of detection is 99%. An efficient design thus requires frequency diversity to lower power requirements and reduces acquisition time. For angle search and track, Raytheon proposes an array electronically scanned in elevation and mechanically rotated in azimuth, which can provide up to hemispherical coverage if required. Of interest is that the array antennas mechanically scanned in azimuth and electronically scanned in elevation are the only type of array radar systems that have been produced in large quantity. This basic concept appears to be the best choice even if the specified coverage volume is greatly reduced.

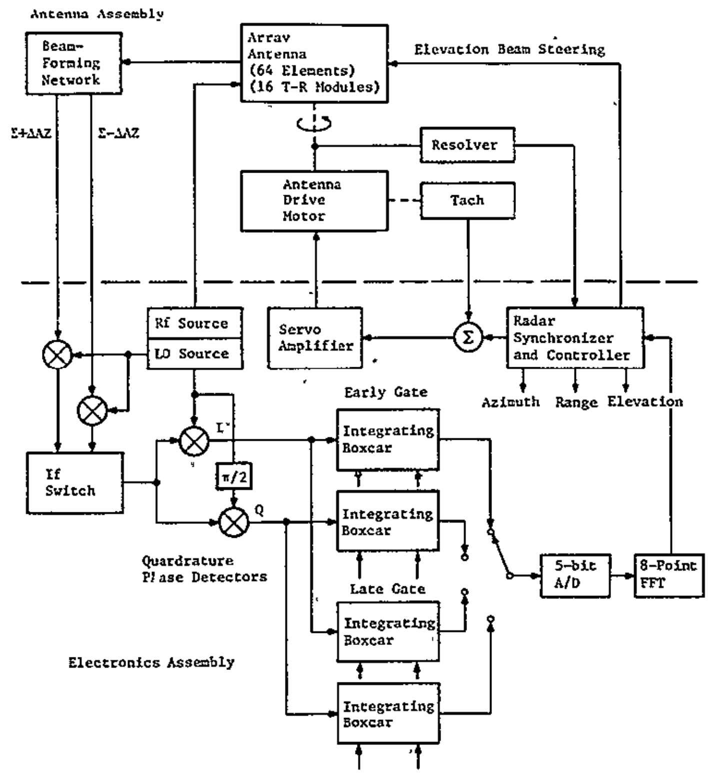


Figure VIII-9 Raytheon Rendezvous System

When the target is acquired, the antenna is mechanically moved to the target azimuth and tracking commences. The transmit waveform is changed to transmit 16 consecutive pulses at each frequency, For elevation tracking, the first 8 pulses are transmitted and received in 1 beam position, then the next 8 are transmitted in the adjacent beam position. Each set of 8 is coherently processed in the FFT, and then, if the signal-to-noise is high enough, the error information is provided to the tracking loop. A similar operation is provided for azimuth track, except the pulses are transmitted on a sun beam and received, first on a beam squinted left, and then on a beam squinted right.

During tracking, frequency agility is still being used between 16-pulse subgroups. AGC is accomplished by shortening the transmitter pulse length to maintain the correct signal-to-noise ratio for tracking from the transmitted frequency that gives the largest target signal return. The net result of this combination is a very low transmitter average power requirement in the track mode. A very short range during tracking, the operating configuration of the antenna array will be changed so that transmission is accomplished with the outer eight element subgroups in the array, and reception is done with the inner eight subgroups. This mode of operation eliminates the necessity to change from transmit to receive in a single transmit-receive module in a fraction of a microsecond.

Random errors in the angle and range outputs and their derived rates are shown in Figures VIII-10 and VIII-11 for the extreme range case The angle error is for either azimuth or elevation. About one minute after detection the angle errors reach specified three sigma levels of 10 MR and 0.2 MR/sec.

A range rate requirement of 1 ft/sec three sigma error dictates the use of a pulse compression scheme at about 100:1. This would be implemented with an acoustic wave filter approach, a technique Raytheon is using on such programs as SAM-D. Range error is more than an order of magnitude below a specified level of 390 feet, three sigma at this range, indicating that the need for very accurate range rate should be re-examined. Three sigma range error at minimum range is specified as 120 feet.

An MTBF of 3000 hours is projected for the system at full performance level. This includes 6250 hours for the 16 solid-state T/R modules, each of which would have an MTBF of at least 10^5 hours, as being verified by life tests now in progress. The effective system MTBF would approach 6000 hours for operation below rated performance because all T/R modules need not be working to provide a useful system output.

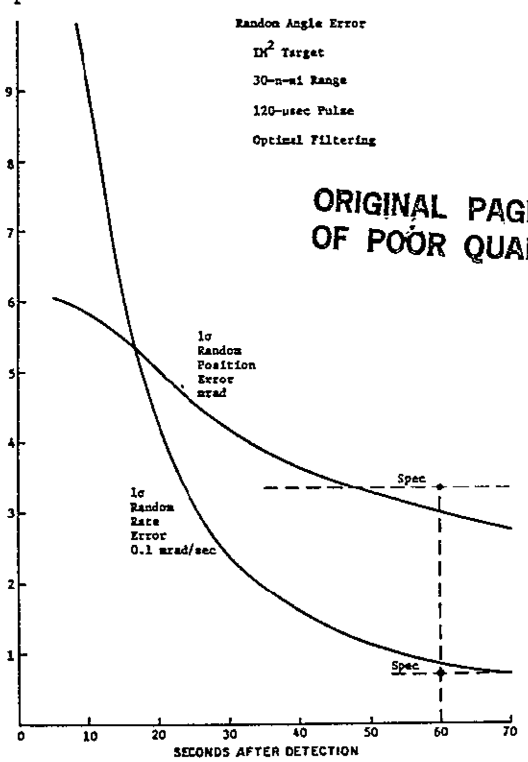


Figure VIII-10 Random Angle Error

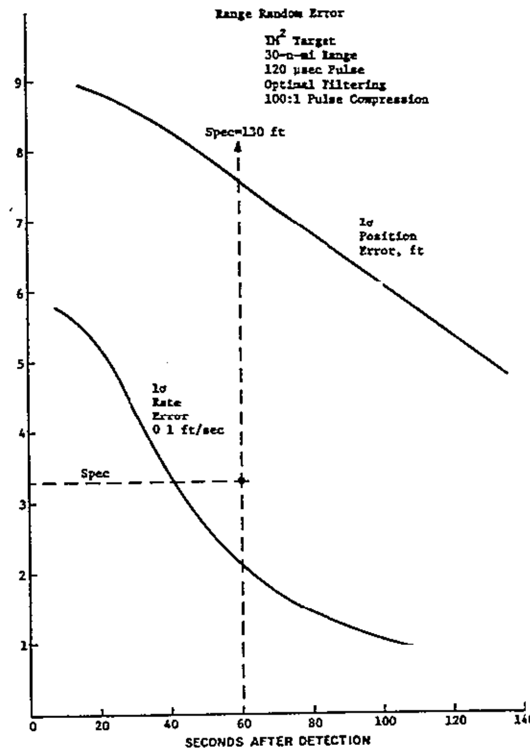


Figure VIII-11 Random Range Error

Group D: VHF

1. Apollo VHF Ranging System - This group contains VHF ranging systems that can be adapted for the rendezvous mission. Unfortunately, none of these systesm provide angle data which is so important for rendezvous and docking. The first system considered is the Apollo VHF ranging system. The ranging equipment which modulates the VHF transmitter and processes the received ranging signals are the DRG (digital range generator) and the RTTA (range tone transfer assembly), the latter located on the target vehicle. Communication and ranging are performed simultaneously except for a 15 second period for ranging acquisition. A clock driven tone generator produces the ranging frequencies. :The 3.95 kHz signal is transmitted first, next a module two combination of 3.95 kHz and 246 Hz, and finally the 31.6 kHz signal is transmitted. After fine tone acquisition time of 15 seconds, slant range data can be visually displayed or read out to the onboard computer. The 31.6 kHz signal provides the range measurement accuracy while the 3.95 kHz and 246 Hz signals provide unambiguous measurements for the 370 km operating range. In the target vehicle the two low-frequency tones are simply retransmitted back but the 31.6 kHz tone is tracked with a VCO loop in order to reconstitute the signal to be transmitted.

A frequency sensor determines which tones are being received and switches the VCO tracker in and out accordingly.

In order to make this system suitable for rendezvous missions, a number of changes would be required. These are listed below:

a) For long-range rendezvous (R=2680 km) the transmitter power must be increased from 2.5W to 9.6W.

b) In order to increase the maximum unambiguous range to 2680 kms, another low modulation frequency of 33 Hz must be added to the transmitted spectrum.

c) In order to furnish Doppler data at close range, the transponder must be modified so that it retransmits at an RF frequency derived from the received frequency.

d) Additional circuitry must be added in the interrogator to extract the Doppler data.

e) A substantial redesign would be required to improve the range accuracy to better than 10 m (1σ).

The major advantages of this system are the dual use of the VHF link for ranging as well as voice transmission and the fact that the equipment is available. However, neither the range nor the range rate accuracies obtainable by differentiation of the range data are good enough for most rendezvous missions. A block diagram of both the interrogator and the transponder are shown in Figures VIII-12 and VIII-13.

2. Modified VHF Range and Range Rate System - A modification to the Apollo VHF Ranging System has been proposed by RCA. The improved system has the following capabilities:

a) Unambiguous ranging to 2400 kms.

b) Simultaneous ranging with digital data and digitized voice transmission without degradation in performance.

c) Improved ranging accuracy at close ranges (3m (1σ) T/R = 18.5 km)

d) Range rate extraction

Block diagrams of the duplex ranging system and the receiver (transponder or interrogator) with dual-channel ranging demodulator are shown in Figures VIII-14 and VIII-15. As in the unmodified system, ranging tones and the digital data and voice information are modulated on the RF carrier. The receiver has a wide-band front end and two matched IF channels. The top channel has a mixer which receives the biphase modulated version of the range code to strip off the

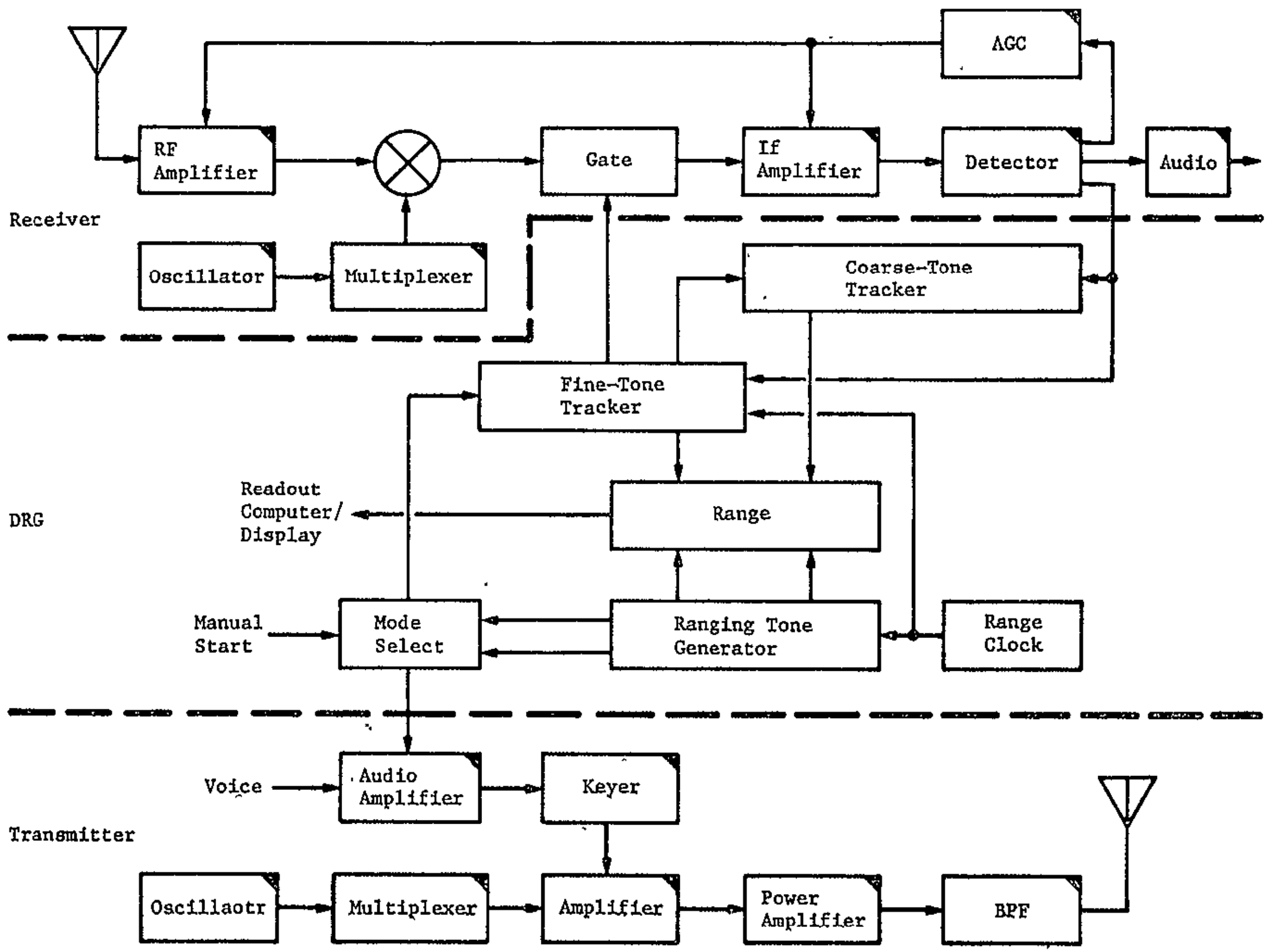


Figure VIII-12 Apollo VHF Ranging, CM VHF Radio (Ranging Function)

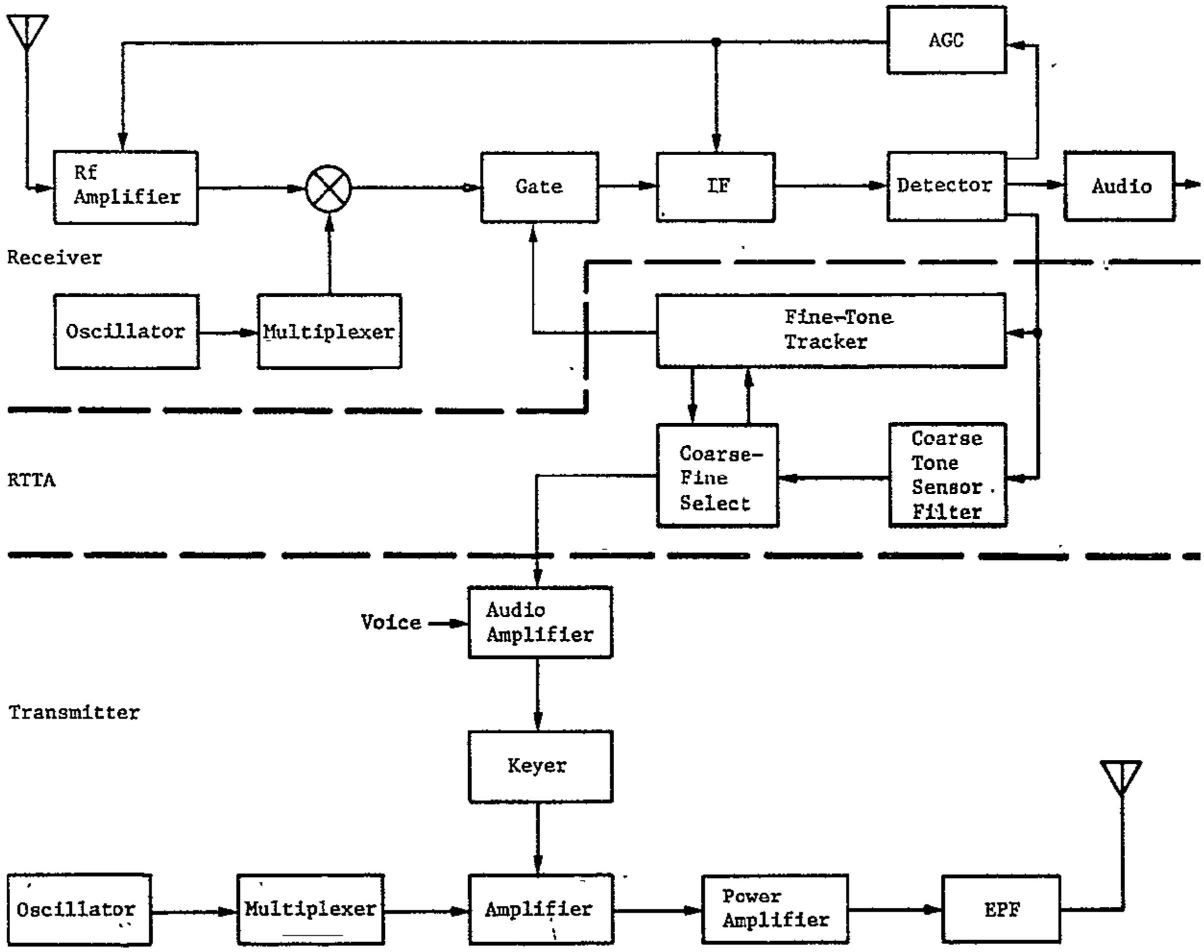


Figure VIII-13 Apollo VHF Ranging, LM Transponder (Ranging Function)

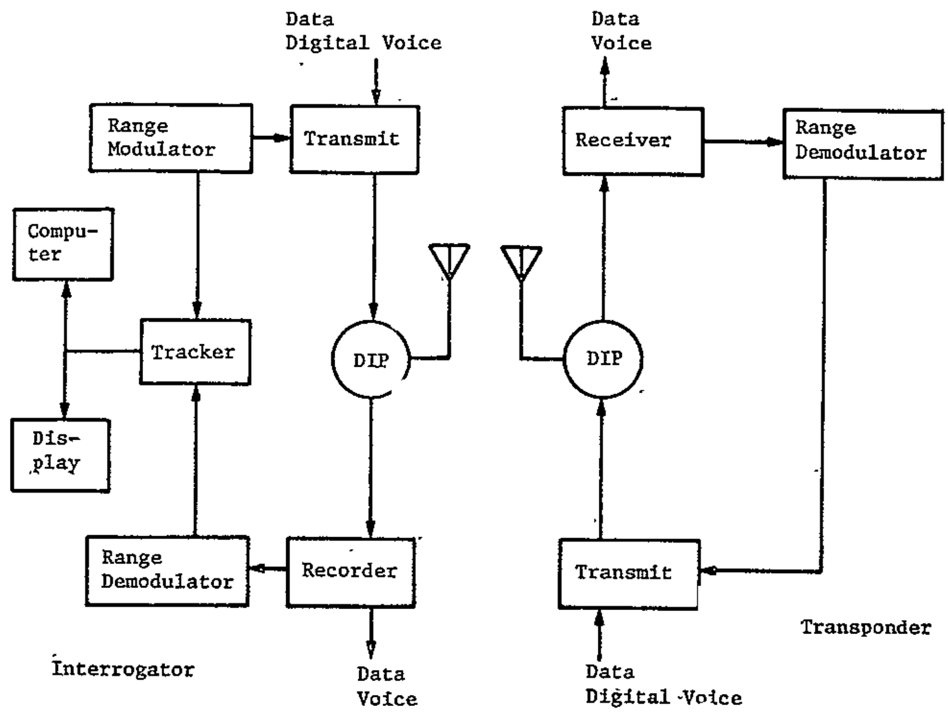


Figure VIII-14 Modified Apollo VHF Ranging System, Duplex Ranging System

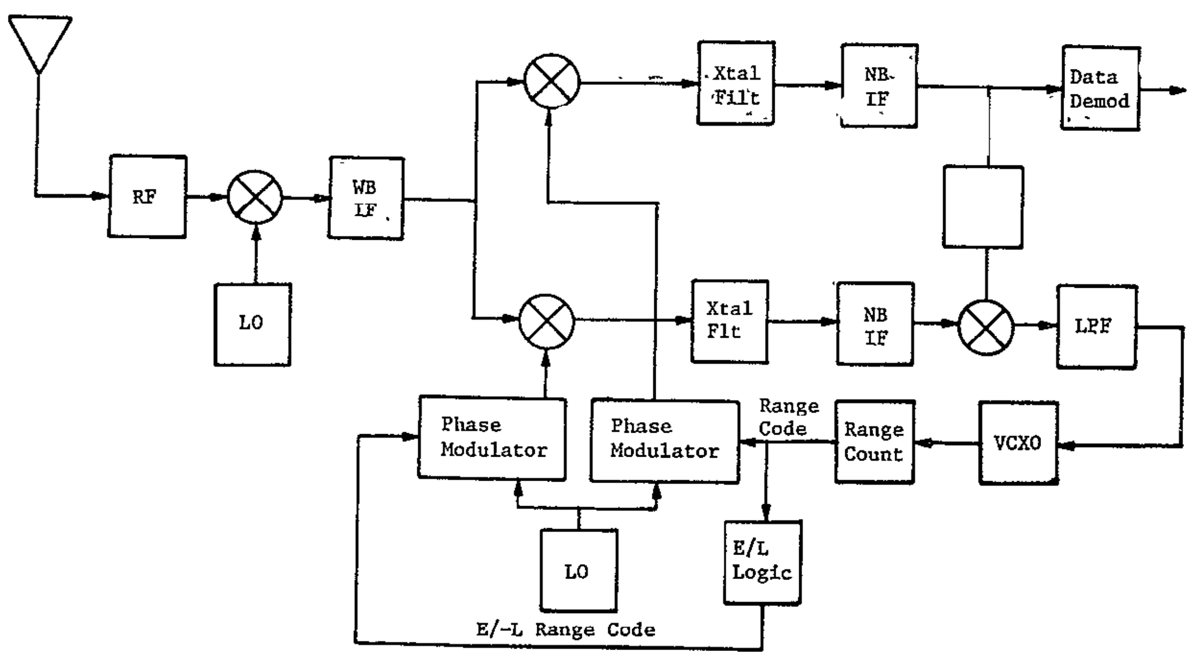


Figure VIII-15 Modified Apollo VHF Ranging System - Receiver (Interrogator and Transponder) with Dual-Channel Coherent Ranging Demodulator range modulation from the data signals. It is subsequently filtered, amplified, and applied to a coherent data demodulator. The bottom channel uses an early/late ranging code to bi-phase modulate the second local oscillator. The signal is then used as an injection for the mixer. The output from the mixer is filtered, amplified, and the data stripped off by using a limited sample from the top channel. The error signal is coherently demodulated and the output is a DC error signal, which is filtered and used to "steer" the ranging clock.

In order to obtain good range rate data, RCA proposes to carry out a coherent Doppler measurement. At the transponder, the receiver rf is tracked and used to synthesize the transmit frequency. In time, the interrogator receiver tracks this signal and compares it to the transmitted frequency, thus allowing extraction of the Doppler frequency. This provides a range rate accuracy of 1 cm/second for an averaging time of about 60.8 seconds.

The proposed modified system has the advantage of correlating the ranging signal prior to subjecting it to the variable-delay effects of narrow band filters. Also, the proposed ranging technique permits the simultaneous transmission of digital data, digitized voice, and ranging data. The improved range accuracy could be achieved in the rendezvous phase, but not in ranging to ground transponders since the latter involves large ionospheric propagation errors at 260 mHz. This effect can be reduced substantially by increasing the RF frequency to S-band. However, in this case, some of the other S-band systems discussed before would probably be more desirable for both rendezvous and ground ranging applications.

3. Motorola Range, Range Rate System - The last system analyzed in this group is a rendezvous range and range rate system proposed by Motorola which is also based on the Apollo VHF system. The system has been carried to the feasibility stange and appears to meet the Apollo rendezvous requirements. A block diagram of the system is shown in Figures VIII-16 and VIII-17. It consists of a Master Unit in the chaser vehicle, which displays the range and range rate data, and a Remote Unit in the target vehicle. The system employes a BINOR ranging code and delta modulation for voice. The BINOR code provides extremely fast acquisition times. System acquisition time of 2 seconds per terminal is based upon 125 m sec. per code component and one second for the clock loop. The range code generator is a straight binary counter operating at 8 mHz. The BINOR code is obtained from the upper half of the counter while the lower, least significant portion provides the "within bit" range resolution. Binary range is extracted directly from the

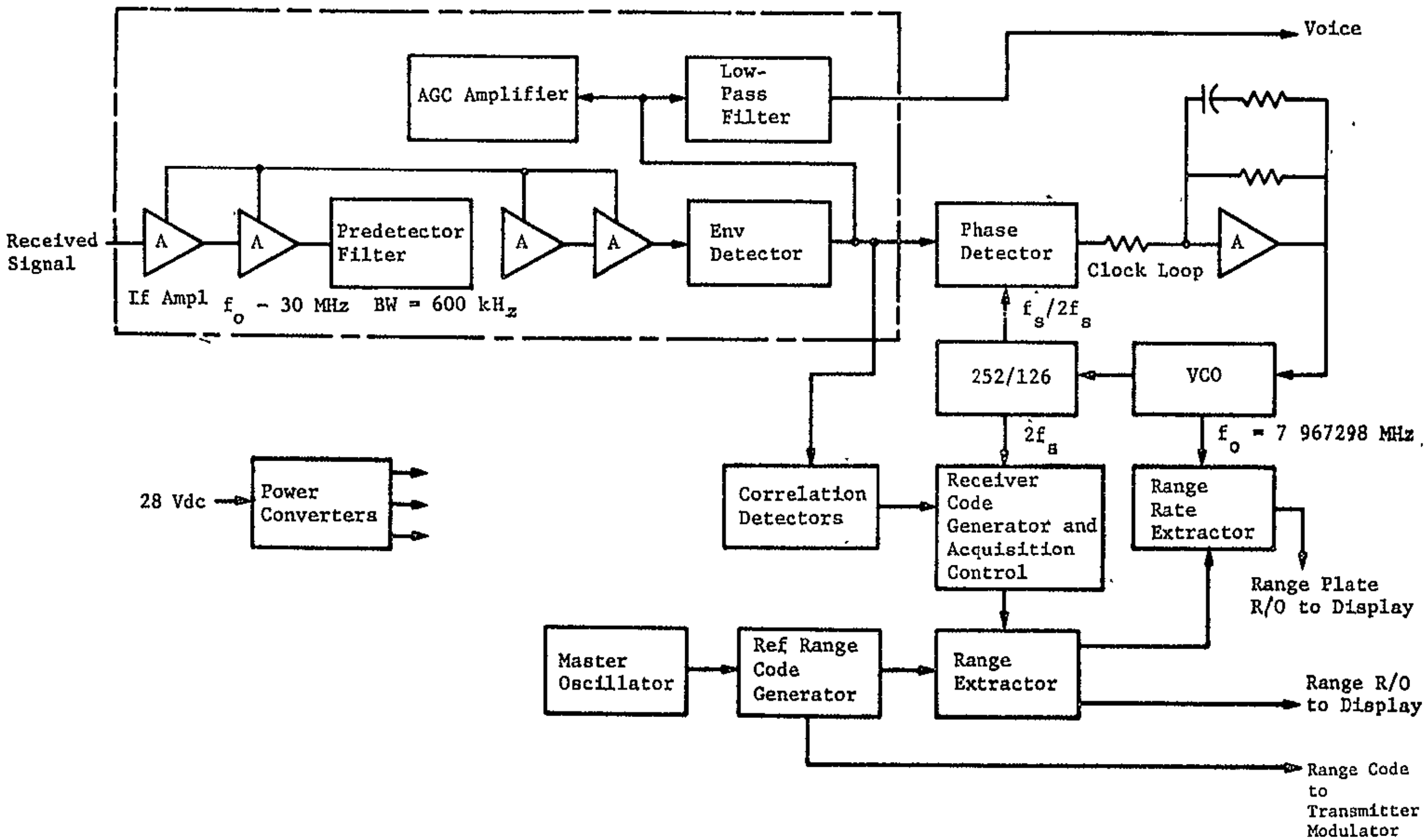


Figure VIII-16 Motorola Rendezvous Range/Range Rate System Master Unit, Functional Block Diagram (1)

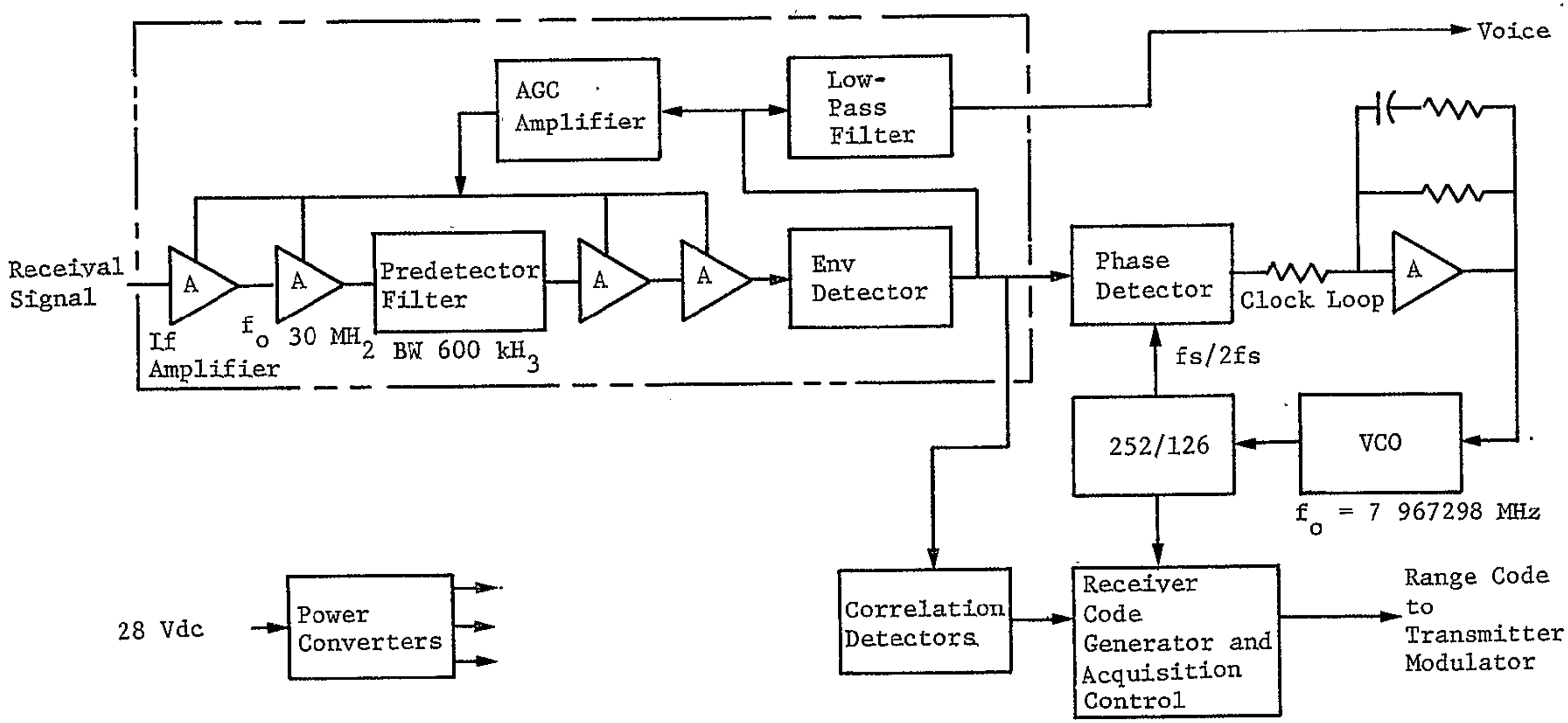


Figure VIII-17 Motorola Rendezvous Range/Range Rate System Remote Unit, Functional Block Diagram (1) reference BINOR counter. Range is supplied as a 15 bit binary number and also as a 5 digit decimal number. Range rate is extracted directly from the two-way Doppler frequency that exists on the range code. By a process of filtering, mixing, and multiplication, the Doppler is transferred onto a 126 kHz bias. The bias frequency plus or minus the Doppler is counted over a fixed time interval of 2 seconds. Range rate accuracy is limited by the low code clock rate of 8 mHz. The maximum unambiguous range determined by the lowest code component frequency, is too small for some applications and should be increased by a factor of four by adding two more low frequency components to the BINOR code. Range rate, although of reduced accuracy, is extracted in the system and is thus available to the pilot.

All of the systems in Group D lack angle measurement capability, and in order to effect space rendezvous this feature will have to add such a capability to an existing range/range rate system than to select new, expensive rendezvous systems which will require additional development and space qualification.

Group E: C Band

This group contains a collection of miscellaneous systems that could be adapted to the rendezvous and docking mission. The AN/SPS-105 and AN/MPS-36 precision tracking radars supply precise target position information to remote computers, displays, or plotting boards and could, therefore, be employed in rendezvous task. The OWDONS and AROD systems can be employed as cooperative rendezvous systems in conjunction with suitable ground equipment. The OWDONS system is a one-way doppler system which performs range difference measurements to two or more CW ground transmitters by integrating Doppler frequencies over consecutive time intervals. The onboard computer then provides spacecraft position and velocity. The AROD system performs range and range rate measurements to ground transponders. The system is fairly complex and requires a separate VHF control link. Range and range rate data are available simultaneously for up to four ground transponder links four times per second. There are always at least four satellites in view of use with only three required for a position fix. Four ground stations serve to update satellite position information whenever satellite is in view of ground station.

Optical Systems

Group F: Laser Rangers

1. ITT Scanning Laser Radar (SLR) - The scanning laser radar (SLR) was designed to determine relative position and attitude of two vehicles engaged in rendezvous and docking. The system can be classified as a passive cooperative one where the target is fitted with a number of optical reflectors. The heart of the system (Figure VIII-18) consists of a GaAs (.9um) laser transmitter, a piezoelectric beam steerer, retroreflector, mounted on the target vehicle, receiver optics, and a scanning optical detector synchronized to the beam stearer. The outputs of the hardware are also listed in Figure VIII-18. During acquisition, the .1° laser beam is steered over a 30° x 30° field of view (FOV) in .1° increments. The steering is accomplished through the use of a mirror mounted to piezoelectric crystals (Figure VIII-19). As voltage is applied across the crystal, it deflects thereby resulting in a rotation of the mirror about an axis. Simultaneously to the scanning of the laser transmitter, the .1° x .1° instantaneous FOV of the receiver is scanned so that the two FOVs coincide. Steering of the receiving FOV is accomplished through the use of an image dissector (Figure VIII-20). As illumination from the receiver optics strikes the photocathode surface, electrons are emitted and the path of these electrons is deflected by amagnetic field produced by current within the coils surrounding the chamber. Some of the electrons then pass through a small aperture into a photo-multiplexer which detects their presence. The incident angle of the electrons reaching the photo-multiplier and, therefore, the instantaneous FOV is dependent upon the current passing through the coils and is controllable.

The range to the target is determined by measuring the later pulse propagation time from the transmitter to the target and back to the receiver in increments of .67 nanoseconds (1498 mHz). Using this configuration, range can be determined to a resolution of ± 10 cm.

The relative attitude of the two vehicles is determined by measuring the range and angle to four retroflectors mounted on the target in a T configuration (Figure VIII-21). A transformation matrix can be derived which will take the four vectors obtained by the SLR and yield attitude offsets for the vehicles.

Rates of the above quantities are determined through differentiation and accuracies are given in Table VIII-1.

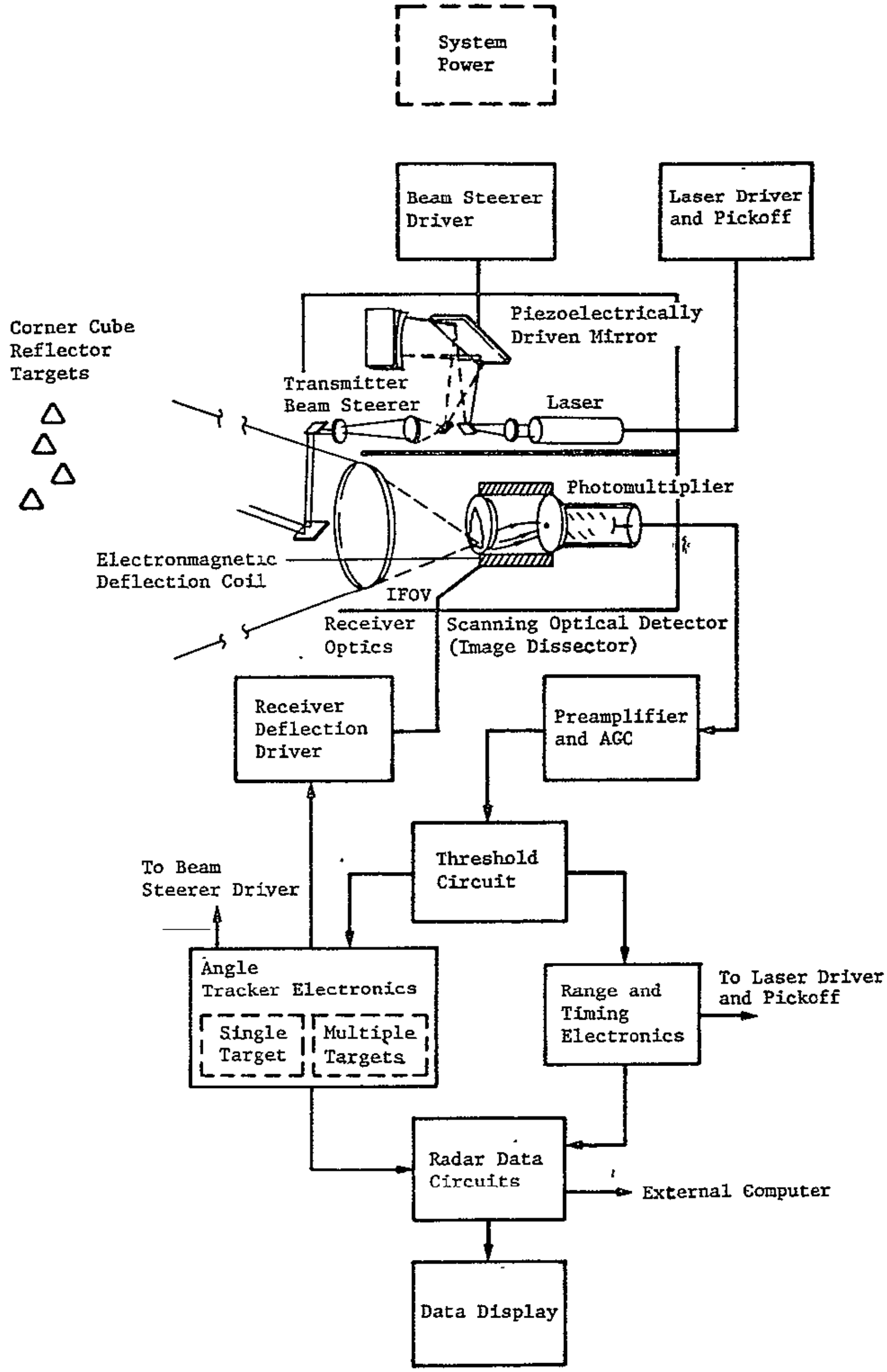


Figure VIII-18 Block Diagram of the SLR Generation No. 3 Transmitter-Receiver and Electronics

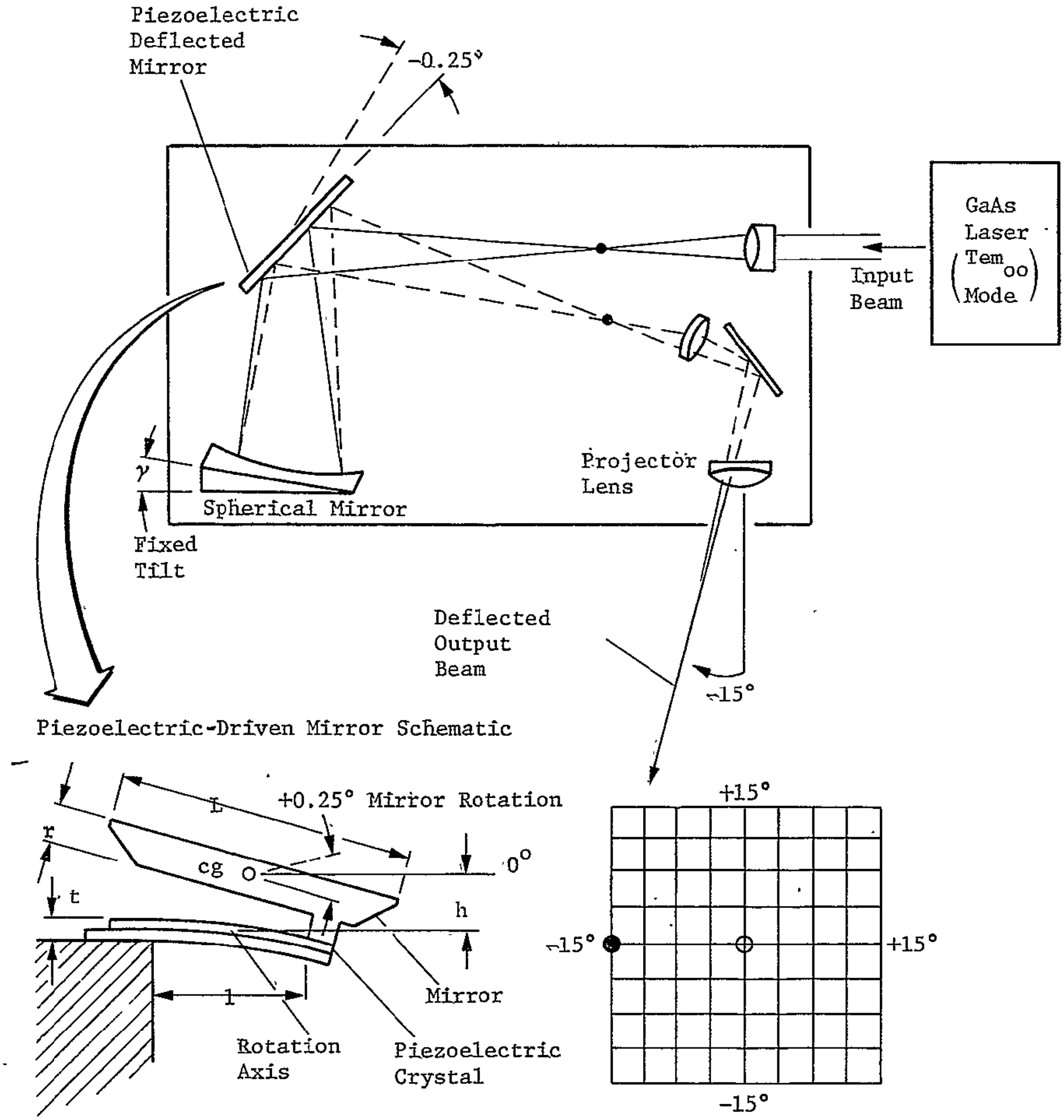


Figure VIII-19 Schematic of the Beam Steerer Optics

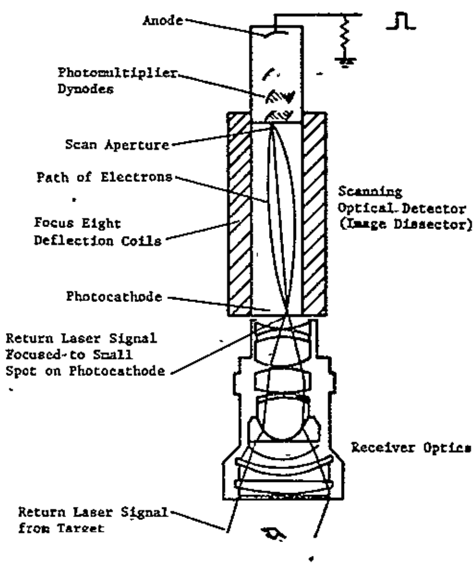


Figure VIII-20 · General Schematic of Scanning Optical Detector (Image Dissector)

2. Optical Docking Sensor (Lockheed) - The optical docking sensor developed by Lockheed was designed for the final docking maneuvers of large space structures. This is reflected in its maximum acquisition range of 300 meters, and range accuracy of 1 mm. The system is capable of measuring range LOS angle to the target, and the relative attitude of the two vehicles. The docking sensor utilizes a multitone, amplitude modulated CW Nd:YAG laser transmitter which illuminates a variable FOV ranging from $5^o \times 5^o$, during acquisition, to $20^o \times 20^o$ during final docking. Three retro-reflectors placed on the target vehicle return a fraction of the light, and the receiver resolves each target in angle and range. The output of the sensor is then fed into an onboard processor which determines the relative position and attitude of the two vehicles. The laser is modulated by an electro-optical modulator designed to impress either or both a 375 kHz sinewave and a 75 mHz sinewave. By allowing two frequency, high accuracy tone can be eliminated. In this configuration, the low frequency tone is used to measure range down to a few meters while the high frequency tone is used to measure range to 1 mm.

Table VIII-1 Optical Docking Sensor

System	Manufacturer	Sensor Type	Target Type	Spectral Range	Total Field of View	Instantaneous Field of View	Accuracies: Range	Accuracies: Range Rate	Accuracies: Angle	Acquisition Range and Time	Projected Size, Weight, Power	Computation Requirement
Scanning Laser Radar	ITT	GaAs Laser Ranger with piejo electric beam steerer and image disector tube	Cooperative with retro-reflectors	0.9 mu	30x30	0.1°	0.1 m	Range found by differentation 1.0 m/sec	1.0%	95 mi	Size 1.5 ft^3 weight 55 lb power 50 W	None Output in digital format
Optical Docking Sensor	Lockheed	Nd: Yag Laser ranger using a variable FOV for the illuminator and an image disector receiver.	Cooperative with retro-reflectors	1.06 um	Variable 5x5 acquisition 20x20 final docking	3 mradians (varies)	1 mm	1mm/sec	3 mr	1000 ft	Size weight NA power	(None) Hardwired if attitude is to be determined some processing must be made available
Optical Docking Sensor	RCA	GaAs illuminator-range determined through geometrical calculations of retroreflector position Silicon VADICON receiver	Cooperative with retro-reflectors	Natural or artificial illumination 0.9 um	Variable 3 to 30	0.002 to .02	0.76 final	Found by differentiation	0.4° final	1000 ft 4 ft min range acquisition time proportional to scan time	Size - 5.6x6.9x 19 weight 20 lb Power 65 W	Memory-220 words 1500 equivalent add cycles per computational cycle
Automatic Stereoscopic Camera	Martin Marietta	Stereo TV using processing to yield range and angle information	Cooperative or non-cooperative (accuracy increase with use of retroreflector)	Natural or artificial illumination	30x30 (in study)	Not applicalbe	6.5%	Found by differentiation tion	0.04°	Dependent on type of illumination and power	N/A	N/A
Centroid Tracker	Martin Marietta	TV Camera with specialized scan control, video processor and an onboard processor video data to yield target position and attitude	Noncooperative or cooperative	Natural or artificial illumination	30x30	Sensor dependent (0.02° for SI Vidicon)	Innacurate for Range Determination		Dependent on optics	Dependent on type of illumination and power	N/A	N/A

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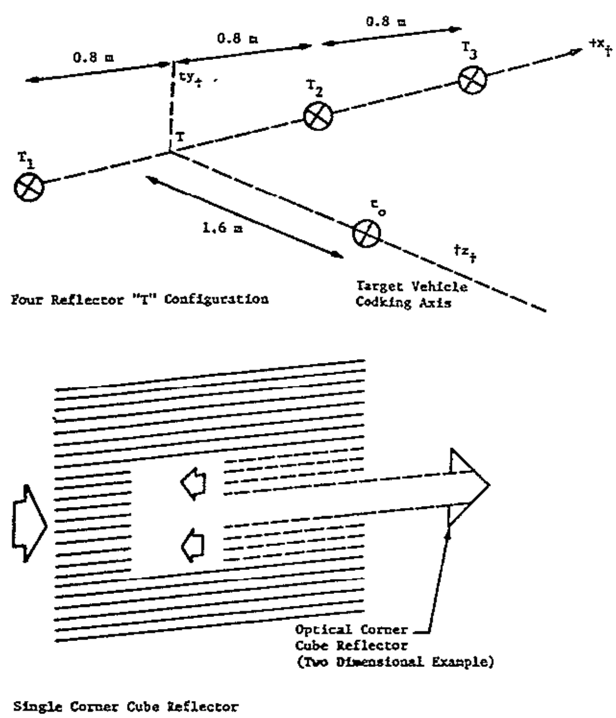

Figure VIII-21 The T Configuration for Retroreflectors

The receiver subsystem consists of receiving optics (including a narrow band optical filter), an image dissection tube, which acts as an optical demodulator, and a phase locked loop for estimation of the phase of the tone. The instantaneous FOV of the receiver is scanned over the variable FOV of the system in increments of 3 mr (Figure VIII-22). When a target retroreflector is within the instantaneous FOV, the signal out of the ID tube exceeds some preset threshold and a scan stop command is initiated. At this time, the signal is input to an appropriate phase locked loop for estimation of the phase of the tone.

The 75 mHz output from the PLL is then heterodyned to derive a 75 kHz signal with the same phase. The phase of the 75 kHz derived signal is then compared to the phase of a 75 kHz reference signal using a digital technique. The zero crossing of the reference signal initiates a start clock pulse and the zero crossing of the received signal stops the clock count. The number of clock pulses, each having a period of about 7 nanoseconds, is then directly proportional to the range of the target. This time-expansion technique is used to derive more accuracy. Angle to the target is derived by measuring the current in the coils of the image dissector. Complete relative position and attitude in-

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formation is derived by measuring the range and angle to each of three retroreflectors mounted on the target. These image and angle measurements are then processed by an onboard computer.

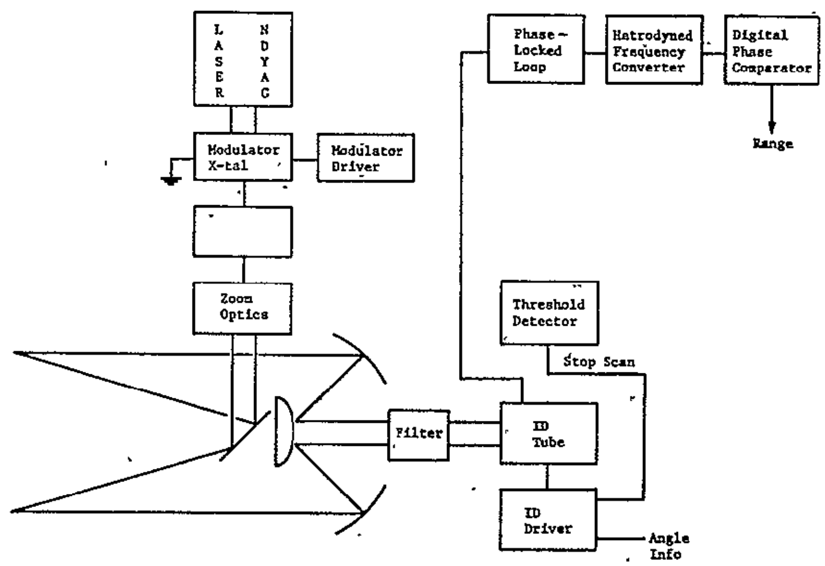


Figure VIII-22 Lockheed Optical Docking Sensor

Group G: Optical Rangefinders

1. RCA Optical-Docking Sensor - The RCA optical docking sensor utilizes the apparent angular separator of several retroreflectors placed on the target vehicle at known positions to be determine the range and relative attitude of the two vehicles. The system was designed for the terminal phase of the docking mission (2000 ft - 0). The sensor consist of a GaAs laser illuminator, receiver optics consisting of a 10:1 zoom lens, a silicon vidicon, a video processor, a local processor, and other electronics associated with the optics and camera (Figure VIII-23).

The GaAs illuminator transmits light in the .7 m portion of the spectrum towards the target vehicle. Each corner cube retroreflector then returns a portion of the light energy towards the receiver optics bypassing the illuminator, which is physically small. The optics focuses the image of the retroreflectors on the photo surface of silicon vidicon cameras. The camera and the video processor then converts the optical signal into digital information concerning the location of each point source (X_i, Y_i coordinates). The local processor then uses this information

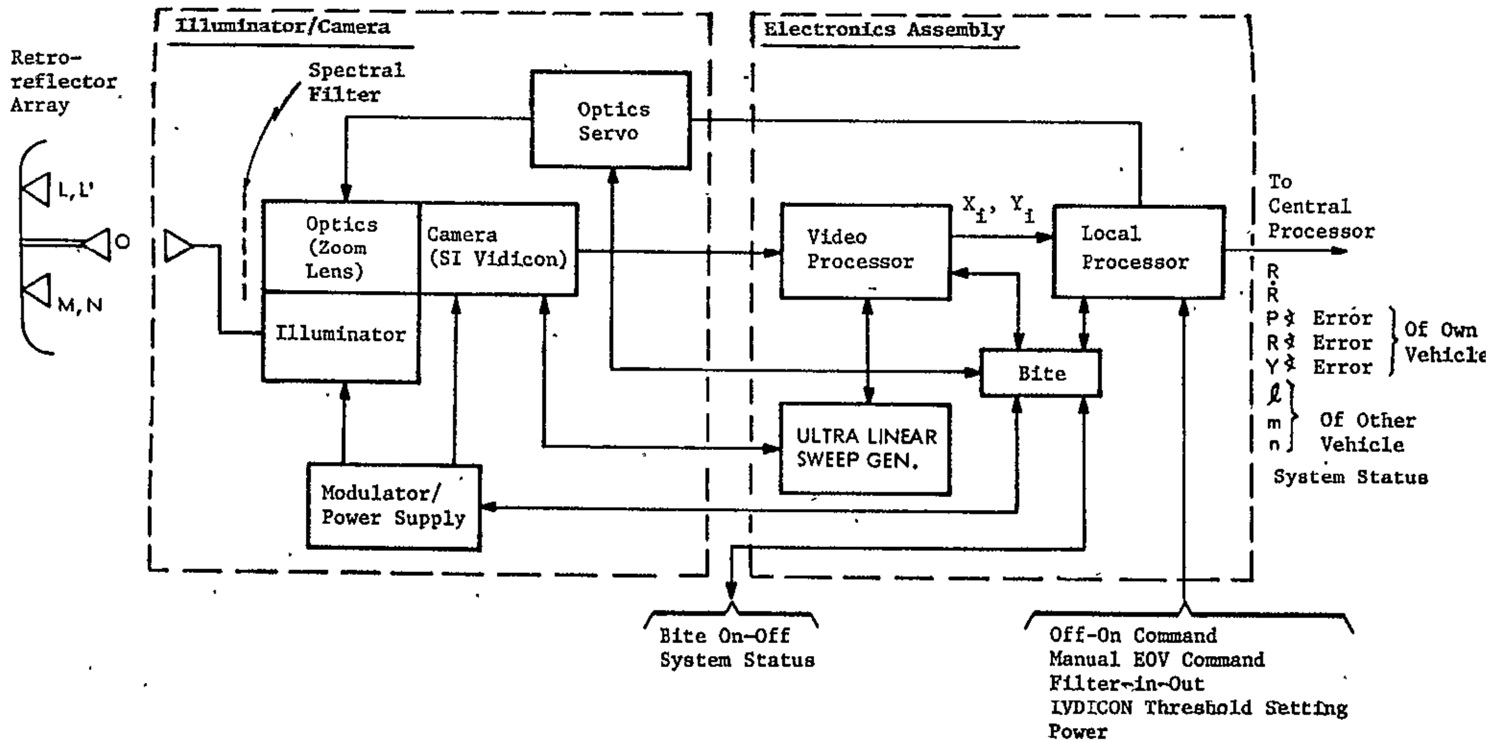


Figure VIII-23 Docking Sensor Functional Block Diagram to determine range and attitude errors. Using a 10 N baseline separation for the target reflectors, the accuracy obtainable with this system is 7% for range and 4% for attitude errors.

The RCA system is a relatively simple configuration in that no sophisticated innate dissector or laser scanner is required. In addition, the sensor could be used as a video monitor for inspection purposes. However, the useful range is severely limited by the illumination power. As a result, the usefulness of the system would be limited to missions such as TRS or perhaps manned maneuvering unit or TV maneuvering unit.

2. Automatic Stereoscopic Camera - Martin Marietta developed an automatic stereoscopic camera (ASTCAM) in the early 1970s to be used for the final docking phase of missions. The system consists of two cameras (either CCD or vidicons), optics to converge to FOVs of the cameras, and a video processor (Figure VIII-24). ASTCAM is capable of determining the range to a remote object through the use of paralax. The geometrical configuration of the setup is shown in Figure VIII-24.

Advanced Concepts

Several rendevous systems are presently in the conceptual or early design phase of development and lend themselves to either full or partial automation. These systems range from the more traditional RF type systems to optical systems where algorithms are used to process the image data.

1) FM Doppler Radar

Cursory examination of Doppler systems indicates that a frequency in the region of 13.3 GHz (Ke) is commonly used for range and velocity radars. Various components that have had sufficient use to provide reliability data in this band are readily available, and development costs of new components would be avoided. In addition, available components influence the type of system modulation to be used.

Ranging Doppler systems normally employ one of three possible types of modulation;

1) Sinusoidal FM (Bessel);

2) Linear FM, either triangular or sawtooth;

3) ICW.

ICW is simply a form of pulsed radar. If a solid-state transmitter is to be used. CW would be preferred over pulse, to limit peak power and voltage on the solid-state device. Some form of FM is therefore desirable.

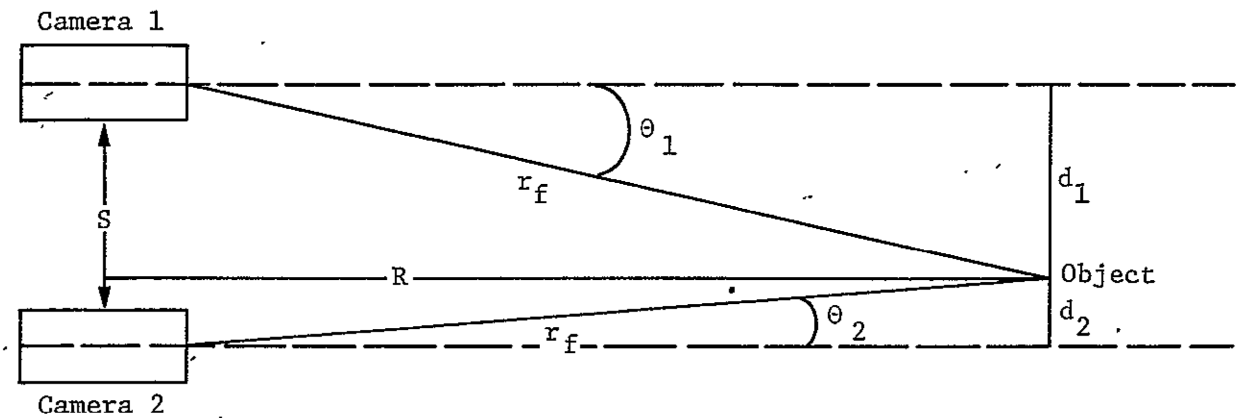


Figure VIII-24 Geometry for ASTCAM

Where:

θ_1 - displacement angle of object as seen by camera 1

θ_2 - displacement angle of object as seen by camera 2

S - physical separation of cameras 1 and 2

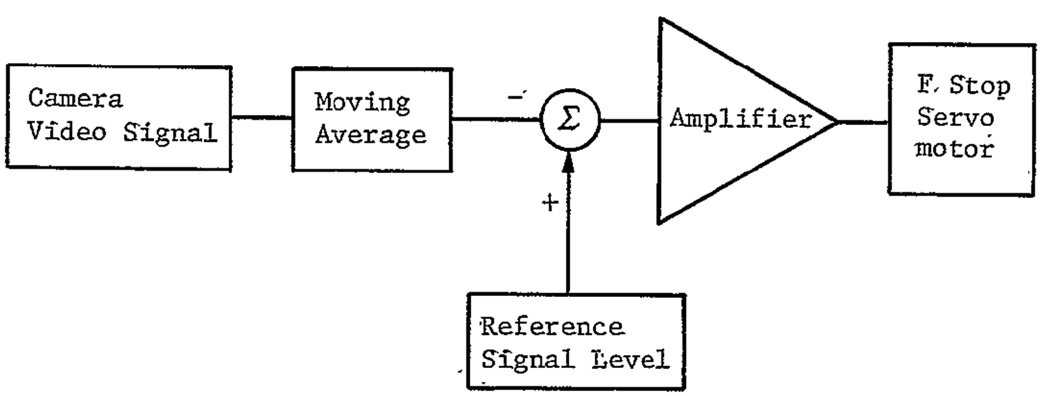
R - range of object

r_f & r_{f2} - focal distance from cameras to object

d_1 & d_2 - distance of object from the center line of each camera

One feature which is required to make the ASTCAM system feasible is an automatic focus mechanism. This may be accomplished by examining video signals of a single scan (Figure VIII-25). When the picture is out of focus (Figure VIII-25), the scan will show very little contrast, while the characteristic of a well-focused picture will be one of sharp contrast. By using a differentiator on the video signal, it is possible to obtain a measure of the sharpness. With this measurement, it is possible to maximize sharpness or contrast through the use of a servo on the lens focus adjustment.

Another camera function which must be implemented autonomously is the f stop setting. This function may be accomplished by taking the average light reading the camera, comparing it to a reference level and adjusting the f stop on the lens for each camera until the two levels are the same. The following diagram illustrates the system.



This system would be a good candidate for short-range missions where a video overview by astronauts is required. Such missions might include TRS, MTV, MMU and others.

Group H: Advanced Concepts

Several rendezvous systems are presently in the conceptual or early design phase of development and lend themselves to either full or partial automation. These systems range from the more traditional RF type system to optical systems where algorithms are used to process the image data.

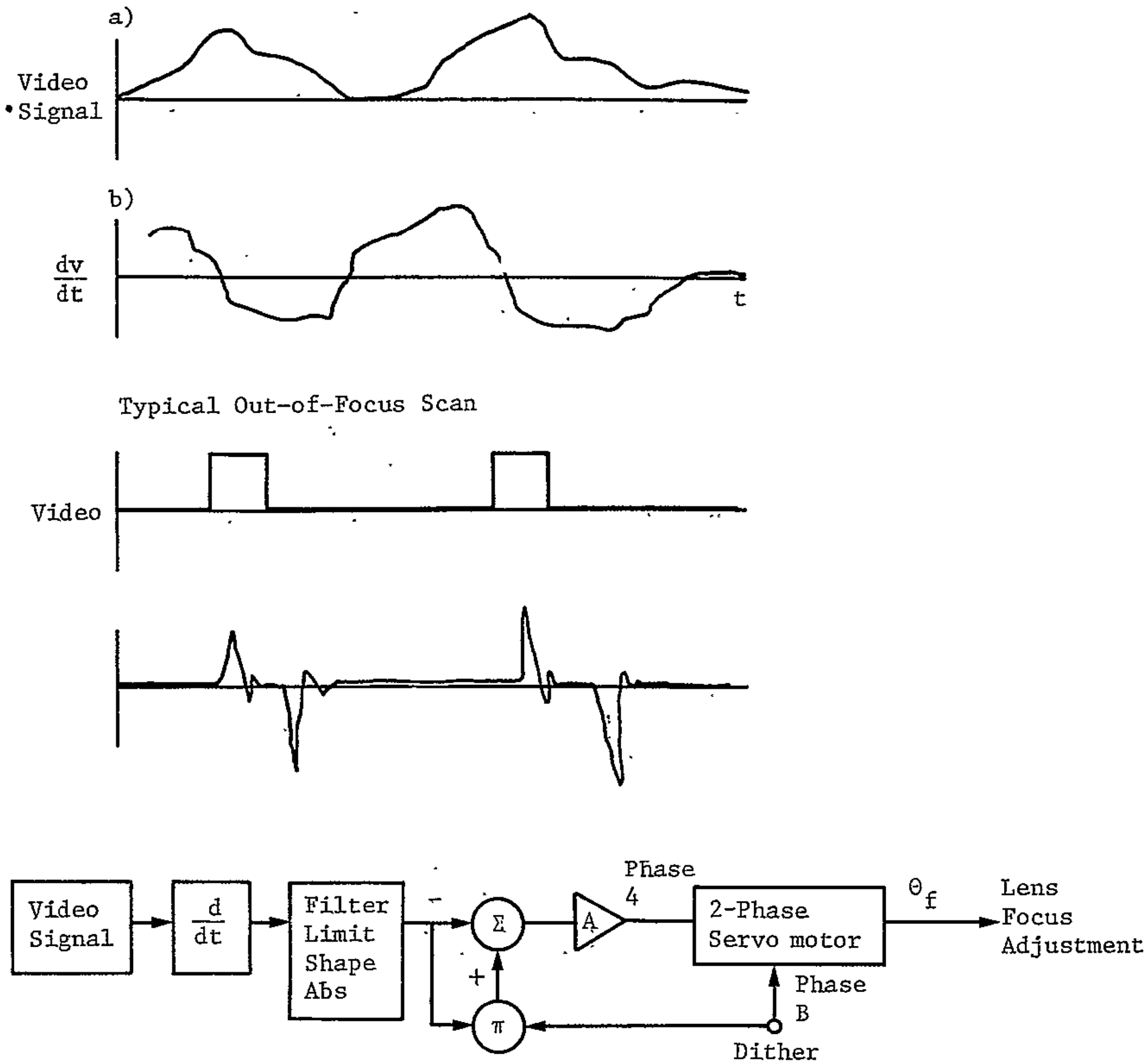


Figure VIII-25 Automatic Focus

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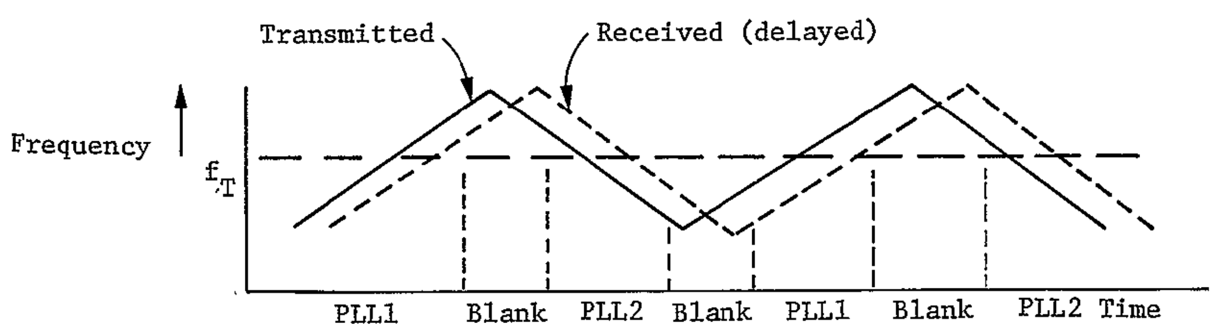
ICW is simply a form of pulsed radar. If a solid-state transmitter is to be used, CW would be preferred over pulse, to limit peak power and voltage on the solid-state device. Some form of FM is therefore desirable.

Sinusoidal FM is easily generated, but is difficult to process because of the nonlinear change of frequency. Of the linear FMs, triangular FM offers advantages over sawtooth FM:

1) The spectral spread is less for a triangle than for a sawtooth;

2) The triangle produces a spectral output that is easily processed to produce the desired range and velocity information, as explained in the following paragraph.



Let the difference in frequency between the transmitted ramp and the received ramp, which has a range delay, be defined as f_R.

During the upramp, the received frequency is $f_{Doppler} - f_R$. During the downramp, the received frequency is $f_{Doppler} + f_R$. Two time-gated phase-locked loops may be independently locked to the two received frequencies. The processor then simply adds and subtracts the two output frequencies to obtain twice the ranging frequency and twice the Doppler frequency. The simplicity of processing makes triangular FM very attractive.

The phase-locked loops of the dual-frequency tracker exist in both analog and digital form from many sources. The data processor would be a very simple form of microprocessor or T^2L logic, both of which exist in space-qualified forms.

At 13 GHz, the transmitter could be an IMPATT diode or a Gunn oscillator. Space-qualified IMPATT diodes are available, but are quite expensive. Gunn transmitters are used in commercial equipment such as police Doppler radars, but are probably lacking in reliability and/or uniformity. Because of the carefully machined cavities required, the transmitter will probably be the most expensive component of the radar.

The type of antenna to be used will require study because there is a tradeoff between beamwidth and gain. A beamwidth of about 20° is probably desired because a narrow beam would be difficult to point, and a wide beam would have low gain. Various types of waveguide or stripline phased arrays are already designed or could be easily designed. Mission requirements could well dictate a design. For example, if angle tracking becomes a requirement, to maintain accurate pointing on a tumbling target, the resultant four-channel monopulse could well dictate antenna design, as well as requiring four matched receivers.

Figure VIII-26 shows a possible triangular-FM design. The recommended transmitter would have an IMPATT diode, followed by a varactor diode phase modulator, each in machined cavities. The combination of these two elements can be made to produce a very linear FM ramp with a linear driving voltage, even though each element by itself has a nonlinear function of frequency versus voltage. Ramp modulation is required to obtain ranging information. Rf power output would be about 250 mW.

The triangle generator (ramp modulator) would consist of a squarewave multivibrator or driven FF followed by an integrator. This type of generator produces the least possible residual noise.

The lowest-cost antennas would probably be dual stripline phased arrays. If monopulse is required, considerably greater complexity will result in both the antennas and the receiver.

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The lowest-cost antennas would probably be dual stripline phased arrays. If monopulse is required, considerable greater complexity will result in both the antennas and the receiver.

A coupler would provide a low-level sample of the transmitter output to an SSB modulator for generation of the receiver first LO. The offset (LO) oscillator is required to obtain zero-frequency Doppler response.

A superheterodyne receiver is recommended rather than a homodyne in order to reject the low-frequency mixer noise that would otherwise enter the zero-frequency Doppler region. A first IF of 30 MHz would be suitable because many well-designed amplifiers exist, and image rejection is not of concern. The first IF amplifier needs only enough bandwidth to pass range and velocity offset components and allow for any drifts of the IFLO because the FM ramp is effectively removed in the first mixer. The 30-MHz IFLO would have no particularly stringent specifications because the second IF is independent of the IFLO. The IFLO could be counted down to provide other required signals such as the data processor clock.

The second IF amplifier would be a high-audio amplifier passing roughly 9 to 11 kHz, with no particular specifications other than stable gain.

The dual-frequency tracker would contain the two time-gated phase locked loops. The PLLs would be common microminiature units like those available from the Viking design. The time gates would basically freeze the loop during the off time, so that each PLL tracks only a selected portion of the received waveform.

The data processor would add and subtract the two input signals to obtain independent range and velocity signals. Cycle counters would then determine range and velocity, which would then each be scaled for their independent displays. The data good signal would be derived from thresholds in the tracker.

If deemed worthwhile, another piece of information could be derived in the data processor and displayed separately, i.e., change of range with time, which is not to be confused with velocity. Proper processing could show the variation of range about the average range, which would therefore provide tumble as a cyclic output. Such information could be useful in preventing collisions with rotating members of the target.

Doppler accuracy of the triangular linear FM system is directly proportional to the ratio of FM bandwidth to center frequency. If FM bandwidth were 50 MHz, and the center frequency were 13,300 MHz, accuracy would be 1 part in 266 or 0.376%. Quick calculations indicate that S/N will always be so high that front-end noise effects will be obscured.

Range resolution is proportional to FM sweep rate. A linear section of the sweep must be longer than the round-trip ranging time, which sets a minimum on sweep duration. Therefore, to increase sweep rate, total sweep bandwidth must be increased. If triangular FM is to be used, study will be required to select the optimum waveform.

2. FM/CW Harmonic Docking Radar - The FM/CW Harmonic Docking Radar was designed to provide range, range rate, and angle data during the final 100 meters of a mission and is immune to spurious reflections and clutter effects. The system employs a passive target enhancement device that uses a frequency doubler to translate the carrier frequency from X-band to Ku-band, so all reflections from the target vehicle at the fundamental frequency are rejected by the radar receiver. This approach also eliminates transmitter-receiver isolation problems normally encountered with single-antenna FM/CW radars. A five-horn monopulse system is employed in the receive mode to provide the angle information. It is estimated that a system weighing 10 lb and requiring 10 W of prime power could be built using current state-of-the-art techniques.

Figure VIII-25 is a block diagram of the reference channel of the docking radar. It is a linear-sweep FM/CW radar that uses a harmonic generator in the target vehicle to translate the received signal to twice the frequency of the docking radar transmitter. The homodyne radar receiver is then tuned to this harmonic so reflections from the target vehicle at the fundamental radar frequency are automatically rejected. A frequency doubler was employed because it is nonradiating, requires no power supply, and at short ranges incurs minimum signal loss to the incoming signal from the radar. The major advantage of this system is that all clutter is eliminated because any such reflections from the target vehicle cannot produce echoes at the second harmonic of the transmitter frequency. The system is also impervious to multiple reflections and rejects strong reflections from the target vehicle that do not originate at the docking port, because these echoes also occur at the fundamental frequency.

The radar is an all-solid-state sensor operating at X-band and Ku-band (8 and 16 GHz). The solid-state transmitter consists of a varactor-tuned Gunn oscillator frequency-modulated over a 50-MHz range at a modulation rate of 3 kHz. Good linearity is easily achieved over this range, and all radar components will operate satisfactorily over this bandwidth. As large a bandwidth as possible is desirable to minimize the basic error in the range measurement. The 50-MHz frequency excursion yields a step error of 75 cm, which is satisfactory for the docking radar and still allows design of all RF circuitry within current state-of-the-art constraints. Furthermore, as long as there is relative motion between the chaser and the target vehicle, continuous shifting of

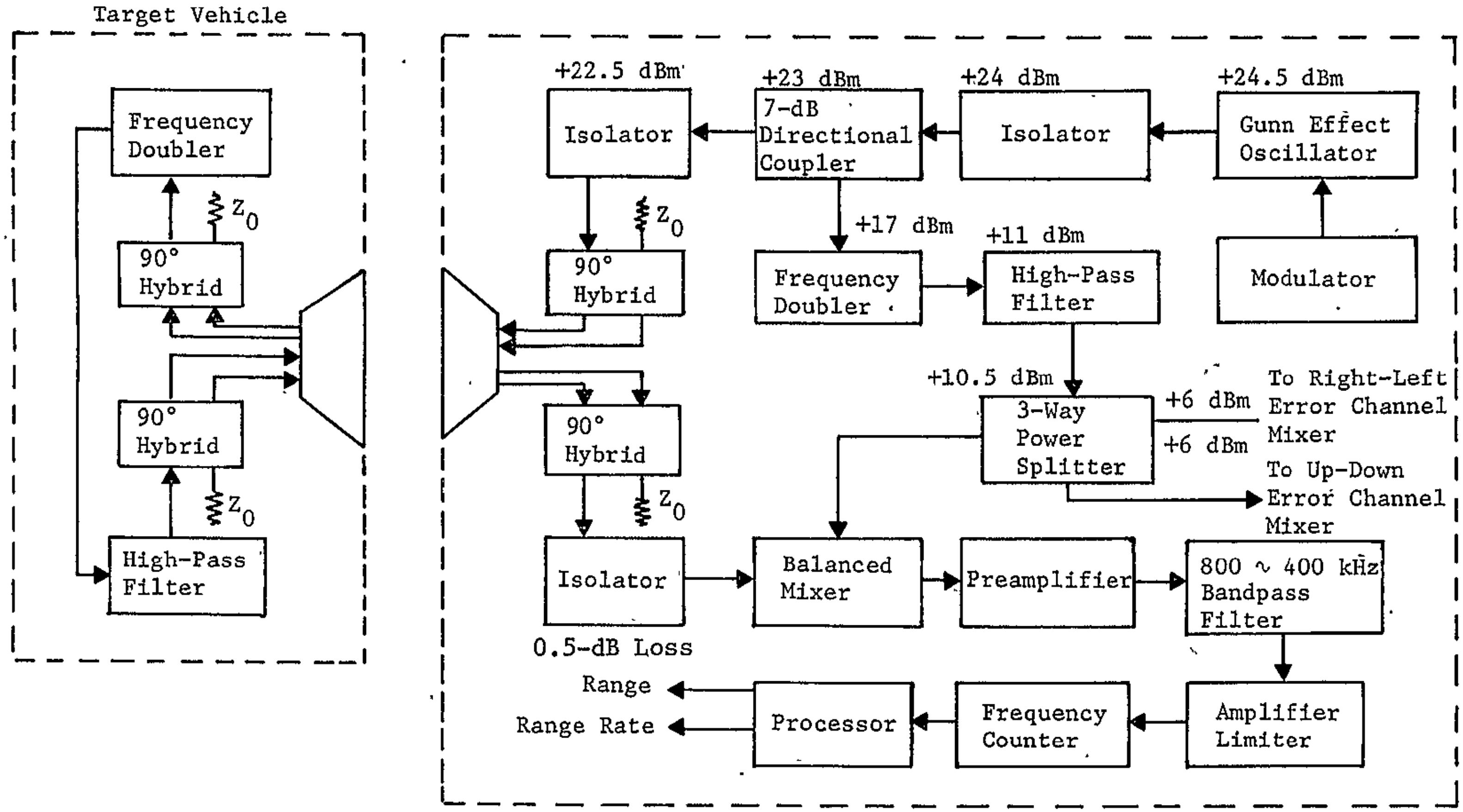


Figure VIII-25 FM/CW Docking Radar Reference-Channel Block Diagram the RF phase will average the step errors, and the absolute error in the range measurement will be reduced to about 20 to 25 cm.

The output from the 275-mW transmitter is fed through an isolator to a 3-dB directional coupler, where a fraction of the transmitted signal is used to drive a frequency doubler. The frequency doubler operates at a relatively high level (+17 dBm) and supplies the local oscillator drive required for the reference and error channel balanced mixers. Because 12 mW is required to drive the three balanced mixers, a gallium-arsenide varactor doubler is employed to provide a conversion efficiency of about 25% and a bandwidth of about 10%. The output from the directional coupler is fed through an isolator to a broadband circularly polarized antenna that must be capable of transmitting the X-band signal and simultaneously receiving the Ku-band signal from the target vehicle. A similar antenna is employed at the target vehicle except that it is a lower-gain antenna to provide a reasonably broad beam pattern. The harmonic signal is then received at the radar antenna and fed through an isolator to the reference-channel balanced mixer. Output from the mixer is fed through a preamplifier, an 800-Hz to 400-kHz bandpass filter, and an amplifier limiter to a frequency counter and a processor. Processor circuitry also derives a voltage proportional to the first derivative of range, so the reference channel of the radar provides simultaneous data on range and range rate.

Figure VIII-26 is a block diagram of the error channel and the monopulse feed system required for obtaining angular information.

Because of its superior performance and ease of implementation, a five-horn monopulse system is employed in preference to the more conventional four-horn monopulse system. Each of the five horn radiators uses a 90° hybrid to receive a circularly polarized wave in the square horn antenna. Identical 180° hybrids are employed to connect the two azimuth-channel horn antennas and the two elevation-channel horn antennas, thus producing the familiar null-type difference patterns in the error channels. Because the receiver operates at the second harmonic of the transmitter signal, coupled power from the active center horn is automatically rejected by the receivers, thus avoiding one of the major problems normally encountered with single-antenna FM/CW radars. The output from each error channel mixer is fed through a preamplifier, bandpass filter, and amplifier limiter to a phase detector. A signal from the reference channel is also fed to the phase detector. The output from the phase detector is then an error signal whose magnitude is proportional to the angular error and whose sign is determined by the direction.

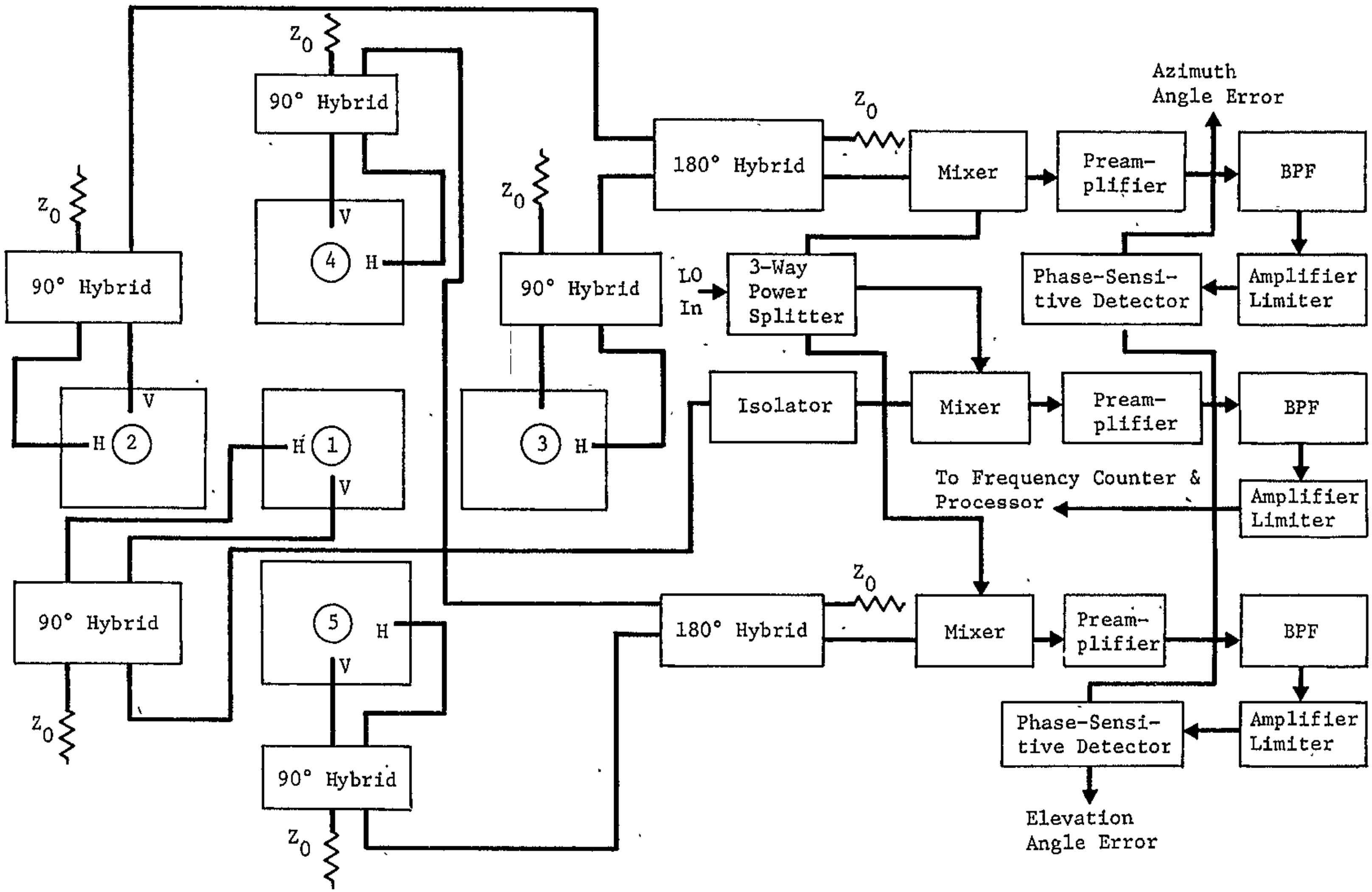


Figure VIII-26 FM/CW Docking Radar, Five-Horne Amplitude-Comparison Monopulse System Block Diagram

3. Cubic SR-100/ELF System - This system is a combination of cubic corporations CR-100 and ELF-3 systems and was conceived for use on large space structures. The former system provides determination of range and range rate data in digital form and operates on the FM-CW Doppler Radar. The latter system, an electronic location finder, gives angular information in analog form and operates on the CW monopulse radar principle. Under the proposed configuration, the two systems will be combined into a single unit sharing antennas, some RF amplifiers, and a power converter-regulator. Given an onboard processor, the system will be able to provide range, range rate, angle, and relative attitude information about the two vehicles engaged in docking. Relative attitude is determined by ranging to three transponders on the target vehicle.

The CR-100 (Figure VIII-27) measures range through a CW phase comparison technique whereby the interrogator measures the phase shift of an electromagnetic wave after a round-trip between the chaser and target vehicle. This technique yields range accuracy to .5% which will be very useful in determining relative attitude. The system measures range rate by determining the Doppler frequency shift of the carrier and is accurate to within .024 ft/sec.

The ELF-III system (Figure VIII-28) measures the line of sight angle to several transponders mounted on the target vehicle through an interferometer technique. The system consists of several pairs of antennas placed orthogonally to one another. Two pair of antennas are used to resolve coarse angle information and eliminate ambiguities for the five angle determination antennas. A wave incident at some angle creates a phase differential between a pair of receiving antennas. The receiver then processes this phase information to yield the incident angle to an accuracy of .1 degree.

The CR-100/ELF system provides good range and range rate accuracies as well as angular information to several transponders on the target vehicle. With this information, an onboard processor will be able to determine all the parameters necessary for an autonomous rendezvous and docking sequence. The system, as it was presented, has two drawbacks for certain missions. First, the minimum range in the existing system is 20 feet. However, this can be significantly reduced by reducing transmitting power as the range decreases. Also, the ELF system has a limitation in accuracy toward the edges of the field of view. For a small target, this is not a problem, but for Large Space Structures, the limitation is intolerable. For this reason, it was suggested that the Cubic TDAME system be included instead of the ELF. The TDAME

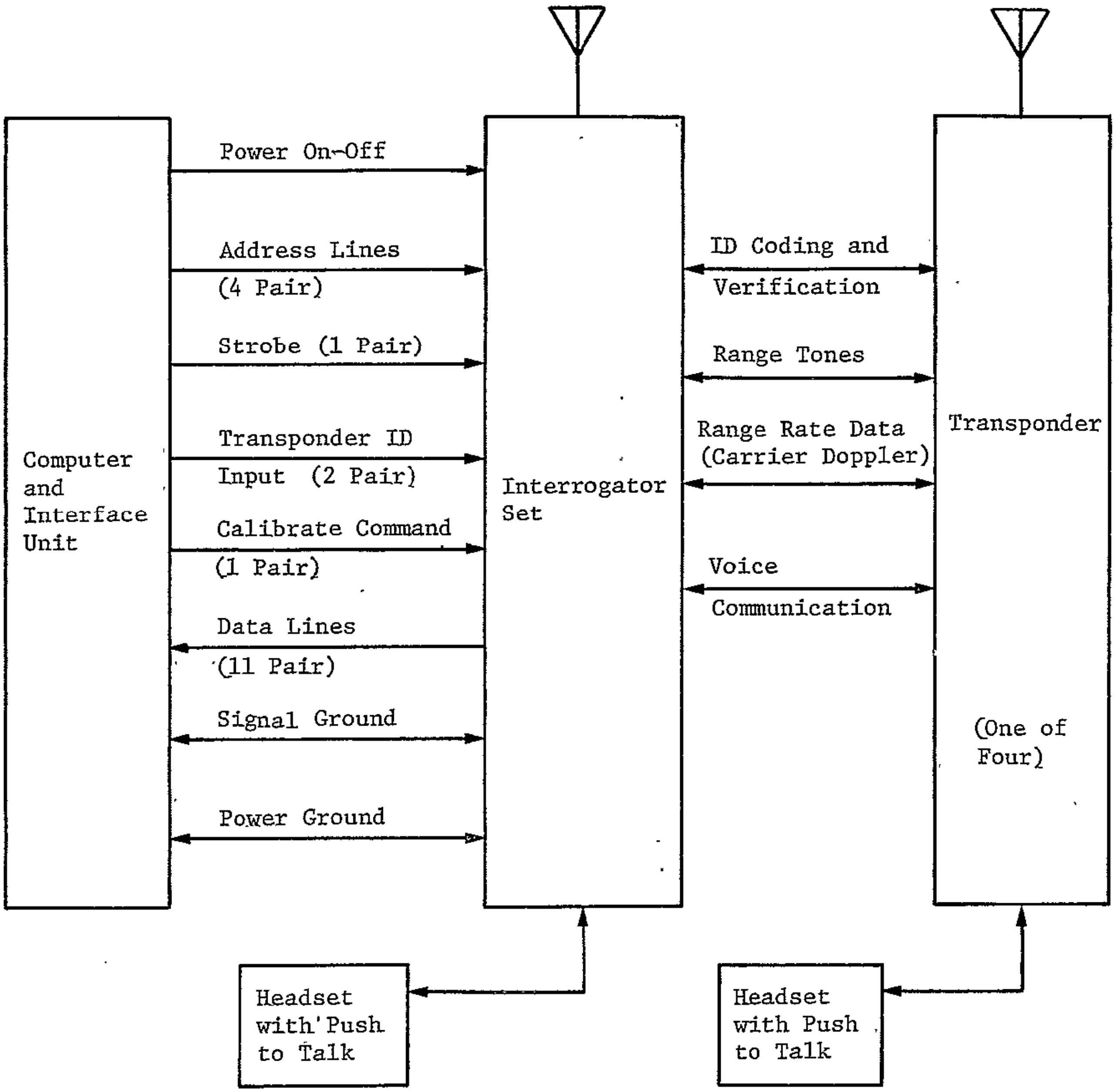


Figure VIII-27 System Block Diagram, CR-100 Range/Range-Rate System

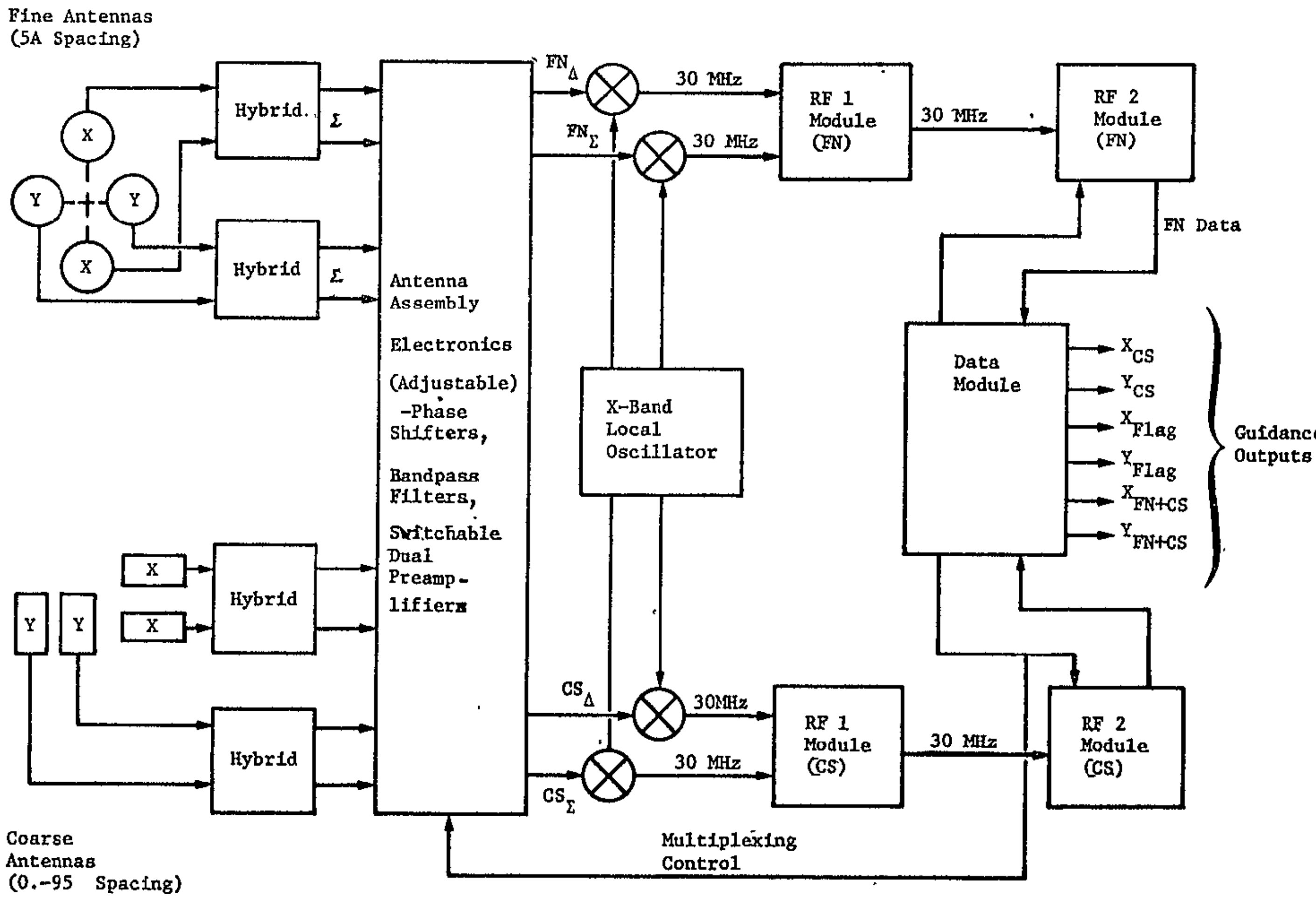


Figure VIII-28 System Block Diagram Rocket-Borne Elf III Tracker system is similar to the ELF except it incorporates a longer baseline, which is used to obtain better resolution toward the fringes of the field of view.

4. TRS Proximity Rendezvous System - The Teleoperator Retrieval System (TRS) is a small unmanned transport vehicle that will be deployed from Shuttle. It will be controlled by man in the loop onboard Shuttle and will be able to dock with vehicles in orbit or place vehicles in orbits not obtainable by Shuttle. Its first proposed mission will be the reboost of Skylab into a safer orbit. (See Figure VIII-29).

The rendezvous system proposed for TRS originally consisted of a TV camera mounted on the vehicle and a man in the loop watching a video monitor onboard Shuttle inputting commands through hand controllers. During simulation of the system, it became apparent that the astronaut may require range-rate information to be consistent. If such information is required for a safe docking, it was proposed that the Viking Terminal Descent Landing Radar (TDLR) be added to the baseline configuration to provide the astronaut with range-rate information. The TDLR was selected because of the time constraint involved with the construction of TRS. The TDLR is already space qualified, and, because a backup Lander was produced, one system is readily available. However, several modifications must be made to make the unit compatible with TRS. The use of the TDLR on TRS by no means implies that the system is a candidate for the proposed application; it is overdesigned, does not provide range information, and cannot provide any relative attitude information.

The TDLR is a third-generation landing radar design built by RYAN. The first two generations were flown on the Surveyor and Lunar Module landing radar programs. The TDLR is a CW Doppler radar that uses separate transmitters for each of four beams, dualbeam array antennas, and a stripline microwave receiver. For the system to be used on TRS, the antenna array must be modified and the size cut down within required limits. Because of the cost of the TDLR and the design modifications, it will probably not be used on TRS systems built in the future but may be used only for the Skylab reboost mission. One of the techniques developed for an advanced TRS rendezvous system is aimed at decreasing the bandwidth requirements of the video link and is described below.

The need for a video data link in the TRS rendezvous system can be eliminated by transmitting only enough pertinent data to reconstruct a graphical picture of the target vehicle. The high

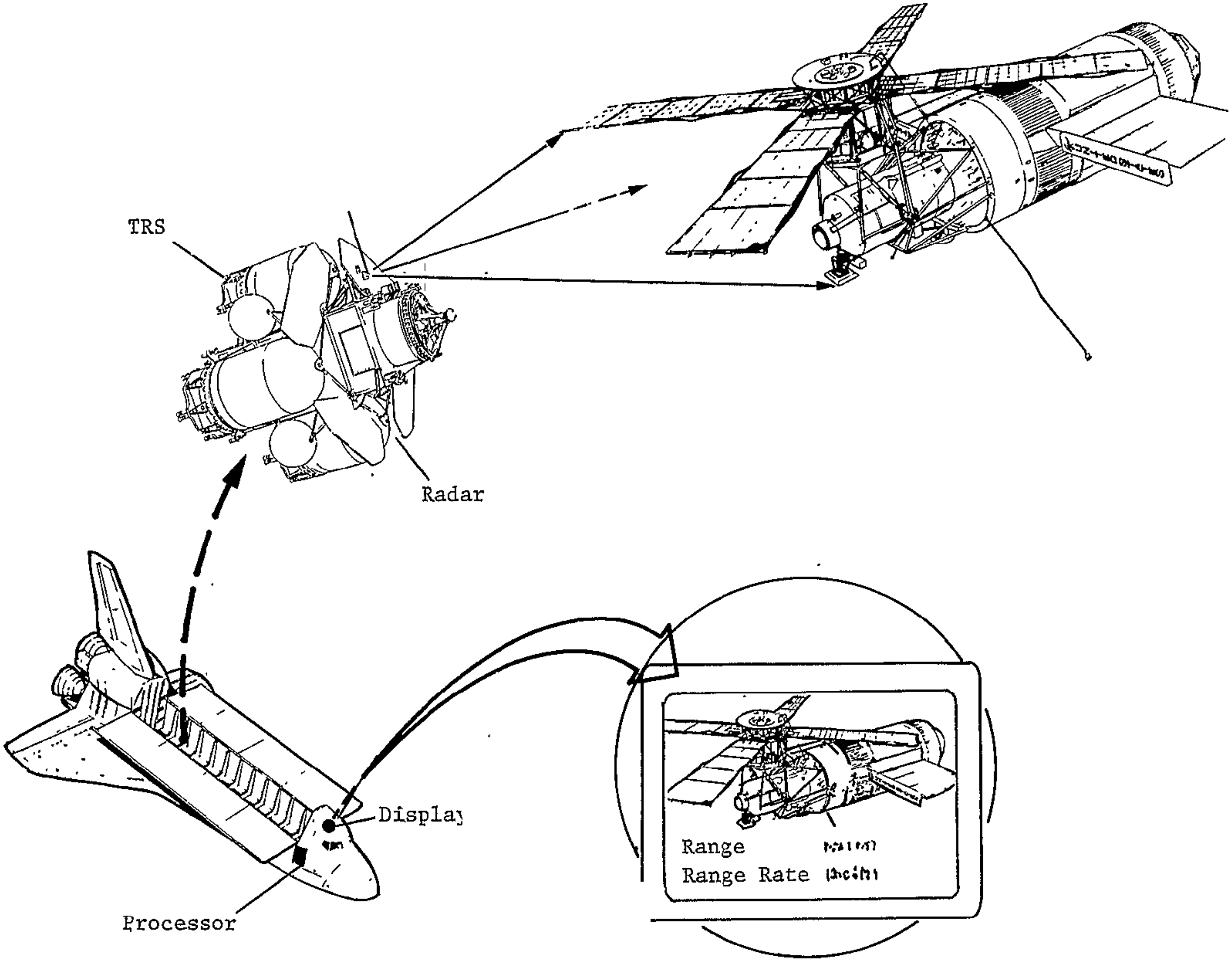


Figure VIII-29 TRS Proximity Rendezvous System rate video data link could therefore be replaced with a low bit rate TM link, thus increasing the maximum transmitting distance between TRS and Shuttle. The total relative navigation system would consist of a range and range rate subsystem and a video camera onboard the TRS. Docking is performed by a man in the loop positioned in the Shuttle or on Earth. The astronaut views a graphical representation of the target vehicle on a CRT and uses a set of hand controllers to input translational and rotational commands to a TRS. Range and range rate information is displayed in alpha numeric form on the lower portion of the screen. The video data displayed has been transmitted to the orbiter from TRS over a low data rate conventional TM link and processed by the Shuttle computer to derive the graphical representation. The low bit rate video data is generated by using the TRS camera to interrogate arrays of light emitters arranged on the target vehicle to produce signatures unique to that side of the target vehicle figure. The camera will be used in such a way as to respond <u>only</u> to these light emitters and produce no signal when the scan does not "see" an emitter. Several techniques for extracting only emitter signature data from full scan systems are available (e.g., the use of color emitters, a color wheel camera and simple gating of scan locations of signal with respect to sync. pulse). Software, at the receiving end, to drive computer graphics from minimum data must be developed, but is practical. Since this technique also produces relatively accurate relative attitude information, the data can be used back at the Orbiter to drive a dynamics computer graphic display of a detailed line drawing of the target vehicle. It is this display which the astronaut actually sees on the CRT. The application of this idea will <u>remove</u> the limitation on the range at which rendezvous and docking using man-in-the-loop remotely viewing a video scene can be employed. The new parts of the idea are: (1) the replacement of the video picture with a line drawing on a CRT which contains all the information of the video picture required for rendezvous and docking; (2) the transmission of all the necessary data for a high frame rate video image at KBPS rates, <u>not</u> MBPS rates.

5. Feature Identification Docking Operation (FIDO) - A number of breadboard systems related to rendezvous and docking that use TV cameras and video signal processing to obtain range information have been developed. In such systems, accuracy depends on the distance to the target--very good accuracy is obtained at close range but not at distances approaching 1 mile. However, features such as a video display, direction information (pitch and yaw angles to the target), and target orientation can

be provided by these systems with very little added hardware. Moreover, the systems can be lightweight and consume little power. They are presented here to illustrate our experience with the problems associated with docking aids and automated rendezvous and docking technology.

The FIDO system (Figure VIII-30) was developed as an autonomous rendezvous and docking system for guidance of the grappler arms on a Large Space Structure (LSS). The heart of the system is the video centroid tracker described below, which uses an image-dissector TV camera as its sensor.

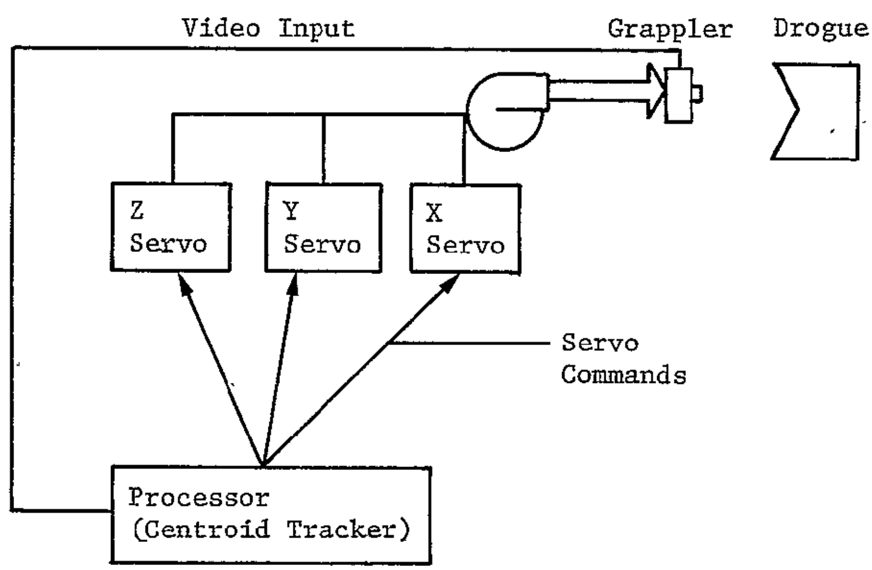


Figure VIII-30 Feature Identification Docking System

Centroid Tracker - Under contract NAS1-13558, "Video Guidance, Landing, and Imaging Systems", a rendezvous and docking experimental system involving a TV camera and electronics mounted on a three-translational-degree-of-freedom servo-controlled simulator was developed. A PDP-9 computer commanded the camera scan position and camera location with respect to the target and sampled the video signal from the camera.

Using this apparatus, algorithms were developed and tested for determining target centroid location with respect to camera boresight, angular orientation of the target, target area, and ranging. For these experiments, a light-colored target against a black (space) background was assumed.

For coarse ranging and target-area determination, the system counted the number of pixels in a raster-scanned scene whose brightness exceeded a threshold. In a flight-instrument case, this would be accomplished by integrating the thresholded video signal over one frame.

The study also demonstrated that fine ranging could be performed using the centroid algorithm and two separated views of the same object. This can be accomplished with two cameras (or with a single camera and mirrors) separated by an interocular distance d. The center of the object is found by the centroid algorithm, and the range is given by

$$R = \frac{df}{\bar{X}_L - \bar{X}_R}$$

where f is the focal length and X_L and X_R are the centroids referred to the focal planes. For moderate range variation, gimbals are not required. Greater accuracy can be obtained by using one camera as a centroid tracker and the other as a TV area correlator using the output of the first camera as a reference scene.

These techniques are readily adapted to logic-circuit implementation using solid-state cameras, and they provide a considerable amount of pointing and tracking capability in a samll, lightweight, inexpensive package. However, accuracies comparable to those of the system that is the subject of this proposal are obtainable only at close range.

6. Bio-Optic Correlation Control Scheme (BOCCOS) - This scheme uses man in the control loop of a remotely located TV guided vehicle to update the physical position of the vehicle with respect to a target. This information is sent to a computer, evaluated, and processed into control signals that are sent to the vehicle. The scheme features mutual comparison of image size and position for developing error signals that are used for closed-loop steering commands.

A TV presentation of an appropriate target is transmitted from the remote vehicle to a local monitor on which either a physical or electronic overlay can be manually positioned. The overlay provides relative six-axis information between vehicle and target to be used to generate control signals for positioning the vehicle.

Referring to Figure VIII-31 a vehicle with an on-board TV camera sends video information of a target to a local monitor, which provides both a wide-and narrow-angle presentation. Two overlays, either physical or electronic, are positioned manually by the operator as he correlates them with the TV image. The X, Y, and θ coordinates and size of the overlays with respect to a reference reticle on the monitor are picked off either mechanically or electronically, and processed by a computer to provide control commands that are sent to the vehicle to move the vehicle in relation to the target image on the monitor, thus providing a closed-loop control system.

This scheme could greatly simplify manual control of a remotely positioned vehicle by reducing the hardware and coordination needed by the conventional control loop of joy-stick-to-actuators-to-vehicle-to-video presentation of target-to-man-to-joystick. The system takes full advantage of natural human attributes of recognition, correlation, manipulation, and coordination while the job of calculation, anticipation, and integration are taken over by the computer, to provide a control system that could considerably reduce the stress on the controller while also reducing the amount of fuel and time required for a given operation.

Simulations

A six degree of freedom simulator can provide a unique environment for testing advanced concepts and as those described above. By implementing the processes in software, elaborate breadboards can be avoided while allowing thorough evaluation of a concept. The TRS, FIDO, and Centroid tracker algorithms have been simulated in this manner and a description of the TRS simulation is included below for reference.

TRS Simulation - The TRS Skylab reboost mission has been simulated in several laboratories with the intent of giving astronauts a preview and training tool for the actual docking of the TRS to Skylab. In the simulations, only the last 60 ft--the most critical portion of the rendezvous and docking mission--are considered. Figure VIII-32 and VIII-33 show the TRS VGL docking simulation system. Figure VIII-34 is the component block diagram.

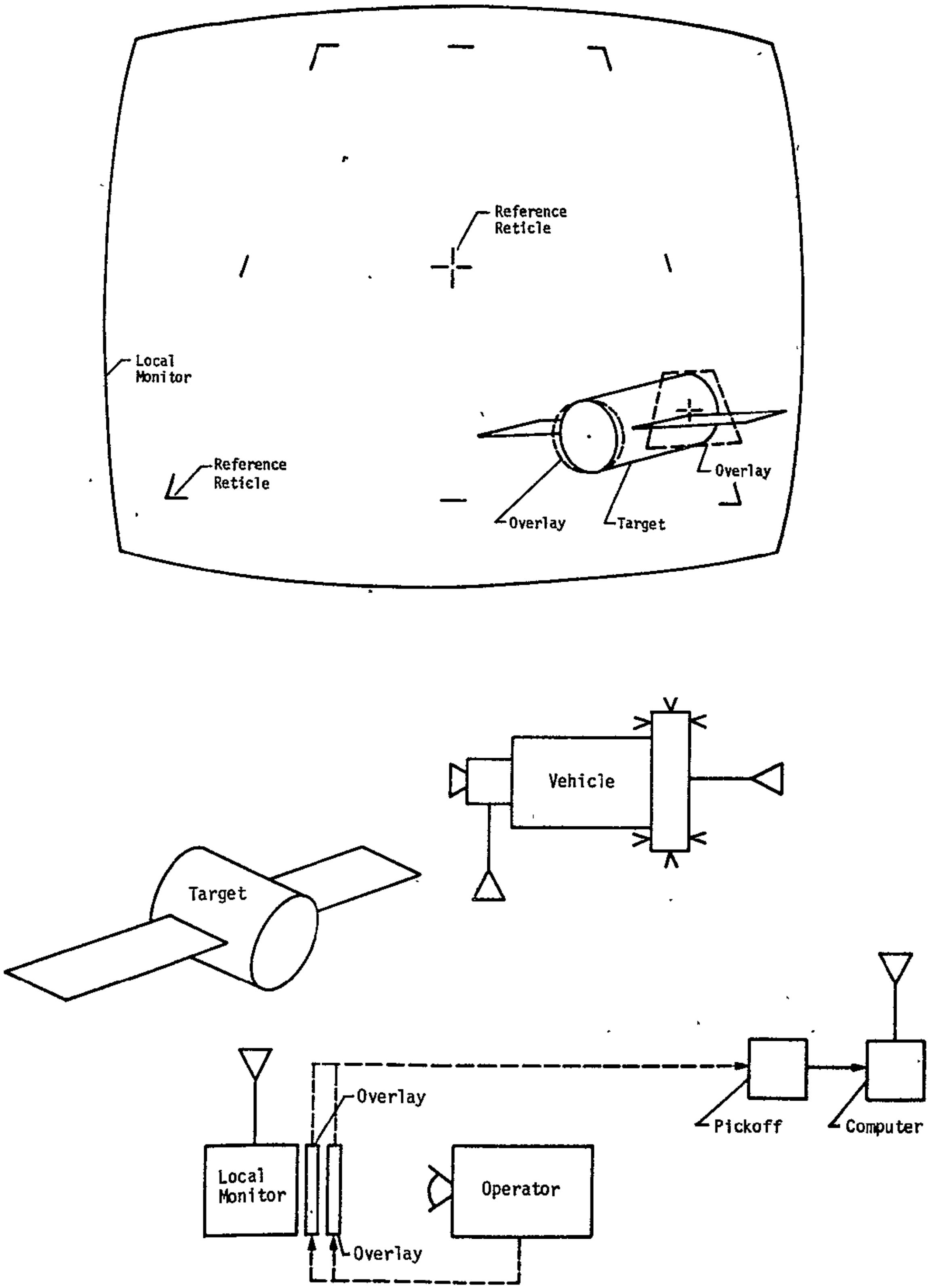


Figure VIII-31 Bio-Optic Correlation Control Scheme

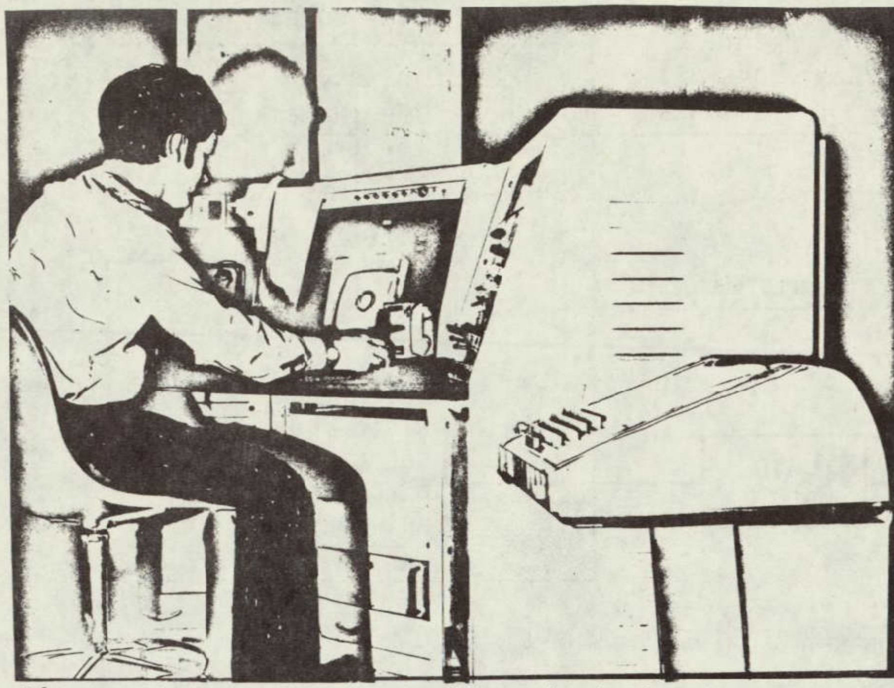


Figure VIII-32 TRS VGL Docking Simulation System Crew Station

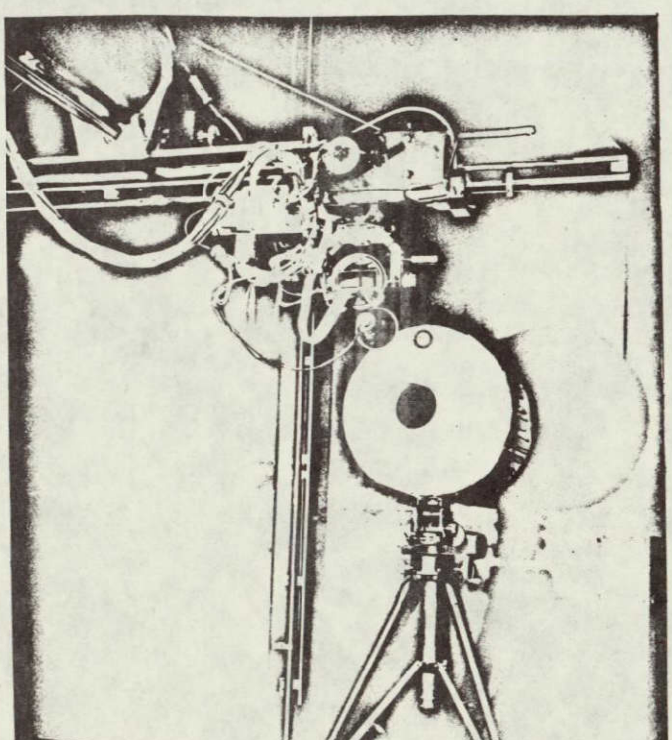


Figure VIII-33 TRS VGL Simulation System Physical Simulator

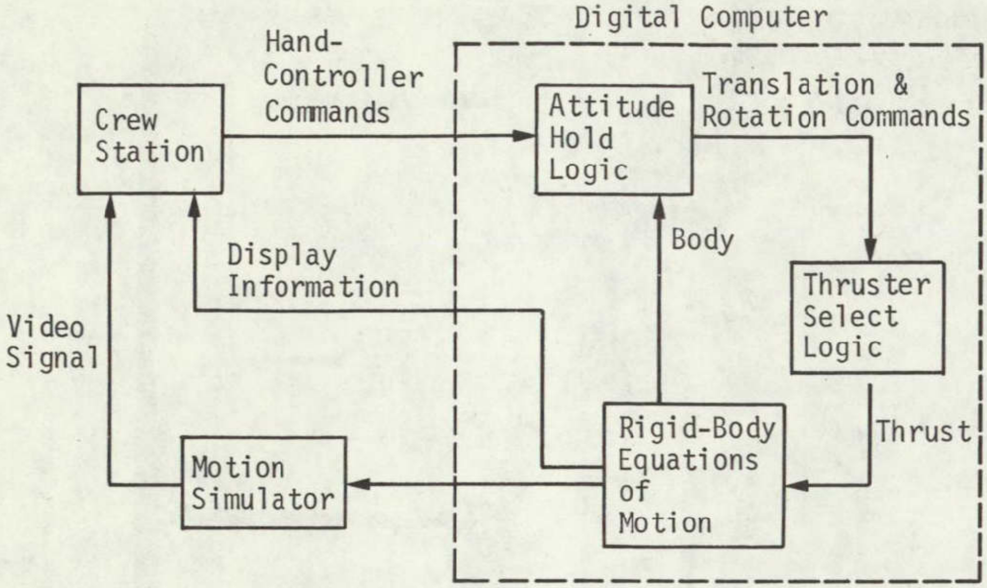


Figure VIII-34 TRS VGL Docking System Flow Diagram

Crew Station - The pilot sits at the crew station (Figure VIII-32), inputs TRS commands via hand controllers, and monitors the relative TRS/Skylab motion on a video monitor. The hand controllers consist of a translation controller (x, y, z) and a rotation controller (roll, pitch, yaw), and will control the thruster firing in the actual TRS. A button is provided on the rotational hand controller to initiate the attitude-hold or rate-hold logic. An audio cue is provided to indicate that a hand controller command is being given. Relative TRS/Skylab motion is fed back to a video display from a camera mounted on the physical simulation.

Attitude hold allows the user to hold the attitude of the simulated body fixed while translator maneuvers proceed. The logic can be best explained through the use of a phase plane diagram (Figure VIII-35). The dotted line in the figure portrays the attitude state of the vehicle; as its attitude increases past some predetermined critical value (A_1), the thrusters are commanded to be pulsed until the attitude rate has been reversed to a sufficient level determined by R_1. If the attitude continues to diverge from its desired value (A_0) and passes (A_2), the thrusters will be commanded to a full-on state until the attitude rate is brought to the desired level. In this method, the attitude of the vehicle can be controlled around any desired value (A_0).

Thruster Select Logic - Implementation of the thruster sleect logic is accomplished using the following algorithm with the commands as input:

$$NC = K_x + 3K_y + 9K_z + 27K_\phi + 81K_\theta + 243K$$

x, y, z are translational commands

ϕ, θ, are roll, pitch, yaw commands

K = 0 if command is zero

K = 1 if command is negative

K = 2 if command is positive

Notice that each weighing function (1, 3, 9, 27, 81, 243) is one larger than the maximum sum of the previous terms. This ensures that there will be 3^6 = 379 unique states corresponding to any combination of inputs. The output of this algorithm is processed to yield one of the 24 thruster states (on or off).

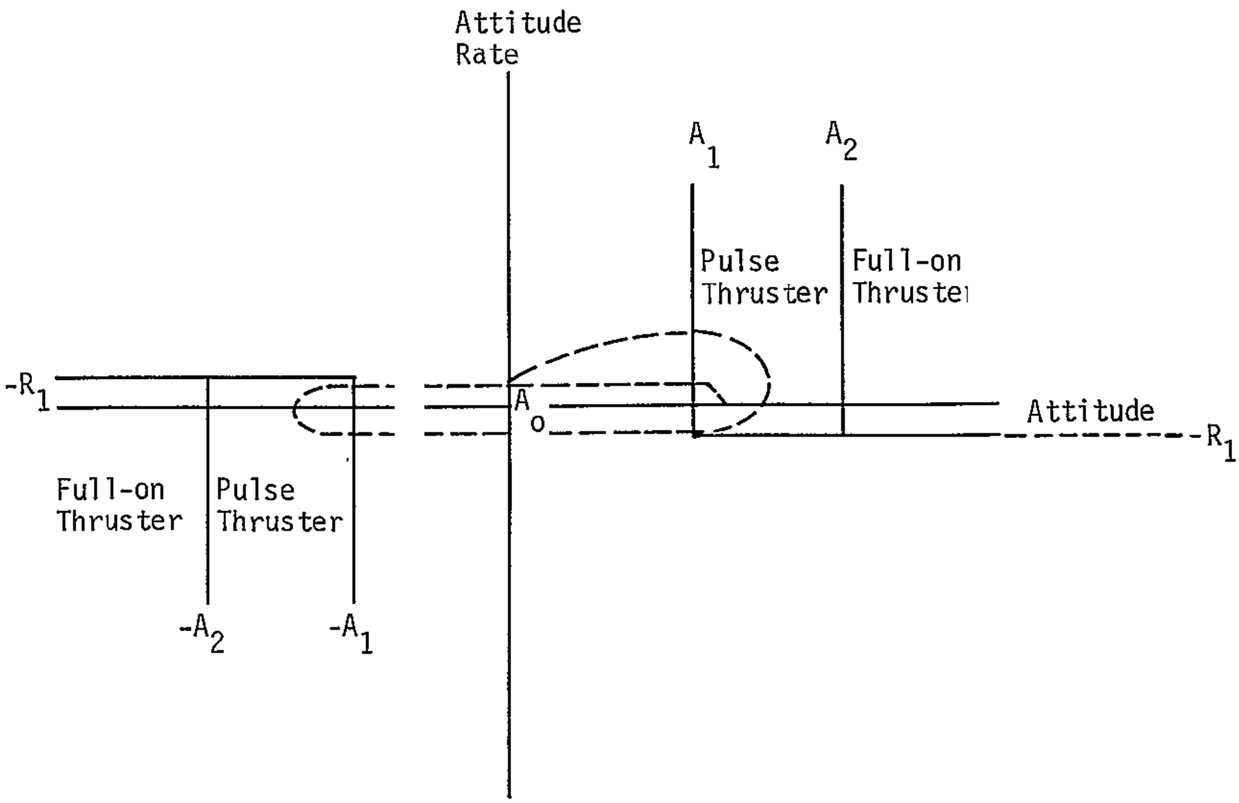


Figure VIII-35 Attitude Control Using Phase Plane Logic

Rigid-Body Equations of Motion - The rigid-body equations of motion convert the thruster state obtained from the thruster select logic into body forces and moments. These equations are integrated to yield velocity, position, rotation rates, and vehicle attitude. This information is used along with the camera position on the vehicle and target position to determine the relative position and attitude of the camera with respect to the target. The effects or orbital mechanics and Skylab motion are included at this point to allow a realistic simulation.

Physical Simulator - The physical simulator (Figure VIII-33) consists of a TV camera mounted on a set of servo-driven gimbals and rails. This configuration results in a six-degree-of-freedom simulation (3 translations and 3 rotations). A 1/5-scale model of the docking target and port is mounted in a fixed position within the simulator's range of motion. Input to the various servos comes from the rigid-body equations of motion. The output of the video signal is sent to the crew station, where the man closes the loop by issuing hand-controller commands. Hardware constraints result in the following scaled limits:

Translation	Min Limit (ft)	Max Limit (ft)
X	-62.5	0.0
Y	-23.8	8.3
Z	-26.7	17.7

Rotation	Min Limit (deg)	Max Limit (deg)
Roll	-179.0	179.0
Pitch	-45.0	45.0
Yaw	-46.8	46.8

Centroid Trackers - The centroid tracker, developed by Martin Marietta, is capable of providing automatic steering and stationkeeping on an unmanned vehicle for rendezvous and inspection of other spacecraft. The system is composed of a TV camera, specialized scan control, analog preprocessor or dedicated microprocessor, and an onboard digital computer as shown in Figure VIII-36.

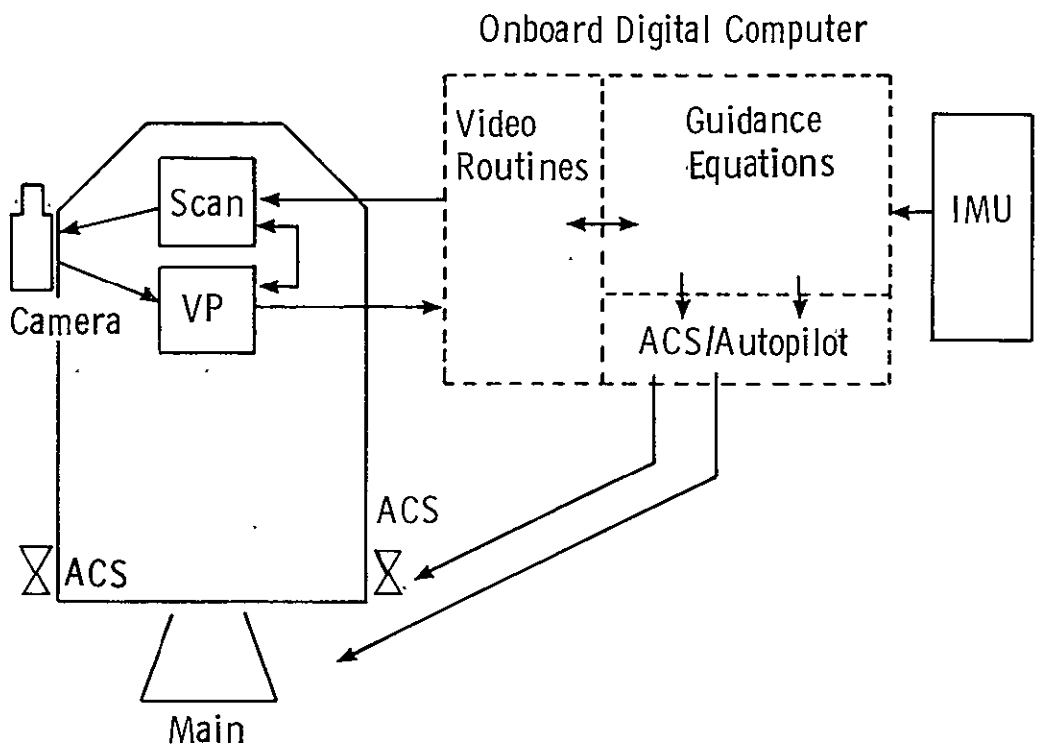


Figure VIII-36 Centroid Tracker Block Diagram

The system components function in such a way that the digital computer provides a supervisory and overall control function while wide band, high data rate computations are performed in the camera and analog preprocessor (Video Processor (VP). Large data block storage in the digital computer and high speed A/D converters are, therefore, not required.

The several different tasks will be based on a common sequence of processing. The camera is commanded to scan a certain area of given coordinates and size of scan in the field of view. The VP operates on the camera data and issues discrete values to the spacecraft digital computer at the end of a frame. The digital computer then decides what the next camera operation and VP function will be.

A wide variety of tasks may be accomplished in this manner, depending on the digital computer software. With this scheme, all mundane calculations are performed in the VP, leaving the digital computer free for other work.

In the application of rendezvous, the first function which the system must perform is that of detection. Assume the target is a bright disk on a black background. The digital computer would set the camera frame size to be approximately twice the expected area of the target. A frame would be scanned in which the VP would take the following functions: integral video (threshold), first X passed to the digital computer, which calculates area, diameter and center in X and Y coordinates. The next step would be to scan four smaller frames that would be positioned to cross the limb in each of four directions for more accurate measurement. The digital computer would command each of these in sequence and retain the results for a precise determination of target relative position. During scans, the digital computer is free for other tasks while the VP is collecting data.

The basic functions required in the VP, of course, depend on the particular task, but it would appear the following are adequate for most and are surprisingly easy to accomplish in the analog hardware or microprocessor as well. The following functions are to be calculated over one frame of scan.

AVE	Integral of the video signal
S_X	First moment of the video signal in the X direction
S_Y	First moment of the video signal in the Y direction
I_{XX}	Second moment of the video signal in the X direction
I_{YY}	Second moment of the video signal in the Y direction
I_{XY}	Cross moment

Figure VIII-37 shows a block diagram of the VP. As shown, some thresholding and filtering of the video is required. The system shown is a small analog version of the processor, commanded directly by the digital computer. It is also possible to mechanize these functions in a microprocessor.

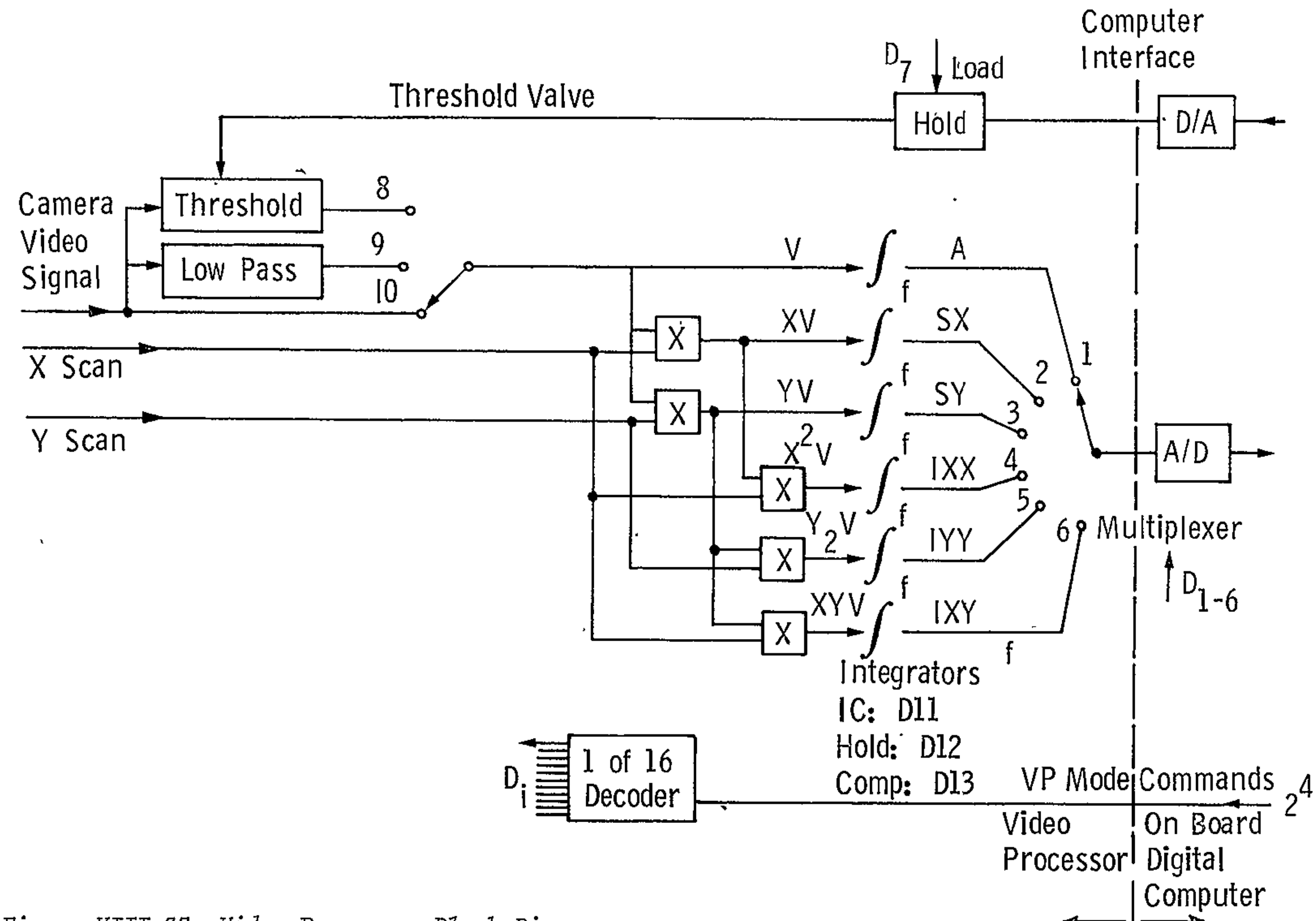


Figure VIII-37 Video Processor Block Diagram

A hypothetical mission would proceed as follows directed by the digital computer logic.

(1) Far Steering

(2) Near Steering

(3) Feature Detection and Inspection

(Track Intermittently)

(a) Spacecraft Sizing

(b) Axis Orientation Determination

(c) Spin Rate Determination

(d) Home on Predetermined Features

(e) Determine Whether Dock is Feasible

(f) Docking Maneuver

Figure VIII-38 shows a simplified scene and the associated video functions required for far steering. The guidance equations determine when the object is within range and field of view. Then a frame is scanned and the VP takes A, S_X, and S_Y. The digital computer then calculates $\bar{X}$ and $\bar{Y}$ in camera coordinates and translates these to steering signals. At the appropriate distance, determined from A (proportional to size) or a ranging device, the system changes to near steering logic, as shown in

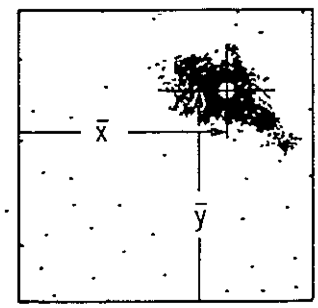
Figure VIII-39 shows one possible method of near steering and stationkeeping to be performed while doing other tasks. This approach may be used while taking pictures, looking for predetermined features, or performing surveillance maneuvers.

Following are a few of the functions that may be performed with appropriate software additions and the same basic hardware.

- Geometric Area
- Average Brightness
- Object Center in the Field of View
- Major and Minor Axes of an Equivalent Elipse
- Angular Orientation of Major Axis
- Search for Predetermined Feature
- Track Feature (i.e., automatic docking)
- Determine Spin Axis and Spin Rate

This system, being optically oriented, is limited by illumination power. However, it appears to be well suited for close-range applications.

Figure VIII-40 shows a simplified scene and the associated video functions required for far steering. The guidance equations determine when the object is within range and field of view. Then a frame is scanned and the VP takes A, S_X, and S_Y. The digital computer then calculates $\bar{X}$ and $\bar{Y}$ in camera coordinates and translates these to steering signals. At the appropriate distance, determined from A (proportional to size) or a ranging device, the system changes to near steering logic, as shown in Figure VIII-37.



Scene

VP Functions per Frame of Scan

$$A = \int_f \text{Video}^* \, dt$$

$$S_X = \int_f \text{Video} \;\; X_S \, dt$$

$$S_Y = \int_f \text{Video} \;\; Y_S \, dt.$$

Then the Object Center is Defined by

$$\bar{X} = S_X/A$$

$$\bar{Y} = S_Y/A$$

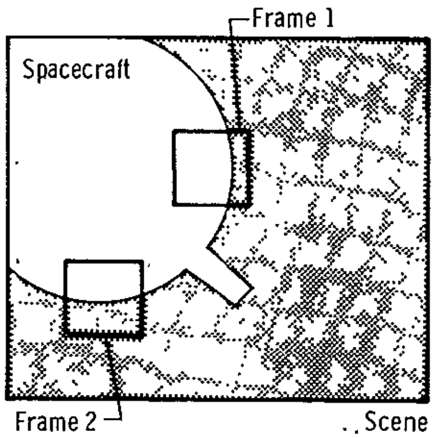
* It May be Required for Low-Pass Filter to Remove Starfield and Modify the Gray Scale for Contrast Enhancement

Figure VIII-38 Far-Steering Video Functions

Figure VIII-41 shows one possible method of near steering and stationkeeping to be performed while doing other tasks. This approach may be used while taking pictures, looking for predetermined features, or performing surveillance maneuvers.

Following are a few of the functions that may be performed with appropriate shoftware additions and the same basic hardware.

Geometric Area
Average Brightness
Object Center in the Field of View
Major and Minor Axes of an Equivalent Ellipse
Angular Orientation of Major Axis
Search for Predetermined Feature
Track Feature (i.e., automatic docking)
Determine Spin Axis and Spin Rate



Given Approximate Data from Previous Scans, Accurate Relative Position Is Obtained by Scanning an Area Covering Object Boundaries in Two or More Places. By Knowing the Commanded Scan Coordinates and the Relative Placement of the Camera, Frame 1 Yields Data on the X Position of the Spacecraft Edge, While Frame 2 Shows What the Y Coordinate Is.

Figure VIII-39 Near-Steering and Stationkeeping

Conclusions

As in each of the technologies discussed by the automation study, the application of rendezvous sensors is totally mission dependent. Therefore, any evaluation of sensors must be conducted on a mission-oriented basis.

RF systems have been extensively developed, tested, and used in the space environment so that extensive development for space qualification is not necessary. Laser rangers have either one of two problems. Either the laser is scanned to obtain maximum ranging, or the laser remains fixed in order to obtain accuracy. If the laser illuminator is fixed, the maximum acquisition range is severely limited by illumination power. If the laser is of the scanning type, the maximum acquisition range can be extended, but is still a limitation. In addition, the scanning can be extended, but is still a limitation. In addition, the scanning mechanism, a mirror mounted on a piezo-electric crystal, is sensitive to misalignments caused by shock and temperature. Other optical sensors are severely limited in range and are, therefore, not applicable to the general rendezvous mission.

An added requirement of autonomous rendezvous is that relative attitudes be measured; therefore, the system used must be able to track several transponders on the same target. The task could be accomplished by using IRUs mounted on each vehicle; however, alignment of the two units in inertial space and the communication link required would add an unjustified expense. By obtaining range and angle information to several transponders on the target, relative attitude can be computed by an onboard processor thus eliminating much of the expense. Of the RF system presented, the cubic corporation SR-100/ELF III system seems to be the most promising. The SR-100 is a flight proven instrument that has demonstrated a high degree of accuracy. The system was designed to track several targets at the same time, so minimum modifications have to be made. The ELF system is also flight proven and its incorporation into the CR-100 will allow complete determination of Range, Range-rate and angle to several targets.

It appears that the one missing link for a completely autonomous rendezvous system is the processor which takes the output from the rendezvous system and computes relative position and attitude of the two vehicles. Although the processing requirements of the rendezvous mission are not strenuous, the processing capability of existing computers such as NSSC-1 must be evaluated to determine whether they are adequate.

For missions where man is in the loop, a first judgement would say that no automation is required. However, experience gained from several rendezvous simulations at Martin indicates that optical control is more easily obtainable by providing the astronaut with information concerning the state of the two vehicles. Such information might include range, range rate, and relative attitudes. Since such missions will include imaging systems on the chaser vehicle, a simple low-cost system could be included whereby the imaging data is processed by a computer onboard shuttle using the centroid tracking algorithms. Increased accuracy can be obtained by adding another camera and using the stereoscopic techniques discussed. By combining these two systems, the centroid tracker and the stereoscopic techniques, the sensor/processor could provide the astronaut with not only video data, but quantitative data concerning the relative position and attitude of the vehicle.

Recommendations

There are three areas or systems which must be developed if autonomous rendezvous is to be made feasible. These areas are the RF system to be used for general purpose rendezvous, optical systems used in short range missions where observation is a primary goal, and the processors which derive usable quantities from sensor outputs.

The primary area of development must be in the area of flight qualified general purpose fault tolerant processors. The algorithms associated with video processing are only applicable if the flight processor is available. Also, many preliminary studies indicate that the NASA Standard computers are not going to be adequate for a wide variety of applications. Therefore, significant funding should be oriented toward evaluating the requirements of a future onboard processor and then the breadboard and development of a suitable system. It is more wise to establish a firm foundation of which computers are a major part for the field of space automation than try to design around the inherent weaknesses. Since the field is young, there is an affordable luxury of spending a little extra time and money establishing the basic component, the processor, of future systems.

Although maneuverable TV units and TRS will not be considered autonomous, some automation should be incorporated into their design. Such missions might be able to use the Shuttle computer rather than carrying an onboard processor. In addition, an autonomous system might make use of the cameras used on these missions. For example, by adding one more camera to the vehicle and by using the Shuttle computer, an astronaut would be provided with quantitative data concerning the state of the two vehicles. The development of such a system does not involve significant funding and yet the results will be meaningful.

IX FAULT TOLERANT PROCESSORS

Computer Requirements in Space Automation

Space missions that require long unattended spacecraft service will require extensive implementation of automation technology. Planetary missions require autonomous spacecraft capability for navigation, attitude control, and payload data management. Orbital missions in which spacecraft are in ground contact for only a small portion of the mission also require a high level of automation.

Autonomous operations on long duration missions involve heavy computing loads for spacecraft navigation, attitude control, and data processing. These requirements place greater emphasis on computer capability and especially on computer reliability.

Evaluation of Fault-Tolerant Computers

The early history of data processing on satellites and in missile systems is one of extremes. At one end of the spectrum missile and booster guidance and control systems were characterized by short operational life, long self life, and severe operating conditions. At the other end were requirements for long life devices with brief periods of maximum activity; e.g., planetary fly-bys and relatively benign environments. In both situations, requirements were considered specific enough to warrant development of special purpose computers for each mission.

This situation was changed by the increasing use of satellites requiring continuous operation at near-maximum computing loads. High data rates are often necessary and orbital rather than boost and re-entry environments are characteristic. High development costs and similarity of many mission requirements generated an interest in broadening the applicability of computer systems to meet the needs of many missions. From this evolved the idea of using a central general purpose computer, and with it, the concept of a modular design to provide the flexibility in sizing to a particular mission.

The concept of a single computer to handle several data processing tasks on a spacecraft required a reliable general purpose processor. In addition, the trend toward longer on-orbit lifetimes for increased system cost effectiveness placed greater emphasis on the contribution of the data processing subsystem to overall system failure rates. Thus, there was renewed interest in improving computer reliability through fault detection and recovery; i.e., the fault-tolerant computer.

Fault-tolerance in a computer means the ability to render at least the essential level of service after the occurrence of a fault. This implies a measure of redundancy because there must have been initially some resources (parts or circuitry) that were not used for the basic level of service. But redundancy itself; i.e., duplicating components, is not sufficient for fault tolerance. There must also be provisions for recognizing a fault and re-establishing the essential level of service. In spacecraft applications, fault tolerance requires that the entire process of recovering from a fault must be accomplished automatically onboard. If error detection, reconfiguration, or recovery depends on diagnosis and reconfiguration by ground control or other external functions, then it is a redundant installation rather than fault-tolerant. Redundant computer installations are in use today in several spacecraft, e.g., the Space Shuttle Orbiter, but the individual components are not fault-tolerant. The fault-tolerant approach has the advantage of providing immediate response and reducing the dependence on ground support or external hardware.

To date, fault tolerance techniques have gained only limited acceptance in spacecraft. The obstacles have been primarily the large development costs and the onboard resources required for such a computer. The computational needs have been met either by relegating most of the data processing to the ground or by using multiple onboard computers with the capability of switching from one to the other by ground command.

The obstacles to employment of fault-tolerant computers on spacecraft have been reduced considerably because of advances in computing architecture and particularly due to the progress in semiconductor technology which permits the required logic functions to be realized at lower weight, power consumption, and cost. At the same time, there is a greater need for fault-tolerant computers due to longer mission durations, more demanding mission objectives in terms of spacecraft management and payload data processing, and the need for greater degree of automation for autonomous operation.

Fault-tolerant computing for general applications is today a well established discipline. At present, the furthest progress along the road to hardware realization has been achieved in the Fault-Tolerant Spaceborne Computer being developed for the Air Force Space and Missile Systems Organization (SAMSO). The arrival at this state of development can be described as an evolutionary process.

The first fault-tolerant digital equipment to be developed for spacecraft and the only one that has flown so far is the Primary Processor and Data Storage (PDPS) for the NASA Orbital Astronomical Observatory (OAO). This was not a stored program computer. This equipment, designed from 1960 on, employed quad-redundant logic, triple modular redundant (TMR) delay lines, and duplex memory with error detecting code.

The first spacecraft computer design that made extensive use of dynamic fault tolerant techniques was the JPL-STAR (self-test and repair) computer. The concept originated in 1961 but major activity on the project was concentrated between the mid 1960s and the early 1970s. During that period there was much interest in missions to the outer planets which involved spacecraft operation of up to 10 years. Only very limited communication with earth was possible during these missions and a computer was required for navigation, experiment control and housekeeping.

Features of the STAR computer have been carried over into later designs. Examples are a modular bus oriented architecture, and a separate restarting function called the test and repair processor (TARP) that activates and deactivates modules and controls the program recovery following a hardware fault.

A breadboard of the computer was built which demonstrated the value of many fault-tolerance techniques as well as overall system capabilities. Due to curtailment of budget for outer planetary missions, the STAR never progressed to flight worthy implementation.

In 1966, work began on a fault-tolerant computer for the general aerospace environment, including spacecraft applications. The computer employed a multiprocessor organization and single instruction restart capability. In the multiprocessor, tasks are shared among a number of processors. If one becomes faulty, the computational tasks can be transferred to the remaining processors. A software scheduler can assure that the most essential tasks are given the highest priority when such a fault occurs. In this way, faults result in "graceful degradation" rather than in complete loss of computing capability. The single instruction restart capability is provided by having three independent scratch-pad memories associated with each processor that contain the temporary data for the program being executed. The scratch pad output is voted, and upon detection of an error, auxiliary memories can be brought into use. Triplication assures that no temporary data are lost, and re-execution can start at the last completed instruction.

The program was partially sponsored by the NASA Manned Spacecraft Center, and some of the developed concepts have influenced the design for the Space Shuttle computer configuration. So far, no specific application of this design for satellites has developed.

The multiprocessor approach was also employed in a computer, designed under the Space Ultra-Reliable Modular Computer (SUMC) program called the Automatically Reconfigurable Modular Multiprocessor System (ARMMS). A later version under the acronym ARMS was intended as a feasibility model and in it the multiprocessor capability was deleted. Triple modular redundancy with voting was employed at the level of the memory, central processor, and I/O processor. The design was applicable to launch vehicles space stations, and deep space probes. No planned application of this concept for satellite computing is known.

In 1974, a European study was started to define a fault-tolerant computer that would use hardware under current development in a simplex onboard computer. Two candidate configurations were defined, one utilizing duplex processing for error detection and another using a microprocessor with software error detection. It was concluded that differences between the two configurations were too small to permit selecting the least expensive one but that both candidates were well suited to the application. There had been no hardware implementation as of January, 1977.

The Fault-Tolerant Spaceborne Computer

The Fault-Tolerant Spaceborne Computer (FTSC) is being developed by the Air Force to support long duration space missions. The overall objective is to provide a 5-year on-orbit capability to perform the computational tasks for attitude control and pointing telemetry processing, command, supervisory, subsystem management, and recovery, and payload processing.

At the user interfaces, internal buses and in memory, the information is protected by an error-detecting and correcting code. There are seven main memory modules, four of which are normally designated as active and three as spares that can be activated for memory failures not corrected by the use of codes. In the central processing unit, dual operation is used for error detection, and a rotating replacement is used for reconfiguration. The timing modules and the power modules each have redundant internal monitors that detect deviations from specified output. In these modules and in the interface modules, detection of a failure will cause switchover to a standby spare. The configuration control unit, which is the ultimate control for error recovery, the circumvention unit, and the hardened timing unit are all operated in triple modular redundancy. The computer is designed to operate in high radiation environments, which makes it a prime candidate for outer planetary missions.

The predominant semiconductor type used in FTSC is CMOS/SOS, selected because it provides adequate speed at very low power consumption. The memory is non-volatile; i.e., memory content is retained while the memory is unpowered, permitting memory modules to be powered down when not in active use. Non-destructive readout prevents garbling due to transients while a read operation is in progress.

The instruction set of the FTSC supports 32-bit fixed point and floating point arithmetic as well as vector operations that are particularly useful in navigation and pointing control functions. Direct and indirect addressing and predecrement and postincrement are available. These features permit writing very compact codes and thus reduce storage requirements. The ground override can disconnect individual elements of the configuration control unit, permitting thorough checking of this vital element of the computer.

Many application programs in the FTSC service closed loop systems, such as attitude control, in which temporary loss of data or bad data during one or two computing cycles, can be tolerated. Data for these applications need no specific protection. On the other hand, certain navigation quantities cannot be recovered autonomously within the spacecraft if they are destroyed due to memory failure. For these data, a "store double" instruction that permits virtually simultaneous access to the same location in two independent memory modules. These data are thus protected in case of a memory module failure.

When reconfiguration following a failure is complete, the computer resumes operation at a memory location whose address is stored in a special location. This address is called the rollback point; computations supporting closed-loop routines or housekeeping need only one rollback point at the start of each task. In a few cases, e.g., updating of position by velocity increments, complete repetition of a task could produce undesirable double incrementing. To prevent this, the programmer can insert a rollback point immediately after a position update. Upon subsequent recovery from a failure, the program will restart at the last rollback point, thus assuring accurate position information.

The FTSC is expected to meet the needs of a large variety of Air Force and NASA missions in the post-1980 time period. The computer provides adequate performance and reliability for these missions without excessive weight and power requirements. The modular nature of the computer permits tailoring memory size and sparing for all functions to meet a variety of application needs. The incorporation of the fault-tolerance provisions in a single component where they can be tested at the factory simplifies design, compared to an assembly of separately packaged components with external redundancy provisions.

Recommendations for Further Research and Development

There are several areas in which research and advanced development are needed to fully exploit the capabilities of the FTSC and to lay the groundwork for more extensive application of fault-tolerant data processing for future spacecraft.

a. Reliability calculations for the FTSC and other computers have been carried out using an exponential reliability model at the part levels. Although the validity of this model has been questioned for some time, the deviations from it were felt to be tolerable in view of the simple calculations that resulted from the exponential assumption and the extensive modeling capabilities that existed specifically for the exponential failure law. Recent investigations of on-orbit failure rates have shown drastic deviations from the exponential failure law. This suggests that its use for predicting failure rates for long duration missions should be avoided. The existing data points to high failure rates during very early mission stages followed by successively lower failure rates for each six-month interval up to three years. Although these observations lead to optimism regarding the success of long duration missions, they point up the need for new analysis tools. A failure law that is applicable to computer architecture rather than the part level is needed, as well as a concensus as to the form of a reliability function for predicting on-orbit reliability for long duration missions. Considerable research and organizational effort in this area will be required to fully exploit the capabilities expected from components currently under development (see Electronics Chapter).

b. The software that is to be executed on a fault-tolerant computer for spacecraft is another important issue. Although the reliability of the hardware can be demonstrated at a level that will allow computer control of critical spacecraft functions, the full advantages of the fault-tolerant computer can be realized only if the software has equivalent fault-tolerance. Neither testing nor formal verification can assure the total correctness of the software. Fault-tolerant software techniques are available but are costly to implement because of the memory required. However, with the advent of magnetic bubble memories, it may be possible to keep alternate programs in a backup store and this may facilitate the use of fault-tolerant software techniques for essential program elements. The effort necessary to define appropriate software and memory architecture that will permit full realization of the potential of the fault-tolerant computer seems fully justifiable.

c. Some tasks that need to be carried out at high speeds impose burdens on the fault-tolerant computer or are completely impossible to secure in this manner. Examples are frequency control of communications equipment and sensor data compression. In both, a sample instruction repertoire is sufficient and a minimum of local storage is required. Microprocessors seem well suited to these tasks and can be made fault-tolerant by being tested periodically and, if necessary, replaced under the control of a program residing in the fault-tolerant computer. The use of microprocessors to supplement and adapt fault-tolerant central computers to a wider range of spacecraft applications is an area that deserves further research and development effort.

X SPACECRAFT PERFORMANCE MANAGEMENT AND MONITORING

Automation for autonomous operation of spacecraft for long-duration missions demands extremely high reliability of all spacecraft systems. The fault-tolerant computer offers the potential for reliable data processing in the 1980s time period. Navigation and control system components and present-day computers are not fault-tolerant. The level of reliability necessary for long life missions can be achieved only by means of redundant component configurations, which, in turn, require provisions for hardware performance monitoring, fault detection, and reconfiguration.

The Space Shuttle Orbiter avionics system represents the highest level of technology in redundancy and redundancy management in use today. That system and the redundancy management techniques are discussed in order to provide a basis for future technology projections.

Space Shuttle Requirements

Because of economic importance attached to successful mission completions and safe vehicle recovery, very strong emphasis has been placed on the reliability of the Space Shuttle Orbiter. The approach adopted to meet the reliability requirement has been the use of redundancy throughout the system, along with fault detection and identification (FDI) and system reconfiguration capabilities.

The Shuttle Vehicle Specification imposes the requirement that all flight vehicle subsystems exhibit at least fail-safe (FS) characteristics; i.e., no single failure shall result in loss of the vehicle or crew. The general requirement imposed on all avionics is that no single failure shall cause inability to achieve mission objectives (fail-operational) (FO), while the second failure is still FS. For this FO/FS requirement to be met, the critical guidance, navigation and control (GN&C) and related subsystems must provide automatic reconfiguration as required for time-critical failure modes, with manual reconfiguration override capability available. In addition, redundant components must be physically separated where possible so that an event that damages one component is not likely to damage another. Redundant circuits should not be routed through one connector, and redundant boxes should be housed in separate bays.

Although the external tank (ET), the solid-rocket booster (SRB), and the payload (PL) avionics equipment are developed in accordance with these same FO/FS and physical separation requirements, only the Shuttle orbiter avionics redundancy management will be considered here.

Redundant System Configuration

Table X - 1 lists the redundancy levels of guidance, navigation, and control (GN&C) and related subsystem elements that require automatic reconfiguration because of time-critical failure modes. Other avionics subsystems have dual or triple component redundancy but are not addressed here because they perform non-time-critical functions and failures are neutralized by manual reconfiguration.

TABLE X - 1 AVIONICS REDUNDANCY

DEVICE	REDUNDANCY LEVEL
Inertial Measurement Unit (IMU)	3
Star Tracker (ST)	3
Rate Gyro Assembly (RGA)	4
Accelerometer Assembly (AA)	4
General Purpose Computer (GPC)	5
Multiplexer/Demultiplexer (MDM)	8
Aerosurface Servo Amplifier (ASA)	4
Ascent Thrust Vector Control Driver (ATVC)	6 (Orbiter)
	4 (SRB)
Reaction Jet Driver (Nominal)	4
Microwave Laser Beam Landing System (MSBLS)	3
Tactical Air Navigation (TACAN) System	3

Data Processing

The Data Processing Subsystem is the central member of the GN&C mechanization. The heart of the subsystem are five General Purpose Computers (GPCs), each having a modified IBM AP-101 processor and core memory comprising the central processor unit (CPU), with a special input/output processor (IOP) that interfaces with 24 serial digital data buses. Eight buses carry flight-critical data between eight flight critical MDMs which interface with redundant sensors and the GPCs. During the critical flight phases, ascent and descent, as many as four of these computers may be simultaneously solving the GN&C problems redundantly. The fifth computer can be programmed for the system management (SM) tasks, and during non-critical phases, the two mass memories can be used to load one or more computers for other tasks such as payload support. To eliminate divergence (the possibility of outputs from multiple machines drifting apart, especially where there are integrators), the computers are synchronized. Thus, computers having the same requirement are computing identical outputs from identical inputs.

The input/output processors are controlled so that only one processor and hence one computer is assigned at any time to transmit over a given bus. Every bus has a computer assigned to it. However, although only one computer can transmit on a given bus (for example, to request sensor data) any computer can receive data from any bus except the GPC-unique instrumentation buses. Data is transmitted by time division multiplex techniques at a one mega bit data rate.

Each computer has self-test and built-in test equipment (BITE) to determine its own health and the computers, utilizing the five intercomputer buses, are capable of comparing answers to static (sample) problems and dynamic (current) problems and comparing answers as a further means of fault detection and isolation.

Since all computers can "listen" to all other computers over the intercomputer buses, a computer can be disabled (output transmission terminated) in its IOP as a result of being voted out by other computers if two or more are in disagreement. A computer can also be disabled by its own fault detection.

Multiplexer/Demultiplexers

The multiplexer/demultiplexer (MDM) is a key element in the GN&C system. It interfaces between the serial data bus and the several subsystem elements connected to it. These element interfaces can be analog, digital or discrete, and data can flow in either direction. The MDM uses only one bus at a time, and two buses are available to each of the eight flight-critical MDMs for redundancy. Accordingly, every flight critical bus is connected to one forward and one aft MDM for redundancy. Each set (forward and aft) processes, or at least is set up to process the same signal set. Each MDM has dual electronics so that an alternative path is available in the event of a failure. Critical interfacing elements other than GN&C components, such as controls, displays, event controllers, and the main engines, are also serviced by these eight flight critical buses. Sensor data are introduced through the MDMs. Four flight-critical MDMs are located in the forward avionics bays and four in the aft avionics bays. A typical forward MDM converts one set of inputs from various redundant sensors. Two MDMs receive data from dual redundant sensors, three from triple redundant sensors, and four from quadruple redundant sensors. In this way, redundant sensor data reach the computers through redundant MDM paths. As noted, all computers listen to all data inputs, so that each computer has, for example, three IMU inputs to listen to. Also brought in through forward MDMs are manual controlled inputs, tripled inputs from both left and right sides of the cockpit. These include RHCs, THCs, SBTCs and RPTs. Various panel switch positions also are entered through the forward MDMs.

Similarly, the aft MDMs introduce from quadruple redundant pitch, roll and yaw rate gyros located on the orbiter and the solid-rocket boosters. Also, triple redundant lateral and longitudinal accelerometers are serviced by the aft MDMs.

Control Effectors

The control effectors consist of the aerodynamic control surfaces (inboard and outboard elevons, rudder/speed brake, and body flags), the thrust vector control (TVC) system (3 orbiter main engines) and solid-rocket booster (SRB) engines (one on each of the two SRBs) and the reaction jets (38 nominal jets and six vernier jets). The various control effectors are driven by computer commands through the MDMs.

Each aerosurface actuator is controlled by a valve spool which has four rigidly attached pistons or secondary actuators. The control valves for each secondary actuator is driven by an aerosurface servo amplifier (ASA) which is commanded from a separate computer through a flight critical MDM. The valve spool acts as a majority voter in the presence of a failed channel.

This parallel operation provides downstream protection against undetected computer or MDM failures. Since the total loss of any surface can cause vehicle loss and the built-in test equipment (BITE) failure coverage of drivers and valves is low, four strings are used to provide a majority vote on second failure.

The ascent thrust vector control drive mechanization is similar to that of the aerosurface drive with some distinct differences. Like the aerosurface drive, one flight-critical MDM drives one secondary actuator if each of the ten gimbal actuators (six on the three main engines plus four on the two SRBs). However, each actuator requires only three secondary actuators. This is because two failures must occur to disable (causing recentering) an actuator, and safe flight is still provided with one gimbal centered. However, protection must be provided against a single computer or MDM failure disabling one string on all actuators. Such a computer failure is neutralized by reassignment of the fourth GN&C computer by bus reassignment, and an MDM channel failure is precluded bv the dual-channel electronics of the MDMs.

The baseline RCS configuration consists of 14 nominal and six vernier jets forward and 12 nominal jets on each side aft. Although certain crossfeeds are possible, one set of fuel tanks supplies each of these fore and aft reaction jet centers. Each center has four fuel manifolds for nominal jets, arranged so that safe vehicle control is still possible after two manifolds have been shut down (for example, to shut off failed-open jets). The forward and aft flight critical MDMs are arranged so that the manifolds within a center are driven by separate MDMs and computers.

Redundancy Management

Redundancy management includes determination of how the ultimate output of a redundant sensor set is used and the manner in which faults are neutralized. This involves fault detection, identification and reconfiguration.

The major role of the computer in redundancy management is support of sensor control. It is centrally involved in failure detection, identification, rejection and bad sensors, signal selection, and crew input application. Data properly received from qualified sensors are made available for signal selection for each redundant set by the selection filter function. There the proper algorithms are implemented, depending upon the number of available inputs. Generally mid-value-select (MVS) is used for three inputs and average or prime/standby for two inputs. Then, as a parallel offline separation, sensors are qualified by analysis of various tests. These include review of consecutive input data checks such as data present, parity, format constraints and BITE which are reported to the redundancy management (RM) control function. Other tests, accomplished as a part of the FDI function, may include comparisons among strings, comparisons against other similar sensors, and data reasonableness tests. The RM control function uses these test reports to update a table of redundant element status and to define qualified entries in the selection filter. The RM control function also controls external subsystems other than sensors (e.g., disabling aerosurface driven channels) and receives pertinent failure status reports from other subsystem monitors. Redundancy configuration and failure status are presented to the crew by cathode ray tube (CRT). System control by the crew (for manual reconfiguration override or for manual control of non-time-critical functions) is through the RM control function by means of keyboards at the crew stations.

System Management

The system management (SM) function deserves mention here although it is concerned with non-time-critical functions. The SM function supports vehicle redundancy management in several essential ways. Through its network of sensors for non-time-critical functions, including many non-avionics functions, it provides automatic fault detection. SM-detected failures are neutralized by manual subsystem reconfiguration, thus giving the crew an opportunity to survey the existing situation before making a system change. The SM system (including instrumentation, data processing, recording, display, annunciation, and controls) is designed to continue functioning after a single failure, although individual sensors and controls are typically not multiple. Common portions of the instrumentation system, such as pulse code modulation (PCM) masters, recorders, and the MDM multiplex interface adapters (MIAs) are duplicated. However, the individual sensed parameters are themselves fed in single-string fashion through signal conditioner channels into the MDM input channels.

Summary

The Shuttle avionics redundancy management system is designed to meet stringent functional and physical requirements to general FO/FS characteristics. Avionics redundancy is managed through a comprehensive system of fault detection and identification followed by system reconfiguration. Failure detection and identification is provided by one or more of BITE indications, data transmission checks, comparison tests, reasonableness tests, or crew observations (for non-time-critical functions). Reconfiguration is either automatic or manual, depending on the time criticality and form of redundant element status information. In addition, appropriate means of formulating composite outputs from multiple inputs (signal selection) are provided to minimize failure effects.

Recommendations

The example provided by the extensive use of multiple redundant components and redundancy management in the Space Shuttle should be used as a guideline in the design of automated spacecraft systems for long duration missions. In the interest of saving weight and power, and in reducing software complexity and computer loads, dual redundant elements operating in prime/standby mode may be necessary. To support this approach, emphasis should be placed on improvement in individual component reliability, internal component fault tolerance, and self-test and BITE capabilities.

REFERENCES

AIAA Paper 75-571 - Space Shuttle Avionics Redundancy Management Eugene A. O'Hern, Rockwell International, Downey, California, AIAA Digital Avionics System Conference, Boston, Massachusetts, April 2-4, 1975.

Architectural Design for Near 100% Fault Coverage, J. J. Stiffler, Raytheon Company, Sudbury, Massachusetts,

Fault-Tolerant Computers for Spacecraft, H. Hecht, The Aerospace Corporation, El Segundo, California

Rollback Point Insertion Strategies, Frank J. O'Brien, Logicon, Inc., San Pedro, California

Software Recovery in the Fault-Tolerant Spaceborne Computer, Dario DeAngelis and Joseph A. Lauro, Logicon, Inc., San Pedro, California

Specification of the Fault-Tolerant Spaceborne Computer (FTSC), Dale D. Burchby and Larry W. Kern, USAF, Los Angeles, Air Force Station, Los Angeles, California, Walter A. Sturm, The Aerospace Corporation, Los Angeles, California

XI AUTONOMOUS NAVIGATION

Increases in the complexity of scientific and operational endeavors onboard modern spacecraft as well as the growing need for rapid assimilation of resulting data has made space automation a necessary goal for future space missions. It is estimated that by the year 2000, the return of imaging data from Earth applicatons satellites will be 10^{13} - 10^{15} bits per day; an increase of three to four orders of magnitude above the present rate. Previous missions have already provided large stores of data which are becoming obsolete as they await analysis and interpretation, and the increased data rate will become more of a burden than a benefit unless some sort of autonomous preprocessing is implemented onboard the spacecraft.

Two operations which require a large percentage of the total processing time are sensor data annotation and stationkeeping of the satellite. Data annotation involves combining latitude, longitude, and time information with image sensor data, and is currently performed by man/machine systems which are both awkward and time consuming. Stationkeeping presently involves uplinking navigation and attitude corrections in order to keep the satellite within a desired window for its mission. Annotation of image data can be derived if the position and attitude of the satellite and the pointing angle (relative to the spacecraft axes) of the scientific sensor are known precisely. Latitude, longitude, and time information can then be assigned to each picture element, thus reducing much of the time required for image data annotation. Conversely, knowledge of the sensor's line of sight allows determination of the satellite's position and attitude. Autonomous navigation systems such as the ones described in this paper were conceived and developed partially for these reasons.

Autonomous navigation systems can be separated into three groups -- position-sensitive angular measurements to celestial objects, Earth-based target reference measurements, and range measurements to known beacons. Systems utilizing angular measurements to celestial objects are an outgrowth of the concept proposed by Farell in the mid 1950s. In this system, the satellite determines the local vertical to the Earth's surface and then measures the angles to three separate known stars. From these measurements, position and attitude information can be derived.

Systems using Earth-based target reference measurements to determine position and attitude depend on Ground Control Points (GCP) or landmarks which can be identified from space. GCPs have many forms -- pinpoint light sources, EM emitters, linear features, or specified areas on the ground. Although different systems detect different forms of GCPs, the parameter measured is always the angle to the center of a feature whose coordinates are known. From these measurements and measurements taken from an attitude determination system, complete position and attitude information can be obtained.

Systems relying on known beacons determine range to three or more known points and triangulate to solve for position. Range is determined by acquiring some sort of navigation signal transmitted by the beacons. The signal contains information about the position of the transmitter and time of signal origination. The receiver then solves for the propagation time by knowing the time of signal reception; then assuming a constant propagation velocity, the range is calculated. Each system described in this paper is separated in Table XI-1 into one of these categories.

Table XI-1 Autonomous Satellite System Concepts

Position sensitive angular measurements to celestial objects	
LES 8/9	Sun - Local vertical
SS-ANARS	Moon - Star
AGN	Planet - Star
Earth-based target reference measurements	
Natural landmark identification	
1) area correlator	
2) linear feature detection	
Artificial landmark identification	
1) Systems using optical emitters (lasers, search lamps)	
2) ILT using microwave emitters (radars)	
Range Measurements to known beacons	
GPS	Earth orbital beacons

The following section briefly describes the theory of operation for seven autonomous navigation systems being developed.

CANDIDATE SYSTEMS

Autonomous Guidance and Navigation (JPL)

"For nearly two decades, until Viking and Voyager, unmanned non-landing spacecraft have been totally dependent upon Earth-bound perception and intelligence. Except for Viking and surveyor landers, they are still totally dependent upon Earth-bound intelligence. Earth-based orbit-determination technology has been honed to a fine edge, but the ultimate limitations of this approach are becoming apparent. Delays introduced by the round-trip light time and manual intervention in the navigation process are thwarting potential and guidance capabilities for the distant planets." (Bierman, 1977).

Autonomous Guidance and Navigation is a research project at the Jet Propulsion Laboratory (JPL), whose purpose is to develop technology for combining mobility, perception, and intelligence on unmanned spacecraft. (Kohlhase, 1977 and Klump, 1977). From the research invested to date, a system is being designed and is proposed for use onboard the 1982 mission to Jupiter. The theory of operation is described below.

Theory of Operation - A lens system projects a rectangular field of view (FOV) of two to three degrees onto a CCD sensor composed of an nxn array of detectors. The output of each detector is quantitized and is proportional to the illumination. The CCD, therefore, transforms the image into a digital matrix which can be processed by a computer. The sensor/processor detects the presence of stars and illuminated bodies such as planets, asteroids, or satellites within the sensor's FOV. The centers of the target bodies are computed, using models of target shape, by a microprocessor within the sensor determining the precise (within several micro radians) directions to target bodies relative to the direction of known background stars. The absolute directions of the background stars are determined by measuring their absolute and relative magnitudes, their relative directions, and the angular separation between neighboring stars. This information is compared to data stored in an onboard star catalog to obtain the star identities, and absolute directions. By determining the absolute direction to several known stars, the satellite's attitude and position can be determined. Knowledge of the spacecraft's position and attitude as well as information about the relative direction to target objects allows not only precise determination of target ephemeris but also autonomous updates in the vehicles' guidance equations.

The initial system developed for the next five years relies on Earth-based ground support to determine navigation corrections and optical orbit determinations. In the future, however, the systems will be completely autonomous.

Lincoln Experimental Satellite Autonomous Stationkeeping System

In 1975, two satellites, LES 8 and 9, were launched by the Air Force with the ability to autonomously navigate. Although the stationkeeping accuracy was somewhat modest and the application to various missions limited, the project has demonstrated the feasibility of autonomous navigation onboard space vehicles. (Srivastava, 1972, 1973, 1974)

The first design objective for the LES 8/9 stationkeeping system was to maintain a desired station as accurately as possible with a high fuel efficiency.

Autonomous stationkeeping requires control of the mean longitude of the satellite. The daily variation of the longitude about a mean, which results from the eccentricity and inclination of the orbit from the equator, is uncontrolled and goes through a predictable periodic cycle. The input required for the above stationkeeping system is the angular position of the satellite in its orbit plane. Angular position is determined on LES 8/9 through the use of two sun transit sensors, horizon sensors, a solar ephemeris synthesizer, and hardwired algorithms. The sun transit sensors are mounted pointing in opposite directions with their line of sight being coincident with the roll axis of the satellite. The sensors consist of optics which focus a very narrow FOV onto a threshold detector. Twice an orbit the sun crosses the optical axis of one of these sensors and a signal is triggered. If the yaw axis of the satellite is perpendicular to the Earth's surface (corresponding to no attitude error) during this transition, then the angle formed by the Earth's center, the spacecraft, and the sun would be 90 degrees (Figure XI-1), and the longitude of the satellite would be known. However, attitude errors can exist so horizon sensors have been added to determine the local verticle by detecting the edge of the Earth's limbs. From these measurements, two degrees of attitude, roll and pitch, can be determined. The yaw error is assumed to be slight and unimportant. With knowledge of the satellite's attitude and time of sun transit, the satellite can stationkeep to an accuracy of .02 degrees in its orbit plane.

The LES 8/9 stationkeeping system incorporates a backup shadow sensor (Figure XI-2) to detect position. A satellite in synchronous orbit is constrained to pass through the Earth's shadow during its orbit. The longitude of the satellite can be determined by observing the time of entering and leaving the shadow. The event of passing through the shadow is independent of attitude and can be detected by using a very simple sensor. The shadow measurements have a small random error, but the error due to the eccentricity of the satellite cannot be eliminated. Therefore, the shadow sensor is used as backup for the sun sensors.

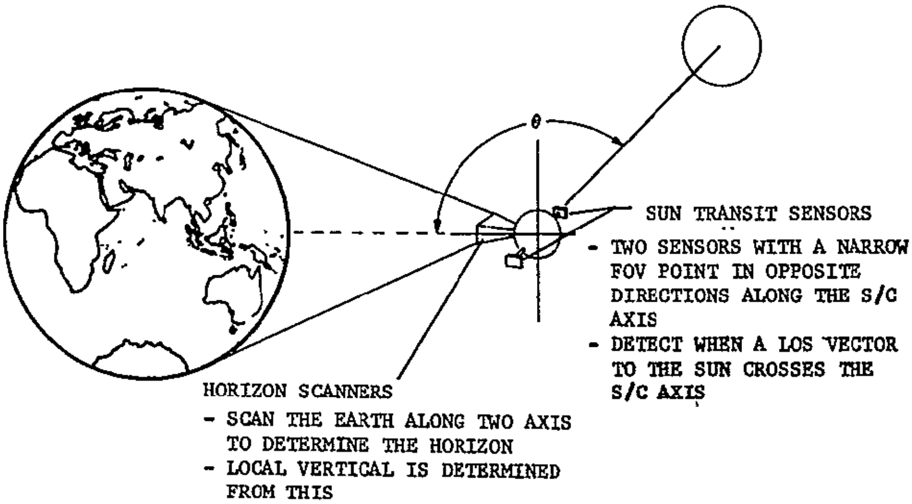


Figure XI-1 LES 8/9 Sun Transit Sensor

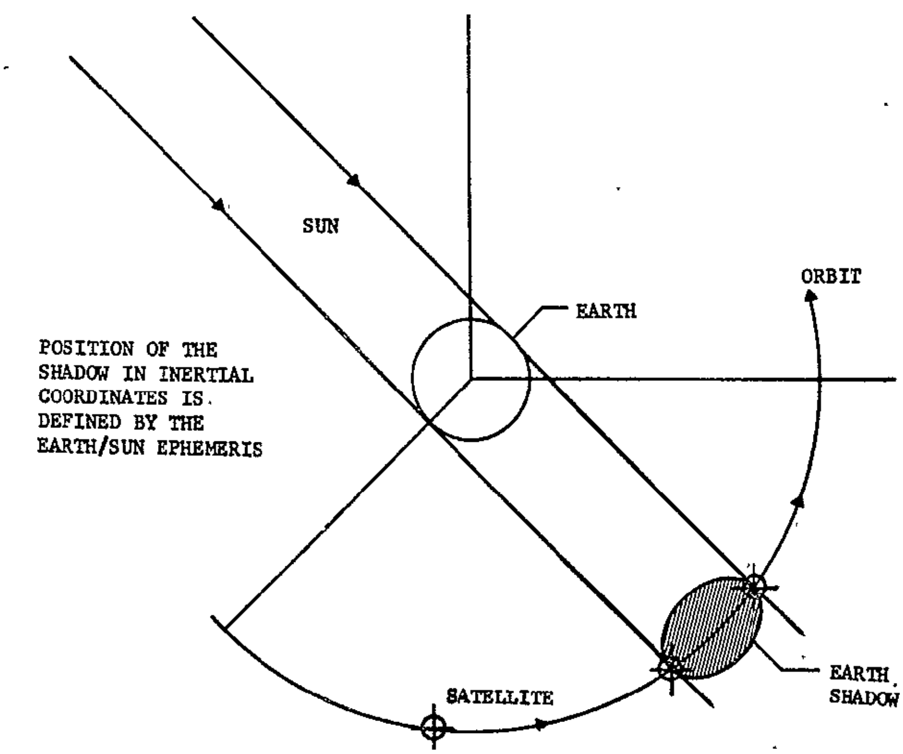


Figure XI-2 LES 8/9 Shadow Sensor

The second requirement of the LES 8/9 stationkeeping system is to execute an automatic station change at a desired drift rate whenever required and acquire a new station with fast settling time and a minimum number of overshoots. A stationkeeping filter provides an estimate of the satellite drift rate using the measured satellite drift computed by the navigation logic. Both estimated drift and estimated drift rates are used in the controller for computing the thruster firing time to provide minimum overshoot during a station change.

Global Positioning System

The Global Positioning System (GPS) consists of twenty-four navigation satellites in three 12-hour orbits (10,900 N miles), (Figure XI-3). (Van Dierendonck; Martin, 1977; Schaibly, 1976; Fuch and Wooden, 1977). This configuration insures that at least four GPS satellites will always be within the satellite's (vehicle using the GPS message for navigation) FOV.

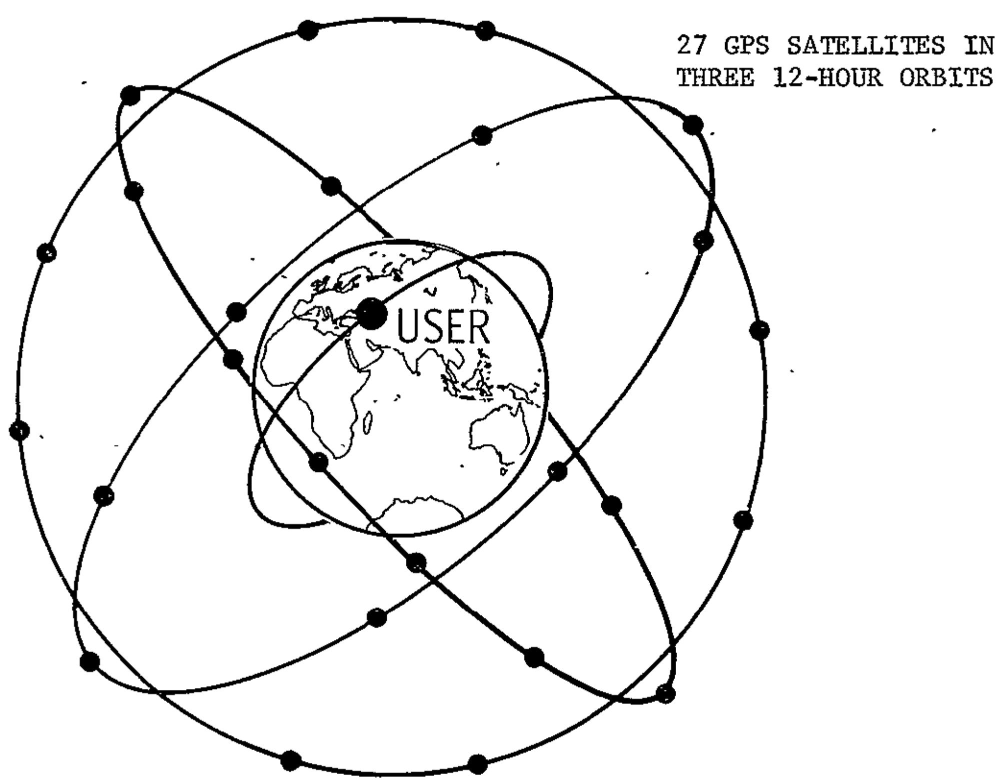


Figure XI-3 The Global Positioning System Configuration

Each of the GPS satellites transmits a navigational signal which contains information about the vehicle's time at signal transmission, its clock correction relative to a master time, ephemeris parameters, the health of all GPS space vehicles, and text messages.

The navigation message is transmitted in two separate codes on both the L_1 and L_2 frequencies. The C/A (clear/acquisition) code is transmitted only on the L_1 frequency and is intended for civilian use. The P (precision) code is transmitted on both frequencies and is intended for military and high precision uses. The navigation message is in the form of a 50 bit per second data stream modulated on the carrier codes. The data stream is common to both the C/A and P codes, but it is only possible to correct for ionospheric delay errors using the P code. Because this code is transmitted on two separate frequencies, processing of the phase information will yield propagation delays. Both codes are repeated periodically, the C/A code repeats every seven days whereas the P code has a 208-day cycle. In order to use the P code, the user must know the form of the code and the beginning of its cycle.

The navigation message is contained in a data frame that is 1500 bits long and repeats every 30 seconds. The data frame is divided into five subframes. "The first subframe contains the space vehicle's clock correction parameters and ionospheric propagation delay model parameters. The second and third subframes contain the space vehicle's ephemeris. The fourth subframe contains a message of alpha-numeric characters (for military use). The fifth subframe is a cycling of the almanacs of all the space vehicles (one per frame) containing their ephemerides, clock correction parameters and health." (VanDierendonck) Information received from one GPS satellite allows the user to determine the precise position and time of that space vehicle, and less precise positions and times of the three other GPS satellites yet to be acquired. Upon acquisition of all four GPS satellites, the precision position of each vehicle as well as a standard time is obtained. From this information, the user can solve for the range of each of the GPS satellites and triangulate to determine its own position (Figure XI-4). Range to a GPS satellite is found by solving the following equation:

$$R_i = c(t_R - t_{Ti}) - C \Delta t_{Ai}$$

where:

R_i = range to the i^{th} satellite

C = speed of light

t_R = time signal was received

t_{Ti} = time signal was transmitted

Δt_{Ai} = propagation delays

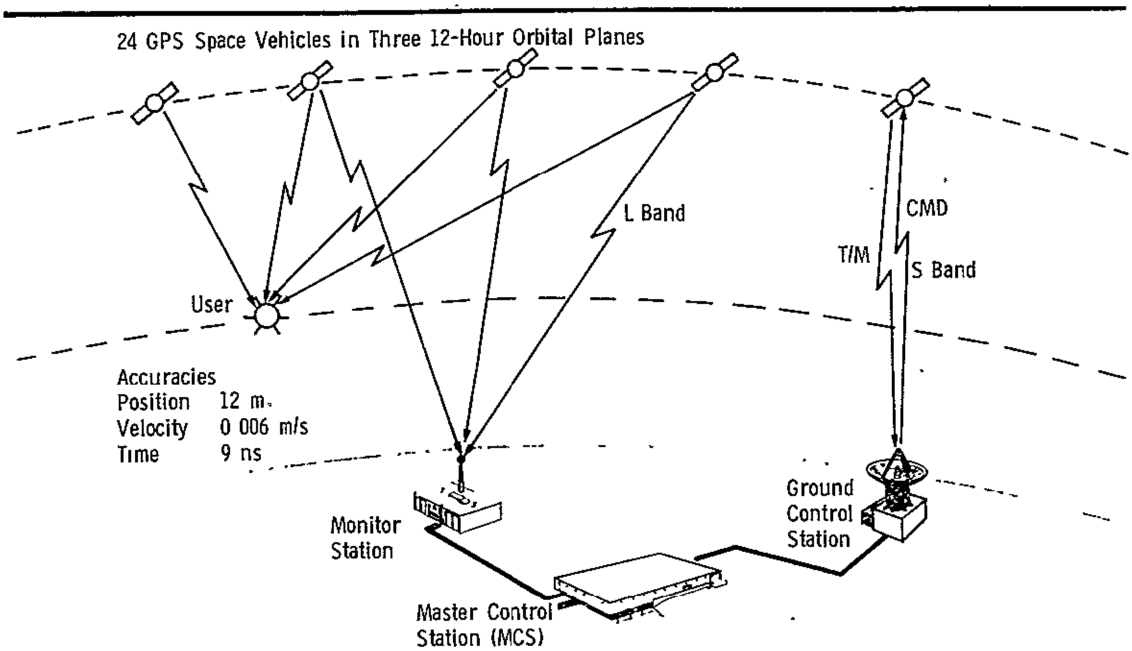


Figure XI-4 Position Determination Using GPS

Velocity can be determined through consecutive determinations of position separated by known time intervals. The determination of position, time, and velocity is accomplished using an onboard receiver/processor built by Magnavox and the Applied Physics Lab at Johns Hopkins University.

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Space Sextant

Space Sextant, an autonomous navigation system being developed for the Air Force with an orbital demonstration scheduled for 1980-1981 computes both position and attitude to very high accuracies. (Garcia and Owen, 1976; SS-ANARS Study 1975; SS-ANARS Study 1977). The Sextant consists of two Cassegrainian telescopes, an angle measurement head, gimbals that provide three angular degrees of freedom, and a reference platform consisting of a planar mirror, porro prism assembly, and a gyro package, (Figure XI-5).

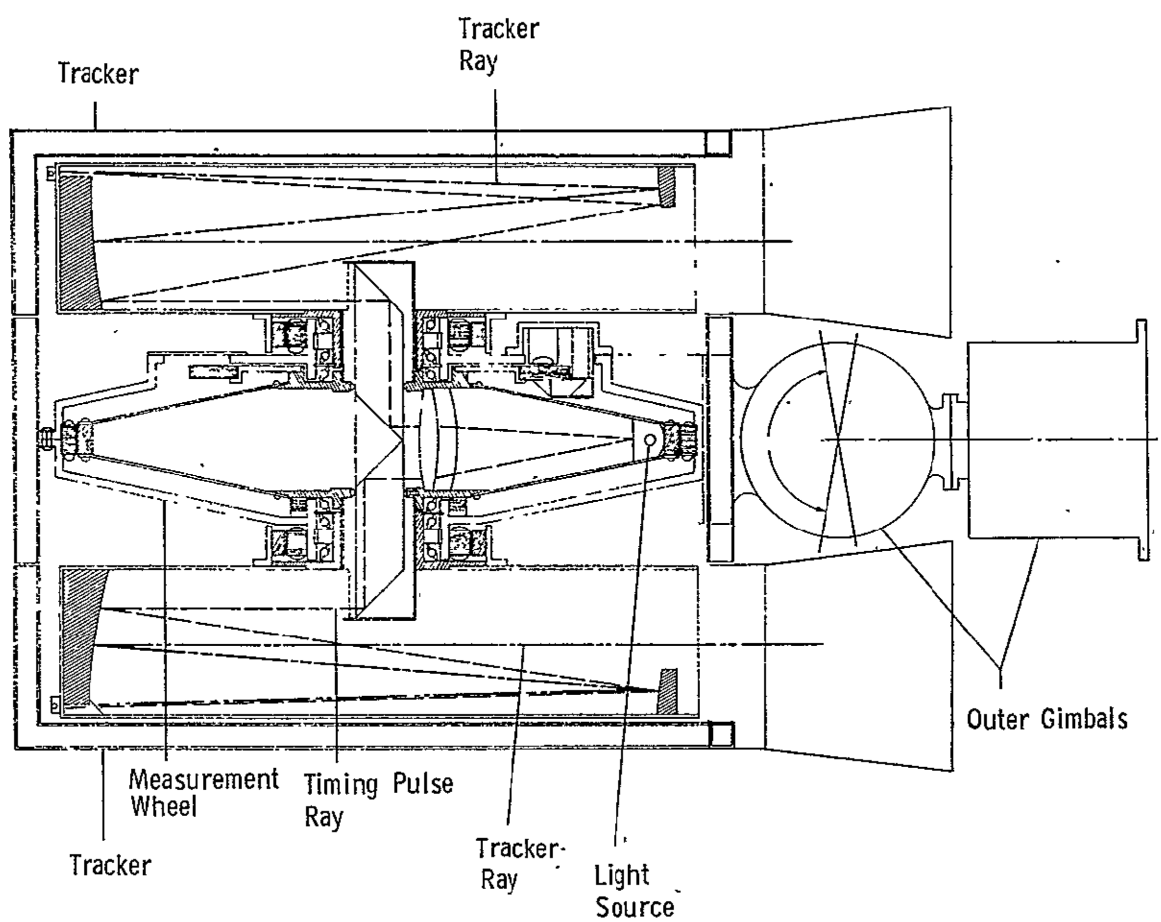


Figure XI-5 Space Sextant Configuration

Navigation using the Space Sextant is accomplished by making angular measurements between the bright limb of the moon and the brighter stars (visual magnitudes 3.0), (Figure XI-6). Reduction to the moon's center of figure (approximately the center of mass) including the compensation for asphericity and lunar terrain effects, is accomplished by the onboard software system. The essential data required to determine the spacecraft state are, therefore, the measured angle, the moon's stored ephemeris, a model of the lunar terrain, and the precisely recorded time of measurement.

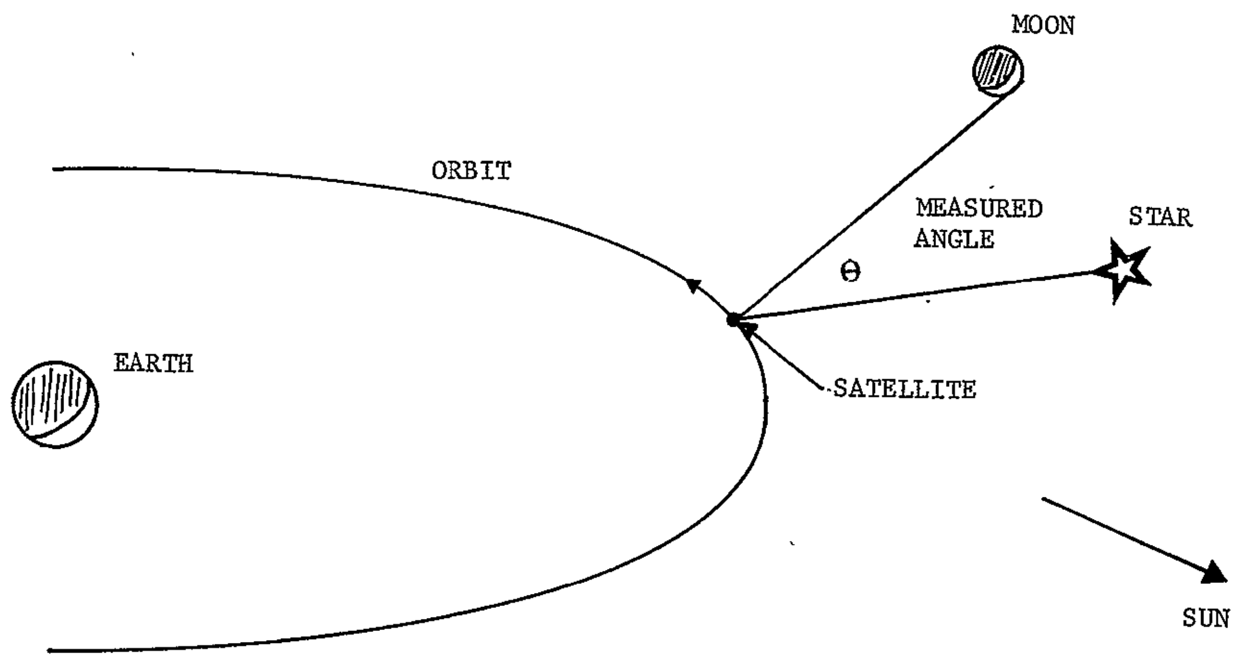
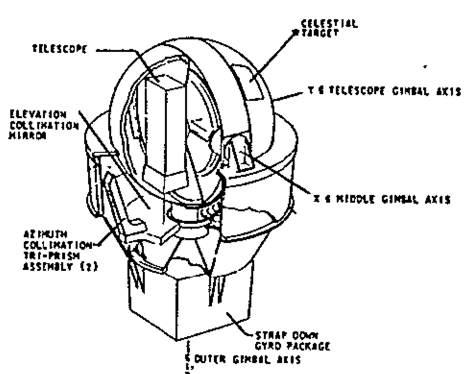


Figure XI-6 Space Sextant Navigation Concept

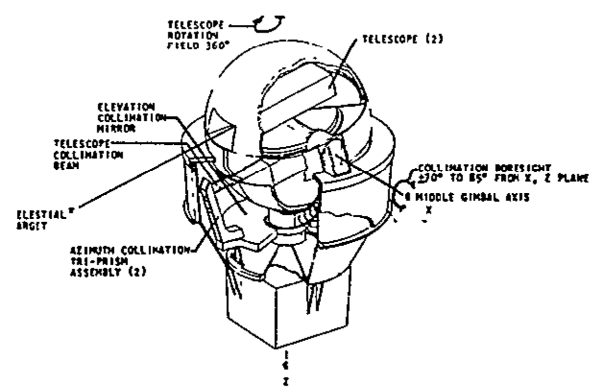
The moon's limb and the stars are tracked using the two independent Cassegrainian telescopes each of which focus their field of view onto an eight element detector. The outer four elements of the detector provide coarse tracking (four arc min) information while the inner four elements are used for fine tracking to within one arc second. The outputs of each of the elements are appropriately differenced and transformed to provide each gimbal with tracking servo error signals in order to maintain track. When the two telescopes have locked on fine tracking, the angle measurement head, incorporating a wheel, rotating at a rate of 10 revolutions per second, and a light source, sends out a very narrow illuminating window to each of the telescopes. The window rotates with the wheel at a constant angular velocity. As the window passes through the optical axis of tracker A, a timing sensor associated with the tracker sees an impulse of light at T_1. As the window continues its arc, it will pass through the optical axis of tracker B, at which time a second timing sensor will see an impulse at T_2. The included angle is then just:

$$\Theta = \omega \, (T_2 - T_1)$$

Attitude is determined by first making included angle measurements between two or more stars and a reference mirror fixed at the base of the sextant. A light source within the telescopes allows it to autocollimate off the mirror, thus providing a reference for one of the telescopes, (Figure XI-7). The included angle relative to the reference mirror fixes the attitude in one direction with respect to inertial space. The other attitude directions are fixed by making included angle measurements between one or more stars and a porror prism assembly mounted on the reference mirror in such a way that the prism is elevated above the plane of the mirror, again Figure XI-7. The two measurements are then processed to yield precise attitude information.



Sextant Collimation Off Mirror - Elevation Measurement



Sextant Collimation Off Porro Prisms - Azimuth Measurement

Figure XI-7 Attitude Determination with Space Sextant

Known Landmark Tracker-Area Correlator

Autonomous guidance of the spacecraft does not insure accurate sensor pointing because small perturbations in the orbit combined with bias errors and sensor pointing errors degrade the sensor pointing accuracy. Therefore, for evaluation of scientific experiments to be effective, it is necessary to determine the location of the ground scene within the sensor's FOV. The known landmark detector does this. (Gilbert and Majan, 1977).

The concept of using landmarks to register images is common in the field of image processing. Registration is carried out by correlating sensor data with data representing a significant feature within the area being searched. A landmark, also known as a Ground Control Point, (GCP), Registration Control Point (RCP), or anchor point is a small area, relative to the sensor's FOV, which contains a significant feature (highway crossing, airport, reservoir, etc.) located at a known latitude and longitude, (Figure XI-8). Landmark images and their locations are prestored in digital format in the satellite's onboard memory for access by the flight computer. By locating the position of a known landmark within the image data being received from the sensor, and by knowing the resolution of a picture element, the entire scene can be accurately registered.

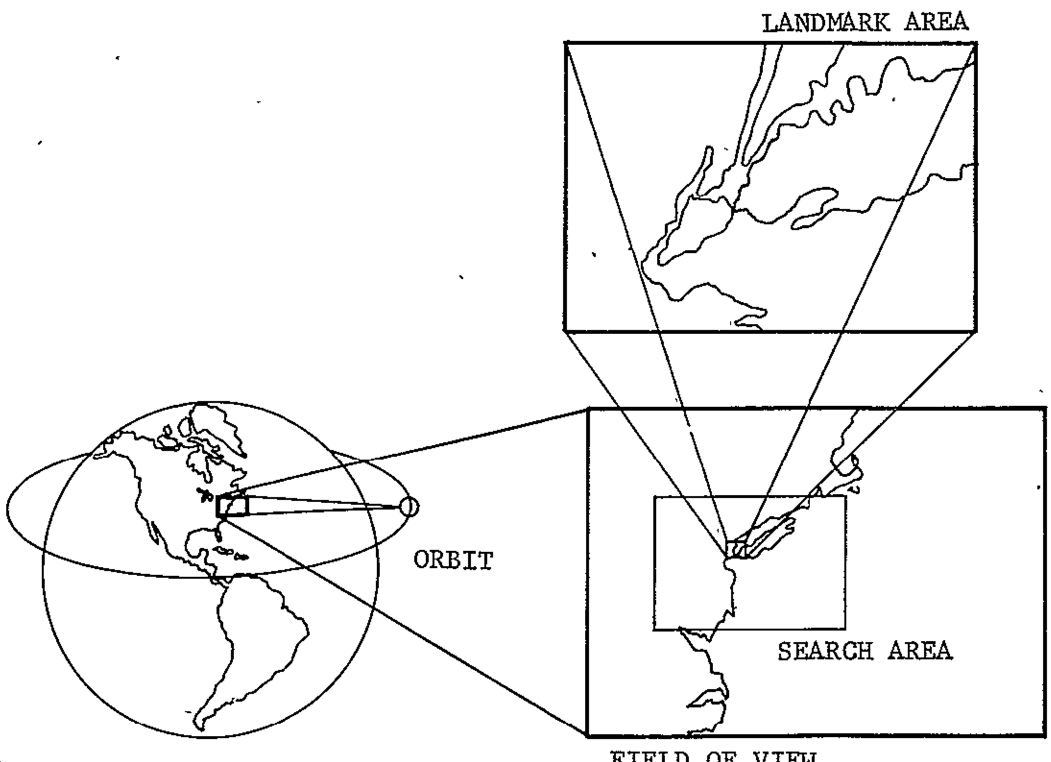


Figure XI-8 Landmark Tracker

It is not necessary to search for the landmark within the entire FOV but only within some smaller area called the search area, (Figure XI-8). The size of the search area is directly related to the satellite position uncertainty as well as the pointing errors. As these uncertainties decrease, the area in which the sensor/processor must search for a landmark decreases and so the time required for image registration is reduced as well.

Actual registration is accomplished by correlating picture element by picture element, each n x n area (corresponding to the size of the landmark chip) within the search area with the picture elements of the stored landmark. The area which produces the most favorable correlation coefficient represents a whole pixel registration of the scene, and can be labeled with the same latitude and longitude information characterizing the landmark chip, (Figure XI-9).

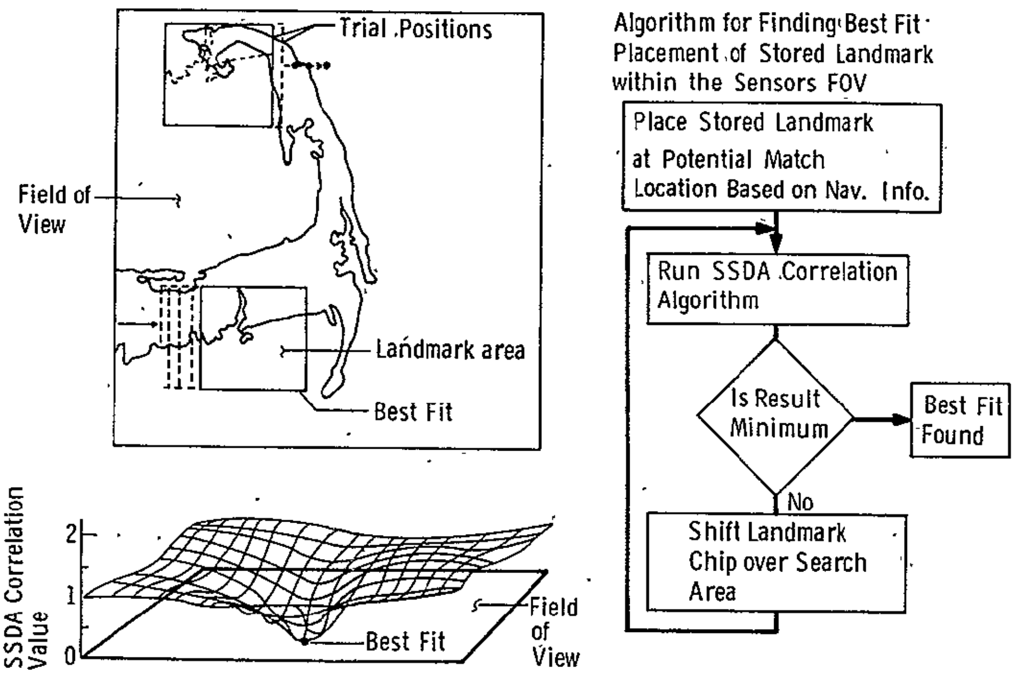


Figure XI-9 Registration of the Landmark

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There are two predominant correlation techniques being investigated at this time. The classical correlation coefficient involving square roots of sums and products requires that the calculations be carried out over every picture element pair (Scene and landmark) before a numerical answer is derived (Figure XI-10). This technique wastes a great deal of time because the processor must compute n^2 (n is the size of the landmark area) calculations regardless of whether the area being evaluated represents a best fit. Since one of the prime limitations of area correlation systems is the time required to find a position of best fit, the classical correlator may not be attractive. The sequential similarity detection algorithm (SSDA) involving sums of absolute differences between image pixels and landmark pixels significantly reduces the time required for registration (Figure XI-10).

The SSDA incorporates a decreasing cutoff threshold to allow partial processing of the correlation coefficient. During the process of correlation using the SSDA, if the sum exceeds some threshold value, the calculations are halted and a new placement is tried. If the previous threshold is not exceeded prior to completion of the calculations, then the placement of the landmark on the search area becomes the current best fit and the value of the sum becomes the new threshold.

Using the SSDA approach with a decreasing threshold to allow only partial processing, the scene can be registered quickly (2-3 sec) to whole pixel accuracy.

After the whole pixel registration has been found, techniques exist which allow further registration to subpixel accuracy (as accurate as 1/10th of a pixel). These techniques involve image resampling (interpolation between pixels) in order to change the image so that it represents the same scene shifted to one side by some subpixel distance. The resampled image is then correlated using the SSDA with the stored landmark to determine the goodness of fit. The resampling and correlating process is continued until the image has been registered to one tenth of a pixel accuracy. Registration to subpixel accuracy requires an additional 5-10 seconds, making total registration a 7-15 second process. The time required for data registration will be drastically reduced with the advent of the massively parallel processor. The time may be reduced to as little as a few seconds making the system very attractive.

Landmark registration allows the sensor/processor to determine the location of its FOV to a very high accuracy (8-80 meters for LANDSAT data). This information can be used to derive satellite position through the use of transformation matrices and attitude measurements.

$$\text{Sequential Search Detection Algorithm} = \left[\sum_{i=1}^{n} \sum_{j=1}^{n} (\text{Chip}(i,j)-\text{Chip mean})-(\text{Window}(i,j)-\text{Window mean})\right]$$

$$\text{Classical Correlator} = \frac{\left[\sum_{i=1}^{n} \sum_{j=1}^{n} (\text{Chip}(i,j)-\text{Chip Mean})(\text{Window}(i,j)-\text{Window Mean})\right]}{\left[\sum_{i=1}^{n} \sum_{j=1}^{n} (\text{Chip}(i,j)-\text{Chip Mean})^2 \sum_{i=1}^{n} \sum_{j=1}^{n} (\text{Window}(i,j)-\text{Window Mean})^2\right]}$$

Where:

Chip (i,j) is the ijth pixel value in the n x n array of pixels representing the landmark.

Window (i,j) is the ijth pixel in the n x n segment of the m x m array of pixels representing the overlapping area of the search area and the chip position.

$$\text{Chip mean} = \sum_{i=1}^{n} \sum_{j=1}^{n} \text{Chip}(i,j)/n^2$$

$$\text{Window mean} = \sum_{i=1}^{n} \sum_{j=1}^{n} \text{Window}(i,j)/n^2$$

Figure XI-10 Correlation Algorithums

Known Landmark Tracker - Linear Feature Detection

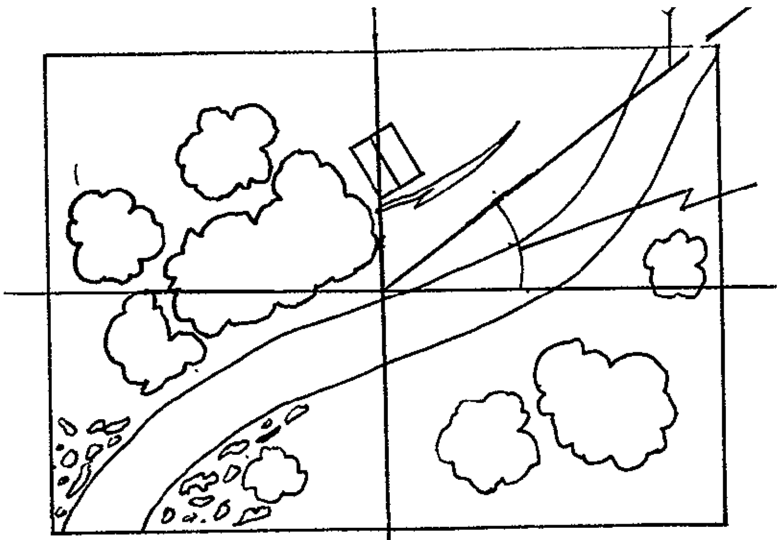
Both known and unknown landmark navigation systems have benefits and limitations. For low earth orbital applications, known landmark navigation approaches are typically less sensitive to pointing errors than tha unknown landmark approaches but identification of the landmarks is time consuming and requires an extensive onboard memory. The unknown landmark approaches, on the other hand, are attractive in that the task of landmark identification can be eliminated, making the system operationally simple and fast. However, the accuracy of such a system is limited.

The detection of known linear features using a single strapdown sensor combines several assets of both the known and unknown landmark navigation systems. (Kau, 1977). The detection of linear features requires few computations making it a time efficient process and knowledge about the landmark's position and orientation improves the navigation accuracy over that of unknown landmark detectors. Although the accuracy is improved above unknown landmark sightings, only one component of position is determined, thus limiting the accuracy of the navigation update and the usefulness of the technique for applications such as a sensor pointing algorithm.

Candidate landmarks consist of highways, coastlines, rivers, and other linear features. Due to the simple nature of linear features, a strapdown sensor with a relatively small FOV can be mechanized for detection of landmark crossings. Such a sensor might be composed of a CCD or a linear array of detectors. Digital images from these sensors are processed to derive linear landmark sighting information for system navigation updates. "Due to the deterministic signature of linear landmarks, deterministic image processing techniques such as thresholding and edge enhancement are used." (Kau, 1977).

Such techniques involve the detection of sharp discontinuities in the intensity profile of the scene data. Sharp discontinuities correspond to the edge of the linear feature involved. The measurement provided by the down sensor is the LOS-vector to the centroid of the segment of a linear Earth feature that falls into the sensor's FOV, (Figure XI-11). This is obtained from processing the discrete image for detection of a linear feature and for extraction of feature orientation and the segment centroid location.

a. Field-of-View



b. Digitized Image

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DIGITIZED IMAGE
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6 6 6 6 6 6 6 6 6 6 6 6 6 6 6 2 2 6 6 2 2 6 6 6 6 6 6 6 6 6 6 6 6 20 20 6
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6 6 6 6 6 6 6 15 15 15 15 6 2 2 2 10 10 6 6 6 6 15 6 6 6 2 2 2 20 20 6 6 6
6 6 6 6 6 15 15 15 15 15 15 15 6 6 10 10 6 6 15 2 2 15 2 2 2 20 20 6 6 6 6
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19 20·20 20 19 19 19 19 6 6 6 6 6 6 6 6 6 2 2 2 2 2 2 2 6 6 6 6 6 6 6 6 6
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Figure XI-11 Processing of the Linear Feature

Detection of a linear landmark provides one component of position information; for example, if the landmark is oriented perpendicular to the flight path of the satellite, the sensor/processor derives downtrack information. Similarly, if the feature is oriented parallel to the light path, cross track information is derived. Several landmark sightings are required to provide complete position information. Because complete position information is not determined in one sighting, errors might arise due to the drift of the satellite between sightings, and thus the load placed on the onboard filter is significantly increased.

Knowledge of the ground position of the sensor's FOV can be manipulated to provide satellite position information by combining attitude information from a gyro system with periodic updates from a sun or star sensor.

Artificial Landmark Tracker

High intensity point source radiators such as xenon search lamps are easily recognizable in video data received from multi-spectral scanners (MSS) mounted onboard the LANDSAT satellites. (AEOSIS, 1974; AEOSIS, 1975). Due to their recognizability, Draper Labs has suggested the use of artificial landmarkers, consisting of search lamps, placed at known locations on the Earth's surface to aid in the navigation of Earth viewing satellites. Their study was carried out with the intent of performing all processing activities on the ground and uplinking the navigation updates. However, the concept could be easily adapted for autonomous systems using onboard processing.

An onboard processor, using the image data taken from the Earth viewing sensor, would determine the line-of-sight direction to the high intensity point source, (Figure XI-12). This determination would yield two components of position information. An inertial reference unit and perhaps a star sensor would allow attitude determination, and with the knowledge of the landmark's location, satellite position could be calculated.

Interferometer Landmark Tracker

An Interferometer Landmark Tracker (ILT), together with a complementary inertial reference system and star sensors for precision attitude monitoring, will satisfy the navigation requirements of an autonomous navigation system for a wide variety of missions. (AEOSIS, 1976 and Aldrich, 1974). The ILT passively exploits ground radars with known geographic location to aid in fixing the position of the spacecraft. Unknown landmarks (radars

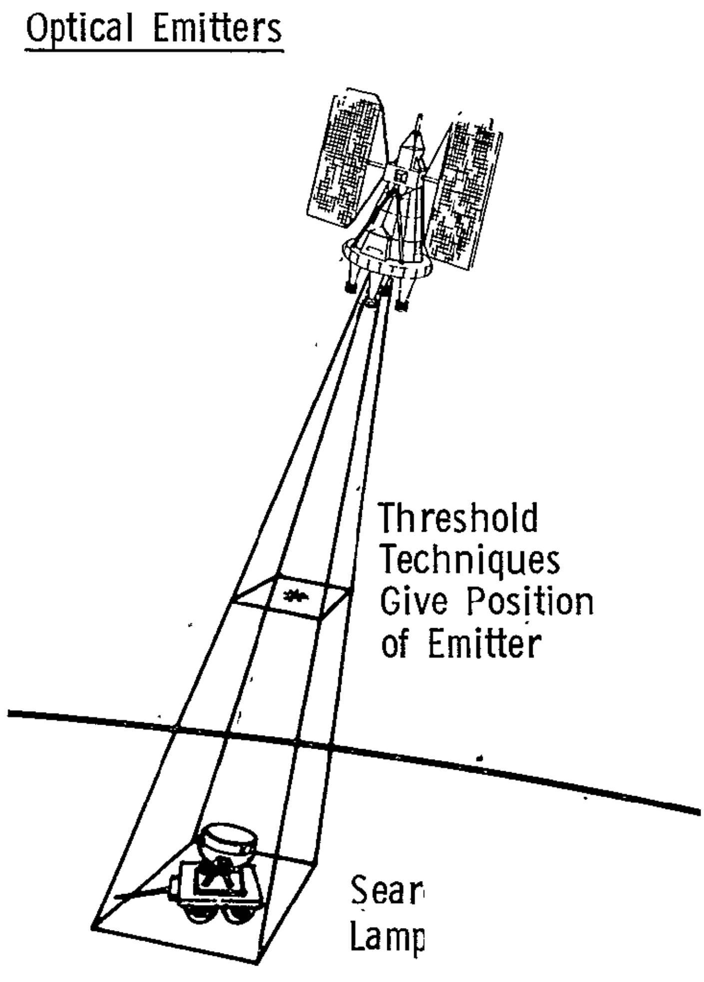


Figure XI-12 The Artificial Landmark Tracker

without known locations) may also be observed in order to provide an additional geometric constraint in orbit determination when several sequential measurements are made of the direction to the same ground position.

The ILT is a strapdown sensor composed of two orthogonal phase interferometers operating in the R/F frequency band and providing a 120° field of view (FOV). The two interferometers measure the phase difference of the radar signal received by two pairs of spiral antennas. Phase information allows precise determination of the angle toward the emitting source (Figure XI-13). Calculation of the line-of-sight direction to a transmitting radar, along with previous knowledge of the radar's position and the spacecraft's attitude, allows determination of the satellite's position. Attitude determination is provided by a strapdown star sensor in combination with an inertial reference unit.

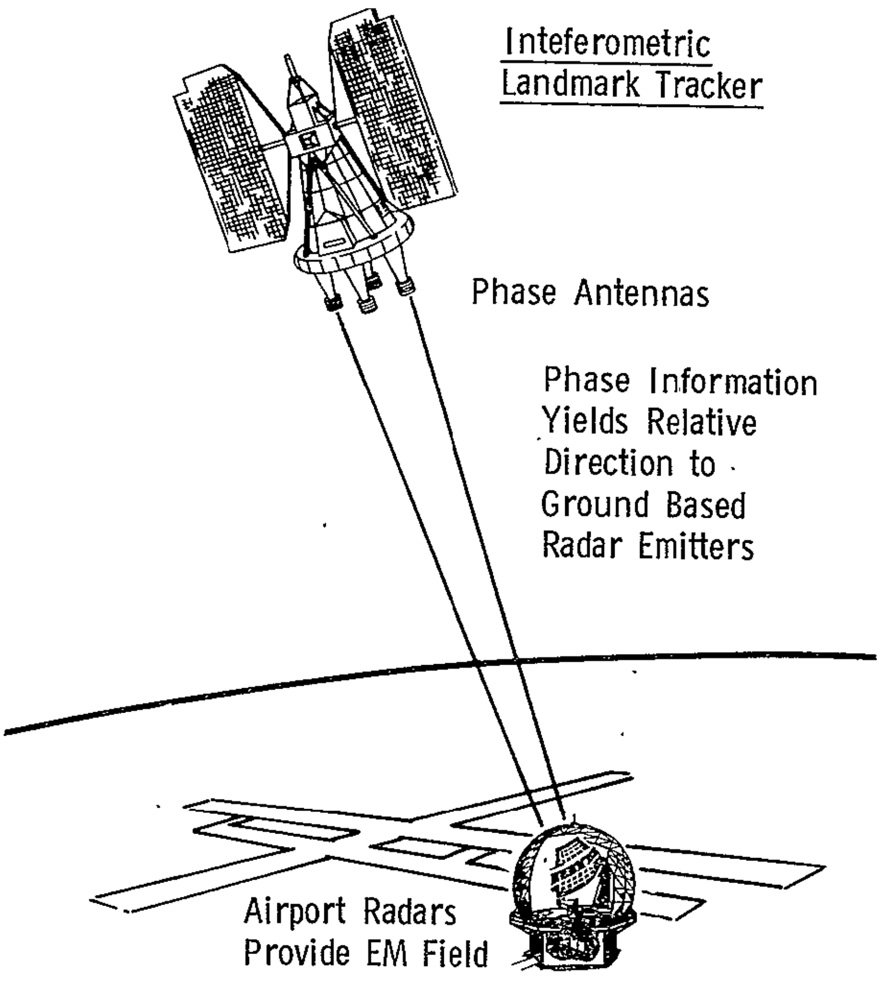
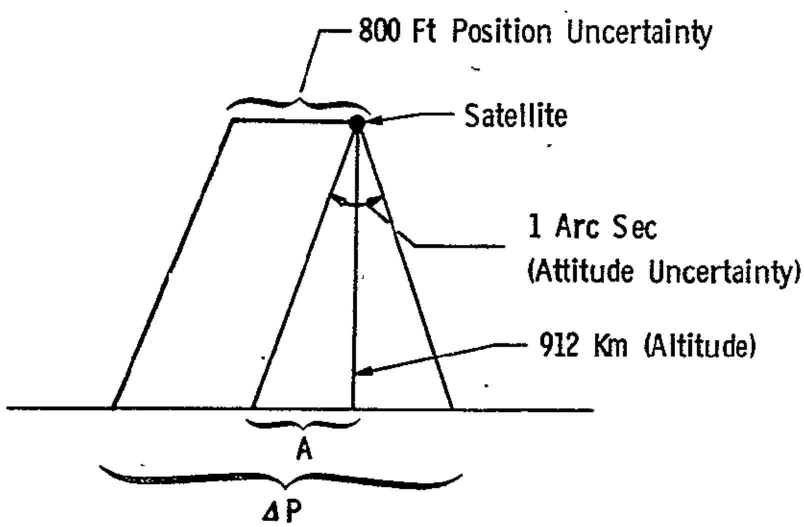


Figure XI-13 The Interferometer Landmark Traker

The following table is a technical summary of the seven candidate systems described above.

whose accuracies are quoted as being 244 meters (800 feet) for position and 1 arc sec for attitude, would have an uncertainty in ground position, as seen by the sensor of approximately 250 meters (820 feet) assuming no pointing errors for the sensor, (Figure XI-14). This uncertainty value may not be acceptable for some experiments such as feature identification or mapping functions; therefore, a landmark tracker should be used to minimize the ground position error to within acceptable limits (perhaps 30 meters (100 feet)).



$$\Delta P = 800 \text{ Ft} \times \frac{1 \text{ Meter}}{3.28 \text{ Ft}} + 912000 \tan (1 \text{ arc sec}) = 250 \text{ Meters}$$

Figure XI-14 Ground Position Uncertainty

Landmark trackers have the ability to determine the ground position of the sensor's FOV, but it is difficult to differentiate between errors in the position and attitude of the spacecraft. If the satellite were tilted along its pitch axis, the FOV of the sensor would be very similar to the FOV seen if the satellite were translated (maintaining its attitude) along its flight path, (Figure XI-15). Using landmark techniques, it is difficult to determine the position of the satellite without knowing the attitude and vice versa. However, the landmark techniques are ideal for image registration since they determine the absolute ground position of the sensor's FOV.

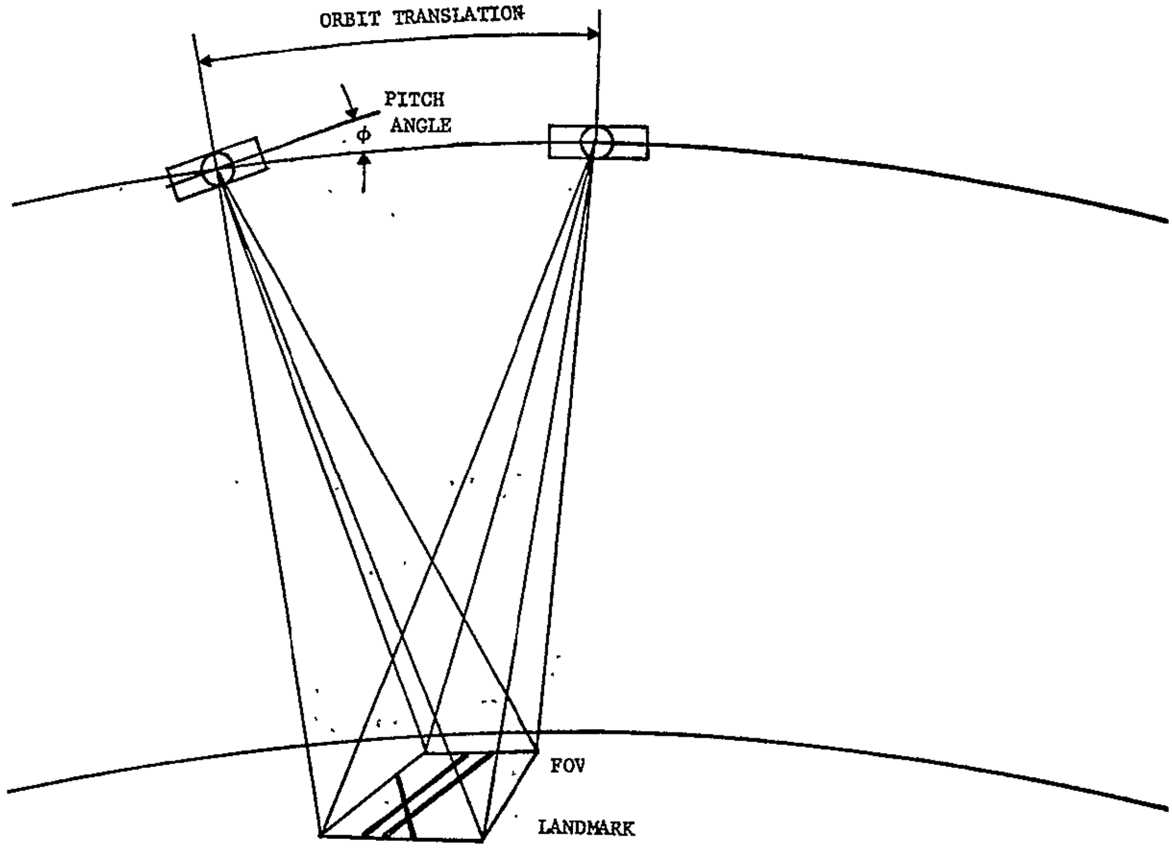


Figure XI-15 Position/Attitude Crosscoupling of the Landmark Tracker

Benefits and limitations of each approach to landmark identification are shown in Table XI-3. From this tradeoff, it is apparent that the area landmark registration approach is superior to other approaches. Although the technique requires significant computational support, processors are currently being developed which could handle the increased load. Also, using a single processor, the registration technique requires an excessive amount of time to provide a position update, however initial studies showed that this time could be drastically reduced by hardwiring the correlation algorithm. The technique of area registration not only provides inputs to a navigation system, but a general purpose image processor allows autonomous annotation of sensor data, inputs to a sensor pointing system, and deterministic data acquisition and transmission.

For Earth orbital missions the GPS system and Space Sextant are the prime candidates. It is difficult to label one of these systems as being superior due to the difference in concept. GPS will realize a greater position accuracy, but does not have a capatility for autonomous attitude determination. In addition, the ground support system is both costly and vulnerable. The space sextant provides both navigation and attitude information, but is limited by its size and weight. The use of either of these systems will depend totally upon mission requirements.

Table XI-3 Tradeoff of Landmark Tracking Techniques

Landmark Tracking Techniques	Benefits	Limitations
Artificial L/M Tracker	Fast Relatively small computational support for one L/M sighting	Requires upkeep of ground based emitters Accuracy limited to pixel resolution Small percentage of clouds can obscure the emitter Study was conducted with the idea of using ground processing System cannot provide ot other functions
Interferometer L/M Tracker	Fast Relatively small computational support for one L/M sighting	Requires upkeep of ground based emitters Limited accuracy Cannot provide other functions Requires sizeable antenna which is generally not a primary sensor
Linear Feature L/M Tracker	Fast No upkeep of Ground Systems	Severely effected by clouds Does not yield complete position information for each sighting Cross-coupling between crosstrack and downtrack errors

Table XI-3 Tradeoff of Landmark Tracking Techniques (Cont).

Landmark Tracking Techniques	Benefits	Limitations
Area L/M Tracker	Not severely effected by clouds when using the FILE cloud detector Provides other functions: Sensor Pointing Image Enhancement Data Management Yields complete position information No upkeep of ground systems required	Heavy computational support Could require significant time for registration

RECOMMENDATIONS

NASA has established very few programs in the area of autonomous navigation. The systems presented in this paper, with the exception of the interplanetary AGN system and artificial landmark tracker, have been funded and developed by DOD. NASA can significantly reduce the development costs of an autonomous system by adapting these systems to their unique requirements. Two systems which would be adapted are Space Sextant and GPS.

The Space Sextant, at present, is a large and heavy system but is capable of providing both autonomous navigation and attitude determination. The prototype sextant has not taken advantage of the new advances in superlight materials or LSI technobogy, and by combining these advances with the current configuration, a significant savings in size and power might be achieved. In addition, NASA's requirements generally do not call for the accuracy of the sextant, so a scaled down version which is smaller, lighter, and less espensive might be desirable. This would take funding, but the initial development expenditure is not required.

The GPS concept currently works only for orbits below the GPS orbit (10,900 n mi). There has been some interest in using the GPS signal above these altitudes and studies should be conducted on this possibility.

For earth observation satellites, the end-to-end data management and the navigation problems can be significantly reduced through the use of an area registration landmark navigation system. Studies have shown that sufficient technology exists to perform this task and that only limited hardware development would be required. It is recommended that a development plan for a landmark navigation system be established.

For interplanetary vehicles, the JPL AGN system is well designed and should be continued. The project is not only developing a good interplanetary navigation system but is pushing the state-of-the-art in CCD image devices and in onboard processors. These technology areas will, in turn, benefit other projects and broaden the scope of future automation.

While assimilating data for this paper, a strong bias against autonomous navigation systems was encountered. These biases must be overcome if automation is to progress at any reasonable rate.

REFERENCES

Aldrich, D. H., "Interferometer Landmark Tracker Navigation System," IBM Electronics System Center, NAECON 1974 Record, p. 592.

Barnea, D., and Silverman, H., "A Class of Algorithms for Fast Digital Image Registration," IEEE Transactions on Computers, Vol. C-21, No. 2, pp. 179-186, February, 1972.

Fuchs, A. J., and Wooden, W. H., "Autonomous Satellite Navigation with the Global Positioning System," AAS/AIAA Astrodynamics Specialist Conference, September, 1977.

Garcia, H. A., and Owen, W. J., "Design and Analysis of a Space Sextant for High Altitude Navigation," Journal of Spacecraft and Rockets, Volume 13, No. 12, pp. 705-711, December, 1976.

Gilbert, L. E., and Majan, D. T., "On-Board Landmark Navigation and Attitude Reference Parallel Processor System," presented at the NASA Flight Mechanics/Estimation Theory Symposium, Fall, 1977.

Kau, S. P., "Autonomous Satellite Orbital Navigation and Attitude Determination," presented at the NASA Flight Mechanics/ Estimation Theory Symposium, Fall, 1977.

Klump, A. R. et. al., "Autonomous Guidance and Navigation Design Concepts with Applications to a Jupiter Probe, Volume I and II," Jet Propulsion Laboratory, California Institute of Technology, Pasadena, California, March, 1977.

Kohlhase, C. E., "Autonomous Navigation Preparations for Future Unmanned Space Missions," The Institute of Navigation, Thirteenth Annual Meeting, June, 1975.

Martin, E. H., "GPS User Experiment Error Models," Magnovox Advanced Products Division, presented at the Institute of Navigation 33rd Annual Meeting, June 1977.

Schaibly, J. H., et. al., "Simulated and Projected Performance of the NAVSTAR GPS Control Segment," General Dynamics Electronics Division, August, 1976.

REFERENCES (Continued)

Srivastava, S., "Design of a Solar Ephemeris Synthesizer for LES 8/9," Massachusetts Institute of Technology, prepared for Air Force Systems Command, AD-752-250, October, 1972.

Srivastava, S., "Navigation Schemes for LES 8/9 Autonomous Stationkeeping System," Massachusetts Institute of Technology, prepared for Air Force Electronics Systems Division, AD-759-806, April, 1973.

Srivastava, S., and Brockett, R. W., "Design of an Autonomous Stationkeeping Controller for LES 8/9," Massachusetts Institute of Technology, prepared for the Air Force Systems Command, AD-775-144, February, 1974.

VanDierendonck, A. J., et. al., "The GPS Navigation Message," Technical note written at General Dynamics Electronics Division, San Diego, California.

"Space Sextant High Altitude Navigation System (SS-ANARS) Study," Phase 0 Final Report, MCR-75-5, Vol. 1-6, Martin Marietta Aerospace, Denver, Colorado, 1975.

"Design and Laboratory Testing of Self-Contained High Altitude Navigation System, Phase 1 - The Space Sextant Autonomous Navigation Attitude Reference System (SS-ANARS)," Phase 1 Final Report, MCR-77-196, Martin Marietta Aerospace, Denver, Colorado, 1977.

"Advanced Earth Observation System Instrumentation Study (AEOSIS)," Interim Report No. 2, Contract No. NAS5-21917, Charles Stark Draper Laboratories, Cambridge, Massachusetts, July, 1974.

"Advanced Earth Observation System Instrumentation Study (AEOSIS)," Interim Report, No. 3, Contract No. NAS5-21917, Charles Stark Draper Laboratories, Cambridge, Massachusetts, October, 1975.

"Advanced Earth Observation System Instrumentation Study (AEOSIS)," Final Report, Contract No. NAS5-21917, Charles Stark Draper Laboratories, Cambridge, Massachusetts, May, 1976.

XII ATTITUDE DETERMINATION

The design and implementation of an autonomous attitude determination system, as with any system, depends on several variable factors. The pointing accuracy requirements of the onboard scientific sensors along with the desired life of the spacecraft place the most stringent requirements on the design of the system. Other features such as sensor characteristics (does the sensor provide scanning? Do electronics require shielding? etc.) and the type of control system used also help to define the architecture of the overall system.

The variety of mission requirements has led to a number of unique attitude determination sensors. These sensors can be characterized in one of two ways. Either the sensor is part of a real-time reference system where attitude information can be obtained directly at any time, or the sensor takes periodic measurements. The first type of sensor is generally an Inertial Reference Unit (IRU), and the second usually consists of a sensor which measures a line of sight to some observable. The most common observables are the stars, planetary limbs, the sun, radar or other electromagnetic emitters. The number of sensors developed under each of these categories is too large to warrant a description of each one, so the emphasis will be placed on summarizing generic types of sensors.

In addition to the various sensors being developed for attitude determination, the basic algorithms used to determine attitude relative to some reference frame vary from mission to mission. Also the reference frame used varies as a function of mission requirements. For example, for nadir pointing vehicles, the earth reference frame is used, whereas 3-axis stabilized vehicles normally use the earth centered inertial frame.

Automation of the attitude determination system requires onboard processing capabilities. Variation of the mission requirements and configuration place certain demands on the architecture, speed, and processing capabilities of the onboard computer. If the computer is too slow, the system will be data bound, and if the processing capabilities are not adequate to solve the problem, the system will be processor bound. All of these factors must be understood and weighed in order to incorporate automation into the satellite system.

There are four basic considerations leading to the design of an autonomous attitude determination system. The context of this chapter is separated into these sections with the addition of a summary at the end. The organization is shown as follows:

- Types of missions requiring autonomous attitude determination
- The attitude problem and its solution
- Specific types of sensors
- Processing requirements for autonomous attitude determination
- Conclusion

Types of Missions Requiring Autonomous Attitude Determination

Attitude determination is closely related to the control system of the satellite, thus it is possible to classify determination schemes in terms of the type of control system used. Figure XII-1 gives a general breakdown of attitude determination systems using this method. The lefthand branch in this figure contains passive systems which require no automation of attitude determination and will not be treated in this report, even though a significant percentage of missions that have flown incorporated these schemes. Active systems are broken down into spin stabilized spacecraft and non-spinning vehicles. This delineation between spinning and non-spinning vehicles is the single most prevalent factor influencing attitude determination sensor and system design.

The problem of attitude determination and control is significantly reduced in a spin stabilized vehicle, but the accuracy is generally not as good as a 3-axis stabilized vehicle. By spinning a spacecraft around an axis, the vehicle obtains a rigidity about the other two axes, thereby reducing the attitude determination and control frequency required for those axes. The spinning motion can also be utilized by the attitude determination sensor in order to provide the scanning motion, thus making the component simple, reliable, and small. The absence of a scanning mechanism is one of the primary differences between sensors designed for spinning vehicles and those designed for non-spinning vehicles. Typical examples of sensors designed for spin-stabilized vehicles are horizon crossing indicators for earth orbiting missions and star mapping techniques for interplanetary and earth orbital missions. The information derived from these sensors can be used to compute spin rate, spin orientation, and spin direction for use in the vehicle control systems.

Of the non-spinning mission types, three axis stabilized vehicles have the most stringent set of requirements. This is because no form of passive or semi-passive (as in spin vehicles) control is provided. The lack of passive control requires that

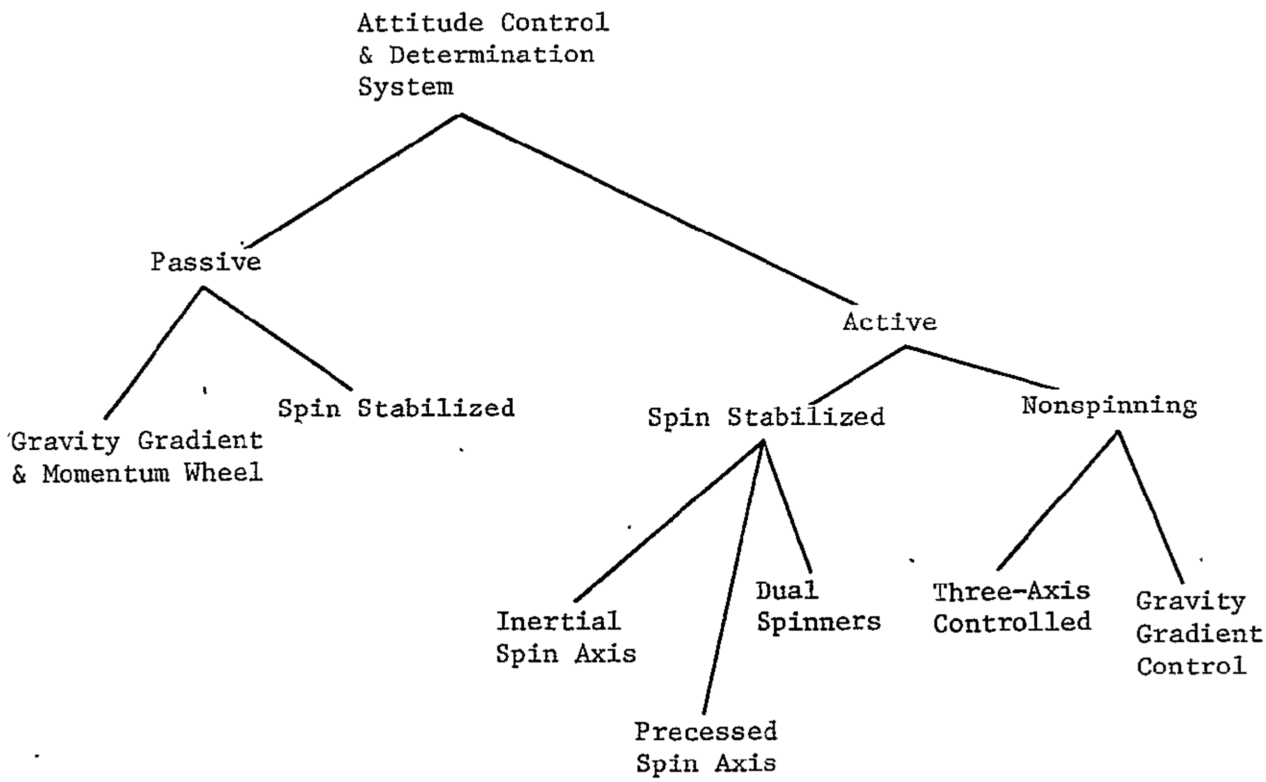


Figure XII-1 Hierarchy of Attitude Control System

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vehicle attitude be sensed and controlled with a greater frequency, thus placing a heavier requirement on the processing elements. In addition, the accuracy requirements of the attitude determination system are often much greater for the three axis stabilized vehicles. This often calls for a sensor with greater resolution. In addition to this requirement, sensors must provide their own scan which is usually accomplished by either gimballing a spin type sensor; using an image dissector type; or using solid-state electronics which can be scanned electronically. All of these features add to the system weight, size, and power requirements, but provide higher accuracy and a stable platform.

The Attitude Determination Problem

Attitude determination involves solving for the angular offsets between a coordinate system fixed within the body of the spacecraft and a reference coordinate system established by the mission requirements. The primary reference systems used to determine attitude are described below:

Inertial Reference Frame - The inertial coordinate system most commonly used has its origin fixed at the earth's center. The X axis is oriented along the 1950 epoch vernal equinox; the Z axis lies along the symmetrical earth rotation axis (North Pole); and the Y axis forms a right-hand triplet. This system is usually chosen when a star tracker or other star detection device is used because star catalogs have been established in these coordinates.

Earth Reference Frame - The earth reference frame, like the inertial reference frame, has its origin fixed at the earth's center. The X axis of this frame has its direction fixed along the zero longitudinal plane; the Z axis points through the North Pole as in the inertial reference frame; and the Y axis completes the ortho-normal coordinate system. This coordinate system is sometimes used for missions utilizing gravity gradient and other earth sensitive attitude sensors. It is also an important reference frame for researchers using data from scientific sensors onboard the vehicle because data must eventually be related to the latitude and longitude from which it came. A transformation matrix converts the coordinates of a point in the earth reference frame to coordinates in the inertial reference frame. If point P (Figure XII-2) is defined as a point in the earth reference frame with longitude ϕ and latitude λ, then the position of P in the inertial frame can be found through the equation

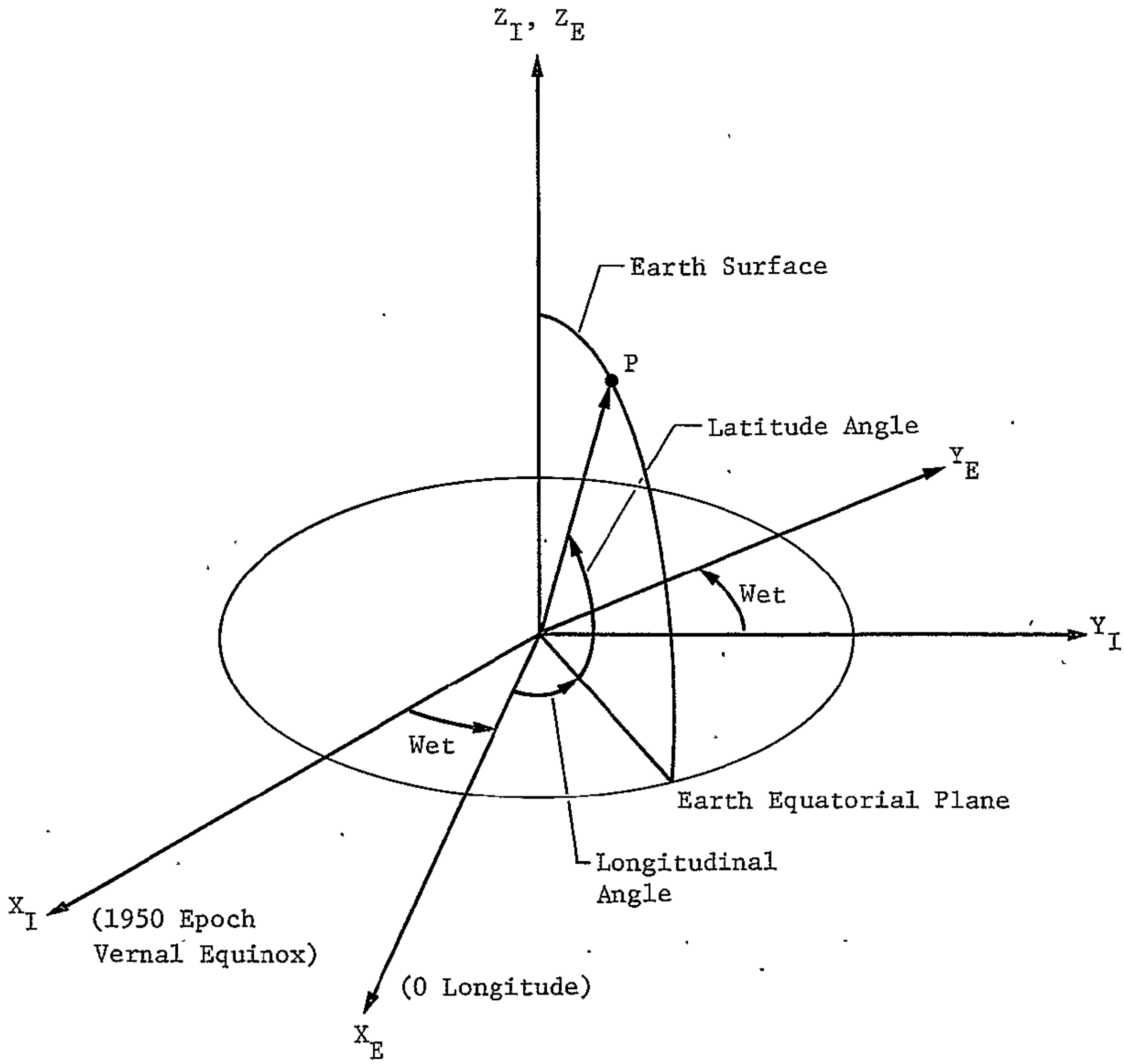


Figure XII-2 Inertial and Earth Reference Frame

$$\left\{U_I\right\} = \left[{}_I T_\varepsilon\right] \left\{U_E\right\}$$

where

$$\left\{U_\varepsilon\right\} = \left\{\cos \lambda_p \cos \phi_p,\ \cos \lambda_p \sin \phi_p,\ \sin \lambda_p\right\}$$

$$\left[{}_I T_\varepsilon\right] = \begin{bmatrix} \cos W_\varepsilon t\ , & -\sin W_\varepsilon t, & 0 \\ \sin W_\varepsilon t\ , & \cos W_\varepsilon t, & 0 \\ 0\ , & 0\ , & 1 \end{bmatrix}$$

W_ε = earth rotation rate

t = time

Local Vertical Frame – The local vertical referencé frame (Figure XII-3) is centered within the spacecraft. The X axis is oriented along a vector pointing from the earth's center to the spacecraft; the Z axis is located normal to the flight path pointing in the direction of the angular momentum vector; the Y axis completes the right handed system. For a circular orbit the Y axis is in the direction of the velocity vector. The transformation matrix from the L frame to the I frame is:

$$\left[{}_I T_L\right] = \begin{bmatrix} c\Omega c\phi - s\Omega s\phi ci, & -c\Omega s\phi - s\Omega c\phi ci, & s\Omega si \\ s\Omega c\phi + c\Omega s\phi ci, & -s\Omega s\phi + c\Omega c\phi ci, & -c\Omega si \\ s\phi si\ , & c\phi si\ , & ci \end{bmatrix}$$

where

c – cos

s – sin

Ω – longitude of ascending node

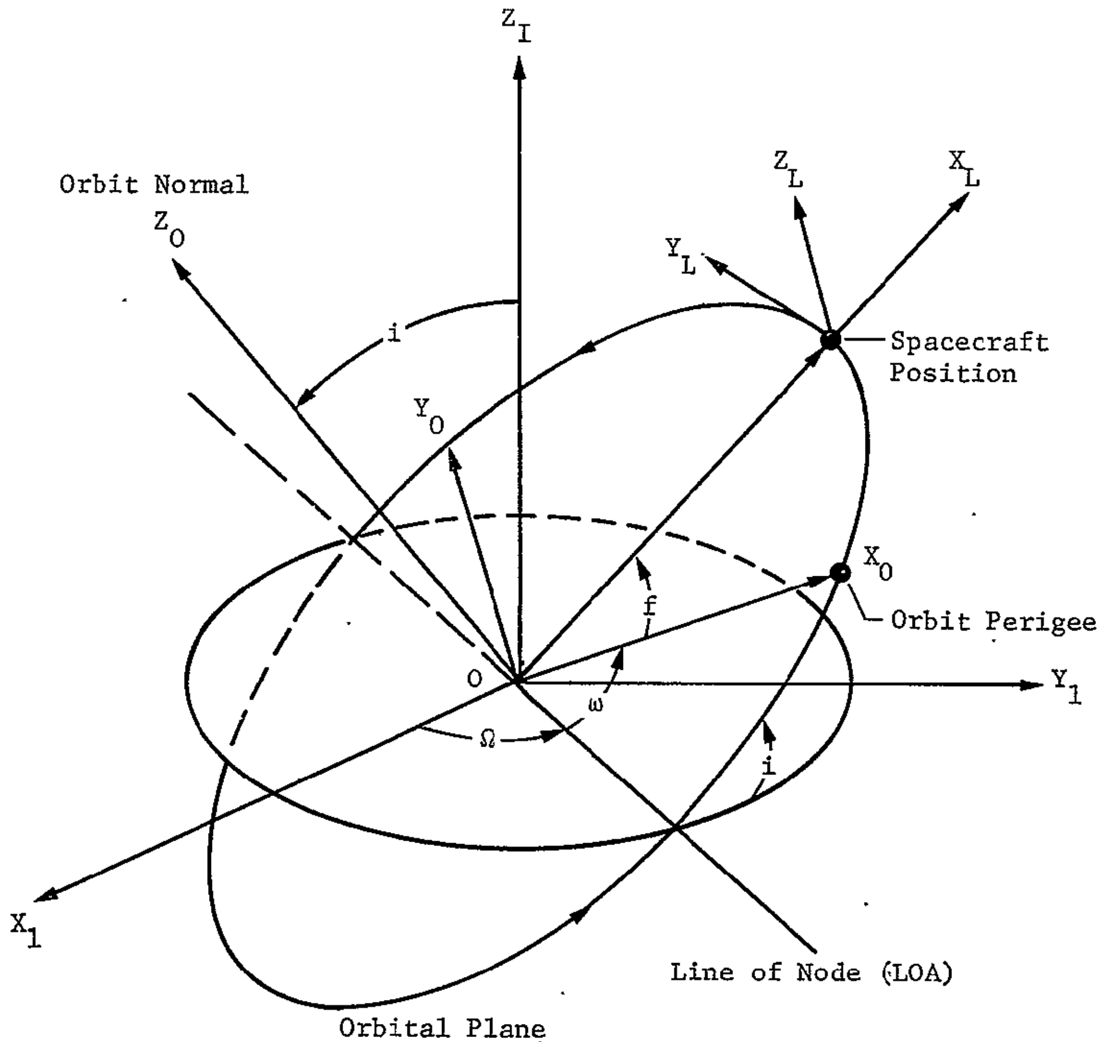
i – angle of inclination

ω – argument of perigee

f – true anomaly

$\phi - \omega + f$

Body Fixed Frame – This coordinate system is centered in the spacecraft and represents the structure of the vehicle. Attitude is determined by resolving two or more vectors in one of the reference coordinate systems relative to the body frame. A transformation of this data is then made to derive pitch, yaw, and roll errors. There are four ways to represent the spacecraft attitude: 1) direction cosines; 2) quaternions; 3) Euler



Where: Ω - longitude of ascending node

i - inclination angle

ω - argument of perigee

f - true anomaly

Figure XII-3 Rotation Between I-Frame, O-Frame and L-Frame attitude angles; and 4) Gibbs vector representation.

Direction cosines are the most straightforward method. The other methods can be derived from the direction cosine representation and are directly related to each other. The Euler attitude angle representation provides a better physical visualization of the attitude and is generally the quantity delivered to the user even though the system may use a different set of internal variables.

For onboard attitude processing, the direction cosine matrix variables and quaternions are the two methods most widely used. The Euler angle representation suffers from the singularity problem and special function usage (sine, cosine functions, etc.) which may introduce extra errors during computation. They are also more difficult to implement in digital computers.

Attitude determination using the direction cosine method utilizes the following differential equation:

$$[\dot{c}] = [c][\Omega]$$

with initial conditions $\left[c\ (t = t_o)\right] = \left[c_o\right]$ known

$\left[c\right]$ is the nine element direction cosine matrix and Ω is a skew symmetric matrix defined by

$$[\Omega] = \begin{bmatrix} 0 & -W_z & W_y \\ W_z & 0 & -W_x \\ -W_y & W_x & 0 \end{bmatrix}$$

W_x, W_y, and W_z are the body axes angular rates expressed in the inertial frame and are obtained from the attitude sensors. The direction cosine matrix contains nine state variables to be integrated. This can be reduced to six variables and the other three are derived by requiring the direction cosine matrix always remain orthonormal. Orthonormality is maintained through the following method. If c = (i,j,k) is the direction cosine matrix then the vectors i and j can be found directly by integrating the rate gyros. The following requirements are then applied to i and j:

$$i \cdot j = 0 \quad \text{and} \quad \begin{matrix} |i| = 1 \\ |j| = 1 \end{matrix}$$

This assures that i and j are normal to one another and are unit vectors. k can then be computed by calculating the crossproduct k = i x j. This method, however, introduces extra skew errors in the computation.

The attitude determination differential equation using the quaternion method is given below:

$$[\dot{q}] = 1/2[\Omega^*]\,[q],$$

with initial conditions $\left[q\ (t = t_o)\right] = \left[q_o\right]$ known.

The quaternion state vector q is a (4x1) vector containing four state variables, $[q]^T = [q_1, q_2, q_4]$. The matrix Ω^* is a 4 x 4 skew symmetric matrix defined by:

$$\left[\Omega^*\right] = \begin{bmatrix} 0 & W_z & -W_y & W_x \\ -W_z & 0 & W_x & W_y \\ W_y & -W_x & 0 & W_z \\ -W_x & -W_y & -W_z & 0 \end{bmatrix}$$

The tradeoff between the use of direction cosines or quaternions has been investigated by many, i.e., Wilcox of TRW, Mortenson of UCLA, Marcus of MIT, etc. In general, the quaternion method is favored due to the following advantages:

1. It has inherent zero skew error and reduces the scale and asymmetry error. It, therefore, reduces the computational error.

2. It uses only four state variables instead of 9 for the direction cosine method, therefore it reduces the storage requirement and iteration steps in the numerical integration.

3. The direction cosine requires three calculations to insure orthonormality whereas the quaternion requires only one.

Attitude Sensors

Attitude determination systems exist in a variety of configurations of different sensors (Figure XII-4). Most of the advanced systems incorporate some sort of Inertial Reference Unit (IRU) and a reference update system. This combination is advantageous because it minimizes the shortcomings of an IRU and a reference update unit. The IRU is used to maintain a reading of the spacecraft attitude. However, due to the properties of gyroscopes, this reading will drift from the actual value. At some limit of attitude uncertainty, the reference unit will then update the gyro by providing a precise input of the attitude. Attitude uncertainty will be a function similar to that shown in Figure XII-5.

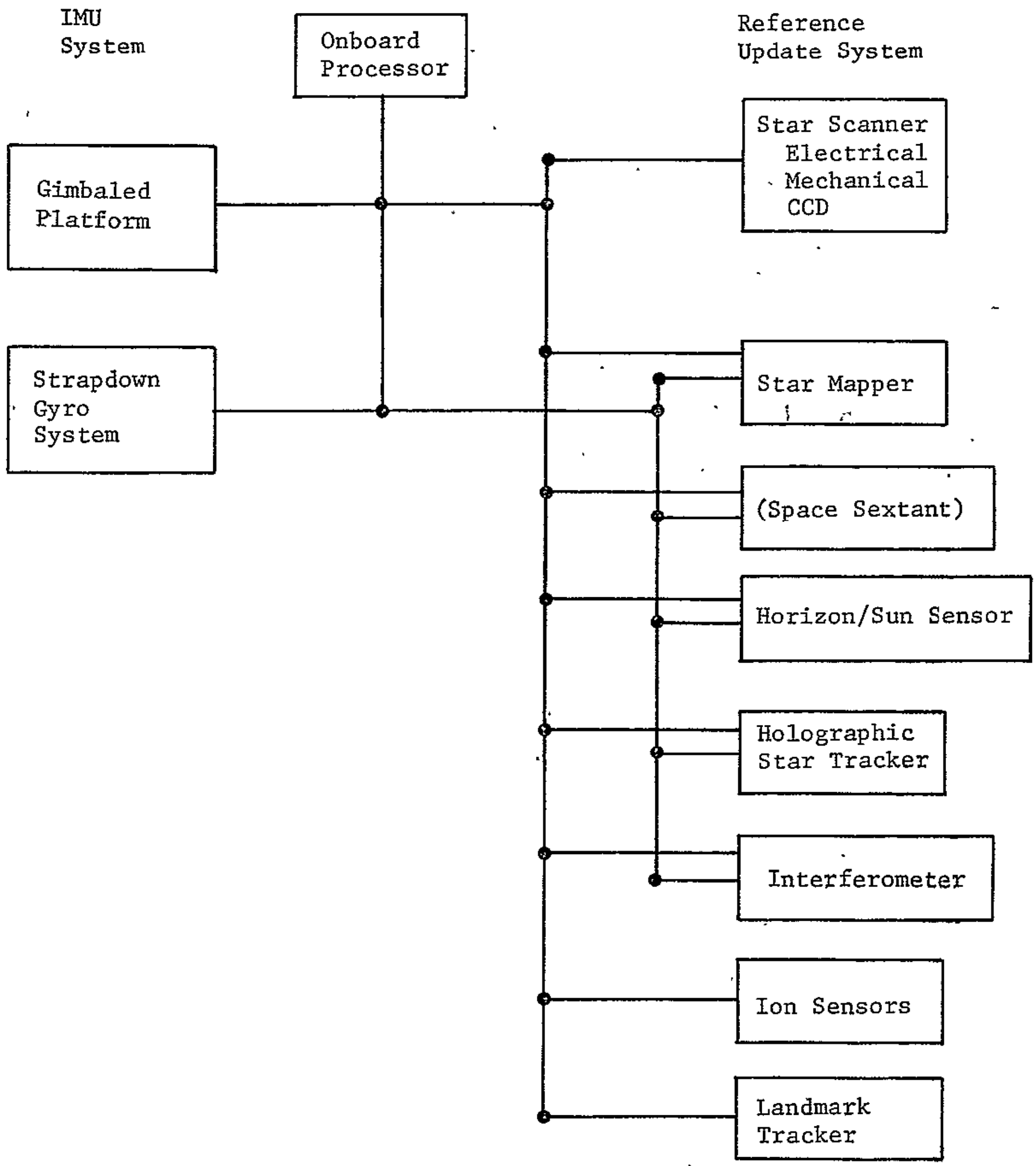


Figure XII-4 Alternative Attitude Determination Sensor Configurations

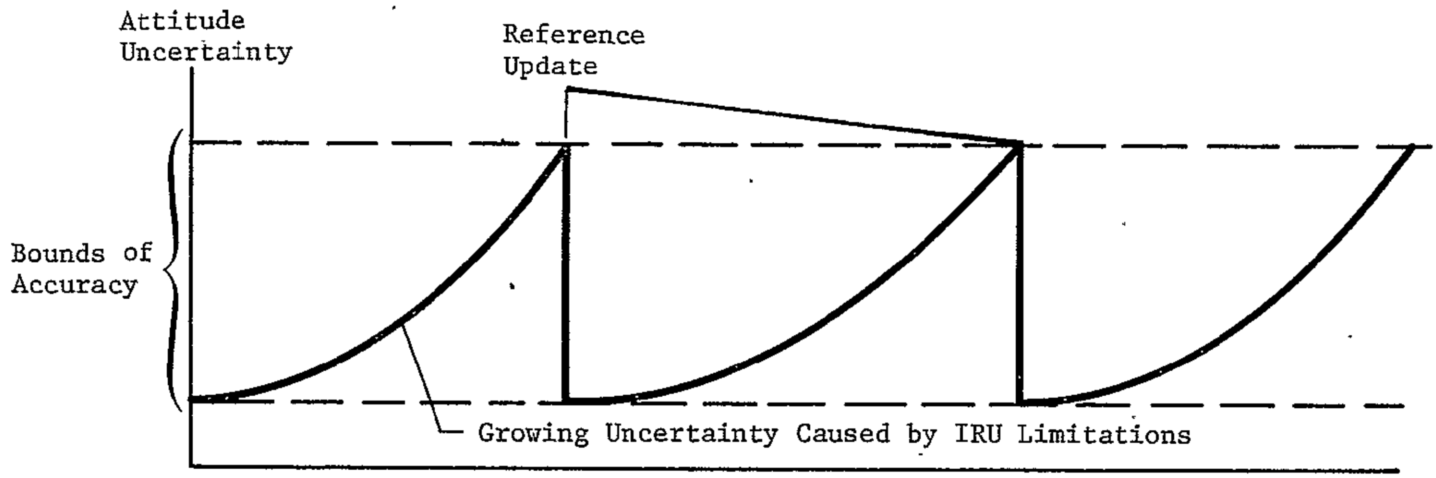


Figure XII-5 Attitude Uncertainty As A Function of Time

Inertial Reference Unit

The two basic types of Inertial Reference Units are the gimballe platform and the strapped down gyro system. A comparison betwee these two generic systems was performed under an internal research task based on the system concept, sensor impact, software impact, and calibration impact (K. Yong, et al., 1978). A summary of this tradeoff is provided in Table XII-1. It was generally concluded that gimballed platforms are superior for short duration missions due to the limited software and calibration requirement. However, for long term missions where reliability and accuracy become the major driving forces, gimballed systems should not be considered. The advantages of using a strapdown system over the gimballed platform in a long life mission are listed below:

1. Eliminates all errors associated with platform stabilization. This increases the long-term reliability.

2. The gimballed platform has the limitation of working within a defined range for each gimbal. The strapdown system, being free from gimbal lock, allows all attitude motion.

3. Due to the absence of mechanical platform gimbals, the strapdown system is smaller in size, lighter in weight, more rugged in mechanical structure and consumes less power.

Strapdown inertial reference systems were studied extensively under NASA contract OADS (NAS4-23428) and a DOD contract. The initial study concluded that two degree of freedom (TDF) gyro systems may be considered above single degree of freedom (SDF) gyros. The advantages of using a TDF gyro package are as follows:

1. Provides higher reliability for the same number of gyros used (Figure XII-6).

2. Less effect of sensor accuracy

3. Provision for more redundant measurements for better data reduction.

A tradeoff study was performed between various TDF gyros. Results of the comparison between TDF gyros are summarized in Table II. The gyros considered are Litton's G-1200 and G-6 series TDF gyros and the DRIRU II's SDG-5. Referring to XII-2, it is observed that the SDG-5 is superior to the G-6 gyro in almost all respects, and although the stability performance of the G-1200 is slightly better, its dynamic range is far too narrow to be applied in a strapdown environment. Therefore, the SDG-5 gyros are considered to be the best available

Table XII-1 Strapdown/Gimballed IMU Comparison

	Concept	Sensor Impact	Software Impact	Calibration Impact
Gimbaled IMU	Inner Platform remains inertially fixed. Resolver/encoder outputs measure space vehicle attitude (Euler angles) directly.	Gyros are operating a benign environment that is ideal for maximum performance. Drift rates of less than 0.01°/h are obtainable.	Mimimal compared to strapdown system.	IMU calibration is minimal compared to strapdown system.
Strapdown	Gyro outputs are intergrated to estimate spacecraft attitude.	Very demanding in areas of scale factor stability, linearity and asymmetry coning motion, alignment vibration, noise and bandwidth.	Very demanding in computational requirements such as truncation, quantization, roundoff and bandwidth. Sensitive to spacecraft motion.	Requires calibration of gyro scale factor, drift, alignment.

Table XII-2 Comparison of Performance

Parameter	Units	SDG-5	G-6	G-1200
G-Insensitive Drift				
Absolute Value	deg/h	<0.5	4.0	0.23
Stability				
Random Drift	deg/h 1σ	0.0005	0.003	0.0009
Shutdown	deg/h 1σ	0.0016	0.01	0.0023
Long-Term	deg/h/yr	0.01	0.03	0.015
Temperature Sensitivity				
Uncompensated	deg/h/°F	0.00059	0.002	0.0014
Compensated	ppm/°F	1.0	Not Compensated	
G-Sensitive Drift				
Absolute Value	deg/h/g	<1.0	5.0	0.23
Stability				
Continuous Operation	deg/h/g 1σ	0.0007	0.0003	0.0005
Shutdown	deg/h/g 1σ	0.008	0.008	0.0035
Long-Term	deg/h/g/yr	0.02	0.04	0.015
Temperature Sensitivity				
Uncompensated	deg/h/g/°F	0.0032	0.02	0.0017
Compensated	ppm/°F	<1.0	Not Compensated	
Torquer Scale Factor				
Absolute Value	°/h/ma	160	1250	33
Stability	ppm 1σ	27	50	50
Linearity	ppm Peak	25	30	No Date
Asymmetry	ppm Peak	3	30	No Date

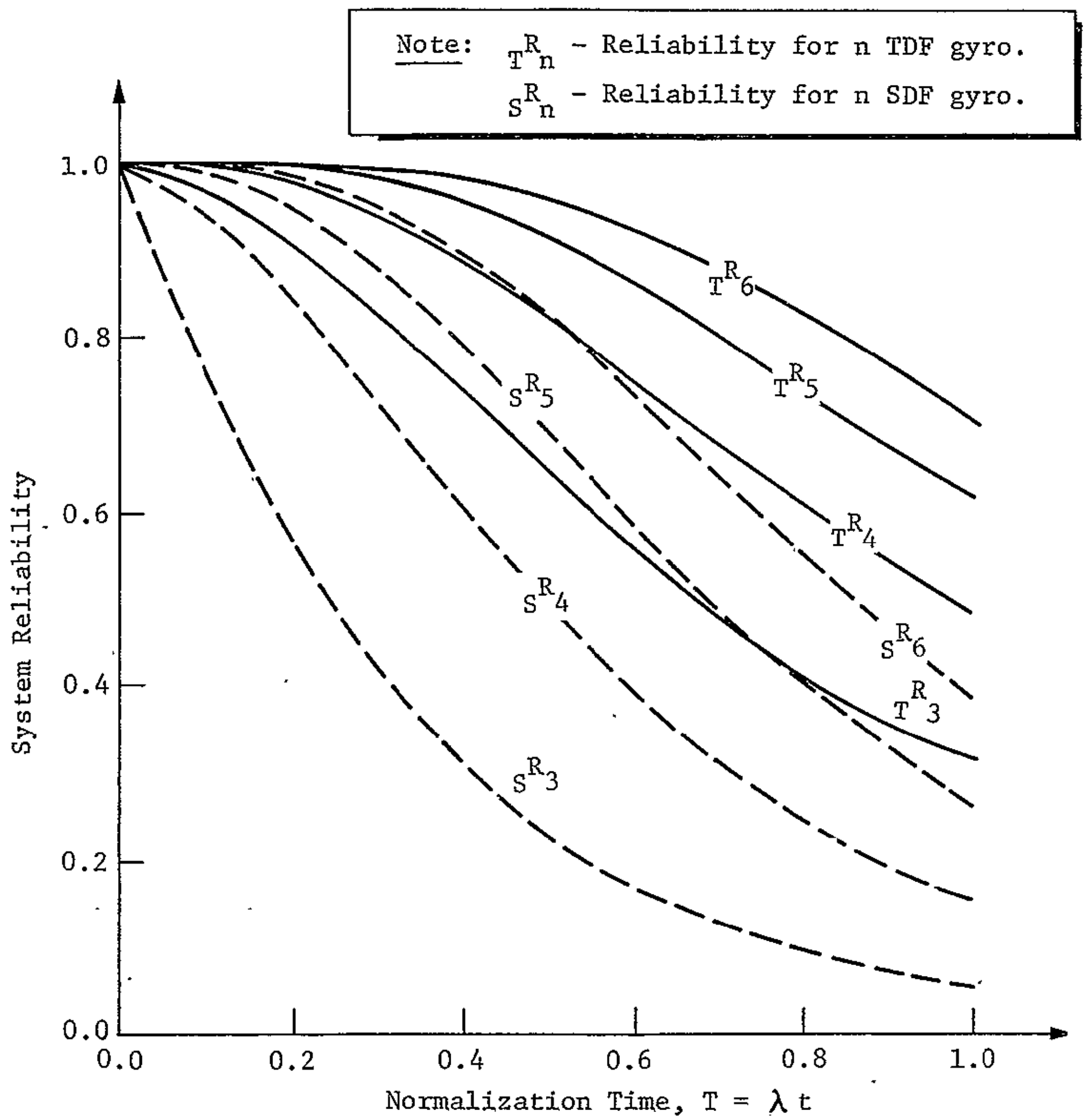


Figure XII-6 Raliability of N and Two-Degree-of-Freedom Gyros with Time

The DRIRU-II (Digital Redundant Inertial Reference Unit-II) is the NASA standard IRU and is manufactured by Teledyne. It consists of three SDG-5 two-degree-of-freedom (TDF) dry tuned gyros mounted orthogonally in a single unit (Figure XII-7). The characteristics of the DRIRU-II system are shown in Tables XII-3 and XII-4. The Bendix IRU systems consisting of six SDF 64 PM gyros has been added for comparison.

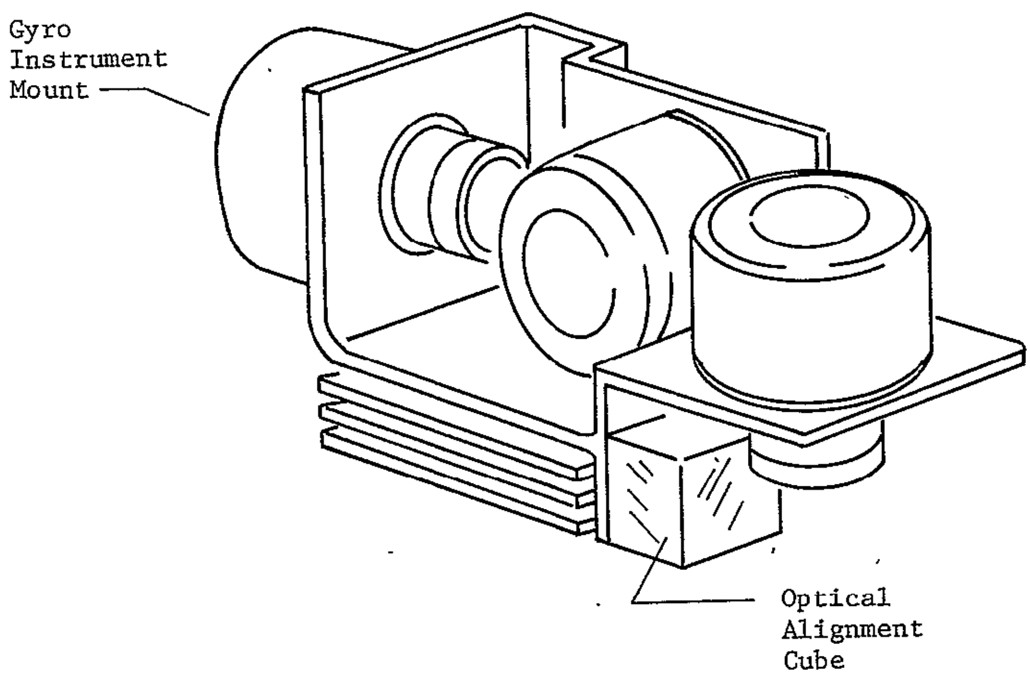


Figure XII-7 DRIRU-II Configuration

As stated earlier, the DRIRU-II is far superior to other existing gyro systems operating in a strapdown environment. Also, in a system configuration, two DRIRU-II units can be used with one as a primary and the other as a backup. With this configuration, the system redundancy is extremely high.

Table XII-3 Gyro Error Characteristics Comparison

	Driru-II	Bendix 46 PM
Random Drift, deg/h - 1σ	0.0005	0.0005
Long-Term Bias Stability, deg/h/yr - 1σ	0.01	0.09
Torquer Scale Factor, deg/h/ma - 1σ	0.6	230.0
Torquer Linearity, ppm - 1σ	25.0	37.0
Torquer Asymmetry, ppm - 1σ	3.0	27.0
Angular Rate Capability, deg/s	100.0	20.0
Angular Momentum, gm-cm^2/s	1×10^6	0.43×10^6
Aniscelastic Drift, deg/h/g - 1σ	0.01	0.04

Table XII-4 Comparison of Physical Characteristics and Reliability

	Driru-II	Bendix System
Weight, lb	25.0	65.0
Power, Wa	21.0	115.0
Cost	200k	800k
*Reliability (2 years)	0.958	0.914
*Based on 6-gyro configuration for both systems.		

Steller Sensors - Steller sensors measure a Line of Sight (LOS) angle to a star whose position is known. Each star sighting yields two components of attitude information which is used to update a current estimate. There have been several approaches to solving the star sighting problem, and the sensors which have evolved can be separated into four classes:

1. Gimballed star trackers;
2. Electronically scanned star trackers;
3. Star mappers; and
4. CCD star trackers.

Gimballed Star Tracker - The gimballed star tracker searches for and acquires known stars using a mechanical gimbal action. The sensor has a relatively small instantaneous field of view (FOV) (1^{o} x 1^{o}) with the gimbal motion providing a much larger effective FOV. Pointing control is usually provided through the use of a null seeker electronics package which causes the gimbals to move so that the star remains centered within the instantaneous FOV. Gimballed star trackers, such as those used on ATM and OAO, have achieved accuracies of 30 arc seconds. Other gimballed star trackers have accuracies ranging from 1 to 60 sec. This type of sensor, however, has several serious disadvantages:

1. Gimbal apparatus reduces long term reliability.

2. Possible to track either the wrong star or particles such as paint chips.

3. Errors in determining star position with respect to null, and gimbal angle readout errors effect the overall accuracy.

4. Increased size and weight due to gimbal mount.

A summary of some of the existing gimballed star trackers is shown in Table XII-5.

Electronic Star Tracker - This type of star tracker is an electro-optical device which electronically scans a small instantaneous FOV over a larger effective FOV in order to acquire stars brighter than some fixed threshold. The scanning pattern is usually produced by an image dissector tube and associated electronics. During acquisition the scanning pattern is a raster type until a star is detected. At this point, the raster scan is normally halted and the star is tracked using a much smaller scan pattern until the star leaves the effective FOV.

The electronic star tracker has no moving parts so it is usually lighter, smaller in size, and has a longer life time

Table XII-5 Gimbaled Star Trackers

Identification	Manufacturer	Scanned FOV	Gimbaled FOV	Accuracy	Sensitivity Star Magnitude	Size in. 3	Weight, lb	Power, W
OAO Star Tracker	Kollsman	1x1	4°	10 arc-s	+2	11x17x15 + 11x16x4	23.6 + 18.5	1.75 + 10.0
K5-199 MOL	Kollsman	1x1	30°/axis	15 arc-s /axis	+1.8	9.5x15x9.5 + 7x8.6x8	30.0	23.0
OAO Backup Star Tracker	BC/ITT	1x1	60°	9 arc-s	+2.5	5 5/8x5 1/4 x5 1/4	6.0	4.5
Star Tracker	Honeywell	~	20°	27 arc-s	+1.0	185	5.5	7.0

Table XII-6 Electronic Star Tracker

Identification	Manufacturer	Scanned FOV	Accuracy	Sensitivity Star Magnitude	Size in. 3	Weight, lb	Power, W
Standard Star Tracker (SST)	BBRC	8x8°	10 arc-s	+6,+5,+4, +3	6.6x7.1x 12.2	17.0	18.0
Photon-Counting Star Tracker	Honeywell	2x2°	1.5 arc-s	+8			
PADS Star Tracker	TRW	1x1°	1.5 arc-s	+10			
Dual-Mode Star Tracker	ITT	8x8°	9-12 arc-s	+3	5x10½x5	8.5	5.0
Canopus Tracker	BEC/JPL	4x11°	0.1°	Canopus +2	4x5x11	5.0	1.5

reliability than the gimballed star tracker. In addition, it generally has a higher sensitivity, greater signal to noise ratio, and is relatively more rugged mechanically than the gimballed type tracker. However, it too has disadvantages as discussed below:

1. Subject to errors from stray electronics, magnetic field variation, and temperature variations,

2. Because of the finite acquisition time, a maximum attitude rate limit is imposed to ensure quality output data,

3. Narrow field of view might limit the mission applicability.

The more promising star trackers which have been flown and/or developed are summarized in Table VI. The three prime candidates, based on accuracy, applicability, and sensitivity, are the NASA standard SST, the Honeywell photon counting star tracker, and the TRW PADS tracker. Although both the Honeywell and TRW trackers have better accuracy, there are two major disadvantages of these systems which should be pointed out. First, the star tracker sensitivity for those two trackers are +8 magnitude stars and brighter. This creates numerous data storage and processing problems for autonomous systems because there are about 14,000 stars within this category. Secondly, the small field of view of these sensors impose a problem in the dynamic environment of the spacecraft, i.e., fewer good star acquisition signals will be obtained as the attitude rate of the vehicle is increased. These two points are not considered disadvantages for all missions, but they tend to limit the general application of the sensors.

<u>Star Mappers</u> - The star mapper generally has a slit type aperture (Figure XII-8) which utilizes the spacecraft rotation to provide a scanning motion for the sensor during steller acquisition. The FOV of the sensor is thus scanned over the celestial sphere.

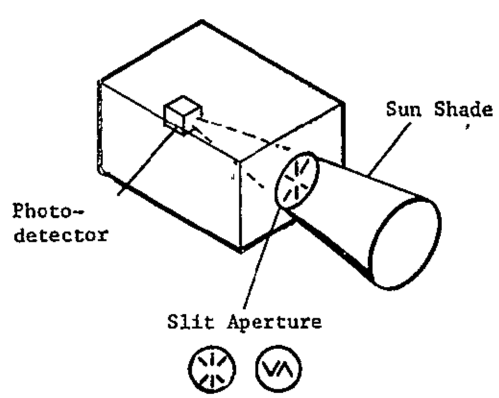


Figure XII-8 Star Mapper

As a star image on the focal plane passes a slit, the star is sensed by the detector. If the signal is above a set threshold, a pulse is generated by the electronics signifying the presence of a star. The slits in the focal plane are arranged in such a way that the crossing time of the star through the first slit and the elapsed time between this and the crossing of subsequent slit(s) together with a star catalogue yield attitude information as well as information about spin rate.

Like the electronic star tracker, the star mapper has no moving parts and is relatively simple in structure. The elimination of the image dissector is accompanied by a decrease in the complexity of the electronics and a decrease in the vulnerability to fluctuations in the local EM field. However, because the sensor relies on the motion of the spacecraft to provide scanning, the system applications are limited to missions where the spin rate is at least .07°/sec. If these mission requirements are met, the accuracy obtainable can be as good as 2 sec. There are also some features which make automation of the system difficult.

1. Interpretation of measurements is difficult if the spacecraft motion deviates from a uniform slew rate.

2. In-flight calibration is difficult because this normally requires sighting the same star on two successive passes.

With these considerations in mind, typical star mappers and their characteristics are summarized in Table XII-7.

<u>CCD Star Trackers</u> - The CCD Star Tracker uses a charged-coupled imaging array as a detector in place of an image dissector. The detector is a buried-channel, line-transfer, charge-coupled device (CCD), with vertical and horizontal picture elements. A typical detector contains 488 vertical by 380 horizontal picture elements within an active image area of 8.8 mm by 11.4 mm. The detector is cooled to an operating temperature below 0°C.

The detector array is read out with high speed microprogrammable logic. At those places in the field of view where star energy is detected, the operation is slowed to allow analog to digital conversion of the signal charge of each picture element, or "pixel" in the region. A micro-processor is employed to compute the location of the centroid of the star images to an accuracy of about 1/10 of the inter-pixel distance and to provide sequencing and control functions. The CCD unit possesses some distinct advantages over other types of star sensors. Those are: the ability to track multiple stars simultaneously, no sensitivity to magnetic fields, and improved accuracy. At the present

Table XII-7 Star Mappers

Identification	Manufacturer	FOV	Sensitivity Star Magnitude	Accuracy	Slit Configuration
Spars Star Sensor	Honeywell	8.8° Wide	+3.15	2 arc-s	Six - Slit
Block 5D/DMSP Star Scanner	Honeywell	10.0° Wide	+3.7	2 arc-s	Six - Slit
CS - 205	BBRC	10' Wide	+5	0.25 arc-s	V Configuration
Star Sensor	Bendix	6° Cone	+4	±5 arc-s	Six - Slit

time, TRW, BBRC, and Honeywell are evaluating the performance of CCDs in the laboratory using experimental breadboard models. The preliminary characteristics of both the BBRC and TRW CCD units are presented in Table XII-8.

Horizon Sensors - The combination of a two axis digital sun sensor and a horizon scanning sensor has often been applied to the problem of attitude determination. The two axis digital sun sensor will provide the two axis attitude information and, with the aid of a horizon sensor, can provide three-axis attitude information. It is generally a low cost, reliable sensor system with less software support required. However, because of the low resolution of the sensors and the lack of definition of the targets they sense, the sun/horizon sensor combinations are used only where relatively coarse attitude information is required.

Horizon sensor systems fall into several basic design classifications which are discussed below.

Conical Scan Sensors - A rotating optical component within the sensor head creates a conical scan pattern for a relatively narrow instantaneous FOV (Figure XII-9). For a two-axis performance, two sensor heads are generally mounted with the center of their scan pattern facing in opposite directions symmetric with the local vertical. A rectangular pulse is generated when the visible boundary of the earth is traversed by the optical scan of each sensor. A reference pulse is also generated at the time when the optical scan position is coincident with the vehicle pitch position. The square wave signals from each of the sensors are processed to yield roll information, and the phase of each sensor is compared to that of the reference signal to compute pitch. The accuracy obtainable by this type of sensor system can be as good as $.1^{\circ}$. Although this figure is nowhere near the accuracy obtainable by other sensors (star trackers, etc.), the system is operationally characterized by wide acquisition angles, fast response times, high reliability, and flexible adaptation to varying mission requirements making it a good candidate for some missions.

Radiation Balance Sensors - Radiation balance sensors are designed and implemented for applications where extremely high reliability, as in long orbital missions, is more important than high accuracy. This type of sensor contains no moving parts so it is generally small, lightweight, and consumes little power. Roll and pitch attitudes are obtained by measuring the imbalance in the radiant energy received by two sensors whose FOV are pointed in opposite directions towards the earth's surface. The errors associated with this type of sensors are quite large and uncertainties can be as high as 2°.

Table XII-8 Preliminary Characteristics of CCD Star Tracker

Characteristic	Units	TRW	BBRC
Accuracy (1 sigma)			
Vertical	arc-s	2.4	
Horizontal	arc-s	4.1	
Total	arc-s	4.75	5.0
Physical			
Weight	lb	7	7
Volume	in.	6 x 6 x 12	
Power	W	9.5 at 28 Vdc	26 at 28 Vdc
Development			
Status		Breadboard in Test	Breadboard in Test
Field of View			
Total	deg	6.0 x 8.53	7.1 x 9.2
Instantaneous	arc-min	0.81 x 1.35	
Optical System			
Focal Length	mm	76.0	70.0
f/No.		0.87	
Transmission		0.75	
Detector			
Type		Fairchild CCD	
Number of Elements		488 x 380	488 x 380
Image Area	mm	8.8 x 11.4	8.8 x 11.4
Configuration		Front Illuminated, Interline Transfer	Front Illuminated, Interline Transfer
Electronics			
Intergration Time (for +6 M Star)	s	0.100 Max	0.100 Max
Readout Rate (for +6 M Star)	s	0.100	0.100
Star Position Output			
Vertical	Digital	12-bit Serial	
Horizontal	Digital	12-bit Serial	
Star Magnitude	Digital	12-bit Serial	
Update Interval (for +6 M Star)	s	0.100 Max	0.100 Max

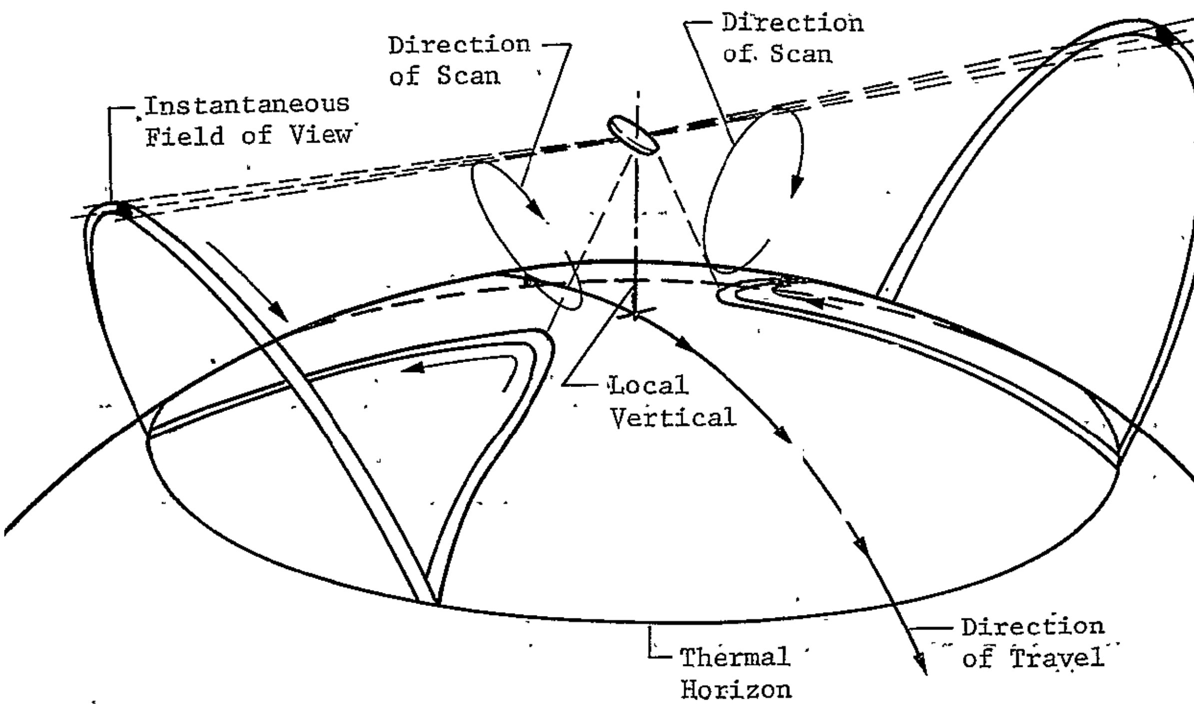


Figure XII-9 Conical Scan Geometry

Horizon Crossing Indicator - The horizon crossing indicator is a body mounted horizon sensor commonly used for spin stabilized spacecraft. The scanning motion is thus provided by the spacecraft rotational motion. The operation of the sensor is similar to the horizon scanner with the exception that the outputs include spin rate as well as attitude information. The accuracy of the sensor is effected by the rotational motion of the spacecraft and the non-uniform spin rate as well as the errors common to other horizon sensors. The accuracy obtainable with such a device can be .3°.

Sun Sensors - Sun sensors can be divided into analog and digital types. However, analog sensors have several disadvantages and will not be treated here. The major component of ditigal sun sensors consists of a mask which encodes sun angles as digital numbers (Figure XII-10). Light passing through a slit on the front surface of a fused-silica reticle forms an illuminated image of the slit on the binary-code pattern which is on the rear surface. The image's position is dependent upon the angle of incidence.

Behind each column of the code pattern is a silicon photodetector. If the light falls on a clear portion of the pattern in a particular, column, the photocell behind is illuminated producing an output "one"; if it falls on an opaque segment, the photocell is not illuminated and the output is "zero". The outputs of the cells are amplified, stored, and processed as required to furnish suitable output to telemetry or other data processors.

The gray code most commonly used for encoding quantitizes the field of view into 128 increments. Therefore, the accuracy obtainable is dependent on the front end optics and the width of the reticle. The accuracies which have been obtained are on the order of $.1^{\circ}$.

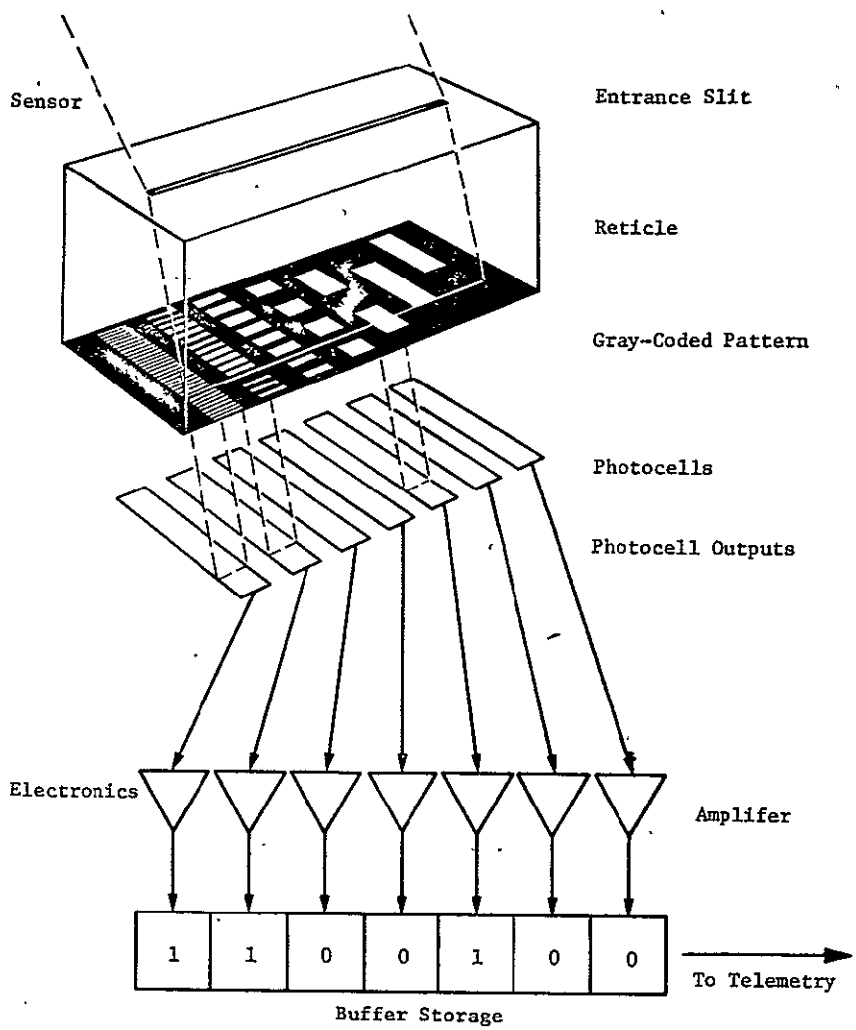


Figure XII-10. The Digital Sun Sensor

Space Sextant

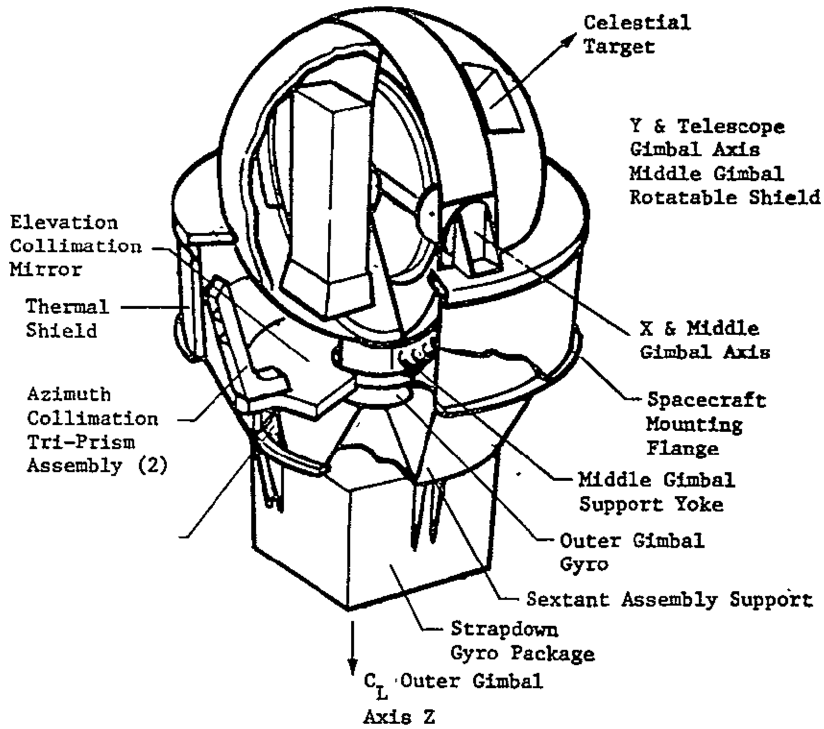
The space sextant approach to attitudes determination utilizes the angle between several stars and two reference points located on the base of the sextant (Dale Mikelson, 1977). The sextant consists of two Cassegrain telescopes, an angle measurement head, gimbals that provide three angular degrees of freedom for the telescopes, and a reference platform consisting of a planar mirror, porro prism assembly, and a gyro package (Figure XII-11). The space sextant was primarily designed for autonomous navigation, but the addition of the reference package allows attitude determination as well.

Attitude is determined by first making included angle measurements between two or more stars and a reference mirror fixed at the base of the sextant. A light source within the telescopes allows it to autocollimate off the mirror, thus providing a reference for one of the telescopes (Figure XII-11). The included angle relative to the reference mirror fixes the attitude in one direction with respect to inertial space. The other attitude directions are fixed by making included angle measurements between one or more stars and a porro prism assembly mounted on the reference mirror in such a way that the prism is elevated above the plane of the mirror (Figure XII-11). The two measurements are then processed to yield precise attitude information.

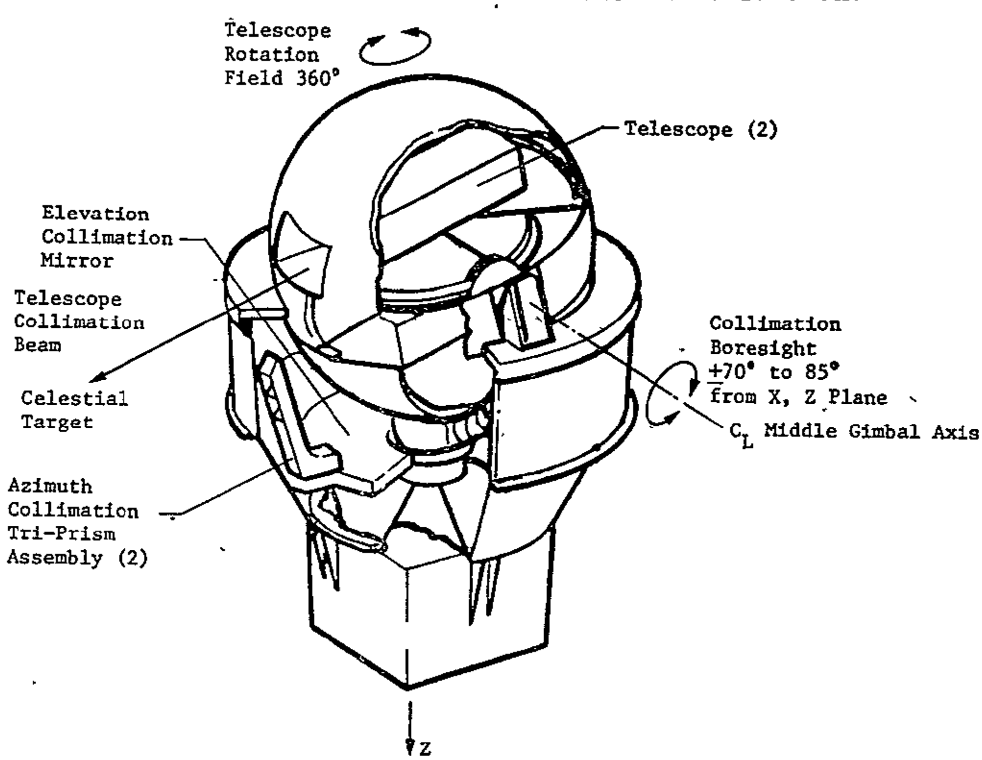
This attitude information is then used by an on-board filter to update the gyro uncertainty. Using this configuration, the three axis attitude uncertainty can be kept below 1/2 arc sec. The system characteristics are shown in Table XII-9.

Table XII-9 Space Sextant Characteristics

Instantaneous FOV	Accuracy 3-Axis	Size, in.	Weight, lb	Power, W	Life, yr
6 arc-min	0.5 arc-s	21.3x20x21.3	60	30	5



Sextant Collimation Off Mirror - Elevation Measurement



Sextant Collimation Off Porro Prisms - Azimuth Measurement

Figure XII-11 Attitude Determination with Space Sextant

Landmark Trackers

Landmark trackers utilize sightings of known earth features to yield attitude information (Gilbert, 1977; Sugihara, 1971; and Kau, 1975). There are many types of earth features ranging from radar emitters to natural features such as lakes, and the methods used to detect these features differ as greatly. However, all the concepts rely on obtaining the Line of Sight (LOS) angles to some point on the earth whose position is accurately known and stored onboard.

The accuracy which is theoretically obtainable can be as good as 1 arc sec. for some types of Landmark trackers. However, Landmark trackers are not as well suited for attitude determination as other systems relying on sightings of celestial objects. The qualities which led to this conclusion are listed below.

Crosscoupling Between Attitude and Position - There is a severe crosscoupling between the position of the satellite and its attitude when LOS measurements are being made (Figure XII-12) A downtrack error can easily be mistaken as an error in pitch. There are proposed methods of obtaining several sightings within the field of view and using this information to derive both position and attitude. However, such a system would be nowhere near real time, and the onboard memory requirements would be tremendous.

Algorithms Too Involved - The algorithms to derive attitude from a Landmark sighting, assuming that position is known, are much more involved than those required by a star tracker or horizon/sun sensor combination. Because of the virtually infinite distance of stars, their coordinates can be stored in an inertial reference frame, and attitude can be derived directly from several sightings. Likewise, horizon/sun sensor systems directly derive the local vertical whereas landmark trackers do not.

Development Required - The technology required to implement a Landmark tracker attitude reference system has not evolved to the degree that star trackers has. Most of the development in these trackers has been directed at solving the autonomous navigation problem, not attitude determination.

Landmark trackers are much better suited for navigation updates than for attitude updates, and so they should be dismissed from this arena.

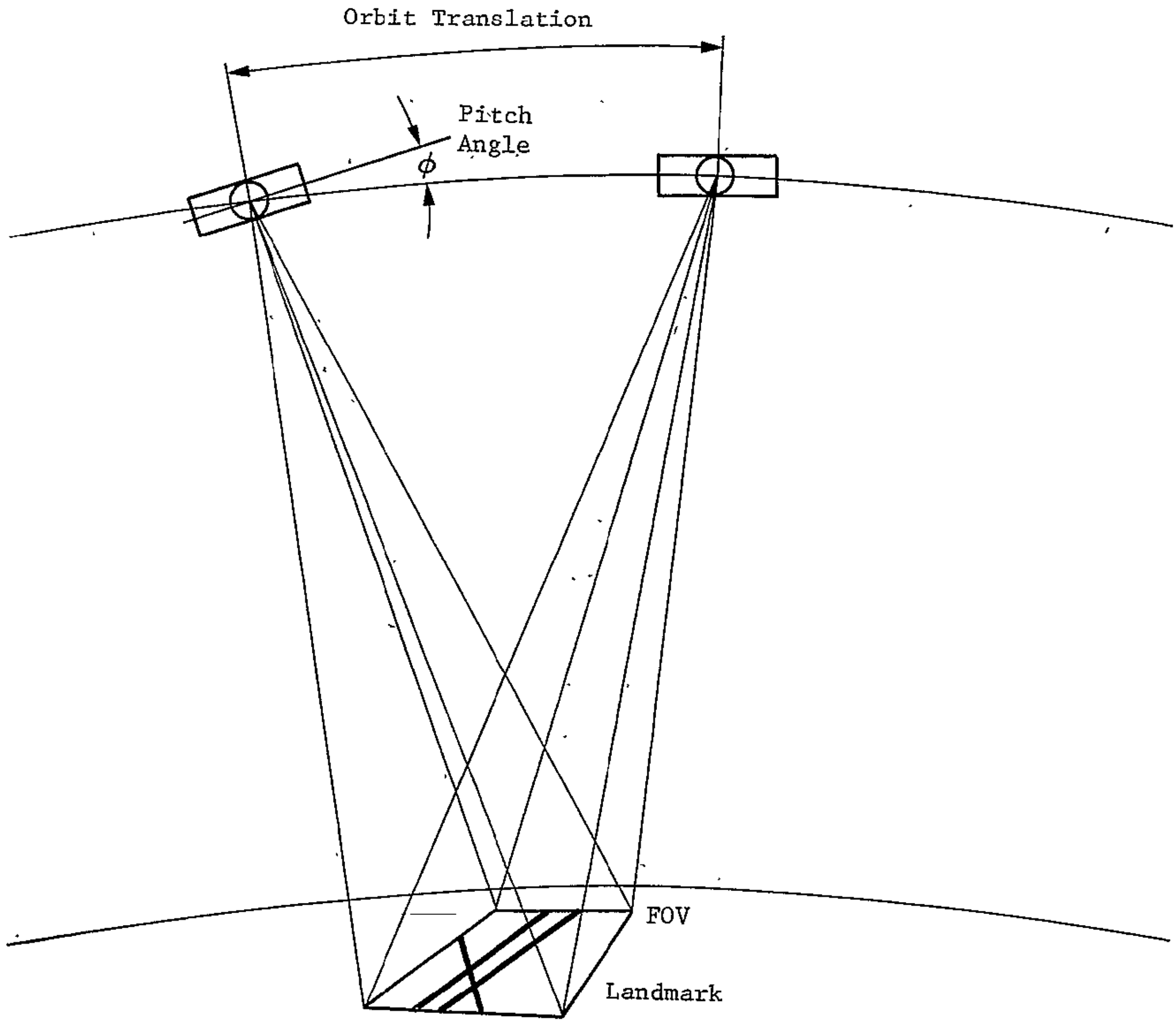


Figure XII-12 Position/Attitude Crosscoupling of the Landmark Tracker

Interferometers

Interferometers can be used to derive attitude information by measuring a line of sight angle between the spacecraft axis, defined by an antenna array, and an RF emitter whose position is accurately known. The angle is measured by detecting the phase difference of an RF signal arriving at two pairs of receiving antennae (Figure XII-13).

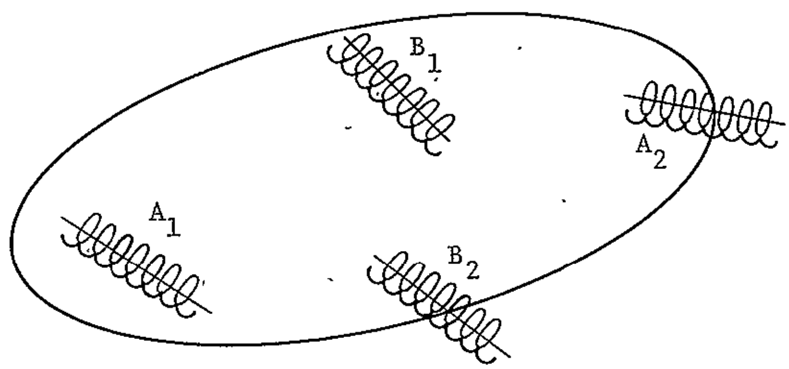


Figure XII-13 The Interferometer Setup

Angular information can be derived from knowledge of the phase difference through the well known equation:

$\Phi = \frac{2\pi D}{\lambda} \cos \Phi$ (Figure XII-14) Φ = phase difference

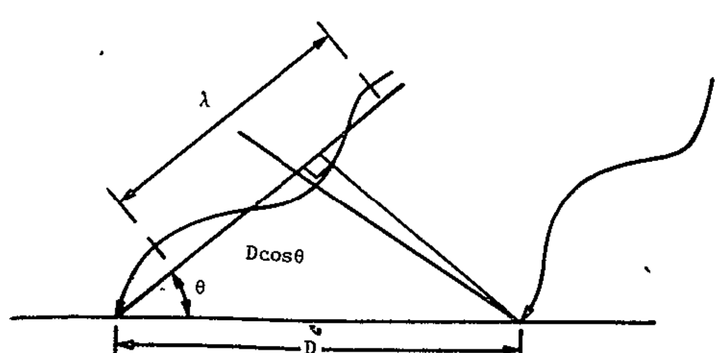


Figure XII-14 Interferometer Setup

Complete attitude information can be obtained by knowing the position of at least two transmitters, and the angle of arrival of their signals. There have been two investigations for using an interferometer as an attitude determination sensor. The first, designated as the Interferometric Landmark Tracker (ILT), utilizes ground based radar emitters whose positions can be accurately stored onboard (Aldrich, 1974). These emitters include airport radar systems and perhaps tracking stations. Although the ILT differs from optical Landmark trackers in several ways such as the speed involved in measuring the LOS angles, and the memory required to store each landmark, the system is still limited by the crosscoupling errors.

The second approach to using an interferometer is in the conceptual stage only, but the method involved is promising (Ellis and Creswell, 1978). The Global Positioning System (GPS) is designed to be a navigation aid. However, through the use of an interferometer placed onboard the user spacecraft, attitude information can be derived as well.

Unique determination of three axis attitude information involves measuring relative to the spacecraft axis, two linearly independent vectors to GPS satellites using two independent interferometers. The accuracy obtainable from such a system is between $.02^{\circ}$ to $.6^{\circ}$.

Processor Requirements

In addition to the attitude determination sensors, onboard processors are one of the most important prerequisites to an autonomous system. As the capability of the sensors and the scope of the mission increase, the burden placed on these processors will rapidly increase. A point will soon be reached whereby it will not be economically feasible to develop a new computer for each mission as is quite often done presently. It is necessary to design a processor which can be reconfigured for different mission requirements.

Two in-depth studies were recently conducted to analyze the computational requirements of an onboard autonomous attitude determination system (Carney, et al., 1978 and Mikelson, 1977). The mission configurations studies were the OADS concept and the Space Sextant. This section will briefly summarize those studies in order to portray the general requirements of an autonomous system.

Onboard Attitude Determination System (OADS)

The OADS system proposed consists of one NASA standard IRU (DRIRU-II) as the primary attitude determination sensor, two improved NASA standard star trackers (SST) for periodic update of attitude information, a GPS receiver to provide onboard space vehicle position and velocity vector information, and a multiple microcomputer system for data processing and attitude determination functions. The processing requirements for the automated system were broken down into two categories: IRU processing and star tracker processing.

IRU Processing - Analysis of the OADS mission requirements showed that the gyro inputs required sampling every 50 milliseconds. This figure set a maximum limit on the cycle time for IRU processing; however, to minimize the impact on other processing requirements, this figure has to be as small as possible. IRU processing is typically a computational oriented problem which requires solving a second order Runga-Kutta integration. Microprocessors have, for some time, been circuit simplification devices as well as small application controllers. Only recently have the computational capabilities of these devices been examined for processing oriented applications. Two processors which were evaluated were the Intel 8080 and the Am 9511. The 8080 microprocessor is an 8-bit general-purpose processing unit. Its relatively primitive instruction set (as compared to minicomputers) makes the 8080 undesirable for performing the arithmetic computational requirements needed for onboard attitude determination. The Am 9511 on the other hand, is tailored to perform arithmetic computations but its data management capabilities are extremely limited. The OADS study, therefore, initially analyzed the IRU processing using a system containing both an 8080 and an Am 9511.

In the analysis, the worst case timing requirements for each of the algorithms was performed and the results are shown in Table X. It was determined that all nominal IRU processing could be accomplished, using this configuration, in 49.988 ms. Although this figure meets the requirement for IRU processing time, it should be noted that there would be no room for software expansion and there are severe effects on other priority processing requirements. An analysis of the timeline characteristics suggested that the throughput could be increased significantly by the addition of a second APU. Such a multiple APU configuration would take advantage of parallelisms in the IRU algorithms and would be less complex, in hardware and software terms, than a dual microcomputer configuration. The timing analysis of the dual APU configuration is shown in Table XII-10.

Table XII-10 OADS IRU Throughput

	ESTIMATED IRU THROUGHPUT USING SINGLE APU		ESTIMATED IRU THROUGHPUT IN DUAL APU SYSTEM	
ALGORITHM	ALGORITHM PROCESSING TIME (ms)	ACCUMULATIVE PROCES'ING TIME (ms)	ALGORITHM PROCESSING TIME (ms)	ACCUMULATIVE PROCESSING TIME (ms)
Data Edit	.132	.132	.132	.132
Rate Computation	2.157	2.289	1.245	1.377
Compensation	13.242	15.531	7.198	8.575
Data Reduction	21.534	37.065	13.521	22.096
Integration	12.924	49.989	7.324	29.42

Star Tracker Processing - The primary difference between IRU processing and star tracker processing is that star tracker processing is a two-phase problem. The first phase involves determining what star is in the tracker's field of view. The OADS analysis showed that it requires over 1100 milliseconds to accomplish a linear search for a star catalog containing 1500 stars (using a dual APU configuration). This time might be reduced by using a binary search, however, the 8080 microprocessor instruction set does not lend itself to this type of algorithm for large tables. Another approach considered was to use an indirect indexing table. If the star catalog is sorted by right ascension, an indirect indexing table containing 360 entries is constructed. Each entry in the index table corresponds to one degree of right ascension and points to the star catalog where stars of corresponding right ascension are stored. For example, assume that a given V and H reading and a quaternion produce a predicted star right ascension of 263.875^{o}. This right ascension value is truncated and used as an index table position. The 263 entry in the index table contains an address in the star catalog where stars whose right ascension is 263.xxx degrees are stored. The star catalog entries around this point are then searched (using the star identification algorithm previously described) to determine which star best fits the predicted star right ascension and declination. It is estimated that this may require searching 100 star catalog entries and use 114 milliseconds (in the dual APU configuration).

Once a star in the tracker's field of view has been identified, searching the star catalog is not required again until the star leaves the field of view. Phase two of star tracker processing involves using V, H and IRU quaternion, and star catalog data to produce an updated quaternion. This updated quaternion is then integrated forward in time and used in succeeding IRU processing. Our analysis shows that for a single star tracker, generation of a corrected quaternion will require 152 milliseconds in the dual APU configuration. If two trackers are used, each having their own microcomputer system, star identification processing can be overlapped but most of phase two processing must be executed sequentially. This results in 236 millisecond processing time when two star trackers are active. The following table summarizes this information.

Table XII-11 Multiple Phase Star Tracker Throughput

	PROCESSING TIME WITH SINGLE APU (ms)	PROCESSING TIME WITH DUAL APU (ms)
Phase 1 - Star Identification	265.6	134.3
Phase 2 - Update Quaternion Single Star Tracker	231.6	152.2
Dual Star Tracker	366.0	235.8

IRU and Star Tracker Integration - From the beginning of the OADS study, it was felt that the combined IRU Star Tracker processing would place the greatest demands upon an onboard microcomputer system. IRU processing time is constrained by the fact that gyro readings are to be made every 50 milliseconds. Star Tracker processing must be synchronized to IRU processing and must be performed in a sufficiently short time to correct the quaternion to the required accuracy. Because of the high demands on the IRU and star tracker microcomputers, it seemed appropriate to integrate these subsystems before investigating the remaining elements of the OADS system.

Since IRU data is required for star tracker processing and vice versa, a shared memory unit is anticipated for the microcumputer systems. It would have been possible to directly connect the microcomputers using input and output ports; however, this technique would require higher software overhead. Because of the independent processing of the IRU and star tracker microcomputer systems, a synchronization mechanism is required to insure reliable results. Two levels of synchronization are anticipated. First, a hardware semaphore is needed to prevent inconsistencies in shared data memory. For example, the IRU microcomputer must not be allowed to modify rate and quaternion data the star tracker microcomputer is reading. The hardware semaphore would prevent this by permitting only one microcomputer system to access shared memory at a time. To avoid long access delays, individual microcomputers could move shared data into local memory and then operate upon that data while it is in local memory. The hardware semaphore need not be complex circuitry; in fact, it need only emulate a slow input/output port. The techniques for implementing such microcomputer logic are well known.

The second level of synchronization is required to logically associate IRU and star tracker data. For example, the time between gyro readouts and star tracker readouts must be known to associate V & H readouts with IRU activity. Obviously, a common clock and time tagging hardware is an essential element in this synchronization. It will also be necessary in software to carry time tags (in the form of counter values) along with rate value, V & H values, quaternion values, etc. This type of software logic is common in most process control applications.

It was mentioned earlier that if two star trackers are operating simultaneously, star identification may be performed in parallel but that correction of the quaternion (using two stars) is basically a sequential process. For this reason, it is desirable to put the two star tracker microcomputers in a master-slave relationship. This can be achieved by means of a "smart" switch. The function of the smart switch is to direct the data generated by the first tracker to lock on to a star to the master microcomputer system. The master microcomputer system may then proceed to identify the star and correct the quaternion. Should the second tracker acquire a star during this time, its data would be directed by the smart switch to the slave microcomputer system which would then proceed to identify the star. When the master microcomputer has finished correcting the quaternion based on data from the first star tracker, it would check with the slave microcomputer to determine if a second quaternion correction can be performed. If the Kalman filter can be run again, it is done at this time by the master microcomputer using data supplied by the slave microcomputer.

The master-slave relationship between star tracker microcomputers was suggested because it reduces the complexity of software needed for star tracker processing and because it minimized the interfaces between the IRU subsystem and the star tracker subsystem. Implementation of the smart switch is not envisioned to be a difficult problem. Even if star tracker electronics cannot be extended to make a smart switch, it is possible to use normal switching logic driven by the master microcomputer system.

The processor configuration of the OADS system is shown in Figure XII-15. The basic building blocks consist of the IRU processor, the master/slave star tracker processor, and the orbit generator/resolver processor. Each processor block consists of an 8080 MPU and two Am 9511 APUs. This configuration, which is one of the latest onboard processor designs, emphasizes the use of distributive processing. A comparison of the NASA Standard processor, NSSC I, and the multi-microprocessor configuration was based on performance. The results are summarized in Table XII-12.

Table XII-12 Summary of Estimated OADS Processing Time for Baseline Microcomputer System

	Proposed System Processing Time (ms)	NSSC I Processing Time (ms)
IRU Processing	29.42	44.575
Star Tracker Processing		
Phase I (star identification)	134.3	183.98
Phase II (1 tracker-quaternion correction)	152.2	314.983
Phase II (2 trackers - quaternion correction)	235.8	609.468
Orbit Generator Processing	3.421	187.527
Resolver Processing	38.518	

It was concluded that the NSSC I is incapable of supporting the operations required for the OADS approach to autonomous attitude determination. This results from the fact that the OADS

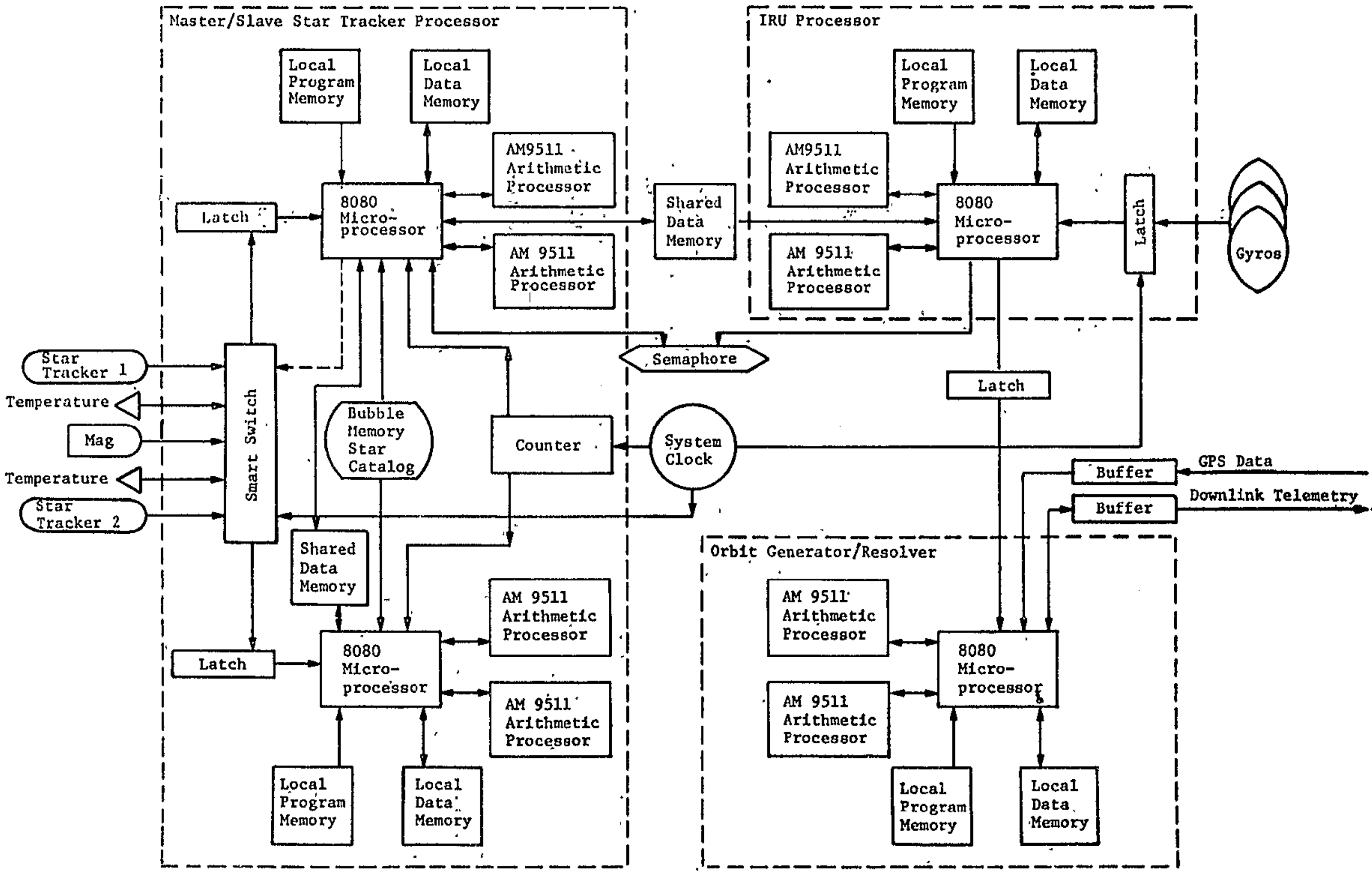


FIGURE XII-15 OADS PROCESSOR CONCEPT

algorithms are computationally oriented whereas the NSSC I is tailored to perform data management tasks.

Space Sextant Processor Requirements

The SS-ANARS flight test minicomputer will perform all computations necessary for the operation of the space sextant. The mini-computer must perform five major functions:

1. Control of telescope positioning servos;
2. Integrate the measured angular rates from the gyros;
3. Integrate equations of motion for position and velocity;
4. Store command information and earth/moon ephermeris, and
5. Perform attitude and position estimation by processing measurements taken by the sextant.

Servosystem Software - The four sextant gimbals will be controlled by a four degree-of-freedom sampled data servosystem. The servosystem software will process the gimbal data to derive the logical processes which control the sextant. This software is also responsible for processing the included angle measurement data. Execution of the software will provide functional control of the sextant through an interface with a special purpose electronics package. Approximately 6K words of 16-bit memory will be required for the construction of the servosystem software. The prime executable routines are:

- Shuttle and autocollimate light control - Telescope servo-lock to the reference platform annular mirror or porro prism assembly will be preceded by activation of that telescope's autocollimate light. Reflection of the autocollimate light is used as a target which insures the telescope will be perpendicular to the reference planes defined by the mirror and prism.

- Bias Set - Prior to each included angle measurement, sensor bias levels will be determined and compensated for within the hardware and software to insure amplifier nonsaturation and minimal measurement offsets.

- Gain Set - Sensor amplifier and servoloop gain levels will be updated for each measurement so that variations in star visual magnitudes and limb illumination have a minimum effect.

- Sensor interrogation - Each sensor element output will be sampled at a 5-ms interrogation rate and the resulting data (12-bit resolution) formatted and stored in memory. Sensor element output levels control the logical processes for acquisition and compose the servo feedback channel for target lock.

- Gimbal position - Telescope gimbal position information (24-bit resolution) will be provided at a 9-Hz sample rate (wheel speed) while the yaw and roll support gimbals (18-bit resolution) are updated at a 100 Hz rate. Gimbal positions will be utilized for slew servoclosure, coordinate transformations, motor commutation, and included angle determination.

- Servoclosure - Following the formation of each loop error signal, gimbal torque commands will be derived via signal shaping using the digital filters and in-line trapezoidal integrators.

- Slew control for target acquisition - Following the receipt of initial gimbal angles, the four gimbals will be simultaneously activated, slewing 5 to 10 deg/s toward the commanded position.

- Scan control - Due to the small field of view (6 arc min) of the telescopes, the raster scan will be performed during initial acquisition. Target position determination in sensor coordinates and transformations to gimbal commands will provide the servo inputs for scan execution.

- Acquisition logic - Following initial target detection, precise gimbal rate control will be effected to move the telescope in order to center the target within the field of view.

- Target lock transformation logic - Once the target acquisition has been accomplished, the sensor data will be processed in order to insure that target lock is maintained.

- Motor commutation - Gimbal torque will be effected via two-phase brushless dc torquers. These torquers are software commutated, thereby removing any requirements for such hardware devices.

- Trigonomeric generation - Sine and cosine trigonomeric functions required for the Euler transformations and motor commulation computations will be generated by a 9-bit lock-up table.

- Status control - During the slew, scan acquisition, and lock-on phases of servo activity, the monitor will be provided with reasonability checks of sensor outputs, gimbal angles, and torque commands for failure identification.
- Data processing of included angle measurement - The included angle measurement will be derived by differencing telescope 1 and 2 gimbal position (24-bit resolution). The effects of servo activity on the measurement value will be reduced by averaging sine angle measurements.

Attitude Transformation - The minicomputer must integrate the measured body angular rates from the strapdown gyro package to obtain attitude information. The spacecraft attitude information is necessary to decide whether the required bodies (moon, earth, stars) are in view of the sextant and, if so, this information is necessary to correctly point the telescope(s). Also, the data obtained by integrating body angular rates will allow the spacecraft attitude to be estimated, through filtering, at times when an attitude update is not being measured.

The measured angular rates will not be integrated directly. The measured quantities will first be corrected for compensatable errors and then transformed to the principal axes of the sextant. The transformed and compensated quantities will then be used to calculate the change of four components of a quaternion relating the body axis system to the reference coordinate system.

The stellar observations serve to orient the sensor's fixed axis in inertial space. Because the fixed mounting of the space sextant containing its principal axes and the gyro unit are statically connected, there is an invariable relative orientation between the respective coordinate systems. Attitude measurements made with the space sextant can be related to the gyro axes and may, therefore, be used to update that system's inertial orientation.

State Estimation and Predication - The minicomputer must process measurements from the space sextant to estimate the position and attitude state vectors of the spacecraft and then integrate differential equations to predict the state of the spacecraft at times other than actual measurement times. Between measurements, the attitude of the spacecraft will be estimated by integrating the measured body angular rates measured by the gyros. The position and velocity of the spacecraft will be estimated by integrating the differential equations of motion defining the spacecraft state in rectangular inertial coordinates.

Ephemerides and Star Tables - The minicomputer must contain a table specifying the locations of the stars the space sextant will observe. This table will consist of the three components of the unit vector pointing in the direction of the star. The coordinate system used to locate the stars is defined by the mean standard equinox at epoch 1950. The star table must also contain data describing the electrical current produced by the star's light in the silicon photodetector so gain parameters can be set for the telescope electronics for the acquisition and tracking logic. The data in the star tables will first be used to direct the space sextant to search a particular area of the sky to locate a particular star. After the star is located and the included angle between the telescopes is measured, the position of the star must be accurately known so the minicomputer (or FTSC) filter can update the position or attitude estimate for the spacecraft using this measurement.

Ephemerides for the moon, sun, and earth must also be calculated by the minicomputer. The position of these three bodies is important because they may obsecure a star from the sextant's view. The minicomputer must determine if a star is visible before directing a telescope to search for it. Each of the three coordinates of the moon's position will be approximated with a Chebychev polynomial. Experience has shown that a Chebychev polynomial of degree 20 for each of the three coordinates is adequate to calculate the lunar position for a period of .25 days.

Sextant Measurement Calculations - The space sextant measurements will be used for both spacecraft position estimation (navigation) and spacecraft attitude estimation. The navigation measurements are the measurements of the included angle between a star and either the bright limb of the moon or earth. The attitude estimation measurements are the included angle between a star and a light beam reflected from either the base mirror or the sextant porro prism.

The predicted value for each measurement must be calculated for comparison with actual measurements made by the sextnat. The difference between the actual and predicted measurement is used by the Kalman filter to update the estimate of either position or attitude. The nominal value for each measurement is determined by the geometry between the star, spacecraft, and moon or earth, or by the attitude of the spacecraft. However, the accuracy of the measurements made by the sextant requires that these measurements be precisely corrected.

Using these functions as a design guideline, the basic requirements of the SS-ANARS minicomputer were established. It was determined that the CPU must be able to perform the following mix of instructions within 8.5 milliseconds.

Instruction Type	No. of Instructions
Floating Point Multiply	95
Other Floating Point (1 Divide)	396
Load, Store, Add, Sub, I/O . . .	921
Shifts (Assume Average of 12 Bits)	37
Jumps	449
Trig Table Look-Up (80 μsec each)	6
Double Precision	26

Comparison of Processing Systems

It is interesting to compare the two proposed computer systems with the new generation NASA standard processor NSSC-II. Table XII-13 lists the time required for various operations in each of these systems. While the NSSC-II is a great improvement over the NSSC-I processor, its capabilities can be greatly increased by utilizing the dual APU approach as was done for OADS. This configuration would also allow the processor to be custom tailored to mission requirements rather than making extensive software redesign from mission to mission.

Table XII-13 Processing Times for Various Systems

	Fixed Point (32 Bit)				Floating Point (32 Bit)			
	Add	Subtract	Multiply	Divide	Add	Subtract	Multiply	Divide
OADS, Single APU AM9511	10.5 μs	19.0 μs	104.0 μs	104.0 μs	37.5 μs	37.5 μs	84.0 μs	85.0 μs
SS-ANARS, Single CPU Minicomputer	0.75 μs	0.75 μs	7.5 μs	10.25 μs	3.5 μs	3.5 μs	10.5 μs	10.75 μs
NSSC-II*	2.8 μs + N	2.8 μs + N	30.4 μs + N	51.8 μs + N	21.2 μs + N	21.8 μs + N	33.7 μs + N	48.6 μs + N

*For the floating point values, 1.4 μs must be added for each digit in the calculation.

CONCLUSIONS/RECOMMENDATIONS

The possible cost benefits obtainable by automating the attitude determination system onboard satellites fall into two categories: mission support costs and data handling. Many missions presently require a group of people to work three shifts seven days a week to perform these tasks. Through automation, this figure could possibly be reduced to one shift seven days a week (or less), thus bringing a significant cost savings to the overall mission.

These reductions in mission support costs are essential in making many future NASA missions feasible. Without such a reduction, the number of new start missions will continue to decrease.

The increased scope of future NASA missions has led to development of high data rate sensors. Data from these sensors will be of little added value unless new data handling technques are developed by automating several operations. These functions include:

- onboard image distortion corrections
- selective data acquisition
- precise registration of sensor data
- precise sensor pointing on command

Each of these functions implies a high-precision attitude determination system which must be implemented autonomously onboard. Without automation, the real-time decision process required for each of these operations will not be possible.

In addition to realizing goals of future NASA missions, automation of the attitude determination system will bring a cost savings to the mission by reducing the amount of ground processing required for the analyses of data. Presently all image data is transmitted to ground stations for geometric distortion corrections, registration and navigation information extraction. This process results in reams of unusable data due to either cloud coverage or undesirable features (ocean, etc.). Excess data implies an excess of processing, and indeed somewhere in the vicinity of 50% of LANDSAT data is unusable due to these effects. Automation of the navigation/attitude system would thus allow the development of an autonomous pointing and data acquisition system and cut this portion of ground support by 50%.

In addition to the reduction of image processing, autonomous attitude determination would cut the amount of telemetry and its processing proportional to the amount of attitude information contained therein. A reduction in the telemetry also leads to a reduction in the number of people required for mission support and therefore leads to a cost savings.

In order for autonomous attitude determination to be implemented on a large number of missions, several key technologies must be developed or propagated. First, radiation hardening of large scale integrated hardware is essential. Many LSI circuits have been specially constructed and space hardened for specific missions, but the vast number of components which were developed for use in ground systems are generally not qualified for space operations. By investing in the hardening of commercially available components, NASA can take full advantage of the rapid advancement of the LSI technology and reduce the development costs of some of the more advanced automated systems.

Distributive operations and processing must be emphasized to allow the development of automated systems. Most of the missions which contained onboard processors for automation have relied on a central processor. As the requirements of the automated systems increase, this type of architecture will become less effective for various reasons.

1. Check and validation of such machines becomes increasingly difficult as the size and processing capabilities increase.

2. The data bus between peripheral sensors and the processor will become saturated unless extremely high speed busses are implemented, thus leading to a higher cost factor. In addition, peripheral sensors would have to be time shared under this system, which is not always desirable.

Distributive architecture solves these problems in the following manner:

1. Fault detection can be isolated in either a single peripheral or the central processing unit, which makes debugging an easier process.

2. Since raw data is processed by the peripheral units, the data bus and the central processor are reserved for higher level operations.

3. Redundancy is more easily implemented in this configuration.

Although the current trend is leaning towards distributive processing, it is important to maintain and develop this course.

The cost of software in the development of autonomous systems will be a major limitation unless several software tools are developed. These tools include higher order languages with efficient translators into machine language (efficient in the number of machine cycles), more advanced debugging techniques whereby the software development computer is actually responsible for some of the debugging tasks; fast emulation of the onboard processor to aid in the critical areas of software development.

With the increasing use of shuttle and its refueling capabilities, the number of three axis stabilized spacecraft will begin to increase. This development will be due to several factors.

1. The fuel economy of the spacecraft will not be as important a factor as it presently is. This will be due to the refueling capabilities of shuttle.

2. The development of high data-rate sensors leads to the requirement of accurate sensor pointing and platform stabilization.

3. Many missions can be implemented using a single spacecraft frame otherwise known as the multi-mission spacecraft.

For systems such as this, the attitude determination system will probably assume the architecture shown in Figure XII-16. The system shown consists of an IRU which is used to read attitude directly and a star tracker to update the system as uncertainties in the gyro drift increase.

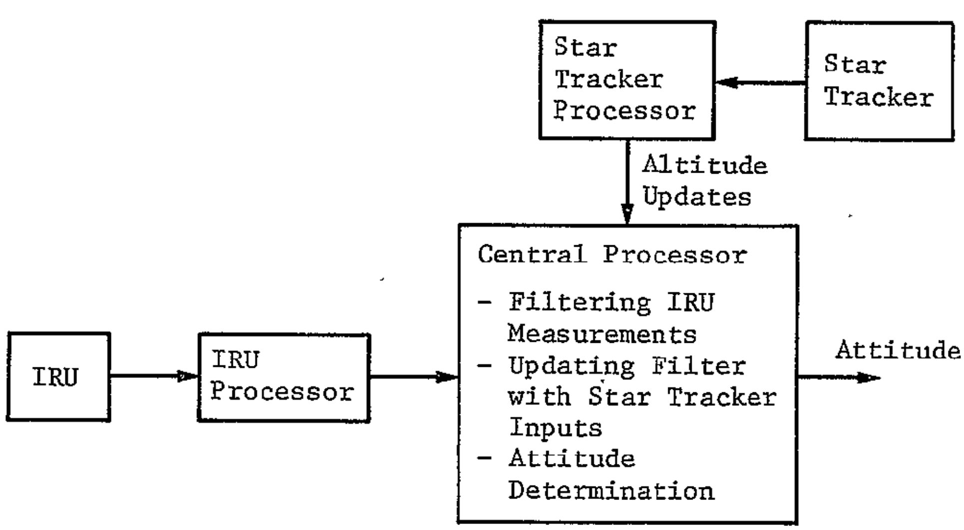


Figure XII-16 Architecture of an Autonomous Attitude Determination System

Although 3-axis stabilized spacecraft will see an increasing use as automation technology is implemented; the other types of missions will not be discarded. The system which requires the most attention, other than the 3-axis type, is the dual spin spacecraft. This type of vehicle maintains the benefits of a stabilized platform while at the same time exhibiting the qualities of a spin stabilized vehicle. However, with the increasing data rates required between the sensor and the processor, the present method of using slip rings for data transmission across the spin-non-spin boundary will become a severe limitation to these vehicles. Other transmission techniques such as an optical or microwave link must be explored and developed.

CCD or other solid state star tracking devi ces should be developed. These sensors have the following advantages over the conventional ID tubes used:

1. Ability to track several stars simultaneously
2. No sensitivity to magnetic fields
3. Possible improved accuracy
4. Smaller, lighter, and consume less power

Modeling of IRU errors presently require extensive processing which will tend to saturate processors onboard an automated spacecraft. Therefore, in addition to emphasizing distributive processing, it will be advantageous to develop algorithms which reduce the processing required. This will become especially important in future systems where smaller and smaller errors will be modeled.

In addition to technological advances which should be propagated, there are systems being developed which deserve special attention. The first of these is the system which combines an interferometer with the GPS receiver in order to determine attitude. This is an interesting concept because:

1. It can be incorporated into a completely autonomous navigation/attitude determination system.

2. Not effected by magnetic fields as ID star trackers are.

3. Onboard catalogues are not required as in star trackers.

Much more investigation is required as to the feasibility of using the system onboard a satellite, but its potential benefits warrant the expenditure for feasibility studies.

The second of these systems is Space Sextant. As a pure attitude determination system, the space sextant is awkward relative to other sensors; it is large, heavy, and consumes more power. However, the fact that SS-ANARS also performs autonomous navigation tends to make the system more promising. In addition, the development expenditures have already been provided by DOD so the initial cost can be significantly reduced. It would be beneficial to perform a trade study to see if the reduced development costs and the available accuracy would offset the increased cost per unit and the size, weight, and power.

In addition, studies are currently underway looking into methods which would minimize the size, weight, and power of the Space Sextant. Improvement would be made through the introduction of LSI circuitry, new lightweight materials, and perhaps through the reduction in telescope size.

REFERENCES

Aldrich, D. H., "Interferometer Landmark Tracker Navigation System," 1974, 592-NAECON.

Carney, P., Millilo, M., Tate, V., Wilson, J., Yong, K., "On-Board Attitude Determination System Final Report," Martin Marietta Aerospace, 1978, NAS4-23428, Mod. 27.

Creswell, G. A., Ellis, J. F., "Interferometric Attitude Determination with the Global Positioning System," AIAA Paper No. 78-1250.

Gilbert, L., Mahajan, D., "On-Board Landmark Navigation and Attitude Reference Parallel Processor System," presented at the Goddard Symposium on Flight Mechanics/Estimation Theory, October, 1977.

Kau, S. P., "Autonomous Satellite Orbital Navigation Using Known and Unknown Landmarks," presented at the AIAA Guidance and Control Conference, 1975, Paper No. 75-1096.

Mikelson, D., "Design and Laboratory Testing of Self Contained High Altitude Navigation System, Phase 1 - The Space Sextant Autonomous Navigation Attitude Reference System," Martin Marietta Aerospace, 1977, SAMSO TR-77-123.

Sugihara, S., "Attitude Determination Using Known Landmarks," 1971, Aerospace Corporation, Report No. TOR-0172(2129-01)-50.