NASA Conference Publication 3355

Sixth Alumni Conference of the International Space University

10th Anniversary Summer Session Program 1997

Herring Hall, Rice University

Houston, Texas, USA

July 11, 1997

INTERNATIONAL® SPACE UNIVERSITY

Sponsored by
ISU U.S. Alumni Organization

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National Aeronautics and Space Administration

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The 1997 ISU alumni conference would not have been possible without the work and talents of many people. The co-chairs of the conference wish to thank all who participated in the organization and accomplishment of the many 1997 ISU alumni conference and weekend events.

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Preface

The sixth alumni conference of the International Space University, coordinated by the ISU U.S. Alumni Organization (IUSAO), was held at Rice University in Houston, Texas, on Friday July 11, 1997.

The alumni conference gives graduates of the International Space University's interdisciplinary, international, and intercultural program a forum in which they may present and exchange technical ideas, and keep abreast of the wide variety of work in which the ever-growing body of alumni is engaged. The diversity that is characteristic of ISU is reflected in the subject matter of the papers published in this proceedings.

This proceedings preserves the order of the alumni presentations given at the 1997 ISU Alumni Conference. As in previous years, a special effort was made to solicit papers with a strong connection to the two ISU 197 Summer Session Program design projects: (1) Transfer of Technology, Spin-Offs, Spin-Ins; and (2) Strategies for the Exploration of Mars. Papers in the remaining ten sessions cover the departmental areas traditional to the ISU summer session program.

Of special note in 1997 is that it is the first year of the Design Project Roundtables, working discussions between alumni and students, which are intended to strengthen the alumni involvement in the Summer Session projects, and to introduce 1997 students into the distinguished ISU alumni family.

The IUSAO wishes to thank the NASA Johnson Space Center for sponsoring the 1997 ISU Alumni Conference proceedings publication, especially Luanne Jorewicz, for her hard work. NASA provided a tremendous service to the ISU through their willing assistance and significant technical expertise in the publication of these proceedings.

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Contents

Contents
(continued)

Technical Paper *Page*

Room 113

Contents
(continued)

Technical Paper *Page*

Contents
(continued)

Contents
(continued)

Technical Paper *Page*

Session 1
Technology Transfer

S1-81
040713
289486
p. 6

COMMERCIALIZING U.S. FEDERAL LAB TECHNOLOGIES AND PROSPECTS FOR PRIVATIZATION

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Abstract
This paper presents the general process for commercializing U.S. federal lab technologies, with focus on NASA's current practices. Understanding the details of this process is useful when considering ways to make federal technology commercialization more effective. Furthermore, it is asserted that many of the activities now performed by the federal government can be done more effectively by a for-profit market-driven enterprise, and that a whole industry can be created around commercializing the $70 billion per year[1] that the U.S. government spends on R&D. Examples of four classes of potential profit centers are discussed, in order to open the dialogue for discussions along these lines, for the benefit of the global space program, and the alumni and staff of the International Space University.

Background
NASA has recently embraced commercialization as an integral part of its mission. Much of the estimated $5 billion per year that NASA spends on technology research and development has application to commercial non-aerospace industries. In the early 1990's, the Clinton Administration's National Performance Review[2] encouraged all federal labs to more effectively commercialize the results of their internal R & D activities. NASA's response was to assemble a high level working group that recommended creating a coordinating function at NASA headquarters and commercialization field offices at each of the ten NASA field centers. This became NASA policy, as documented in the June 1994 *Agenda for Change* [3]. Building on that policy, an aggressive, proactive, disciplined process for identifying, packaging, and transitioning these technologies into commercial enterprises has proven to be more effective than a haphazard, serendipitous approach to commercialization.

The History of Commercialization at the Johnson Space Center
Technology Transfer and Commercialization at the Johnson Space Center (JSC) started to emerge as a primary mission while it was a division-level organization within the New Initiatives Office. It included Technology Utilization (TU) and Small Business Innovation Research (SBIR). Patenting and licensing was done by the patent attorneys in the JSC Chief Counsel's office.

TU's primary activity was to recommend articles on JSC innovations for publication in NASA Tech Briefs. For 20 years, this monthly periodical was the only regular source of information about NASA technologies available to outside industries. When a company was interested in a particular technology, they could request more detailed information in the form of a Technical Support Package. After this, options were limited. There were few mechanisms established to facilitate development of joint activities or to patent and license the technologies with the highest commercial potential.

The federal SBIR program provides $1 billion per year to small U.S. R&D companies. It was created by Congress to give small businesses access to federal R&D funds. Each federal agency administers the SBIR program slightly differently. NASA provides Phase I awards of up to $70,000 each and Phase II awards of up to $600,000. Phase I

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[1] *Federal R&D Funding by Budget Function: Fiscal Years 1994-96.* An SRS Special Report. Ron L. Meeks, Principal Author. NSF 95 342. National Science Foundation. See http://www.nsf.gov/sbe/srs/s1296/htmpdf.htm

[2] The recommendations for NASA are documented in an accompanying report to: *From Red Tape to Results: Creating a Government that Works Better and Costs Less,* Report of the National Performance Review. September 7, 1993. See http://www.npr.gov/library/reports/nasa.html

[3] *Agenda for Change*. June 1994. NASA Headquarters. See http://nctn.hq.nasa.gov/nctn/Agenda/Contents.html

provides funding for a 6 month concept development. Phase II provides funding for a two year proof-of-concept. Currently, at NASA, about 1 in 10 applicants receive a phase I award. Of those, about 50% receive phase II funding.

The Johnson Space Center has always been fairly aggressive in pursuing patent protection. Historically, JSC receives about 30 patents per year. Contrary to popular believe, technology developed by the federal government does not immediately go into the public domain. This used to be the case. However, NASA discovered that many companies were not interested in technologies unless that were given some sort of exclusive rights. They needed some assurance that they would not be beaten to the marketplace by a competitor after they had invested millions of dollars in product development. Learning this, NASA started patenting technologies, then granting royalty-free licenses to whomever expressed interest. They found that this also was not an effective means of commercialization. Since companies didn't have anything invested in the technology, they were less committed to actually doing something with it. At that point, NASA started asking for upfront and ongoing royalties, as well as a commercialization plan containing due diligence milestones for patented technologies.

Even though JSC patent attorneys were quite successful at getting patents, they did not have the resources or experience to effectively evaluate technologies based on commercial potential. Nor were they able to aggressively market the patents, once received. Unless a company discovered a technology via Tech Briefs or heard about it through a champion within JSC, the patents would languish.

JSC's Technology Transfer and Commercialization Office[4] was formed in March 1994. It elevated commercialization at JSC from a division to a directorate level office. The philosophy was to keep the office small. This office would coordinate commercialization activities versus manage every aspect. The office currently has 20 civil servants and 4 contractors. Commercialization officers and patent attorneys work closely with points-of-contact within the 24 technology divisions and the 20 contractor companies. In addition, the office works closely with the NASA commercialization offices at the other 9 field centers, the NASA headquarters commercialization office, the 6 Regional Technology Transfer Centers, the National Technology Transfer Center, and the Research Triangle Institute.

Shortly after its formation, the patent attorneys were transferred from the general legal office to the new commercialization office. This resulted in much closer ties between patenting, evaluation, marketing, and licensing functions. JSC now pursues patent protection primarily based on the commercial marketability of the technology and aggressively markets these technologies once the patent application is filed with the Patent and Trademark Office.

Generally, NASA technologies, as well as all federal lab technologies, are known as a pretty good deal. NASA's motive is the successful commercialization of the technology, not in securing the maximum royalty fees. In most cases, NASA technologies are available for a fraction of the street value.

The three primary functions at the NASA-JSC Technology Transfer and Commercialization office are: 1) Technical assistance, 2) Development of cooperative R&D agreements, known as Space Act Agreements, and 3) patenting and licensing of JSC technologies that have the highest commercial potential.

The focus at JSC is patenting and licensing. Other field centers may take a different approach. For example, at the Marshall Space Flight Center in Huntsville, Alabama, the focus is providing technical assistance to U.S. companies. A company will present a specific technical problem to which they are seeking solutions. If Marshall has they needed expertise, they will attempt to provide solutions. If not, they will pass the problem on to another NASA field office or commercialization center

The Commercialization Pipeline

Figure 1 diagrams the flow of technologies from Invention Disclosure to Commercial Products. The four primary activities are: Inventory & Evaluation, Patenting, Marketing, and Commercialization. The process starts when Invention Disclosures or New Technology Reports are received by the Commercialization Office. However, an

[4] The latest detailed information about the JSC Commercialization Office can be found on their web site at: http://technology.jsc.nasa.gov

important activity preceding this is called Marketing InReach. Marketing InReach involves providing consulting services and briefings on the commercialization process to managers, scientists, and engineers within the science and technology divisions and within the contractor companies. It also establishes and maintains the network of contacts among the technology divisions and contractor community. The ultimate purpose of Marketing InReach is to incentivize innovators to file disclosures on all new technologies developed.

Generally, civil service inventors will file an Invention Disclosure (NASA Form 425) and contractors will file a New Technology Report (NASA Form 666A). This provides the Commercialization Office with information on the development of the invention, possible public disclosures, plus detailed technical information. This is used to assess the invention's technical merit, patentability, and commercial viability.

Rights to the technology are generally first offered to the contractor. In the great majority of cases (excluding SBIR contractors) the contractor defers to the government. An Evaluation Team within the Commercialization Office reviews the technology for: technical merit, patentability, and commercial viability. Documentation is then sent to a technical reviewer at JSC to determine technical merit. Patentability is determined by discussing the new, novel, useful and non-obvious features of the technology with the patent attorney on the Evaluation Team. Commercial viability is determined based on the experience of the Evaluation Team members. The commercial viability of those technologies that appear to have some potential will be confirmed by a 10-20 hour "Quicklook" market assessment.

Under the direction of the NASA JSC Marketing Team, the marketing Quicklook is performed by graduate students working for the Mid-Continent Technology Transfer Center, one of the 6 Regional Technology Transfer Centers. The process is fairly straightforward. The students review technical and marketing materials about the technology, then brainstorm with others on possible commercial applications. They will then look into various periodicals and data bases, notably, *Gale's Directory of Associations*. A few calls to various associations usually identifies industry leaders. Once the right people within the various companies are found, a few detailed questions will usually indicate whether the technology has market potential within the company's industry.

The technology is then prioritized for marketing, based on its commercial potential. Marketing materials are prepared, including a scrubbed version of the Quicklook Commercial Potential, a 2 page Technology Opportunity Sheet, the patent application (without the claims, unless it has issued), licensing information, plus any other information that will help companies make a informed decision on whether they want to license the technology.

Marketing starts with establishing a marketing strategy for the technology. For example, should the technology be licensed exclusively to one company or should it be licensed to a number of companies for various applications. Also, what sort of upfront and ongoing royalties should be expected. The marketing case manager for the technology will then start with companies outlined in the Quicklook assessment. He may also go into various corporate and financial data bases to find promising leads.

Interested companies will send the Commercialization Office a license application. This application is assessed for two aspects: the company's commitment and capability to successfully commercialization the technology. Commitment is gauged by the detailed breakdown and milestones in the commercialization plan. Capability is gauged by such things as the experience of the management team, financial commitments, and product development experience. Entrepreneurs are at somewhat of a disadvantage. However, enthusiasm and drive can help balance a lack of experience and financial wherewithal.

Terms of the license are all negotiated. It may include in-kind support from one or more of the labs at JSC. This could also be contained in a separate agreement, called a Space Act Agreement.

Incentives and De-incentives

Although the U.S. government has come a long way towards effective commercialization of federal lab technologies, many feel that there is still a long way to go. It's useful to assess the barriers to more effective commercialization in order to find ways through them.

Although senior management has espoused the merits of commercialization, there are still significant cultural barriers at all levels within federal organizations. Effective commercialization requires long-term, steady commitment. It often takes many months, even years, to realize a return on commercialization investments. This commitment can at times wane and become unfocused, in light of near-term crises. Needed resources are reallocated and commercialization momentum is lost. The case for continued resources can be difficult to make when competing against "mission critical" projects.

Public organizations also often suffer from a lack of accountability. Not dependent on licensing revenues for survival, the licensing of valuable federal technology can take on an academic attitude. This is in contrast to the for-profit business world, where inadequate revenues can mean dismissal, unmet payroll, bankruptcy, etc. Loss of livelihood can be a strong motivator.

Finally, all federal labs, NASA included, have a technical heritage, not a business one. Often the characteristics that make one a good scientist or engineer are directly opposed to those needed to create and build a business. Technical people don't usually know much about, or have much interest in, for example, business development, product development, finance and accounting, marketing, sales, distribution, etc. Therefore, the activities needed to successfully commercialize a technology are foreign. To them the steps needed by the lab to identify and package these commercially viable technologies are often not well understood within the lab.

Prospects for Privatization

This leads one to consider what subset of commercialization functions might be better performed by an outside organization, particularly a for-profit, market-driven enterprise. In fact, a case can be made that a whole industry can emerge based on commercialization of the virtually untapped federal lab R&D investments of an estimated $25 billion per year. Four classes of activities come to mind for near and mid-term profitability:

- Combining the right management teams and financial resources with the best federal lab technologies
- Commercialization consulting to federal labs, contractors and businesses
- Commercializing SBIR (Small Business Innovation Research) technologies
- Creating decentralized Rapid Prototype Labs

Effective Technology Commercialization

Effective technology commercialization requires; "*The Right Technologies, The Right Packaging, The Right Team, The Right Financing*". Packaging includes aspects such as sufficient development to prove the viability of the technology and the right technical, marketing, and product development information so businesses and investors can make informed decisions. Three fundamental things are needed for any enterprise to succeed: ideas, people, resources. The most difficult of these is people, particularly the management team. Next are the right ideas. Surprisingly, the easiest to find are resources, namely money. There is an enormous amount of venture capital available, particularly in the global marketplace, for the right management team around the right technologies. Although, this is the core of a potential for-profit, private industry, it's conceivable that this profit center, based on fees and equity positions, could take 3 to 4 years to become profitable. Near-term cash flow would need to be generated by the additional activities.

Commercialization Consulting Services

Many federal labs are arriving at the conclusion that they could use some help in commercializing their technologies, particularly considering the current trends towards downsizing and outsourcing. An experienced organization with clearly defined, no-nonsense accountability, could meet this need. This enterprise could be structured similar to Big 8 accounting and technology consulting companies such as McKinsey, Andersen Consulting or EDS. Services could also be provided to federal contractors, who know their markets are shrinking and badly need to diversify. In addition, many outside companies are looking for new sources of technology, but are unclear as to how to get these technologies out of the federal labs.

Commercializing SBIR Technologies

As discussed, the federal government spends about $1 billion per year on Phase I and Phase II SBIR contracts. Although the primary purpose of the program is to give small businesses access to federal research funds, there is increasing pressure for the results of this investment to more quickly reach the marketplace. There is an unfunded Phase III that is supposed to follow Phase II. The SBIR contractor is supposed to develop commercial products, based on the technologies developed in Phases I and II. The great majority of the time, this does not happen. In fact, companies often become "SBIR factories", winning a series of SBIR awards from various agencies, then reapplying and winning more when the first set expires. The phenomenon described above, where the best technical people do not always make the best business people, applies here too. An organization that partners with the most promising Phase I awardees could fill this niche.

Rapid Prototype Labs

Often, very promising technologies will not be sufficiently developed to prove to the commercial world that investment is warranted. Before investing millions in product development, business leaders want to have some assurance that the technologies will perform as advertised.

On the other hand, federal labs will often have sufficient funding to develop a promising concept, but will lack the funding to develop a proof-of-concept or prototype. A lab, or series of labs, that specialize in this, could be profitable. It might even be possible to partner with various federal labs, with, for example, the labs providing facilities, and the enterprise providing personnel. These decentralized, perhaps even "virtual" labs, could develop the most promising ideas from all sources, providing value-added to both the enterprise and the sponsoring labs.

Conclusions

The paper outlines the general process of commercialization within NASA, which parallels commercialization activities at many of the other 700+ federal labs. Understanding the details of this process is useful when considering ways to make federal technology commercialization more effective. An outside enterprise, indeed a whole industry could emerge to fill this large market niche. There are likely to be more classes of services that this industry can provide. It's hoped that this paper will open the dialogue for discussions along these lines.

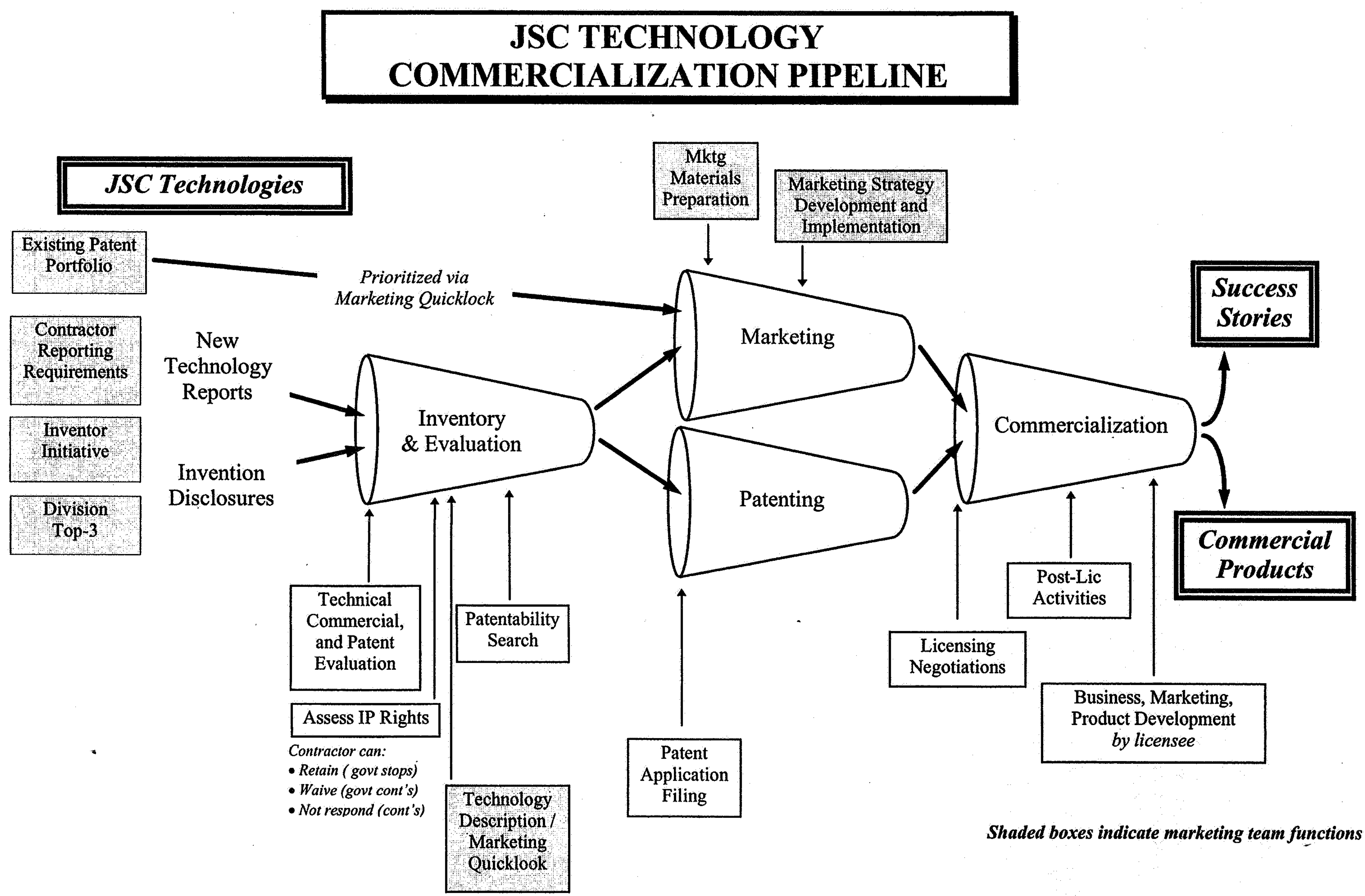
JSC TECHNOLOGY
COMMERCIALIZATION PIPELINE
JSC Technologies
Existing Patent Portfolio
Contractor Reporting Requirements
Inventor Initiative
Division Top-3
New Technology Reports
Invention Disclosures
Prioritized via Marketing Quicklock
Inventory & Evaluation
Technical Commercial, and Patent Evaluation
Assess IP Rights
Contractor can:
• Retain (govt stops)
• Waive (govt cont's)
• Not respond (cont's)
Patentability Search
Technology Description / Marketing Quicklook
Mktg Materials Preparation
Marketing Strategy Development and Implementation
Marketing
Patenting
Patent Application Filing
Commercialization
Licensing Negotiations
Post-Lic Activities
Business, Marketing, Product Development by licensee
Success Stories
Commercial Products
Shaded boxes indicate marketing team functions

Figure 1

52-82
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P7

APPROACH TO CREATION OF A CONVERSION DEVELOPMENT INFORMATION SUPPORT SYSTEM (CDISS)

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Abstract

The problem of information support is the key one for any decision-making process. Especially important it appears to be for conversion and technology transfer projects where huge amounts of information of various types must be gathered and processed at both business-plan development phase and project running phase. Therefore, an information support system is a critical issue especially when conversion and technology transfer activities merge into a single routine (conveyor-type) process. This paper details some possible approaches to creation of such an information support system for enterprise level and industry level.

Introduction

Over the last years, conversion and technology transfer became very critical for the aerospace and defense industries because of reduced government funding and significantly lower public support. Thus, national agencies (ministries) and enterprises themselves have been trying to find alternative applications for their products, production capacities and high-qualification labor.

The problem of information support is the key one for any decision-making process. Especially important it appears to be for conversion and technology transfer projects [1] where huge amounts of information of various types must be gathered and processed at both business-plan development phase and project running phase.

Thus, there is required a new approach with the major objective that, when conducting conversion or technology transfer project, it is necessary to understand the conversion process not as a unique one but rather as a continuous (integral) routine process. This can only emphasize how much critical the information support system is. Within the framework of such a integral routine process the transfer of technologies and information has to go all the way from the developer down to the end user. Therefore, both organizational and financial schemes of technology transfer process must be revised.

At the moment, there are two basic levels of conversion which may be considered: a nation-wide conversion, and conversion within the framework of a single enterprise or company. Nation-wide conversion is dealt with at the corresponding national agencies/ministries, various national programs are being developed - different for each country and industry. However, the enterprise-level conversion seems to be not developed well enough by now, whereas it is, in fact, the most critical (and difficult) issue.

In general, the goal of the enterprise-level conversion may be stated as follows: *increase profitability for the manufacturer and ensure profit for the investor(s).*

There is a number of reasons why an enterprise may be seeking a conversion opportunity. It may be because of a drop in profitability/payability of the goods/services market(s) for the enterprise's usual specialization; insufficient government funding or its absence (for instance, in cases of defense or space industries); or because specialization (profile) of the enterprise became morally obsolete in the light of rapid development of new sciences and technologies where the particular case may be a planned restructurization of the enterprise's production profile by the government.

Finding a solution to a conversion or technology transfer problem means that the enterprise develops a business-plan for the new product/services proposed and guarantees its reliability, whereas the investor provides funding. Both enterprise and investor then work together (or just the enterprise alone) to implement the accepted and approved business-plan.

The enterprise and the investor have different tasks to solve (questions to answer) at both the business-plan development and implementation phases. The enterprise has to conduct the market study based on the enterprise's current profile (in terms of available technologies and labor qualification), assess its own production capacities, develop a business-plan (standard methodologies for which exist [2]) and support it by trustful guarantees for the investment.

The investor has to make sure the proposed business-plan is realistic and acceptable, and safe enough to invest in it. Also, the investor may be interested in having an investment insurance in addition to the guarantees brought up by the enterprise.

Solution of all these problems requires a great amount of business, financial, technological, legal and social information to be obtained and handled. Obviously, this only can be done if one has some sort of an information support system (ISS). The better this system is, the more efficiently the conversion process will work out. Here below the approaches to creation of an enterprise-level and industry-level ISS are discussed.

Enterprise-level Information Support System

An enterprise-level CDISS involves both the enterprise and the investor and works for both of them. The goals of such a CDISS are: ensuring that both the manufacturer and investor get *all the necessary information* for further analysis and decision-making and creating an efficient operating system/mechanism to control the process of conversion or technology transfer.

The major functions of such a CDISS are:

- - collection and verification of the relevant information from various sources (including but not limited to: commercial on-line databases, news and all types of mass media, Internet, corporate database resources, etc.);
- - conversion of the information obtained into the necessary format which is convenient for further work;
- - ensuring the group work with the documents and their automated control;
- - establishing a joint trust control over the enterprise for both the owner and the investor.

The investor and the enterprise will need information of various types. One type is business and political information (business news, stock exchange information, tariffs for third parties' services, future trends and analysis for the given technology, information on investment projects and investors, political and economical situation in the regions of manufacturing (including manufacturing of supplied parts) and product marketing, legal aspects, etc.). Also important is information on the manufacturer (production capacity and costs, feasible technologies that may be used, enterprise specialization, personnel qualification) and cooperation (potential suppliers of the raw materials and parts, their business reliability, product quality and prices). Possible sources of information form an information retrieval system (Table 1).

Table 1.

sources of information	type of source
- mass media of all types	newspapers, TV, radio, Internet
- stock exchange	hot line

- legislation	database
- specifications for the manufacturer's equipment	enterprise's database
- personnel qualification	enterprise's database
- database on suppliers	database
- analytical reviews on given technology	commercial on-line databases

A general information flow chart and the proposed CDISS block-scheme are shown on Fig. 1 and 2.

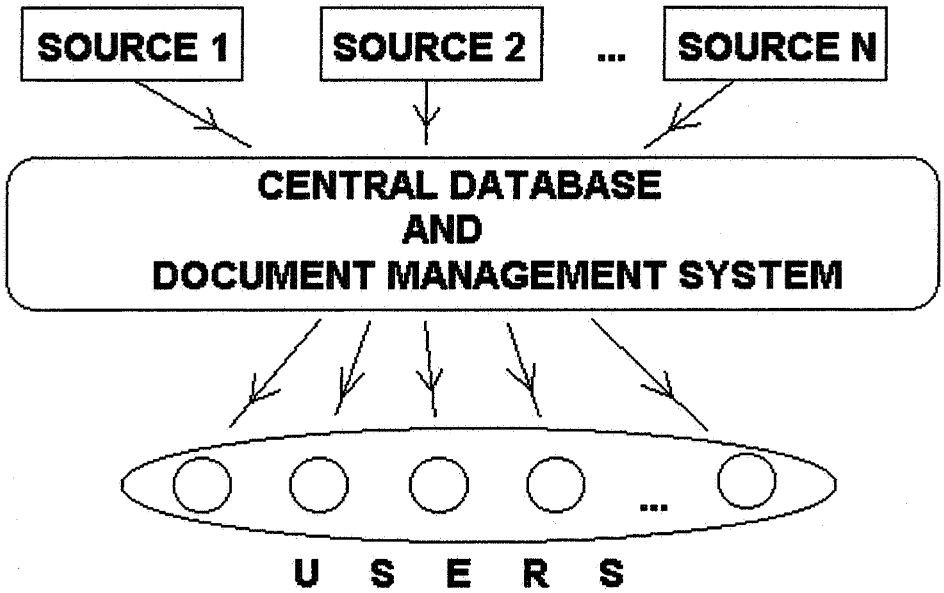

Fig. 1. General information flow chart

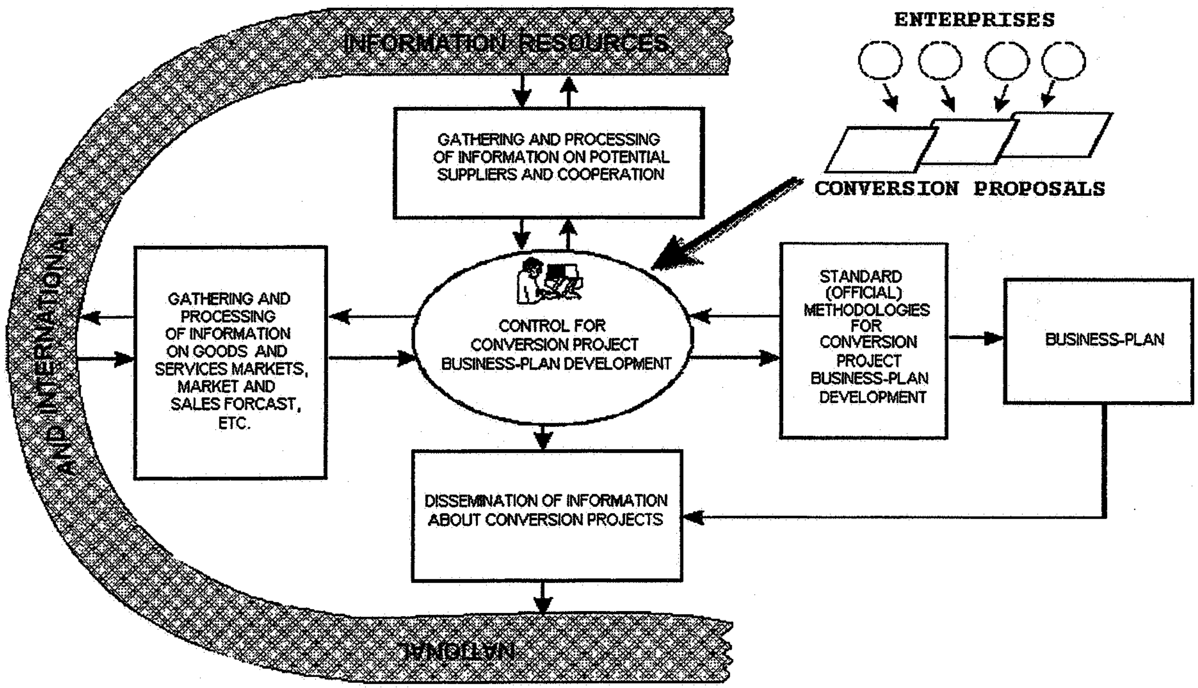

Fig. 2. CDISS block- scheme [3]

Such a CDISS may use client-server, mainframe+terminals or Intranet architecture (technology). The users of such a CDISS may be split into several groups: enterprise top management, investor(s), auditors, shareholders and mass media (information generally available for public)

Industry-level Information Support System

Quite different approach is necessary when considering conversion process at an upper level: the level of the entire country or international cooperation.

The conversion at this level has the following goals: restructurization of some industries or branches; increase of investment efficiency for promising scientific studies and research; maintaining of the country's so-called "safety factor" for strategic/critical fields from economic and political points of view.

Conversion and technology transfer at this level means integration of the conversion development itself, its adaptation process, technology transfer legal and financial support, and information transfer - into a single system which ought to enhance the efficiency of each of these component processes, and guarantee the overall success of the particular projects.

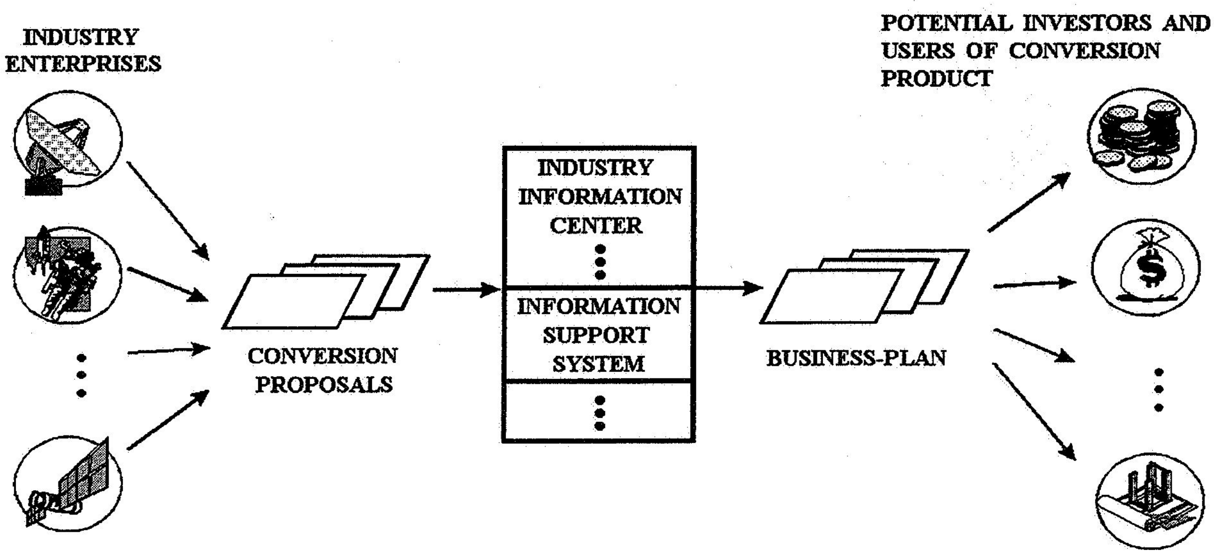

Fig. 3. Place of the Information Support System in the industry's conversion activities structure [3]

As was mentioned above, the financial and organizational schemes of the conversion process need to be revised if the light of this new approach proposed. Such a restructurization, first of all, implies creation of a united information center (at various levels - industrial, inter-industrial, regional, national, international) for the actual information collection and processing. A system of organizations doing financial, legal, patent support, etc. also will have to emerge or be formed. There will be needed a system of scientific-technical centers for technology development and adaptation, as well as organizations to do the job relating to technology assimilation at the end user organization (including marketing services, psychological and sociological support, public relations, etc.).

Conclusions

In conclusion, it is necessary to emphasize that an enterprise-level CDISS built on the principle proposed for it will allow to better aim and more correctly develop the business-plan, select the most reliable and safe way to invest, better ensure the investment return, provide information on potential investors (critical for the enterprise) and potentially profitable enterprises (critical for the investor), control and manage the project in the real-time as well as establish an integral and "transparent" system to monitor investment spendings and project progress.

On the other hand, an industry-level CDISS based on the principle proposed for it will allow to optimize the national technology transfer cycle by means of forming an all-inclusive and end-user-oriented flow of technologies and related information, minimize the high-tech development or dual technology investment return period, control the process of absorbed technologies integration into the production as well as attract additional investments for venture operations.

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S3-84
040 715
289493
p8

SEA LAUNCH SPIN-INS

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Abstract - The Sea Launch program is an excellent example of diverse spin-ins for the commercial space launch industry. It also relies on a high degree of international cooperation, which has been made increasingly possible and necessary within the last decade. Sea Launch's specialized operational concept necessitates the creation of new technologies as well as the pioneering of unique applications of existing technologies. As a new commercial program, Sea Launch is free to apply new technologies based on relatively unconstrained trades.

Sea Launch spin-ins are highlighted in the table below:

Sea Launch Element	Source of Spin-in
Zenit Stages 1 & 2	Russian military space launch vehicle
Block DM-SL Upper Stage	Russian military & civil upper stage
Launch Platform	Japanese oil rig, modified by Norwegian shipbuilding company
Assembly and Command Ship	Norwegian Roll-On Roll-Off cargo ship
Communications and Tracking	Russian surveillance ship, Russian & American communication satellites

Sea Launch also includes other elements not normally associated with space launch processing, such as a helicopter for transportation of personnel between the Assembly and Command Ship and Launch Platform, a Global Positioning System for ship navigation, etc.

This paper will give an overview of the spin-ins mentioned above, and will suggest creative consideration of other options for technology transfer.

Introduction - Sea Launch is an international joint venture created to launch commercial satellites from a platform in the Pacific Ocean. The Sea Launch system includes the following main elements: the Zenit-3SL launch vehicle, the Launch Platform, the Assembly and Command Ship, the spacecraft and launch vehicle processing facilities, and tracking and communications assets. The Launch Platform and Assembly and Command Ship are based at the Sea Launch Home Port in Long Beach, California, USA.

The Sea Launch operational concept was chosen over a traditional land-based approach because of several significant advantages. Launch will take place in the international waters of the Pacific Ocean about 1600 kilometers south of Hawaii. At such a remote location, due to freedom from terrain overflight considerations, the vehicle can be directed at any azimuth. For satellites bound for equatorial orbits, launching at sea permits launching directly from the equator, which eliminates costly orbit plane change maneuvers and takes advantage of the earth's rotation. The combination of benign weather and the extreme stability of the submersible platform makes it possible to launch year round.

The joint venture is made up of four corporate partners. Boeing Commercial Space Company of the USA conducts overall system integration, marketing, and mission analysis and provides the Home Port, payload fairing, spacecraft adapter, and interface skirt. RSC Energia of the Russian Federation provides the Block DM-SL upper stage and the automated support equipment. KB Yuzhnoye/PO Yuzhmash of Ukraine provides the first two rocket stages. Kvaerner of Norway provides the Assembly and Command Ship and Launch Platform.

Launch Vehicle Spin-ins - The first two stages of the Zenit-3SL are adapted from the two stages of the Zenit launch vehicle, originally developed for rapid deployment of military satellites. One of the key technologies for Sea Launch is the automatic fueling and mating system, which makes over 200 connections autonomously. After all personnel have been evacuated from the Launch Platform to the Assembly and Command Ship and the launch vehicle is erected on the pad, the automatic fueling and mating system autonomously performs the necessary disconnections and retractions to protect the Launch Platform and its equipment from the effects of the launch. This autonomous capability makes it possible to launch at sea with a minimum crew.

The Block DM-SL upper stage is adapted from the Proton upper stage (Block DM), which traces its heritage back to the fifth stage of the Soviet N-1 moon rocket, and has enjoyed a legendary record of over 160 flights. The Block DM-SL is being fitted with a state of the art control system to allow launching at sea.

The pneumatic actuators which help jettison the graphite composite fairing halves are adapted from automobile hatchback dampers. The acoustic protection system inside the fairing is adapted from foam designed for muffling sound in bridges and buildings. The vent cover material for the interface skirt is adapted from refrigerator magnets. Some of the spacecraft command sequencer avionics rely on commercial (instead of space-rated) sensors and chips, saving approximately 47,000 US dollars per chip.

The integrated launch vehicle is shown in Figure 1.

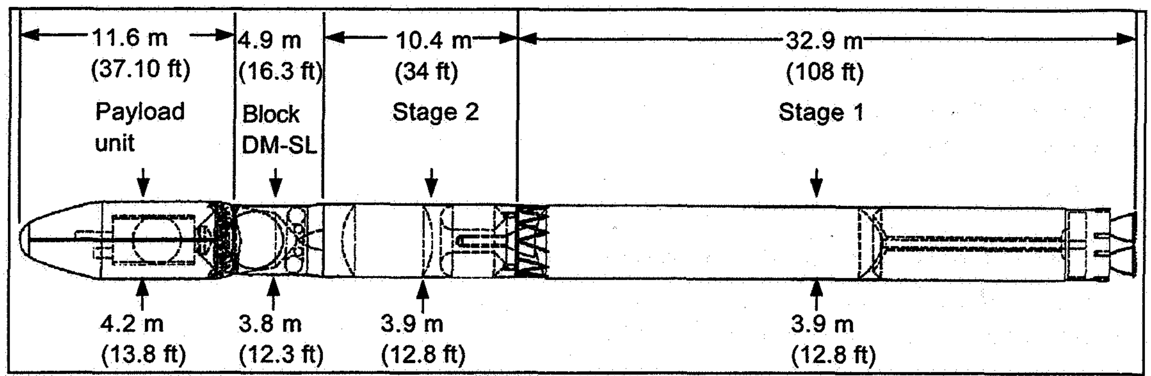
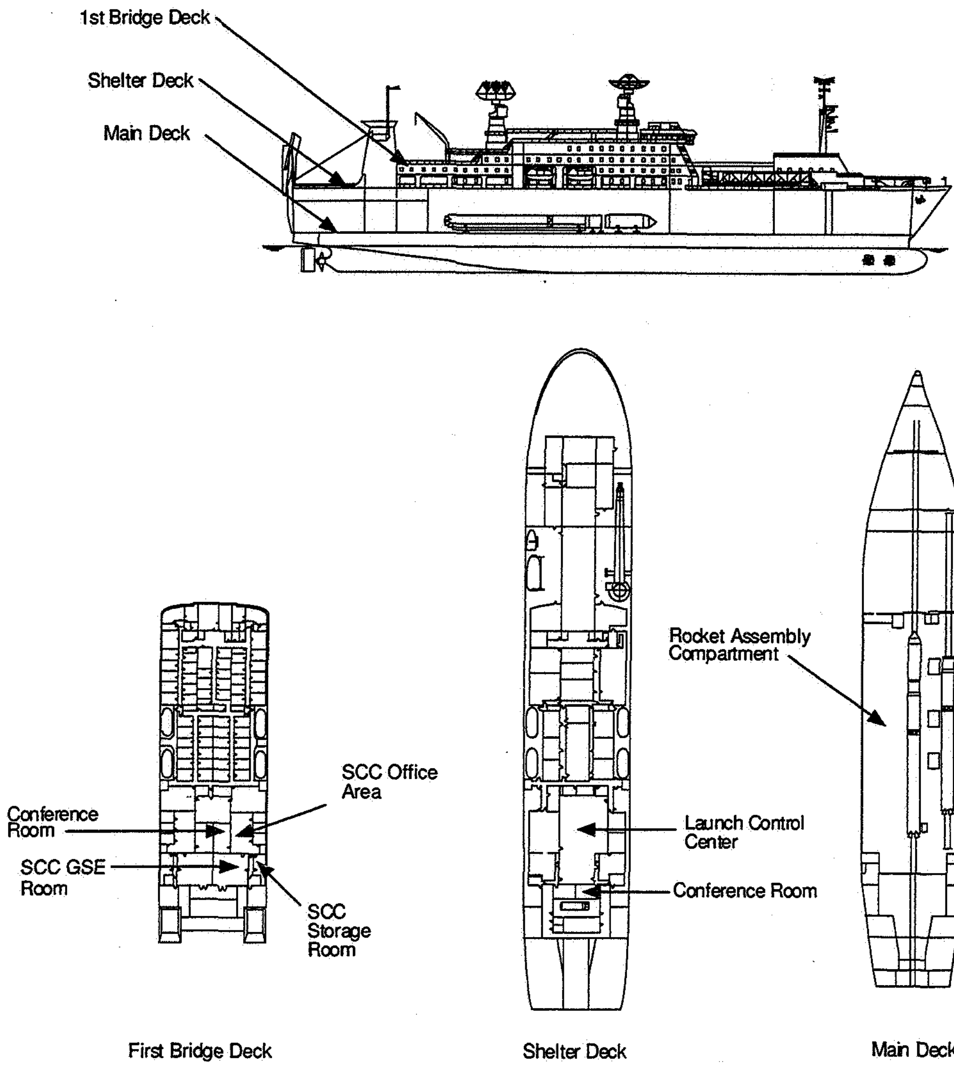


Figure 1 - Integrated Launch Vehicle

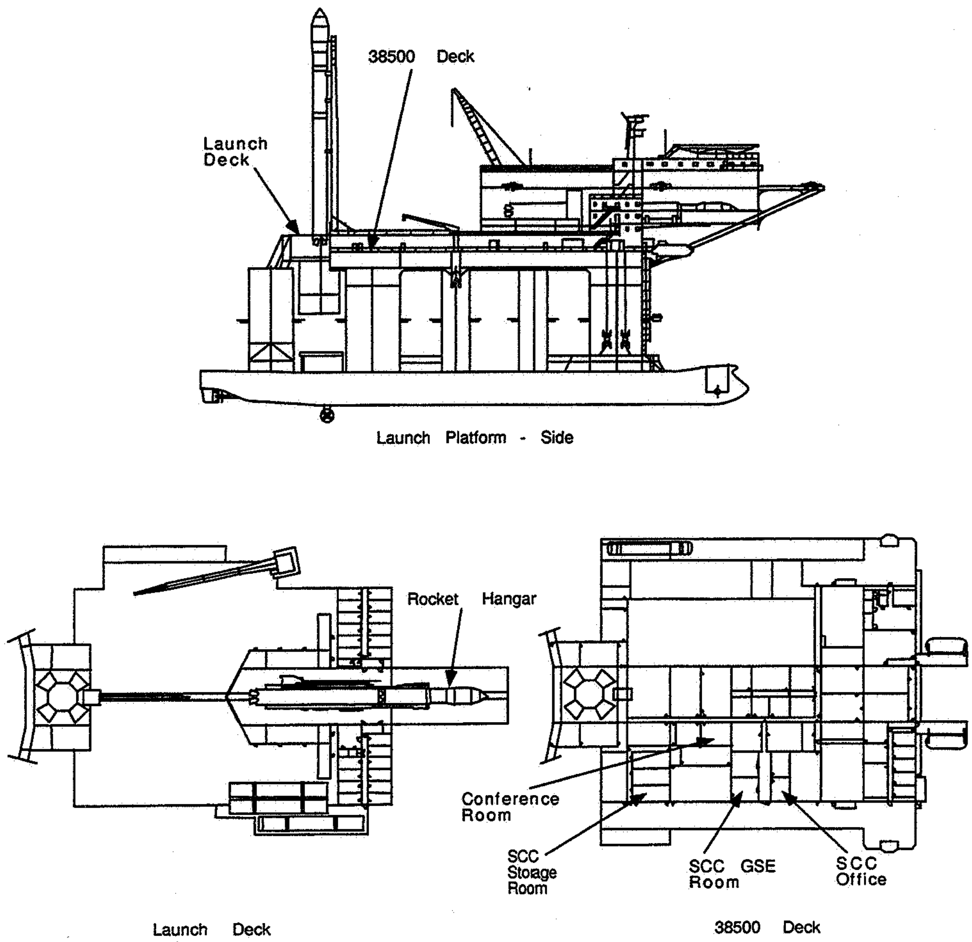
Marine Systems Spin-ins - The Assembly and Command Ship, "Sea Launch Commander" (shown in Figure 2) was constructed especially for Sea Launch, and is based on a standard "Roll-On Roll-Off" cargo ship design (the term "Roll-On Roll-Off" refers to the capability to load and unload cargo via a ramp in the ship's stern). The Assembly and Command ship also features a swimming pool and staterooms adapted from a cruise ship design. The Rocket Assembly Compartment, Launch Control Center, and support equipment rooms are similar to traditional land-based processing facilities. Launched December 12, 1996, the 200 meter long, 33 meter wide ship is being readied for installation of launch support systems.



GSE - Ground Support Equipment
SCC - Spacecraft Contractor

Figure 2 - Assembly and Command Ship

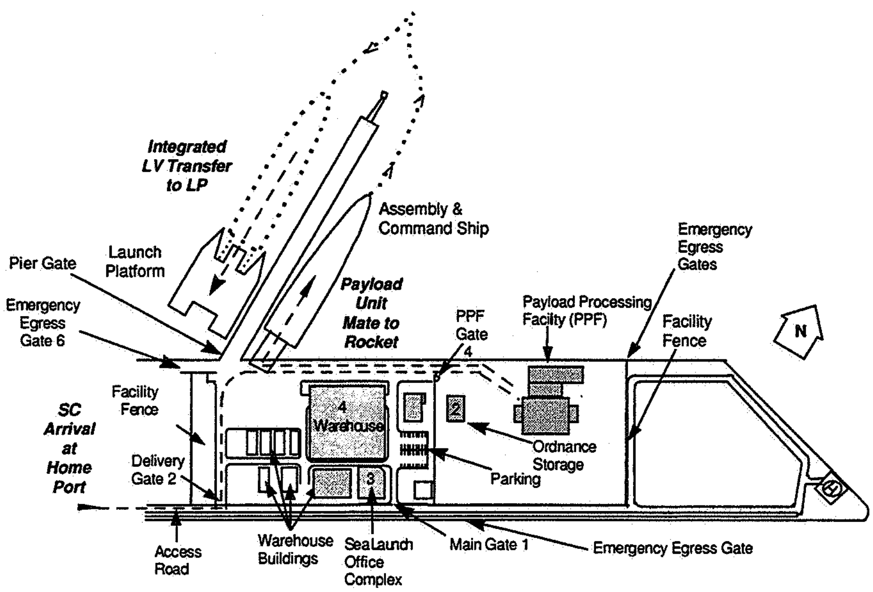
The Launch Platform, "Odyssey" (shown in Figure 3) was originally built in Japan, and served as an oil platform in the North Sea. It is 110 meter long, 70 meter wide, self-propelled and semi-submersible to approximately 22 meters. Modifications include the addition of a launch pad, hangar, and rocket transporter/erector. The Launch Platform achieves precise stationkeeping through the use of the Global Positioning System. Prior to launch, Launch Platform personnel are transported to the Assembly and Command Ship via helicopter. The Launch Platform began sea trials in early May 1997.



GSE - Ground Support Equipment
SCC - Spacecraft Contractor

Figure 3 - Launch Platform

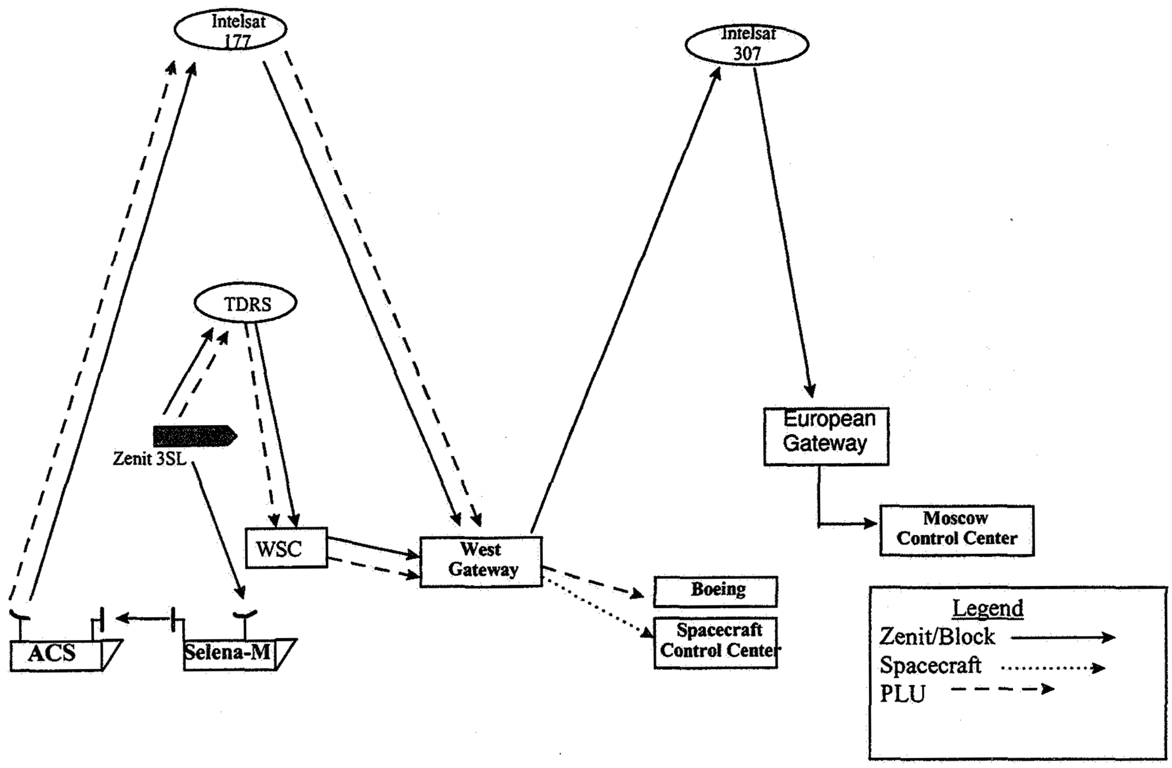
Home Port Spin-ins - The Home Port site is part of the former Long Beach Naval Station and is located on the end of the "Navy Mole" which is a large, artificial breakwater. Some of the original Navy buildings are being renovated to serve as office space and warehouses. The Assembly and Command Ship and Launch Platform are moored at an existing 335 meter long pier. Groundbreaking ceremonies were held August 8, 1996. The facility will be completed in October 1997, and is shown in Figure 4.



LP - Launch Platform
LV - Launch Vehicle
SC - Spacecraft

Figure 4 - Sea Launch Home Port

Communications and Tracking - The communications and tracking architecture shown in Figure 5 for the post-fairing-jettison phase ensures complete visibility of the Sea Launch trajectory. It includes antennas onboard the Assembly and Command Ship and the "Selena-M" tracking ship, formerly used for Soviet military surveillance. Space-based communications elements vary by phase, and include commercial Intelsat services, NASA's Tracking and Data Relay Satellite, and the Russian "Luch" satellite (also known as "Altair").



ACS - Assembly and Command Ship
PLU - Payload Unit
TDRS - Tracking and Data Relay Satellite
WSC - White Sands Complex (TDRS ground terminal)

Figure 5 - Sea Launch Communications Architecture

Implications for International Space University Design Projects - Students are advised to adopt a "consumer mentality" in examining seemingly-unrelated technologies for use in the space sector. Often by the time a technology is marketed commercially, its entire life cycle is sufficiently well understood to be adapted for space use. Ironically enough, technologies which started as spin-offs from the civil space program of the 1960s, such as microelectronics, have continued to develop along commercial lines, and are now being used as "spin-ins" back into the space industry.

Conclusion - The entire commercial space industry may be thought of as a gigantic spin-off from the civil space industry, which was originally a thinly-veiled demonstration of ballistic missile technology during the height of the cold war. Today, the commercial space sector is enjoying record profits and mounting backlogs, creating new products and jobs, and increasing living standards. This success is a perfect example of technology transfer leading to the development of an entirely new business sector.

Sea Launch management has implemented an open-minded technology selection approach since the inception of the program. This policy allows engineering teams to choose innovative concepts and processes based on their technical merits and commercial value.

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NASA INTELLECTUAL PROPERTY NEGOTIATION PRACTICES AND THEIR RELATIONSHIP TO QUANTITATIVE MEASURES OF TECHNOLOGY TRANSFER

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Abstract
In the current political climate NASA must be able to show reliable measures demonstrating successful technology transfer. The currently available quantitative data of intellectual property technology transfer efforts portray a less than successful performance. In this paper, the use of only quantitative values for measurement of technology transfer is shown to undervalue the effort. In addition, NASA's current policy in negotiating intellectual property rights results in undervalued royalty rates. NASA has maintained that it's position of providing public good precludes it from negotiating fair market value for its technology and instead has negotiated for reasonable cost in order to recover processing fees. This measurement issue is examined and recommendations made which include a new policy regarding the intellectual property rights negotiation, and two measures to supplement the intellectual property measures.

Introduction
NASA long perceived as a leader in technology transfer has in recent years taken an even more proactive stance towards technology transfer. Concurrently, the sentiment of the public and Congress has pushed agencies like NASA to demonstrate their utility to the common good and justify their funding. This has placed additional emphasis on NASA's technology transfer efforts.

While federal legislation has allowed for more innovative ways of performing technology there is increasing accountability put on the agencies performing technology transfer [Bayh-Dole 1980, Stevenson-Wydler 1980]. Congresspersons want to be able to tell their constituents what they are getting for their investment (tax dollars). Typically, the interest of the Congresspersons is in a tangible, current commodity like increased jobs, saved production costs, new companies, new products and markets, improved quality of life or some other like measure. In addition, the benefits must be distinguishable in a short time frame (say, a Congressional term). Long-term benefits are of little use or interest to Congresspersons who are driven by re-election concerns, similar to American businesses fixated by the stockholders short-term interests. The current dilemma of NASA is to show its impact on the private sector to its stakeholders (Congress, public, industry, etc.) within the context of their stakeholder's measures.

Recently, a low-level debate has stirred within NASA on appropriate measurement methodologies for technology transfer and just as important, best practices to achieve maximum economic impact. The policies and philosophies are not consistent across the agency, nor perhaps should they be. For example, NASA Langley Research Center and Johnson Space Center focus their efforts on intellectual property by attempting to market its technologies. In contrast, Marshall Space Flight Center focuses its attentions on technical assistance through the solution of problem statements. This diversity of focuses is a strength for the agency but also points out the difficulty in establishing a single-minded policy for metrics. Metrics should be tailored to the specific goals and objectives of each particular organization. Many of the participants of an inter-agency working group to define common metrics, and the participants of a recent technology transfer metrics summit in Santa Fe will attest to the diversity in missions, objectives, and goals of the various agencies performing technology transfer. Both of these groups had significant difficulty in just arriving at a commonly agreed upon definition for technology transfer.

The difficulties in measurement arise when attempting to place a quantitative figure on the impact of the technology transfer program to the private sector. One of the few easy methods of deriving measurements is to utilize the intellectual property data (i.e. licenses, royalties). This data is typically well documented, easily defendible and offers simple numerical values. Previous analysis of intellectual property data shows a strong correlation coefficient between licensing and royalties [Bush 1996]. In addition, some case studies have shown that licensing can be a good indicator of success [Bush 1996]. In these previous studies the importance of utilizing other metrics in conjunction is also emphasized, like the amount of royalties per license. Herein lies the crux of the measurement issue for some agencies with licensing philosophies similar to NASA.

Historically, NASA has not negotiated for fair market values, settling only for reasonable sums. Therefore, while the number of licenses by NASA indicates an active organization in terms of technology transfer, the royalties indicate a below-average return but not necessarily below-average impact. Typical upfront licensing fees negotiated by NASA are between $5,000 and $10,000 with sales royalties dependent upon factors including field of use and others but historically ranging from 3% to 5% of the sales. If these royalties are used as the sole or primary measure of economic impact the benefits of the technology transfer are woefully undervalued since these values do not represent the fair market value, but typical minimum processing costs. And even more importantly that intellectual property returns are but one measure of the impact of technology transfer. Many transfers occur through other mechanisms.

A prime example of this philosophy occurred during licensing negotiations for a NASA developed material. NASA Langley Research Center was negotiating the licensing fees and fields of use for this new material with several companies. When the upfront licensing fees were disclosed by NASA under the older NASA licensing policy/philosophy, the companies found the fees to be significantly lower than industry standards. Evidence indicates that these companies were willing to pay upfront licensing fees equivalent to the fair market value of approximately $100,000 or more [Manuel 1996]. Prior to this project, this amount of licensing fees were viewed as excessive by technology transfer management within NASA and were not negotiated or expected. This philosophy is at the crux of the current debate.

Those who support the status quo opine that NASA should not be negotiating large royalty fees but should be providing a public service. In addition, those in support of small royalties feel that a large upfront royalty payment or a royalty rate (based on sales) that is too high may prevent a small capital-challenged company from competing for licenses or succeeding when they do receive a license. Those who would like to see a change feel that the current system does not fit well into a capitalistic system and puts the government in the role of "picking winners" and this is an inappropriate activity for civil servants. In addition, they charge that a fair market value is reasonable and in the words of Terry Willey of the Purdue Research Foundation: "If I have the opportunity to obtain a return, one that fairly reflects my contribution to a licensed product, and that return is used to perpetuate a program that is important to local and national initiatives, then it is my responsibility to do so. In fact, it may be irresponsible for me not to" [Willey 1996]. The model followed by the majority of universities and agencies is the latter.

NASA Technology Transfer

In this study an attempt to quantitatively assess NASA's intellectual property technology transfer efforts began by reviewing it's intellectual property royalties and technology transfer budget. The sales extrapolated from annual royalties were compared with the technology transfer expenditures/budgets to assess the quantitative effectiveness of the technology transfer activities. The technology transfer expenses were selected over research and development budget since NASA would perform the research for its primary mission objectives of aeronautics and space regardless of the technology transfer mission. The technology transfer budget reflects the additional effort required to commercialize the technology to the non-aerospace market as opposed to the primary market of aeronautics and space. The royalties were selected since they were the only available quantitative data measurement of sales for the agency. Currently an agency wide study is being performed to obtain all of the quantitative and qualitative metrics for technology transfer. The results of that study will be available after the publishing of this paper and will be subject to further publication.

Industry sales based on NASA intellectual property activity is calculated by extrapolating NASA's annual royalties received data. The NASA royalties database contains both upfront and running royalties received. To derive the approximate industry sales, it was assumed that all of the royalties received were running royalties negotiated at five percent (5%). Therefore the royalties received were multiplied by 20 to determine the sales. This is undoubtedly a rough estimate of the economic impact but considered to be within reason and may be somewhat balanced by any lost revenues due to non-payments which are not accounted for here.

These calculated sales are compared to the technology transfer budget for the same year. Many studies of product commercialization have determined that the time to commercialization can be anywhere between 6 months and 20 years with typical times being somewhere in between [Griliches 1988]. Therefore, it may be fairer to compare the extrapolated sales to the technology annual transfer expenditures 4 to 5 years prior. Though, in an effort to keep the assumptions conservative a conscious decision was made to compare the sales and budget for the same year.

The comparison of extrapolated sales to budget shows the sales trailing the expenditures (Figure 1). This evaluation demonstrates the pitfall of relying entirely on existing quantitative data. If the analysis of the NASA program were to stop here, the economic benefits of the technology transfer would be questionable. But a deeper probe of the program finds evidence that the program is both economically and socially beneficial.

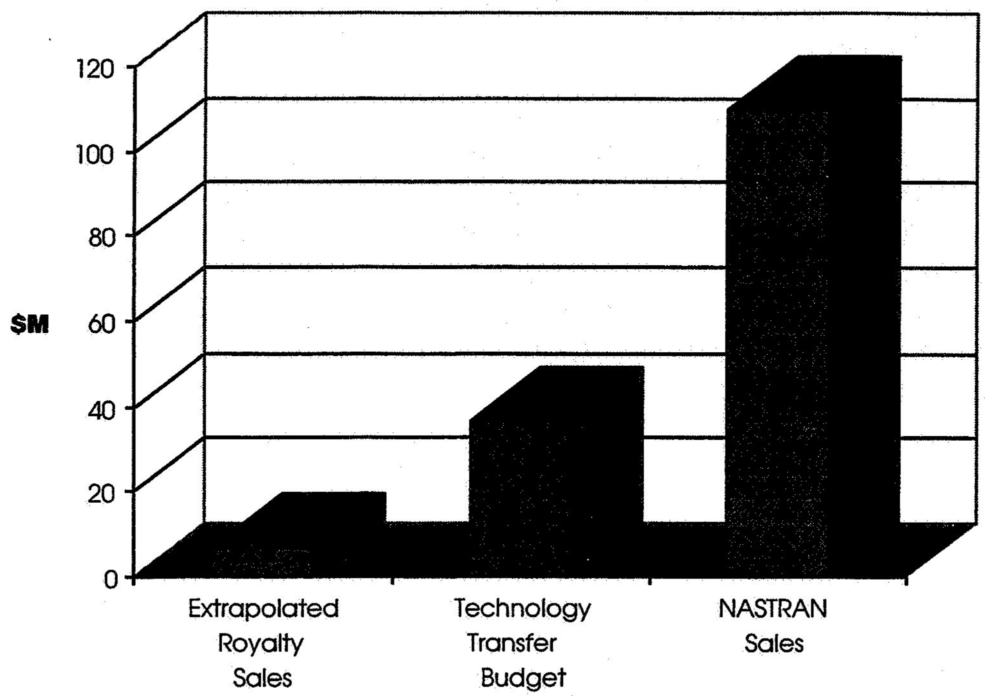

Figure 1. Comparison of NASA Technology Transfer Budget, Extrapolated Sales Based on Royalties, and NASTRAN Sales (1995 values)

Several case studies of NASA technology transfer have been previously performed [Bush 1996]. One case study alone accounted for sales in the private industry for the same year of $110M based on a NASA technology transfer. (Figure 1). In this case, NASA initially invested $3M from 1965 to 1968 into the development of a structural analysis code entitled NASTRAN [MacNeal 1988]. The code eventually became the staple commercial product of the MacNeal Schwendler Corporation, NASA's partner in development. This case study demonstrates the value of performing both quantitative and qualitative measures.

In order to account for lost opportunity costs an alternative of 10% compounding interest was assumed on the initial $3M investment over the 30 years since NASA's investment. This investment would now be worth a mere $27M which is approximately one third the magnitude of the current MacNeal Schwendler Corporations sales of $110M.

A plot of the investments by both NASA and the MacNeal Schwendler Corporation and the sales by MacNeal Schwendler Corporation demonstrate the phenomena of induced investments and success (Figure 2). Induced investments are expenditures incurred by partners attempting to commercialize transferred technology. Universities have long since recognized the value of utilizing induced investments as a measurement of the potential commercial success of a technology transfer [Pressman 1995]. MacNeal Schwendler Corporation began investing their own money into the product in 1969, induced by the initial NASA investment and the potential for profits. Within five years, (1974) NASA MacNeal Schwendler Corporation began making a profit and was no longer being supported by NASA funding. This case demonstrates the utility of using induced investments as an indicator of potential commercial success.

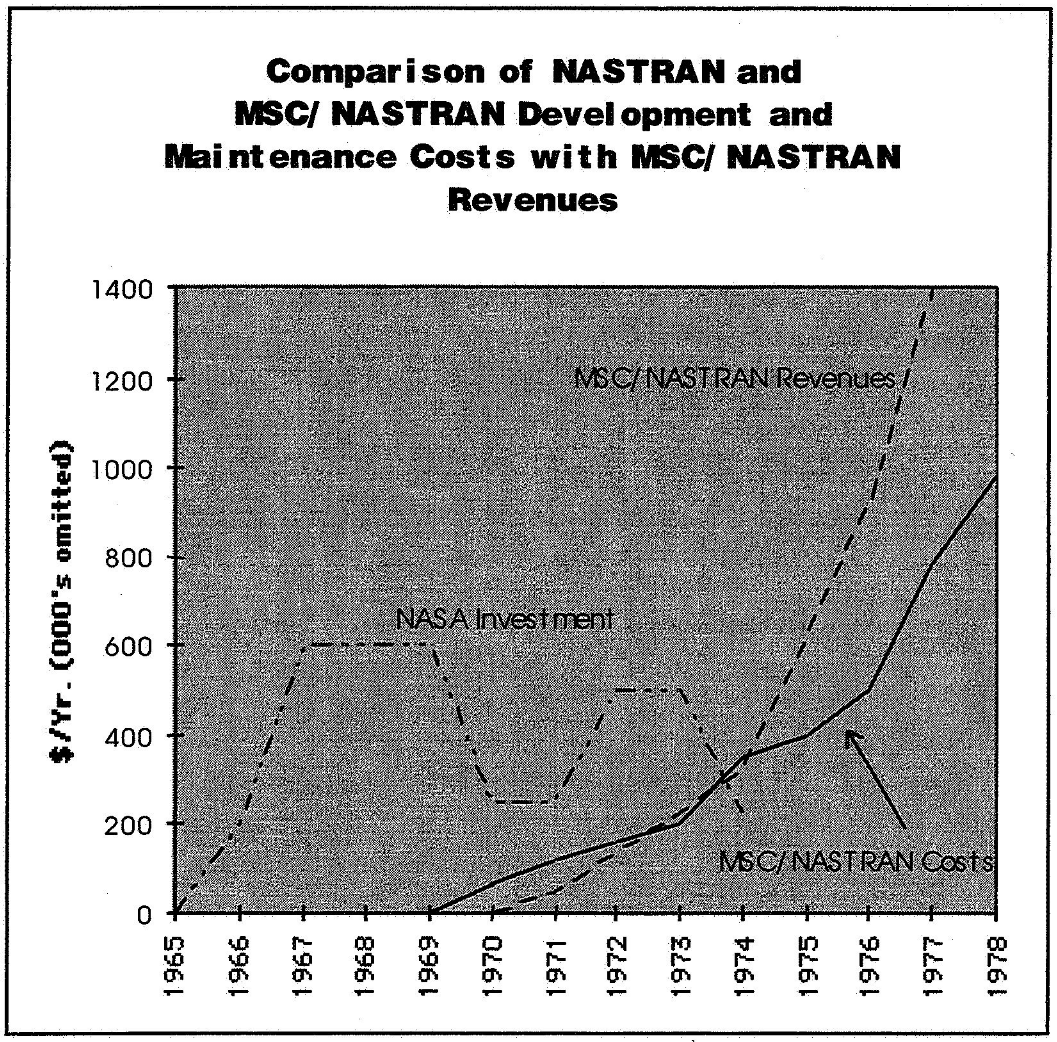

Figure 2: Induced investments in the NASTRAN case

The MacNeal Schwendler Corporations case study demonstrates the pitfall in analyzing a technology transfer program solely on quantitative intellectual property data. Intellectual property stream data is but one measure of economic impact, other worthy techniques include case studies and a portfolio-type approach. More importantly, measures must be selected based on the unique programs objectives and stakeholders interest.

In the case of NASA, there is good news concerning its future technology transfer efforts. The current approach of negotiating for reasonable sums is being challenged by the philosophy of negotiating for fair market value. Since taking a proactive stance in technology commercialization, in the four most recent years there has been an average annual increase of approximately 54% in royalties. Technology transfer projects require long periods of time to achieve commercial success, usually on the order of years. Therefore, continued measurement of the technology transfer programs must be performed in order to evaluate the programs progress.

Conclusions

In conclusion, NASA administration and other like agencies should carefully consider adopting a change in philosophy to negotiate fair market values. This potential change must be agreed upon by the stakeholders. Case studies along with quantitative analysis should be considered in evaluating the impact of the technology transfer programs. Induced investments can be an indicator of potential market impact as they are direct commitments by a private concern. Lastly, any evaluation must be performed over time in order to gauge the true progress of the program.

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Session 3
Strategies for the Exploration of Mars

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THE POLAR EXPLORATION OF MARS

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ABSTRACT

Apart from the Earth, Mars is the only planet in our Solar System to possess significant and traversable polar caps that could potentially play host to a long term program of human polar exploration. Such explorations may provide valuable information on the structure of the martian poles, both geologically and from the point of view of gaining an increased insight into volatile cycling in the martian atmosphere, past and present. Here some initial considerations are made on the nature and methods for the human exploration of the martian poles and the scientific objectives that might be prioritized for such missions. The first proposed strategies and routes for overland Mars polar expeditions, using the nature of terrestrial polar expedition attempts as a template, are also suggested.

1. INTRODUCTION

The martian polar caps are a significant feature of the martian surface and were observed by Giacomo Maraldi in 1719 (Sheehan, 1996). The obliquity of Mars (25.2°) is similar to the Earth and for that reason Mars is subject to similar seasonal variations in climate to the Earth, which are reflected in the extent and behavior of the martian polar caps. Because the poles represent one of the few geological features of Mars that undergo significant variation over relatively short time periods (from season to season) and because of their involvement in the deposition and cycling of volatiles, particularly CO2, which also has an effect on martian atmospheric pressure and winds, they represent an important target of martian exploration, both robotic and human. Through the analysis of core samples removed from polar laminations, studies of the early martian atmosphere might be undertaken.

As well as the scientific data that may be derived from the study of the martian poles, they also possess interesting potential from an exploration perspective. Polar exploration on Earth has been a significant part of human exploration achievement. Given that Mars has significant polar caps it is conceivable that the insatiable drive that humans have for exploring new frontiers will lead them to attempt polar and transpolar assaults on the martian poles as feats of exploration endurance and endeavor as well as the scientific information that may be derived from such regions. Routes for such endeavors are proposed.

2. THE MARTIAN POLAR CAPS

Similarly to the Earth, the axial inclination of Mars has created conditions that are favorable for the formation of permanent polar caps at its poles. Unlike the Earth, however, the lack of oceans or the confining restraints of continental boundaries has created a situation where these poles are able to retreat and extend in size over large areas of the martian surface to a greater degree than their terrestrial counterparts.

Mars has an orbit that is highly eccentric (0.093 compared to Earth at 0.017). The martian south pole experiences summer at perihelion (closest approach to the sun). During this summer seasonal carbon dioxide deposited during the winter at the pole is vaporized into the martian atmosphere leading to a reduction in the size of the pole.

Because of the short hot summers the reduction of size of the south pole is greater than that for the north pole and it shrinks to some 300km across. During the contraction and expansion of the polar ice caps clouds of carbon dioxide or 'polar hoods' are frequently generated that obscure the forming caps from Earth observation.

Each of the residual polar caps are also cut across by huge valleys, up to around 100 km across that swirl in a counterclockwise direction from the center of the poles outwards (Cutts, 1973, Sharp, 1973). They swirl in a direction opposite to that induced by Coriollis acceleration (Cutts, 1973, Sharp, 1973, Cutts et al, 1979), leading some workers to suggest that they are caused by the vaporization and deposition of volatiles along the escarpments, facing towards greatest solar warming (Howard, 1978) and implying a dynamic landscape of polar valleys.

These valleys also reveal the presence of layered deposits within the polar regions, horizontal strata of deposits along the edge of the valleys that because of their smoothness are believed to suggest a young age of around 100 million years (Plaut et al, 1988). They suggest a thickness for the north residual pole of around 5km and a thickness of about a fifth of this for the south residual pole (Dzurisin and Blasius, 1975).

The further study of the martian polar caps may provide us with an insight into the cycling of volatiles in the martian atmosphere and the linkage between polar behavior, wind and atmospheric behavior. Additionally, it may also provide us with some insight into the atmospheric behavior of early Mars. The study of the behavior of the martian poles, as well as providing fundamental geological knowledge about Mars and its history may also enhance predictive capabilities for martian climate that may have long term application for Mars exploration and colonization.

3. RESEARCH OBJECTIVES AND PRIORITIES FOR MARTIAN POLAR EXPLORATION

The exploration of the martian poles may be accomplished within a broader strategy for the exploration of Mars. For reasons discussed later, the exploration of the martian poles is not likely to be an initial research priority, but is likely to run on from the establishment of a permanent scientific outpost in alternative regions, with polar geology, life sciences and atmospheric study issues adding to research data being gathered in other regions.

Here some initial scientific research priorities associated with the martian poles are proposed.

3.1 Atmospheric and climatic studies

Measurements of thickness of seasonal ice deposition may be made in various regions of the poles and volatile composition of these deposits measured. Using the calculations of drop in atmospheric pressure at the time of polar cap formation, more accurate studies of the correlation between atmospheric pressure drop and seasonal $CO2$ deposition may be made, contributing to our understanding of $CO2$ cycling in the martian atmosphere. Similar calculations may be made to understand the contribution of the poles to the martian hydrologic cycle.

The formation and direction of polar winds may be studied in more detail and the contribution of temperature differentials between the polar caps and circumpolar regions to martian wind generation may be understood (Kieffer and Palluconi, 1979).

The role of these winds in both local weather patterns and global dust events may be understood, assisting in the development of predictive weather models for future martian exploration and settlement plans. The knowledge of volatile behavior and wind patterns may then be used to understand more fully the origin of the polar valleys.

Within the context of these studies more effective models may be derived concerning the relationship between the concomitant retreat and expansion of the south and north polar caps respectively at perihelion and visa versa at aphelion and the role that this process has in volatile cycling and climatic conditions.

Finally, because of the current lack of oceans on Mars and the lack of a significant biotic contribution to climatic and weather processes, Mars can be seen as a simplified 'atmospheric laboratory', offering the opportunity for understanding weather processes and the origins and contributions of particular weather systems on Mars. These may be compared to our understanding of similar patterns on the Earth and through this comparative planetological approach, our understanding of both Earth and Mars weather and climate systems may be greatly enhanced.

3.2 Studies on the past history of Mars

The polar laminations provide exciting research possibilities for studying the past history of Mars. Measurement of the thickness of strata may provide valuable information on past climatic conditions and the quantity of past polar deposition. Core drilling will provide evidence of whether older layered deposits exist beneath the visible younger deposits in the polar valleys. By comparison of ancient deposits at the south and north pole, the role of the martian 51,000 year climatic cycle (Leighton and Murray, 1966), caused by the combined effects of rotational axis and orbital precession, in determining the preferential pole at which dust laden ice is deposited can also be studied, thus providing an insight into the past linkage between Mars dust storms and polar expansion and retraction. The study of past deposition may also provide critical insights into the $CO2$ pressures on Mars in past epochs which will depend upon the total carbon dioxide available in the polar cap-atmosphere system (Fanale, 1982). The study of the patterns of laminations across the poles may also provide information on the large scale nature and size of the martian poles over time and may also provide further insights into potential polar wandering (Schultz and Lutz, 1988).

Finally, the potential contribution of changes in obliquity and eccentricity not only in the past atmospheric and polar behavior on Mars, but also in the putative generation of stable liquid water at the martian surface may be investigated (Paige and Pathare, 1994).

3.3 Future settlement

The possibility of using the poles as a source of water, both in early settlement stages (Jakosky, 1990) and for long term terraforming proposals (McKay, 1991, Clark, 1994) may be examined. In this respect the explorations of these regions would provide valuable logistical information for the exploitation of this resource and essential information on the operational logistics within these regions required for determining water mining and exploration strategies.

4. HUMAN VERSUS ROBOTIC EXPLORATION OF THE MARTIAN SURFACE

Before engaging in a discussion in methods of exploring the poles it is worth briefly addressing a question that has come to the forefront of space exploration in recent years and that is important in determine strategies for planetary exploration. That is, human versus robot planetary exploration. In this paper it is assumed that these two methods of exploration are complementary, rather than competitive and indeed each one may be used to enhance and improve the other. Despite the advances in virtual reality, computer and calculation solving capabilities, there are several facets that humans possess that make them greatly superior to the robot explorer. Some of them are listed below based on informal discussions at the fourth session of the International Space University in 1991.

1) Pattern recognition skills. Humans have the ability to recognize patterns in data that do not have any immediately obvious programmable or mathematical relationship. They are also capable of responding to unusual events and discoveries and redirecting research objectives accordingly and in real time. They have the ability to recognize information that is of paramount importance to them, but may at the time of recognition be peripheral to the immediate research objective. Additionally, humans have the ability to draw on libraries of information within their brains from vastly disparate disciplines and with the advantage of information having been collected from the world around them since birth (covering many years of assimilation). Because of this long time span of information collection and sifting, which has often been partly specialized through their education they are often programmed with highly versatile and adaptable methods of data interpretation.

2) Ability to tend to and repair equipment. This has been proven in many unforeseen EVA's on board orbital space stations and lunar explorations.

3) Using their skills as above, humans may operate telepresence stations on Mars in real time, responding directly to research questions and priorities in utilizing telepresence robots to further their research interests.

There may be other advantages of humans over robots, but the qualities mentioned above are perhaps some of the most important reasons for human use.

Aside from the merely objective analysis, there is also a valid philosophical objection to be raised to the prospect of a purely robotic approach to planetary exploration. Can advocates of purely robotic exploration say that had a robot been dropped at the summit of Everest in 1950, the 1953 attempt on the summit by humans would have been abandoned or indeed that machine dropped at the terrestrial south geographical pole in 1910 would have caused Amundsen and Scott to abandon their expeditions ? It is reasonable to suspect that it would have had little effect on these expeditions and indeed from the experience of human reaction to the photographs of martian plains taken by the Viking landers, it is valid to say that photographs and data collected by robot explorers often merely entice and tantalize the human imagination to an even greater degree and provide a greater and more exact basis on which speculation, planning and potential human exploration achievements can flourish. It is unlikely, even if machine learning and data analysis techniques progress to a significant level, that humans will ever accept obsolescence or replacement by robots. Although scientific and economic arguments may be made for a purely robotic approach to planetary exploration, this is not likely to be a reflection of the human condition now or at any stage of advanced robot and machine technology that may occur in the future.

From this point of view and the point of view of the human advantages discussed earlier, a human-robot approach to planetary exploration will be assumed and this will be the basis for the discussions on martian polar exploration.

5. STRATEGIES FOR THE POLAR EXPLORATION OF MARS

Martian polar exploration might be incorporated within the long term strategy for the robotic and human exploration of other regions of Mars (Stoker, 1989). Methods of implementing a long term polar exploration program are suggested here with a phased approach leading to the human exploration of the polar regions.

5.1 Precursor missions.

A great deal of data may be assimilated by precursor missions and may provide basic knowledge as well as information for human exploration. An essential feature of Mars polar exploration is study over a significant period of time. Many of the features of martian polar behavior that are of interest occur over seasonal time scales. For this reason rovers, both macro- and micro, have a limited use. Seasonal deposition over the polar caps can amount to several tens of centimeters and so aside from the temperature constraints it is doubtful whether such vehicles could maintain useful operational life spans and would be capable of operating with complete reliability. The most effective study of the polar regions prior to a human-robotic effort could be accomplished with the use of a polar orbiter (sun-synchronous to maintain light conditions for optical analysis). The Mars Observer would have provided much of the precursor capability required. However, a satellite with specific polar exploration objectives might be envisaged. Such a polar orbiter may also have a series of communication instruments in anticipation of communication requirements for human polar exploration and thus may act as a communication node between human polar excursions and Mars settlements.

5.2. Rationale for the establishment of a martian polar base

Following the establishment of a human presence on Mars and the construction of a permanent Mars scientific base it will be possible to undertake excursions to the polar regions. Short duration human excursions to the poles could be imagined that would allow for ice thickness measurements, core drilling in polar laminations, studies of winds and other activities. Such excursions could be accomplished through the use of a sub-orbital hopper, allowing for landing at polar regions of interest followed by EVA's of a few tens of hours and then return to the primary base. Such a hopper could be equipped with basic scientific instrumentation, core drilling devices, meteorological equipment and sample containers, allowing for basic data collection prior over a short time period.

Valuable though this approach would be, the majority of polar phenomena of interest occur over seasonal and yearly intervals. The value of Mars polar exploration would therefore be greatly enhanced by being part of a longer term program that would allow for year round observation of changes occurring at the poles.

6. A MARTIAN POLAR BASE

6.1. Timing for the construction of a martian polar base

Several reasons preclude the establishment of a permanent human outpost at the martian poles in the first stages of consolidating a human presence on Mars. A polar outpost would be a later objective. Theses reasons are :

1. Temperature and ice deposition. The low temperatures (approx. -70°C at the north polar cap and -110°C at the south polar cap in summer) (Jakosky and Martin, 1987) would make such a base difficult to operate and would increase energy requirements for

maintaining a human habitable base. Secondly, ice deposition would make such a base difficult to maintain in operational order, similarly to terrestrial polar stations, although the thin martian atmosphere would alleviate some of the wind problems associated with these conditions. Polar conditions would also make the development of closed system ecosystems required for long term martian existence difficult to sustain. For these reasons a polar station should be a temporary study base or one that is constantly resupplied from a more major martian colony. A self-sustaining independent human presence at the martian poles is an unlikely and indeed, unnecessarily complex, objective.

2. Other areas of Mars, particularly the Tharsis region, have very a great deal of interest from a geological point of view and also for resource utilization (e.g. Plescia, 1990, Cliffton, 1989). They may also provide answers to other fundamental questions on martian history such as the previous existence of water and the evolution of life on Mars in more clement conditions, particularly evidence that might be found in ancient river bed strata. For these reasons and others, initial martian scientific exploration is likely to be focused in areas other than the martian poles.

6.2. Location

Potentially any area of the poles, both north and south, could be interesting. In seeking to locate such a base the location of the primary Mars infrastructure must be considered and a base established that minimizes energy and logistical requirements in reaching it. A base that lies roughly along the same line of longitude to a primary base, creating the shortest traveling distance would be preferable. The ablation and deposition in the polar valleys that is believed to generate some movement in the valleys (Howard, 1978) may be a matter of concern in the establishment of a base at the north pole. A base established at the geographical pole itself, south or north, similarly to the Amundsen-Scott Station on the Antarctic might also be considered as this would allow for human and telepresence excursions to any region of the pole with minimum traveling distance to all regions.

6.3 Fundamental design requirements

In order to reduce cost and set-up technicalities such an outpost might be capable of supporting three astronauts at the martian surface. Many of the complexities of closed-loop systems would be avoided by resupply from a primary martian base. Orbital space stations provide a good model for a martian polar base as gases are continuously circulated, but food is resupplied, allowing for long term independent operation, but not a self-sustaining existence. A similar philosophy might be adopted for martian polar bases. Power could either be supplied by small nuclear units or has been previously suggested, by methane, which would be generated by local resource utilization (catalytic hydrogenation of CO) (Meyer and McKay, 1989, ISU, 1991) at the main base and resupplied in canister form by sub-orbital hopper or other transport methods. Other essential resources could also be generated by local resource utilization at the main base (Meyer and McKay, 1989) with hopper supply, including structural and repair materiels (Boyd et al, 1989). Water could be extracted locally and prepared by a heat exchange unit attached to the main power supply with a simple sifting device to remove polar dust deposits. Such exchange systems are already in use in terrestrial Antarctic polar bases (Cornelius, P.E., 1990) In general a base should be designed to minimize servicing and repair, minimize energy and labor input by its occupants and should be cheap, allowing for the construction of another base at an alternative site when the serviceable life has expired (which because of conditions might be expected to be low) or should be capable of easy dismantling for reconstruction in another area.

7. MARS POLAR EXPEDITIONS

PROPOSALS FOR OVERLAND ASSAULTS ON THE MARTIAN SOUTH AND NORTH GEOGRAPHICAL POLES

Science is the primary and often critical reason for exploration endeavors. But frequently exploration for its own sake, or to test human capabilities to the limits drive people to make attempts on significant frontiers. Polar exploration has a strong historical precedent on Earth and explorers such as Amundsen, Peary and Scott undertook science to underpin their principal interest - achieving assaults on the terrestrial poles. With sub-orbital hoppers and rovers, overland assaults are not explicitly required in order to reach the geographical poles of Mars or to achieve assaults on other significant frontiers such as the summit of the highest mountain in the Solar System, Olympus Mons. However, this does not remove the interest and excitement that such challenges pose to testing the limits of human and human-technological capabilities, tests that underpin our ability to challenge physical and natural obstacles in our push to explore and settle the Solar System.

Using NASA Viking 2 and Mariner 9 data, routes and strategies are suggested here for overland assaults on the martian south and north geographical poles, routes that may either be incorporated into a scientific study plan, may be used for validation of polar technologies or may similar be undertaken as a challenge to polar exploration. In selecting periods for the expedition, times when polar hoods would reduce visibility to a minimum should be avoided. Also periods when dust storms are prevalent, should be avoided.

7.1. Assault on the martian north geographical pole - 'spiral in-spiral out'

The martian north polar cap is perhaps the most challenging as its aphelion retreat is less that the summer retreat of the south polar cap and so its size creates a greater challenge even at minimum extent. It also has polar valleys that cut huge swirling swaths from the geographical pole itself out to the edge of the ice cap. An overland expedition to the pole from the edge of the ice cap either in winter or summer would therefore be most wise to use what might be referred to as a 'spiral in - spiral out' strategy for the expedition, following the polar valleys. Based on the valley profiles visualized by Viking in 1977 of the martian north polar region such a strategy would increase the expedition time by about 80 to 100%. However, crossing the polar valleys in order to attempt a shortest distance approach would not only be logistically difficult, but the need to climb and descend polar valleys would make such an expedition laborious.

It is suggested here that an expedition would be best to use the 'spiral in-spiral out' strategy following the bottom of the polar valleys. Although winds on Mars are significantly weaker because of the low atmospheric pressure, the polar valley floors may afford at least some protection from the winds. The seasonal deposits along the valley escarpments may make polar traveling difficult along these routes and this would benefit from more detailed precursor appraisal.

Figure 1 shows the basic concept for an overland transpolar assault on the martian north geographical pole. The choice of starting point may depend on the position of a polar base or a primary Mars habitat. Since the polar spiral valleys exist across the whole polar region, any direction in and out of the pole could be attempted, although avoidance of the Chasma Boreale would be wise as this creates anomalies in the directions of polar valleys.

7.2 Assault on the martian south geographical pole

The retreat of the martian south pole is extensive during perihelion and to such a degree that the martian south geographical pole is sometimes not actually covered in ice making an overland attempt from some directions more of a desert expedition than a polar one. In winter the matter is different and the pole is fully covered by the seasonal deposition of CO_2.

Whether in summer or winter, a route is proposed here for an assault on the martian south geographical pole with transpolar assault based on a previously suggested strategy (The London Times, 1992). The direction that has been chosen assumes the dropping of the team at the starting point, perhaps by sub-orbital hopper, with their equipment. The Chasma Australe is suggested as an access route to the pole, in a similar manner to large glacial valleys that are used in attempts on the terrestrial poles. It would afford some protection from martian winds and act as a guiding route to the pole. Chasma Australe also generally tends to cut across the trend of cratered terrain and therefore provides a direct access route to the pole with minimal distance traveled. Deposition at the floor of this valley may also have fewer crevasses and other dangerous irregularities that may be associated with the old terrain surrounding the remainder of the pole.

On clearing Chasma Australe there is an approximately 100km direct dash to the south geographical pole itself, thus achieving an overland assault on the martian pole.

On returning from the pole, the expedition may either retrace its tracks or, if attempting a transpolar assault, continue in a direction of approximately 60° from the pole, close to the Cavi Augusti, which offers a relatively clear exit from the pole over the residual polar ice, which may be used to achieve faster exit than over the irregular older terrain that surrounds it. Figure 2 shows the basic proposed route.

8. CONCLUSIONS

The poles of Mars offer an essential insight into volatile cycling in the martian atmosphere. Because of the lack of oceans and the relative dormancy of other geological processes on Mars, they represent an important contributor to the changing weather patterns on the planet, given their seasonal fluctuations in extent. For the long term exploration of the planet, the poles represent a potential water resource and the understanding of their behavior may enhance the possibility of developing predictive weather models for future settlements and exploration plans. They also potentially represent a window to the past recent climate of Mars through the study of polar laminations.

Mars polar exploration therefore represents a potentially significant part of a long term integrated strategy for Mars exploration, both robotic and human. Mars polar exploration may become part of a precursor data assimilation exercise and, once a scientific outpost has been established on the martian surface, a small temporary polar base can become part of long-term Mars exploration efforts.

Here strategies for Mars polar exploration were suggested and some methods of approaching them discussed. Basic research priorities and objectives at the poles were proposed with directions that such research might take. Finally, proposed overland exploration routes for expeditions making attempts on the martian south and north geographical poles were suggested based on Viking 2 and Mariner 9 data.

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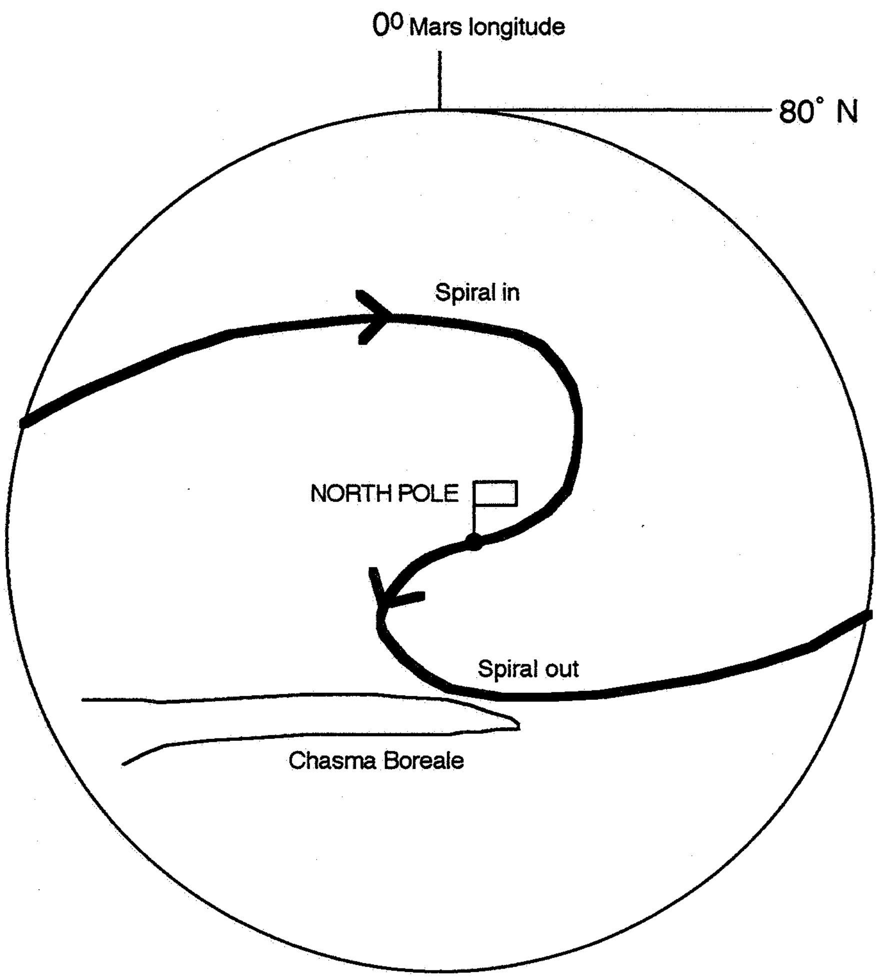

Figure 1. Proposed route for an overland and transpolar assault on the martian north geographical pole, using the 'spiral in-spiral out' expedition route

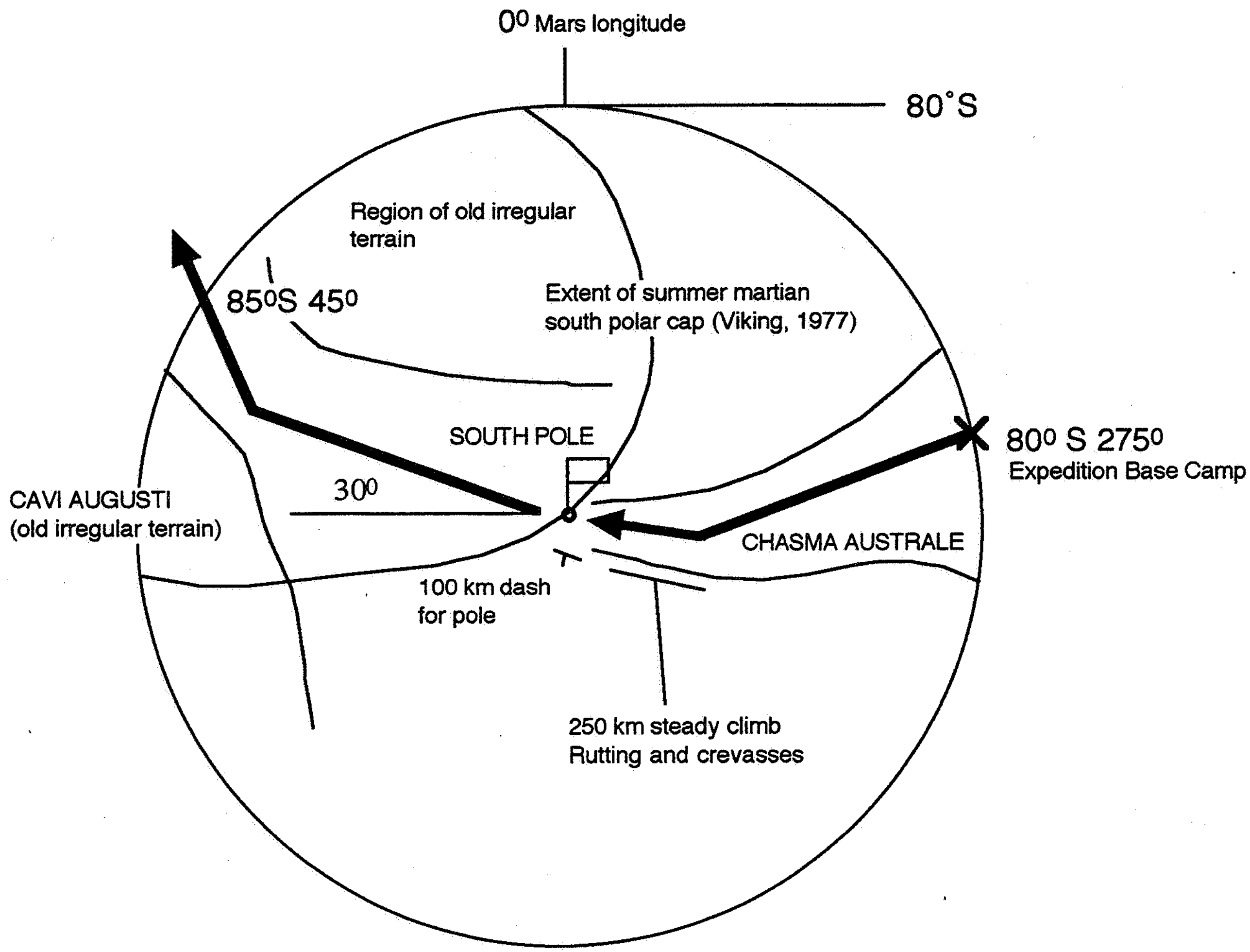

Figure 2. Proposed route for an overland and transpolar assault on the martian south geographical pole

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STRATEGY FOR THE EXOBIOLOGICAL INVESTIGATION OF THE MARTIAN SUBSURFACE

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Abstract

Since the discovery of potential biogenic origin of features on ALH84001 in 1996, the interest for exobiology has increased steadily. The way back to Mars in order to study the possibility of prebiotic or biotic evolution on our neighbour in the solar system seems more likely than ever. The present and upcoming missions (Pathfinder Mission, Global Surveyor and Mars Surveyor), offer a perfect opportunity to learn more about Mars. Together with new knowledge about extremophile lifeforms on Earth, the selection of potential landing sites can be addressed. Because of the harsh surface environment on Mars today, evidence for extinct and extant lifeforms can probably be easier identified in the martian subsurface. Subsurface drilling from a lander platform is the most advantageous way to carry out that task.

1. Rational for subsurface investigation of Mars

It is evident from the Viking data that the surface conditions of Mars are very inhospital to life at present (Klein, 1996). However, several lines of evidence (Carr, 1986; Carr, 1987) are leading to the conclusion that the climatological and surface conditions on Mars were more temperate and probably similar to Earth (3 Gys ago (McKay, 1986). At roughly about the same time, life started its evolutionary path on Earth (Schopf, 1993). The age of the carbonate inclusions of the martian meteorite ALH84001 is also determined to be about 3.6 Gys (McKay et al., 1996). Although the discussion about evidence for biogenic processes that was discovered on this SNC is not over, the assumption that they contributed to the observed features of ALH84001 is still a strong one. This is particulary true after the conflicting results about the hot (Harvey and McSween, 1996)/moderate (Kirschvink et al., 1997; Valley, 1997; Romanek et al., 1994) temperature origin of the carbonaceous inclusions being shifted more toward the moderate temperature regime, which is necessary if biology is included in the process.

As will be discussed in the next paragraph (1.1.), it is unlikely to find living organisms or the remains (especially organic carbon) of extinct species on the surface because of the rough conditions that are present there today. Therefore, if life has survived the change in the climatological history on Mars then a retreat to subsurface regions is very likely. Comparision with terrestial subsurface biological systems can help to identify prospective candidates and biota that may hold the relics of previous evolution or existing life on Mars. This comparision will be addressed in 1.2. The interest and the scientific rational for subsurface investigations is already stated in several publications dating back to the Viking era (Klein, 1979; McKay and Stocker, 1989; Ivanov, 1990; Boston et al., 1992; Klein, 1996; NASA SP-512; NASA SP-530). Finally, after the interesting and initially controversial results of the Viking metabolic experiments it is time to find the answers to the still open question of (Life on Mars).

1.1. Surface Conditions

From the Viking experiments we know the bulk composition (mainly SiO2-45(5 wt% and Fe2O3-19(3 wt%) and the elemental composition (mainly Si, Fe, Mg, Ca, S, Al) of the martian surface on the two landing sites which were separated by approximately 6000 km (Clark III et al., 1977). No organic compounds were identified within the detection limit (ppm for simple and ppb for more complex compounds (Biemann et al., 1977; Klein, 1979)). This is quite intriguing because organic compounds and organic carbon in particular are found on several different places throughout the solar system (interplanetary dust particles (Clemett et al., 1993), carbonaceous chondrites). Therefore, organic carbon should be present on the surface through influx of meteorites (Klein, 1979). There is also a way to produce organic carbon with the UV-radiation interacting with the atmosphere (Hubbard et al., 1971). As a result of these production mechanisms and the apparent lack of organic carbon on the surface, a mechanism that actively destroys organic carbon is probably at work (Klein, 1979).

In relation to that, an interesting discovery was made with the (biology package(on Viking. The experiments (pyrolytic release, gas exchange, labelled release) targeted on finding metabolic activity, identified a highly reactive surface layer - probably highly reactive oxidants (Klein, 1977). According to models this oxidation layer consists partly of hydrogen peroxide (H2O2), produced continously photochemically in the atmosphere (Hunten, 1979). To explain all the features of the Viking biological experiments, other thermally stable oxidants must be present too (H2O2 is not thermally stable). KO2 and CaO2 derived from hydrogen peroxide are possible candidates for that (Hartmann and McKay, 1995). This oxidant layer can actively destroy organic carbon and therefore eliminate an important criteria to identify biogenic activity. There is a more fundamental issue in that respect. If life was present on Mars and if it evolved in a way to form amino acids, it may be of great interest to study the chirality of these amino acids. The chirality of amino acids can be preserved under certain conditions that may have been present on Mars (Kanavaroti and Mancinelli, 1990; Bada and McDonald, 1995). On Earth, living organism are only using L-amino acids (the asymmetry comes from the asymmetry in the carbon building blocks). If life on Mars uses the same chirality as organisms on Earth, then there is maybe a connection between life on Earth and on Mars. Alternatively, a new deeper level of understanding of the biological processes is necessary. It is possible to lose the information of chirality by a process called racemization (Bada and McDonald, 1995) but the rate depends on the environmental conditions. The further decay process is followed by forming hydrocarbons (no hydrocarbons were detected at both Viking landing sites) and oxidation to kerogen and graphite (Kanavaroti and Mancinelli, 1990). Another effective way to destroy organic carbon and organic compounds is through photochemical degradation via UV-rays in the range from 190 to 300 nm (Boston et al., 1992) (highly destructive for living organisms on Earth) present on the martian surface.

Although there are some critical remarks on the metabolic experiments of Viking (like the higher temperature of the measurements compared to the ambiant martian temperature, the increased pressure level, short incubation time and uncommon temperature set points for the different experiments), the discovery of the highly reacting surface layer is probably the reason why the search for biogenic activity on the surface might be doomed to be unsuccessful from the beginning.

1.2. Potential Lifeforms

There is the possibility to find prebiotic or biotic evidence for life on Mars. Information about prebiotic evolution is not accessable on Earth because of its highly active tectonic history. On Mars, where plate tectonism is assumed to be absent (NASA SP-530), we may find the missing part of organic evolution. For the biotic part, we can asses the possibility of finding life forms under several different environmental conditions by taking the terrestrial case as an example.

There are three groups of organism we know - bacteria, archaea and eukarya. Of special interest are the so called extremophiles. These are organisms that can flourish under extreme conditions like high temperature (thermophiles/hyperthermophiles - up to 110(C, archea and bacteria), low temperature (psychrophiles - down to -12(C, eucarya and bacteria), high salt content (halophiles), high acid level (acidophiles) and high alkali levels (alkalophiles) (Madigan and Marrs, 1997). More specifically, anaerobic chemolithoautotrophes are organism that build their cell components from mineral components (Ivanov, 1990). They are totaly independent of any other kind of photosynthetic process. They can live for example from CO2 and H2 that is available in an basaltic ground water environment (Stevens and McKinley, 1995). Basalt is available on Mars, the necessary ground water can be provided by hydrothermal activity in conjunction with the permafrost layer. Evidence for even recent (in geological terms - millions of years) thermal activity has been found in the SNC meteorites. One of the SNC(s obviously cristalized in between 200 and 400 million years ago (Laul, 1986). Present hydrothermal activity may be difficult to identify. It can be subsurface or very local so that high resolution thermal mapping is required. The surrounding of hydrothermal vents can be identified by the temperature and therefore chemical gradient of the adjacent material (Walter and DesMarais, 1993). Hydrothermal vents also offer a unique opportunity to have different forms of metabolism on a well concentrated spot (Walter and DesMarais, 1993).

There is rising evidence that the common ancestor to the tree of bacteria, archae and eukarya was a sulfur matabolising chemolitoautotroph hyperthermophile (Lake, 1988). Microorganisms can be found in sulfur and sulfide bearing modern and ancient vulcanic terrestrial rocks (Ivanov, 1990). The activity of sulfur metabolising organisms can be seen macroscoplicly. The result of the metabolic process is sulfur acid which leaches the surrounding rock (Ivanov, 1990).

Investigations in Sibiria found viable microbial life (procaryotes) in the permafrost (Gilichinsky et al., 1993). The interesting result was that within the permafrost only 93 - 98% of the water is ice. The rest is liquid surrounding the microorganisms. That provides a cryoprotective layer, with the ice acting as a cryopreservative. This discovery opens the door for the possibility of finding dormant lifeforms in the permafrost layer on Mars. Because of the small percentage of liquid, they can be well preserved.

1.2.1. Extinct/Extant Life

Basic ingredient for life is the presence of water. There are a number of geomorphological features on the surface of Mars that suggest the presence of liquid water either as a river like flow with a lot of tributaries (mostly in the old terrain) or as catastrophic outflow channels (Carr, 1987). Because favourable atmospheric conditions vanished after a relatively short time (Carr, 1987), liquid water was not able to exist in reasonable amounts for most of the history of Mars (not for > 3 Gys). If life developed during the more clement period, sedimental deposites and inclusions in rocks can carry the remains of the martian evolution. But because of aeolean deposites (global dust storms) and the destructing mechanism of the surface environment, the actual search will be difficult. If life was able to conquer the subsurface threatened by the harsh climate, it is possible that the remains are better preserved or even dormant (in the permafrost). It is therefore probably easier to focus on life that lived in shallow subsurface areas where thermal activities were present. To find extant live, the detection of hydrothermal activity related to a shallow permafrost layer is probably necassary.

1.2.2. Identification of Lifeforms

There are different ways to identify biogenic activity. Macroscopic or microscopic fossils (usually in the micron range for bacteria), mineral deposites (e.g. magnetite (Lovley et al., 1987)) and the leaching of rocks (Ivanov, 1990) are some ways. Organic carbon (especially the right isotope ratio of 13C/12C (Schidlowski, 1988)) and organic molecules like hydrocarbons are strong evidence for a metabolic process too. The nitrogen or nitrogen to carbon ratio is also indicative for life (Banin and Navrot, 1979). Amino acids are maybe present either as L/D-form or in their mixed L+ D form (Bada and McDonald, 1995; Kanavaroti and Mancinelli, 1990).

Fractionation of the martian volatile inventory however, can interfere with the 13C/12C biogenic signature (Jakosky and Jones, 1997). Therefore, a clear protocol for seperating chemical/physical from biological processes has to be established. Because some of the processes can be explained by biological as well as non biological activities, the relationship of several indicators will be necessary to get an unumbigous result. Mastering this problem will be one of the main issues in non-human in situ verification.

1.3. Necessary input for landing site selection

In order to find an appropriate landing site for a subsurface investigation it is essential to characterize the different regions on Mars as precisely as possible. The Mariner and Viking missions provided a wealth of useful data (NASA SP-4212). However, the most important information - the presence of near surface water (permafrost layer) and the distribution of potential hydrothermal activity are still to be answered. But even with the limited information we already have, thanks to Mariner and Viking, areas of high interest can be specified. The Pathfinder landing side (19.5(N, 32.8(W - Ares Vallis) is at the terminus of a catastrophic outflow channel originating in chaotic terrain and carving its way through highly cratered old terrain (highlands). Evidence for sedimental deposits (on crater rimes in the old terrain and on teardrop shaped islands on the border to the lowlands) and fluvial erosion (teardrop shaped islands) are within the landing ellipse (100 x 200 km) of Pathfinder (Golombek, 1997).

Evidence of near surface water is of great interest because it offers the possibility to find existing lifeforms that are maybe dormant or to find the remains of ancient life cryopreserved in the permafrost layer. Existing hydrothermal activity together with the melting of subsurface water reservoirs in the permafrost layer can form local life bearing ecological niches. The interaction of hydrothermal or volcanic activity with subsurface water ice or permafrost is evident on several locations on Mars (Aeolis Mensae - 3(S, 220(W, Dao Vallis - 33.2(S, 266.4(W) (Squyres et al., 1987). The Tharsis bulge (resulting from volcanic activity) and the formation of the Valles Marineris (biggest canyon system on Mars) is another example of a possible hydrothermal/water interaction (McKay and Stoker, 1989). The layered sediment deposits in Valles Marineris are best explained by the presence of standing water (Nedell et al., 1987). This would again provide an environment where life can flourish. In general these events are dating back to the ancient martian history. Nevertheless, it is important to asses the likelyhood of prebiotic or biotic activity within these environments because the primary ingredient - liquid water may have been present (current estimates are from 5.9 m to 1000 m spread over the entire surface (Carr, 1986; Rassol and LeSergeant, 1977)). In general it is only feasable to conduct subsurface investigations within a few meters (maybe 3 to 10 meters). Therefore it is mandatory to concentrate on areas on Mars where there is a possibility to find shallow subsurface water reservois (permafrost). Due to sublimation these areas are limited to latitudes from about 40(towards the poles (NASA SP-530; Squyres and Carr, 1986).

The upcoming Global Surveyor Mission is equipped with the neccessary instruments (MOC-Mars Orbital Camera, Mars Orbital Laser Altimeter, Thermal Emission Spectrometer) to compose a topographic map with the related surface morphology (global 7.5 km, local 1.4 m) and mineralogy (NASA SP-530). The MVACS 98 (Mars Volatiles and Climate Surveyor) lander will concentrate on the volatile inventory of Mars using a robotic arm to get shallow subsurface samples (MSP 1998). The main analytical instrument will be the TEGA (Thermal and Evolved Gas Analyzer). The Mars Surveyor 2001 orbiter will carry a (-ray/neutron spectrometer and a MMI (Mineralogical/Morphological Investigation package) to study the global near-surface hydrogen and elemental distribution, and global mineralogy (MSP 2001). Together, these missions will enable mission planners to determine prospective landing sites better than with the information we have today. Nevertheless, a reasonable concept for the search strategy based on the discussion in 1.2. has to be implemented. Therefore, having a concept of (what to look for(together with the geomorphorphological and chemical information will provide the neccessary prerequisite for determining a landing site and for conducting a potentially successful mission.

2. Carrying out the search

During the ongoing exploraton of the solar system, several missions already investigated subsurface areas. Robotic drilling and subsequent sample return missions to the moon (Luna 16, 20, 24 (Johnson, 1979)) were successful and managed to explore the subsurface environment to a depth of 2 meters. They used rotary percussive drills. The Apollo astronauts used different equipment to characterice the lunar regolith to a depth of about 3 meters (Allton and McKay, 1990). In general they used battery powered rotary percussive drills that were operated by the astronauts. They learned from experience to use drill cores with a higher diameter in order to get undisturbed cores. The analytical procedure also changed as more experience was accumulated during the Apollo program. The Viking lander used the robotic arm to get martian soil samples from a depth of about 10 cm. Commercial (oil and gas rigs, mines) and scientific subsurface activities are widespread under the most different conditions on Earth. As a result of that, a variety of different techniques and experience has been accumulated. Therefore, a more sophisticated strategy for the exploration of the martian subsurface is not a question of insufficient technological prowess.

2.1. Comparision of applicable techniques

There are several ways to get information on the subsurface environment of Mars. One can use penetrators, penatrators combined with drills (Bilodeau et al., 1990), a rover with a drilling device or a lander with a drilling device on board. The advantage of penetrators is the simplified landing system and procedure. Penetrators however, have the disadvantage of high impact loads (700 g (Dornier, 1993)). This restricts the scientific payload. It is not possible to guarantee the survival of the penetrator with only crushing sections on the penetrator projectile itself. An alumnium honeycomb skirt around the penetrator is necessary to reduce the impact loads on the payload (Dornier, 1993). A simple penetrator is limited to penetrate the subsurface to a certain depth (usually about 1 meter (Dornier, 1993)). Although it is possible to fix sensors on the outer skin of the penetrator, sample acquisition for combined analysis with different instruments may not be possible. A penetrator combined with a driller is a very interesting concept already developed in detail (Bilodeau et al., 1990). This will guarantee flexibility in probing the subsurface to different depths and to acquire samples for comparable analysis. The proposed concept, however, does not provide the opportunity to get undisturbed fixed core profiles but only undisturbed soil transport. Verification of fossils in combination with other morphological features such as adjacent mineral deposits is therefore not possible. Drilling in general allows more flexibility. The depth profile can be characterized from zero to the maximal drill depth. The acquisition of undisturbed core samples is possible (e.g. Apollo (Alton and McKay, 1990)) and enough techniques and experience is already available in subsurface drilling on the Earth and on the Moon. A lander or rover as carrier for the drilling device offers more space for scientific equipment and the necessary sample transportation and distribution system. The question of using a rover or a lander as carrier is a difficult one. Because Mars has different features like the rims of rampart craters or the fluvial systems, local sedimental deposits and so on, it is of interest to study as many of these features as possible. The problem is that some of the features are far apart (ranging from several hundreds to thousands of kilometers) or difficult to access like the sedimental deposits on crater rims or chaotic terrain. In general it is neccessary to characterize the whole extent of geological features on Earth in order to get a complete understanding of it. This may be a problem if one considers sizeable features like the Tharsis bulge or the Valles Marineris on Mars. Both have dimensions in the range of thousands of kilometers and depths on the order of kilometers. Rover technology today is still not advanced enough to guarantee extensive autonomous operation, which would be required in the case of Mars. The microrover of the Pathfinder mission has to stay within sight of the lander and that means even in the case of an extended mission scenario, it has a maximum distance of 500 m (Golombek, 1997). It has just a small number of instruments to characterize the environment. A rover that has to go for long distances (hundreds of kilometers) or that has to handle rough terrain (access to craters or descending into valleys) has to have a certain size in order to accomodate the scientifc instruments and housekeeping activities. Just to go around for a few meters on flat terrain is appropriate for a technology demonstrator like the microrover Discovery class mission but not for a real exploratory mission. Taking together the problem of accessing difficult terrain and the extended nature of geological phenomena it is more appropriate to use several different landers instead of mobile rovers for the first step. After the initial characterization, rovers and especially human presence in combination with rovers will be neccessary to really explore our neighbour in the solar system. A lander with a drill platform offers therefore the best near term solution.

2.2. Scientifc interest

After a landing site has been selected based on near surface water content, mineralogical and elemental distribution, and geomorphological features aquired from orbiters, the on site investigation is ready to start. In order to find the remains of extinct life or the presence of extant life, the features described in 1.2. and 1.2.2. have to be addressed. An important part at the beginning is to define the distribution of reactive oxidants on the surface and in the subsurface. As mentioned in 1.1., it is unlikely to find organics within this layer. Because the penetration depth of the oxidation layer is diffusion limited (in absence of plate tectonism), the model dependent depth of this layer is up to a few meters (Bullock et al., 1994). Leaving the oxidation layer, determination of organic carbon and organic molecules like hydrocarbons will be the next step. The measurement of the 13C/12C ratio (Schidlowski, 1988) combined with 18O/16O isotope ratio which determines the carbon temperature history, as well as the carbon nitrogen ratio (Banin and Navrot, 1979) are fundamental to distinguishing physical/chemical from biological processes. Because of near surface water measurements from orbit, it should be not too difficult to find the permafrost layer. From ground zero, elements on rock surfaces and mineralogical composition of the depth profile has to be determined in order to find mineralogical deposits of metabolic processes and the distribution of biogenic elements (C, H, N, O, P, S). Optical analysis (in stereo-color) of the morphology (fossils, mineral deposites (Lovley et al., 1987), leaching because of metabolic activity (Ivanov, 1990), microbial communities, cell and subcellular structures) with and without microscope is required. The presence of extant life can be addressed by a specific metabolic experiment that is able to measure the activity of chemoautotrophic as well as photosynthetic organism (on the surface and for the first few centimeters). Measurement of volatile organics in this respect is advised. Finally, the identification of amino acids and their chirality (Bada and McDonald, 1995; Kanavaroti and Mancinelli, 1990) is recommendet.

3. Proposed concept for a Mars-Lander

The following is a proposed concept for a Mars-Lander with a focus on finding evidence for biogenic activity in shallow subsurface regions. No orbiter is included, as the commlink is direct from a high gain antenna on the lander to NASA(s Deep Space Network. The lander includes the scientific equipment, the drilling device and a sample handling and distribution system that is placed partly in the driller and partly in the lander. After a soft landing either with an active terminal descent obstacle avoidance system or the air bag style Pathfinder landing system, the lander is fixed to the ground with pyrotechniques. This may not be necessary because of the heavy lander but the fixation to the ground will provide a more stable platform for the drilling. The active terminal obstacle avoidance system is implemented because the size distribution of rocks encountered by Viking was from mm to meters (Soffen, 1977).

In the terminal descent stage, spring released sensors (a few dozens) are ejected in order to study the distribution of the oxidation layer close to the landing site (see for example Zent and McKay, 1994). These small devices have an omnidirectional beacon to report the presence and the concentration of specific oxidants to the lander. The scientific hardware to study the environment is partly mounted on the driller and partly in the lander.

3.1. Driller description

The driller consists of the front end part with the drill bit and several stem segments. The front end part with the drill bit is rigid, about 1 meter long and massive to generate the neccessary force to push the drill bit against the ground with its own weight. The following instruments are mounted in this front end part: sensors for identifying and measuring the concentration of the oxidants (together with the surface sensors, this will give the first three dimensional local concentration and identification of oxidants); a (-p-x-ray spectrometer to identify elements on the rock surfaces (Rieder et al., 1997); a (-ray/neutron spectrometer to measure the water concentration and mineralogy close to the sampling site at the respective depth (within about 0.3 m (NASA SP-530)). The oxidant sensors are fitted in small trenches that go along the helical flute of the drill stems. The (-p-x-ray spectrometer is normally retracted in the double hull front end drill stem and can be brought into contact with the soil surface (after stoping the drilling process) by means of small hydraulics. The (-ray/neutron spectrometer is positioned within the double hull and has a small window to the outside. Because it can work during the drill process, a complete 360 degrees view is achieved.
The segmented drill stems (10 - 12 individual parts, each about 50 cm in length) are detached from each other. The only lose connection is a flexible sample tube in the inner hull and wire strings in the outer hull that are fixed to the front end part. After touchdown and initiation of the drilling process, the individual parts, stored horizontally in a box during the first part of the mission, connect themselves: force is applied to the wire strings and a quarter turn of each part in respect to the contacting one engages the connection. The inner hull of the front end part and the drill stem assembly consists of a flexible plastic tube. Wires are implemented perpendicular to the tube axis within the outer skin of the tube. During the drilling, the soil sample fills the tube. After the maximum depth (about 5 meters) is reached, the drilling stops and the sample transport to the instruments and the following analytical procedure can start. For that, the tube with the sample will be transported vertically to the dissection chamber in the lander. This can be done with a conveyor belt arrangement.

3.2. The dissection chamber

The dissection chamber consists of the sample tube inlet, a sample preparation device, a small conveyor belt, an optical stereo camera, a robotic arm and the inlets to the different scientific instruments. The sample preparation device is a simple two blade structure with a central piston. The blades are perpendicular to the tube axis and parallel to the conveyor belt. The top blade cuts off the upper end of the tube and stays in this position for the rest of the mission. The bottom blade cuts the tube further down so that the piston can push a 4 cm thick slice to the conveyor belt. On the conveyor belt the sample (still engulfed by the tube material on the sides) can be preanalysed with the stereo camera and a magnifing lense mounted on it.

3.3. Lander description

The instrument package in the lander consists of the following devices: A SIMS (secondary ion mass spectrometer - with a time of flight mass spectrometer (Inglebert et al., 1995; Zschegg et al., 1992)) in order to measure the 13C/12C ratio and the equally important 18O/16O isotopic ration. An x-ray diffraction device to measure the mineralogical composition of the sample. An optical color stereo camera to investigate the morphology of the sample in combination with an optical microscope. An advanced gas analysis/calorimeter (ESA SP-1117) to identify specific trace gases (H2, H2S, CH4, SOx, NOx) (Klein, 1996) in combination with a special metabolic experiment that has to be determined. Parts of the preanalyzed sample can be selected for further analysis in one or sucessive scientific instruments. With a plasma etching device in the microscope chamber, it is possible to prepare cuts to investigate the interior of grains. The selection with the robotic arm is controlled by scientists on Earth. Sample exchange from one instrument to the others must be possible. Purging the dissection chamber and the individual instrument chambers with nitrogen or with martian atmosphere is neccessary. With a core diameter of about 4 - 5 cm and a drill depth of about 5 meters it should be possible to get in total several kg of sample material (> 10 kg) in several dozens slices. A (-ray spectrometer to identify the surface mineralogical composition close to the landing/drilling site is positioned on the lander structure.

4. Conclusion

There is a general consensus that the investigation of the martian subsurface environment is of great interest in relation to exobiology. Nevertheless, there is no mission specified yet for carrying out the search. There are three critical areas involved in this and similar mission scenarios related to exobiology: First, determine a simple, practical and effective protocol to verify the biological heritage of prebiotic or biotic evidence. Second, solve the sample handling system. Third, determine an effective metabolic experiment which includes chemoautotrophic and photosynthetic metabolism. Using a geocentric approach is only appropriate at the basic level (carbon based life and the need for liquid water).

The above described mission scenario is just a first rough idea about how to get as much information as possible with reasonable resources and hardware. Especially considering the hardware, the approach is maybe conservative. Using off the shelf parts like the SIMS, the (-p-x-ray spectrometer and a (-spectrometer however, reduces development costs and time. Some new features like the sample transportation and distribution system need some further study. The selection of the samples for further investigation by remote scientists using a built-in manipulator arm is a way to include the unique capabilities of humans into this robotic mission. The hardware for the mission was chosen in order to fullfill the requirement stated by the Viking Biology Science Team Report: (The observations chosen should complement each other in such a manner as to confirm results and minimize ambiguities.((NASA SP-4212).

The opinions expressed in this article represent the views of the author and do not represent those of ESA.

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What Does the First Manned Mars Mission Cost?

Technical and Economic Issues of a Manned Mars Mission

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Abstract: In 1953, Wernher von Braun first proved the technical feasibility of a manned Mars mission. In the following decades further investigations concerning the general feasibility of such a mission were performed in the USA as well as in Europe especially with regard to a continuation of manned space exploration after the Apollo Program. Beside a wide variety of scientific, technical, political and cultural objectives, special benefit of a manned Mars mission could be expected in the field of exobiology: After the discovery in 1996 of possible fossile life forms in Martian asteroids, some scientists believe that the complex search for life on Mars should be accomplished by astronauts respectively scientists. The contents of the paper is based on a comprehensive system analysis approach, where different manned Mars mission scenarios are investigated, compared and optimized with the in-house developed software tool FAST (Fast Assessment of Space Technologies). Possible landing sites on Mars and technical requirements, which result from manned missions to Mars of up to 1000 days are also considered. Furthermore, different Earth-Mars transfer trajectories and strategies to get into a Mars orbit, the design of manned space vehicles at subsystem level and a final life-cycle cost analysis with the calculation of development, production and operational cost are included. The investigations show that the total mass of the space vehicle at Earth departure and the total cost can be reduced by more than 50 percent if a low-energy Earth-Mars trajectory in combination with an aerocapture maneuver at Mars arrival is selected. However, the roundtrip-time of this mission scenario is about 1000 days. Assuming a project-duration of ten years, beginning with the development-phase and ending with the return of the six crew members, the annual average cost amount to about $5 billion at a total space vehicle mass in low Earth orbit of approximately 1000 metric tons. Depending on the achieved cost effectiveness in space operations at that time, the total cost of the first manned Mars mission range from $40 to $60 billion (US-Dollar buying rate of 1995). For the scientific operation on the martian surface (including the required scientific infrastructure) additional cost of 10 to 20 percent are expected. Furthermore, the life-cycle cost analysis shows that a considerable reduction of at least $10 billion can be achieved, if habitat modules derived from the planned international space station and existing main propulsion systems (slightly modified) are used.

1. BACKGROUND

The manned Mars mission (Fig. 1), whose technological feasibiltiy was first proved by Wernher von Braun in 1953 [1], was further investigated especially in the USA and partly in Europe with respect to the continuation of the manned Apollo program in the following decades [2,3].

After the manned Moon landings from 1969 to 1972, the manned space activities were exclusively limited to low Earth orbit. At the end of the eighties, the manned return to the Moon and a following manned Mars Mission was again discussed in the public. The temporary highlight in this discussion represents the speech of former US-president Bush in 1989 on the occasion of the 20th anniversary of the first manned Moon landing when he proclaimed the manned return to the Moon, the establisment of a permanently manned lunar base and a following manned Mars mission. Also in Europe at a pioneering ESA conference in Beatenberg, Switzerland in 1994, a four-phased Moon program was developed whose last phase consists of a manned return to the Moon including the build-up of a lunar base. These Moon programs are regarded as a useful step for the preperation of a manned Mars mission, because the Moon offers - as well as the international space station - unique enviromental conditions as a testbed for the development and qualification of required technologies. Because of decreased space budgets in the countries of the western world and enormous enconomic difficulties in Russia since the beginning of the nineties, it is uncertain when the ambitious plans for manned lunar and martian missions can be carried out. Especially from the estabishment of the international space station Alpha around the new millennium, it is expected that manned activities are positively stimulated and therefore also lunar and martian missions.

Furthermore, the discussion about possible simple fosile life forms on Mars increased the public interest in a manned Mars Mission. So, some scientists believe that the complex search for life on Mars should and could only be accomplished by astronauts who are respectively scientists.

Significant benefit is also expected by a manned Mars Mission as a technology driver with regard to the necessary development of propulsion systems, electrical power systems and life support systems as well as advances in automatic and robotic systems (especially with respect to required unmanned precurser missions) [5]. Beside political issues like the implementation of a manned Mars mission in international cooperation, this project is also regarded as an important cultural task of mankind with the objective to globalize the philosphy of life and thereby to solve local conflicts on our planet. In any case the manned Mars mission meets human nature, which tends to expand to new regions.

2. INTRODUCTION

In the frame of a closed system analysis approach, it is the objective of this investigation to address the following questions:

1. "What are the main scientific objectives of the first manned Mars Mission?"
2. "Which interplanetary Earth-Mars mission profile should be prefered?"
3. "What is the design of the manned spacecraft on subsystem level"
4. "What does the first Mars mission cost?"
5. "Which potential of cost reduction does exist if infrastructure systems from the international space station Alpha or from a manned lunar programm, which is established earlier, are used?"

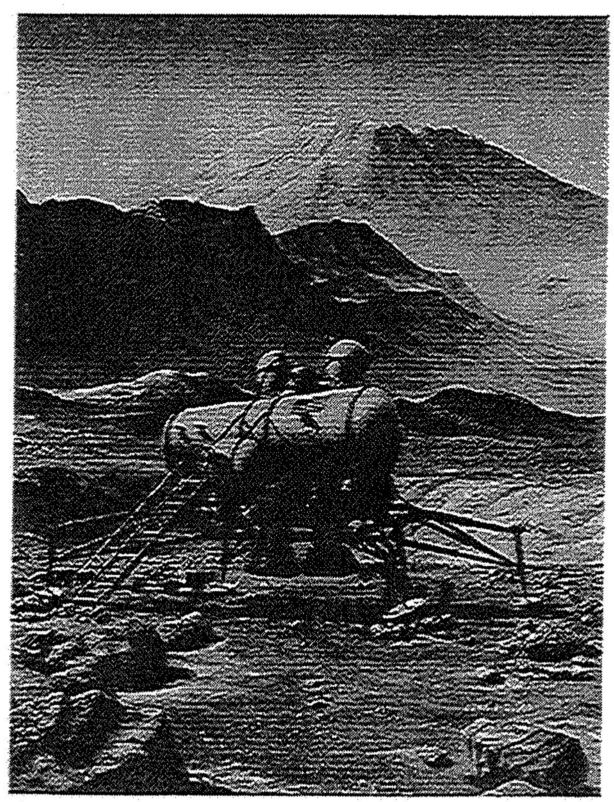

Fig. 1: The first manned Mars Mission
(Artist View [4])

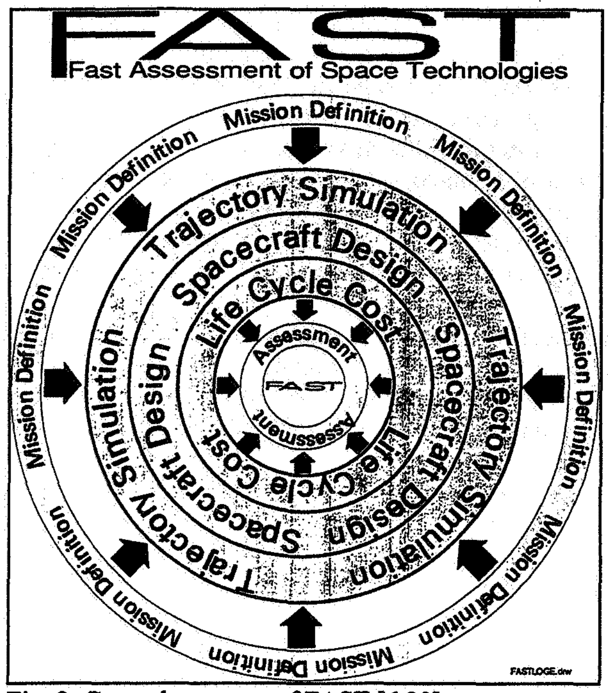

Fig. 2: General structure of FAST [6,20]

To answer the previous questions, suitable tools have to be devoloped which enable a user to investigate, compare, optimize and assess missions on a conceptional level based on a closed system analysis approch. This is accomplished by the in-house developed modular software tool FAST (Fast Assessment of Space Technologies), whose general structure is shown in fig. 2. The tool integrates all software modules of FAST to a procedure corresponding to the investigated task and, furthermore, it ensures the required data transfer beetwen the different modules by input and output data files. The principle of the analysis process is depicted in fig. 2. Based on the definition of the mission, the software tools "trajectory simulation", "spacecraft design" and "life cycle cost" are applied so that different mission concepts can be analyzed depending on the calculated results and the preference of a user. The practice shows that this is very often an iterative process that requires several loops if, for example, a mission is not feasible or upper cost limits are exceeded.

Besides the calculation of the total cost, it is another objective of this investigation to emphasize the complexity of a manned Mars mission which depends on many critical and influential parameters due to the technological requirements which result from long-term missions of up to 1000 days. The investigation ends with the identification of the mission concept that offers the lowest life cycle costs.

3. DEFINITION OF THE MISSION AND SCIENTIFIC OBJECTIVES

Since the total cost of a manned Mars mission strongly depends on the crew number, it is aimed to minimize this number as much as possible. Each qualification, however, which is necessary for survival (pilot, navigator, software and electronics engineer, mechanician, medical doctor) must be represented by at least two crew members, so that redundancy is guaranteed in case of sickness. From cost and safety aspects 6 crew members seem to be acceptable as an absolute minimum.

The mission profile chosen is similar to the lunar Apollo-strategy:

- After arrival at Mars two crew members remain in orbit while four go down to the surface. This will increase mission safety, because if unexpected problems occur with the spacecraft, which stays in orbit for up to 500 days, and will also return to Earth could be solved very flexibly by man.
- The total landed mass on Mars, is about 146 metric tons of which 25 tons are foreseen for scientific exploration (e.g. scientific laboratory and rover). The scientific payload on the return flight comprises ground samples and experiments up to one metric ton.
- The spacecraft are expendable and crew return to Earth is performed with a capsule. Contrary to the Apollo-program, however, the spacecraft are equipped with hydrogen/oxygen propulsion systems which are presently the best high-performance/high-thrust systems available.

For the scientific exploration of Mars - besides questions of exobiology, i.e. the search for extraterrestrial life forms and the investigation of the behaviour of life outside our biosphere - the topics geology, mineralogy and atmospheric research are very important. General aim is to understand planetary formation and evolution processes. Mars is the planet most similar to Earth, and the question of climatic changes - especially the disappearance of water and perhaps of atmospheric gases - is fascinating and relevant for understanding the Earth, too. Amongst others similarities with Earth comprise: the Mars day (rotation period) lasts about 24.5 hours; temperatures at the equator may reach levels above the melting point of water and there are icy clouds in the thin atmosphere. Overall, Mars is relatively cold and dry today with significant amounts of H_2O/CO_2 ice in the polar regions. Due to a tilt of the rotation axis towards the ecliptic similar to Earth, seasonal effects can be observed on Mars, of which melting or growing of the icy polar caps and global dust storms are the most spectacular ones. During winter, white frost is deposited on the surface at higher latitudes. Because of the low atmospheric pressure of less than 1% of the terrestrial one, no liquid water can exist on the surface today. Dry river beds (fig. 3), however, indicate the presence of huge amounts of water and a denser atmosphere at a time about 3 billion years ago. During this warmer and wetter period Mars was probably even more similar to Earth than today, and it seems plausible that at least in those times simple life forms could have existed on Mars. The

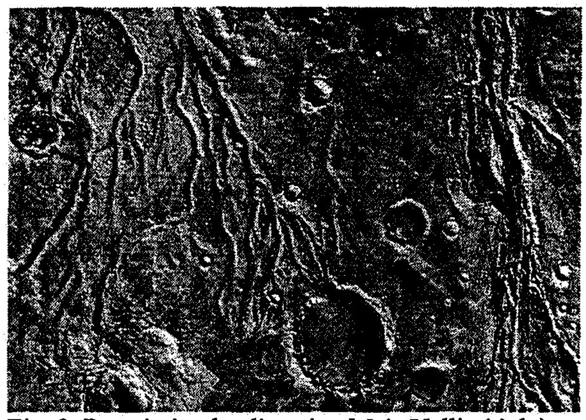

Fig. 3: Promissing landing site: Maja Vallis (right) and Vedra Vallis (left) [9]

search for morphological or chemical indications of this life is one of the primary and most exciting goals of Mars exploration. The recently published discoveries of possible fossil life forms in Mars meteorites [7] rouse optimism. Furtheron, the question whether life could exist even today perhaps in special "oases or refuges" (e.g. geological formations below the surface with favourable conditions) remains open. Impressive signs of past vulcanism are shown by the large vulcanos; Olympus Mons with a height of about 27 km is the most famous in our planetary system.

Intelligent sample selection and collection (esp. from drilling), field studies via geologic traverses and in-situ experiments (chemical or mineralogic analyses) are essential for a successful exploration. The presence of man with his cognitive, explorative, combinatoric and manipulative capabilities can help significantly. The unknown terrain and the limited possibilities to predic events together with the long time taken by signals between Earth and Mars (10 to 45 minutes bidirectional) require great flexibility and a talent for improvisation, which presently can only be provided by man. Therefore also in the opinion of NASA scientists the complex search for life on Mars should be accomplished by astronauts respectively scientists [8].

Obviously, for a cost-effective Mars exploration an appropriate mix of unmanned and manned activities must be found which supplement each other in a logical way (e.g. selection of landing site by unmanned precursor missions). In this context also the development and test of technologies for in-situ resources utilization (e.g. propellant from the CO_2 of the Mars atmosphere or from the water ice) should be taken into account which would allow future even more costeffective missions [6].

The selection of the landing site depends on a number of criteria, e.g. crew safety, possibilities for extended rover excursions and quality of expected scientific informations - to name just a few. With respect to the last mentioned criterion the question of proximity of the landing site to important geological units (vulcanos, polar caps, river beds etc.) and chance to find fossil or even existing life in suitable regions ("refuges or oases") is important. Candidates for landing sites are : areas near the rim of the northern polar cap; Olympus Rupes (southern flank of the largest vulcano Olympus Mons); Valles Marineris (large canyon-like fracture formation in the equatorial region); Maja Valles (dry river channels between Lunae Planum and Chryse Planitia north of Valles Marineris; see fig 3).

Due to the imposed long stay time on Mars of the astronauts (up to 500 days) extended excursions with a pressurized rover (100 km and more) could and should be performed, which would significantly enhance the scientific merit of the mission.

4. TRAJECTORY SIMULATION

The trajectory calculations are performed by numerical integration procedures, based on Newton's equation of motion and are added by standard analytical methods. The objective is the calculation of major parameters like the Δv-requirement, flight times and critical velocities e.g. at Earth and Mars atmosphere entry. With regard to the considered chemical oxygen/hydrogen propulsion system, impulsive orbit changes are assumed due the relative short burning times and high thrust levels. This means for the flight to Mars that firstly an impulse in low Earth orbit is performed to accelerate the spacecraft to escape velocity and secondly when arrived at Mars an impulse is necessary to decellerate the spacecraft and to get it into a martian orbit. Additional Δv-requirements, which result from gravity losses, mid-course maneuvers, changes of inclination and hovering maneuvers during the final landing phase are considered by analytical methods, which also provide the initial conditions for the numerical integration procedures.

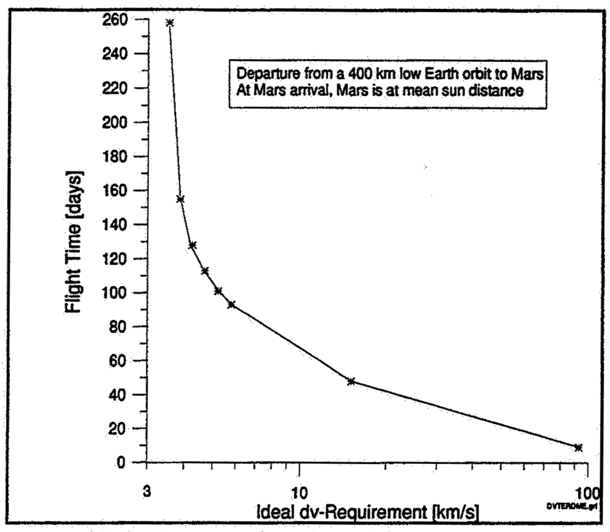

Fig. 4: Flight time as a function of the Δv-requirement for an Earth-Mars transfer trajectory [6,20]

Mars revolves around the sun in an elliptical orbit (eccentricity=0.093) whose mean values are a distance to the sun of $227.9*10^6$ km and an orbit velocity of 24.1 km/s. The eccentricity of the Mars orbit causes, that the distance from the sun varies between ±21 million kilometers with respect to the mean distance and that the orbit velocity of Mars ranges from about 22 to 26 km/s. Furthermore, it must be noted that the Mars orbit is inclined by 1.85° with respect to the ecliptic plane. Because of these circumstances, similar launch windows to Mars exist only every 24 months. This launch windows are characterized by an Earth/Mars constellation in which Earth and Mars are at the same heliocentric angle with repect to one another. The eccentricity of the Mars orbit results in different heliocentric velocities between Mars and Earth and causes strongly varying values for the Δv-requirement and flight times, depending on the selected type of transfer trajectory as shown later in the numerical calculations and that identical launch windows are repeated only roughly every 15 years.

In principle, Mars can be reached from Earth in a very short time if the spacecraft is equipped with sufficient Δv capacity and no accelleration limits have to be considered. Figure 4 shows this for the example of a departure from a 400 km low Earth orbit to Mars. Even with a departure Δv of about 3.5 km/s Mars can be reached on a hohmann similar ellipse in 260 days. If the departure Δv is only slightly increased by a few km/s, the flight time is already halved and could be reduced to less than 10 days for departure Δv's around 100 km/s. Due to the fact, that the considered spacecraft with LO_2/LH_2-propulsion systems provide only a limited Δv capability ranging from 5 to 10 km/s to achieve a reasonable

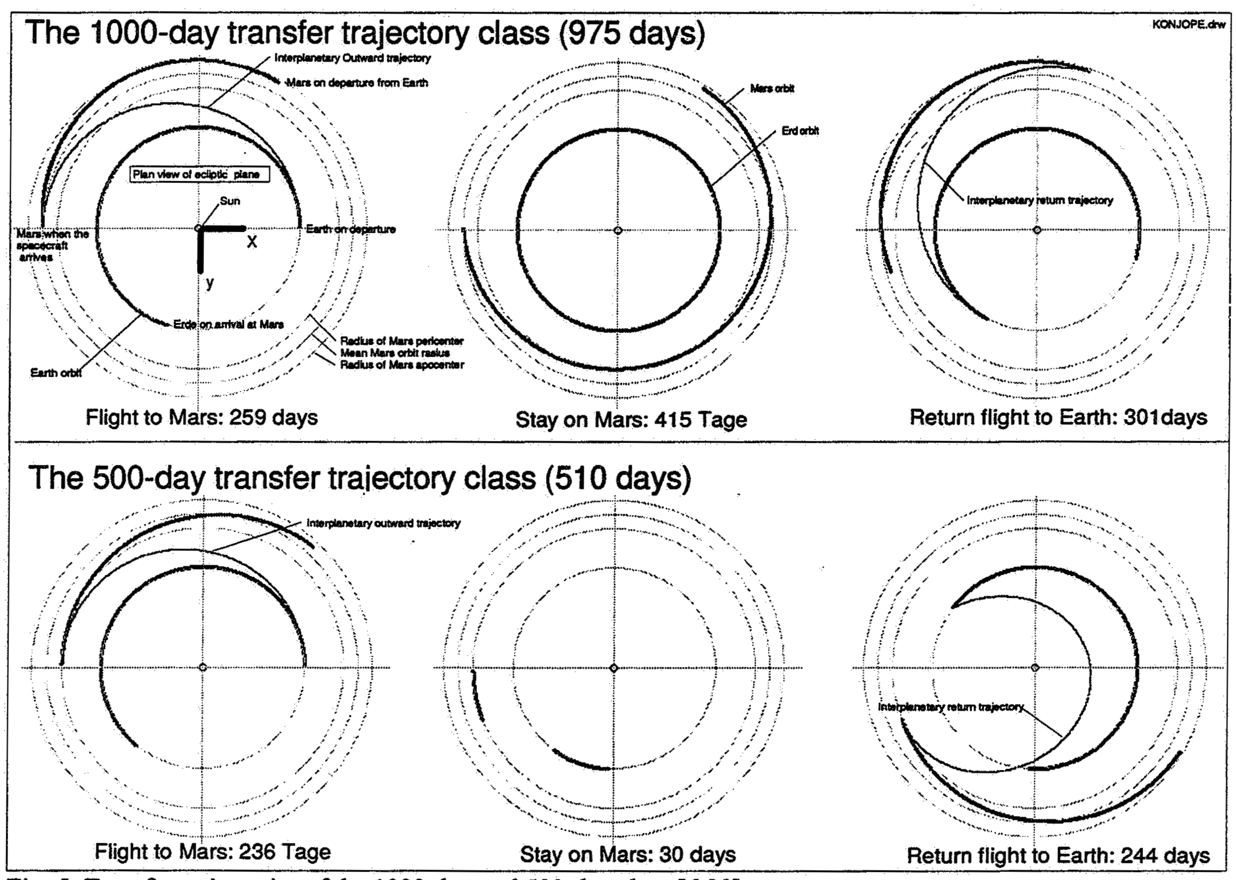

Fig. 5: Transfer trajectories of the 1000-day and 500-day class [6,20]

payload fraction and further Δv is required for the landing on and launching from Mars as well as the return to Earth, this investigation focuses on transfer trajectories below a Δv requirement of 10 km/s and transfer times of more than 100 days. This leads to the two classical types of transfer trajectories: The 1000-day low energy transfer trajectory and the fast 500-day high energy transfer trajectory.

The 1000-day trajectory class

Figure 6 shows the result of the numerical trajectory calculation for 3 Earth/Mars transfer trajectories of the 1000-day and 500-day trajectory, Mars beeing located firstly at the pericenter, then at the mean distance and finally at the papocenter (furthermore a Venus-Swing-By is depicted with data from [10]). Figure 6 shows the total ideal Δv-requirement as a function of the 3 required maneuvers TMI (Trans Mars Injection), MOI (250 km Mars Orbit Insertion) und TEI (Trans Earth Injection). The average real Δv-requirement from 30 NASA trajectories calculations [11] is illustrated for a rough

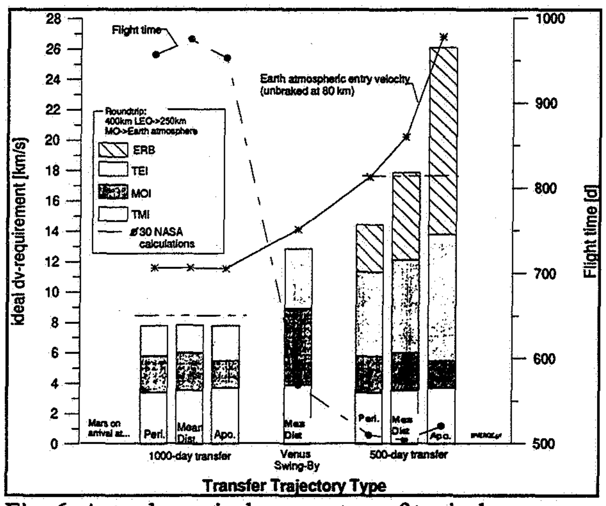

Fig. 6: Astrodynamical parameters of typical Earth/Mars transfer trajectories [6,20] (Data for Venus Swing-By from [10])

verification. The respective flight time and Earth atmospheric entry velocities are also shown. It can be seen here that the mission profiles in the 1000-day transfer represent low energy flight paths since they use the advantages of a similar Hohmann trajectory transfer, which is in general characterized by the optimum-energy transfer between 2 coplanar circular orbits. Major characteristics of the 1000-day transfer are the limiting of the outward and return flight trajectories by the Earth and Mars orbit (cf. also fig. 5), low Δv-requirement of between 8 km/s and 10 km/s, and long mission durations of around 1000 days, which result result from outward and return flights of 200 to 300 days in each case and necessary stays on Mars of 350 to 550 days. The long stay on Mars is required since it is necessary to wait for an energy-favorrable launch window to the Earth for the return flight, which permits a return trajectory similar to a Hohmann trajectory. The Δv-requirement for the different lauch windows varies only slightly because due to the long stay on Mars the outward and return flight takes place alternately, for example at a Mars pericenter or aopcenter position. A further Advantage results from the moderate atmospheric entry velocities at Mars and Earth, which are only slightly above the respective planetary escape velocities, so that the accelleartion stresses and , in particular, the thermal stresse during atmopheric entry remain within tolerable limits. There is therefore no need for any braking maneuver (ERB: Earth Brake Maneuver), which consumes propellant on arrival at the Earth.

The 500-day trajectory class

In contrast to the 1000-day trajectory the 500-day mission profiles represent high energy flight paths with an increased Δv-requirement of typically between 14 km/s and 25 km/s dar (Fig. 6). This results from the short stays on Mars (around 30 days), since the Earth has then already moved out of phase for an energy-optimum return flight similar to a Hohmann trajectory to such an extent, that it can still be reached only by using a high TEI-Δv. The fast transfer trajectory intesects in this case the earth's orbit (cf. Fig. 5) and can also cross the orbit of Venus, depending on the energy level. As a result of a fast transfer trajectory (outward flight or return flight) and short stays on Mars, the total mission duration is approximately halved, at around 500 days, in comparison to the 1000-day mission profile.

Furthermore, 500 day transfers are characterized by the disadvantage of high Δv sensitivity with regard to different Earth/Mars constellations, and very high atomospheric entry velocities for Mars and Earth. Using the example of the Earth, a braking maneuver (cf. in fig. 6: ERB=Earth Return Brake) is required before atmospheric entry inorder to reduce the entry speed to at least 14.5 km/s, because of the maximum permissible acceleration and thermal loads. Some of the Δv values which have to be applied to do this are considerable and may reach values of more than 10 km/s.

Venus-Swing-By

The transfer trajectory of a Venus swing-by uses during the Venus fly-fy the gravitational force for -depending on the mission requirement - accelerating or deccelerating the spacecraft so that a significant amount of propellant can be saved. With respect to the Δv-requirement, flight time and atmospheric entry velocities, a Venus swing-by is characterized by average values compared to the 1000-day and 500-day transfer trajectory class (Fig. 6).

Arrival/Departure Orbit at Mars
Different strategies for the arrival in and the departure from (circular low Mars orbit or high elliptical Mars orbit with/without aerocapture) allow the Δv-requirement and thus the total spacecraft mass to be reduced considerably depending on the selected transfer trajectory type. This includes:

- One alternative to the low 250 km Mars circular orbit (MO) is to enter into a 250×33800 km high elliptical Mars orbit (HEMO) with a period of 24.6 hours that is identical to the Mars rotation so that the trajectory pericenter always occurs above the same point with respect to the planet's surface. Since the mass-dominant interplanetary spacecraft does not penetrate so deeply into the gravitational potential of Mars in this case, this results overall in propellant saving, although the smaller lander and ascent stages consume slightly more propellant because of the increased Δv-requirement.
- Entry into the respective Mars orbit either using propulsion or via a Δv-saving aerocapture maneuver.

The calculations for the outbound and return flight show, that a Δv of approximately 1200 m/s can be saved, if the spacecraft orbits into a HEMO instead of a MO. If an aerocapture maneuver is applied, the Δv for both the HEMO and MO to get into an Mars orbit is reduced to 200 m/s which represents a Δv-saving of 1000 m/s respectively 2000 m/s. For both orbits, the atmosphere entry velocities at Mars are below the critical atmosphere entry velocity of 6500 m/s [12] which requires no additional propulsive brake prior atmosphere entry.

Landing on and ascent from Mars
Because of the presence of a thin Mars atmosphere, which predominantly consists of CO_2, a large proportion of the Δv-requirement for landing can be provided, almost without propulsion, via an aerobrake maneuver as it is state of the art in the Space Shuttle program today. In this case a braking maneuver is initiated in the aopcenter of the HEMO or MO which reduces the pericenter from 250 km to 50 km and thus allows an entry into the upper layers of the Mars atmosphere. By the aerobrake maneuver, the spacecarft's velocity is reduced to about 800-1000 m/s at an altitude of 5 km [12] and the selected landing point is subsequently flown by using propulsion, taking account of a hovering maneuver. The accent to the HEMO and MO is carried out exclusively by propulsion and requires a Δv of 5452 and 4229 m/s, respectively. The ascent from Mars therefore represents one of the most Δv-intensive sections. This does not include any Δv-gain from the rotation of Mars since landing sites are not specified in any more detail for the purpose of this work. The output data of the trajectory calculations represent important input data for the following spacecraft design model.

5. SPACECRAFT DESIGN MODEL

A computerbased spacecraft design model is developed, which allows the calculation of the total spacecraft masses on subsystem level including their relevant performance parameters. This means for the example of the main propulsion system that - besides the calculation of the propulsion system mass - further subsystem specific parameters like the thrust level, propellant mass flow and burning time can be calculated. The spacecraft model offers the design of unmanned and manned space vehicles above a total mass of 10 metric tons for cislunar (Earth-Moon) and interplanetary Earth-Mars missions including the required luanar and planetary landers. To perform a detailed design, the spacecraft is first of all divided into its subsystems. Based on the classical division of a spacecraft in payload, dry mass and propellant mass, fig 7 shows the structural breakdown on subsystem level.

As fig. 7 indicates, the spacecraft design model focuses on the design of subsystems which belong to the dry mass. It is noteable that the present state of the art in space technology allows the design of spacecraft, whose dry mass fraction (without payload and propellant) is

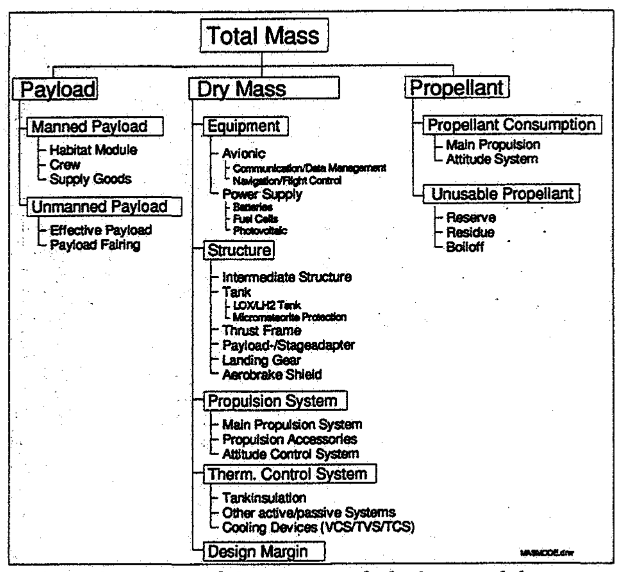

Abb. 7: Structure of the spacecraft design model at subsystem level[6,20]

only ranging from 6 to 15 %, which results in the extreme case in propellant mass fractions of up to 94% with respect to the total mass (without payload). Although the dry mass subsystem portion is generally very small, it is very often the dominant factor on the life cycle costs (particularly the development and production costs) so that an estimate which is as exact as possible is very important. To achieve an accurate total mass and cost estimation, the major susbsystems, which contribute a big portion to the dry mass and the total vehicle costs, should be modelled as accurately as possible, while inaccurate mass estimates for subsystems which represent only a small portion of the weight have only minor effects. Different analysis procederes are used for mass estimation within the spacecraft design model:

1. Methods based on empirical values
2. Analytical methods (mass estimation is carried out for known loads using rules for strength and stabilty calculations)
3. Statistcal methods (mass estimation is based on regression analysis by data from the literature)

In the case of the statistical methods, which are used most frequently, the mass estimation generally is a functional exponential expression in the form:

$$m = (F \cdot X^E + K) \cdot f \qquad \text{(Gl. 4-1)}$$

In this case, m represents the mass of the respective subsystem and X the corresponding influencing variable. The factor F and the exponent E, which together define the qualitative functional behaviour, are obtained by a regression analysis. The cosnstant K provides the capability for adding or subtracting fisxed values. A factor f (frequently also called the technology factor) also allows the substem mass to be varied lineraly. By the factor f, it is possible to simulate future technologies if, for example the use of modern materials (e.g. CFRP) leads to a mass reduction. The verification of the the spacecraft design model with data from the literature resulted in a maximum deviation of ±10% on the main subsystem level [6].

According to the "Worst-Case" design philosophy, mainly conservative assumptions are implemented in the spacecraft design model as e.g. by consideration of a 15% mass margin and generoursly calculated reserves for propellant so that an overall design on the safe side is achieved. As shown above, the spacecraft design model allows the conceptional definition and design of a space vehicles on subsystem level. This provides a powerful tool which enables an user to compare and assess different mission scenarios and, furthermore, to optimize them according to the specific preferences (e.g. minimum total mass and/or costs)

The configuration of the space transportation system in the LEO-departure orbit is depicted in fig. 8. The TMI-stage (Trans Mars Injection) accelerates the spacecraft to the interplanetary Earth-Mars transfer trajectory and is then ejected. The MTV (Mars Transfer Vehicle), which accomodates the crew during the interplanetary outward and return flight, performs at Mars arrival an aerocapture maneuver to get into a Mars orbit and later, it injects the spacecraft to the interplanetary return trajectory back to

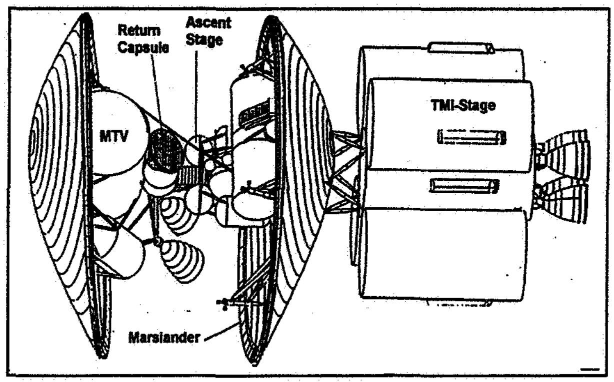

Fig. 8: The space transportation system for the first manned Mars mission in the LEO departure orbit[10]

Earth. The lander, on which the Mars ascent stage is mounted on top accomplishes the descent to the martian surface. The crew returns to the Mars orbit, where the MTV is waiting for the interplanetary return flight, by the ascent stage. The detailed investigations show, that active cooling devices are necessary to avoid boil-off of the cryogenic propellant. Furthermore a life support system with almost closed water and gas cycles is required to achieve a mass minimized spacecraft design.

For the example of the 1000-day class trajectory (cf. section trajectory simulation), fig. 9 shows the considerable potential of mass savings with respect to the spacecraft and propellant mass, if the spacecraft orbits without aerocapture in a HEMO (total spacecraft mass: 1096 metric tons) instead of a 250 km MO (total spacecraft mass: 1712 metric tons). In the case of the HEMO, the landing and ascent stages has to be sized bigger according to the increased Δv-requirements, but the mass-dominant MTV can be designed significantly smaller, because it does not enter so deep into the martian gravitational potential.

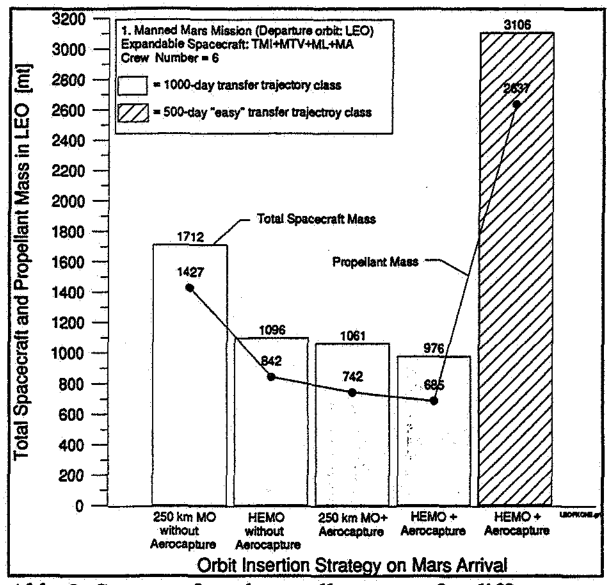

Abb. 9: Spacecraft and propellant mass for different transfer trajectory classes and orbit insertion strategies [6,20]

This results in an overall mass saving of about 35% compared to the MO without aerocapture. A similar mass saving can be achieved, if the orbit insertion in MO is supported by an aercapture maneuver, because of the significantly reduced propellant mass. The lowest total spacecraft mass of 976 metric tons can be achieved for an orbit capture in HEMO in combination with an aerocapture maneuver. The mass savings amount to about 43% compared to an orbit insertion in MO without aerocapure and is therfore selected as the reference trajectory for the further investigation. The total mass for the most favourable case (circularization in HEMO with aerocapture) of the „easy" 500-day transfer class already amounts to more than 3000 metric tons and is therefore not considered any more. Due to the fact, that the development, production and Earth-LEO transportation costs dominate with respect to the considered expandable spacecraft for a first manned Mars mission and that the corresponding costs mainly depend on the spacecraft mass, it is expected, that the total costs are at least double of that for the most favorurable 1000-day low energy transfer trajectory class.

6. LIFE CYCLE COST ANALYSIS

Space programs are - especially if they are carried out manned as in this study investigated - cost-intensive undertakings. The cost spectrum for previous space programs ranges from a few million Dollars for small experimental satellites to about 100 billion Dollars (in the value of 1995) for the manned American Apollo program. Cost aspects are more and more frequently the critical factor on wether space programs are approved, particularly since national budgets are "tight" and space budgets have stagnated or even been reduced, worlwide. The main objective of this cost model is therefore to estimate the transportation life cycle costs (LCC) for the manned Mars program investigated here. For different mission scenarios the cumulative costs can be calculated and a comparative and qualitative assessment can be caried out as a decision basis to select one alternative. In this content the word "qualitative" has particular significance. Wheras the trajectory simulation and spacecraft design model provide an accuracy of 1% and 10%, respectively (since they are based on the fundamental laws of orbital mechanics and on spacecraft subsystem design, which are already largely available) this calculation accuracy can not always be achieved within the cost model. Particulary when estimating operation costs which are accomplished in space - e.g. refuelling and launch preparation costs - in some cases the empirical values are inadequate or do not exist at all so that the corresponding cost estimation depends on the assumptions made by the respective user. The final sensitivity analysis in which cost influencing variables which are subject to relatively large uncertainties are varied over their expected and possible fluctuation range also permits relatively accurate quantiative cost estimations, however.

By a sensitivity analysis a so-called "what would happen if" analysis can be carried out. This allows the simulation of possible future developments and corresponding variables which are subject to major uncertainties and to assess their influence on, for example, the total program costs or their potential to achieve cost savings. By the sensitivity analysis the the investigation is given considerably more validity and important interactions within the complex simulation model can be clarified.

The present cost model is based on the TUBCOST cost model [13] developed under the leaderchip of H.H. Koelle at the Technical Universty of Berlin and the TRANSCOST [14] cost model of D.E Koelle at Daimler-Benz Aerospace. In addition, the cost model includes self-developed models [6] which are generally based on the regresssion analysis with data derived from literature.

The life cycle of a program and therefore the corresponding costs can be divided in the phases of developement, production and operation. The structure of the life cycle cost model, which is also able to consider missions with reusable spacecraft and the use of extraterrestrial propellants, is shown in Fig. 10 and is based on the subsystem structure of the spacecraft design model (cf. fig 7) with respect to the development and production costs. All costs are initially calculated in MY (Man Year) in order to allow the cost calculation to be carried out independently of exchange rates and inflation influences. Futhermore, the initial calculation in MY has the advantage to have a unit which can be dealt with and estimated easily particularly if working hours have to be calculated or older cost data have to be converted to present-day cost values.

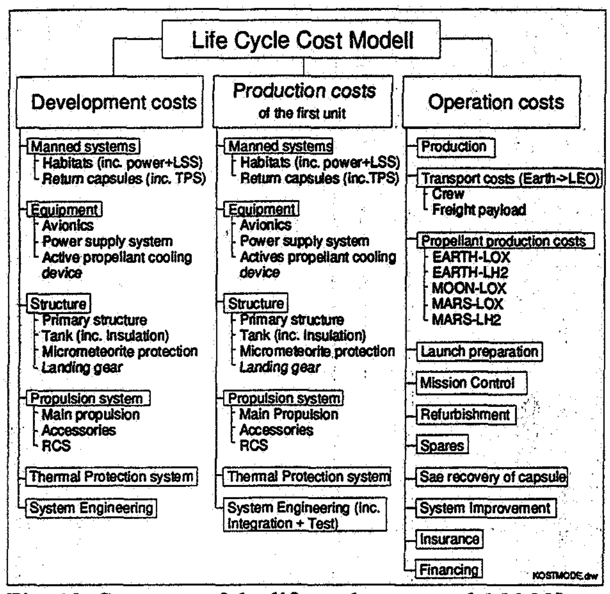

Fig. 10: Structure of the life cycle cost model [6,20]

The life cycle cost model, in which the output data from the trajectory and spacecraft model represent important input data, allows the calculation of the total transportation costs. By this cost model, special cost-relevant issues can also be detailed investigated in the course of a sensitivity analysis, e.g. the influence of different Earth to LEO transportation costs. In this investigation, inter alia, the Earth to LEO transportation costs are varied with respect to 3 differently advanced future manned Moon programs, in order to investigate their influence on the total costs of a manned Mars Mission. These 3 lunar programs are defined as:

1. A conservative cislunar scenario, where no manned return to the Moon is carried out since the Apollo Program (assumption: 10000 $/kg Earth to LEO transportation costs which represent todays costs).
2. A nominal cislunar scenario, where a small manned lunar base is operated mainly for scientific research (assumption: 1000 $/kg Earth to LEO transportation costs).
3. An optimistic cislunar scenario in which already on the Moon and in cislunar space commercial activities are carried out (e.g. He-3-export for the future "clean" terrestrial fusion technology or the production of structures on the lunar surface for solar power satellites to support our planet with energy (assumption: 100 $/kg Earth to LEO transportation costs).

With respect to these 3 cislunar scenarios, different cost-effectivities can be expected in space programs. In the case of the optimistic scenario, significant Earth-LEO traportation costs reductions on the order of 2 magnitudes (down to 100 $/kg) will be possible from which also the first manned Mars mission will profit. This for the future possible cost reduction may not be neglected because at present transportation costs of 10000$/kg only the transportation costs for the manned spacecraft (total mass: 1000 mt) amounts to $10 billion. Further costs reductions can be achieved with regard to developement, production and operation costs, if the gained experience, e.g. in the optimistic cislunar scenario, is transfered to the manned Mars program.

Figure 11 shows the life cycle costs - divided into development, production and operation costs - for the first manned Mars mission depending on the 3 differently advanced cislunar scenarios. The development and production costs are further devided into the respective spacecraft stage TMI, MTV, ML, and MA (beginning at the bottom). For the conservative scenario, which represents today's cost, the total cost for the manned Mars mission amount to $58.3 billion. By an improvement of the cost-effectiveness in the nominal and optimistic cislunar scenario, which mainly results from decreased operation costs, total costs of $44.4

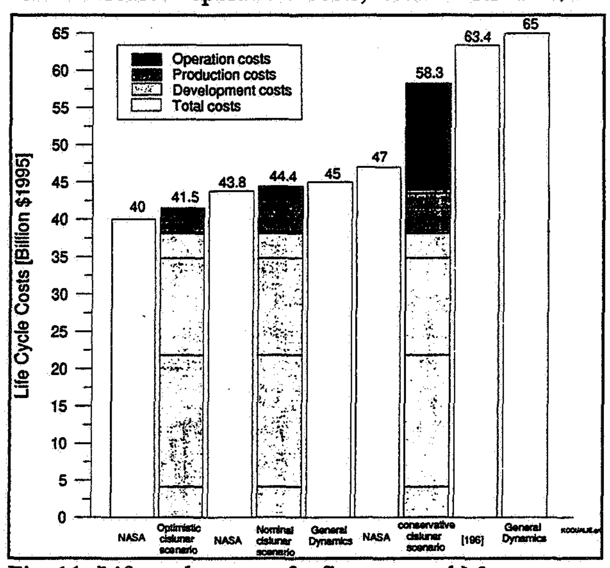

Fig. 11: Life cycle costs of a first manned Mars mission [6,15,16,17,18,19,20]

respectively $41.5 billion can be achieved which represents a cost saving of 20-30% compared to the conservative scenario. Thus it can be shown, that an improvement of the cost-effectiveness - especially by the initating of a more or less advanced manned lunar program - results in considerable cost advantages for the first manned Mars mission. For the operation of the scientific research on Mars (including development and production of the required scientific systems) which is not considered in this investigation. additional costs of 10-20% are expected.

It is remarkable, that only the development costs for the two spacecraft stages MTV und ML amount to about $30 billion which is more than 50% of the toal costs. The reason for that are very expensive manned habitat modules for accomodating the crew. So ensures the habitat module of the MTV the accommodation of up to 6 crew members for almost 1000 days as well as the ML, which accommodates 4 crew members during the long 415-day stay on the martian surface. With regard to accomodation requirements, both habitat modules are comparable to the planed international space station. This indicates, that a considerable cost saving potential does exist, if for the first manned Mars mission slightly modified habitat modules are used which are derived from the international space station. This cost saving philosophy can also be applied for the very expensive main propulsion systems, if slightly modified SSMEs (Space Shuttle Main Engine), RL-10 or the European HM7- respectively HM60-engines are used. With the assumption, that 50% of the corresponding developement costs can be saved by using modified habitat modules and propulsion systems, a cost reduction of about $10 billion can be realized for the first manned Mars mission. In order to compare and for a rough verification, fig 11 shows also the total costs of first Mars missions of feasibilty studies from NASA and American space companies [15,16,17,18, 19]. The differences in total costs are mainly caused by different assumptions and mission concepts. In the previous mentioned studies operation costs are partly neglected and different interplanetary trajectory classes, technologies, propulsion systems, Earth-LEO transportation costs und number of crew members are considered. Nevertheless, both the magnitude of the total costs and the cost tendency show a good accordance with cost values, calcuclated in this investigation.

7. SUMMARY

Beside a wide variety of scientific, technological, political and cultural objectives, the first manned Mars mission might provide considerable benefit in the area of exobiology, which means the search for extraterrestrial life forms.

For the first Mars mission with a crew of six and a four-staged expandable spacecraft which is injected from a low Earth orbit to Mars it is shown that by an aerocapture maneuver at Mars arrival to insert the spacecraft in an high elliptical orbit the total spacecraft mass can be reduced by 40% compared to a low circular orbit without aercapture. The selection of a hohmann similar low energy interplanetary trajectory on the one hand results in doubled round-trip times of 1000 days but on the other hand significantly reduces the total spacecraft mass by at least the factor 3, compared to a high energy 500-day transfer trajectroy class. For this most favorable case, the total spacecraft mass amounts to 976 mt.

The total costs range from about $40 to $ 60 billion depending on the cost-effectiveness achieved at that time in the space sector, e.g. by the initiating of a manned Moon Program. For the operation of the scientific research on the martian surface (including the development and production of the required scientific systems) additional costs of about 10-20% are expected.

A further considerable cost saving potential in the order of $10 billion does exist, if modified Habitat modules of the international space station for accomodation of the crew and slightly modified existing main propulsion systems can be used. Based on a 10-year program for the first manned Mars mission, which begins with the development phase and ends with the return of the crew, the average costs per year amount to $5 billion, which represents about 15% of the today's civil space budgets worldwide.

Thus, the manned Mars mission seems affordable within the first decades of the next century, especially if it is carried out in international cooperation with a corresponding distribution of the expenditures.

If the first manned Mars mission among other scientific reserach activities, discovers fossile life forms on Mars - maybe even still active life forms - this monetary effort seems acceptable, because this would answer one of the fundamental questions of humankind "Are we alone in space?".

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PHOTOVOLTAIC POWER FOR MARS EXPLORATION

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Mars is a challenging environment for the use of solar power. The implications of the low temperatures and low light intensity, solar spectrum modified by dust and changing with time of day and year, indirect sunlight, dust storms, deposited dust, wind, and corrosive peroxide-rich soil are discussed with respect to potential photovoltaic power systems. The power systems addressed include a solar-powered rover vehicle and a human base. High transportation costs dictate high efficiency solar cells or alternatively, a "thin film" solar cell deposited on a lightweight plastic or thin metal foil.

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Session 5
Systems Architecture & Mission Design

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P. 16

BOLAS: A CANADIAN - US IONOSPHERIC TETHER MISSION

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Abstract

Everyday, international broadcasters, ships, and aircraft use a naturally conducting atmospheric layer, the ionosphere, to reflect communications signals over the Earth's horizon. A better understanding of this layer, with its irregularities, instabilities, and dynamics, would improve communications transmission and reception. This atmospheric layer is also a lens that can distort signal transmissions from communications, navigation, and surveillance satellites. The ionosphere over Canada and other high latitude countries can carry large currents and is particularly dynamic, so that a scientific understanding of this layer is critical. The BOLAS mission would characterize reflective and transmissive properties of the ionosphere by flying two satellites, each with identical HF receivers, dipole antennas, particle probes, and GPS receivers. The satellites would be connected by a non-conducting tether to maintain a 100 m separation, and would cartwheel in the orbit plane to spatially survey the ionosphere. The six-month mission would fly in a high inclination, 350 x 600 km orbit, and would be active during passes over the auroral region of Canada. This paper discusses the system requirements and architecture, spacecraft and operations concepts, and mission design, as well as team organization, international cooperation and the scientific and technological benefits that are expected.

INTRODUCTION

Bistatic **O**bservations using **L**ow **A**ltitude **S**atellites (BOLAS) is a proposed space science experiment to exploit a unique set of capabilities from recent advances in tether and microsat technology. A multi-disciplinary Canada-US team with interests in space plasmas and miscrosats/tethers proposes a scientific experiment implemented with low-cost spacecraft, comprising two payload packages that are separated by a 100 m tether and in a bolas (cartwheel) rotation in low earth orbit [1]. The spacecraft would be launched as a secondary payload on the Delta Launch vehicle.

The objectives in basic space science would be to improve the understanding of two classes of ionospheric dynamic processes that redistribute plasma energy in its flow from the sun to the low atmosphere. We focus on the class of the fluid processes around the peak of the ionospheric F region that give rise to large-scale density irregularities, such as the gradient-drift instability. Spaceborne BOLAS radio instrumentation would view these irregularities and allow us to see density structures from a new perspective. The other class of processes is in the realm of microscale plasma instabilities. The simultaneous observation of thermal and suprathermal particles and concomitant waves would lead to improved models of the formation of ion conics, cavitons and other phenomena that must be part of the transport phenomena that control energy and mass flux in the collisionless topside auroral ionosphere. The electron density distribution would also be measured with tomography, using transmissions from GPS satellites to GPS receivers aboard BOLAS. With reference to the CSA's "Magog Manifesto," the significance of the anticipated new science would arise from its contribution to improved models of the magnetosphere-ionosphere.

Our approach would be to coordinate the operation of the two-point (bistatic) payload with ground radio-science and other facilities to yield space perspectives on auroral density structures hitherto only observed on the ground. The primary facility for radio-science objectives would be phase coherent receivers on both ends of the tether for measuring the direction of arrival, signal delay and other parameters of the transionospheric waves. Particle detectors on both ends of the 100-m tether would be associated with the receivers in the study of spontaneous auroral processes whose spatial extent approximates the tether length. The bolas rotational motion of the ensemble would allow the double probe to investigate the dependence of measured parameters on the direction with respect to the local magnetic field .

BOLAS would employ established sounding rocket and tether technology as a base, but also would turn the page to new chapters, particularly in tethers, microsats and the use of the Global Positioning System (GPS) in orbit. The CSA Space Technology Program and the NASA Marshall Space Flight Center propose to be contributing partners in the project, with the CSA Space Science Program (SSP). These organizations would use and extend expertise in tether technology and in Secondary Payload integration acquired in the SEDS, PMG and TIPS orbital missions and in the Observations of Electricfield Distribution in Ionospheric Plasmas a Unique Strategy (OEDIPUS) suborbital flights. The first-time application of GPS technology to instrument synchronization and to differential determination of the inertial direction of the tether would be new applications of this technology, which also would supply spacecraft ephemeris. Proposed technology demonstrations of the mission have significance to future Canadian microsat and smallsat missions, and to future space station-related and interplanetary missions of NASA.

NASA is interested in the BOLAS configuration to study the long term orbital stability of large spin-stabilized structures, for both artificial gravity applications and for LEO-to-GEO tether transportation systems. The configuration also has interest to the NASA science community for in-situ atmospheric and ionospheric measurements.

The BOLAS concept has been conceived in response to the Announcement of Opportunity in July 1996 by the SSP for small payload experiments. The proposing team is drawn from various agencies interested in the above topics. We think that we have achieved a balanced mixture of institutional experience in radio science, auroral physics, the space dynamics of tethers, space technology development and space operations. This mixture is the key to achieving an exciting synergism within the tight financial constraints of the participating agencies.

MISSION OBJECTIVES

Science Overview

The BOLAS experiment is a novel approach to improved understanding of the high-latitude ionosphere. It utilizes two payloads separated in space by about 100 m to focus on two major areas of current research: (1) density irregularities that affect radio waves, and (2) small-scale instabilities, as well as ionospheric density distribution measurements, all linked to the dynamics of the auroral plasma. Although other multiple-satellite missions are being operated or sought abroad, BOLAS would occupy a special niche by virtue of its small payload separation and its relatively low altitude at and just above the ionosphere-magnetosphere interface.

The scientific objectives of BOLAS are to:

(1) Investigate ionospheric density irregularities that affect radio wave transmission, using an in-space two-element direction finding array coordinated with ground transmissions from Super Dual Auroral Radar Network (SuperDARN) and Canadian Auroral Digital Ionosondes (CADI) ground sites.

(2) Investigate kinetic instabilities of the auroral plasma involving low-energy ions and electrons using field and particle probes separated by about 100 m.

(3) Measure the two-dimensional electron density distribution in the ionospheric space between a GPS spacecraft and a BOLAS GPS receiver, to provide the basis for improved global density models.

The BOLAS concept takes advantage of opportunities for novel investigations that can be carried out with small payloads. First, we propose to use a space tether as the basis for a two-element direction-finding array, working in concert with prearranged, collaborating transmitters on the ground. Second, we want to apply the tethered satellite pair for bistatic studies of the generation and propagation of spontaneous electromagnetic (EM) waves. The 100 m tether length represents a probe scale size that is intermediate between previously attained separations.

Arguments in favour of coordinated ground and space observations of the ionosphere-magnetosphere have had currency since spacecraft explorations began. Ground radars operate through finite time intervals to produce integrated images of the spatial distribution of various parameters. Spacecraft move relatively quickly through part of the radar coverage yielding snapshots of the same parameters. Brought together, these two data sets permit us to understand the complete spatial-temporal behaviour of atmospheric dynamics.

Admittedly there are objectives adequately addressed with ground facilities alone. Data from incoherent backscatter, coherent HF backscatter and ground ionosondes when compared yield consistent measurements of certain quantities, for instance the drift velocity of the convecting ionospheric plasma. These parameters tend to be of the bulk-parameter or large-scale variety. However, other scientific objectives unavoidably require in-situ, space observations. These include micro-scale observations of plasma processes in general, and, in the context of electromagnetic (EM) wave spectrum, observations of wave parameters which simply are not accessible from the ground. Objectives (1) and (2) exploit the potential of a tethered payload for these two kinds of in-situ observations. Objective (3) is a unique and novel method for tomography of the ionosphere, which would be done with orbiting GPS receivers that are necessary for Objectives (1) and (2).

Science Objectives

Ionospheric density irregularities that scatter transionospheric EM waves are the signature of fluid instabilities that are constantly at work at auroral latitudes around the magnetosphere-ionosphere interface. Micro-physical wave-particle interactions play important intermediary roles in the flow of energy across the same interface. Both of these lines of research respond to the "Space Weather" priorities of the SSP's 1994 Magog Manifesto.

The International Solar Terrestrial Program (ISTP) has been organized to provide a global view of processes that determine the flow of mass and energy through geospace. An important input to this program is the data from ground-based observatory networks. Canadian and U.S. scientists are engaged in the SuperDARN, CADI, and Canadian Auroral Network for OPEN Unified Study (CANOPUS) facilities as parts of national contributions to the ISTP. The facilities support near space research through observatory modes of operation. The interpretation of these observatory data is based on physical notions, such as coherent backscatter or ionospheric reflection. The ground data have a particular perspective, and therefore have limitations. Our proposed SuperDARN and CADI collaborations seek to overcome limitations of their ground data by combining them with BOLAS data, as already described.

One objective of this proposal is to gain improved information about the three-dimensional shape and spatial distribution of high-latitude density structures. We believe that this will lead to better understanding of the plasma instabilities that created them. Phenomena such as nightside blobs or polar patches may be observable in different phases in this mission. We acknowledge that incoherent scatter radar are able to map density structures in two dimensions [2], but little has been done so far in comparing those measurements with observations from overlapping coherent backscatter radars. The collocated Saskatoon and Kapuskasing radar zones, the CADIs and the CANOPUS network constitute a unique target of opportunity for comparative studies

> **Questions**: What can the simultaneous SuperDARN-BOLAS or CADI-BOLAS data tell us about the shapes of density enhancements that are routinely observed on the ground? What is the distribution of HF scatterers within large-scale density structures like blobs or patches? What is the three-dimensional scattering cross section for coherent scatter hitherto usually seen only in the backward direction? Are there classes of structures that have been invisible hitherto under the backscatter geometry requirement?

The heating and upward transport of ions at auroral latitudes has become a space physics question of considerable interest. Theories for the transverse acceleration of ions to form ion conics have called principally on electrostatic ion cyclotron wave heating or acceleration by lower hybrid waves [3]. Theoretical and simulation research on these concepts have typically involved setting up models for interaction regions, and then checking to see if temporal and spatial scales predicted agree with observations. These models include such features as free-energy sources in energetic electrons and density cavities. The scales predicted for cyclotron and lower-hybrid instabilities are different.

> **Questions:** Is there one dominant ion heating mechanism in the topside ionosphere? What are the characteristic spatial scales for the interaction, parallel and perpendicular to the magnetic field? What is the source of the driving waves? Are suprathermal electrons and/or ions the drivers?

All three BOLAS Objectives correspond to highly active topics of research internationally, and through the type of instrumentation planned for BOLAS, this mission can make unique and meaningful contributions to the overall international effort.

Science Measurements and Specifications

The primary radio observations would be of waves with signal levels between 1 and at least 1000 mV m^{-1}. The frequency range would be 100 Hz to 20 MHz. Important observables would be amplitude to ±10% and signal delay to ±20 ms. The measurement of the DOA angle to ±5° is stated as a special target that would depend on the development of the differential phase measurement. The received signals would need to be sampled at the tether extremities with less than 10 ns differential error in the clocking rates.

Electron and ion fluxes would be required at energies between 0.1 and 50 eV. The detectors should give good angular (10° pitch angle by 40° azimuth bins) and energy (10%) resolution at high time resolution, the goal being to measure drift energy, direction, temperature, and density of the core population every 1-10 ms. Both swept-energy and fixed-energy modes would be required for obtaining the overall energetics and spatial resolution of small structures, respectively. A complete energy-pitch angle distribution should be determined within one second. A mode of operation interleaving fixed- and stepped-energy measurements would be required.

The BOLAS experiments would take place in the auroral oval and its neighbourhood. It is proposed to conduct primary BOLAS operations when the spacecraft flies near the centre of the overlapped coverage areas of the SuperDARN radars at Kapuskasing and Saskatoon, shown in Figure 1. The large fan-shaped areas shown in the figure are the coverage of the SuperDARN radars at 350 km altitude where each fan shape is subdivided azimuthally into 16 individual beams. The overlapping coverage permits the vector velocity of each beam-range cell to be determined. At some points during the orbital passes, at least near the beginning and end, the radars would produce area-wide maps of the distribution of back scatter. At other times, the selection of a beam or of several beams could be tailored to the satellite earth track across each wedge. Additionally, primary BOLAS operations would be conducted when flying over the CADI ionosondes. The circles in Figure 1 are the coverage areas at 350 km altitude of the CADI ionosondes presently operating. The sites at Alert and Eureka are of lower priority than the other more southern CADI sites. Coordinated operations would also be conducted at the CADI sites that are in view of the spacecraft during an orbital pass. Whether for SuperDARN or CADI collaboration, in either case the decision about spacecraft operating mode would follow a prediction of the orbital path and what can be predicted about the state of ionosphere in the sector.

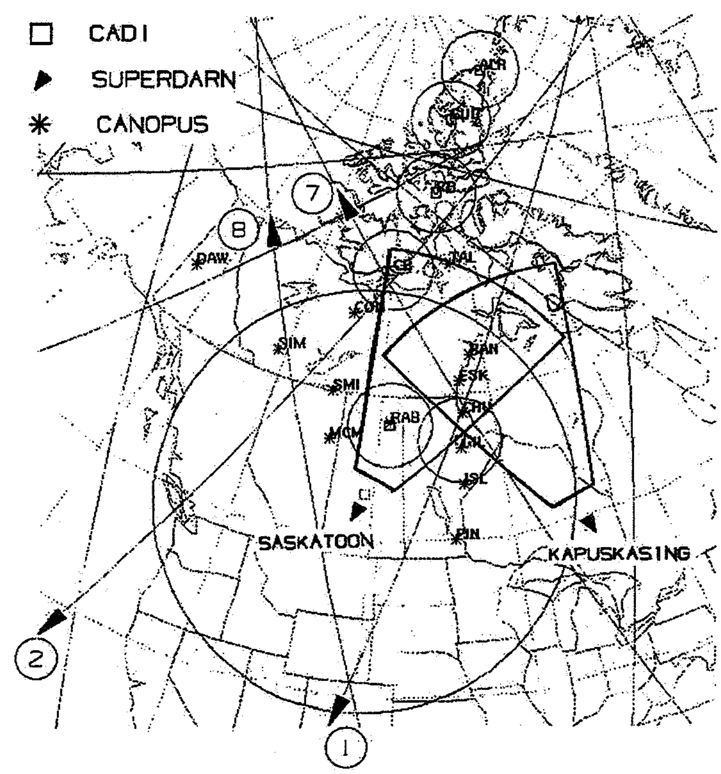

Figure 1 Ground tracks of typical ascending and descending node BOLAS passes over the centre of the region of collaborating ground facilities

Table 1 Summary of the Science Measurements and Specifications

Measurement Planned	Method and Accuracy	Comments
Radio observations	5 m tip-to-tip (min) dipole antenna on each satellite; align dipoles along tether direction to ±10°; 1 - 1000 μV/m signals; 100 Hz - 20 MHz; 10 ns synch. of receivers; ±5° DOA knowledge	The principal observables would be signal level amplitudes to 10% and the signal delay to ±20 μs. The two sets of dipole antennas, one on each subsatellite, will allow for direction of arrival (DOA) measurements of an incoming signal. The dipole antennas on each subsatellite used for the radio observations should be aligned along the tether line.
	Region of interest over SuperDARN & CADI sites; 2-3 passes per day; Measurements in both dayside and nightside auroral ovals is required	The primary science will be conducted when passing through the region covered by the SuperDARN radars and the CADI ionosondes shown in Figure 1. The CANOPUS network will also provide information on the state of the ionosphere before and during a pass. During interesting ionospheric activity, 2-3 orbital passes will be required per day for a typical duration of 3 days. At other times a lower frequency of passes is possible to allow for other operational modes. Flexibity in the science operations should be maintained to operate in other regions relative to the earth.
Electron & ion measurements	0.1 - 50 eV 10% energy resolution (approx) 10° x 10° angular resolution 1-10 ms sampling time	Goal is to measure drift energy, direction, temperature and density of the core electron and ion population. Both swept energy modes and fixed energy modes are required, and a complete energy pitch angle distribution is required every second. Measurements to be done while radio observations are being made
Tomography using a GPS receiver	L1 & L2 carrier phase precision of 0.2mm; selection of GPS satellites	The GPS tomography studies require the use of the L1 and L2 frequencies, and the ability of the GPS receiver to select specific GPS satellites that are near the horizon to get maximum travel of the GPS signal through the ionosphere.

The ionospheric occultation investigation requires that the GPS receivers process the relative phase of the L1 and L2 frequencies. The GPS units must have software control that allows selection of the relevant spacecraft near the earth horizon. A summary of the science measurements and specifications is provided in Table 1.

Technology Demonstration Objectives

The science objectives described above require measurement locations in space that are separated by at least 100 m. The requirement may be met using emerging tether technology that has been developed in the USA and Canada, and low-cost sub-satellite technology that derives in part from sub-orbital rocket missions.

Tether technology, GPS applications, and low-cost design of satellites are at the forefront of space technology R&D. Elements of BOLAS have potential for future application to solar-terrestrial space science missions, to International Space Station, to future space stations requiring artificial gravity, and to several missions involving GPS in orbit. The technology demonstration objectives include:

- The controlled deployment and spin stabilization with a SEDS deployer in the BOLAS range of tether length (about 100 - 400 m).
- The operation of a Canadian tether retriever, to achieve spin-up.

- The short and long-term passive stabilization of BOLAS, in particular stable orientation of the end bodies, non-decaying spin rate, and predictable motion relative to the orbit plane.
- The survivability of a non-conducting 2 mm tether of Spectra 1000 material, in the orbital debris and atomic oxygen environment for six months and beyond.
- Determination of orientation and position of the large rotating configuration, using GPS in-orbit receivers and differential GPS ground processing.
- Operation of a spacecraft processor suitable for this category of low-cost micro-satellite, with high speed, high capacity, low power data handling based on a MSFC design and Bristol implementation.

The BOLAS project would draw on NASA's experience with the successful SEDS-1, SEDS-2, PMG, and TiPS missions. These missions employed the Small Expendable Deployer System (SEDS), which is being proposed as the tether deployer for BOLAS. The two SEDS missions were launched into orbit as Secondary Payloads on the Delta II launch vehicle in 1993 and 1994 [4]. The TiPS mission was launched by the US Air Force in 1996 and also used a SEDS deployer. The proven tether dynamics modelling software and tether test facilities at MSFC would be used to support design and development of the mission.

Experience gained in the OEDIPUS-A and C suborbital tether missions would be the base for the Canadian part of the tether activity. The Canadian team would have the lead responsibility for the system dynamics and stabilization of BOLAS. The tether retriever for the spin-up maneuver would be developed under sponsorship of the CSA Space Technology Branch. Its design would be based on the successful tether reel design of the OEDIPUS missions. Ground tests of the retriever and stabilization principles would be performed to qualify the system for in-orbit operation.

MISSION ARCHITECTURE

Mission Description

The BOLAS science experiments would be implemented with a low-cost spacecraft consisting of two small and nearly identical subsatellites connected by a tether of about 100 m length. Each subsatellite would carry an HF receiver, a dipole antenna, a GPS receiver and clock, and two instruments to measure electrons and ions in the ambient plasma. The baseline launch service is as a secondary payload with the Canadian Radarsat-II spacecraft on the Delta II vehicle in the year 2001. The tethered subsatellites would rotate in a cartwheel fashion, approximately in the orbit plane, to provide for scanning of the ionosphere by dipole HF antennas and the particle instruments. The required experiment operations involving measurement of direction of arrival (DOA) for RF transmissions would be carried out when the spacecraft is traversing the northern auroral region of Canada at an orbital height of 350 - 600 km. RF transmissions would be received from SuperDARN and CADI sites in northern Canada. The experiments involving reception of signals by the satellite's GPS antenna would be conducted when the line of sight between the BOLAS satellite and a particular GPS satellite passes through the ionosphere.

Orbit, Spacecraft, and Mission Parameters

The requirements of the science experiments outlined Table 1 can be met with two tethered subsatellites in a low-altitude, high-inclination orbit that maintain a fixed distance between each other and also are rotating relative to each other in a cartwheel fashion, to allow observation of the ionosphere in a variety of inertial orientations. Orbital heights of about 350 km altitude are sought for in-situ probe measurements of density irregularities that scatter observable waves. Pass altitudes in the collisionless plasma of the topside ionosphere also are required for the auroral wave-particle investigations. To insure passes through the auroral ionosphere in both cases would require an orbital inclination not less than 60°. A summary of the systems-level spacecraft and mission requirements are given in Table 2.

Table 2 Summary of the Systems Level Requirements

System Parameter	Requirement	Comments
Subsatellite separation (tether length)	50 m - 300 m 100 m (ideal)	The subsatellite separation is the baseline between the two satellites used to calculate the DOA of an incoming wave.
Rotation rate of BOLAS config	10-30 times the orbital rate	The BOLAS rotation permits DOA measurements at various orientations
Orientation of BOLAS config	in-orbit plane (ideal)	A cartwheel spin orientation is desired but is not mandatory
2-body attitude determination	±1°	Attitude determination of the two body vector in inertial space
Orbital altitudes	350 km - 600 km	Altitudes near 350 km are of primary interest, with the higher altitudes up to 600 km of secondary interest
Inclination	above 60 deg	measurements are to be made in and near the auroral oval
Local mean solar time (LMST)	10:00 - 14:00 (primary)	Experiments will be conducted at all LMSTs, however the 10:00-14:00 is the region of most interest. Hence, the orbit plane should remain in the 10:00-14:00 LMST for as long as possible.

Launch Vehicle and Operations

In discussions with NASA MSFC and NASA Headquarters, it has been agreed that NASA will endeavour to provide the launch, support integration of the spacecraft on the Delta second stage, and support the launch related operations until the BOLAS subsatellites are both deployed from the Delta second stage. An application for launch has been made to the NASA GSFC Secondary Payloads Program, jointly by the Science PI and a representative of NASA Marshall in November 1996. If the project is chosen for Phase A, the GSFC Secondary Payload Program Office will formally proceed with a decision process.

The current proposed baseline for launch is with the Radarsat-II mission as a secondary payload. Based on discussions with CSA, it is currently planned to launch in the fourth quarter of 2001. The expected mass margin for Radarsat-II, assuming it is identical to Radarsat-I, is 167 kg, which will allow for accomodating the BOLAS mission. Other launch options are possible, such as a launch with the IMAGE spacecraft in the year 2000, and may be evaluated in Phase A.

The BOLAS as a secondary payload would be manifested inside the same fairing envelope as the primary spacecraft on the Delta II vehicle. The accommodation of BOLAS on the Delta has been discussed with NASA and appears feasible. It is depicted in Figure 2. The BOLAS hardware will come under close scrutiny to ensure that it poses no danger to the primary spacecraft, as per the requirements and procedures of the Delta-II Secondary Payloads Manual [5].

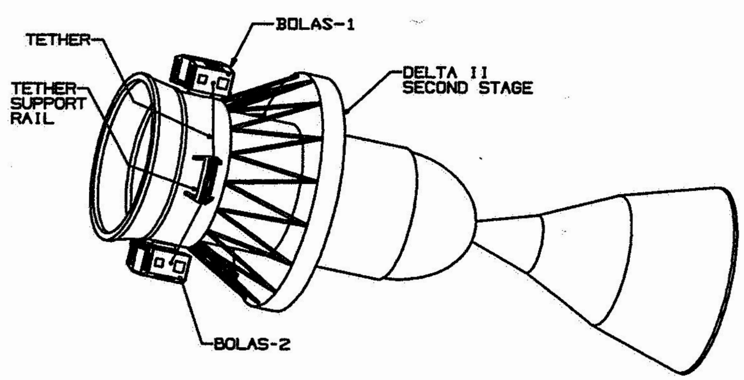

Figure 2 BOLAS Subsatellites Integrated on Delta II Second Stage

Mission Orbit Acquisition and Delta Operations

Immediately after deployment of the Radarsat-II spacecraft, the Delta II second stage and the attached BOLAS spacecraft would be in the Radarsat-II orbit, i.e. a circular orbit with 800 km altitude, and sun-synchronous with dawn-dusk orientation. Launch would be from Vandenberg AFB and the ascending node would cross the equator at 1800 hr PST. The inclination would be 98.64 degrees and its orbital period 100.7 minutes [6], [7].

When the primary science data is taken, the satellite should be passing through the auroral oval at low height. Both daylight and darkness passes are required over the course of the mission. The requirements can be satisfied with an orbit with perigee and apogee of 350 km and 600 km respectively, and a drift over the course of the mission from an initial dawn-dusk orientation, through to and beyond a noon-midnight orientation. The perigee should be near the auroral oval during the main period of science measurements. Figure 3 illustrates the orbit geometry. The BOLAS orbit selected has a drift relative to the sun (Ω-Ω_{ss}) of 120 degrees in six months. This requires an orbit inclination of 102.24 degrees. The corresponding drift rate of the perigee (i.e., line of apsides ω) is approximately -3 degrees per day. The BOLAS SCIFER orbit may be achieved with the Delta second stage after deployment of Radarsat-II with a two impulse orbit transfer strategy.

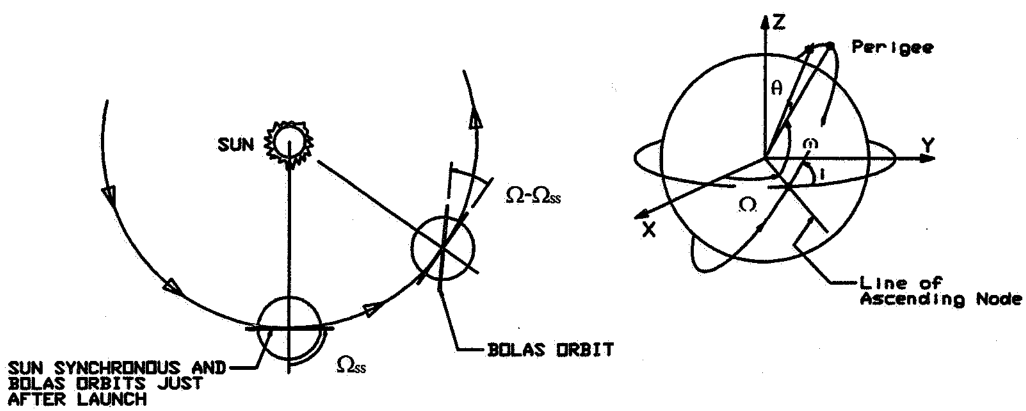

Figure 3 BOLAS Orbit Geometry Showing the Orbit Plane and Perigee Drift

Deployment, Separation and Spin-Up of the Spacecraft

The BOLAS subsatellite separation and spin-up is achieved by initially deploying the tether to a length of about 326 m, which would allow the gravitational forces to initiate a slow rotation of the two-body system. The tether would then be retrieved at a relatively high rate to allow the resulting Coriolis forces to spin-up the system to the final rate of about 0.2 rpm with the tether at 100 m long. An approach using cold gas thrusters has also been investigated, but this approach is not viable as the gas requirements are excessive, and an attitude control system would be required to orient the thrusters in the appropriate direction. With the gravity-gradient assisted spin-up, only a relatively simple tether retriever is required, which would be based on already proven technology developed for the OEDIPUS-C tether mission. The deployment, separation and spin-up of the spacecraft is depicted in Figure 4. A detailed timeline analysis confirms that all maneuvers with the Delta can be accomplished before the Delta batteries are depleted.

To confirm the feasibility of the gravity-gradient assisted spin-up, deployment simulations were conducted by NASA Marshall using flight proven tether dynamics software, with a deployer friction model based on test results of SEDS deployer hardware on NRL's TiPS mission [8]. The spin-up dynamics was jointly analyzed by NASA Marshall and Bristol, using simplified mathematical models that have been verified by comparing independent formulations by Bristol [9], NASA, and those found in the open literature [10].

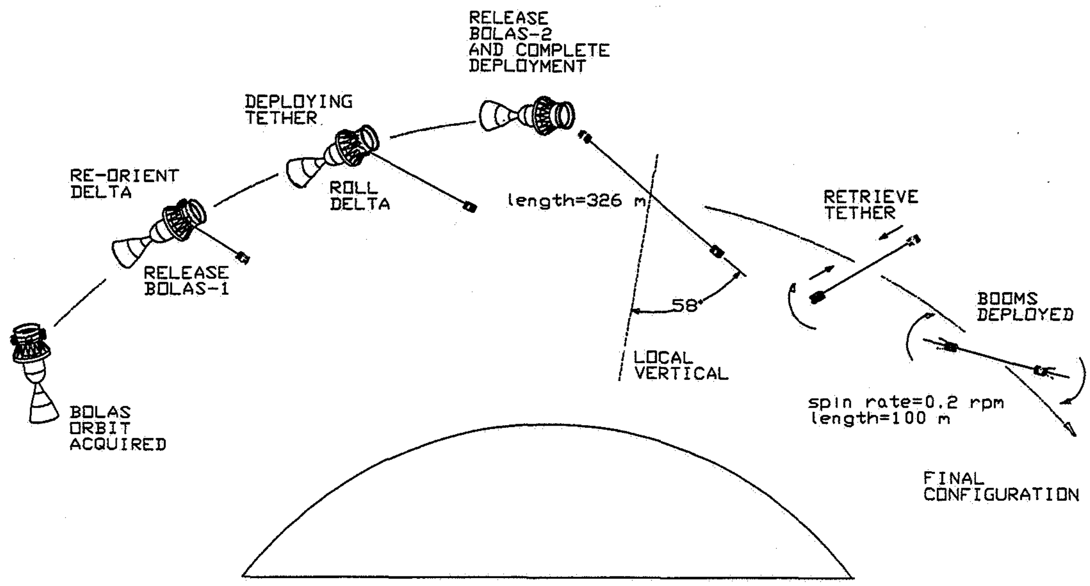

Figure 4 BOLAS Tether Deployment and Spin-Up Scenario

On-Orbit Science and Engineering Operational Modes

The mission will have several modes for operations coordination and liaison. The "Primary Science Mode" is used as the main operational mode for gathering scientific data, which defines the required spacecraft resources (i.e., power and on-board memory). The "Secondary Science Mode" allows for a variety of other scientific operational scenarios that can be carried out, using the resources that are available on the subsatellites (i.e., this could mean running the HF receiver without the SII and TECHS instruments, operating in different regions over the earth, etc.). The "Engineering & Dynamics Data Acquisition Mode" will gather data over several orbits for engineering, diagnostic, and dynamics analysis, adjust system spin rate if necessary, use only the resources provided by the spacecraft (as defined by the Primary Science Mode), and not be a driver on these resources. When the spacecraft is not in any of these operational modes, it is in a sleep mode requiring very low power.

Data Acquisition and Command Ground Station

The spacecraft is conveniently controlled from a CSA ground station in Saskatoon, using existing SHF ground equipment. The human resource requirements are very small as the mission duration is six months and the spacecraft is in sleep mode for much of the time; staffing would be most efficient via an arrangement with existing Saskatoon staff. An alternate to Saskatoon is the (Canadian Centre for Remote Sensing (CCRS) ground station in Prince Albert, Saskatchewan, and this would entail the installation of a low-cost UHF unit for uplink.

SPACECRAFT DESIGN

System Configuration

The BOLAS spacecraft is shown in Figure 5 in its final deployed configuration. It comprises two nearly identical subsatellites attached by a 100 m non-conductive tether with a pair of 3 m booms on each subsatellite aligned along the tether. The entire configuration is spinning at about 18 times the orbital rate, or approximately 0.2 rpm about its center of mass located close to the middle of the tether. The spin of the two body system is achieved by deploying the tether to initially about 326 m, and using the gravity-gradient forces to initiate a slow rotation. The tether is then retrieved at a high rate to its final length of 100 m, which causes Coriolis forces to spin-up the system. To implement this spin-up approach, a tether deployer (the mini-SEDS deployer from NASA Marshall) is used for the initial deployment and a separate tether retriever (based on OEDIPUS technology) is used to retract the tether.

This approach of using separate systems to deploy and retrieve the tether minimizes hardware complexity and allows for capitalizing on the technology developed for the US tether programs (SEDS, PMG, TiPS) and the Canadian OEDIPUS tether missions.

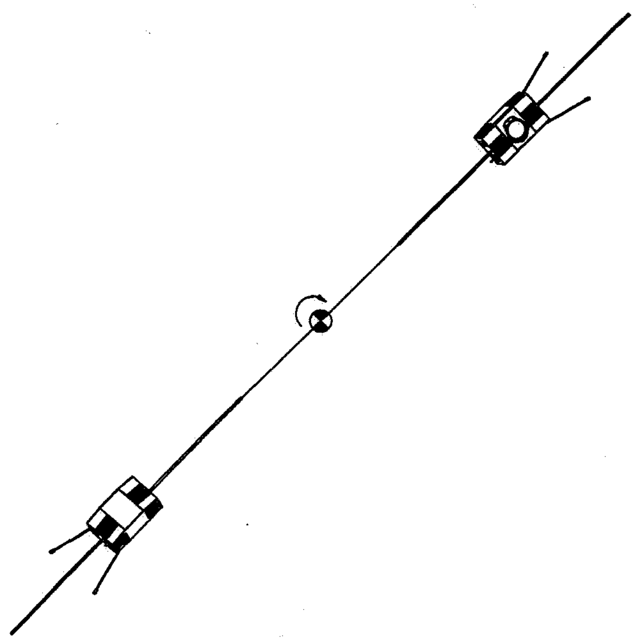

Figure 5 BOLAS in Final Deployed Configuration

Science Instruments

The HF Receiver is a broadband receiver for measuring wave fields from both manmade and spontaneous sources. It has a preamplifier that matches the high impedance of the BOLAS dipoles to the 50-ohm input of the receiver signal processor. The latter comprises two branches. One is for direct amplification at frequencies up to 50 kHz. The other is a double heterodyne for frequencies between 100 kHz and 20 MHz, at 50 kHz steps. The receiver has considerable heritage from the instrument flown on OEDIPUS-C.

The Suprathermal Ion Instrument (SII) images the 2-D ion distribution from 0-50 eV, and provides an integral measure of ion flux at rates sufficient to resolve localized ion heating structures on spatial scales of tens of meters.The two SII's on BOLAS are identical and each consists of three parts: 1) a 2.5 cm diameter cylindrical sensor head with a rectangular baseplate housing electronics, 2) a 1.0 m (nominal) boom supplied by Bristol Aerospace, and 3) a power and control unit housed inside the spacecraft. The SII is based on the design of the Freja Cold Plasma Analyzer [11], with two major modifications: 1) the dimensions will be shrunk by a factor of roughly three, and 2) the detector design will be based on a charge-coupled device (CCD), not a network of charge amplifiers.

The Thermal Electron Capped Hemisphere Spectrometer (TECHS) is an azimuthal imaging tophat electrostatic analyzer. By sweeping the analyzer voltage, TECHS measures a count rate that can be directly related to the distribution function of low energy and thermal electrons [12]. From these measurements, integral moments of the electron distribution function, including density, anisotropic temperature, bulk drift, and heat flux may be derived. A version of the instrument flew successfully on the SCIFER sounding rocket [13]. Similarly to the SII, the TECHS sensor also mounts on the end of a 1 m boom.

Subsatellite Design

The layout of the subsatellites is shown in Figure 6 (note only BOLAS-1 is shown). The overall configuration is driven by the accommodation requirements for the Delta II launch vehicle. Note that the tether from

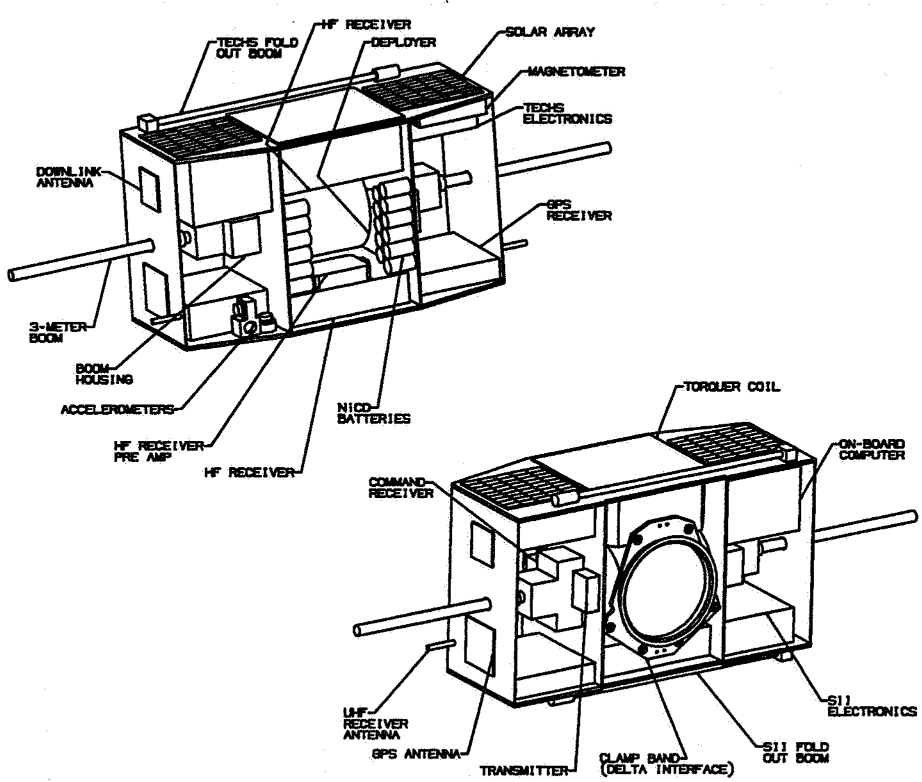

Figure 6 BOLAS Subsatellite Layout

the tether deployer (and similarly from the retriever on BOLAS-2) goes through the boom package and out the tip of the boom. This ensures the booms are aligned with the tether and avoids the possibility of the tether getting tangled around the boom. It also helps to stabilize the payload oscillations relative to the tether. The concept uses two individual BI-STEM boom packages from Astro Aerospace, each deploying a single element, as these allow for easily feeding the tether through the back of the boom package and out the tip of the boom when stowed and when deployed. This type of boom was flown on OEDIPUS-C. As the payloads will be stabilized by the tether tension with damping provided by the BI-STEM booms, subsatellite attitude control about the orthogonal axes to the tether is not required. However, primarily for thermal reasons, a magnetic torque coil is provided to allow for some intermitent open loop control (via the ground) about the tether axis to ensure sufficient spin so that the solar energy can be equally distributed on all sides. Full three-axis attitude determination would be provided by the magnetometer via processing on the ground. A magnetometer-only based attitude determination scheme was used for the US SEDS tether missions, and NASA Goddard will develop an algorithm based on this technique for the BOLAS mission. The accelerometers would be used to provide accurate attitude rate information and also to determine the tether tension.

The main hardware elements of each subsatellite are listed in Table 3. The two subsatellites are identical with the exception of the tether deployer and retriever. To reduce development costs, only one subsatellite design will be developed that will be able to accommodate either the tether deployer or retriever.

Table 3 BOLAS Spacecraft Mass and Power Summaries

BOLAS-1			BOLAS-2		
Item	**Mass (kg)**	**Power (W)**	**Item**	**Mass (Kg)**	**Power (W)**
HF rcvr and pre-amp	9	10.8	HF rcvr and pre-amp	9	10.8
Booms (2 @ 3 m)	2.7	-	Booms (2 @ 3 m)	2.7	-
TECHS electronics	3	5	TECHS electronics	3	5
TECHS sensor & boom	1	-	TECHS sensor & boom	1	-
SII electronics	3.5	10	SII electronics	3.5	10
SII sensor & boom	1.5	-	SII sensor & boom	1.5	-
Tether deployer	1.5	-	Tether retriever	2.5	6.44
Tether (maximum)	2.7	-			
Transmitter (2W)	0.45	18.2	Transmitter	0.45	18.2
Tx Antenna patch (2)	0.2	-	Tx Antenna patch (2)	0.2	-
Command Receiver (2)	0.72	4.48	Command Receiver (2)	0.72	4.48
Antenna - Rx (2)	0.4	-	Antenna - Rx (2)	0.4	-
On-board Computer	6	2.5	On-board Computer	6	2.5
Solar Arrays	2.1	-	Solar Arrays	2.1	-
Batteries	3.4	-	Batteries	3.4	-
GPS receiver	2.3	6	GPS receiver	2.3	6
Magnetometer	0.2	1.12	Magnetometer	0.2	1.12
Torque coil	0.1	2	Torque coil	0.1	2
Accelerometers (3)	0.45	1.80	Accelerometers (3)	0.45	1.80
Wiring Harness	4.5	-	Wiring Harness	4.5	-
PDU & DC-DC conv	0.3	2.2	PDU & DC-DC conv	0.3	2.2
Structure & thermal	14.1	-	Structure & thermal	14.1	-
Contingency (20%)	12.02	12.82	Contingency (20%)	11.68	14.11
TOTAL	**72.1 kg**	**76.9 W (peak)**	**TOTAL**	**70.1 Kg**	**84.65 W (peak)**

The power system consists of two small solar arrays (each made up of 45 2 x 4 cm Si cells) mounted on each face of the subsatellite (except the ends), a 4.5 A-hr battery made up of commercial NiCd D-cells, and a Power Distribution Unit (PDU) and DC-DC converters used to provide switchable power to the instrument and subsystems. The pair of solar arrays on each face provide nominally 28 V which provides the necessary power in any arbitrary orientation of the two-body system and at all orientations of the orbit plane relative to the sun. Also since the current generated from the solar arrays is very low (well below the trickle charge rate), a battery charge regulator and shunt regulator are not needed and the solar arrays are connected directly to the batteries. This approach has been used on other microsatellites as it simplifies the power system and helps to minimze the costs.

The on-board computer (OBC) consists of a processor card, an I/O card, a data handling card, and four mass memory cards. It decodes commands from the command receiver, executes real-time and time-line commands, collects science and housekeeping data, provides 128 Mbytes of on-board storage, and outputs a 2 Mbps serial bit stream to the telemetry transmitter. Low power is a prime requirement and a 80C186 processor has been selected for the baseline concept, as it has adequate capability at very low power and has considerable space flight heritage. The OBC will be developed jointly by Bristol and NASA Marshall. It will be manufactured from SEL-free, military grade parts, where appropriate. Some screened commercial grade parts may be used in non-critical areas to reduce costs without compromising mission critical functions. Through hole construction will be employed also to reduce cost without greatly affecting overall size and mass.

The GPS receivers are modified versions of the TurboStar units from Allen Osborne and Associates. These units provide the necessary features including dual frequency capability and sufficient on-board processing to allow arbitrarily selecting up to eight GPS satellites. This unit has been used in space and is being implemented in a number of smallsat and microsat missions including the GPS-Met/MicroLab-1 mission by Orbital Sciences Corp., NASA's WakeShield facility, the Danish Oersted microsatellite, and others. The units would require minor modifications to provide a clean 5 MHz reference signal that will be used by the HF receiver for synchronization between the signals received from BOLAS-1 and 2 via ground processing (currently the reference signal is corrected by the GPS clock once every second). The baseline downlink transmitter is an Aydin Vector T-100S/L Series S-Band 2 Watt RF Telemetry Transmitter which has been used on the FREJA mission and accepts binary bits and produces a linear or binary phase modulated carrier. Two Aydin Vector RCC-100 Series UHF Command Control receivers are proposed where each connects to a small UHF dipole antenna located at each end of the subsatellite. Both receivers will be on continuously (although strobed to minimize power requirements) so that a command can be received in almost any orientation of the two-body system. The GPS and transmitter antennas are microstrip patches that are located on each end of the subsatellite as shown in Figure 6. The GPS antennas (L-band) will be designed for this mission to provide a nearly omni-directional pattern. The two S-band antennas for the transmitter can be switched via timeline commands, so that only one is used at a given time when it is oriented towards the ground.

In a Phase A study, the addition of some optional hardware will be considered if it can be accommodated from a resource and cost point of view. The possible additional hardware includes a "running line" tensiometer such as that used in the SEDS tether missions. This will provide a direct measurement of tether tension which may provide better tether dynamics data. The other possibility is to add a small digital camera aligned to view the tether and the subsatellite on the other end. The current resources on the subsatellite allow for taking a picture every second for up to approximately 5 minutes, and store on-board for about 12 minutes during an overhead pass, where the data would be downlinked in real time. Once the data, which comprises a series of "still pictures," is obtained on the ground, it can be processed to make a video clip (i.e., an MPEG file) that runs at 30 frames a second. Hence, this will be a 30 fold speed-up of the motion (i.e., 5 minutes worth of data will provide a 10 second video clip). This should be very interesting from a dynamics point of view as the actual dynamics is very slow, so the 30 x speed will make the data more useful. Additionally, this will be very valuable for public relations purposes as video clips could be taken periodically to show actual flight data during different stages of the mission (deployment phase, spin-up phase, at various times through-out the life of the mission). Both the tensiometer and the digital camera may not be expensive add-ons and will be able to operate within the current resources of the subsatellites, hence it may be warranted to include these items in the mission.

Considerable system design and trade studies were conducted over the past year to arrive at the design described above [14]. This included significant contributions from NASA MSFC in the final design iteration. This level of detail was undertaken to firmly establish feasibility of the mission and to establish a solid basis for cost estimates. The effort was made jointly with the science team, the technology team and with NASA Marshall to iterate the design to one that minimizes cost and complexity and yet maintains the capability to undertake leading edge scientific investigations. It is believed that the design achieves a good balance between cost and performance.

Spacecraft Integration and Test Approach

Bristol has developed plans to upgrade their current space payload integration facility to be suitable for supporting the development of smallsats/microsats. This new facility will be installed by the company once a smallsat/microsat mission is approved. Integration of the spacecraft subsystems and instruments will be undertaken at the facility following fabrication of the structural elements and wiring harnesses. Functional tests of the spacecraft subsystems will carried out, both with and without the instruments and in various combinations to check for nominal performance in all of the planned operating modes. Verification tests will also be carried out at the thermal extremes expected during the mission, followed by vibration testing per the DELTA specification levels.

Additionally, the spacecraft hardware-in-the-loop simulation and testing facility, currently being developed by Bristol with support from CAE electronics under contract to the CSA, Space Technology Branch [15], will be used to support the spacecraft design, software development and full spacecraft functional check-out.

Environmental testing is planned to be carried out on the subsatellites at the CSA David Florida Laboratory (DSL) facilities with support from appropriate team members. Antenna pattern characteriztion and performance testing will also be undertaken while at DFL.

POTENTIAL BENEFITS

Mitigation of Undesirable Effects of the Ionosphere

Improved understanding of the ionosphere leads to practical benefits in a number of areas. Ionospheric processes limit the performance of space communication systems. Our current knowledge of these processes is not adequate for removing the perturbations they cause in signals. Better models of density irregularities effectively addresses problems which arise in geosynchronous and low-earth orbit communications at frequencies up to UHF. As in communications, ionospheric irregularities also limit the GPS. Two areas of public concern which could benefit from improved performance of GPS are earthquake prediction and global warming, through the monitoring of the movement of the earth's crust and the reduction of glacier thickness, respectively.

The understanding of the ionosphere is arguably the most significant limiting factor on communication system performance. In many instances the root of the problem is the densest part of the ionosphere, the F region at around 300 km altitude. It is here that dynamical processes are continuously at work to produce turbulence of a wide range of scale sizes. This turbulence is manifested in random distributions of density and temperature of the ionospheric plasma. The randomness of the medium acts to randomize signals being sent through it, at all frequencies in the radio spectrum. The BOLAS transionospheric experiments deal with such a medium.

Even though satellite communications at UHF are now widely available, there are still major perturbations of communications channels during magnetic storms which produced increased ionospheric irregularity. Communications systems operators would like to be able to predict the occurrence, in time and location, of the perturbing irregularities. This means understanding how irregularities form and, once formed, what structure (or spectrum) they have. The latter is important because it determines whether irregularities are a problem for a specific carrier frequency. If the spectral power density of irregularities is low at small scale sizes, higher frequencies, say UHF, are not affected but lower frequencies may be affected.

BOLAS would address these problems by determining the location of irregularity structures and measuring their irregularity spectrum using the particle detectors. Propagation between the ground and BOLAS would be used to test predictions of propagation characteristics when irregularities are present. The measured irregularity spectrum would be then used to predict the fluctuations expected on certain frequencies. The predictions then could be compared with the observations. A theory that agrees with observations could be of use for predicting the deleterious effects in practical communications systems.

BOLAS propagation investigations are carried out at HF where the weaknesses of communications are well documented: the disruptive effects of natural unpredictable events like polar cap absorption and sudden ionospheric disturbances; the difficulty of characterizing manmade interference; and limited bandwidths. Seen as a fraction of the total spending on North American communications, the HF segment at first looks small. But a need for and use of HF clearly persists in northern Canada and Alaska. Low population densities neither justify nor need the expensive communication infrastructure used in the south. HF offers simple equipment and operating procedures. In the military context, the ionosphere is a robust medium that recovers much quicker than other links from natural and manmade disruptions.

With the ever-increasing bandwidth requirements in satellite-to-ground microwave communications, and with the continuous progress in communications equipment technology, the frequency-spread and time-spread characteristics of transionospheric propagation paths are becoming the limiting factors in communications performance, in terms of data rate and error rate. The BOLAS ionospheric science mission could make an important contribution to the understanding of the degradation imposed by the ionosphere. BOLAS could measure ionospheric irregularity characteristics for comparison with the influence on microwave waveforms on paths intersecting the ionospheric regions where BOLAS performs its measurements.

The precision of geodetic position provided by the GPS is affected by the ionosphere. Two areas of national and worldwide concern are earthquake prediction/detection and global warming. Seismologists currently are trying to determine whether motions of the earth's crust can be used to predict earthquakes. For example, in the area of the Juan de Fuca plate in British Columbia, the magnitude of this movement is on the order of millimetres per year. The magnitude of errors in current positioning techniques imposed by ionospheric irregularities can be larger than this. If a prediction method shows promise, GPS users will want to develop techniques for subtracting the noise on transionospheric propagation caused by the irregularity of the ionosphere. Given the nature of the GPS satellite orbits, the physical geometry of the ionospheric scattering will be different from that experienced in the satellite-ground communications links mentioned above, but there will be a common need to understand the origins and characteristics of the turbulence.

Glaciologists, meanwhile, are interested in using GPS receivers on the polar ice to gather information about systematic widespread decreases in glacier thickness indicating global warming. Since the high latitude positions of these receivers necessitate paths to GPS orbit through the turbulent ionosphere, it may become necessary in this context to understand the ionospheric spectrum of irregularity in order to subtract the noise that it imposes on signals from the GPS satellites.

Space Technology

BOLAS offers the opportunity for technology development that is relevant to future North American programs and that is strategic to the industrial partners. The tether technology base developed would be relevant to several future international application areas, such as sample return capability from ISS, interferometric synthetic aperture radar, and rotating space stations with artifical gravity for long duration human space presence. Tethers also offer a unique technology solution that provides for science missions that require two separated locations in space. Additionally, the mission offers opportunities for flight experience in a number of areas that are integral to low-cost satellite development, such as GPS hardware and techniques, and tether hardware.

The mission also provides unique opportunities for developing strategic technologies in industry that can lead to future commerical opportunities. Enhancing the tether expertise provides a niche capability that can lead to involvement in future programs. Tethers are an emerging technology that is now being seriously considered by the international community for a number of applications. The cooperation with NASA MSFC and GSFC provides alliances and knowledge transfer in tether expertise, attitude determination and control, spacecraft computer technology, and secondary payload integration.

The low power high data rate microsat/smallsat class computer is of particular interest to industry, as it not only supports smallsat missions, but is also relevant to commerical initiatives in guidance and control system products. The mission also provides opportunities to gain experience with GPS for space applications and to demonstrate the application of differential GPS techniques in space. The development of a new GPS antenna to provide near omni-directional coverage is also of interest, as it may have commercial applications for spacecraft which undergo arbitrary rotational motions.

ACRONYMS

AFB	Air Force Base
BOLAS	Bistatic Observations using Low Altitude Satellites
CADI	Canadian Auroral Digital Ionosondes
CANOPUS	Canadian Auroral Network for OPEN Unified Study
CCRS	Canadian Centre for Remote Sensing
CSA	Canadian Space Agency
DOA	Direction of Arrival
DFL	David Florida Laboratory
EM	Electromagnetic
GEO	Geosynchronous Earth Orbit
GPS	Global Positioning System
GSFC	Goddard Space Flight Center
HF	High Frequency
IO	Input/Output
ISS	International Space Station
ISTP	International Solar Terrestrial Program
LEO	Low Earth Orbit
MSFC	Marshall Space Flight Center

NASA	National Aeronautics and Space Administration
OBC	On-Board Computer
OEDIPUS	Observations of Electricfield Distribution in Ionospheric Plasmas a Unique Strategy
PDU	Power Distribution Unit
PMG	Plasma Motor Generator
RF	Radio Frequency
R&D	Research and Development
SEDS	Small Expendable Deployer System
SII	Suprathermal Ion Instrument
SSP	Space Science Program
STP	Space Technology Program
SuperDARN	Super Dual Auroral Radar Network
TECHS	Thermal Electron Capped Hemisphere Spectrometer
TiPS	Tether Physics Experiment
UHF	Ultra High Frequency

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SPACE TRANSPORTATION SYSTEMS USING TETHERS

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Abstract

The groundwork has been laid for tether space transportation systems. NASA has developed tether technology for space applications since the 1960's. Important recent milestones include retrieval of a tether in space (TSS-1, 1992), successful deployment of a 20-km-long tether in space (SEDS-1, 1993), and operation of an electro dynamic tether with tether current driven in both directions—power and thrust modes (PMG, 199 3). Various types of tethers and systems can be used for space transportation. Short electrodynamic tethers can use solar power to 'push' against a planetary magnetic field to achieve propulsion without the expenditure of propellant. The planned Propulsive Small Expendable Deployer System (ProSEDS) experiment will demonstrate electrodynamic tether thrust during its flight in early 2000. Utilizing completely different physical principles, long non-conducting tethers can exchange momentum between two masses in orbit to place one body into a transfer orbit for lunar and planetary missions. Recently completed system studies of this concept indicate that it would be a relatively low-cost in-space asset with long-term multimission capability. Both methods of using tethers for space transportation and propulsion are described in the paper.

List of Acronyms and Definitions

Acronym or Initialism	Definition
GEO	Geostationary Earth Orbit
GTO	Geostationary Transfer Orbit
LEO	Low Earth Orbit
LTO	Lunar Transfer Orbit
NASA	National Aeronautics and Space Administration
PMG	Plasma Motor Generator
ProSEDS	Propulsive Small Expendable Deployer System
RLV	Reusable Launch Vehicle
SEDS	Small Expendable Deployer System
TiPS	Tether Physics and Survivability Experiment
TSS	Tethered Satellite System

Introduction

Since the 1960's there have been at least 16 tether missions. In the 1990's, several important milestones were reached, including the retrieval of a tether in space (TSS-1, 1992), successful deployment of a 20-km-long tether in space (SEDS-1, 1993), and operation of an electrodynamic tether with tether current driven in both directions—power and thrust modes (PMG, 1993). A list of known tether missions is shown in Table 1.

NAME	DATE	ORBIT	LENGTH	COMMENTS
Gemini 11	1967	LEO	30 m	spin stable 0.15 rpm
Gemini 12	1967	LEO	30 m	local vertical, stable swing
H-9M-69	1980	suborbital	500 m	partial deployment
S-520-2	1981	suborbital	500 m	partial deployment
Charge-1	1983	suborbital	500 m	full deployment
Charge-2	1984	suborbital	500 m	full deployment
ECHO-7	1988	suborbital	?	magnetic field aligned
Oedipus-A	1989	suborbital	958 m	spin stable 0.7 rpm
Charge-2B	1992	suborbital	500 m	full deployment
TSS-1	1992	LEO	<1 km	electrodynamic, partial deploy, retrieved
SEDS-1	1993	LEO	20 km	downward deploy, swing & cut
PMG	1993	LEO	500 m	electrodynamic, upward deploy
SEDS-2	1994	LEO	20 km	local vertical stable, downward deploy
Oedipus-C	1995	suborbital	1 km	spin stable 0.7 rpm
TSS-1R	1996	LEO	19.6 km	electrodynamic, severed
TiPS	1996	LEO	4 km	long life tether, on-orbit (6/97)

Table 1. Known tether flights.

Tether Applications For Space Transportation

Various types of tethers and tether systems can be used for space transportation. Long non-conducting tethers can be used to exchange momentum between two masses in orbit. Shorter electrodynamic tethers can use solar power to 'push' against a planetary magnetic field to achieve propulsion without the expenditure of propellant. Below are descriptions of the main types of tether space transportation systems.

Electrodynamic Tethers

A predominantly uninsulated (bare wire) conducting tether, terminated at one end by a plasma contactor, can be used as an electromagnetic thruster. A propulsive force of $\mathbf{F} = \mathbf{IL} \times \mathbf{B}$ is generated on a spacecraft/tether system when a current, I, from an on-board power supply is fed into a tether of length, **L**, against the emf induced in it by the geomagnetic field, **B**. This concept will work near any planet with a magnetosphere (Earth, Jupiter, etc.) This was demonstrated by the Tethered Satellite System Reflight (TSS-1R) mission – the Orbiter experienced a 0.4 N electrodynamic drag thrust during tether operation[1].

An electrodynamic tether upper stage could be used as an orbital tug to move payloads within low earth orbit (LEO) after insertion (Figure 1). The tug would rendezvous with the payload and launch vehicle, dock/grapple the payload and maneuver it to a new orbital altitude or inclination within LEO *without the use of boost propellant*. The tug could then lower its orbit to rendezvous with the next payload and repeat the process. Such a system could conceivably perform several orbital maneuvering assignments without resupply, making it low recurring cost space asset. The performance of a 10 kW, 10 km tether system for altitude changes is illustrated in Figure 2. The same system can be used to change the orbital inclination of a payload as well. Figure 3 can be used to determine the available inclination change for a particular spacecraft and payload mass by dividing the 'specific inclination rate' indicated by the total system mass as a given altitude.

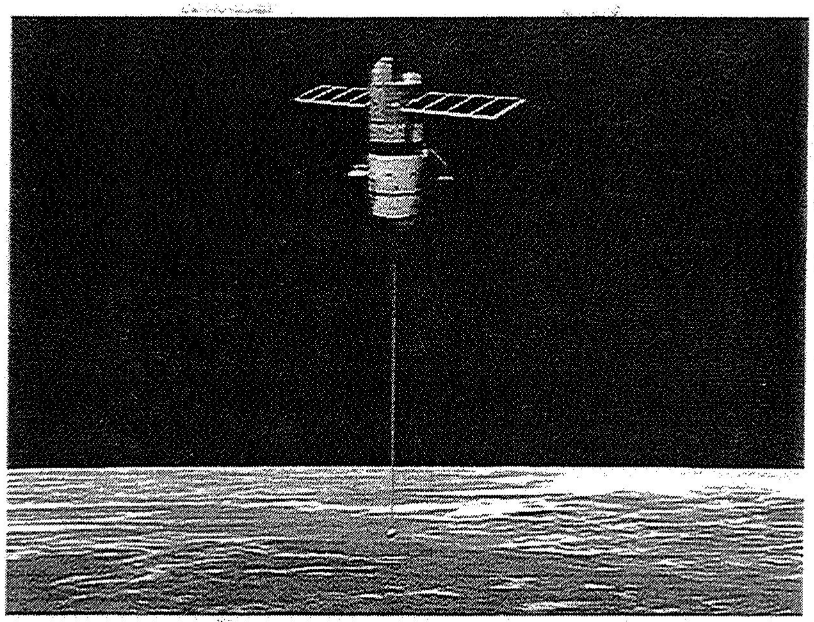

Figure 1. Artist concept of the Electrodynamic Tether Upper Stage Demonstrator.

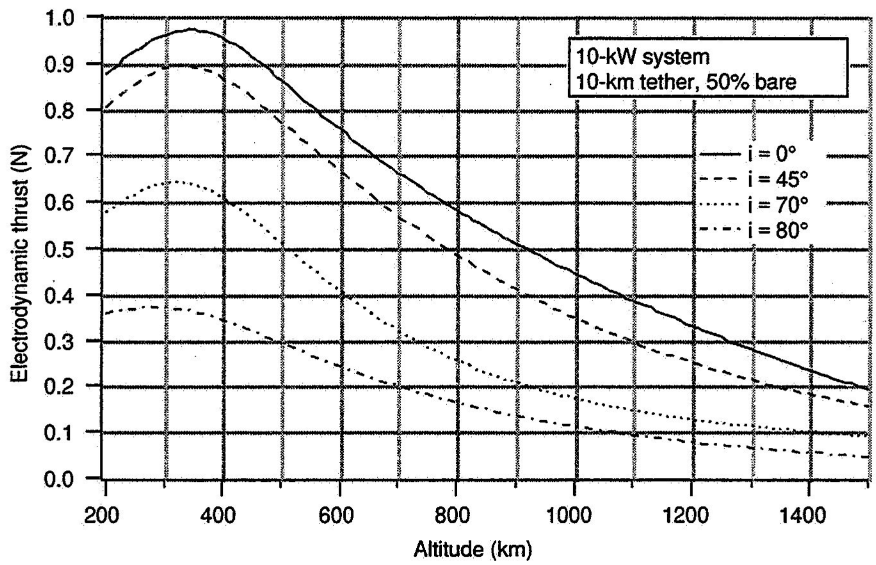

Figure 2. The performance of an electrodynamic tether thruster varies with altitude in the ionosphere (where i is the orbital inclination).

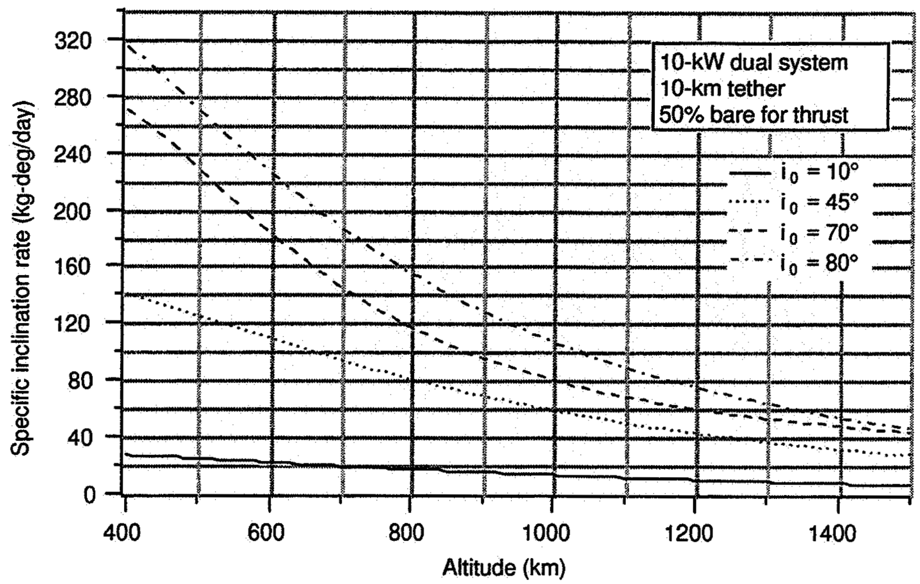

Figure 3. The performance of an electrodynamic tethers thruster for inclination change applications depends strongly on the initial orbital inclination (i_0).

ProSEDS Flight Experiment

From theoretical analyses and preliminary plasma chamber tests, bare tethers appear to be very effective anodes for collecting electrons from the ionosphere and, consequently, attain high currents with relatively short tether lengths[3]. A flight experiment to validate the performance of the bare electrodynamic tether in space and demonstrate its capability to produce thrust is planned by NASA for the year 2000. The ProSEDS (Propulsive Small Expendable Deployer System) experiment will be placed into a 500 km circular orbit as a secondary payload from a Delta II launch vehicle. The flight-proven SEDS will be used to deploy a 5 km predominantly bare copper wire attached to 20 km of insulating Spectra tether and 25 kg endmass.

Once on orbit, the deployer will reel-out the tether and endmass system to a total length of 25 km. Upward deployment will set the system to operate in the generator mode, thus producing drag thrust and electrical power. The drag thrust provided by the tether will deorbit the Delta II upper stage in approximately three weeks, versus its nominal 1.5 year lifetime in a 500 km circular orbit. Approximately 100 W electrical power will be extracted from the tether to recharge mission batteries and to allow extended measurements of the system's performance until it reenters.

'Hanging' Momentum Exchange Tethers

Momentum exchange tethers provide a unique advantage to RLV systems for boosting payloads to higher orbits. This stems from the fact the RLV must reduce its altitude after payload release in order to return to Earth, while at the same time the payload it ejects must increase its altitude to reach the desired orbit. In other words, momentum must be removed from the RLV and added to the payload. Tethers provide a method to make this exchange[4].

Operationally, this can be accomplished by the deployment of the payload upward on a long (~20 km) tether from the RLV (Figure 4). Libration begins and the momentum is transferred from the heavy RLV to the payload. Upon release, the RLV experiences a deboosting force and the payload is inserted into an orbit with apogee many times the tether length higher than previously[5]. This was demonstrated inadvertently when the electrodynamic tether broke on the TSS-1R flight, sending the endmass into a new orbit. The technology for this type of momentum exchange utilizing a tether exists today and was successfully demonstrated with the SEDS in 1993[6].

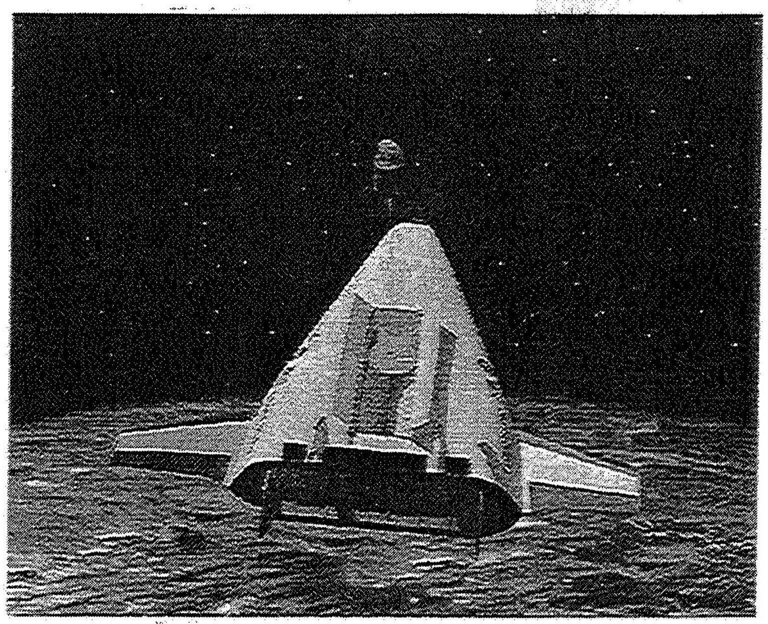

Figure 4. Artist concept of an upper stage tether deployed from the Reusable Launch Vehicle.

The physics governing the momentum exchange is best illustrated in Figure 5. The tether system is stabilized along the local vertical by the gravity gradient force with the center of mass maintained at the orbital velocity for a given altitude. The satellite is thus at a superorbital velocity and the RLV at a suborbital one. After the tether is cut, the angular momentum is exchanges between the two masses: the center of mass of the orbit remains the same while the satellite is injected into a higher orbit with the perigee at the release location. The RLV is injected into a lower orbit with apogee at the release location.

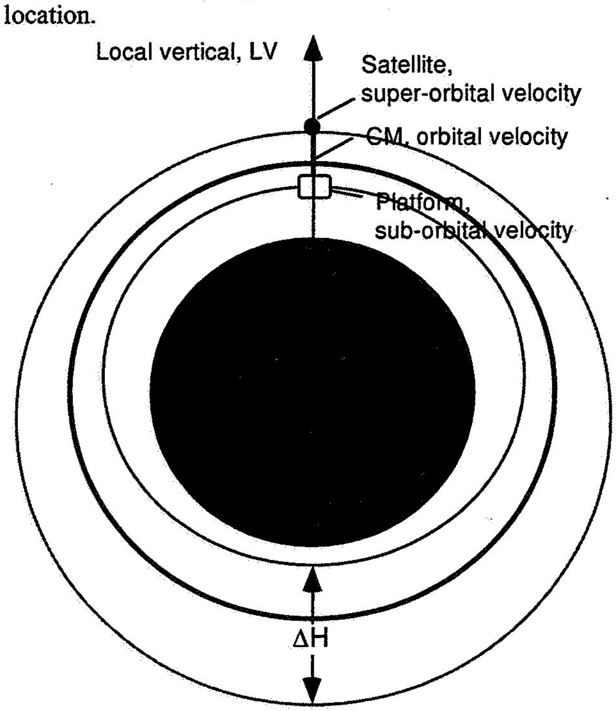

Figure 5. Orbits of the launch vehicle and satellite after release in the 'hanging' tether boost scenario.

Rotating Momentum Exchange Tethers

A spinning tether system can be used to boost payloads into higher orbits with a Hohmann-type transfer. A tether system would be anchored to a relatively large mass in LEO awaiting rendezvous with a payload delivered to orbit. The uplifted payload meets with the tether facility which then begins a slow spin-up using electrodynamic tethers (for propellantless operation) or another low thrust, high Isp thruster. At the proper moment and tether system orientation, the payload is released into a transfer orbit – potentially to geostationary transfer orbit (GTO) or Lunar Transfer Orbit (LTO). A network of such systems could be developed to "hand off" a payload until it reaches the desired location.

The physics governing a rotating momentum exchange system is illustrated in Figure 6. Following spin-up of the tether and satellite system, the payload is released at the local vertical. The satellite is injected into a higher orbit with perigee at the release location; the orbital tether platform is injected into a lower orbit with apogee at the release location. The satellite enters a GTO trajectory and accomplishes the transfer in as little as 5-16 hours, where the lower number applies to a single-stage and the higher number to a two-stage system. The platform then reboosts to its operational altitude using electric thrusters. The system thus achieves transfer times comparable to a chemical upper stage with the efficiencies of electric propulsion.

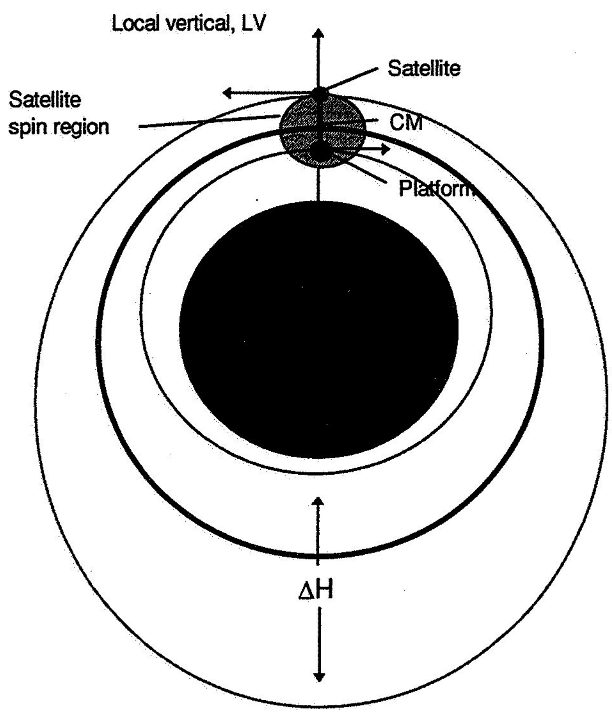

Figure 6. Orbits of the launch vehicle and satellite after release in the 'spinning' tether boost scenario.

As an example, the length of an idealized single-stage system for transferring a 1,000 kg payload from LEO to a higher orbit is illustrated in Figure 7. The maximum acceleration is limited to 10 g. An artist concept for the spinning momentum exchange tether transportation system is seen in Figure 8.

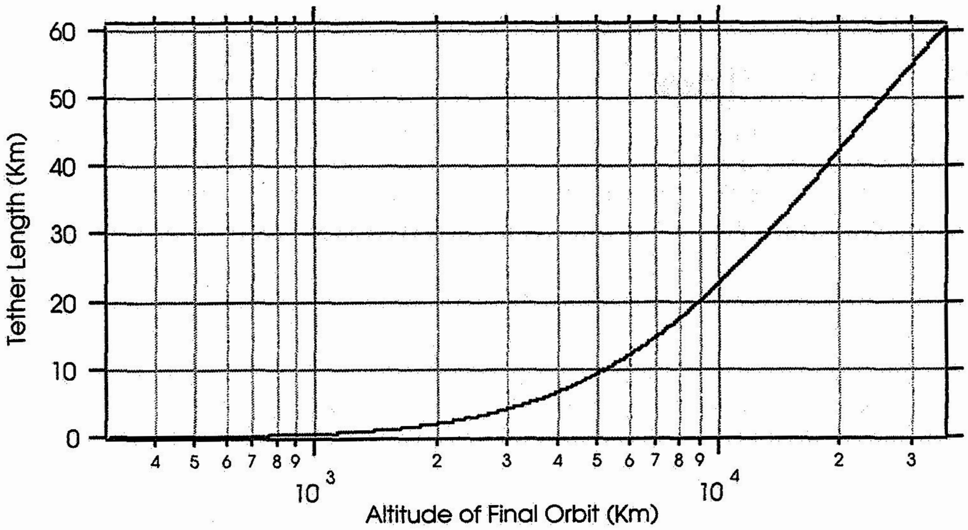

Figure 7. Dimensions of a single-stage LEO tether transfer system.

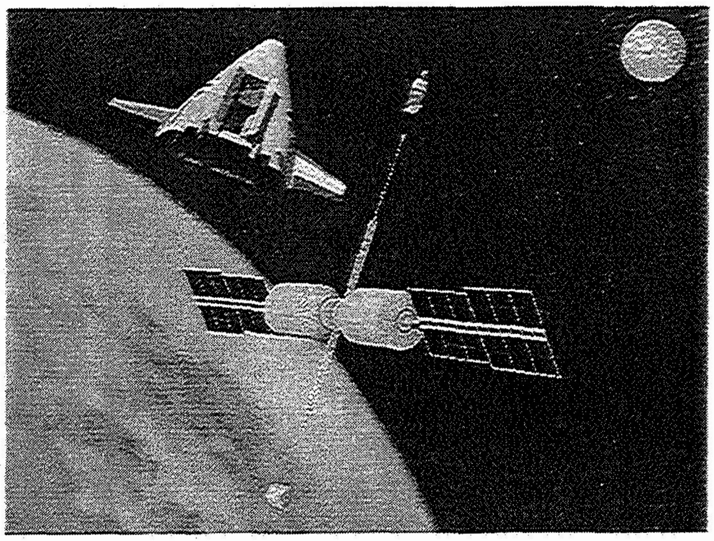

Figure 8. Artist concept of a LEO-to-GTO tether transportation facility.

<u>Other Tether Transportation Applications</u>

In addition to those discussed above, tethers can be used in other configurations for space transportation. Among these are:

1. Lunar Rotovator[6]: A rotating tether facility placed in orbit around the moon takes advantage of the absence of a lunar atmosphere to 'dip' downward to the surface and deposit/grapple payloads. Used in conjunction with a multistage rotating tether system this could make routine access to the lunar surface cost effective.
2. Electrodynamic Catapult: A linear motor accelerates a payload along a long conducting tether 'guide' to achieve a ΔV of 30 - 100 km/s. The concept is basically a space-based rail gun launcher.
3. Tether Launch Towers or Space Elevators[7]: Long tethers (>100's of km) suspended in orbit to transfer payloads via space elevators.

Conclusions

Tether technology has advanced significantly since its inception over 30 years ago. The recent successes of the SEDS system show that tethers are ready to move from experiment and demonstration to application. One of the most promising applications for tethers is in the area of space propulsion and transportation. The use of electrodynamic tethers for reusable upper stages will soon be demonstrated with the ProSEDS mission and the technology is being considered for the proposed Europa Orbiter. Momentum exchange tethers offer the highest payoff for transfer of payloads between LEO and GEO, GTO and lunar space. Further study and development of the technology are planned.

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THE SOLAR SYSTEM CRUISER - AN EVOLUTIONARY APPROACH TO AN INTERSTELLAR SPACECRAFT MISSION ARCHITECTURE

Divya Chander[1]

Although the current space climate is focused on developing cheaper access to space and performing smaller, well-defined science payloads, the ultimate future calls for a permanent human presence in space, crowned with the achievement of interstellar travel.

Whether that expansion comes from practical drives – the taxation of the limits of the carrying capacity of Earth – or more romantic, exploratory impulses, prudence dictates that we begin preparations far in advance of our departure date. There are numerous technological feats to master, in the guise of physical systems such as propulsion and shielding, and biological systems such as life support. Serious issues of design and systems integration will also emerge; their definition, scope, and ultimate resolution may need several testing iterations.

A Solar System Cruiser that cycles between inner and outer planets, improving and evolving along each voyage, represents a phased mission architecture that both utilizes existing infrastructure while validating future technologies and testing human factors limits. As such, it might provide a valuable exercise in design learning that will serve humankind well in future grand undertakings such as interstellar travel. In addition, it may provide valuable infrastructure, material resource and science development for its useful lifetime.

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DESIGN SPECIFICATIONS FOR MESO-ARCHITECTURE OF A MULTI-GENERATIONAL, INTERSTELLAR STARSHIP BASED ON HABITABILITY CRITERIA

Divya Chander[1]

As the limits of space travel are dictated by a velocity far below the speed of light, an interstellar journey will mean the trip several lifetimes–thus, a ship of multiple generations. Furthermore, as manned platforms are currently conceived, spacecraft are designed first and foremost to meet mission specifications, with human factors concerns an appendix. Even some future mastery of the pertinent technological, economic and social hurdles to enable an interstellar voyage would still not produce a starship hospitable to multiple generations of human life.

On an interstellar journey, humans *are* the mission. Therefore, a new approach to large and intermediate-scale design issues ("meso-architecture") must be taken that is dictated first by habitability criteria. Viewing the ship as a self-contained city, an ecosystem in which the ecology is in complete harmony with the human presence, may shift the design approach to producing more of a bioengineered symbiote. Such systems of city architecture were envisioned by Paolo Soleri and others, and are applied here to derive human-driven design constraints for the meso-level design principles of this spaceship. These specifications then suggest solutions to issues as varied as gravity, shielding, life support systems, food production, clustering and separation of human spaces, and accommodation of crew growth.

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Session 5
Space Physical Sciences

ON GIANT EXTRA-SOLAR PLANET FORMATION

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Abstract. Several brown dwarfs and extra-solar planets have been discovered recently. Five confirmed extra-solar planets and three objects which are either planets or brown dwarfs are studied here, all of which circle normal stars, i.e. pulsar planets are excluded. For all of these objects, the orbits are known, i.e. the semi-major axis and (at least) a range in $M \cdot \sin i$ (with mass M and orbital inclination i). If giant planets form around other stars like it is assumed for Jupiter in the solar system, these planets should appear within a certain distance range from the central star. This can be checked using the present data, which is compiled and reviewed in this paper.

1 Introduction

Several brown dwarfs and extra-solar planets have been discovered in recent years. The first such object was detected eight years ago by Latham et al. (1989), who found periodic radial velocity variations for the star HD114762 and concluded that this star is circled by a companion. The mass of this companion is at least 9 M_{jup} (Mazeh et al. 1996), so that it is still unclear whether this object is a giant (Jupiter-like) planet or a brown dwarf. This uncertainty is mainly due to the unknown orbital inclination i, so that one cannot deduce the mass M but only $M \cdot \sin i$ from radial velocity monitoring. More giant planets and brown dwarfs - all circling a primary star - have been discovered using radial velocity monitoring by Duquennoy & Mayor (1991), Tokovinin et al. (1994), Mayor & Queloz (1995), Butler & Marcy (1996), Marcy & Butler (1996), Mazeh et al. (1996), Butler et al. (1997), and Mayor et al. (1997).

A few planets have been discovered by a precise timing analysis of emission from the pulsar PSR $1257+12$ (Wolszczan & Frail 1992, Wolszczan 1994). However, since pulsar planets most certainly form differently than brown dwarfs and planets around main sequence stars, pulsar planets are not considered here. A recent review on brown dwarf and giant extra-solar planet discoveries can be found in Beckwith & Sargent (1996).

Theories on the formation of planets so far rely on data about only one planetary system, our own. To check, whether this system is representative or not, one needs to detect more examples. Other planetary systems

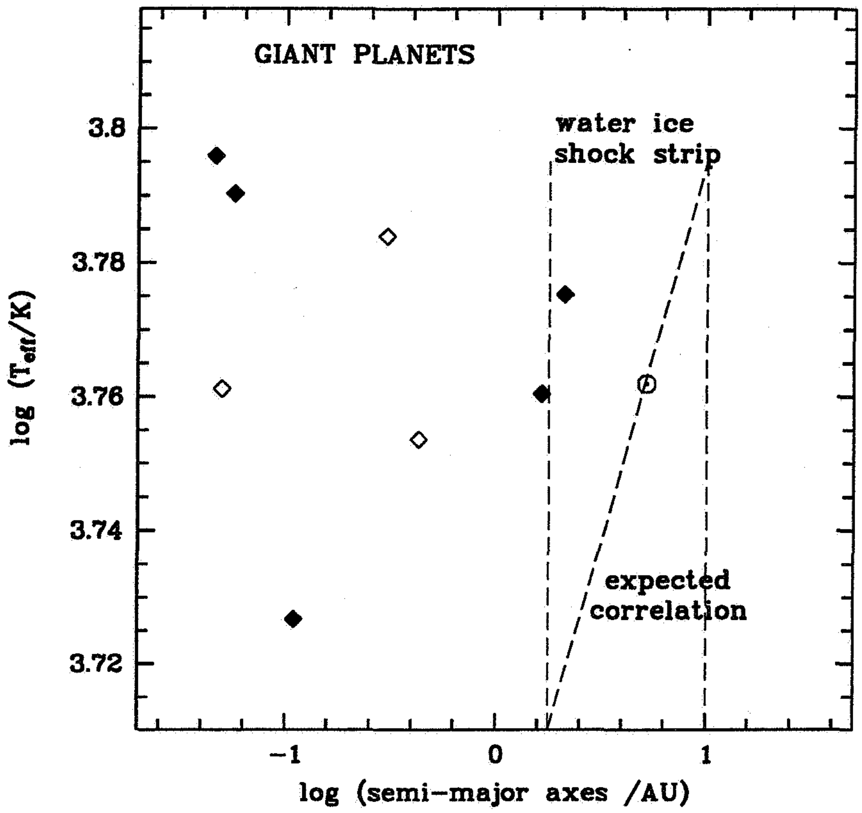

Figure 1: **Giant planets.** Plotted are effective temperature log (T_{eff}/K) of the primary stars versus the semi-major axes (in AU) of the giant extra-solar planets as listed in Table 1: Filled symbols for objects which are certainly less massive than 8 to 13 M_{jup}, the upper mass limit for planets, open symbols for objects which may be either brown dwarfs or giant planets, as well as Jupiter as innermost giant planet in orbit around the Sun ($\odot$). Obviously, there is no correlation present in this data set. Also shown is the strip (broken lines), where giant planet formation is expected to take place, as well as the expected correlation. However, most giant planets are located inwards from this line.

can be expected around stars like the Sun, i.e. around late-type stars. Pre-main sequence stars with spectral type later than mid F are called T Tauri stars and eventually evolve into stars like our Sun. It is known that many classical T Tauri stars are surrounded by circumstellar gas and dust disks (Beckwith et al. 1990), and several such disks have been discovered recently with the Hubble Space Telescope (O'Dell & Wen 1994, McCaughrean & O'Dell 1996). It is widely believed that planets can form in such disks.

Giant planets like Jupiter are widely believed to form in the following way: first, accretion of solid material within the circumstellar disks leads to many planetesimals, some of which grow very fast (run-away growth) to form solid protoplanets (or giant planet cores) with masses of the order of ten Earth masses. Then, these cores can accrete from the gaseous disk to form their atmospheres, to finally become jovian planets. Accretion ceases as soon as the planet's feeding zone is empty, i.e. as soon as a gap in the disk can been created, which also hinders orbital migration of the planet inwards due to friction from the disk material (c.f., Pollack 1984, Lissauer 1987, Wetherill 1990).

2 Discussion

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In the theory of planet formation via planetesimals, it is assumed that the distance between the innermost giant planet and the primary star depends on the temperature profile in the protoplanetary nebula. The innermost giant planet forms at the distance, where water freezes from the gaseous to the ice phase. Inwards to this distance, water is in the gaseous phase. At the distance, where the transition between water ice and gas occurs, the dust density (i.e. the density of condensated material as opposed to gaseous material) in the protoplanetary disk increases sharply, so that there is a so-called water ice condensation shock front (Stevenson & Lunine 1988). This is the distance at which planet accretion via run-away growth is most efficient, so that the innermost giant planet forms here. Jupiter actually orbits the Sun at 5.2 AU, which is perfectly consistent with the location of the water ice condensation shock front given the radial temperature profile in the protosolar nebula (e.g., Boss 1993).

The hotter the star, the hotter the protoplanetary disk i.e. the more distant the cold area (where water freezes), the more distant the water ice condensation shock front. Hence, the innermost giant planet forms further out. So a correlation between the effective temperature of the primary star and the semi-major axis of the innermost giant planet can be expected. Boss (1995) has investigated this relationship quantatively. He found that the distance of the water ice condensation shock front does not vary much with effective temperature (or, similarly, stellar mass) of the primary star: For late-type stars, i.e. spectral types G, K, and M, the water ice condensation shock front lies between $\sim$ 10 and $\sim$ 2 AU (Boss 1995). Again, one should expect a correlation between stellar mass (or effective temperature) and the semi-major axis of the innermost giant planet. A similar relationship between metallicity and the semi-major axis of the innermost giant planet can be expected (Neuhäuser 1992). However, there is neither such a correlation, nor do the extra-solar planets found so far orbit their stars near the water ice condensation shock front, which lies at a few AU. Instead, they are much closer in ($\sim$ 0.05 to $\sim$ 2 AU). The experimental observations are summarised in the Appendix.

According to the most commonly accepted theory of the formation of the solar system, it is believed that terrestrial, i.e. Earth-like planets may be able to form only within the orbit of the innermost giant planet. However, this perception may be biased as we know only our planetary system. If this view is correct, though, one would have to conclude that terrestrial, i.e. habitable planets around other stars may be rare, if most giant planets really are that close to their primary star. However, the currently most successful method for discovering extra-solar planets, the search for radial velocity variations, is biased towards giant planets with very small semi-major axes, because they have the largest impact on the star's orbital motion.

Recently, Lin et al. (1996) and Rasio et al. (1996) as well as Weidenschilling & Marzari (1997) and Rasio & Ford (1996) have argued that giant planets should not necessarily be found in or near the orbit where they were formed. Lin et al. (1996) and Rasio et al. (1996) propose that giant planets can migrate inwards within the protostellar disk. This mechanism can explain why most giant extra-solar planets found so far are very close to their primary star. However, such planets should have circular orbits, but the objects detected around HD 114762, 70 Vir, and 16 Cyg B do not have circular orbits. Weidenschilling & Marzani (1997) and Rasio & Ford (1996) suggest that several (at least three) giant planets form in the outer disk and subsequently interact. One giant planet is ejected out of the system by gravitational scattering, while one other is pushed inwards, and the last one is pushed outwards. Of the two giant planets left in the system, one should be closer to the primary star than the orbit where is formed and one should be further out, while both should have a non-circular orbit, as observed for HD 114762, 70 Vir, and 16 Cyg B.

If large orbital migrations do occur in extra-solar planetary systems, one should not necessarily expect to find a correlation between the effective temperature of the primary star and the semi-major axis of the innermost giant planets, even if giant planets do form near the water ice condensation shock front. The fact that we observe all the hitherto known giant extra-solar planets inwards to the expected strip (i.e. left from the expected correlation as plotted in figure 1) is consistent with them having migrated inwards from the orbit, where they have formed, either by slow migration (e.g. Lin et al. 1996) or by a violent encounter (e.g. Weidenschilling & Marzani 1997).

3 Conclusion

Several extra-solar planets have been discovered by radial velocity variations that they cause on the primary star they orbit. However, almost none of these giant Jupiter-like planets orbit their stars at the water ice condensation point, where they were expected. These expectations, however, were drawn from studying only one planetary system, namely our own. From the currently observed sample of extra-solar giant planets, we can conclude that either most giant planets do not form similarly to Jupiter, or they migrate inwards.

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Appendix

In Table 1, all confirmed extra-solar giant planets are listed. Whether a sub-stellar object is a giant planet or brown dwarf depends on its mass. However, there is no clear consensus yet, on the lower mass limit for brown dwarfs (nor on the upper mass limit for planets). This limit lies between 8 and 13 M_{jup}; c.f., Boss (1996). For all published and confirmed giant extra-solar planets, designation, and effective temperature of the primary stars are given (from Baliunas et al. 1997, Henry et al. 1996, or Simbad) as well as mass ($M \cdot \sin i$, where the orbital inclination i is unknown), semi-major axis, and eccentricity of the objects, as well as remarks and references.

Primary star		Companion				
Designation	T_{eff} [K]	$M \cdot \sin i$ [M_{jup}]	semi-major axis [AU]	orbital eccentricity	rem.	ref.
HD 114762	6080	9	0.3	0.25 ± 0.06	1 $\star$	[14]
51 Peg	5770	0.47	0.05	0.0	2 $\star$	[18]
70 Vir	5670	6.6	0.43	0.4	3 $\star$	[17]
τ Boo	6250	3.87	0.0462	0.018 ± 0.016	4	[8]
υ And	6170	0.68	0.057	0.15 ± 0.04		[8]
47 UMa	5960	2.8	2.11	0.03 ± 0.006	5	[7]
16 Cyg B	5760	1.5	0.6-2.7	0.67	6	[9]
55 Cnc	5330	0.84	0.11	0.051 ± 0.013	7	[8]

Table 1. $\star$ indicates that companion is either a brown dwarf or a giant planet. (1) Probably brown dwarf (instead of giant planet) as inclination i is small (Hale 1995, Mazeh et al. 1996). (2) Inclination i is high (Francois et al. 1996), but less than 85° (Henry et al. 1997); companion may be brown dwarf due to the upper mass limit being 10 M_{jup} (Pravdo et al. 1996). (3) Since the upper mass limit is 38 M_{jup} from HIPPARCOS (Perryman et al. 1996), the companion may be a brown dwarf (Mazeh et al. 1997). (4) Velocity variations first noticed by Duquennoy & Mayor (1991); τ Boo has stellar companion (GJ 527 B) at 240 AU. (5) Upper mass limit is 8 M_{jup} from HIPPARCOS (Perryman et al. 1996). (6) 16 Cyg B has stellar companion (16 Cyg A) at 700 AU distance. (7) 55 Cnc has stellar companion (ρ Cnc B) at 1150 AU distance.

Session 6
Space Policy & Law

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SHOW ME THE TECHNOLOGY: INSTITUTIONAL BARRIERS TO ENVIRONMENTAL TECHNOLOGY TRANSFER

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ABSTRACT

In the environmental arena, both the public and private sectors are exploring technologies, techniques, and information sources which allow them to monitor environmental conditions with fast, comprehensive, and credible methods. However, the technologies are not the only factor limiting application. Institutional issues, such as management practices, agency policies, and legal precedents, pose barriers to the development, transfer and adoption of technologies between the sectors.

The U.S. Environmental Protection Agency (EPA) has recently begun the Advanced Measurement Initiative (AMI) to advance the agency's measurement capabilities, taking into account these institutional barriers. The AMI program sponsors projects that 1) adapt existing technologies (from the public or private sectors) that can meet an identified EPA need, and 2) identify and address institutional barriers to their use and acceptance. AMI projects are primarily focused on fulfilling EPA's measurement needs. However, EPA recognizes that increased capabilities can improve its ability to provide the private sector greater flexibility in regulatory reporting, which, in turn, can increase the search for new monitoring and measurement technologies that the private sector can use and supply.

This paper examines EPA's AMI program in the context of institutional barriers to technology application and implementation. AMI projects provide a broad view of issues to address and approaches to take to resolve institutional barriers. One AMI project, involving the use of remote sensing to characterize a hazardous waste site, is examined in depth. Lessons learned are placed in the context of technology transfer.

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"ALPHA TOWN":
THE INTERNATIONAL SPACE STATION AS A PRECURSOR TO THE FIRST TOWN IN SPACE

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ABSTRACT

As the International Space Station (ISS) project experiences yet another schedule delay, it is becoming clear to many that although this program has been advertised as "the next logical step", it will not really open the door to an expanding frontier in space. In order for ISS to act as a true stepping stone to large-scale space development, the current philosophy of its operations must be changed. Private sector involvement, including the routine operation of the station, is essential in order to greatly increase the pace of space development and to free up government resources for cutting edge R&D and exploration beyond Low Earth Orbit (LEO).

ISS -- THE CURRENT PROBLEM

The International Space Station (ISS) has been delayed again, sliding from an initial planned deployment late this fall to sometime next year. This is not much of a surprise to NASA watchers, as it has already been pushed back some dozen times since the original 1992 completion date that was announced by President Reagan in 1984. Along with each delay (many driven by politics) the cost has also increased, mushrooming from an original tab of $8 billion (US) to the $40 to $100 billion (US) it is running today (depending on which estimate you believe). There have also been so many shifts in its stated purposes as to make the project's current goals almost indefinable. In fact, when pressed, NASA managers now say the main goal of building the station is to learn how to build a space station. This seems strange in light of the legacy of the Skylab, Salyut, and Mir space stations.

So what do we do? Cancel it? No, we could have done so a few years ago, and in fact the Space Frontier Foundation was one of those leading that fight, but now we are too far down the road to do that now. Most of the major elements are already built or close to completion. Tens of thousands of workers around the world have sweated for over a decade to make this thing fly, and the damage to both our old and emerging space industries would be devastating. Finally, to walk away would send a terrible signal to our children and the people of the world about America's ability to lead them across President Clinton's "Bridge to the 21st Century." So, since there is no "lemon law" (like the one that allow us to get our money back from faulty products like cars) that applies to giant government projects, we suggest it's time to start making some lemonade.

The first thing to do is to recall the space agency's mission in our society. If you asked that question of the taxpayers who fund it, they would probably answer that NASA's job is exploration. NASA is today's version of explorers like Lewis and Clark; its job is to blaze new trails, to be pathfinders, to explore, to push back human horizons.

But NASA can no longer afford to do these things, as its budget is increasingly being sucked into the ISS and Space Shuttle programs. Put another way, our proud explorers have been saddled with the job of managing a building in space and driving the delivery trucks to keep it supplied. Thus, exciting news about possible past and present life conditions on Mars and on Jupiter's moon Europa can't be quickly and thoroughly checked out, and there is little or no money to develop new leading-edge space technologies. Meanwhile, its bureaucracy has set up shop just a hundred miles overhead, claimed LEO as its own, and any development and exploitation of the territories it has explored so well in the near Earth domain is stifled. For example, the results released in early 1997 of a 10-month long study by the non-partisan Potomac Institute for Policy Studies were highly critical of NASA's efforts to facilitate commercial space development. It is as if Thomas Jefferson, after hearing the results of Lewis and Clarks' expedition, had decided to turn the American West into a federal reserve, banning all private settlement and development.

So why do we have a crack team of explorers acting as landlords? After all, when stripped of the space mystique, the ISS is merely a combination of port, hotel and research lab in orbit. It will have almost the same categories of costs and overhead as any other such facility on Earth. Under government management and protected from free market forces, one can be sure those costs will be sky high (or higher in this particular case). These costs mean a continuation of the self-fulfilling prophecy of manned space being too expensive for all but deep-pocketed government players. Not only will money that we could be spending on exciting exploration projects such as the search for extra-terrestrial life be going instead to pay space station utility bills, but the high costs will freeze out any who might wish to experiment with new products or carry on scientific research. In the end the ISS, far from being "the next logical step" towards the true opening of the space frontier to thousands of people, would instead be the bar across the door to our future. Historically, there is a fundamental rule about settling new frontiers -- nobody stays until somebody pays. And in our society, it is industry, products and services that write the paychecks.

To make matters worse, the bureaucracy that runs the facility has neither the time nor the inclination to support commercial ventures. If an entrepreneur were to walk into NASA with a billion dollars and try to rent a rack on the wall of the station to house a potentially ground-breaking experiment, they could not do it. There is no system to allow for such activities, because there is no way to put together a fundable business plan based on the ISS operations. No one can tell you how much a kilowatt of power costs, how much an astronaut's time that is needed to run your experiment will cost, or even how to get your materials and products there and back. Most likely, even as they were trying to figure out these basic business questions, NASA managers would be bartering away the needed resources to one of the ISS foreign partners in a government-to-government deal.

ISS -- THE OPPORTUNITY

This must be changed. Running the station is the wrong job for our space agency. Let's face it, such mundane tasks as being a landlord or truck driver are not what an organization like NASA is designed to do. NASA's strengths are as an exploration and advanced research organization, not a construction, trucking and building management firm. Those are jobs that the private sector does, and does very well, in every other environment on Earth, from downtown Manhattan to the extreme conditions of the North Sea.

The way to solve this problem is simple. Indeed, we need look no further than our European partners to see how this can be done for space; Arianespace, while created with the technical and political support of government space agencies and aerospace contractors, focused on the operational needs of their customers to become the world's leading commercial space transportation company. So as soon as the ISS is completed, its management should be handed over to the private sector -- folks who understand things like the bottom line, negotiating contracts, and making money. The space station partners should form an international authority like those used to operate sea or airports. This body will lay out the guidelines for commerce between the station and Earth and within the station between tenants. It will lay out the rules by which the commercial firms will play, enforcing laws on everything from intellectual property rights to arbitration of disputes among users. It should then contract out to a firm or consortium the job of being the station's property managers.

The new commercial landlords will be responsible for everything from signing leases of lab space, fixing and collecting rents, hiring and firing permanent employees (for example, highly paid astronauts are not needed to perform housekeeping, cooking, etc.) and generally managing station operations. All with an eye to pushing costs down and the anathema to government employees -- actually making a profit. The governments who funded and built the facility will of course be the first ones in, acting as anchor tenants, creating an early cash flow, and establishing its credibility as a stable platform for business interests.

To encourage timid business interests to participate in space, tax incentives must be created, just as any town here on Earth puts together special zoning and tax packages in order to get new business to locate in their city limits. In a sense, the space station partners will be acting like the city council in any town hungry for economic growth, new jobs and prosperity. This is a completely different mindset from today's "space is ours and you can watch on TV" attitude. In effect they will be hanging a sign on the docking hatch stating this new place in the sky is open for business. Rather than a financial albatross that weighs down our aspirations in space, the station will become an economic engine that lifts new industries from the Earth out onto the frontier.

In this new commercial environment, firms that have been burned by the space agencies bureaucracy in the past, or those who have heard the horror stories of waiting years to fly, bumped flights or escalating cost estimates, will at last have a stable environment in which to work. A contract will be a contract, a deal a deal, and if someone wants to work on a new potential wonder drug or process in secret, they will be protected by the same laws and contracts we enforce everyday on Earth. Old ideas, like developing ultra high speed electronic components for computers to formulating new medicines and treatments, will be pulled out of the dead file and dusted off. New and untried or undeveloped ideas, like creating light-weight high-strength materials such as foam steel, or making new products like self-lubricating ball bearings alloyed from titanium and lead, can at last be tried. And wild cards we can't even imagine will be pulled from the deck of entrepreneurial ingenuity, as always happens when the game is not stacked, the rules are clear and the game open to all comers.

Using the station as an economic center as well as a research lab benefits everyone. Scientists, critical of the station's high costs and ill defined support capabilities, will benefit from the stability of a well-run building in which to work. They will be able to do much more research with less money, as they will in effect be partially subsidized by the commercial firm's rent payments.

"ALPHA TOWN" -- THE SOLUTION

"Alpha Town" is our proposal to use the power of the free market to revitalize the ISS program. The concept is not about hardware. It is about a mind set. The details are irrelevant. The particular technologies do not matter. It is not a destination or a facility, it is an intellectual framework around which we build the dream of human settlement on the frontier. As opposed to a boring Antarctic style facility on the edge of nowhere and going nowhere, it will be a symbol of hope, the seed corn of a new civilization. But to enable its creation, there are four main principles that must be adhered to.

It will obviously be in the interests of all concerned to bring down the costs of servicing tenants and getting customers' supplies and products to and from orbit, just as it is with any new commercial development here on terra firma. Therefore, Alpha Town Principle One is that after completion of the assembly phase, all U.S. government transportation needs to and from Alpha will be competitively bid for by U.S. private firms. It is hoped that the European partners will also contract their needs out to commercial firms.

National rockets like the government-run and operated Space Shuttles must become a thing of the past, and be replaced by privately run rocket fleets. Considered by many to be the key ingredient to successfully opening space to human settlement, Cheap Access To Space (or CATS, as coined by the Space Frontier Foundation) will come as a natural result of free market forces as launch companies compete to carry various payloads to and from this new commercial nexus in space. Additional payload space can then be sold to companies wishing to set up their own facilities or to paying passengers, such as academics, commercial scientists and even tourists.

Many potential tenants of space stations, such as astronomers or those engaged in delicate crystal growth experiments, need extremely stable platforms with which to work. For them, the comings and goings of a bustling space facility will be unacceptable. Biologists working on easily contaminated experiments or creating vaccines for easily spread diseases may wish for isolated facilities of their own. Again, in this new economic based model for space station growth, market forces will determine and drive new alternatives. To facilitate this, Alpha Town Principle Two and Three are: all expansion of habitable physical structure of the U.S. portion of ISS will be commercially leased from U.S. private firms; and the government will put in place legislation to actively encourage and regulate the development and recycling of all space assets.

With these principles in place, other facilities will spring up near ISS in order to serve the divergent needs of the various users. For example, Russia's old Mir station, which under current plans might end up as orbital junk, will be prime "adjacent" real estate. And the giant 17 story tall Space Shuttle External Tanks that are now dumped into the Indian Ocean can be converted into new "buildings." Once seen as a potential threat to the station's funding by some NASA managers, commercial firms wishing to convert these government surplus assets into new real estate will be encouraged to do so by tax and investment incentives in the orbital space enterprise zone, as precedents set on the station begin to spread beyond its airlocks.

As transportation costs drop and "space" in space becomes available, the first orbital hotels will be constructed. It may sound like pie in the sky right now, but tourism is one of the largest industries on Earth, and there are entire nations on this planet whose economies are based on the tourist dollar. NASA and industry funded research has shown that when the cost of getting into space drops from today's' $5,000 a pound to about a $100 per pound (the long term goal of the commercial follow on to the NASA/Lockheed Martin X-33 rocket program) there are people who will pay for the ride, the same ones who now sometimes pay hundreds of thousands of dollars for global "adventure" tours.

We have the land; we have a transportation system; now we need energy. The need for electricity in space will be a major show stopper early on. As the ISS quickly becomes covered with gossamer sails of solar cells, it will become a space pilot's nightmare to dock with the station without blowing away a few million dollars worth of equipment. At the same time, each and every other facility in space will need considerable power. But large arrays of solar panels in LEO also increase a facility's drag, requiring expenditures of expensive station-keeping propellants. These needs converge nicely with the need to explore and develop what may become the space equivalent of an oil strike -- space generated power distributed to users with Wireless Power Transmission (WPT). Alpha Town Principle Four is that all additional energy requirements aboard the ISS after assembly is completed will be supplied by commercial vendors, preferably off-site.

With the ISS as first a test bed in a two ended system, then as a customer, a well-placed constellation of free flying or tethered satellites in higher orbits that can capture solar energy and then use WPT via microwaves or lasers (lets call them SolSats) will remove the need to further encumber facilities onboard the ISS. Interestingly, there are many proposals for deep space missions to fly on beamed power, which can be produced at the same facility. Thus, one exciting possible customer will be exploration spacecraft powered by beamed energy technologies.

And as the other LEO projects come on line, they can also begin to buy electricity from the SolSat. They will save millions of investor dollars on structural enhancements, complicated docking maneuvers and station-keeping as the technology matures in the the space environment. Once the technology becomes routine and the commercial operators of the SolSats develop expertise and credibility, they may well find profitable markets to sell the first beamed energy to Earth. But obviously, planning must begin now to use the ISS as the test bed and proving ground for these technologies and to foster the development of a diverse space energy industry.

As each new market sector is created in and around the station, economic forces go to work and costs begin to come down. Soon will come the long awaited chance for average citizens like you and I and our children to go there ourselves. Before long, we will see the birth of the first true town in space. "Alpha Town" -- the "First Town" in space will be born.

SUMMARY

Imagine, just as we enter the 21st century, the intrepid human and robotic explorers of NASA will be freed from going in endless circles around the Earth, and will once again push out "to go where no one has gone before." Meanwhile, the shopkeepers, business operators, and settlers who invariably follow such explorations will move out into near Earth space and begin to build there a new human domain.

If one looks to history, each time that careful exploration has been followed by the might of the free enterprise machine, miracles have happened that far exceeded even the wildest dreams of those initiating the quest. We in the Space Frontier believe that we must find a way to unleash those forces in the ISS program. For if this model is adopted, it is quite possible that in your lifetime humanity will begin the irreversible settlement of space. And in no more than a few decades hence, stretched out above us in the night for all on Earth to see, a string of tiny pearls of light representing humanity's first community in the sky will remind a pessimistic world that the greatest age in human history has just begun.

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SPACE & THE FRONTIER MYTHOS: A RE-EXAMINATION

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Abstract

Some modern space philosophies emphasize the historical parallels between the conquest of the Martian frontier and the expansion into the American West. These parallels are used as a rationale for planning the exploration and large-scale settlement of Mars. However, many of these historical allegories are in error. Vast differences exist between the geographical realities of Mars and the western frontier. Furthermore, westward expansion took place within a set of political, economic, social, and technological circumstances that are unlikely to be repeated in Martian exploration. The physical conditions and geographical realities of the Martian frontier in exploration and settlement terms are closer to the Antarctic continent. The American West, which had been inhabited by humans for thousands of years, offered a relative degree of comfort to new arrivals. We do not find large numbers of people migrating to the Canadian High Arctic or other polar regions to live. The inhospitable nature of these polar regions are the major limitation to the human desire to live there. The conquest of the American West took place within the context of a complex set of circumstances. Americans believed they had a "manifest destiny" to extend their civilization across the continent. Many people also went west in search of a better life: for gold, land or other opportunities. The technology also existed at that point to allow large numbers of people to head west on their own at minimal cost. The settlement of Mars will be far different. It is for these reasons that we believe Antarctica is a much better analog for the exploration of Mars. Therefore, Martian exploration and settlement plans should be adjusted accordingly.

Introduction

In the early 1960's, a Hollywood producer named Gene Roddenberry was trying to sell a concept for a new television series called "Star Trek." Seeking to convince skeptical network executives (who were not well disposed toward science fiction), he pitched the series as a sort of "Wagon Train to the Stars." The analogy invoked the rugged pioneers who had crossed the American plains in covered wagons a century earlier.

Roddenberry was probably hoping to capitalize on the popularity of Westerns that dominated American television and cinema during that era. Romantic images of rugged cowboys, frontier towns, and brave cavalrymen occupy a special place in American culture. It is not surprising that Americans in the 1950's and 1960's—beset by rapid urbanization, dizzying technological progress, and Cold War concerns—would hearken back to an earlier, supposedly simpler era.

But there was another aspect to the pitch that was attractive. Roddenberry accurately sensed the view that many Americans held concerning the Space Age, opened only a few years earlier by the launch of Sputnik. Many people came to view space as the new American West, a place where we would relive the glories of past exploration and conquest while elevating human civilization to a new level. This view is still very strongly held by many today.

The fact that space colonies and human voyages to Mars are still distant dreams is seen by many space enthusiasts as a failure of will and vision, akin to Columbus' having turned back after discovering the New World, or America failing to follow up on the discoveries of the Lewis and Clark Expedition. However, the relatively slow progress of space exploration to date may imply something else: that the Frontier Mythos is in need of serious re-evaluation.

The successful exploration and colonization of a frontier is dependent upon many variables: the physical conditions of the frontier; the level of technology; cultural norms; political imperatives; economic needs; and other constraints. Many proponents of the Frontier Mythos have distorted the historical record regarding the conquest of the American frontier. They have also greatly underestimated the differences between the American West and outer space. What made sense for our ancestors in exploring and settling the Americas may not have much relevance to the exploration and colonization of the space frontier.

Frontier Myths

The romantic image of the American West many space enthusiasts passionately evoke is itself an egregious and dangerous misconception. Many false comparisons have been made between the westward expansion and space exploration; in fact, as the discussion below will illustrate, these comparisons may have seriously inhibited the advancement of space exploration.

The moons and planets of our solar system are more extreme and dangerous environments than anything on the surface of the Earth, including the American frontier of centuries past. European colonists settling the Americas may have encountered severe weather and other hardships, but they never had to worry about running out of air, and could usually find fresh water and food. Eventually these and other indispensable commodities could be manufactured in space or on the surfaces of other planets, but for the foreseeable future, "colonies" on other worlds will be sealed, confined, and completely artificial environments, providing living conditions utterly unlike anything that has been previously experienced by pioneers.

Many space advocates claim the effort at meeting the challenges of space exploration will bring great benefits to humanity. They credit the frontier experience for spurring both advances in freedom (through the American Revolution) and unparalleled wealth and opportunity through unregulated capitalist economic policies. They claim that exploring and colonizing space will stimulate a renewal of the frontier values-resourcefulness, individual initiative, and technical innovation-that allegedly made America a wealthy and powerful nation by the end of the 19th Century.

These arguments, however, are grossly oversimplified. The Founding Fathers were certainly liberated by the relative freedoms offered by the New World, which provided social and political opportunities unavailable in aristocratic Europe. But they were also influenced by stay-at-home European philosophers such as John Locke, and by the British Parliament and colonial legislatures, as models for a new form of government.

The idea that the challenge of surviving on the American frontier motivated the development of new technologies and the formation of the "rugged individualist" personality is clearly exaggerated. The founders of the first New England colonies were undoubtedly brave and hearty individuals. But the early European colonists were also highly dependent on native populations for their survival. American Indians helped clear their land for agriculture, and taught the Europeans farming techniques as well as other critical skills; many colonists noted in their journals that the colonies wouldn't have survived, let alone flourished, without the help of native Americans. Admitting this is often difficult because acknowledging the need for assistance in some way diminishes the accomplishments of the early colonists.

Sadly, the American Frontier experience didn't guarantee democracy or equal economic opportunity for Indians, women, or people of color. Much of the wealth generated by the United States in earlier centuries came about through the exploitation of slave labor coupled with a diverse and plentiful endowment of natural resources. The fight for equality continued long after the frontier was settled. Women couldn't vote until 1920, and it wasn't until 1965 that black people in the South, for all practical purposes, could exercise this basic right. The fight for a just and equitable society for all Americans still continues today.

A serious fallacy invoked repeatedly in the Final Frontier argument is that correlation implies causation. It is remarkable that scientists and engineers, rigorously trained in scientific reasoning, should continue to make this fundamental mistake in logic. Space-as-frontier promoters often point out that China had a marvelous fleet of sailing vessels in the fifteenth century and explored much of the globe, only to retreat into isolation at the beginning of the Ming dynasty, in the belief that the outside world had nothing to teach Chinese science and culture. As a consequence of this decision, so the argument goes, China ceased to be a world power and began a long decline into technical backwardness and political oppression.

But, of course, correlation does not imply causation. China didn't necessarily "decline" because it stopped exploring; technical innovation and political freedom in China may have declined for reasons that had nothing to do with the decision of the Ming emperor to turn back from world exploration. It could easily have been otherwise. There is no logical reason to assume that a country that does not expand its geographic horizons cannot flourish intellectually, scientifically, and culturally. In fact, most of the technical innovations and social advances democratic societies take for granted today came into being long after the last frontiers of Earth had been explored. The truth is that most technical and scientific advances—including those that led to spaceflight—have been forged by the imperatives of warfare and the desire to develop weapons superior to those available to enemy nation states.

The Antarctic Analog

Antarctica serves as a far better analog for future space exploration than the American West. Early Antarctic explorers were not unlike the first space explorers: they went there to prove they could. The first trips to the South Pole were tests of skill, endurance, and courage that paved the way for later expeditions.

Today, the Antarctic continent is dotted with research bases housing scientists who conduct a wide range of investigations. An increasing number of space scientists also work in the Antarctic; the harsh conditions there are in many ways analogous to the space environment. The comparison between the south pole and Mars is particularly striking in terms of topography and climate. For example, the average temperature at the equator of Mars is -30°- to -50°-C, with daytime temperatures barely reaching the freezing point even in mid-summer. In some parts of the Antarctic, temperatures can dip to as low as -80° to -90° Celsius. The major difference is that Mars has a thin, poisonous atmosphere that requires humans to wear pressure suits.

During the brief Antarctic summer, the U.S. McMurdo Base, the largest facility on the continent, is superficially similar to a 19th century American frontier town. At peak occupancy, up to 1,000 people live on the crowded base in weather-beaten buildings. The community offers comfortable if cramped living space, complete with a bar, movie theatre, and dance hall. A pioneer from the 19th century might well feel at home there.

There are, however, several crucial differences. One is that day-to-day life is tightly controlled at the south pole. People do not leave the base without substantial logistical support; to do so would risk serious injury or death. Conditions on the American frontier were vastly better.

Antarctic settlements are concentrated into a few main areas, with no development on the periphery of established bases. In the pattern of American settlement, towns and cities were typically established on waterways, forming ports that handled the shipment of crops, raw materials, and finished goods. Farms frequently surrounded cities.

One of the most powerful magnets for the settlement of the Americas was the ability to own and farm land. Emigrants and their families could escape economic problems and political oppression at home and stake their claim on a new continent. This encouraged the migration of Europeans to America during the colonial period and helped drive westward expansion during the 19th Century.

Families: The Key to Community

There are no families at the McMurdo base, and few married couples. The Antarctic Treaty preserves the vast continent for scientific research, forestalling any national land claims. Because of the treaty and various limitations imposed by national research programs, few people stay much beyond 18 months at the pole before heading home to warmer climates. Early American settlers usually brought their wives and children with them (if not immediately, eventually). They established towns and cities where families resided for many generations, providing the stability needed for a community to thrive. Without such stability, it would have been impossible to develop a permanent culture and establish social norms.

Even without these legal restrictions, it is difficult to image anyone wanting to stay in Antarctica longer. Rarely among even the most extreme "loners" do you find a desire to build a house and raise a family in this harsh climate.

Certainly there are very few people living in the far northern latitudes of Canada, where there are no restrictions against building communities. People choose where to live based on a variety of factors: proximity to work, housing quality, safety, recreational facilities, quality of services, educational opportunities, and overall quality of life. Standards may be different depending upon social circumstances; single people might be willing to tolerate more inconveniences than a married couple with children.

Antarctica offers few amenities. As summer at McMurdo Base comes to an end, workers batten down the buildings and most people on the base clear out. Of the 1,000 summer residents, fewer than 100 will ride out the six-month-long cold and darkness of the Antarctic winter.

It is very likely that similar limitations will prevail on space settlements. Settlers' lives will be tightly controlled, less for political purposes than for reasons of safety. Space colonists will live in enclosed, completely artificial habitats. A walk outside—whether in space or on the lunar or Martian plains—will be a hazardous undertaking. No one will be able to own and develop land in the way we can on Earth.

Space colonists will also find the confinement emotionally difficult. Those returning after only a month at McMurdo have found themselves startled by the contrast between the sensory deprivation they experienced in a quiet polar world—dominated by white snow, black rocks, and perpetually blue skies—and the broad spectrum of stimulation we take for granted in the non-polar world. Stepping off the transport plane at Christ Church, New Zealand, Antarctic workers are frequently overwhelmed by the brilliant colors, loud sounds, and sweet smells of a diverse biosphere. The same will almost certainly be true of returnees from colonies in Earth orbit, on the moon, or Mars.

Much immigration to the Americas was motivated by the possibility of a better life in the new world. Immigrants moved to America for better work opportunities, or to escape persecution and poverty. Extraterrestrial bases will no doubt grow in sophistication and quality over time, but it will be a very long time before they come close to matching the simplest amenities of the home planet. The lack of demand for moving off the Earth will in turn depress the space housing market, limiting innovations needed to make the frontier a better place to live.

Space might eventually look more attractive if political, environmental, and economic conditions on Earth deteriorate substantially. However, that outcome is totally within humanity's power to prevent. If we properly manage population growth and the use of natural resources, there is no reason why the Earth's biosphere cannot be maintained in a good state in perpetuity. The Frontier Mythos of simply starting over again in a new place may promote a "disposable Earth" philosophy that could prove to be very costly in the long run.

The Tourism Model

In addition to scientists, Antarctica is increasingly becoming a destination for another group of adventurers: tourists. A growing number of people are spending upwards to $50,000 in order to visit the remote region. The exotic nature of the Antarctic appeals to wealthy individuals seeking an alternative to the usual vacation spots. The increased interest in visiting the south pole corresponds with the increasing popularity of vacations to rain forests, savannas and other remote sites.

Many people believe that space holds a similar appeal to tourists. They foresee a day, perhaps in the early 21st century, when tourists will vacation at orbiting hotels or visit the historic Apollo 11 landing site where Neil Armstrong took his giant leap for mankind. Advocates believe that the establishment of inexpensive space transportation, which NASA is now attempting to develop with its X-33 program, will spur a thriving tourism industry.

This may be a reasonable prospect, but the question remains whether space tourism will in turn lead to large-scale migration of people into space. Space may be a better place to visit than to live.

Industrial Applications

Many space enthusiasts believe that the coming era of inexpensive transportation will lead not only to tourism but a thriving commercial applications sector. Advocates believe these applications will include mining the moon and asteroids, manufacturing super-pure drugs and crystals in orbiting facilities, and the construction of solar power satellites to beam unlimited energy down to Earth.

These industries are expected to stimulate colonization as workers move to fill jobs in these new industries. The raw materials will be processed into materials for constructing large-scale settlements for the new arrivals. Comparisons can be drawn to mining and cattle towns that sprang up on the American frontier during westward expansion.

But industrialization and resource extraction during previous centuries was very labor intensive. These same activities on the space frontier may well be highly automated, requiring only limited human involvement. It is possible that space will thrive as a commercial arena without the need for supporting frontier communities.

There are several reasons to believe this is likely. To maximize profits, space industrialists will want to limit the number of humans for whom they will have to provide life support systems, water, food, and other expensive consumables. Advances in robotics and tele-operations—tools unavailable in previous centuries—will make automation much more cost effective and reliable by the time we are ready to begin extraterrestrial mining. Finally, some high-technology applications—such as pharmaceutical and crystal production—are best done without humans, whose presence tend to disrupt microgravity manufacturing.

Conclusion

It is not only intellectually dishonest but potentially damaging to the cause of space exploration to perpetuate a false picture of life on the American frontier. Not out of any sense of misplaced guilt over abuses that took place a hundred or more years ago, but simply out of respect for the truth. The modern public is too aware, and perhaps even too cynical, to buy into the "Ben Cartwright" model of the American frontier. Space advocates have tried selling the American public on a more aggressive space program by spouting mythical frontier rhetoric for over thirty years, and it still hasn't gotten humans beyond lunar orbit. It's time to abandon this losing public relations strategy.

Space enthusiasts should concentrate their advocacy on the intrinsic near-term and long-term value of space technology and scientific exploration. Regarding the long-term prospects for space colonization, what will be needed at the very least is a new conception of the frontier experience, cognizant of the realities of spaceflight and the history of the American frontier, not misguided romantic visions of a past that never existed.

In his recent PBS series on the American West, writer/producer Ken Burns observed that the true lesson of the frontier is this: there is a price to be paid for progress, and not every one can win. This is perhaps the one aspect of the American frontier experience that will likely be as true in space as it has been on Earth.

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ARCHITECTING A VISION FOR THE UN OF THE 21st CENTURY : PROTECTING HUMANITY AGAINST A ANNIHILATION THREAT FROM OUTER SPACE

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Since the unleashing of the power of the atom, the governments of the United States and the former Soviet Union have spent truly staggering amounts of man power and resources to build and maintain a mutually assured nuclear annihilation capability. The environmental damage associated with this build-up borders on the incalculable. Ironically, escalation of this and related activity to politically and economically unsustainable levels has forced radical reform in bilateral-multilateral policies and affected global strategic outlook among nations, resulting in an interim peace that prevails today.

The demise of the cold war has brought about the dismantling of this nuclear capability and both countries have now decided, at least in principle, to destroy a large number of strategic and tactical nuclear war heads. The debate now is about how to go about it; burning them up in the core of nuclear reactors or deliberately contaminating the fissile material and embedding them in large monolithic glass blocks in a process called vitrification.

Barring the possibility of a sea change in the policy between nations in these turbulent times, if either of these schemes ever come about in the stipulated two decade period, both options will eliminate these weapons from the theater of engagement for some time.

In this context, we need to explore yet another option. Using prevailing statistical models, however small and insignificant as we may now perceive it to be, consider the possibility of a natural extraterrestrial threat. A comet, rogue asteroid or another large object on a collision course with the Earth. These existing nuclear weapons may be targeted at such an object, if ever the need arises. The combined arsenals of the US and Russia, deployed in a skillfully choreographed campaign, might just be sufficient to avert total annihilation of civilization and life as we know it today.

Fossil records show that such cataclysmic activity have punctuated the Earth's history with some regularity, wiping out entire life forms on the planet and setting biological evolutionary processes back to square one. Scholars differ on when such a terrible occurrence might happen again or the chain reactions and resonant mechanisms that would come into play thereafter, possibly amplifying the calamity several fold, but they are all agreed that we can expect it to happen again, with devastating consequences. The SL-9 series of impacts on our neighboring giant Jupiter a short while ago, for instance, could have triggered such activity, if the Earth had been the target.

How might we construct such a planetary defense system architecture ? While the window of opportunity still exists, let us swiftly move to create a body within/akin to the UN to safely and securely transfer both warheads and launch vehicles from Russia and the US to a mutually acceptable site. Leave the door open for other nations with a nuclear deterrent capability to contribute. Using existing missile silo technology, coordinate and build a few UN zoned facilities around the globe to service and protect this capability. Provide funds to maintain this capability, including resources for bolstering our early warning sky survey capability to search out, characterize and track rogue asteroids and comets with dead target precision.

Using modular, strap-on boosters and stretched tankage if needed, augment the performance of existing ICBMs to achieve interplanetary targeting capacity so that we might engage the potential adversary at a safe distance from the Earth. The further out we are able to intercept the target, the more choices we have in strategic and tactical engagement, including gently nudging the aggressor off collision trajectory. Adapt the MIRV architecture to respond in real time to deep space re-targeting needs, so as to be able to lock on to and go after secondary and tertiary scatter targets arising from the initial encounter. State -of-the-art autonomous guidance and navigation routines as well as rapidly evolving, powerful neural network algorithms would be employed for complex maneuverability. Longer flight times in a deep space environment may entail some other design modifications as well. Though it is desirable to have a dedicated TT&C infrastructure, existing worldwide assets could be networked using high bandwidth links, for the purpose. These are just some of the "tip-of-the-iceberg" system aspects of such an architecture.

Once such a planetary defense system is built, how do we make sure that the system is operating according to specifications ? What might we use for target practice ? We could do trial runs of varying magnitude ranging from collision avoidance tactics to complete annihilation on minor bodies along the asteroid belt and on highly eccentric bodies entering the inner solar system, and in the process tweak the architecture for peak performance. Eventually, as the system evolves, the heavy plutonium warheads might be replaced with more compact yet vastly superior annihilators including anti-matter devices.

What are some of the possible benefits of such an action ? First of all, we would have found another way to use the already existing, horrendously expensive and deadly weapons of mass destruction for a globally beneficial purpose. We would have initiated a capability for a planetary defense mechanism against a natural threat for a fraction of the cost that it might otherwise entail. Such an action could potentially become a nucleus upon which to create a World Space Organization that would eventually coordinate and extend other peaceful space faring activities to all nations. A program of this scope and complexity will also surely push the envelope and test our mettle, as a species, for collective effort and organization.

Are we ready for this kind of an undertaking yet ? Are nation states willing to relinquish control of nuclear assets to an international organization ? Could we, as a species, muster the political will and the resources to create and operate such a system ? That is for the visionary leaders of our nations to decide. The US Air Force along with some think tanks and universities are already contemplating strategies to cope with this sort of an event. In any case, we can be certain that there are things out there that can wreak unimaginable havoc on Earth without the help of wicked aliens depicted in movies like ID4.

However unlikely, insignificant or irrelevant it may be in the course of time, events and history, I suspect that such a planetary defense system architecture will still stand as a unique twentieth/twenty first century testament to humanity's collective approval and decision to preserve itself against a perfectly natural and thoroughly devastating alien threat.

As we approach the new millennium, we live in the most unusual times. The post cold war era offers opportunities that were not imaginable but a decade ago. Current events in the neighborhood provide proof positive that extraterrestrial cataclysmic agents are indeed at work, possibly at a rate higher than anticipated, and statistical astronomers are scrambling to rebuild their models of such activity.

Finally, it is not often that a civilization finds itself pondering what to do with a highly potent stockpile of thousands of "surplus" nuclear missiles. After all, are these not the very same warheads that won us the peace ? Do we simply destroy them or could we reassign them to a noble humanitarian mission; one dedicated to the preservation of our species and life on Earth as we know it ? The window of opportunity may not last long. So, let us seize the moment !

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Session 7
Space Life Sciences

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P1

THE EFFECTS OF LIQUID COOLING GARMENTS ON POST-SPACE FLIGHT ORTHOSTATIC INTOLERANCE

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Post space flight orthostatic intolerance among Space Shuttle crew members following exposure to extended periods of microgravity has been of significant concern to the safety of the shuttle program. Following the Challenger accident, flight crews were required to wear launch and entry suits (LES). It was noted that overall, there appeared to be a higher degree of orthostatic intolerance among the post-Challenger crews (approaching 30%). It was hypothesized that the increased heat load incurred when wearing the LES, contributed to an increased degree of orthostatic intolerance, possibly mediated through increased peripheral vasodilatation triggered by the heat load. The use of liquid cooling garments (LCG) beneath the launch and entry suits was gradually implemented among flight crews in an attempt to decrease heat load, increase crew comfort, and hopefully improve orthostatic tolerance during reentry and landing. The hypothesis that the use of the LCG during reentry and landing would decrease the degree of orthostasis has not been previously tested. Operational stand-tests were performed pre and post flight to assess cremember's cardiovascular system's ability to respond to gravitational stress. Stand test and debrief information were collected and databased for 27 space shuttle missions. 63 crewpersons wearing the LCG, and 70 crewpersons not wearing the LCG were entered into the database for analysis. Of 17 crewmembers who exhibited pre-syncopal symptoms at the R+0 analysis, 15 were not wearing the LCG. This corresponds to a 21% rate of postflight orthostatic intolerance among those without the LCG, and a 3% rate for those wearing LCG. There were differences in these individual's average post-flight maximal systolic blood pressure, and lower minimal Systolic Blood pressures in those without LCG. Though other factors, such as type of fluid loading, and exercise have improved concurrently with LCG introduction, from this data analysis, it appears that LCG usage provided a significant degree of protection from post-flight orthostatic intolerance.

519-61
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040 731

USING COMPUTER SIMULATION FOR NEUROLAB 2 MISSION PLANNING

289979
p1

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This paper presents an overview of the procedure used in the creation of a computer simulation video generated by the Graphics Research and Analysis Facility at NASA/Johnson Space Center. The simulation was preceded by an analysis of anthropometric characteristics of crew members and workspace requirements for 13 experiments to be conducted on Neurolab 2 which is dedicated to neuroscience and behavioral research. Neurolab 2 is being carried out as a partnership among national domestic research institutes and international space agencies. The video is a tour of the Spacelab module as it will be configured for STS-90, scheduled for launch in the spring of 1998, and identifies experiments that can be conducted in parallel during that mission. Therefore, this paper will also address methods for using computer modeling to facility the mission planning activity.

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Session 7
Space Informatics

S20-61
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p1

EFFICIENT ALGORITHMS FOR ONTOLOGICAL HIERARCHY

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Many practical applications of Intelligent Information Retrieval call for the greater use of ontologies. However, various levels of conceptual complexity of Ontology, combined with LISP-based implementations, make this use computationally prohibitive. In the paper, I would like to present preliminary results about C-based efficient data structures and algorithms for Taxonomic Concepts Hierarchy representation. Examples of usage of the algorithms on a variety of domains confirm preservation of rich formalism and generality. The comparison of the algorithm's performance with standard C string manipulation library functions is provided.

S21-61
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WEB-BASED COLLABORATIVE PUBLICATIONS SYSTEM: R&TSERVE

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P1

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R&Tserve is a publications system based on "commercial, off-the-shelf" (COTS) software that provides a persistent, collaborative workspace for authors and editors to support the entire publication development process from initial submission, through iterative editing in a hierarchical approval structure, and on to "publication" on the WWW. It requires no specific knowledge of the WWW (beyond basic use) or HyperText Markup Language (HTML). Graphics and URLs are automatically supported. The system includes a transaction archive, a comments utility, help functionality, automated graphics conversion, automated table generation, and an email-based notification system. It may be configured and administered via the WWW and can support publications ranging from single page documents to multiple-volume "tomes".

OMIT THIS
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Session 8
Space Informatics

522-32
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VIRTUAL COMMUNITIES: MODELS FOR FUTURE SUCCESS

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"Virtual communities" is a term used to describe groups of people who have aggregated around a topical area, rather than a physical locale, in a wide variety of communications media. Traditionally, such aggregations have been somewhat haphazard, offering short-term benefits lacking direction and long-term goals. "Planned" virtual communities, on the other hand, provide opportunities for their organizers to define a direction for the community and to expand both their "market share" and "mind share" in the larger global society, to the success and benefit of the organizers, as well as the community. The concepts of the virtual community business model are developed and application to academic organizations such as the International Space University is explored.

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Session 8

Space Business & Management

S23-81

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The Reorganization of the Canadian Space Agency: a Managerial Insight

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On September 7, 1995, the President of the Canadian Space Agency (CSA), Mr. W. M. (Mac) Evans officially launched the organization's renewal process.

Through a 18 to 24-month reorganization process, the CSA would give itself the tools and structure to facilitate strategic planning of the Canadian Space Program's future, instill a corporate approach to delivering its mandate and programs, ensure broader participation and more transparency in the decision making process and improve the delivery of the essential horizontal support functions.

The new organization would be based on generic functions as opposed to specific programs. This would bring an entirely different approach for selecting and implementing programs and projects. Consistent with the principle of empowerment and the need to be more closely linked to the requirements of the clients, the new organization would also provide for delivering services as close to the client as possible, while maintaining a centralized approach to setting policies and standard practices.

To orchestrate the reorganization work and to deal with the many issues that would arise from such a phased approach, a Transition Team was created. Under the leadership of the vice-president of Corporate Development, Dr. G. M. Lindberg, the Transition Team would plan and manage the activities required to implement the transformation of the Agency. The approach they would propose should provide for the most efficient completion of current major program commitments while allowing a smooth transition to a more flexible structure capable of managing the more varied and complex new activities of the Canadian Space Program.

Using the momentum of the Mission Statement Exercise, employees would be involved in further defining the organizational structure.

D. Bourque (CAN '92) will talk about his experience as a member of the Transition Team and will present the reorganization process that was implemented during the last 18 months in order to give birth to the New Canadian Space Agency.

S2 4-15
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040 736

ARIANE 5: LEARNING FROM FLIGHT A501 AND PREPARING A502

289992
p1

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After the failure of the first test flight of the European Ariane 5 launcher on June 4 1996, an independent Inquiry Board was set up, which presented its findings and recommendations in July. The cause of the failure was identified as due to specification and design errors in the software of the Inertial Reference System. Since then, the Launcher Qualification Board and its specialised audit groups, together with the ESA/CNES Ariane 5 Project team and the industrial contractors are implementing a comprehensive "Post-A501 Action Plan" which includes not only action following the Inquiry's recommendations but also additional verification and analysis on all the launcher's systems. This additional work, which does not call into question the design of the launcher nor its flight readiness, is intended to improve its robustness, increase the operational margins and allow for degraded operating modes. The Action Plan leads to a second test flight in September 1997 and a third one five months later. Commercial operation of Ariane 5 by Arianespace should begin in 1998.

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Session 9
Satellite Applications

S25-63
040737

AUTONOMOUS SPACECRAFT NAVIGATION USING GROUND LANDMARKS

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ABSTRACT

In this report a promising technique of autonomous navigation is considered. It is based on observation of the Earth's surface from the satellite, detection and recognition of landmarks on the acquired image and their use for more accurate estimation of the spacecraft position. Main principles of operation of navigational systems using the considered technique are presented along with various possible configurations of on-board equipment. The navigational problem is conventionally divided into the problem of image processing and the problem of processing of angular measurements based on results of image processing. The latter problem is considered in detail; accuracy of navigation is studied taking into account various factors that influence opportunity to perform measurements and their accuracy. Results of computer simulation are presented which prove the opportunity to implement the considered navigation technique with accuracy of positioning better than 1 km.

INTRODUCTION

In operational Earth monitoring systems the on-board equipment is not used efficiently enough. The obtained images of the Earth's surface are analyzed after they are compressed and downlinked to the Earth. The main defect of such an approach is the low efficiency of decision making based on the obtained images. For example, if an anomaly is detected on a low resolution image, a high resolution image may take several hours or days to obtain. Besides, determining the position of detected objects is possible only if the obtained image can be tied to the reference image. This is impossible when the surface has low information density, such as water or desert. One more defect of this approach is the enormous required bandwidth of the satellite-earth communication channel combined with the low quality of the sent data. Many of the images will be rejected on Earth due to excessive amount of clouds or total lack of information.

But the most interesting fact in modern Earth observation systems that is not used is the inherent information in the images about position of the spacecraft. The modern on-board equipment can process the image and get the information not only about the observed area and objects on it, but also about the position of the spacecraft. In other words, quite a new method of autonomous satellite navigation, namely navigation based on ground landmarks, can be created based on the image processing.

In this report the author would like to:

- describe the main principles of the considered navigation technique,

- consider the influence of various factors on the accuracy of the navigation process, i. e. accuracy of spacecraft positioning,
- offer several configurations of on-board equipment used in the navigational system based on the considered principle to provide the most inexpensive and reliable autonomous navigation technique.

All the results sited in this research are obtained using computer simulation of the navigation process. Special software has been created for this purpose that provides a numerical solution to the spacecraft equations of motion, simulation of various disturbances and limitations, data processing algorithms, estimation of the accuracy, etc.

MAIN PRINCIPLES OF THE PROPOSED TECHNIQUE

There are many variants to implement the discussed technology. Their differences are caused by various tasks and requirements on reliability, accuracy and other characteristics of the system. The variants can differ in content and performance of the used equipment and, hence, in different algorithms. Despite these differences, the principle of the system operation is, in general, as follows:

- obtaining of the Earth's surface image by on-board equipment
- image preprocessing
- image analysis to detect and recognize objects and local changes
- determination of the object coordinates in respect to the image
- determination of the line-of-sight orientation
- solving the problem of the object positioning or spacecraft coordinates estimation.

Consider a scheme of a system with a star sensor (configuration of on-board algorithms is illustrated in Fig. 1). The satellite is observing the surface of the Earth. If anomalies on a low information density area are found, their coordinates are preliminary estimated; the high resolution equipment is pointed on this area; the coordinates are updated. In a similar way the position of the satellite is updated if the observed area contains landmarks.

Some areas with high information density and their coordinates can be stored in the memory of on-board computer. This will permit to tie the obtained images to the reference images and get very accurate (meters) information about the position of the detected objects and also estimate the position of the satellite.

So, such an approach can:
- detect objects on low information density areas and determine their coordinates;
- watch local changes on high information density areas and determine their coordinates with accuracy limited only by resolution of the EO equipment;
- solve the autonomous positioning problem with accuracy high enough to get rid of all other navigational means.

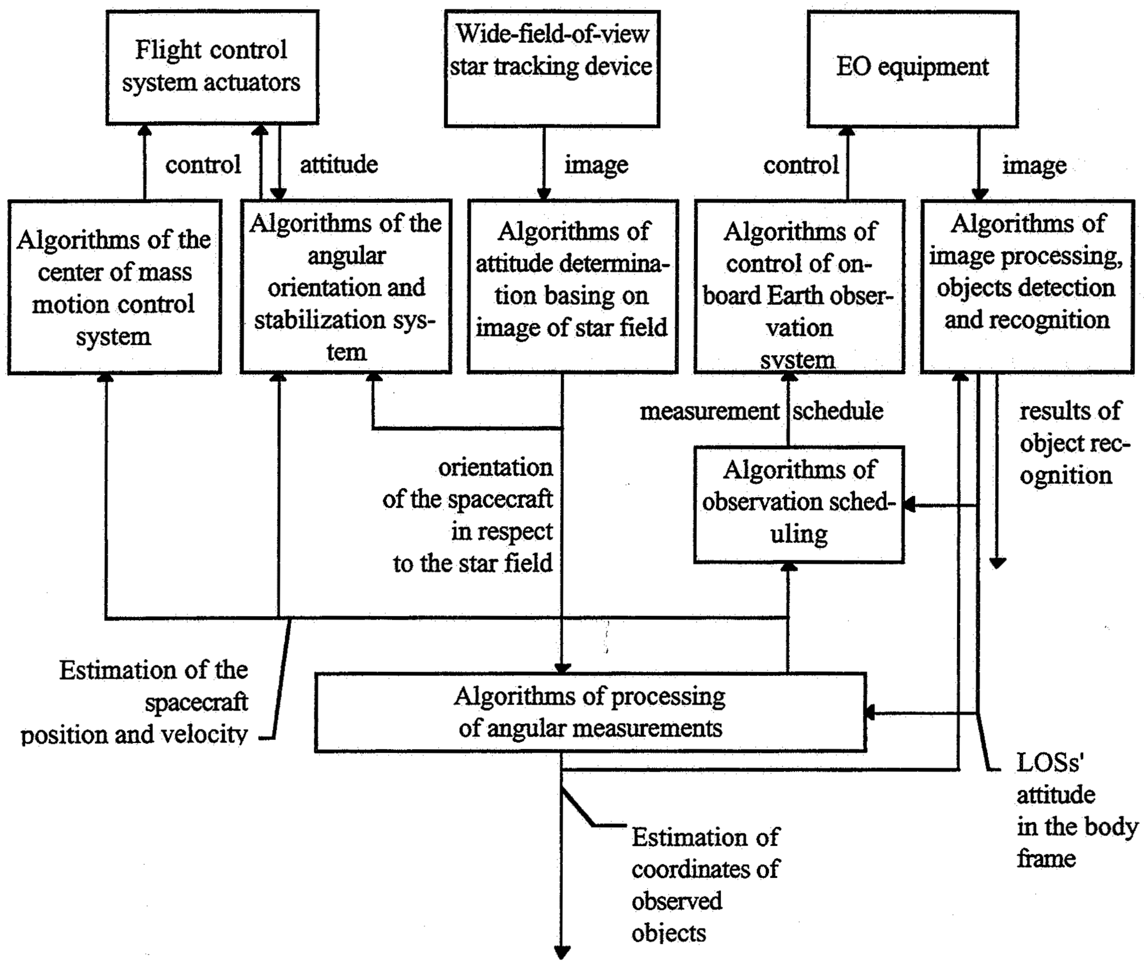

Fig. 1 Scheme of a system with a star sensor

INFLUENCE OF VARIOUS FACTORS ON ACCURACY OF NAVIGATION

Factors with various physical origin have significant influence on the accuracy of autonomous navigation using ground landmarks. All the factors can be divided into two large groups: factors that determine opportunity to perform measurements and factors that determine accuracy of a measurement once it is possible.

Possibility to perform measurements

The factors that determine opportunity to perform measurements influence accuracy of navigation indirectly. Impossibility to perform measurements increases prediction intervals and decreases amount of useful information fed into the navigational system which, obviously, decreases general accuracy of navigation. The contents of this group depend on configuration of on-board equipment, but in most cases the possibility to perform angular measurements depends mostly on possibility to observe a ground landmark. The following principal factors influence this possibility:

- local solar time,
- type of terrain,
- cloud cover,
- readiness of the on-board equipment to take a picture and process it.

Consider these factors.

Local solar time. Local solar time determines whether a given area on the Earth's surface is illuminated and the direction of illumination. In several spectral bands, such as the visual one, absence of solar illumination makes observations totally impossible. In both visual and IR bands direction of illumination and its intensity significantly influence distribution of energetic brightness on the acquired images.

To model the direction of solar illumination is a simple problem due to well-known coordinates of the sun at any moment of time.

Type of terrain. In [1] it was demonstrated that distribution of the landmarks over the Earth's surface significantly influences accuracy of navigation. The subsequent research has shown that it is necessary to use as much surface of the Earth as possible for creation of landmarks. In fact, it means that any informative area, or in other words, area that contains features that make possible to match its images made in a given spectral band with high accuracy, must be used as a landmark. Obviously, such an approach assumes creation of a kind of GIS (Geographic Information System) on board of the spacecraft which will require a high performance on-board computer and a data storage with a big capacity. Any attempt to artificially decrease area used for landmarks will result in increased prediction intervals and, hence, worse accuracy.

Cloud cover. For the EO equipment using waves in the visual and IR ranges, the clouds are a very important and, unfortunately, the most unpredictable factor that determines whether a given point on the Earth's surface can be observed at a given moment of time or if it is obscured.

The problem of modeling the global cloud cover is much more complicated than the two previous ones. In this research a rather simple model of cloud cover is used. It is based on statistical data of cloudiness distribution over the globe for various seasons and does not take into account spatial and temporal correlation between values of cloudiness. For the considered problem such a supposition is more or less correct because the observations take place in various areas of the globe rather rarely, and the effects of correlation are not very strong.

Readiness of the on-board equipment. In contrast to the previous three factors, this one is controllable. It depends mostly on necessity to use the EO equipment for goals different from navigation, e. g. as a payload, and on performance of the on-board computer that determines duration of image processing. If the EO equipment is steerable, the limited angular rate of its rotation will also introduce some delay into the interval between measurements (although with proper scheduling the redirection of the camera can be combined with processing of the received image).

Fig. 2 illustrates influence of minimal interval between measurements on number of observations; the three «natural» uncontrollable factors are also taken into account.

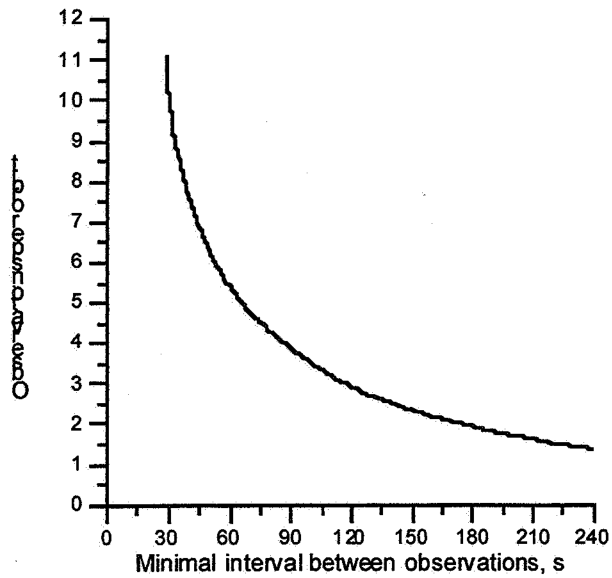

Fig. 2. Influence of interval between observations on the total number of observations

Obviously, the three uncontrollable factors reduce the total number of observations by almost 20 times (with respect to the number of observations under «ideal» conditions which is equal to the period of revolution divided by the minimal interval), and the duration of prediction intervals reaches 2-3 orbits (the orbit period is approximately 6000 s).

Nevertheless, in Fig. 3 one can see that the accuracy of autonomous navigation remains high: not only the level corresponding to the «best» 95% of estimations, but also the maximum error do not exceed 1 km even when minimal interval between measurements is three minutes, and, correspondingly, there are less than two measurements per orbit (see Fig. 2).

At the same time, the threshold of 2-3 minutes can be considered as the highest value of the interval because at such intervals the accuracy decreases much faster. Probably, this fact can be explained very simply: the distance covered by the sub-satellite point during the observation interval, which is equal to the duration of the interval multiplied by ground track velocity, is comparable to the linear dimensions of the largest islands and even Australia. Hence, there is a chance to perform two or more consecutive measurements only above the biggest continents: Europe, Asia, and both Americas.

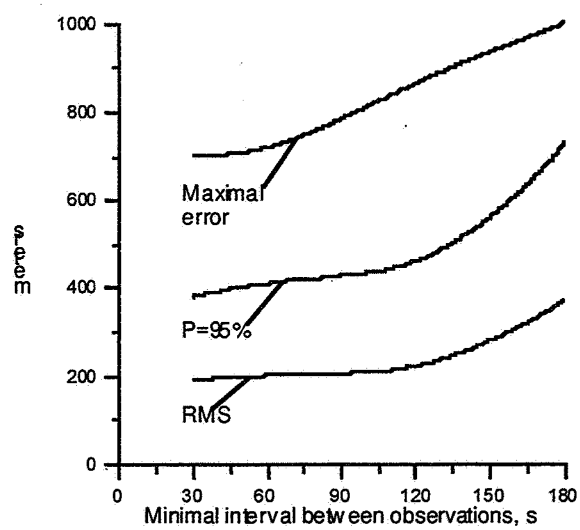

Fig. 3 Influence of minimal interval between measurements on the accuracy of navigation

Accuracy of measurements

Due to various factors, the measured and actual values of the angles that are used for navigation are not identically equal. The sources of the errors can be divided into three groups:

- errors of knowledge of the EO equipment orientation with respect to the body of the spacecraft.
- errors that occur inside the EO equipment due to optics distortions, image processing, etc.
- errors of attitude determination by attitude control system (e. g. based on gyros) or of star sensors' equipment orientation if the angles star — spacecraft — observed object are measured directly.

Some of the errors are negligible: for example, errors of modern star sensors do not exceed 20-30", and often they are much lower, in comparison to several arc minutes, a typical accuracy of traditional means for attitude determination.

Unfortunately, many of the errors have unclear and/or unknown physical origin which does not permit us to create their mathematical models for study of their influence. That is why the errors are modeled using the approach described in [3]. All of the errors are represented as combinations of several components: rapidly changing errors, slowly changing errors with known statistical characteristics, and slowly changing errors with unknown statistical characteristics which can vary only within known limits.

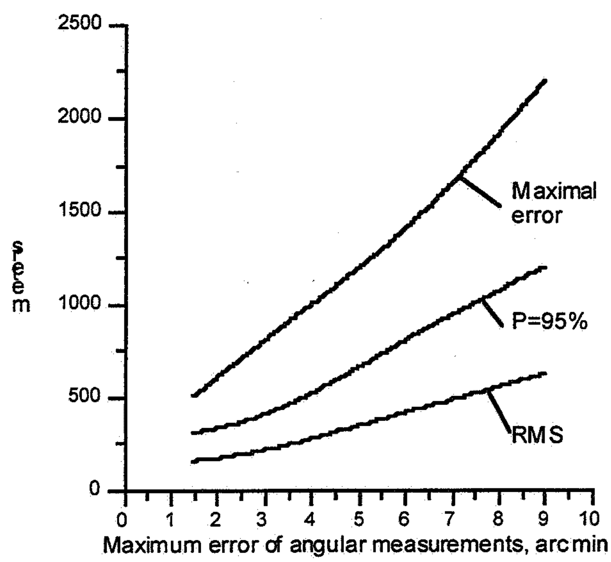

Fig. 4 Influence of accuracy of angular measurements on accuracy of navigation

Fig. 4 illustrates influence of the accuracy of EO equipment (i. e. total maximum errors from two sources: from within the camera and from camera misalignment) on the accuracy of the solution to the autonomous navigation problem.

Although the picture is clear in itself (the dependency is almost linear; maximum error is a bit less than maximal angular error multiplied by altitude), but in the left part of the graph we can see an interesting effect: the dependency loses linearity due to higher weight of other sources of errors which are negligible when the errors of the EO equipment are higher.

It is necessary to note also that neither resolution nor size of the FOV (field-of-view) of the camera are the factors that directly influence the accuracy of navigation. Typically, for matched areas with size about 100×100 pixels, the matching accuracy is better than one pixel. Simple calculations show that spatial resolution will not influence the accuracy of navigation until it is worse than several hundreds of meters: the resolution 300-1000 m adds 1-3 arc minutes to the total error. As for FOV, its influence will be discussed later.

VARIOUS CONFIGURATIONS OF ON-BOARD EQUIPMENT

A camera plus attitude control system

The simplest system able to realize the considered method of navigation consists of a camera and the attitude determination system of the spacecraft. First of all, as in any configuration, attitudes of the LOSs (line-of-sight) of the observed objects are determined in the spacecraft body frame based on image processing. Then they are fed into the estimation part of a data processing algorithm along with results of the attitude estimation or measurements from attitude sensors of any kind (such as: horizon sensor, sun sensor, etc.) and predicted spacecraft position. The result of the estimation will be the estimated spacecraft position. Obviously, the accuracy of positioning in this case will depend mostly on the accuracy of attitude determination. The coarse estimation confirmed by computer simulation has proved the obvious supposition: immediately after the measurement the accuracy is approximately equal to the flight altitude multiplied by accuracy of attitude determination. Obviously, during long prediction intervals the accuracy is much worse.

Systems with and without star sensors

The only obvious exception to the described situation is the case when the attitude control system is comprised of star sensors which are devices with errors that are negligible in comparison with that of the EO camera. But in this case, it is better to convert the data obtained by the camera and the star sensors into angles «axis of star sensor — spacecraft — object LOS». A configuration of on-board equipment with two star sensors pointed in the direction of motion and perpendicularly to the orbit was described in [1, 2]. It provides high accuracy along with high reliability; the results in Figs. 2, 3 correspond to this configuration. The accuracy of a system using the described principle is higher than the accuracy of a system that uses the same equipment, but divides the solution of the navigational problem into two parts: determination of spacecraft attitude and above described solution of the navigational problem. The reason for higher accuracy is the elimination of the spacecraft attitude in the computations. Angles that contain navigational information are derived directly from measurements and not indirectly, as in the case with spacecraft attitude determination. The major reason of inherent redundancy is the star sensors. First of all, a modern star sensor is a telescope able to determine its attitude with very high accuracy with respect to 3 axes. So, if one of the sensors has failed, the remaining one can be used for a backup navigation mode and it will determine the attitude used for navigational purposes.

But even if both star trackers have failed, the system may remain operational. If the spacecraft can maintain a stable attitude so that the camera is directed to the earth, the camera can be used both for spacecraft positioning and as a backup attitude sensor. From the principles of operation stated before, one can see that certain geometrical dependencies tie three principal things: coordinates of the spacecraft, coordinates of the observed object, and the attitude of the LOS of the object in the spacecraft body frame (or, more correctly, in the camera frame). So, if we have several landmarks on an image and attitudes of their LOSs are determined, the obtained data will comprise inherent redundancy for the simple positioning problem; the redundant information will be information on the spacecraft attitude. Such an approach can be supported by purely geometrical explanation: three LOSs form a trihedral with known angles at its top (trivially derived from the measured angles) and a good a priori estimation of its height (altitude of flight).

In contrast to the configuration with star sensors, the accuracy of a system without them significantly depends on such geometrical characteristics of the camera as its resolution and size of the FOV. Fig. 5 illustrates the latter fact. The dependency is due to the small value of measured angles. Unlike the system with star sensors, where it is possible to arrange the sensors to provide measured angles about 90°, the navigational angles of a system without star sensors are always less than the size of the FOV. This fact causes worse observability. Besides, the errors of measurements in a system without star sensors are more correlated with each other. At the same time, the errors emerging within the camera are much less than those of camera alignment. So, the general dependence of the navigation accuracy on the errors of angular measurements has remained similar to one illustrated in Fig. 4, except that the errors vary within 10 to 60 arc seconds. A side effect of this fact is the much higher role of the resolution.

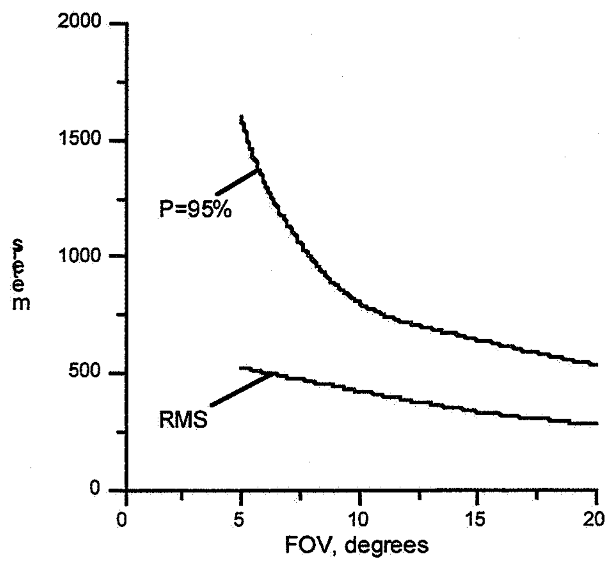

Fig. 5 Influence of the FOV size on accuracy of autonomous navigation without star sensors

A system with a GPS receiver and a magnetometer

Nowadays GPS receivers are becoming a typical device for spacecraft navigation. Advantages of such an approach are obvious, but it has also some disadvantages. Both GPS/NAVSTAR and GLONASS as well as the satellite positioning systems of the next generation create an artificial navigational field, which is why the navigation system using this field is not entirely autonomous. In some conditions, even with intact on-board equipment, solution of the navigational problem will be impossible due to unfavorable conditions or, much worse, due to loss of integrity by the GPS or another reason that destroys the artificial navigational field.

In any case, it would desirable to have an opportunity to provide redundancy of the autonomous navigational system by components using different physical principles. Moreover, such an opportunity will be especially attractive for satellites that have on board a magnetometer and a camera as secondary or even primary payloads. Using the suggested approach, it is possible to provide required redundancy of autonomous navigation.

In the nominal mode of operation, the navigational problem will be solved based only on data from GPS receiver (either for positioning only with gyros used for attitude maintenance or for both position and attitude determination without gyros on board). At the same time, the data from GPS can be used for calibration of the camera and magnetometer. The flight calibration will significantly raise the accuracy of both devices.

In the backup navigation modes (which can be very different depending on contents of operational and failed equipment), the magnetometer will be used as a highly reliable almost continuously operating device that provides excellent convergence and stability of the solution to the navigational problem. The camera will serve as the source of more accurate information on spacecraft position and attitude working with long prediction intervals, as described before.

CONCLUSIONS

The results of the research have confirmed the opportunity to create an autonomous navigation system with maximum positioning error less than 1 km even under influence of such unfavorable factors as cloud cover, varying illumination, and restrictions on the minimal interval between measurements. The autonomous navigation systems using ground landmarks can be used both as standard and as backup navigation systems.

The considered navigation technique can be implemented in the systems considered in this paper, such as: a camera plus one or two star sensors, a camera plus attitude control system, or a camera plus GPS receiver and magnetometer.

Results sited in [1, 2] and in this paper show that simple modifications of the Kalman filter, namely, the extended Kalman filter, are adequate for the considered problem.

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CERES AND THE S'COOL PROJECT

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Abstract. The first Clouds and the Earth's Radiant Energy System (CERES) instrument will be launched on the Tropical Rainfall Measuring Mission (TRMM) spacecraft from a Japanese launch site in November 1997. This instrument is a follow-on to the Earth Radiation Budget Experiment (ERBE) begun in the 1980's. The instrument will measure the radiation budget - incoming and outgoing radiant energy - of the Earth. It will establish a baseline and look for climatic trends. The major feature of interest is clouds, which play a very strong role in regulating our climate. CERES will identify clear and cloudy regions and determine cloud physical and microphysical properties using imager data from a companion instrument. Validation efforts for the remote sensing algorithms will be intensive. As one component of the validation, the S'COOL (Students' Cloud Observations On-Line) project will involve school children from around the globe in making ground truth measurements at the time of a CERES overpass. Their observations will be collected at the NASA Langley Distributed Active Archive Center (DAAC) and made available over the Internet for educational purposes as well as for use by the CERES Science Team in validation efforts. Pilot testing of the S'COOL project began in January 1997 with two local schools in Southeastern Virginia and one remote site in Montana. This experience is helping guide the development of the S'COOL project. National testing is planned for April 1997, international testing for July 1997, and global testing for October 1997. In 1998, when the CERES instrument is operational, a global observer network should be in place providing useful information to the scientists and learning opportunities to the students.

Acronyms and Symbols

ADM	Angular Distribution Model
AVHRR	Advanced Very High Resolution Radiometer
CERES	Clouds and the Earth's Radiant Energy System
DAAC	Distributed Active Archive Center
EOS	Earth Observing System
ERBE	Earth Radiation Budget Experiment
F	Radiative flux, W/m^2
I	Measured radiance, W/m^2-sr
NASA	National Aeronautics and Space Administration
NOAA	National Oceanic and Atmospheric Administration
R	Anisotropic factor
RAPS	Rotating Azimuth Plane Scanner
S'COOL	Students' Cloud Observations On-Line
TOA	Top-of-Atmosphere
TRMM	Tropical Rainfall Measuring Mission

Background

The Earth's climate is determined in large part by its energy, or radiation, budget. A schematic summarizing the radiation processes in the Earth's current climate is shown in Figure 1. A large part of the reflected (shortwave) component of the outgoing radiation is due to clouds, while more than a third of the emitted (longwave) component comes from clouds. Thus an understanding of clouds and how they interact with the rest of the climate system is necessary before accurate predictions of climate change can be made. Better knowledge of cloud processes may also lead to improved long-range weather forecasting.

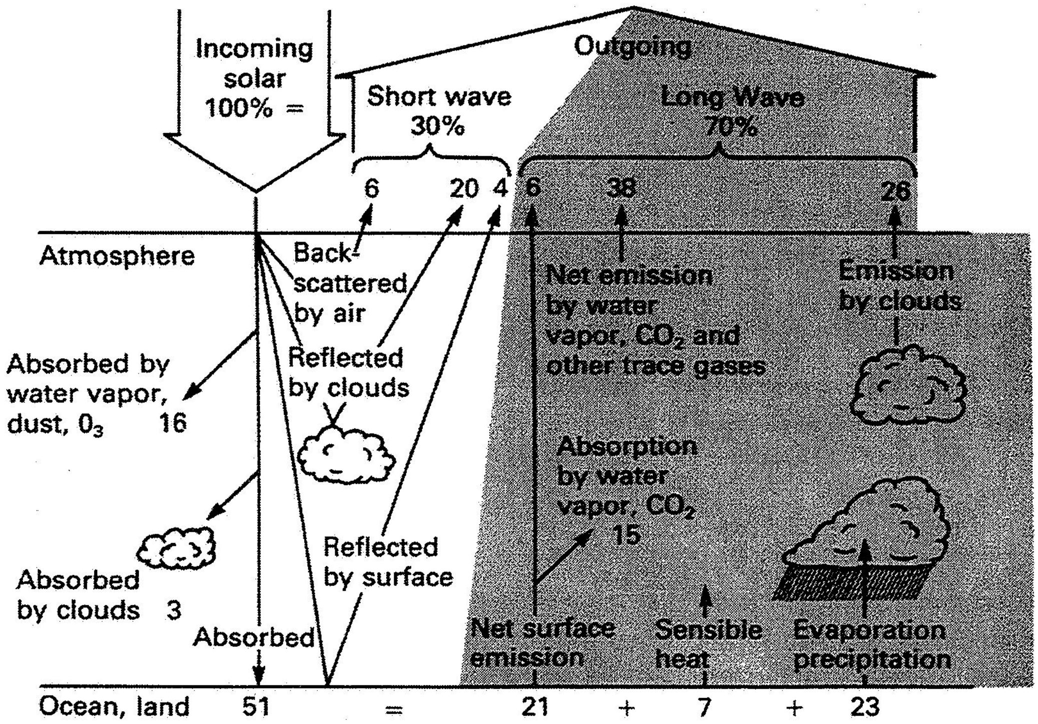

Figure 1: Earth Radiation Budget Processes (from Ref. 1)

The Earth's radiation budget has been monitored on a global basis using spaceborne instruments since the mid 1970's (Nimbus-6 and 7). The ERBE project (Ref. 2) operated instruments on a number of satellite platforms starting in 1984, some of which continue to function today. Recent measurements by the SCAnner for RAdiation Budget (SCARAB; Refs. 3, 4) augmented this dataset for about a year. These instruments were designed principally to measure the energy fluxes at the top of the Earth's atmosphere (TOA). ERBE classified the radiative behavior of the Earth-atmosphere system using 12 scene types, such as "clear over ocean", and "mostly cloudy over land or desert". For each scene type it used an angular distribution model (ADM) built from Nimbus-7 data to convert from satellite-measured radiance to hemispherical flux: $F=\pi I/R$, where R is the anisotropic factor from the ADM. ERBE's field of view was small enough to allow it to see clear areas between clouds at least part of the time. A comparison of clear and cloudy fluxes could then be used to measure the net radiative forcing of clouds, i.e., the net effect of clouds on the Earth's energy budget. ERBE found that on a global basis, in the current climate, clouds act to cool the Earth during all seasons (Ref. 5). A closer look at the data, in combination with global data on cloud distribution, shows that different clouds have different effects (Table 1, adapted from Ref. 6). Furthermore, the tropics are found to be in near radiative balance, while cooling is greatest over mid-latitude oceans. These findings have led to increased interest in clouds within the climate community, such that this issue is one of the highest priorities in global change research.

Table 1: Climatological Effects of Clouds

Cloud Class	Effect	Magnitude	Area Coverage
High thin	Warm	+23	10%
High thick	Cool	-80	9%
Mid thin	Warm	+10	11%
Mid thick	Cool	-103	7%
Low	Cool	-62	26%

The ERBE data are available to researchers and interested individuals around the world from the Langley DAAC (Ref. 7).

CERES

The CERES experiment (Ref. 8) is being designed as a follow-on to the aging ERBE instruments. CERES instruments will operate on a number of platforms as part of the NASA's Mission to Planet Earth over a period of at least 15 years. The first will be launched from Tanegashima on the Japanese TRMM spacecraft in November 1997. CERES will make measurements with better spatial resolution than ERBE, and will fly on satellites carrying imagers that will assist in identifying and classifying clouds within the field of view. Cloud parameters to be obtained include the cloud fraction, cloud height, and cloud optical depth, as well as the particle phase and size. Fundamentally, CERES will provide an extension of the ERBE record of TOA radiative flux data to allow detection of any climate change signal. The high resolution imager instruments will enable improved determination of the radiative scene type -- up to 200 scene types and associated ADMs are planned -- and therefore produce improved measurements of cloud radiative forcing. CERES will also provide an initial determination of radiative fluxes within the atmosphere and at the surface, using sounding and cloud property information from companion instruments. A factor of 2 to 3 reduction in TOA radiative flux error is anticipated compared to ERBE.

A CERES instrument consists of a three-channel scanning broadband radiometer. It has a nadir field of view of about 20 km. The three channels measure the total (0.2 - 100 μm), shortwave

(0.2 - 5 μm) and longwave window (8 - 12 μm) radiation; the latter to assist in obtaining within-atmosphere and surface fluxes. The instrument scans across the earth and back every 6.6 seconds, with an internal calibration in the middle of each scan. Solar calibrations are performed every two weeks to monitor the stability of the instrument sensor. The scanner can operate either in cross-track mode (scanning left and right as it moves along the orbit) or in rotating azimuth plane scanner (RAPS) mode. RAPS mode increases the sampling in angular space (solar zenith, solar azimuth, view zenith, view azimuth) in order to more rapidly obtain measurements to build ADMs. On the TRMM platform, which has a precessing orbit, CERES will provide data between latitudes of 35 S to 35 N. Additional CERES instruments will be launched on the EOS-AM platform in June, 1998 (sun-synchronous polar orbit with equator crossing at 1030 local time) and the EOS-PM platform in 2000 (1330 sun-synchronous polar orbit). These instruments will provide global coverage.

Validation. Validation of TOA fluxes is only possible by comparison to other space-based measurements. Few, if any, will be available in the CERES timeframe. As in ERBE, therefore, consistency checks will be used to validate the TOA results. Validation of the within atmosphere and surface fluxes will rely on data from intensive field campaigns and on-going validation studies such as the Department of Energy's Atmospheric Radiation Measurement program and the Baseline Surface Radiation Network. Validation of cloud property retrievals will also rely on these sites. Areas of active research include detection of clouds in polar areas and over snow, ice, or bright surfaces in general; detection of broken low level clouds, multilayer clouds, and subvisible cirrus; and discrimination of clouds from smoke or dust. Some data for validation of cloud property retrievals will be provided by an educational outreach project, described in the following section. Based on the ERBE experience, CERES validation efforts will likely focus first on the climate months of January and April 1998.

The S'COOL Project

History. The S'COOL project was initiated in December 1996 following an informal conversation with a middle school (6th grade) teacher from Gloucester, Virginia, USA. She was seeking a simple, safe, and cheap experiment that her students could do, the results of which could be sent to NASA. She viewed the connection to NASA as a way to both motivate her students, and to connect them to the world around them. If the results were of use to NASA, so much the better. Using school students as cloud observers for CERES fits perfectly into the teacher's requirements

and provides information to researchers which would not otherwise be available. The short lifetime of clouds (5-10 minutes on average) means coincident observations are very important for comparison to satellite data.

Concept. School students around the world will go outside at the time a CERES instrument in orbit is viewing their location and make an observation of the clouds in the sky (cloud type, fraction, height, and an estimate of opacity), some related meteorological data (temperature, pressure, relative humidity), and the ground cover (snow, foliage, etc.). They will report their observations to a central archive at the Langley DAAC, keeper of the CERES data (Ref. 7). Once the satellite observations are processed, the students will be able to obtain these measurements via the Internet to compare with what they reported. The students' observations will be kept in a database which will be available for use as validation data by the CERES Science Team and other researchers, as well as for educational use by school teachers.

Development. The S'COOL project is being developed in an incremental fashion (Table 2), with test phases during January, April, July, and October 1997. During the test phases existing operational satellites are being used as a stand-in for CERES. The January pilot test was a feasibility test, using the Advanced Very High Resolution Radiometer (AVHRR) instrument on the NOAA-14 spacecraft, to determine how students reacted to the project and whether it was worth pursuing. It was carried out after minimal materials were developed, with personal visits to the classroom by several NASA researchers to instruct the students about the project. The response exceeded all expectations, so development has proceeded apace. Interim tests were carried out with a 6th grade class in Big Timber, Montana, USA to exercise the e-mail interface, and with a 4th grade class in Poquoson, Virginia, USA to test the Internet interface. All tests resulted in identification of issues needing work, and also generated additional ideas for the S'COOL project. In particular, the teacher of the Poquoson class reported that her students learned from the project "across the curriculum", from math and science to language arts and vocabulary.

Table 2: Development Schedule for S'COOL Project

Phase	Date	Scope	Test	Goal
1	January 1997	Local	Concept	Go/Nogo
2	April 1997	National	Interface	Automation
3	July 1997	International	Interface	More automation
4	October 1997	Global	Capacity	Max. automation
5	January 1998	Global	TRMM spacecraft	Correct orbit
Operational	April 1998 ->	Global	Periodic or on-going activation	Validation data

A national test in the USA at the end of April 1997 is planned to test automated operation of the project without school visits by NASA researchers. Invitations were sent to 30 individuals in different states in an attempt to obtain broad geographic coverage and thus increase the range of cloud and surface types sampled. Ten schools around the country have asked to participate (Figure 2). The teachers involved will implement the project using the information on the S'COOL website (Ref. 9) and a one page instruction sheet that was mailed to them. They will be asked to provide feedback on their experience with the project and on the materials developed to date, as well as to provide comments on other materials they believe still need to be developed. Schools east of the Mississippi River will make observations coinciding with overpasses of the AVHRR instrument on NOAA-14, while those in the west will coordinate with GOES satellite observations at 1800 GMT. Limited feedback will be provided, with comparisons to one day of satellite measurements for each participating site.

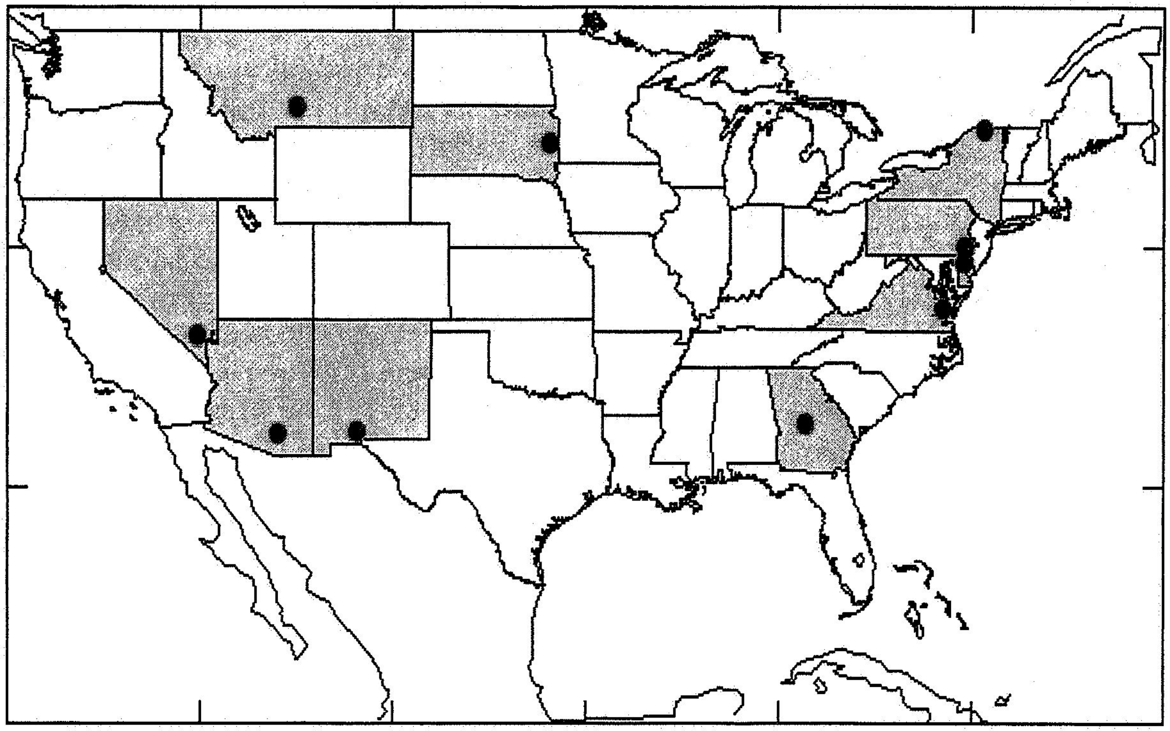

Figure 2: S'COOL Participant Locations for Phase 2, April 1997

Given school schedules, the July 1997 test will be an international test focussing on the southern hemisphere. This test will continue to expand geographic coverage (initial contacts have been identified in Argentina, Australia, Brazil, Chile, Korea, the Philippines, and South Africa) and explore issues such as language and technology barriers for international participants. Automated access to the AVHRR overpass times will be essential for this phase of the development. A test in October 1997 is planned to be of the operational S'COOL system, with participation from around the globe and automation of all elements of the interface. By January 1998, when the CERES instrument on TRMM is expected to begin providing regular measurements, a global network of student observers should be in place. Observations may then be made as they fit the schedule of each participating class, or on a focussed basis for validation periods.

Design. The S'COOL project is being designed to exploit the Internet and related technology to a maximum extent in order to minimize the manpower required to run the project. A website has been created (Ref. 9) and is intended to be the primary interface with participants. Any print material that is found to be necessary will be developed from the information on the website. In particular, registration and report forms are available on-line. An orbital interface, to allow participants to determine CERES overpass times for their location via a simple Internet form, is in the planning stages. On-line access to the database of global S'COOL observations is also being designed. This will allow participants to retrieve all S'COOL observations for a given time period and study the variation of clouds in different regions of the Earth. A companion database of satellite results is being contemplated as well. Development of this part of the project will depend on the availability of validated CERES products for release.

Summary

The climate of our Earth is controlled by a complex system of interacting parts. Monitoring the system over the long term and on a global basis is necessary to understanding our planet and the changes humans may be inducing. Global monitoring of the radiation budget began with satellite instruments launched in the mid-1970's and will continue this year with the launch of the first CERES instrument in November. CERES represents a large improvement in our ability to monitor the Earth radiation budget and the role of clouds on a global basis, in order to understand how they interact with the climate system. As part of the validation efforts for CERES, the S'COOL project is being developed. S'COOL will involve school children around the globe as cloud observers. Participating classes will provide cloud observations and related measurements at the time of a CERES overpass and transmit them for archive in the Langley DAAC. These observations will be available to the CERES Science Team for use in validating and improving cloud retrieval algorithms; and to educators worldwide for educational purposes. Anyone interested in further information on S'COOL, or in becoming a participant, is invited to contact the first author.

Acknowledgments

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Earth Observing System Constellation Design Optimization Through Mixed Integer Programming

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Abstract

This paper presents the successful development of a configuration design algorithm for optimizing a select class of multi-satellite constellations for performance per cost with managed risk. The Earth Observing System (EOS) mission design has been revised many times due to the shifting sands of space policies and budgets. In past re-design efforts, limited numerical modeling and heuristic attempts have been used to determine feasible, though non-optimal, configurations of the EOS multi-satellite constellation - with varying degrees of technological, economic and scheduling risk. The need for a mission design tool that performs rapid configuration design optimizations and sensitivity analyses has now been answered by employing Mixed Integer linear Programming (MIP) model formulation strategies, complemented with advanced "Branch & Bound" optimization solution techniques. This paper briefly describes the methodology of the algorithm, and presents the results of its application in the EOS configuration design optimization. The research focuses on optimization for overall system performance and cost, while subject to science, technology, and policy mission constraints. Management of risk tolerance in the formulation is also through these constraint equations. The sensitivity analysis addresses the design areas of scientific instrument selection, payload definition and spacecraft scaling/launch vehicle selection.

Past multi-satellite constellation design efforts have assumed the use of identical spacecraft, and concentrated only on the feasible configurations for satisfying coverage requirements. The ambitious range of measurements to be made by EOS is beyond the capability of a single spacecraft, so these various tasks must be distributed amongst dissimilar spacecraft using different, overlapping instruments - while still meeting coverage requirements. These factors contribute to make the design of a Distributed Task Constellation (DTC) a vastly more complex challenge than that of a simple Coverage-Based Constellation (CBC). DTC design becomes all the more difficult when considering issues such as performance vs. cost, system redundancy and reliability, international collaboration and launch vehicle issues. As a result, the "brute force" CBC design process of numerical modeling with full enumeration must be abandoned for the more powerful operations research approach of analytical modeling using linear equations. The model is formulated in a series of "mixed integer" linear equations involving continuous, integer, and binary variables. The use of these types of variables reflect the discrete integer nature of certain design parameters (one doesn't launch 2.37 spacecraft), and the binary nature of "go/no-go" flight selection decisions. The equations capture the mission goal (objective function), and the cost, performance and physical constraints to be modeled. As successfully demonstrated in the EOS mission design, a MIP approach with an intelligent "branch and bound" search enables large problems involving over one thousand variables and thousands of constraint equations to be solved for *global* optima.

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1.0 Introduction

The aerospace industry is currently experiencing a fundamental restructuring driven by the ending of the Cold War. Inefficiencies that had been allowed to perpetuate in an era of superpower competition were tolerated to ensure mission success in a geopolitical context at the expense of cost-effectiveness. Today, increasingly-constrained budgets necessitate a new approach to coordinated space systems design and operations that fully integrates performance optimization with strict cost controls. It is imperative to address the aerospace design *process*.

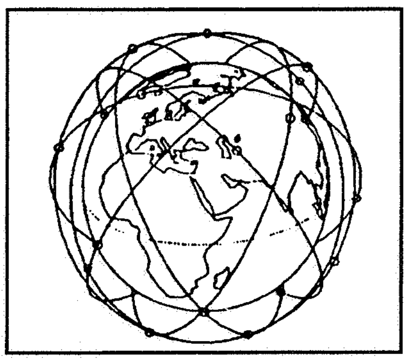

Figure 1.0 Typical symmetric multi-satellite constellation configuration.

The role of constellations of many spacecraft acting in concert is of increasing importance in the new aerospace business environment. (See Figure 1.0) Using constellations to achieve mission goals necessitates a very different design development and production approach than that of traditional single satellite missions. Consequently, the Earth Observing System, a scientific remote sensing constellation of dissimilar spacecraft, requires different configuration design optimization tools. Furthermore, it is a constellation that needs to be optimized not only for mission performance (in order to fulfill its nominal mission), but also for economic efficiency, to reflect realities of the fundamentally different and dynamic current fiscal environment.

1.1 Distributed Task Constellations

While most constellation architecture is selected purely due to orbital coverage requirements, an alternative type of constellation is the *distributed task constellation* whose dissimilar satellites fulfill different tasks in a complementary fashion to achieve an overall mission goal. (See Figure 1.1) An example of this is the Earth Observing System. EOS satellites comprise a constellation of dissimilar remote sensing platforms (whose payloads vary in instrument composition), yet all the observations taken together will form a cohesive investigation into the Earth's natural and anthropogenic processes. While constellation designs of similar satellites typically focus on the selection of the orbital parameters and the number of satellites, a distributed task constellation is primarily concerned with covering the set of tasks that make up the larger mission goal within

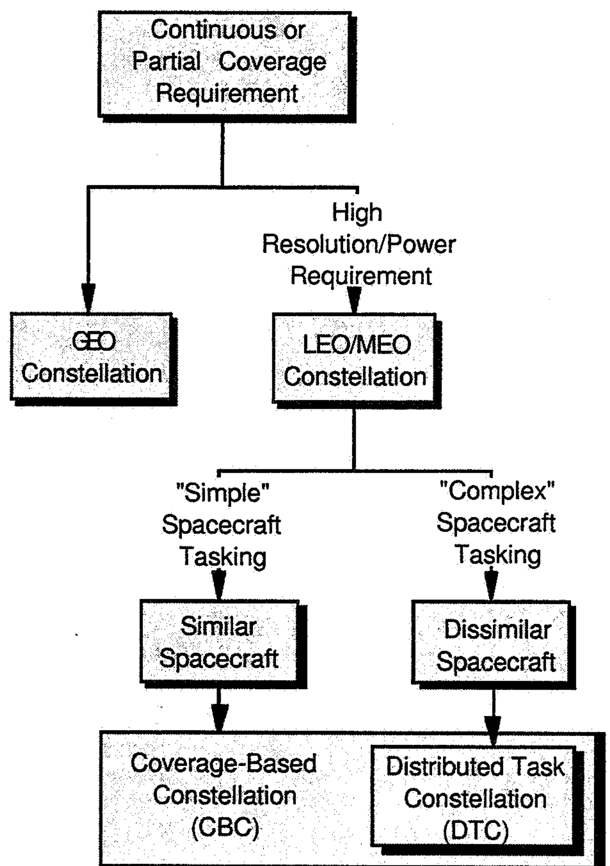

Figure 1.1 Constellation design flow chart.

Table 1.0 EOS Program History (US$)

Mission Planning	1982-87
Announcement of Opportunity	1988
Announcement of Selection	1989
New Start ($17B)	1990
Restructuring Process ($11B)	1991-92
Restructuring Confirmation	1992
Rescoping Process ($8B)	1992
Rescoping Confirmation	1993
Launch Vehicle Change	1994
Rebaselining ($7B)	1994
2 New Reviews/ $2.7B cut?	1995

various system constraints.

1.2 The Earth Observing System

The somewhat tortured design history of EOS demonstrates the need for flexible design tools that strive to balance the collection of high-quality scientific data with a cost-effective design. Congressionally enforced re-designs have also echoed the paradigm shift within the industry as it struggles to adapt to a post-Cold War fiscal environment. (See Table 1.0)

In 1991, the initial EOS program planned for two large Titan-class spacecraft series, attached payloads on an ambitious Space Station *Freedom*, as well as instruments selected to fly on the planned Japanese and European Polar-Orbiting Platforms. The current configuration is a far less ambitious mix of mid-sized spacecraft and a handful of smallsats. Throughout these re-designs, the overall focus of the program science goal has stayed fairly constant. (See Figure 1.2) This prioritization of the various science measurements is vitally important for the instrument selection of the DTC because it provides the relative weighting of the measurements that each instrument addresses.

1.3 Operations Research

The coordination of large-scale national and international space science efforts has many potential benefits. It is possible to enhance both the quality and quantity of data collected, as well as to potentially reduce the budget requirements due to increased cost-sharing and less duplication of efforts. In particular, investigations involving multiple-spacecraft platforms provide unique opportunities for sophisticated missions involving constellations in Earth orbit. To obtain these savings, intelligent planning tools must be available to policy makers and mission planners, in order to exploit potential benefits in science enhancement and cost-savings. The field of operations research has

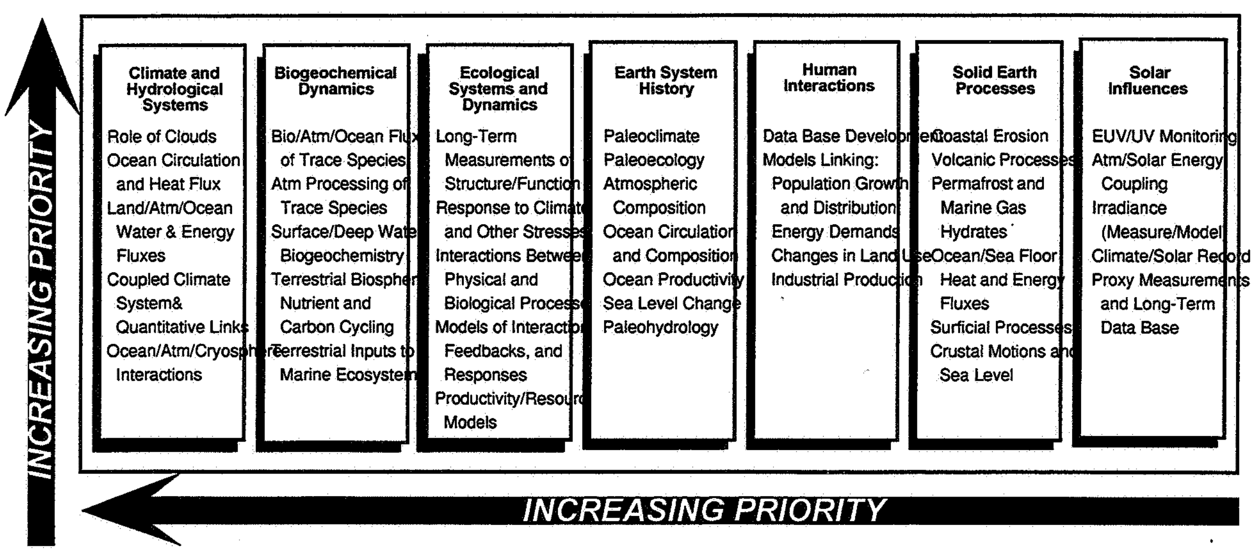

Figure 1.2 Earth Observing System Measurement Priorities

developed several applicable optimization techniques that can aid in meeting this challenge. One of these methods, mixed integer programming, is used in this research. MIP consists of linear constraint equations and a linear objective function to be maximized or minimized, and is solved with an iterative application of the Simplex method called "branch & bound". It should be noted that branch & bound does not guarantee that the optimal solution is unique - there may be another different, yet equally optimal solution.

2.0 Modeling Strategy

The inordinate number of re-designs that the EOS program has suffered clearly demonstrates the demand for a capability to identify feasible configuration solutions, and to efficiently search through the solution space to converge on an optimal configuration. Of course, there is the additional restriction that changes in cost, technology, space policy, and science emphasis must be accommodated, with managed risk.

2.1 Methodology Selection

A robust Phase A (feasibility study) or Phase B (preliminary design) DTC configuration design would include,

- selection of scientific instruments;
- payload definition;
- spacecraft scaling/launch vehicle selection,

while addressing the scientific objectives of the mission with sufficient reliability at an acceptable program cost and cost *risk*. Orbit selection can also be included in the algorithm, however EOS spacecraft generally use a frozen, Sun-synchronous, 16-day repeat orbit so it is not discussed here.

The permutations of 30 candidate instruments, with various inter-instrument complementary and antagonistic relationships, flying on a half dozen possible types of launch vehicles is an extremely large problem. An analytical approach, using equations instead of manipulating vectors, was therefore selected over a numerical method. An analytical approach is also more attractive for an early phase design tool because it affords greater modeling breadth at the expense of modeling depth. Using the results of the analytical model's "head start", a detailed numerical model can then be developed for solution verification.

Given an analytical approach, it is clear that there is a large incentive to use the linear equations of LPs if at all possible. Linear models are in wide use, have extremely sophisticated commercial, off-the-shelf (COTS) solution software available, and allow much larger problems to be solved than if nonlinear equations were used. The algorithm discussed in this paper was successfully tested with the commercially available Cplex version 3.0 MIP solver for systems of linear equations. Solution times on a dedicated DEC Alpha workstation were on the order of minutes.

The challenge of linear programming is to capture the essence of the design process within the confines of linear equations. One way to accomplish this is to restrict certain variables to be integer or binary. Integer variables reflect the reality that solutions of certain variables, such as number of rockets, can only be discrete quantities. The further restriction of binary variables, for items like instrument selection decisions, are used when "yes/no" decisions must be made. This is commonly used in decision-support software.

The final result is an algorithm using mixed integer linear programming. The instrument selection is similar to an OR "set covering" problem, while the integrally linked payload definition and spacecraft scaling/launch vehicle selection is analogous to the classic "bin packing" problem. The problem size is still dangerously large, however the branch & bound solution approach can successfully optimize the problem if strict attention is paid to formulation tightening.

Priority	Measurements	Instruments: MODIS-N	MODIS-T	HIRIS	ASTER	MISR	EOSP	MIMR
98	Soil Features	60	80	85	90	85		50
98	Surface Temperature	60	95		60			50
98	Vegetation	90	95	90	90	90		40
100	Clouds	85	90	80	80	75	95	10
98	Water Vapor	40	60		40		80	90
98	Snow	70	80	85	90	75		75
100	Earth Radiation	50	50	60	75	50		

Figure 3.0 Sample of the Instrument Performance Matrix

3.0 Model Formulation

A description of the input data and the performance section of the formulation are presented below. More detailed discussion of the constrained optimization process used in this research may be found in [Matossian][1].

3.1 Input Data

The most difficult input data to collect is the candidate instruments' performance matrix and the projected instruments' costs. In the case of EOS, the matrix is a mapping of each instrument's performance of each of the measurement goals of the mission. In Figure 3.0, a sample evaluation can be seen that compares the utility of some of the EOS instruments in performing subset of the measurement goals. (The full matrix is 34 measurement rows by 35 instrument columns.) In this example, a broad qualitative assessment has been made, though multiple columns could exist for each instrument to include more specific criteria, such as accuracy, precision, etc. Another alternative would be to use multiple columns for each instrument based on configuration, orbit altitude, etc. Also of note is the priority weighting of the different measurement goals of the mission. These values were roughly extrapolated from the weighting in Figure 1.2 - the *only* openly published science prioritization for EOS.

Instrument costs are difficult to predict, however the Multi-variable Instrument Cost Model (MICM)[2] and it's associated cost *risk* refinement[3] provide a powerful cost estimation capability. Other costs are more easily collected or estimated: spacecraft and ground systems costs can be estimated based on parametric models[4]; while launch vehicle and insurance costs can be directly ascertained from the sources.

Standard technical and performance data for the instruments and launch vehicles are entered into the model, as well as certain technical parametric relationships for the spacecraft.

3.2 System Performance

The first step of processing for the set covering problem, is to distinguish between

Example: Soil Features

Primary Instruments

$$P_a + x_2 + x_3 + x_4 + x_5 \geq 1$$

Secondary Instruments

$$P_b + x_2 + x_3 + x_4 + x_5 + x_9 + x_{14} + x_{15} \geq 2$$

Total Penalty

$$2\,P_a + 4\,P_b = P_1$$

Penalty Structure

Selected Instruments: Primary	Secondary	Total Penalty
0	0	10
0	1	6
1	0	4
0	2	2
1	1	0
2	0	0

Figure 3.1 The constraints for the soil features measurement illustrates how both measurement-covering and a robust system design requirement can be achieved using linear equations.

"primary" and "secondary" instruments in addressing each measurement. A primary instrument is specifically designed for performing a given measurement, while a secondary instrument provides significant correlative information of lesser utility. In the formulation, the primary and secondary instruments' constraint equations are separated as in Figure 3.1. (Each x is a binary variable representing a specific candidate instrument for flight.) The structure of the equations demands at least one primary instrument to be flown, as well as an additional primary or secondary instrument to be flown as a backup. If this is not possible, such as for cost limitations, then the P_x variables take on non-zero values in order to satisfy the constraint equations, reflecting the decreased robustness of the system to reliably collect that measurement.

Table 4.0 The measurement table shows how well the selected instruments cover the set of measurements.

Measurements Table

Measurement Category	Penalty max 10	Priority max 100	Instrument_Counts Primary	Secondary
Soil Features	0	98	2	0
Surface Temperature	0	98	2	1
Vegetation	0	98	2	0
Clouds	0	100	3	0
Water Vapor	0	98	1	2
Snow	0	98	2	0
Earth Radiation	0	100	1	1
Solar Radiation	2+0= 2	100	0	2
Precipitation	0	98	1	2
Evapotranspiration	2÷0= 2	98	0	2
Runoff	2+4= 6	86	0	1
Wetland Areal Extent	0	98	1	1
Phytoplankton	0	88	2	0
Turbidity	0	86	2	0
Bioluminescence	0	88	2	0
Surface Elevation	0	98	2	1
Waves	2+8=10	96	0	0
Inland Ice	0	78	2	1
Sea Ice	0	99	1	2
Atmospheric Comp.	0	89	2	1
Ozone	0	97	1	1
Aerosols	0	90	2	0
Temperature	0	98	2	0
Winds	2+4= 6	97	0	1
Lightning	0+4= 4	100	1	0
Emission Features	0+4= 4	100	1	0
Electric Fields	0	39	2	0
Rock Unit Minerology	0+4= 4	46	1	0
Surface Structure	2+4= 6	50	0	1
Gravity Field	2+4= 6	45	0	1
Surface Stress	0+4= 4	99	1	0
Oceanic Geoid	2+4= 6	99	0	1
Magnetic Field	0+4= 4	45	1	0
Plate Motion	0+4= 4	45	1	0

Weighted Mean Penalty = 1.8212

4.0 Sample Output

The model's output is an optimal configuration for the distributed task constellation for the given input conditions. Currently, the output is in three sections: a measurements table, a payload configuration table, and a spacecraft table. The following data discussed in this section is from a run without penalty or cost constraints, other than being minimized together in the objective function with a neutral scaling factor of penalty to costs of value 1. This resulted in:

Total Penalty + Total Costs
1607.4 + $1405.3

= Objective f
= 3012.7

for costs in 1996 US$M.

4.1 Measurements

Table 4.0 gives the results of the set covering sub-problem in the formulation. In the case of the soil features measurement discussed previously, we can see that no penalty was assessed in the objective function. Two primary instruments and one secondary instrument for soil features measurements were selected for flight, thus satisfying the measurement reliability criteria of having at least one primary

Tables 4.1 & 4.2 The payload and spacecraft tables show the flight configurations and the characteristics of each spacecraft.

Payload Table

Launch Vehicle	Instrument Stability	Viewing Target	Instrument Payload
Taurus	mixed	solar+	ACRIM SOLSTICE_II GOS IPEI
D7930	active	Earth	MODIS-T ASTER TES
D7930	passive	Earth	AIRS MHS NSCAT_II GLAS SWIRLS LIS XIE

Note: "mixed" = active & passive; "Earth" = Earth nadir or limb; "+" = field.

Spacecraft Specifications Table

Spacecraft Class	Mass P/L Kg	Usage Prcnt	Mass S/C Kg	Power P/L W	Usage Prcnt	Power S/C W	Data Kb/s	Cost 96$M
Taurus	255	89%	1020	146	91%	365	30	181
D7930	910	98%	3640	1009	75%	2018	111776	618
D7930	842	90%	3368	1078	80%	2156	1649	607

Note: P/L = Payload; S/C = Spacecraft.
D7930 = Delta 7930; AIIAS = Atlas IIAS.

instrument and one backup instrument.

For certain measurements, a penalty has been assessed for not sufficiently covering the measurement. This can occur because the selection of an additional instrument was inefficient in the balance of cost and penalty (in the objective function,) or simply because there was an inadequate supply of instruments in the candidate pool for that measurement.

The weighted mean penalty is a useful parameter for comparing the science performance of different DTC configurations. It is calculated by weighting the penalty for a given measurement with the corresponding priority.

4.2 Spacecraft & Payload

The bin packing results can be seen in Tables 4.1 and 4.2. Three different launch vehicles were selected from a choice of US rockets: a solar viewing payload (ACRIM & SOLSTICE II) has been formed on the Taurus-class spacecraft, augmented with some particles and fields instruments; the remaining payloads' instruments target in the direction of Earth nadir and/or limb. The first payload configuration combines a mixture of active and passive instruments, but the latter payloads consist of an exclusively active or passive instrument payload. This latter characteristic is necessary due to the passive payload constraint of the GLAS instrument, leaving the second Earth-looking payload for active instrumentation, though it could have been mixed with "active-tolerant" passive instruments.

Table 4.2 indicates the approximate mass, power, data rate, and cost values for each class of spacecraft. In this case, the optimization model has achieved excellent packing factors for the available payload mass. Similarly, the power constraint, (though less sensitive than the mass constraint,) was satisfied. Overall spacecraft mass and power budgets are estimates based on ratios derived from historical data. The science data rate and spacecraft cost provide further insight into the configuration for the user.

5.0 Sensitivity Analysis

The balance of performance and cost with managed risk for a DTC can be fully investigated with this modeling approach. The model can be solved to minimize the cost, to minimize the science penalty (*i.e.* maximize the science performance), or to minimize both together with a scaling

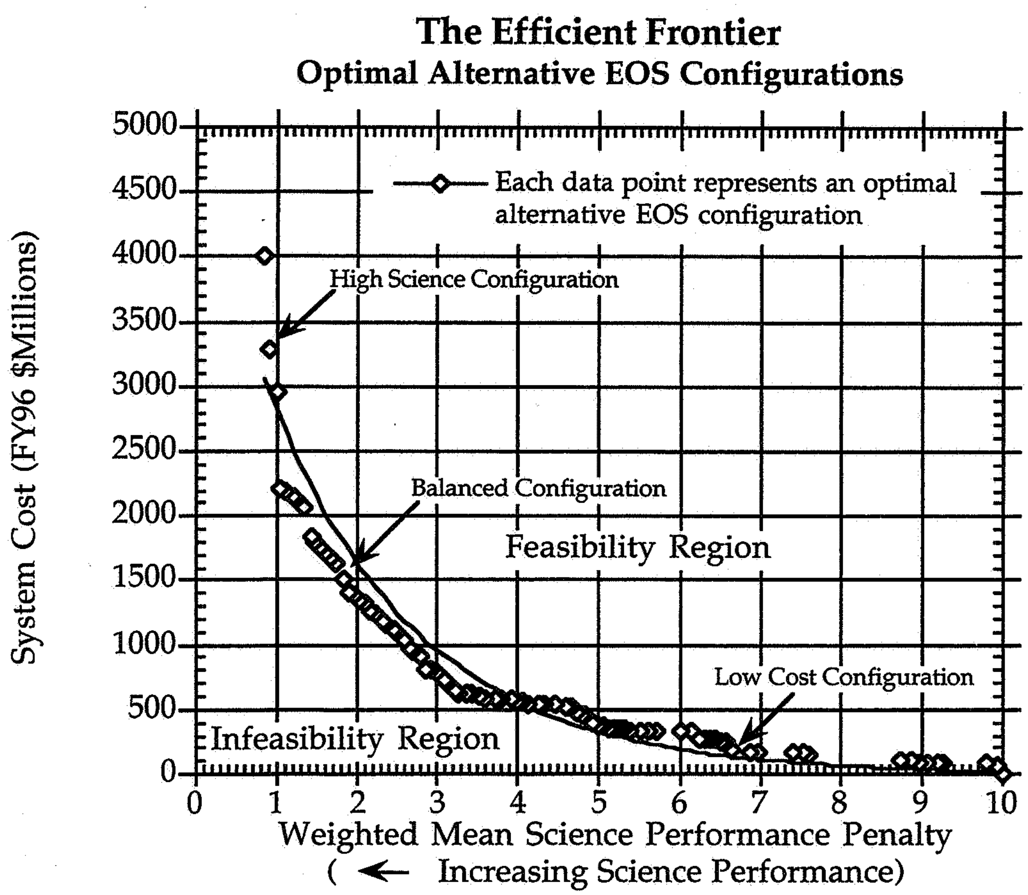

Figure 5.0 This graph indicates the most efficient optimized designs on a performance per cost basis nearest the origin. The "Balanced Configuration" is the same as that shown in Tables 4.0, 4.1 & 4.2.

multiplier to relate the penalty and the cost units. This scaling factor can be set to indicate the desired weighting for cost vs. science and is a powerful feature of the model. Alternatively, a search routine can be activated to find all the discrete solutions throughout the range of performance vs. cost weightings.

Risk tolerance is addressed in both cost model as *cost risk*, and also by adjusting the redundancy and penalty factors in the model formulation..The cost risk extension to the Goddard Multi-variable Instrument Cost Model is integrally embedded within the DTC model. This enables a capability to set the instrument cost risk tolerance of the optimal solution. Other ways to manage risk include adjusting of the payload mass and power fractions, and manipulating the penalty calculation coefficients which dictate the measurement reliability and science robustness standards.

Some of the highlights of the sensitivity analysis relating to performance and cost are presented below, however a more complete treatment may be found in [Matossian][5]

5.1 The Efficient Frontier

At the two extreme ends of balancing science performance and cost are either configurations that are relatively inexpensive

Tables 5.0 & 5.1 The optimum DTC configuration design for an extreme science performance emphasis. (Corresponds to High Science Configuration in Figure 5.0.)

Payload Table

Launch Vehicle	Instrument Stability	Viewing Target	Instrument Payload
Taurus	mixed	solar+	ACRIM SOLSTICE_II GOS IPEI
D7930	passive	Earth	AMSU-A NSCAT_II GLAS SWIRLS LIS
AIIAS	mixed	Earth	HIRIS ASTER MODIS-N/AIRS TES XIE
AIIAS	passive	Earth	ALTimeter/MHS EOS_SAR

Spacecraft Specifications Table

Spacecraft Class	Mass P/L Kg	P/L Usage	Mass S/C Kg	Power P/L W	P/L Usage	Power S/C W	Data Kb/s	Cost 96$M
Taurus	255	89%	1020	146	91%	365	30	181
D7930	665	71%	2660	873	65%	1746	218	554
AIIAS	1651	91%	6604	1649	73%	3298	221130	1024
AIIAS	1441	80%	5764	1917	85%	3834	180085	1540

but do not address the science goals of the mission, or costly designs that utilize as many of the available instruments, spacecraft and launch vehicles as necessary to fulfill the mission science goal. Figure 6.0 displays the full range of the *optimal* DTC configuration designs for EOS. Of greatest interest is the "elbow" of the curve. This is the most sensitive design area of the curve, and indicates not only the most efficient configuration on a performance-per-cost basis, but also the point of diminishing return for budget cuts.

The sample output used for illustrative purposes in the previous section can now be viewed as an important solution. With a weighted mean penalty of 1.8212 and a cost of 1996 US$1,405.3M, it is well positioned at the most efficient performance/cost region in the "Efficient Frontier" graph.

Generation of a single Efficient Frontier curve for EOS, such as that shown in Figure 5.0, required 4-12 hours runtime on a dedicated 250 MHz DEC Alpha dual processor workstation. The software configuration included CPLEX v3.0 (COTS MIP solution software) and our proprietary software with an extremely efficient MIP formulation of the EOS system.

5.2 Performance vs. Cost

The balance of performance and cost and its effect on the configuration is readily apparent through examinations of a science-weighted solution, and a contrasting configuration with a cost emphasis. A science-oriented solution can be found in Tables 5.0 and 5.1, and is denoted as the "High Science Configuration" on the vertical section of the Efficient Frontier curve. This solution projects a cost of US$3,297.5M to achieve a 0.8518 weighted mean penalty. This is the best science configuration that can be flown with the available resources of instruments and the reliability requirements for system robustness. The non-zero value for the weighted mean science penalty indicates that there are insufficient resources in the set of candidate instruments to completely satisfy all the mission goals for science and reliability. The vertical section of the Efficient Frontier is then revealed to be the extremum point, and not the approach to an asymptote.

The heavy weighting on science in the optimization resulting in Tables 5.0 & 5.1 is immediately apparent in the inclusion of larger Atlas IIAS boosters. It should also be noted that complementary instrument pairs (MODIS-N/AIRS & ALT/MHS) were selected for flight.

Tables 5.2 & 5.3 The low cost - but low science - configuration while maintaining risk tolerance standards. (Corresponds to High Science Configuration in Figure 6.0.)

Payload Table

Launch Vehicle	Instrument Stability	Viewing Target	Instrument Payload
Taurus	mixed	Earth	MODIS-T EOSP SAGE_III

Spacecraft Specifications Table

Spacecraft Class	Mass P/L Kg	P/L Usage	Mass S/C Kg	Power P/L W	P/L Usage	Power S/C W	Data Kb/s	Cost 96$M
Taurus	229	80%	916	159	99%	397	3264	183

The heavily cost-weighted configuration shown in Tables 5.2 and 5.3 is a bare minimum mission consisting of a single Taurus-class spacecraft. As an optimal solution for that weighting, the payload resources usage is very efficient, but the solar mission and many other instruments are not selected. At $183M, this solution costs far less than the $3+ billion dollars for the science-weighted configuration, though its weighted mean science penalty dramatically rises nearly 780% to 6.6492. It is interesting to note that the power budget becomes a limiting factor in this austere program, but it would be best not to generalize based solely on this data point.

6.0 Results

The sensitivity analysis on the design of a complex system such as EOS is a somewhat daunting task, however it is made much easier through the use of a DTC model. A DTC sensitivity analysis was conducted as a proof-of-concept exercise that contrasted instrument development cost risk, payload mass fraction, payload power fraction, spacecraft propellant fraction, instrument performance assessments (primary vs. secondary thresholds), and combinations of the above to capture conservative and aggressive risk tolerance in the design.

6.1 Smaller, Faster, Cheaper?

The notion of conducting future NASA space missions with smaller and less expensive spacecraft that require shorter development times has become the NASA mantra: "Smaller, Faster, Cheaper." But is this always the best approach? For the EOS Program, this question is particularly germane because the configuration started with Titan-class spacecraft, moved down to Atlas-class, was then decreed to shrink further to the Delta by order of the NASA Administrator, and is contemplating more extensive use of smaller "lightsat" spacecraft. In the absence of qualitative data, it is difficult to form a decisive argument either way. With the DTC configuration optimization algorithm, we can now shed some light on this debate.

In Figure 6.0, multiple efficient frontiers were calculated with greater and greater restrictions on the size of the largest spacecraft. Again using neither overly agressive nor overly conservative design parameters, the graph of Figure 6.0 can be seen in the unrestricted case with the full choice of all the available US launch vehicles. The largest launch vehicle selected with these Baseline Specifications[6] is the Atlas IIAS. By eliminating the Atlas as an option, the "Delta or Smaller" curve indicates slightly higher expenses in the $2.0-2.4B range. The extreme cases of the leftmost Baseline data points are simply unobtainable with the smaller rockets and spacecraft because they simply cannot accommodate the most demanding instruments. The third Efficient Frontier curve is further restricted in its spacecraft size to the LLV-1, Taurus- or Pegasus XL-class. This results in a far smaller range of configurations due to the

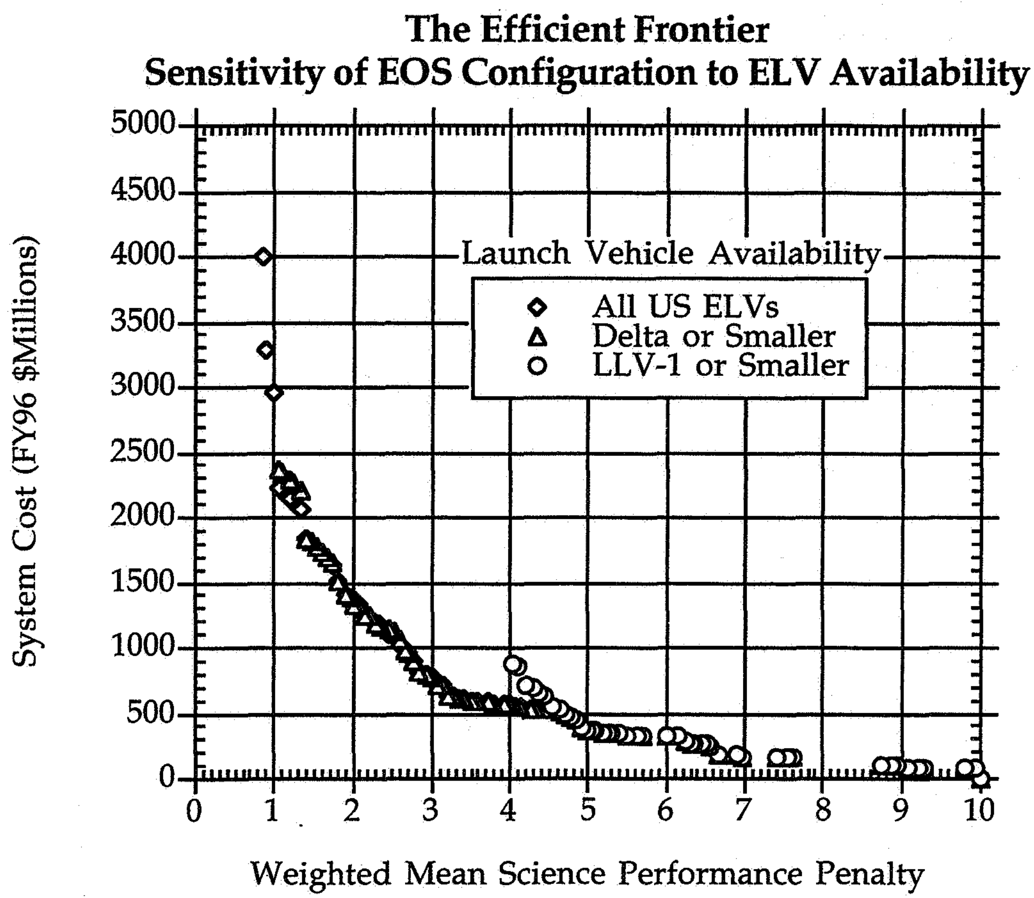

Figure 6.0 This Efficient Frontier graph illustrates the effect of limiting the launch vehicle choices in configuring "an" optimal EOS. The most desirable solutions, nearest the origin, consist of Delta-class spacecraft with complementary smaller spacecraft.

more acute restrictions on instrument payload mass and power. In summary, a configuration consisting of Delta-class spacecraft with smaller complementary spacecraft in the $600M to $1.8B range (in "cost-model dollars") would be most efficient on a performance to cost basis.

6.2 Realistic EOS Re-design Considerations

The Baseline Specifications makes the instrument selection from the large, original pool of candidate instruments. However, program re-design exercises resulted in "down-selections" of instruments that reduced the size of the candidate instrument pool by eliminating instruments from consideration. These eliminations were based on a number of considerations, however Figure 6.1 shows that if the eliminated instruments were meeting specifications and their costs were in-line with their capability, the down-selection process eliminated instruments used in the DTC model's optimal configurations. The Efficient Frontier curve for consideration of only "Current EOS Instruments," deviates from the Baseline at the beginning of the desirable "elbow" of the plot, and becomes even more disadvantageous in the "high-science" region towards the left of the graph.

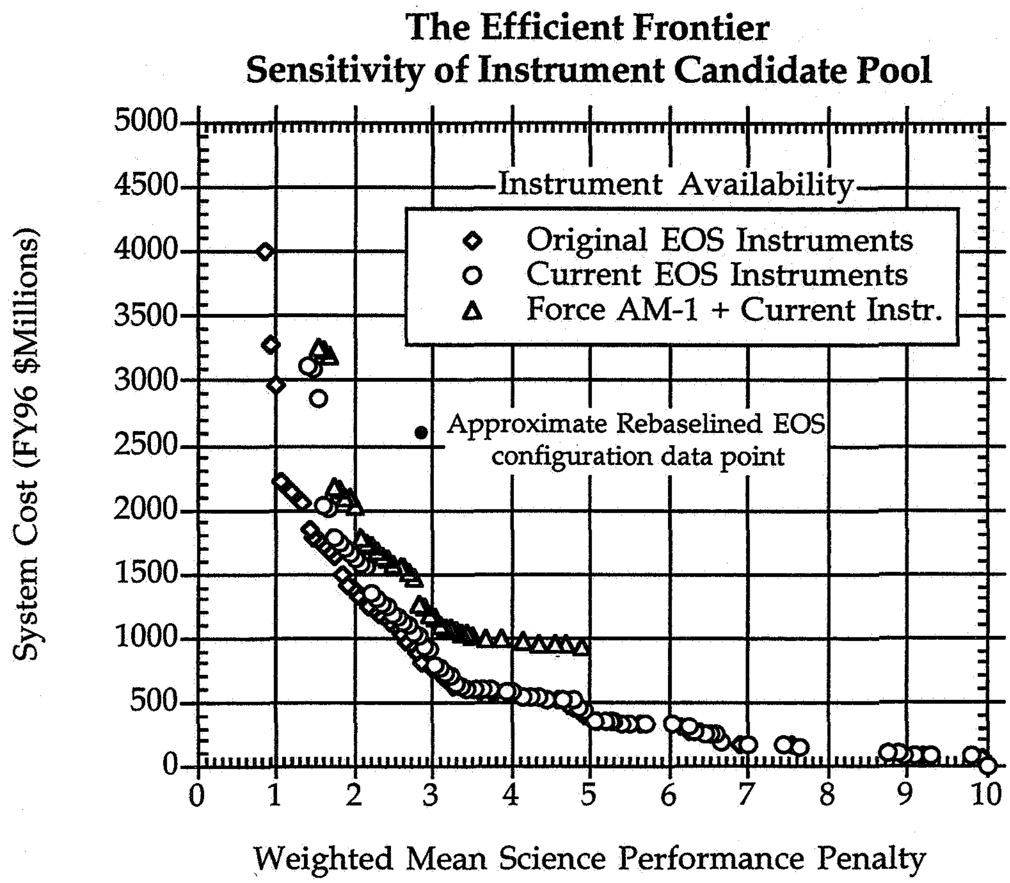

Figure 6.1 Without a systems-oriented optimization tool, it is difficult to know which instruments to eliminate during the original selection process or during downsizing exercises. While the model does not capture all the many factors taken into consideration in instrument selection, the closest analog to the "Rebaseling" re-design process would consider only the surviving EOS instruments as candidates and force the use of the AM-1 spacecraft in the final configuration. NASA's actual Rebaselined EOS configuration is included above, and does not compare favorably with the efficient frontier graph for the analog.

In addition, the re-design process that resulted in the current Rebaselined EOS configuration can also be simulated. Special constraints were added to the DTC model formulation to force the selection of the AM-1 spacecraft, which was already under construction during the Rebaselining and thus was removed from consideration. The model then was run to derive the optimal configurations given the handicap of the (non-optimal) AM-1 spacecraft in the configuration solutions. Even with this severe restriction of including AM-1, we can see that the curve continues to move further out from the origin, resulting in configurations with poorer performance and higher cost.

The most startling result is the score of the actual Rebaselined EOS configuration in relation to the Efficient Frontier. By forcing this configuration in the same manner described above for only the AM-1 spacecraft, and also not allowing any optimization to occur, we can derive the apparent science performance and projected cost (within the bounds of the cost models considerations) of the Rebaselined EOS configuration. This allows to us to compare "apples and apples" with the Rebaselined EOS weighted mean science penalty of 2.9 and a cost of $2.6 billion "cost model" dollars. While this comparison is approximate at best, it highlights the fact that NASA has never conducted a formal EOS configuration optimization program despite multiple re-design exercises.

The Efficient Frontier curve for the forced AM-1 solution in a configuration chosen from only the surviving current instruments, shows substantial room for improvement from the Rebaselined EOS configuration. For the same science capability, a 50% reduction in cost appears attainable in the $1.3 billion dollar configuration directly below the Rebaselined data point. With the larger, original candidate instrument pool of the Baseline Specifications, the savings climbs to 70%. Conversely, a similarly huge 1/3 reduction in the weighted mean science penalty can be obtained by selecting a configuration to the left of the Rebaselined EOS data point - and for $400 million dollars less. Again, the Baseline Specification's Efficient Frontier curve shows even further science performance improvement.

The actual EOS Rebaselining process was performed without any optimization tool such as the DTC algorithm, and the argument for improvement in science performance and/or cost savings is irresistible.

7.0 Conclusions

The Distributed Task Constellation strategy for designing systems of dissimilar spacecraft would be useful if it simply identified feasible configurations. However, its real power lies in its capability to go beyond simple feasibility and to actually *optimize* large and complex constellation design problems for *provably* global optima. A DTC model gives a constellation architect a powerful, flexible tool withwhich to balance performance versus cost, and to constrain the solution space by setting budgetary and other limitations. Furthermore, there are a variety of opportunities to manage the risk, including instrument cost, spacecraft parametric relationships, and data-gathering robustness. This is the first time that a constellation design optimization tool based on powerful operations research techniques has ever been developed.

The range of optimal performance per cost solutions can be combined to map out the entire Efficient Frontier of constellation spacecraft configurations. This curve of optimal solutions dramatically illustrates the best performance to cost ratio configurations, and the regions of diminishing performance or cost return on investment.

The Earth Observing System is one of the most complex DTC applications due to it's scale and complexity. Thusfar, the EOS program has not employed system architecture optimization strategies such as the DTC method, which continues to hinder its congressional support and encourage re-designs. The EOS Efficient Frontier graphs, though approximate, show a dramatic opportunity for such optimization methods to improve both EOS science performance and decrease costs.

The DTC approach shows much promise for adaptation to other non-constellation space mission design problems, and is an excellent phase A/B design tool that can easily and quickly adapt to changing fiscal, technical and policy influences effecting the design. Planned future developments include the implementation of the time dimension to enable optimal project scheduling, as well as other enhancements. In the tight fiscal reality of industry today, design optimization which methods that realistically blend cost and performance with managed risk *are* the future of the aerospace industry.

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[1]Matossian, Mark G., "Earth Observing System Mission Design: Constrained Optimization Of The EOS Constellation Configuration Design," *46th Congress of the International Astronautical Federation*

Proceedings, October 1995, paper #IAF-95-A.6.02.

[2]Dixon, Bernard, and Villone, Paul A., "Goddard Multi-variable Instrument Cost Model," Research Note 90-1, Resource Analysis Office, NASA/Goddard Space Flight Center, May 1990.

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[5]*Configuration Design Optimization of Multi-Satellite Distributed Task Constellations*, Department of Aerospace Engineering Sciences Doctoral Dissertation, University of Colorado at Boulder, December 1995.

[6]Ibid, Section 12.1.

S28-32

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P1

RAIN ATTENUATION IN SATELLITE COMMUNICATIONS

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The propagation of satellite communications signals is influenced by a variety of factors, of which the free space loss is the most important. Rain, however, can severely affect a communications link from time to time, especially for frequencies over 10 GHz, and has to be taken into account in the design of a satellite communications system. This is done by calculating a rain margin for a given link reliability using empirical rains models. This paper describes the way in which rain attenuates a signal by absorption and scattering, and discusses two of the most commonly used methods for calculating rain margins: the Crane model and the CCIR model.

OMIT THIS PAGE

Session 10
Satellites Applications

S29-31

ABS ONLY
020741
290013
P1

COTS SOLUTIONS IN THE ENGINEERING WORKPLACE

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Over the past several years, Commercial-Off-The-Shelf Software (COTS) products have become increasingly accepted as important tools in the engineering environment. From conceptual development to real-time operations, COTS have filled a niche because of their ease of use, standardization, continuous upgrades, low-cost maintenance, and low prices. Much of this success is due to the wide user base which allows COTS development companies to add in features from thousands of users, while having the software tested and validated by these same users. All involved have input into the software and a partial stake in it instead of the entire responsibility for funding, development, and maintenance laying in the lap of one user organization. One such example of a successful COTS product is Satellite Toolkit and its complementary software tools, from Analytical Graphics, Inc (AGI). Over the last eight years, STK has developed into a tool that is increasingly relied upon by organizations such as NASA, NOAA, Lockheed-Martin, Boeing Aerospace and the Air Force, to name a few. Various examples abound how the STK suite of software tools have supported the Hubble Space Telescope Reservicing Mission, launches of payloads by Orbital Sciences Corporation, support of the NEAR and SAMPEX scientific missions. In these cases and countless others, the COTS paradigm, as embodied in STK, has served missions from the design and planning periods through the launch, operations, and data gathering phases All this has been done in an era of tight budgets and higher expectations, exactly the conditions in which COTS can thrive and have indeed prospered. This paper will examine several cases involving STK and its successful entry into engineering organizations that once relied upon large, complex, and single-user development efforts to satisfy their software needs.

S30-18
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290030
p 15

EVOLUTION OF A SATELLITE SERVICE FACILITY IN EARTH ORBIT

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Team Project, Fall 1995, Space Exploration Architectures Concepts Synthesis Studio, Dept. of Aerospace Engineering, University of Southern California, Los Angeles, CA 90089, *Participant, **Instructor/Coordinator

Abstract

This project depicts an evolving vision of an Earth orbital facility that would gradually build up our capability to repair and service satellites in distress. Starting with orbital debris mitigation operations and missions like placing spacecraft in their intended orbits, station relocation and plane change operations or safely decommissioning aging and useless stations in orbit, the proposed orbital infrastructure will grow to be able to handle tasks like the specific replacement of malfunctioning or degraded subsystems aboard suboptimally operating satellites in order to enhance their performance and life expectancy as well as evolve their capability, using a modular strategy. Preliminary studies are presented as to the nature of the projected Earth orbital environment by the turn of the century and expected mission opportunities for this architecture are explored. Alternative concepts for satellite repair facility configurations are depicted and related issues regarding the maintenance and evolution of such an infrastructure are discussed. Potential projects stemming from this open-ended architecture are also portrayed.

Introduction

The Earth orbital regime, stretching from the low earth(LEO) to the geosynchronous(GEO) orbit is already a rather crowded environment with over eight thousand operational, aging and decommissioned spacecraft and related deployment hardware objects. Less than ten percent of this number are functional stations. Artificial debris arising from the degradation of discarded hardware are threatening the safety of operations in this environment. Projecting into the next decade and into the new millennium, several planned constellations of commercial communication satellites in low and medium earth orbits(MEO) will most definitely increase the traffic in the LEO to MEO Earth Orbital environment as well.

Extrapolating from past experience, we can expect to see a proportional increase in spacecraft failures and anomalies during all phases of deployment including launch to orbit, orbital transfer maneuvers, orbit changes, station keeping operations and decommissioning procedures.

In the context of increased activity in this regime, it may be prudent and economically feasible now to develop and establish a permanent space based infrastructure in order to repair, service and evolve spacecraft in a systematic effort to enhance performance, as well as maintain order and stability of these stations in the increasingly crowded Earth orbital environment.

Earth Orbital Environment 2001

Looking at the present trend as well as the unprecedented applications blitz at the International Telemetering Union(ITU) for satellite licenses, we speculate ever increasing traffic in the Earth Orbital Environment for the next decade. Commercial telecommunications and defense surveillance satellites designed to deliver a plethora of ingenious new services will dominate the arena. Unless entirely radical and unforeseen methods of Earth-To-Orbit(ETO) deployment are employed, we will continue to see a normal rise in the number of failures due to normal accidents during deployment and other phases of satellite operations. In addition, a large number of stations in the so called "Clarke belt" along the geostationary orbit will reach their end-of-life by the end of the coming decade, thereby becoming potential targets for additional debris production. Consequently, unless prudent measures are taken now, we will most definitely also see an even more polluted and dangerous orbital debris environment all the way from Low Earth to Medium and Geosynchronous orbits and beyond.

Though ETO insurance premiums might become more and more affordable to users due to the increased activity in this business, the monetary aspects of deployment and operations do not suggest any long term beneficial effect at all on this orbital debris scenario. In fact, all indications are to the contrary. Besides, affordable premiums and the like are no substitute for an on schedule, healthy, operating satellite, because time lost in mounting another campaign(two to three years) can leave a service provider without a viable business, in this fiercely competitive industry.

Unless innovative new systems are put in place even as we plan to exploit the benefits of high bandwidth global personal communications and a host of other applications, we can expect to see substantial impairment of station performance on a global scale, and consequently, an eventual deterioration in the quality of ubiquitous services provided by stations located in the Earth Orbital regime.

Architectures are needed that will continually monitor, assist, correct, repair, remove and dispense of stations and debris in a manner that will enhance traffic in this regime. Several recent studies by NASA and contractors suggest the need for this capability. The Orbital Maneuvering Vehicle(OMV) proposed some time ago by NASA falls into this category. More recently, the European Space Agency(ESA) has also conducted a study for a satellite for on-orbit repair and maintenance of stations. The proposed Satellite Service Facility(SSF) is envisaged as such a system architecture, that draws on the latest developments in technology, to achieve it's goals.

Architectural Elements Of The Satellite Service Facility(SSF)

The elements that make up the SSF architecture are as follows.

1. The Space Tug Mk-1(ST-1)

The ST-1 is a basic phase 1 interorbital maneuvering vehicle with a protected payload bay. Otherwise costly plane changes are done using a unique aerodynamic assist maneuver coupled with a high specific impulse solar electric propulsion system. Its primary mission is to boost incorrectly injected or stranded spacecraft to prescribed station orbits. Its secondary mission is also to rendezvous payload satellites with the STS for manned intervention or return to Earth as well as to return repaired payloads to desired orbits. Deorbiting and safely disposing large decommissioned objects like spent stages and other spacecraft deployment articles through controlled atmospheric ablation and disintegration is also part of ST-1 mission operations.

2. The Space Tug Mk-2(ST-2)

The ST-2 is a more sophisticated system that is evolved as phase 2 of the SSF architecture. This larger vehicle is capable of all ST-1 missions as well as on-site repair and maintenance activities of stations in all orbits. The ST-2 tool kit offers on-the-spot diagnostics, orbital replaceable units, and a high bit rate communications link that allows teleoperation of a highly dexterous set of robotic manipulator systems(RMS).

Both ST-1 and ST-2 are capable of large plane changes employing a unique, detachable aerodynamic plane change assist shell that protects all subsystems and the payload from the extreme thermal loading during these maneuvers. All exposed systems including solar arrays, communication system antennas, observation cameras are designed to fit within or retract into the shell during the plane change operations. The results from the scheduled Aeroassist Flight Experiment(AFE) scheduled to fly soon aboard the NASA space shuttle would be used to optimize the shell design. Data from the recently retrieved Long Duration Exposure Facility(LDEF) as well as the low cost Advanced Photovoltaic and Electronics Experiment(APEX) will be used in selection of materials and system components for the synthesis of this vehicle.

Both ST-1 and ST-2 have similar spacecraft buses in that they share the same kind of photovoltaic array power system with back up batteries, uses a modular hybrid electric/cold gas propulsion system and a high bandwidth communications system. A set of cameras and remote manipulators are also common to both spacecraft. They are both designed to be refueled and serviced on-orbit. The aerodynamic plane change assist shell has a small avionics package that supports its maneuverability during the parking orbit period when it is separated from the rest of the spacecraft and payload.

Advanced Technologies Identification

Advances in electric propulsion as evident in the Hughes XIPS station keeping system as well as the NSTAR system for the New Millennium Program(NMP) Deep Space 1(DS1) is pushing the envelope in high specific impulse electric thruster technology. Advanced cascade solar cell arrays will deliver the power needed to drive these engines. Modular clusters of these engines coupled with the unique aerodynamic maneuver using an advanced aerobraking shell should provide sufficient energy needed for plane change operations. Optical communication systems are employed for compact, high efficiency, interference free, inter-tug, mission control and related high bandwidth links. Autonomous guidance and navigation and high fidelity teleoperation made possible by the advent of super computers, biologically accurate, self educating neural algorithms, highly dexterous mechanical manipulators and end effectors, as well as high strength, light weight alloys and composite materials and the design of smart/hot structures make possible an intelligent spacecraft structure, that is able to maneuver, articulate, capture and redeploy target satellites and objects efficiently in space and at the Earth's atmospheric boundaries, enabling it to conserve fuel and power for repeated cyclic operations.

A Synergetic Supporting Architecture

Five major systems, both existing and envisioned for the coming decade and beyond, are expected to be of synergetic relevance to the SSF architecture evolution. They are:

1. The existing manned national Space Transportation System(STS)
Though SSF will provide its service mainly in a robotic mode, we expect manned intervention for certain tasks. In the primary stages of evolution, phase 1, these manned tasks will be performed in the vicinity of the STS. The SSF/payload configuration will rendezvous with the STS, and crew can access the payload in EVA. Crew may also retrieve payload for a return trip to Earth. All of these mission scenarios, less the SSF, have been accomplished in past STS missions. The Intelsat rescue mission, the LEASAT mission and the Long Duration Exposure Facility(LDEF) retrieval mission all provide precedents to ST-1 mission profiles.

2. Advanced Tracking and Data Relay Satellite System(ATDRSS)
The command, control, communication, and intelligence(C3I) system is crucial to the successful operation of any complex architecture. By allocating a schedule that does not conflict with other NASA activities, the ATDRSS constellation and associated ground segment would have the capability to provide mission control for ST-1 and ST-2 missions. It is envisaged that as the demand for bandwidth increases with the need for multiple missions and in order to accomplish more complex tasks, the SSF architecture will transition to a dedicated telecommunication system. It is also foreseen that radio interference can be expected while conducting operations in the vicinity of target stations. Optical links are suggested as a way to avoid this problem.

Recent experiments using the NASA Advanced Communications Test Satellite(ACTS) promises to open up the Ka Band for high bit rate communications. Optical communications offers the promise of even more bandwidth. Along with advances in the field of real-time telerobotics, we expect to be able to employ very sensitive remote manipulator systems on board the ST-2 to do highly dexterous tasks like handling orbital replaceable units(ORU) on stations without manned intervention.

3. The Orbital Debris Removal System
A system capable of continually keeping track of and eliminating debris is needed. As the orbital environment becomes more and more crowded with stations of all kinds including scientific, defense related and commercial, debris hazard to these stations will also become worse. The ST-1 and the ST-2, between missions, may be used in coordination with an orbital debris removal system architecture to speed up the debris detection and mitigation process. Such an automated orbital debris removal system is being proposed in a separate paper at this conference.(SPACE 96, ASCE)

4. An Artificial Gravity Facility in Earth Orbit
International Space Station(ISS) is a being designed as a facility devoted to the study of human biology in prolonged weightlessness and other microgravity research in the physical sciences. In the decade after the commissioning of ISS, a facility to study the effects of induced gravity on the human system may become operational. The research conducted on board would be invaluable in designing truly long duration spaceflight vehicles. Crew aboard this orbiting facility may be synergetically employed for enhancing SSF missions. In the advanced phase of operations, we speculate that more complex and time consuming manned satellite service missions could be accomplished on board this platform.

5. Lunar Infrastructure Development Architecture
Growth, transformation and the budding of new architectures is a dynamic part of the normal process of evolution in any complex system architecture. Using the advanced SSF ST-2 as a stepping stone, it is possible to leverage that

infrastructure to begin the exploitation of the moon. Starting with the removal and disposal of orbital debris and useless, decommissioned stations in high energy orbits including GEO to safe allocated sites on the lunar surface, we envisage the development of a polar orbiting lunar station that would eventually become an orbital logistics gateway to the lunar surface. This unique orbit provides total accessibility to the lunar surface during the lunar period of rotation. Since the moon orbits around the Earth in a twenty eight day cycle, this lunar polar orbit gateway station also can be used to inject payloads into a variety of useful Earth orbits without the costly penalty of plane change maneuvers. The ST-2 and other orbital transfer vehicles would ply cislunar space between Earth and this lunar polar orbit, while lunar lander shuttles would rendezvous periodically with this gateway station.

6. Spacecraft Salvage Operations Architecture
The deployment of the SSF architecture has immediate ramifications for global satellite operations. A spacecraft that is able to deliberately alter the station of another has far reaching legal implications. New regulations and the creation of an international body to oversee the activities of such a system will be needed. Issues dealing with ownership of salvaged spacecraft, their reuse or the manner of their disposal, the effect of such activity on other stations, guarantee of operational safety in orbital regions allotted or licensed to other parties will all have to be examined and modified to accommodate the smooth operation of the SSF architecture.

Mission Design and Operations

ST-1 Mission Operations

In phase 1, ST-1 is launched into orbit on a Titan IV class vehicle. After check out it is parked in orbit. It could be called upon to assist in a global orbital debris cleanup campaign while not being mission tasked for satellite rescue.

Immediately after an accident where a satellite is injected into a wrong orbit due to a launch or upper stage failure, the ST-1 mission operations are initiated. First of all, the ST-1 engages in a aeroassisted plane change maneuver to approximate the orbital plane of the target satellite. Next, it disengages from the aeroassist shell and leaves it in a parking orbit. Communications antennas/telescopes and solar arrays are unfurled and cameras are exposed. A high bandwidth communication link is secured with mission control and the vehicle proceeds to capture the satellite.

Using well known rendezvous procedures that are routinely employed today, the SSF will use phasing orbits, homing and terminal approach maneuvers, to gradually collocate with the target satellite using on-board electric thrusters. All operations in the vicinity of the target satellite are conducted carefully using selected systems without polluting the immediate environment of activity. Using a small, clean, inert, non-polluting cold gas attitude control system, the remote manipulators and grapples, the satellite is firmly attached to the payload bay of ST-1.

After integrity check out, the total system gradually returns to dock with the aeroassist shell.(Using this approach, the additional mass of the aeroassist shell does not compromise the payload capacity of the ST-1.) After docking with the shell and integrity checkout, the entire assembly does another aeroassisted plane change maneuver to bring the payload into the desired orbital plane. Again, the shell is disengaged and parked while the rest of the ST-1 and payload is slowly thrust into desired orbit. Maneuvers are repeated at the end of the operation to unite ST-1 and the aeroassist shell at the end of the mission.

An application that needs to be studied further is the possibility that a fleet of such vehicles parked in LEO might be used regularly and frequently, as tugs are in the maritme industry, to slowly boost all spacecraft to their desired orbits in a very clean and reusable operation. All pollution and debris associated with orbit transfer maneuvers, their motors, casings, ullage gases and unspent tankage could be eliminated. Such an application of the SSF architecture could have wide ranging implications to the scope and economics of this infrastructure.

Also, this staging procedure in LEO would allow us to thoroughly check out a payload that has just survived the severe ETO launch environment, before sending it gently away to its destination, using the low but constant thrust of the ST-1 electric engines. This concept of using low thrust engines also opens up a new area of spacecraft design where very large yet fragile craft like optical interferometric telescopes and extensive solar sail powered vehicles, or even wide fields of arrays for solar power satellites may be assembled in low Earth orbit and slowly placed in their proper orbits.

However, if we use these low power electric thrusters, we would have to deal with the aspect of the payload being severely exposed to the Van Allen radiation belts through the course of this procedure. Since stations headed to GEO for instance, are sufficiently radiation hardened, this should not pose a big problem.

ST-2 Mission Operations

In Phase 2, the ST-2 vehicle may be carried up on a Titan Class vehicle and deployed fully and checked out with assistance from the crew of a co-orbiting space shuttle in a dedicated STS mission.

The ST-2 is envisioned as a more capable vehicle. It is a larger version of the ST-1 with a more dexterous remote manipulator system and a kit of tools allowing it the ability to do maintenance and delicate repair and replacement operations on spacecraft located in a range of orbits from LEO to GEO. Lessons learned using ST-1 operations are used in the design of ST-2. It may be initially commissioned to be parked in GEO where it might slowly move along the GEO belt assisting the constellation in such activities as temporary station keeping, visual examination of stations for various purposes, stabilizing anomalous conditions aboard stations, providing auxiliary power, removing and replacing ORUs, cleaning and maintaining thermal louvers and blankets and removing decommissioned satellites from orbit. The prevailing technique of moving useless stations to graveyard orbits where they eventually deteriorate and add to debris through accidental exposure to micrometeors or other debris will be totally eliminated. It is the consensus of the USC group that is a prudent philosophy to de-orbit spent stages and other large deployment related hardware, before they break up and scatter more debris over large regions.

Both ST-1 and ST-2 are envisioned as fully reusable spacecraft. Depending on the energy needed for each mission, these vehicles are designed to carry out between five and eight missions before refueling and service operations. These operations are carried out in LEO, possibly in the vicinity of a SSF dedicated STS mission or on board the advanced artificial gravity facility. Modular cold gas and electric thruster propellant tankage units are replaced, all docking and other mechanical hardware including solar array and camera drives are checked out, the aerodynamic shell is tested for integrity and the SSF is ready for its next mission.

Merits and Limitations

A reusable Satellite Service Facility like the ST-1 and ST-2 offers a new way to manage our resources in Earth orbit. Though the demand for satellite rescue missions are quite unpredictable at this time, we can expect to see a sizable number of events in the coming decade, given the projected activity. Such rescue missions could save an otherwise perfectly healthy satellite and circumvent the need for mounting another long launch campaign that could cost the owner their business.

The ability to replace subsystems on board suboptimally operating stations is a highly desirable activity for which an infrastructure is lacking at the present time. The recent spate of subsystem anomalies aboard stations like the Anik, the Palapa C1, AsiaSat 3, JCSat 3 and the AMSC-1 clearly suggest the growing need for such a system.

Besides filing insurance claims for loss or damage, the current strategy, at least in the deployment of multi-satellite constellations like the Global Positioning System(GPS) and the proposed Big LEOs like Iridium and Teledesic, is to provide several fully functional spares on orbit that are turned on by telecommand to take over from a faulty station. However until that need arises, as mentioned before, these spares add to the threat of causing more debris. The SSF offers a different "fix it or safely dispose of it" philosophy and could be a timely architecture for the coming decade and beyond, given the number of spacecraft planned for launch as well as those expected to reach end-of-life during this period.

The propellant penalty of plane change operations has been a prohibitive factor against the development of this sort of architecture. However, combining the emerging technologies of high specific impulse, clustered, low thrust electric engines powered by high efficiency solar arrays employing cascade cell technology and the concept of aerodynamic steering using light weight high strength, composite, hot structures technology, it should be possible to overcome this difficulty.

The term reusability is still not fully applicable to the aerospace world. Reusability is the key to more economically viable architectures. The STS employs a partially reusable architecture. We have to strive for more reusability in all systems. The SSF is designed for reusability. The ST-1and ST-2 will need to be serviced and refueled on orbit. This technology has yet to be perfected and is on the critical path for any major achievement in space.

To assure the success of the SSF architecture, it is imperative that new standards be drawn up and followed closely in the design and engineering of spacecraft for serviceability. Once the design and engineering parameters for the SSF elements at the components level are established, satellite production lines will have to slowly begin the transition to incorporating serviceability features into the assembly sequence. These include the provision of hooks and scars to accommodate Orbital Replaceable Units(ORUs), multiple hard points for grabbing, anchoring and latching on to the SSF, mechanisms that enable deployed satellites to retract solar arrays and antennas back to a stow away configuration, as well as compatible designs for a variety of Remote Manipulator Systems(RMS) and other end effectors on board the SSF elements, just to name a few features. The need for ambient lighting control in the proximity of the target payload during multi-orbit operations is also a factor of importance.

Recommendations

1. New and tighter international regulations are needed to curtail production of orbital debris from current satellite deployment procedures. Though providing spare stations in orbit makes sense economically, it is a dangerous practice, from the environmental point of view, however slight the mathematical probability of explosive disintegration due to a variety of reasons.
2. Begin deployment of orbital debris mitigation systems as soon as possible. It is the consensus of the USC team that all potential debris producing bodies be de-orbited before they the disintegrate and scatter debris over large regions, that are practically impossible to eliminate. This includes deorbiting spent stages and

other deployment relate hardware. The proposed architecture can assist in this mission.

3. New debris-free deployment techniques are needed for satellites to supra LEO orbits. Furthermore, clean and non-polluting systems to be employed during target satellite proximity operations. The SSF staging and operations architecture suggests one strategy. Insurance premiums should reflect the cleanliness of deployment operations.

4. Start designing spacecraft that provide standardized hooks and scars to facilitate serviceability in orbit. They include easily accessible hard points for grabbing and anchoring, protected fixtures that will not fail during service/decommissioning procedures and the incorporation of mechanisms on payload satellites to retract deployed elements like booms and arrays during service and redeployment operation.

5. On-orbit refueling technology demonstration as well as aerodynamic braking/steering are high priority items on the critical path for this architecture to succeed. Build an SSF fleet of vehicles in a phased manner akin to ST-1 and ST-2 evolution that can be serviced and refueled on orbit.

6. Design versatility into fleet architecture for rescue operations, for ORU changeouts, as well as for orbital debris mitigation.

7. Design SSF fleet with interfaces to accommodate interaction with STS as well as an advanced manned station in orbit.

8. Establish an international body to oversee the activities of the SSF architecture. This same body would gather a pool of resources to subsidize the orbital debris mitigation program for companies willing to invest in this mission.

9. Multi-tasking may be the key to satisfying the economic requirements of such an architecture. The creative coupling of orbital debris removal and station decommissioning activities, satellite maintenance, inspection and modular upgrades and satellite rescue maybe the answer to an economically viable, versatile architecture.

10. Satellite deployment/operations insurance company policies to be reviewed and revised to provide new standards and more incentives for cleaner, safer campaigns all the way from "cradle to grave".

Economics of the Satellite Service Facility

Communication satellites costs 100 - 300 million dollars or more per unit to design, build and test. A typical launch campaign from satellite integration procedures through launch preparation, liftoff, transfer orbit injection, circularization and final checkout and handover to operators can easily amount to another 100 -200 million dollars. A typical conservative underwriter will insure the payload for between 20%-30% of the value of this total campaign. At a minimum, a full campaign from start to safe operation will cost the owner between 200 million and a quarter of a billion dollars.

Satellite deployment failures are hard to predict. However, in the last five years alone, several satellites have failed to reach prescribed station orbits. Launch failures and upper stage malfunction were responsible for many of these events. Though we cannot yet portray any definitive statistical model on how often a satellite might end up in this situation, a communication satellite that does not reach or cannot maintain its prescribed orbit, geostationary or any other, is considered a complete loss because it cannot operate effectively in any other regime, technically or legally. Such a stranded station, besides being useless and a liability, can threaten the orbital environment by explosively scattering debris upon accidental impact with other debris or micrometeorites.

Under these circumstances, the SSF becomes an attractive option in order to safely and efficiently place the spacecraft in its intended orbital slot. The typical rescue mission costs would be set to provide both a profit for the SSF service provider and a bargain for the satellite owner, both parties understanding fully well that lost time in mounting another campaign from the beginning could cost the satellite owner their business, in this very lucrative but fiercely competitive industry.

We estimate that a phased SSF architecture design, development, test and engineering(DDT&E) program comprising of three ST-1s and two ST-2s along with mission control infrastructure would cost about three and a half billion dollars. This capital investment is well within the reach of a large private aerospace company or a consortium to field. Building on high power satellite buses nearing production, the ST-1 could be built and flight tested in two years. The ST-2 might evolve in another three. A five spacecraft full-up SSF fleet could be operational shortly after the turn of the century.

Furthermore, between these unpredictable but highly profitable rescue missions, the SSF architecture would pay its way by assisting in debris mitigation activities, salvage operations, visual examination of spacecraft for verification of system or subsystem status for various technical and legal purposes, and for maintaining stations by regular operations that include cleaning solar panels and thermal control louvers, fixing thermal blankets, assisting in temporary station keeping, and decommissioning useless stations. These secondary, less dramatic, but continually available missions should at least pay for the upkeep of the SSF architecture.

Conclusion

The Earth Orbital Regime is getting crowded with satellites providing a host of services. Judging by the applications being filed for new licenses, we can expect to see a great number of stations being deployed in the next decade alone. Barring unforeseen new developments in deployment technology, we can also expect to see a sizable number of deployment failures, resulting in fully functional yet useless, stranded satellites, that will threaten the orbital environment by producing more debris.

The orbital satellite service facility architecture will provide the infrastructure and elements to rescue these satellites and deliver them to their intended orbits. Though such rescue events are considered quite unpredictable, every such mission can be very profitable to both the SSF service provider and the satellite owner. Also, between these missions, the SSF would pay its way through providing a host of other valuable functions including assisting in orbital debris mitigation and other satellite maintenance and salvage activities. Multi-tasking is seen as the key to economic feasibility.

We anticipate that the nature and scope of the SSF architecture and related activities will infringe on the technical and legal domain of satellite operators worldwide. We suggest the creation of an international body to gather the resources and to oversee the smooth operation of such a complex system.

The development of such an infrastructure can then be leveraged to include activities on the moon by first establishing a lunar orbital station for various purposes and then continuing to expand to our activity to the lunar surface and beyond.

Acknowledgments

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AIAA Journal of Rockets and Spacecraft
Acta Astronautica, Journal of the International Astronautical Federation
ESA Bulletin, ESA Journal, Publication of the European Space Agency
SPACE 88-96 Engineering, Construction and Operations in Space. ASCE
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OMIT THIS
PAGE

Session 10
Resources, Robotics & Manufacturing

PROGRAM COMPREHENSION:

MODELING ONTOLOGIES FOR PATTERN RECOGNITION

S31-61
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To understand a program, a software engineer calls upon a range of information from diverse sources. Understanding the system architecture and having information about the problem domain and common design cliches are some of the tasks required for program comprehension. Design cliches or patterns have a strong relationship with the architecture of software systems. They facilitate the process of understanding the structure and/or funcionality of computer programs, and could be used as a basis for reasoning, code generation, and measuring the goodness of a design. A pattern repository that can permit to store and retrieve patterns based on a description of the ideal architecture needed for the application aims, can provide a structure for a systematic method of choosing patterns for a particular application. Patterns in the repository would be associated with a structured data-set representing their meaning. This meaning should be match with the user's query when describing the ideal pattern structured description. Then the pattern description structure and the matching method need to be defined in order to facilitate the recognition of correct pattern for a specific architecture. The structured data-set representing the meaning of patterns can be developed as a set of terms like concepts, relations and attributes, belonging to an ontology. In this study the task to be performed is to model the data set of ontologies that define the patterns. A combination of patterns according to their architectural application is expected when the end user client inputs a query.

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THE NEAR EARTH ASTEROID PROSPECTOR: AN INNOVATIVE ALLIANCE FOR OPENING SPACE MARKETS & DOING CHEAP SPACE SCIENCE

Divya Chander[1], Mike Wiskerchen[2]

Since the Apollo missions, most of humanity's large-scale dreams regarding space development appear to have faded. In more than 25 years, we have not been able to even return to the moon. Our national space transportation system, while a remarkable feat of engineering, is still limited to low-Earth orbit. The International Space Station has become a bumbling political tortoise, and the various proposals by enterprising groups to reestablish a lunar presence, set up solar power stations, or pioneer a path to Mars are fraught with the problems of large development projects that lack an infrastructure extant.

The Near Earth Asteroid Prospector (NEAP) mission represents a novel approach to reopening the space frontier that capitalizes on the joint strengths of private enterprise and hands-on undergraduate and graduate education. The Space Development Corporation (SpaceDev), founded by Jim Benson, stands on the principle that the objects in near earth (NEOs) space represent an untapped wealth of accessible resources that today's markets remain largely undeveloped to exploit. Over the long-term, these resources are cheaper to develop than the moon's due to an insignificant gravity well. The study of these NEOs will also provide insights into the earliest origins of our solar system. At present, this valuable space science isn't being adequately done for the exorbitant costs of missions.

Therefore, SpaceDev's first venture is NEAP: to construct and launch a spacecraft capable of rendezvous and landing on various NEOs, while determining its composition and other scientific parameters. SpaceDev will go on to charter privatization of mining industry in space while developing an investment base and markets. The aim is to keep the cost under $50 million, including both spacecraft and secondary payload launch. The majority of cost savings comes from moving spacecraft design away from NASA and its large sub-contractors to groups like the California Space Institute (CalSpace), that will enable cheap and innovative development. CalSpace and the California Space Grant Consortium, both headquartered at the University of California, San Diego (UCSD) have facilitated a process of allying veteran scientists and engineers (from the private and public sectors) with undergraduate and graduate students under the national Space Grant program. A subset of this student and mentor team partnership has divided into teams that are working on the preliminary design of NEAP for Jim Benson. To date, we have developed an ~800 kg spacecraft with a three instrument payload to collect multi-spectral, gamma-ray, and alpha-emission data on the composition of the asteroid, including the potential presence of water. We have also calculated the Delta-V requirements to reach a sample of candidate Near Earth Asteroids with favorable orbital characteristics and compositions. In addition, we are discussing various launch vehicles, deep-space communications options, and doing a trade-off analysis between using conventional chemical versus ion-electric propulsion (which would also enable a more complex instrument package). For the initial design effort, all student and mentor time has been volunteered as part of the Space Grant training effort to provide hands-on experience for the next generation of space scientists and engineers. In the follow-up design and development phases of the NEAP mission, this organizational structure and process could be formalized in terms of student scholarships and mentor subcontracts. This would not only create substantial savings in mission costs, but would also create a world class vertically integrated (payload, spacecraft, launch vehicle, operations) workforce. Existing CalSpace capability includes the building of a set of micro-satellites known as DragSat, and the running of the KidSat (Shuttle-borne camera for K-12 education) mission control center on the campus of UCSD by California Space Grant students.

Data gathered at the asteroid will be an invaluable scientific and potential first commodity before mining operations can even be realized. Later resource markets include, but are not limited to, electrolyzed ice for rocket and satellite maintenance fuel, and the raw materials to enable both building in space and radiation shielding. Ultimately, successful rendezvous and mining of the NEOs will help establish a badly needed space infrastructure. Such innovative alliances between groups such as CalSpace and the SpaceDev Co. should help invigorate the private development of space in the near future, and enable cheaper access to the scientific treasures that may tell us more about the origins of our early solar system.

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Session 11
Engineering

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THE EFFECT OF AIR ADDITION TO THE INJECTED FUEL ON FLAME HOLDING BEHIND A STRUT BY INCIDENT SHOCK WAVES

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Abstract

Flame holding is an essential technology for Scramjet combustors especially under low Mach flight conditions, under which the airflow temperature is low. An interesting technique for flame holding is the utilization of shock waves. Experiments in a supersonic wind tunnel showed that the incident shock waves enabled stable flame holding in Mach 2.5 supersonic airflow with air temperatures between 360K and 900K. The objectives of this study were to investigate the flame holding characteristics as a function of the amount of air added to the hydrogen fuel jet and to understand the mechanism of the flame-holding stabilization by shock wave. The flame holding limits of hydrogen flow rate with various fuel/air ratios showed that both diffusion- and premixed-type flame-holding existed. To elucidate the mechanism of the flame-holding, visualization of OH species by laser-induced fluorescence and measurements of temperature in the wake were also conducted.

Introduction

The Scramjet Engine is one of the most expected thrust systems for Supersonic Transportation or New Space transport systems, shown in figure 1. It is required for Scramjet Engines to perform efficiently under wide flight Mach conditions, from 4 to 20 flight Mach No. It is one of the most important themes to realize stability of flame holding and high efficiency combustion, through all flight conditions. Especially under low Mach flight conditions, at the combuster, the air temperature is lower than the fuel hydrogen self ignition temperature, so that to stabilize the flame some consideration is needed. One way is to use a strut wake, which is a simple method, as it does not require additional special devices, so it can be utilized many ways. Between stabilization of flame holding in the strut wake and pressure loss, there is a trade-off. It is necessary to understand the mechanism of flame holding for finding the strut shape which facilitates high efficiency flame holding at minimum pressure loss.

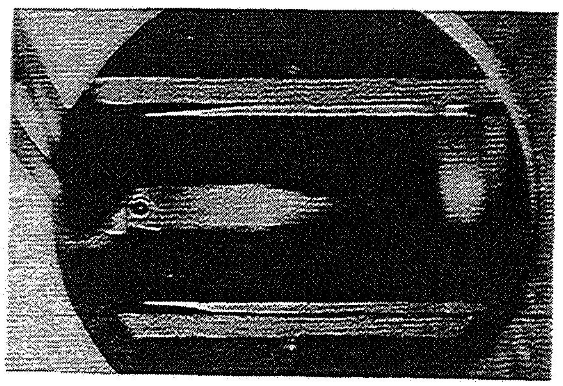
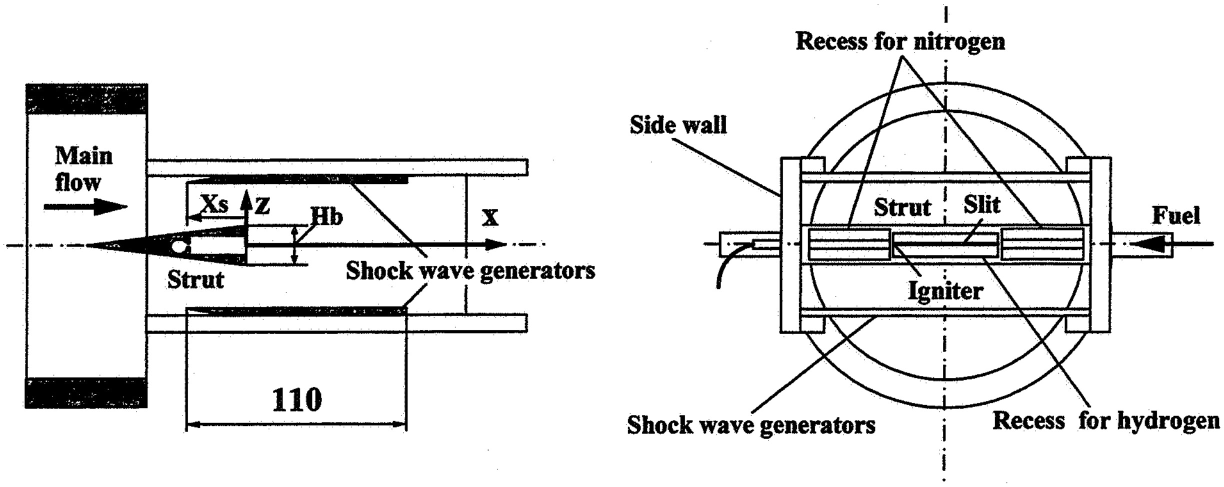


Figure 1. Side view of a strut wake with flame



(a) Cut view (b) Rear view

Figure 2. Test section

The stabilization of flame holding when fuel is injected into the strut wake is determined by not only air speed, temperature, pressure, and area of recirculation region, but also by wake construction which is local fuel distribution, flow field and so on. There were few studies about the mechanism of the stabilization of flame holding in the strut wake.

In a previous study by the authors, as shown in figure 1, the incident shock waves in the strut wake made possible the stable flame holding. In this study we investigate the flame holding characteristics depending on air addition to the hydrogen fuel jet and the mechanism of the flame-holding stabilization by strut wake.

First we illustrate flame-holding utilizing an induced shock wave. Secondly we elucidate the mechanism of the flame holding. Visualization of OH species by laser-induced fluorescence and measurements of temperature in the wake were also conducted. Then we discuss the mechanism of flame holding influenced by air addition to the fuel jet.

Experimental Apparatus and Conditions

Experimental Apparatus

In this paper, a blow-down tunnel (Institute of Fluid Science, Tohoku University) was used for experiments. It operates for 20 seconds at main air flow Mach number 2.5. Main air flow is heated by heater. The test section has 4 windows, which can be used for flow field visualization.

In figure 2, the test section is shown. As a flame holding device, a wedge section 2-dimensional strut (length 80 mm, height 20 mm) is used. The strut and 4 arms are fixed at the exit of the supersonic nozzle, and shock wave generators and side wall are set on the arms. Shock wave generators can slide along the main flow direction, so that the induced shock wave point can be changed without changing shock waves strength. In figure 3, Xs is the distance between leading edge of shock wave generators and the aft end of the strut. Hs is height of strut. The fuel injection slit (height 0.8 mm, width 50 mm) is located inside the recess which is located in the center of the strut behind the face. The recess has

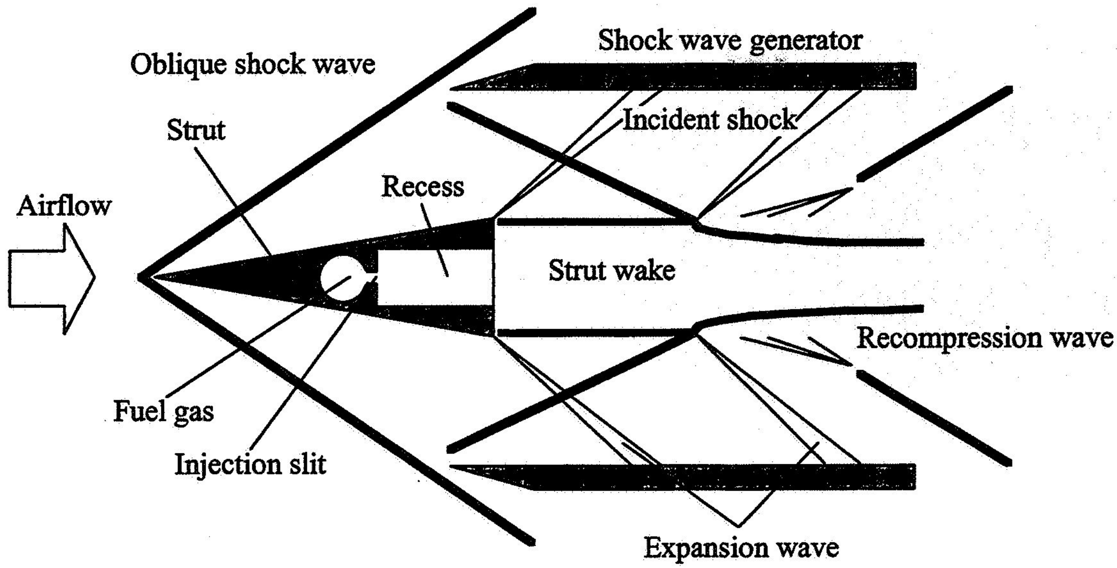


Figure 3. Flow field around strut wake and shock waves

a rectangular section (height 10 mm, width 50 mm, depth 29 mm), to improve flame stabilization. Fuel and hydrogen are supplied to the strut from the side wall through pipes, and injected from the slit into the strut wake. The speed of injection fuel at the exit of the slit is subsonic. Fuel is ignited under subsonic condition and then the main flow transitions to supersonic.

Fuel hydrogen flow rate is controlled by orifice and added air flow rate is controlled by mass flow controller. The flame holding limit is determined by pressure decrease and video monitoring. When the flame is blown off, the strut base pressure suddenly experiences a 5 kPa drop.

To measure the temperature inside the strut wake, a Pt 40%Rh-Pt 20%Rh thermocouple was used.

To investigate distribution of combustion gas in the strut wake, OH species were visualized by laser-induced fluorescence (LIF). A YAG/OPO raiser system was used, and laser wave length was 283 nm.

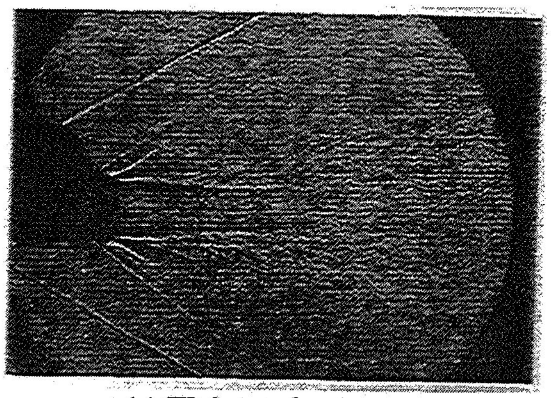
Experimental conditions

We experimented under these conditions as follows. Total pressure of main flow was maintained at 0.5M Pa constantly. Main air total temperature was 360K through 900K. At the entrance of test section, Mach No. was 2.5, static pressure was 30 KPa, and static temperature of main flow was between 160K to 400K.

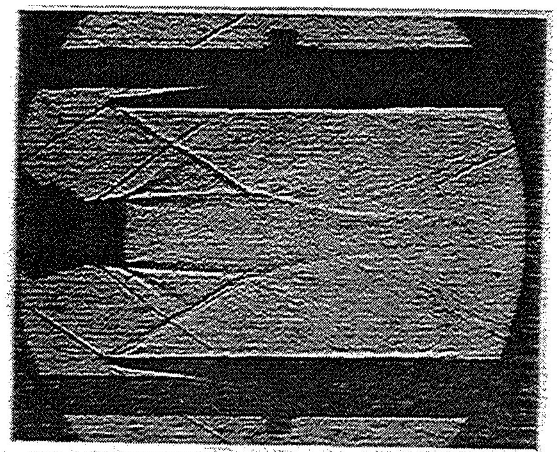
Utilization of shock wave for flame holding

In figure 4, the flow field around the strut wake is illustrated.

At the entrance to the test section, the main flow flows horizontally. Main air flow is tended to the direction along the strut surface by the first shock wave which occurred at the leading edge of strut, and then tended to the horizontal direction by the second shock waves which occurred at the leading edge of the shock wave generators. This second shock wave, we named the induced shock wave. Downstream of the induced shock wave, main flow Mach no. is 1.93, and static pressure ratio between downstream and upper stream is 1.5. The



(a) Without shock waves



(b) With shock waves

Figure 4. Visualization of strut wake by Shadow-graph

second shock wave interfered with the strut wake and the strut wake was spread, as shown in figure 4, which is a shadow graph.

Flame holding characteristics

We measured the flame holding region wth and without additional air. In figure 5, flame holding limits are plotted vs. main air flow total temperature when added air flow rate is constant. In this result, when added air flow rate is constant, and when main air flow total temperature T_{in} increases, Q_{H2} also increases. It is expected that near Upper Blow off limits, the fuel ratio becomes too rich in the strut wake, and air addition makes the fuel ratio proper, then Q_{H2} rises higher. But when Q_{air} = 180 Nl/min, Q_{H2} is decreased more than when no air is added. Then Q_{air} is much increased, Q_{H2} increases, when Q_{air} is more than 750 Nl/min, Q_{H2} increases above no air addition flame holding limits.

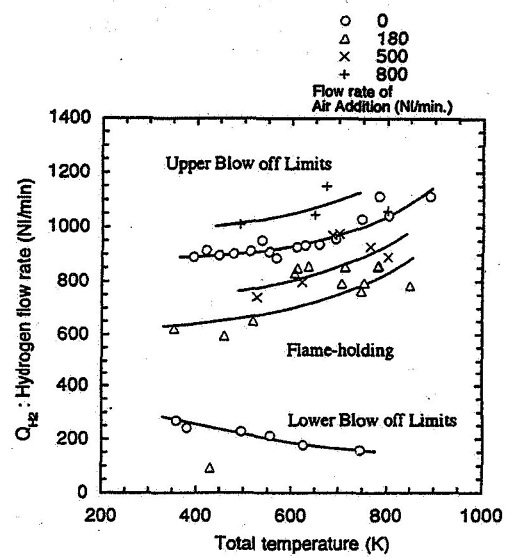
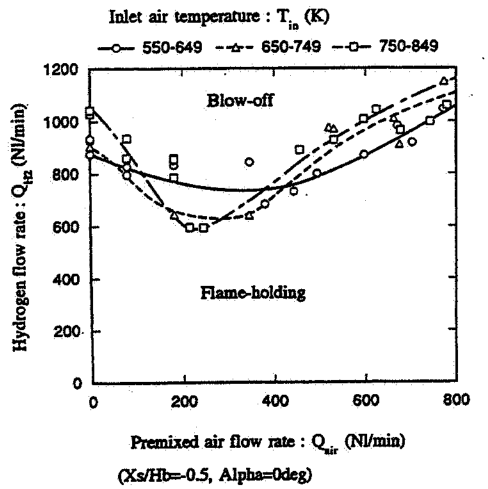


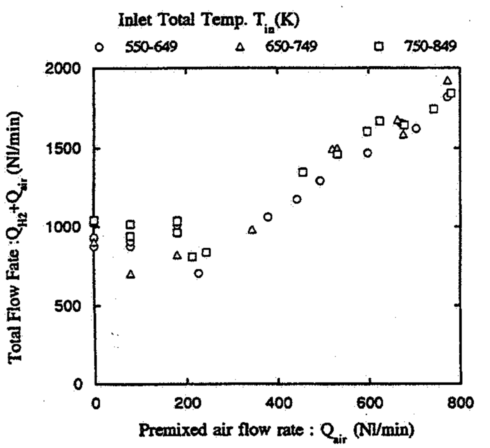
Figure 5. The flame holding region (Additional air flow rate was constant)

In figure 6 (a) Q_{H2} at upper blow off limits were plotted vs. Q_{air} as x axis. When Q_{air} is around 200~300 Nl/min, Q_{H2} has the minimum value, then Q_{air} becomes more over, and Q_{H2} increases.

In figure 7, the equivalence ratio of premixed fuel at upper blow off limits is shown. Under normal temperature, the hydrogen combustible limit equivalence ratio is around 7, now the minimum value is equal to this equivalence ratio. When Q_{air} is above the minimum point, the injected premixed fuel is combustible itself.



(a) Hydrogen flow rate at blow off limits



(b) Total flow rate at blow off limits

Figure 6. The effect of air addition to fuel hydrogen on upper blow off limits

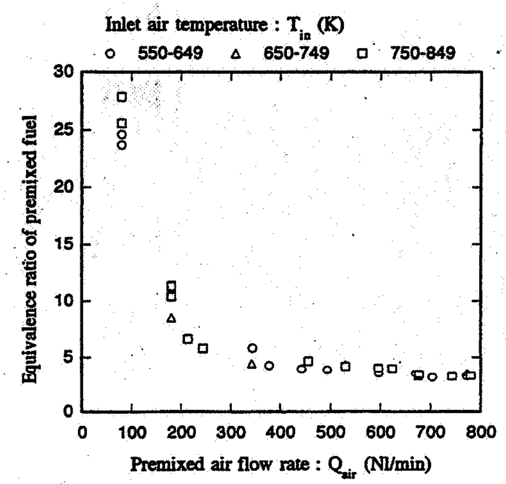


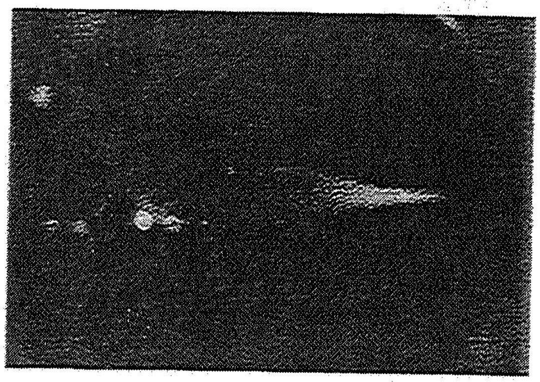
Figure 7. Equivalence ratio of premixed fuel at blow off limits

Visualization of flame holding region

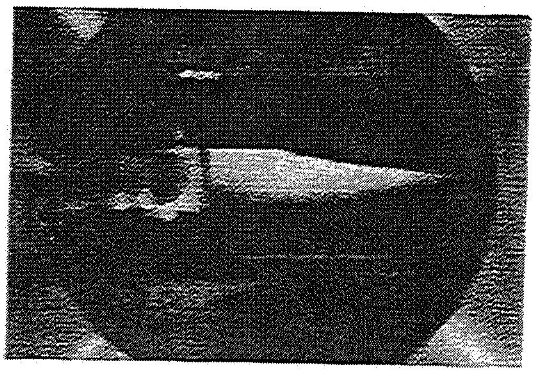
Side view of the flame

Figure 8 shows a side view of the flame, with different amounts of added air. Case (a) is near the upper blow off limit, and the flame exists downstream from the strut wake.

In Case (b), with a large amount of air, Q_{air} = 700 Nl/min, the flame attaches in the recess wall, and cannot exist symmetrically. In this case, the flame holding mechanism is different from case (a), because injected premixed fuel is combustible, so it forms a stability point on the strut wall which has no relation with mixing with main air flow.



(a) Near upper blow off limits



(b) Large amount of additional air Q_{air} = 700 Nl/min

Figure 8. Side view of strut wake with flame when air added

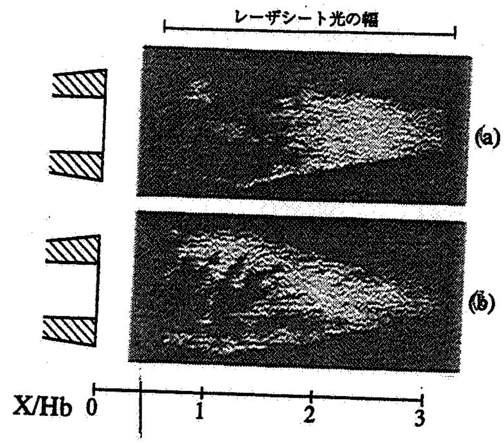


Figure 9. OH species on the strut wake (Q_{H2} = 480 Nl/min, Q_{air} = 0 Nl/min) (a),(b) Mach = 2.5

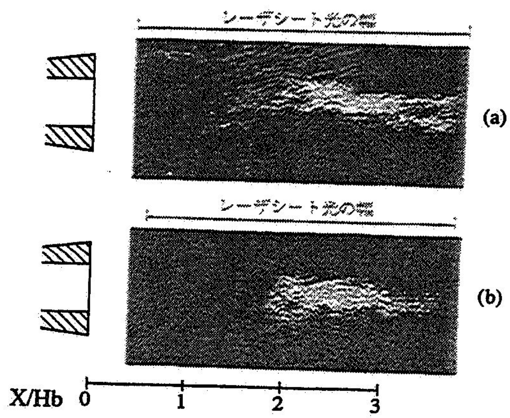


Figure 10. OH species in the strut wake
(a) Q_{H2} = 800 Nl/min, Q_{air} = 0 Nl/min
(b) Q_{H2} = 400 Nl/min, Q_{air} = 180 Nl/min

Visualization of OH species by LIF (laser induced fluorescence)

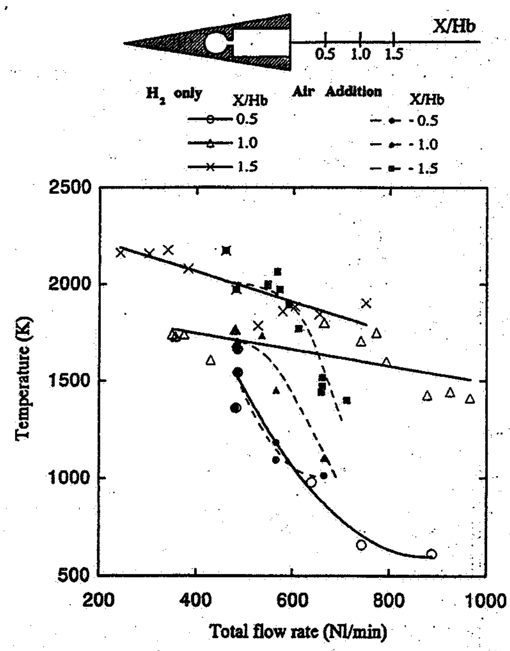
In figure 9 and 10, are shown the instantaneous sectional views at center line of OH species in the strut wake. In these figures, the white part indicates OH species fluorescence, and the high distribution region of OH species is deep white.

In figure 9, both (a) and (b), QH2 = 480 Nl/min constantly and Q_{air} = 0 Nl/min. At the widest area of the strut wake (around X/Hb = 1.2) , the distributions of OH species change drastically. In case (a) combustion gas was injected into the center of the wake from downstream. In case (b) , the boundary of combustion gas and fuel gas is very complex. On the other hand, downstream more than X/Hb = 1.7, there is a constant distribution of OH species.

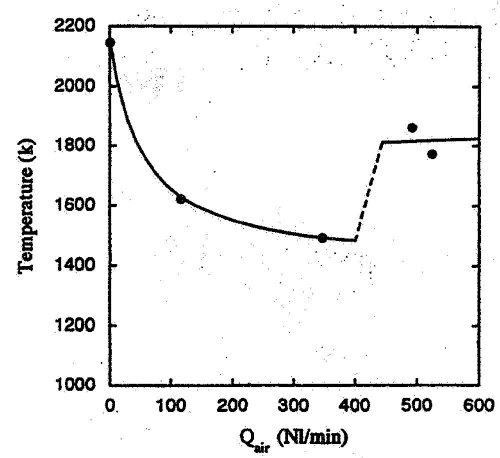
In figure 10, case (a) is QH2 = 800 Nl/min, Qair = 0 Nl/min, and case (b) is QH2 = 400 Nl/min, Qair = 180 Nl/min. For both cases, flame holding conditions were near upper blow off limits. For both case (a) and (b), OH species do not exist in the region near the strut wake. The total flow rate of jet fuel increases, the stagnation point moves downstream, and the recirculation flow shrinks.

Temperature measurement

In figure 11 (a), the horizontal axis is total flow rate of jet fuel, solid lines mean only hydrogen fuel injected, and dotted lines (broken lines) mean air added under Q_{H2} = 480 Nl/min constant. When total flow rate increases, the temperature becomes low in all cases. Especially in the case of X/Hb = 0.5, temperature rapidly goes down. Because recirculation flow shrinks downstream, the region is occupied by cold fuel gas.



(a) Small amount of additional air



(Xs/Hb=-0.5, X/Hb=1.5, Q_{H2}=330Nl/min)

(b) Large amount of additional air

Figure 11. Temperature distribution in the strut wake

In figure 11 (b), the x axis is added air flow rate, and it shows wake temperature influenced by added air flow rate (Q_{H2} = 330 Nl/min constant, X/Hb = 1.5). At first, when a small amount of air is added, the temperature goes down, but when the added air flow rate is over than 550 Nl/min, the temperature rises. This means that the mechanism of flame holding in the strut wake changes from diffusion-type to premixed-type.

Discussion

In figure 12, the difference of flame holding mechanism depending on the amount of added air is shown schematically. Case (a) is no air addition; under stabilized flame holding, reaction occurred near the strut wake, and large recirculation flow exists. Case (b) is a small amount of air added, and because of increase of the momentum of injected premixed fuel, the recirculation flow shrank downstream. The reaction region is small and flame holding conditions approach blow off limits. In Case (c), a large amount of air is added, the flame attaches to the recess wall, and the flame tends to one side in the strut wake.

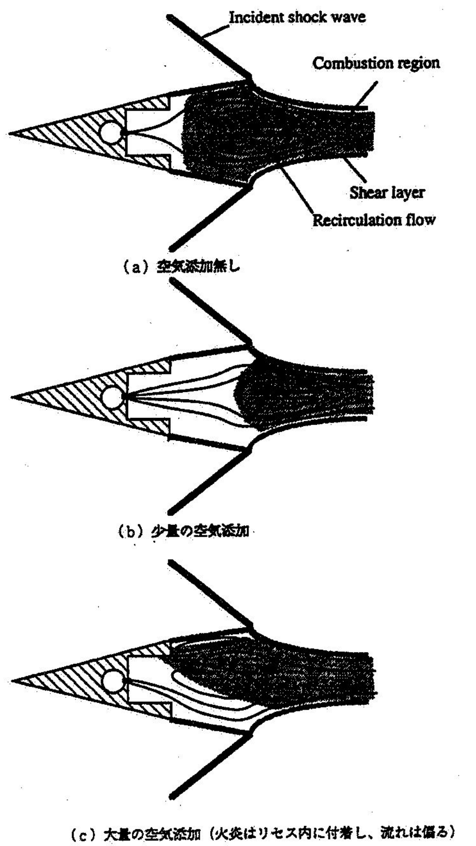


Figure 12. Schematic of the mechanism of flame holding in the strut
(a) No additional air
(b) Small amount of additional air
(c) Large amount of additional air

Conclusion

In this study we investigated the mechanism of flame holding stabilized by shock wave, using additional air added to hydrogen fuel.

(1) To add air to hydrogen fuel, two flame holding types exist, diffusion- and premixed-type flame-holding. In the case of a small amount of air added, the flame-holding type becomes diffusion-type. Increasing the amount of added air, the upper blow off limits become lower. On the other side, the amount of added air increases, and the flame holding type becomes premixed-type. Increasing the amount of added air, increases the upper blow off limits. Upper blow off limits become higher, when air addition makes the reaction term shorter than the registration term.

(2) In the diffusion-type flame holding, increasing the total flow rate through the slit, the recirculation flow is pushed downstream and shrunk by the injected fuel jet, and becomes smaller. In the premixture-type flame holding, the flame attaches on the recess wall, and in the strut wake, the flame is tended toward the upper or lower wall.

Acknowledgment

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SPASIM: A SPACECRAFT SIMULATOR

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ABSTRACT

The SPAcecraft SIMulator (SPASIM) simulates the functions and resources of a spacecraft to quickly perform conceptual design (Phase A) trade-off and sensitivity analyses and uncover any operational bottlenecks during any part of the mission. Failure modes and operational contingencies can be evaluated allowing operational planning (what-if scenarios) and optimization for a range of mission scenarios. The payloads and subsystems are simulated, using a hierarchy of graphical models, in terms of how their functions affect resources such as propellant, power, and data. Any of the inputs and outputs of the payloads and subsystems can be plotted during the simulation or stored in a file so they can be used by other programs. Most trade-off analyses, including those that compare current versus advanced technology, can be performed by changing values in the parameter menus. However, when a component is replaced by one with a different functional architecture, its graphical model can also be modified or replaced by drawing from a component library. SPASIM has been validated using several spacecraft designs that were at least at the Critical Design Review level. The user and programmer guide, including figures, is available on line as a hypertext document. This is an easy-to-use and expandable tool which is based on MATLAB® and SIMULINK®. It runs on Silicon Graphics Inc. workstations and personal computers with Windows 95™ or NT™.

INTRODUCTION

The primary function of the SPAcecraft SIMulation (SPASIM) software is to create a virtual environment to simulate a spacecraft. The simulation includes the spacecraft's operation and the interaction of multiple subsystems as a function of time and resources. SPASIM presents this virtual environment to the user in a graphical/object-oriented interface to enhance usability and integration.

SPASIM defines a hierarchy of block diagrams wired together along with parameters that describe operational and performance characteristics that yields a well documented functional spacecraft model. The top-level block diagram is shown in Figure 1. Each block within a graphical user interface (GUI) window defines a function or a hierarchy of lower level blocks. Blocks at the lowest level invoke MATLAB® or SIMULINK® code. The GUI presents a dialog box to the user that allows changes to be made to a block's parameters before simulation starts. Lines connecting the blocks transmit values such as those used to represent orbital information and spacecraft resources. Examples of these resources are propellant, power, and data.

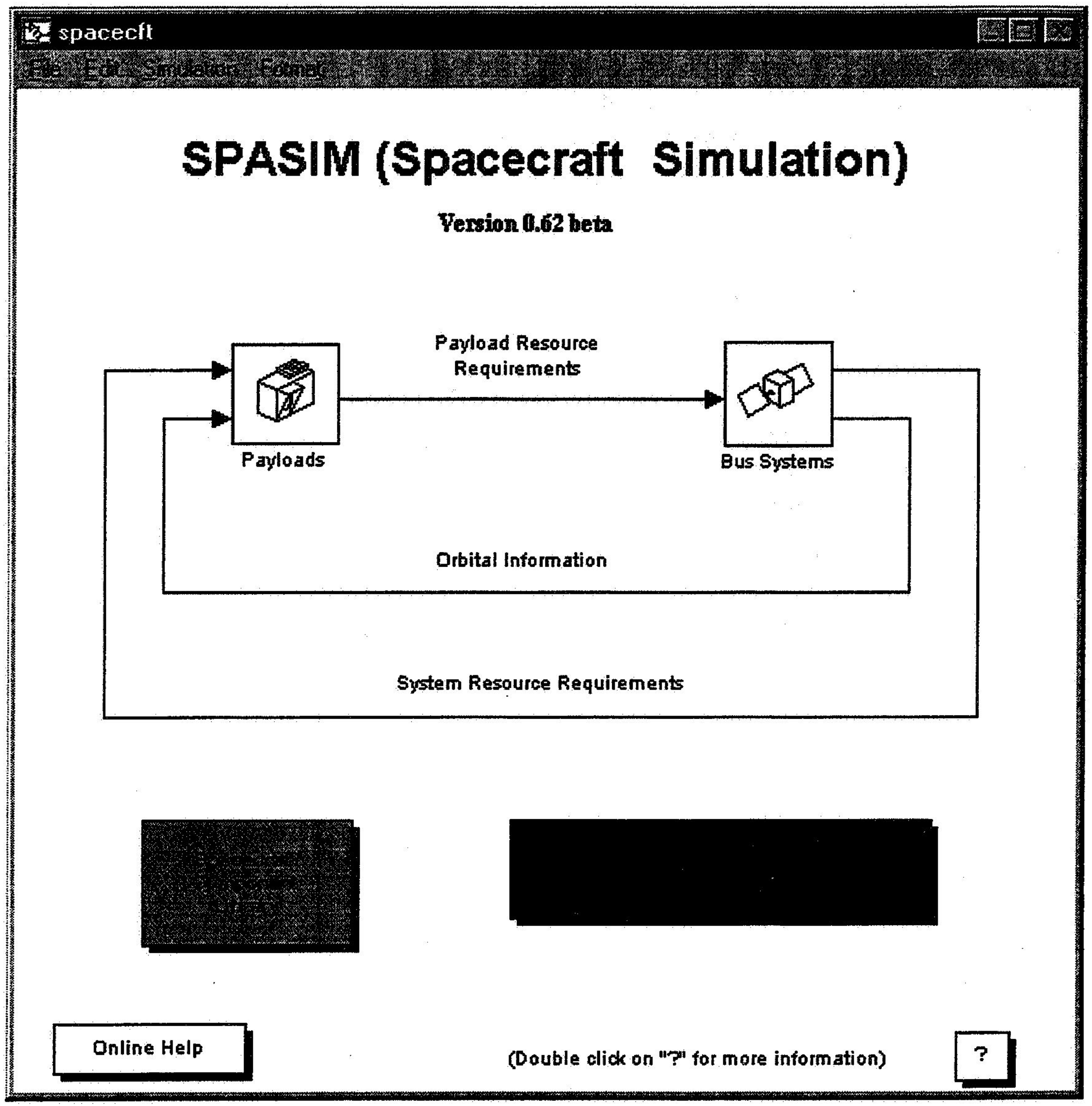

Figure 1: Main Window

The user can analyze subsystem interactions during a simulation by displaying a dynamic plot of any block's input or output values. A large selection of 26 predefined plots may be chosen by clicking on the "Spacecraft Parameters" button and choosing the plot menu from the pop-up window.

SPASIM includes a library of models of spacecraft payloads and subsystem components that represent a range of functionality. The user and programmer guide (Liceaga*et al.* 1997), including figures, may be accessed by clicking on the "Online Help" button. It is also available on the World Wide Web at http://freedom.larc.nasa.gov/projects/spasim/help.html. In addition, each major window has help specific to that window which can be accessed by clicking on the "?" button.

SPASIM is one of the tools in the Satellite System Design and Simulation Environment (Ferebee, Troutman, and Monell 1997). This environment also includes a design and sizing tool and a component database.

SPASIM has been initially implemented for uncrewed, earth-orbiting spacecraft. It can be ported to any platform on which MATLAB® and SIMULINK® can run. It is now running on Silicon Graphics Inc. workstations and personal computers with Windows 95™ or NT™.

SPACECRAFT PAYLOADS

The payloads are the instruments the spacecraft carries to accomplish its mission. By default, there are four payloads, which are shown in Figure 2.

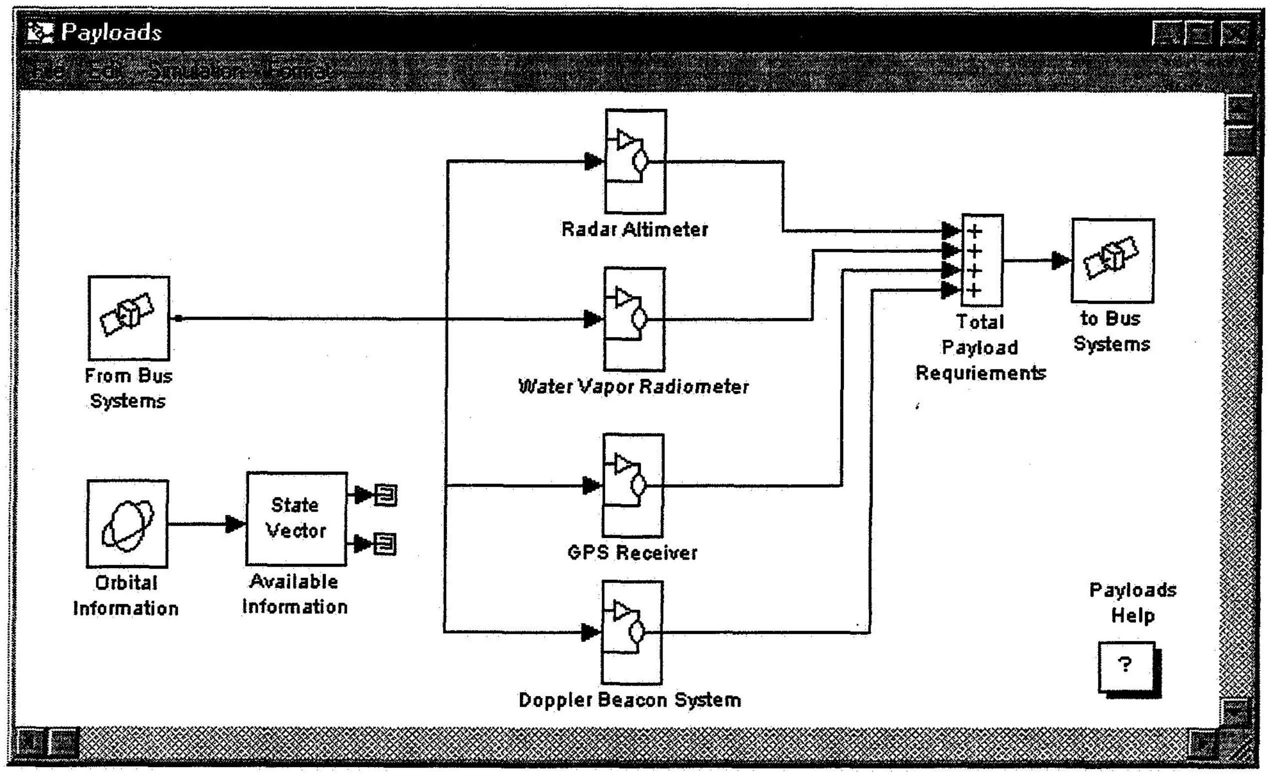

Figure 2: Spacecraft Payloads

Resource requirements and state information from the subsystems flow in from the left. They are distributed to the payloads which output their requirements. Finally, the requirements of each payload are added and feed back to the subsystems as the payload resource requirements.

All payloads share a standard interface. They have access to the same state and orbital information from the subsystems. They can also add to any of the spacecraft resource requirements.

SPACECRAFT SUBSYSTEMS

The spacecraft subsystems provide the resources required for the mission to be accomplished. Together they make up what is commonly called the spacecraft bus. As shown in Figure 3, the spacecraft bus is modeled as being composed of the following subsystems: power; thermal; propulsion; guidance, navigation, and control (GNC); communication and tracking (CT); and command and data handling (CDH).

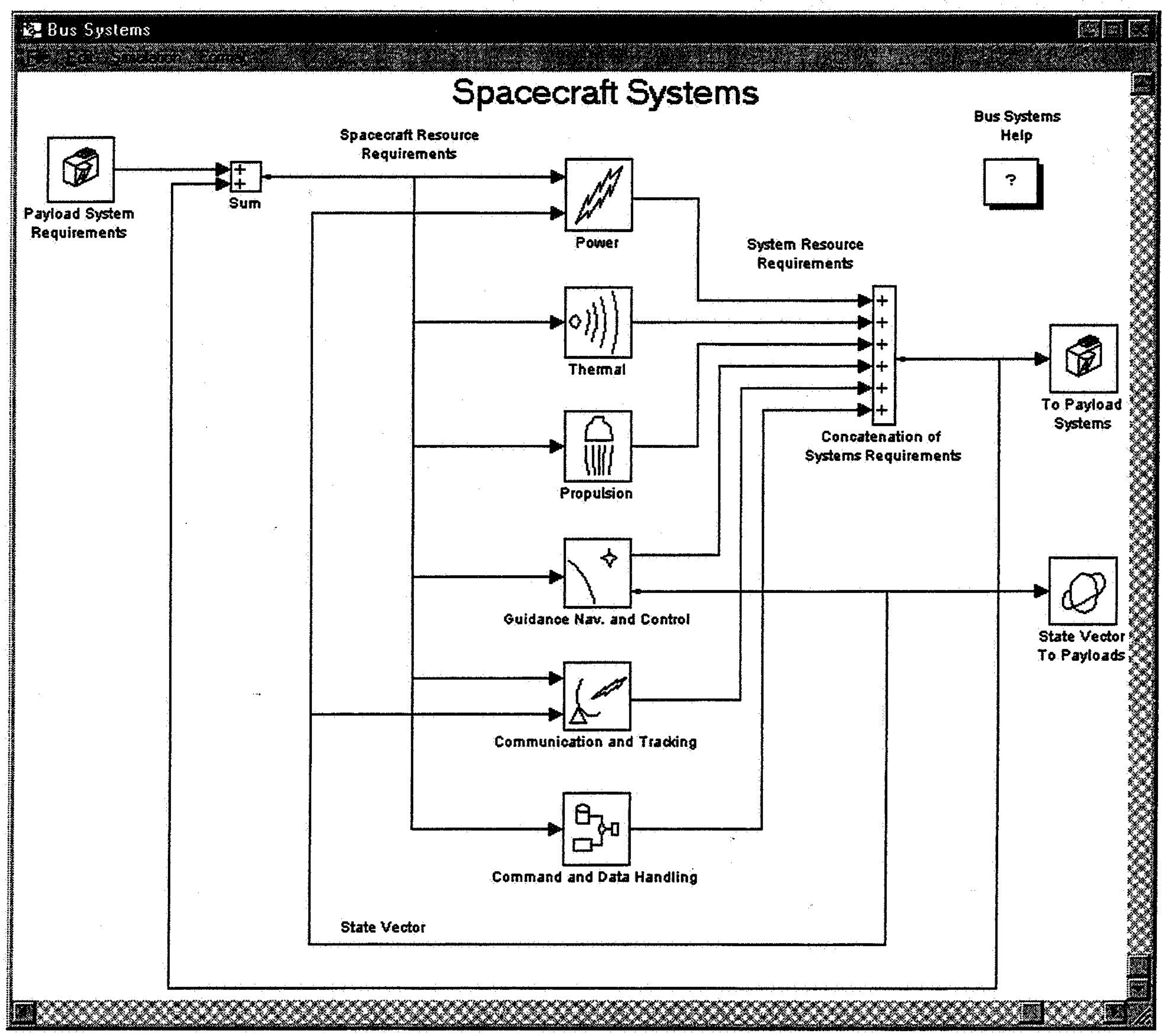

Figure 3: Spacecraft Subsystems

Payload resource requirements flow in from the left. The subsystem requirements are added and feed back to the subsystems as the spacecraft resource requirements.

All subsystems share the same standard interface that the payloads have. They have access to the same spacecraft resource requirements and orbital information. They can also add to any of the spacecraft resource requirements.

This standard interface provides modularity and facilitates independent maintenance of the models by the instrument and subsystem experts. It also facilitates the incorporation of externally developed models.

Each subsystem has a parameter menu. These parameters are used as constants or the initial value of variables.

Power

The purpose of the power subsystem, shown in Figure 4, is to generate, store, and distribute electrical energy. It is implemented through a solar array (produces power), a battery (stores power), and a charge unit (controls power).

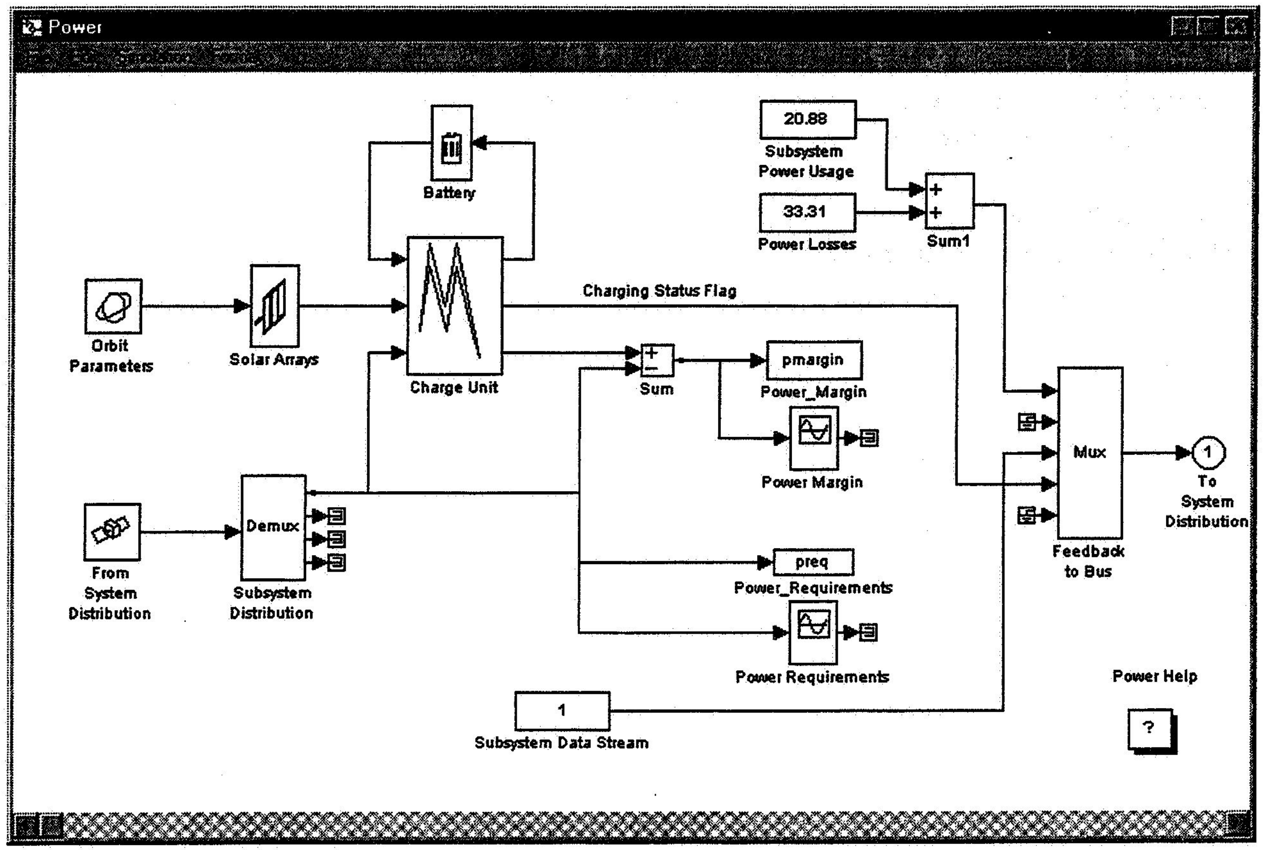

Figure 4: Power Subsystem

The parameters in the main power menu, shown in Figure 5, include the: average minimum solar flux, solar cell type and efficiency, solar array active area, solar cell degradation factor, initial solar array efficiency, solar array shadow file name, solar cell in-service time, power bus efficiency and nominal voltage, and number of degrees of freedom (DOF) of the solar array. The solar array shadow file is an ASCII file, loaded before the start of the simulation, and used by a two-dimensional look-up function in SIMULINK®. This look-up function linearly interpolates the fraction (0.0 to 1.0) of the solar array area available at specific alpha and beta angles for the spacecraft in a nominal local vertical local horizontal (LVLH) flight attitude.

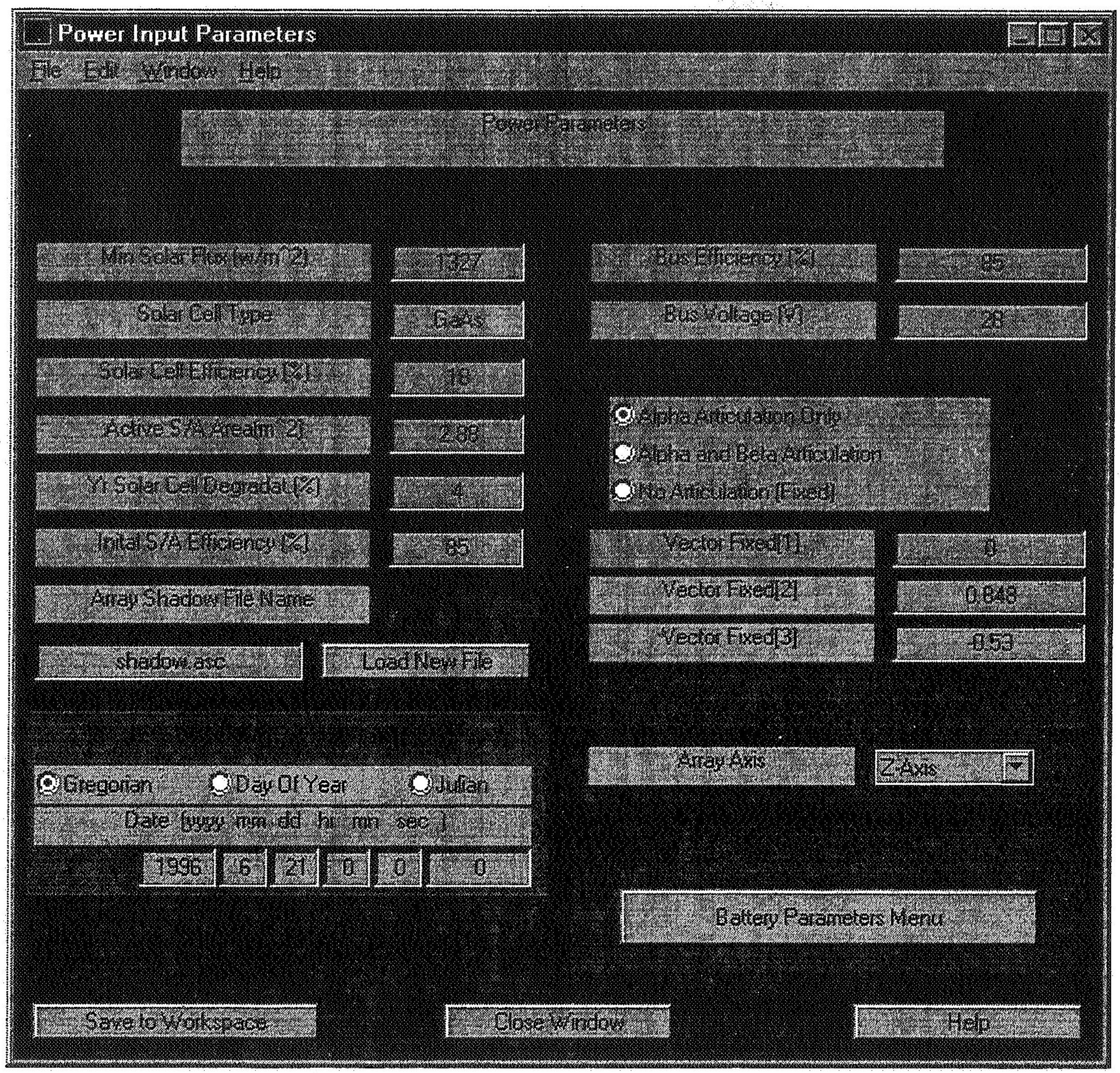

Figure 5: Power Parameters

The parameters in the battery menu, shown in Figure 6, are: maximum end-of-life energy capacity, cell type, charging efficiency, initial charge, minimum depth of discharge (DOD), and maximum charge and discharge currents. The minimum DOD indicates how much the battery needs to be discharged, after obtaining a full charge, before it will be charged again.

The inputs to this model are the: spacecraft power requirement, sun vector, earth vector, and sunlight flag. Its outputs are a battery charging flag, the power it requires, and the rate at which it generates data. This model has predefined plots of time versus: spacecraft power requirements, power margin, depth of discharge, solar alpha, solar beta, shadow table result, initial power factor, and power factor.

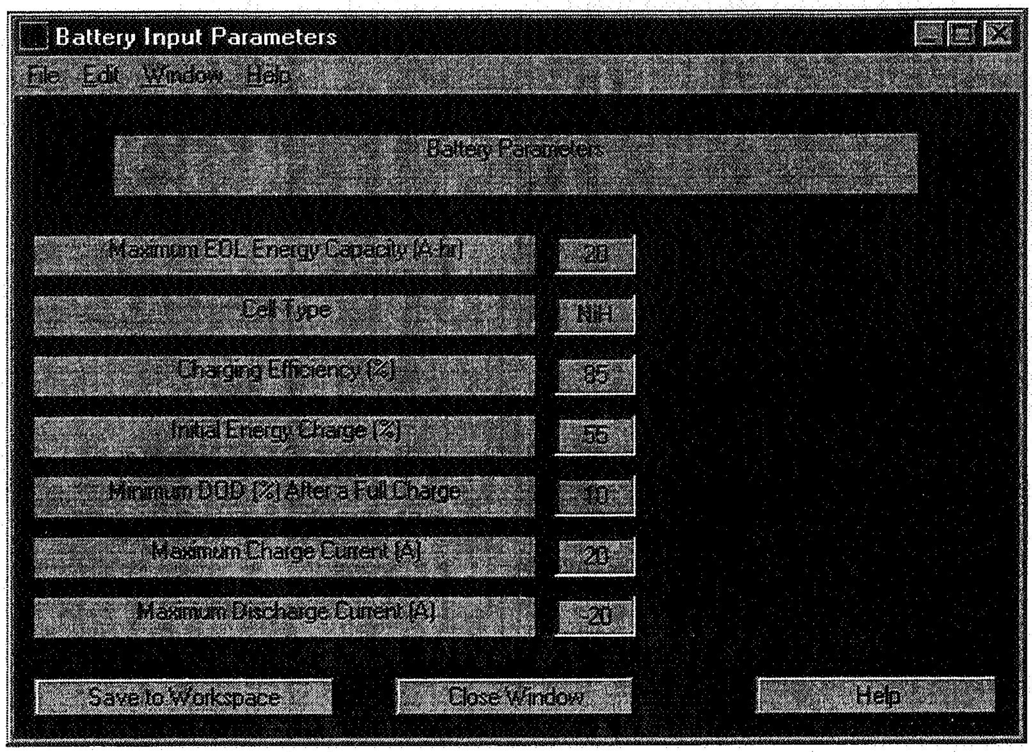

Figure 6: Battery Parameters

Thermal

The purpose of the thermal subsystem, shown in Figure 7, is to maintain spacecraft components within specified temperature limits. The model implemented in SPASIM assumes a cold biased system in which given regions are designed to operate below the upper temperature limits of the components therein. Since this often results in the temperatures, in that region, falling below the lower limits of the components, thermostatically controlled heaters are often employed. Components that can't be effectively cold biased are cooled by passive (stored cryogen and radiative) and/or active (closed cycle and thermoelectric) coolers.

This model also allows the user to simulate custom heater/cooler power profiles. The heaters/coolers are either keyed to a day/night switch or are available on an on-demand basis. Once on, the heaters/coolers will follow a user defined power profile. Note that because of their more restrictive temperature requirements, batteries are assumed to be on a different cold biased loop where heaters are keyed to the charge state of the batteries.

The parameters in the thermal menu, shown in Figure 8, specify the power requirements for: a cryocooler; the battery heater used when neither charging nor discharging; and the payload, propellant, and GNC electronics heaters used during eclipse. The inputs to this model are the battery charging flag from the power subsystem and the sunlight flag from the GNC subsystem. Its outputs are the power it requires and the rate at which it generates data. This model has predefined a plot of time versus thermal power requirement.

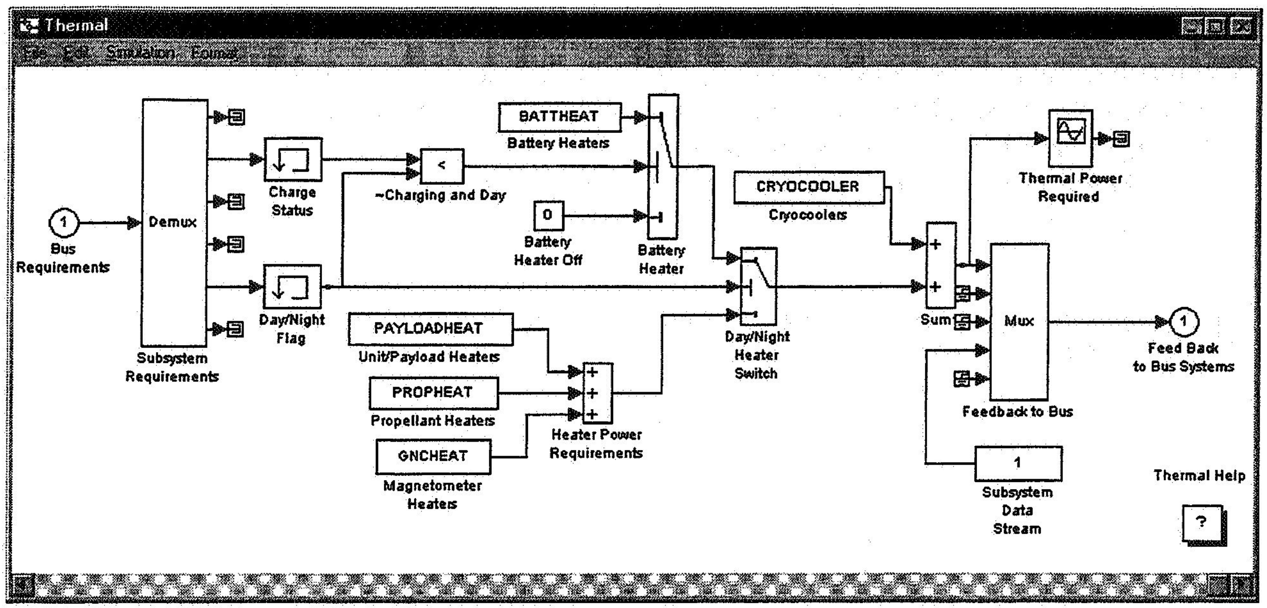

Figure 7: Thermal Subsystem

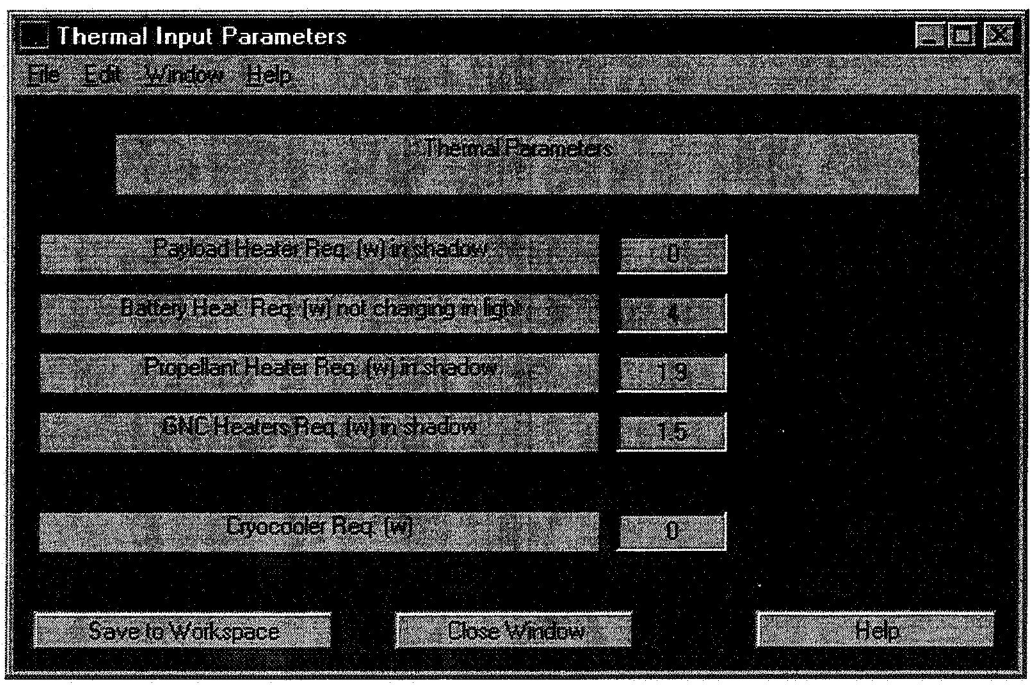

Figure 8: Thermal Parameters

Propulsion

The purpose of the propulsion subsystem, shown in Figure 9, is to provide the thrust required to maintain or change the spacecraft's orbit and attitude. This subsystem uses two resources, propellant and power. The model implemented is strictly an event driven process. Until the GNC subsystem is modeled as a mass-accurate system, this model will stay as a stochastic model. The subsystem is implemented through twelve thrusters. Four thrusters are required to provide the positive and negative torques about each of the three axes.

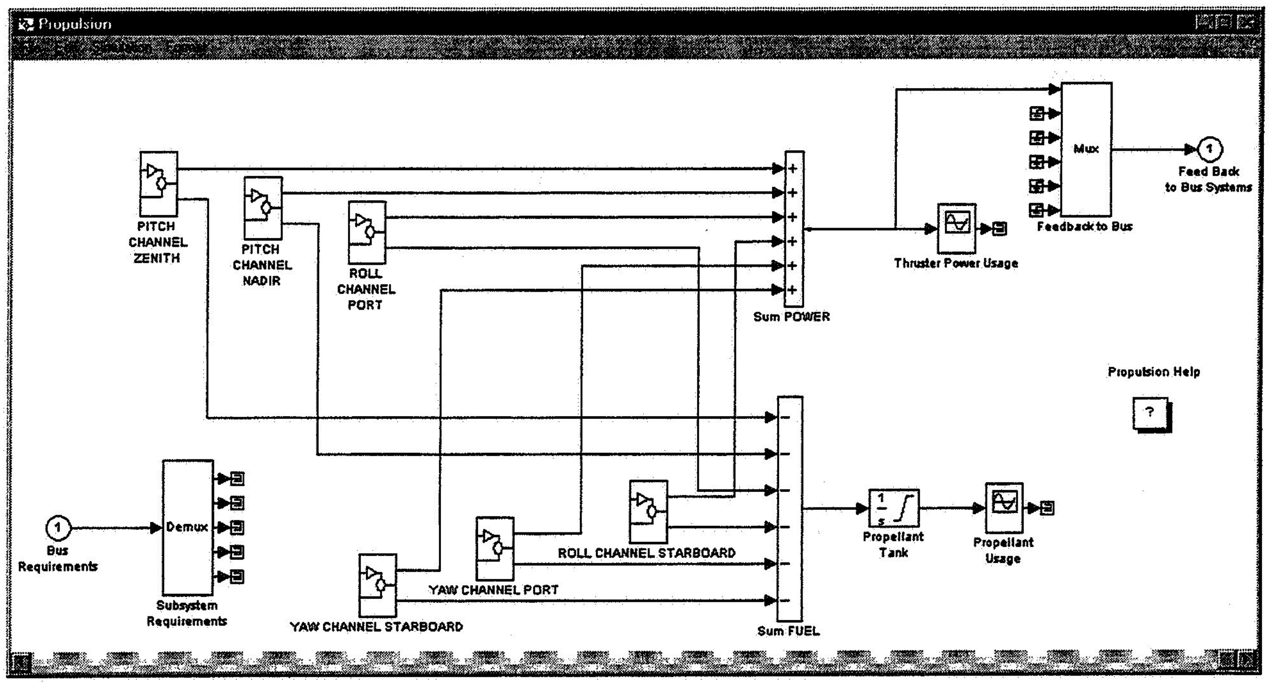

Figure 9: Propulsion Subsystem

The parameters in the propulsion menu, shown in Figure 10, specify the: initial fuel load contained in the tanks for the mission, specific impulse of the propulsion subsystem, minimum time a thruster remains on after an on/off command is issued, power per thruster, amount of fuel burned for each second of engine firing time, number of operational propulsion events to occur in a year, and time the first event will occur from the start of the simulation. The output of this model is the power it requires. This model has predefined plots of time versus thruster power requirement and propellant tank level.

The following three assumptions were used in creating the stochastic resource model for the propulsion subsystem. First, thruster events are modeled as periodic throughout one year. Second, thruster events will have a duration of the thruster's minimum impulse time. Third, a thruster firing sequence will have a duration of two hours.

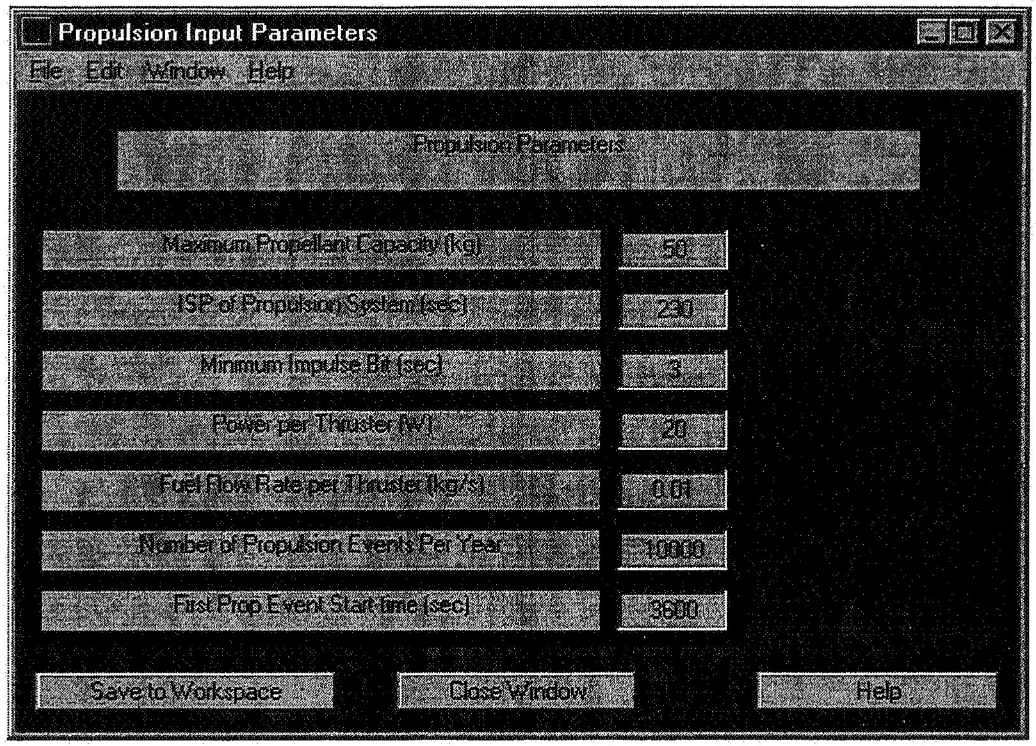

Figure 10: Propulsion Parameters

Guidance, Navigation, and Control

The GNC subsystem is shown in Figure 11. The purpose of a typical GNC subsystem is to provide orbital and attitude determination and control. However, passive or active attitude control isn't simulated. Instead, attitude motion can be prescribed with one of the following four methods: fixed, oscillatory, maneuver, or user prescribed. In fixed, the user specifies an initial attitude and the spacecraft is held fixed at that attitude. In oscillatory, the user specifies amplitudes and frequencies for the three axes. In maneuver, the user specifies up to 10 attitudes and the time in the simulation when they will be reached. In user prescribed, an input file with a proper attitude history is given. This can be the result of an off-line three DOF or a six DOF simulation. It is assumed that the control system can meet the user prescribed attitude profile. There are two attitude modes available to the user. An Earth oriented LVLH mode and an inertial mode. When in inertial mode, the user can specify a spin rate.

The parameters in the main GNC menu, shown in Figure 12, specify the: attitude flight mode, either LVLH or inertial; initial spacecraft attitude; spacecraft spin rate and axis; attitude history type, either fixed, oscillatory, maneuver, or user prescribed; repeat attitude history flag, if maneuver or user prescribed no value matching at beginning or end; oscillatory amplitudes and frequencies for the three axes; and user prescribed attitude history file name. The parameters in the orbit menu, shown in Figure 13, specify the: launch time, simulation start time, spacecraft lifetime, and epoch time. For this epoch time, they also specify the: apogee, perigee, inclination, argument of periapsis, and ascending node.

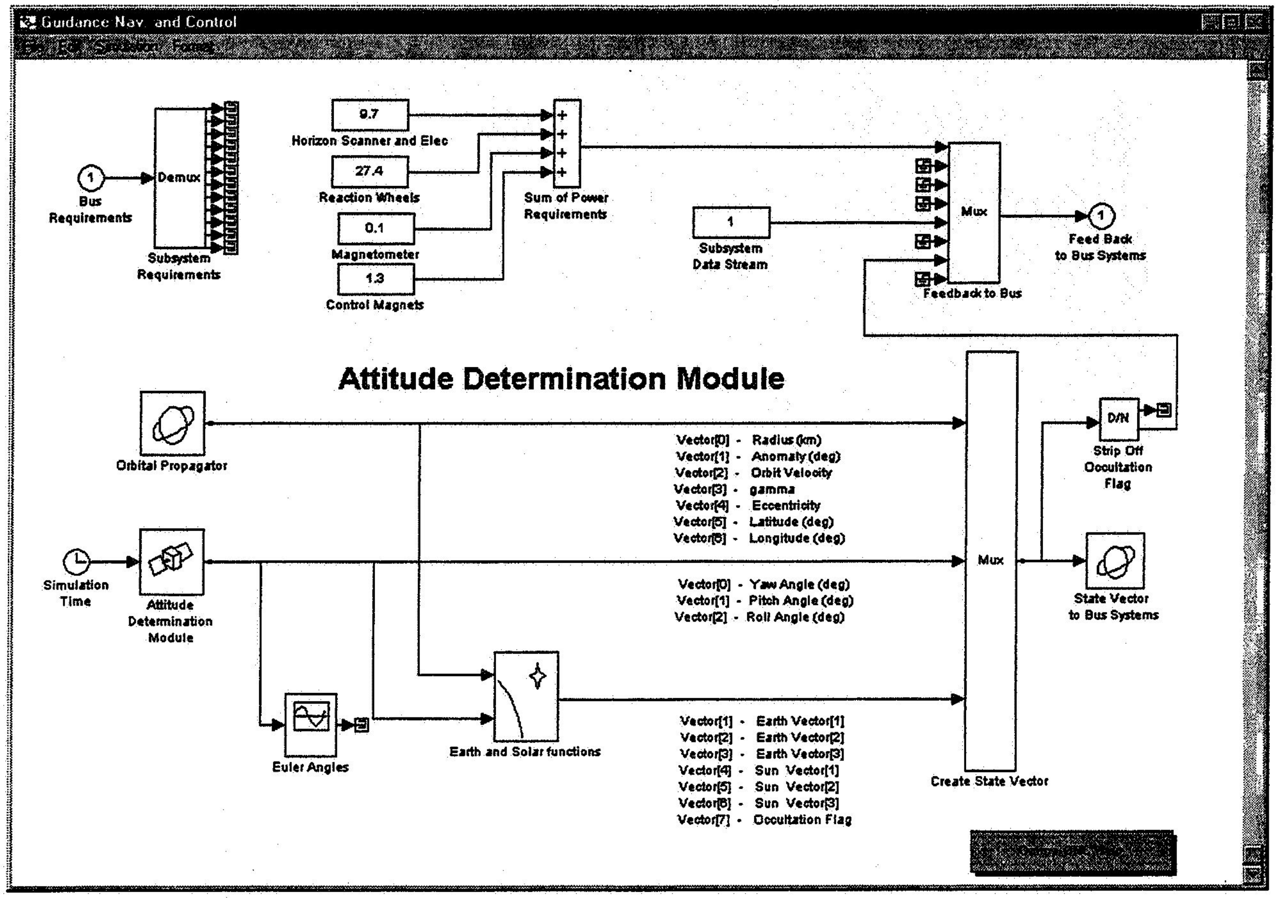

Figure 11: Guidance, Navigation, and Control Subsystem

The orbital information this model outputs is calculated by three submodels: orbital propagator, attitude determinator, and earth and solar pointer. The orbit propagator integrates a simple two body orbit equation using a Runge-Kutta type integrator based on Fehlberg's seventh-order formula to calculate: radius, true anomaly, velocity, flight path angle, eccentricity, longitude, and latitude. The attitude determinator calculates the spacecraft's attitude in terms of yaw, pitch, and roll angles. The earth and solar pointer outputs the: sun vector, earth vector, and sunlight flag.

The model also outputs the power it requires and the rate at which it generates data. This model has defined plots of time versus: yaw, pitch, roll, earth vector, sun vector, and sunlight. It has also defined a plot of latitude versus longitude.

Communications and Tracking

The purpose of the CT subsystem, shown in Figure 14, is to communicate with ground stations to download data and upload commands. The carrier-to-noise ratio of both the telemetry downlink and the command uplink is calculated as a figure of merit for the channel carrying capability of the link.

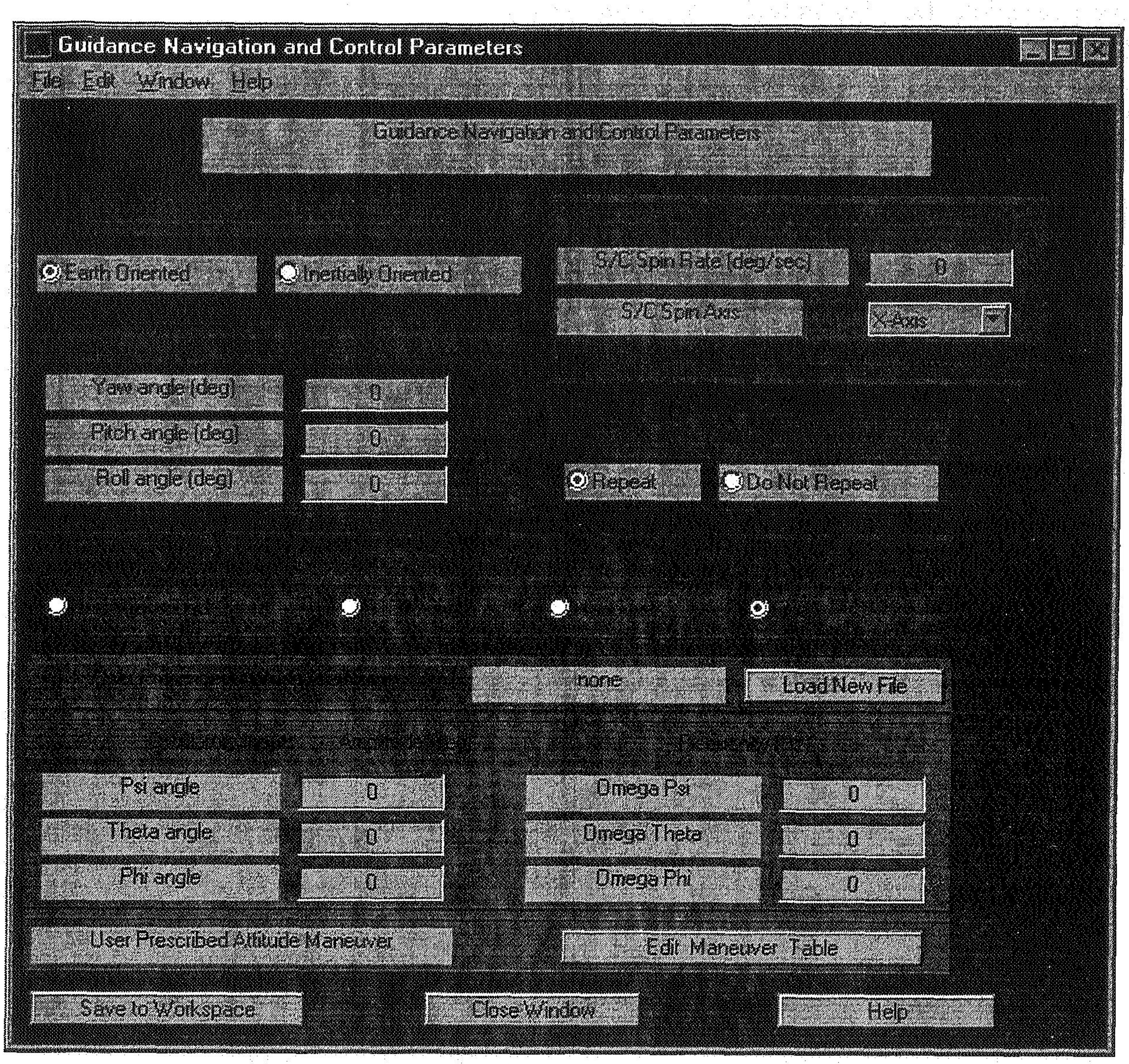

Figure 12: Guidance, Navigation, and Control Parameters

The power flux density (PFD) is calculated at both the ground station site and at the spacecraft during a contact coverage period. This feature helps determine if the link has enough power at the receiver to accept a data transfer link. Because of the change in slant range due to continuously changing position of the spacecraft with respect to the ground station, calculating the PFD gives an indication of the received power range while the ground contact is made. It can help in the design and analysis process for either a ground station or a spacecraft communications subsystem.

The radio frequency (RF) link margin between the space and ground communication segments is calculated to give an indication of RF performance levels available to maintain adequate communication. Based upon requirements such as bit-error-rate, the link margin gives the theoretical potential of the link to perform to certain specifications. If the calculated performance exceeds requirements, savings can be realized by relaxing the ground or spacecraft communications subsystem design.

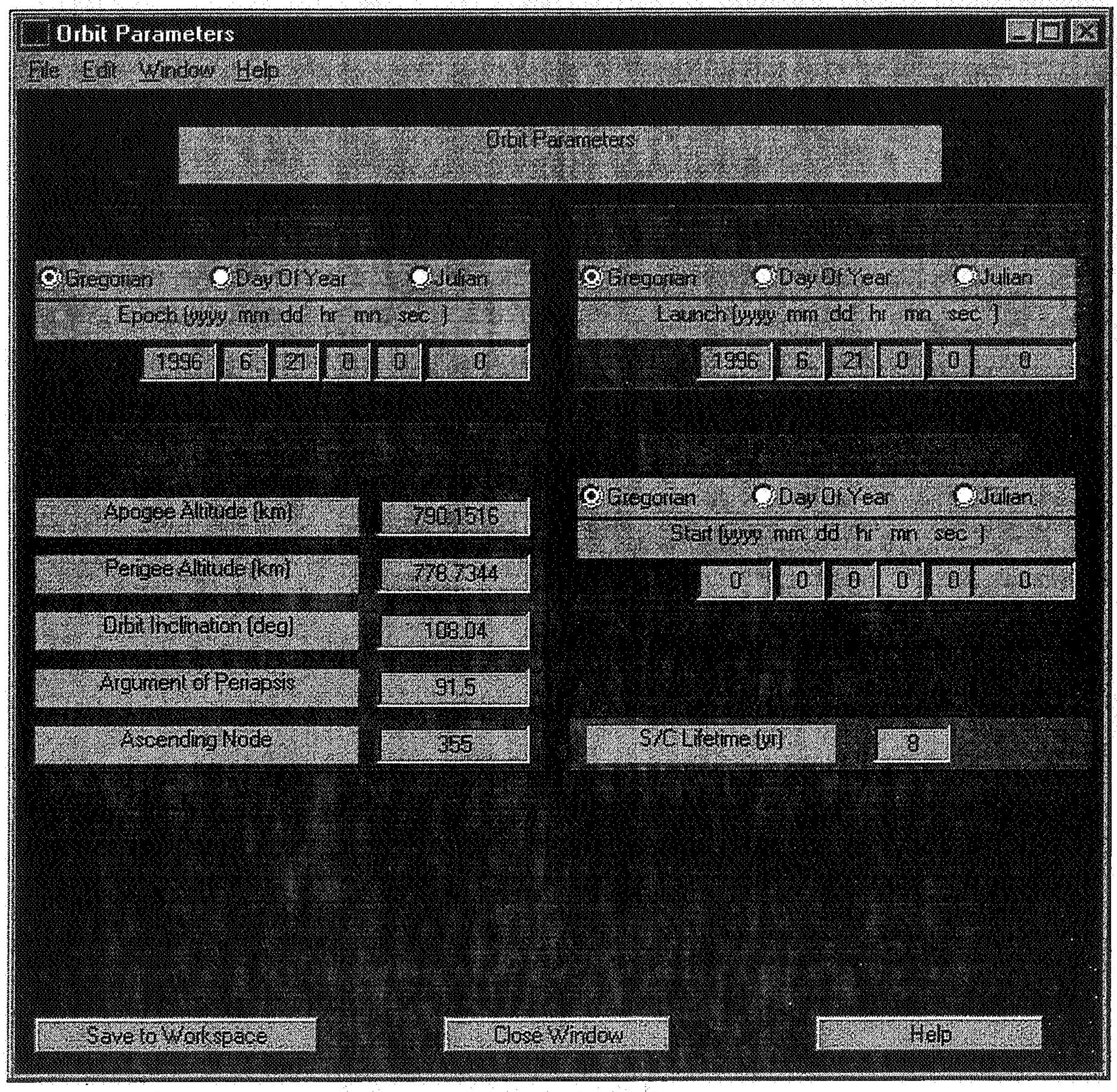

Figure 13: Orbit Parameters

The main CT menu, shown in Figure 15, includes parameters for the: transmitter, receiver, RF link, and typical ground station. It also includes flags to indicate whether Consultative Committee for Space Data Systems formatting and/or Reed-Solomon error coding are used. The transmitter parameters are: frequency, power, antenna beam width, antenna diameter, antenna type, line loss, antenna gain, and effective isotropic radiated power. The receiver parameters are: frequency, sensitivity, antenna beam width, antenna diameter, antenna type, antenna system noise temperature, antenna pointing error, and antenna gain. The RF link parameters are the uplink data rate and energy-per-bit to noise-density ratio (E_b/N_o), and the downlink data rate and E_b/N_o. The parameters for a typical ground station are: antenna system noise temperature, antenna diameter, transmitter power, receiver noise bandwidth, minimum elevation, maximum time to acquire station, and maximum time to loss of signal.

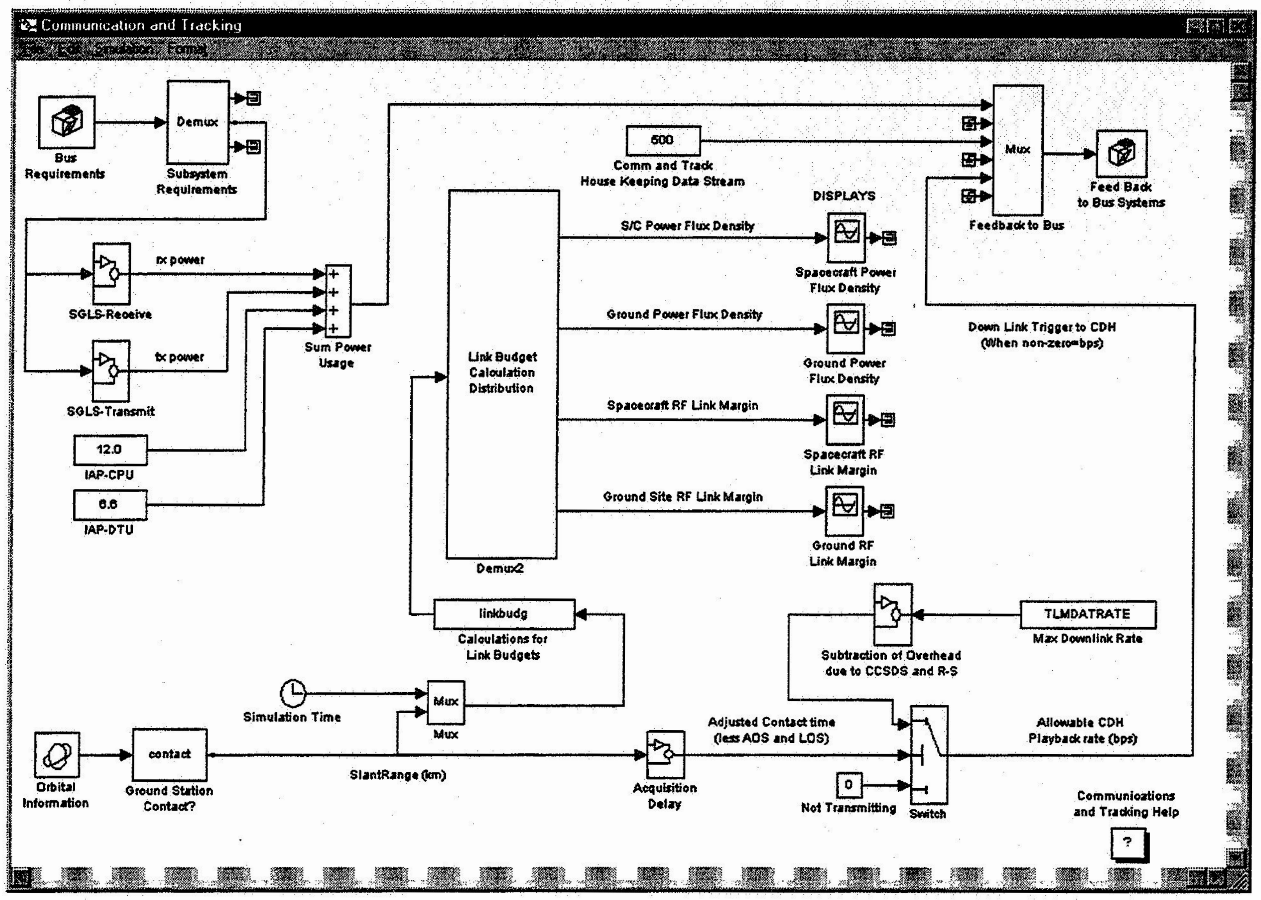

Figure 14: Communications and Tracking Subsystem

SPASIM also allows the user to select which ground stations are active through a ground station menu, shown in Figure 16. The other parameters in this menu are the ground station: name, latitude, longitude, and altitude. Currently there are 18 stations defined. The user can add or delete from this list through this menu.

The inputs to this model are the: radius, longitude, latitude, and net downlink rate from the CDH subsystem. This is the maximum rate at which the CDH subsystem can send data for downlinking. Its outputs are the power it requires, the rate at which it generates data, and the maximum net downlink rate. The maximum net downlink rate is the maximum rate at which the CT subsystem can accept data for downlinking. This model has predefined plots of time versus: downlink status, slant range, spacecraft power flux, spacecraft link margin, ground power flux, and ground link margin.

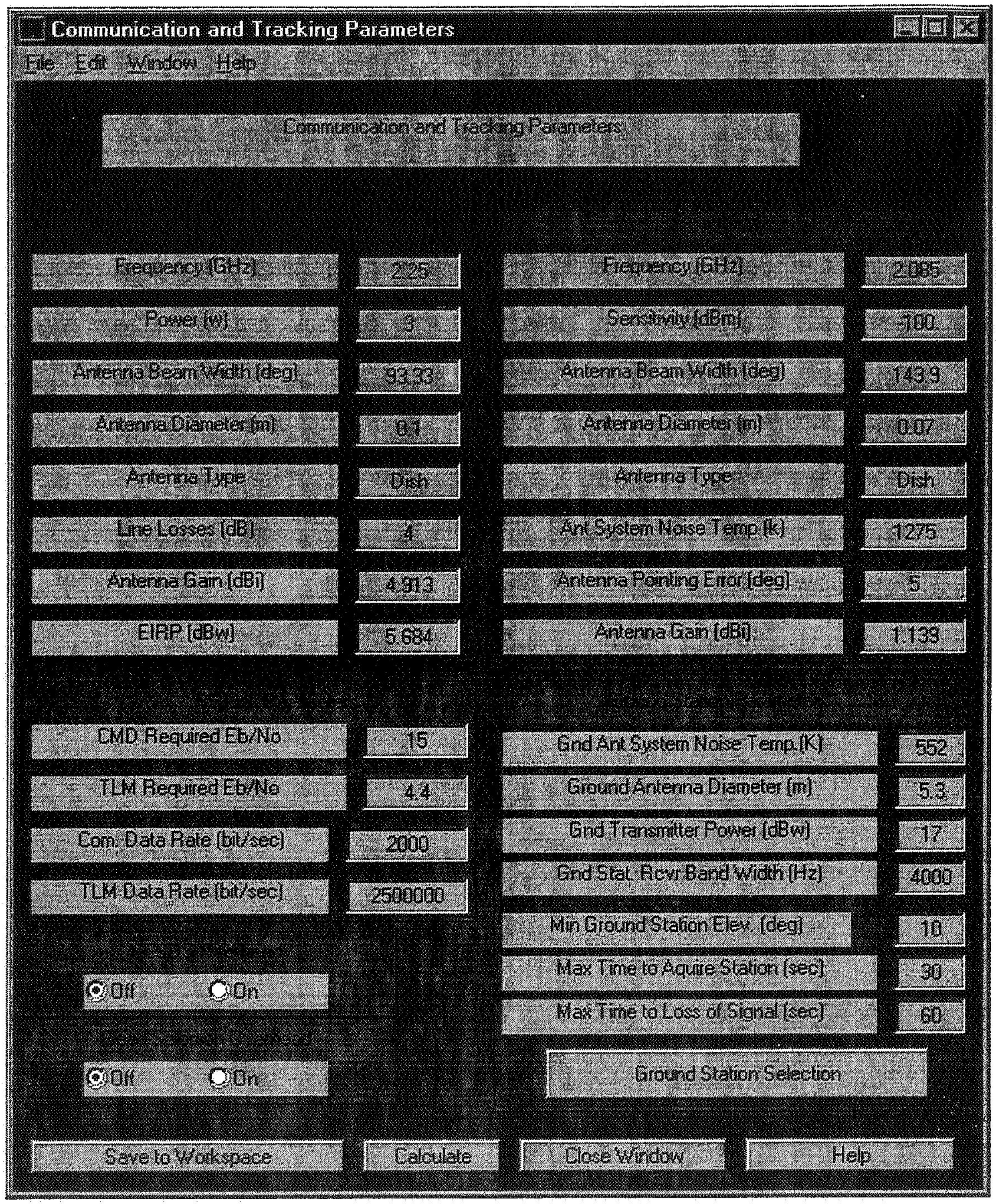

Figure 15: Communications and Tracking Parameters

Command and Data Handling

The purpose of the CDH subsystem, shown in Figure 17, is to store and retrieve data and commands and to execute those commands. Its model simulates the utilization of the data processing and storage capabilities and the requirements placed on other subsystems.

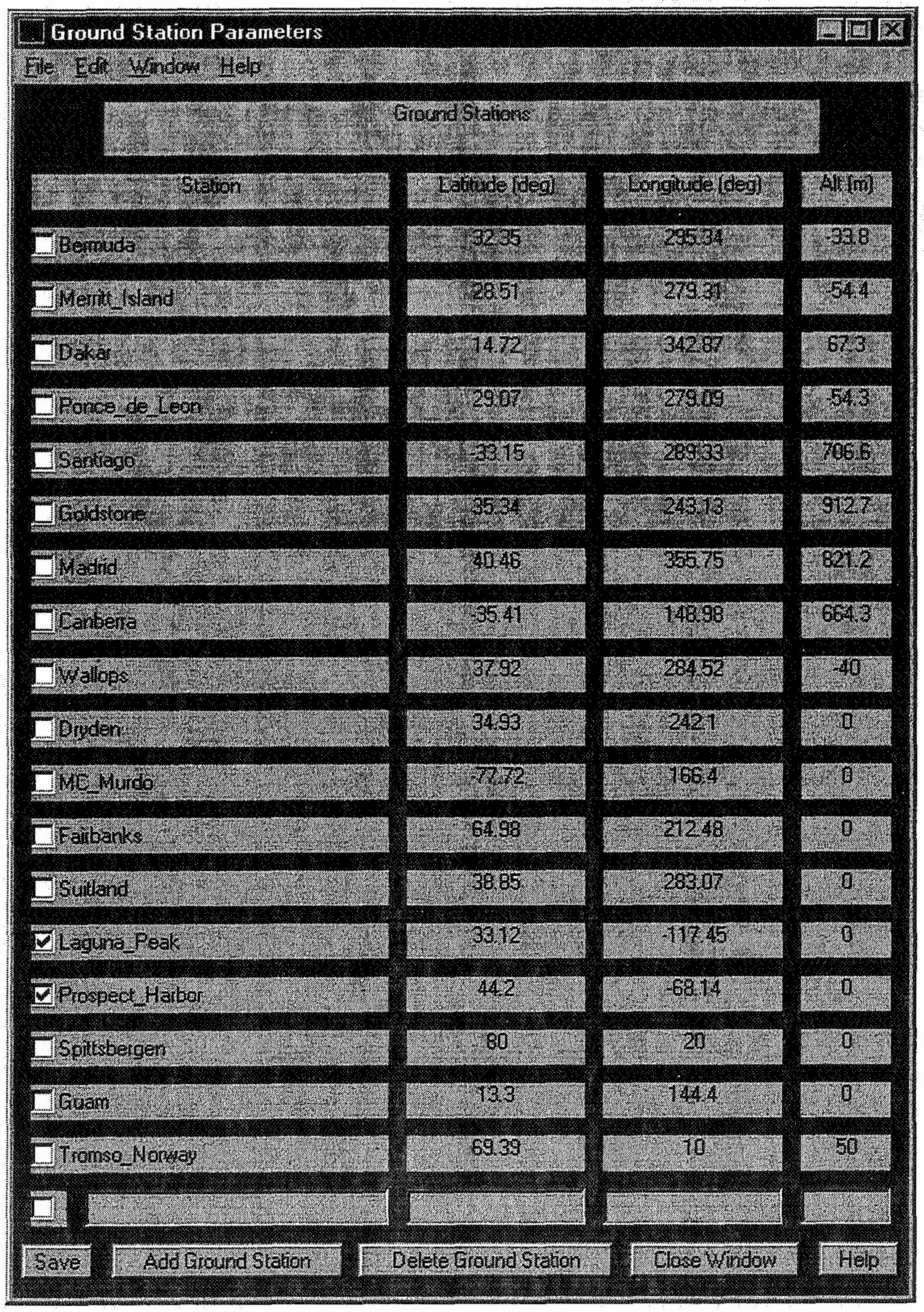

Figure 16: Ground Station Parameters

The parameters in the CDH menu, shown in Figure 18, specify the: maximum net processing rate, maximum storage capacity, maximum record rate, maximum playback rate, pre-storage formatting overhead, pre-storage error coding overhead, sensor sampling rate, number of CDH analog sensors, number of CDH discrete sensors, unallocated data rate, unallocated instruction rate, and time stamp size. The maximum net processing rate is the number of instructions per second that the processor can allocate to the spacecraft's processing requirements. Operating system and other software overheads as well as required processor margins are deducted from the raw processor capability to get this number.

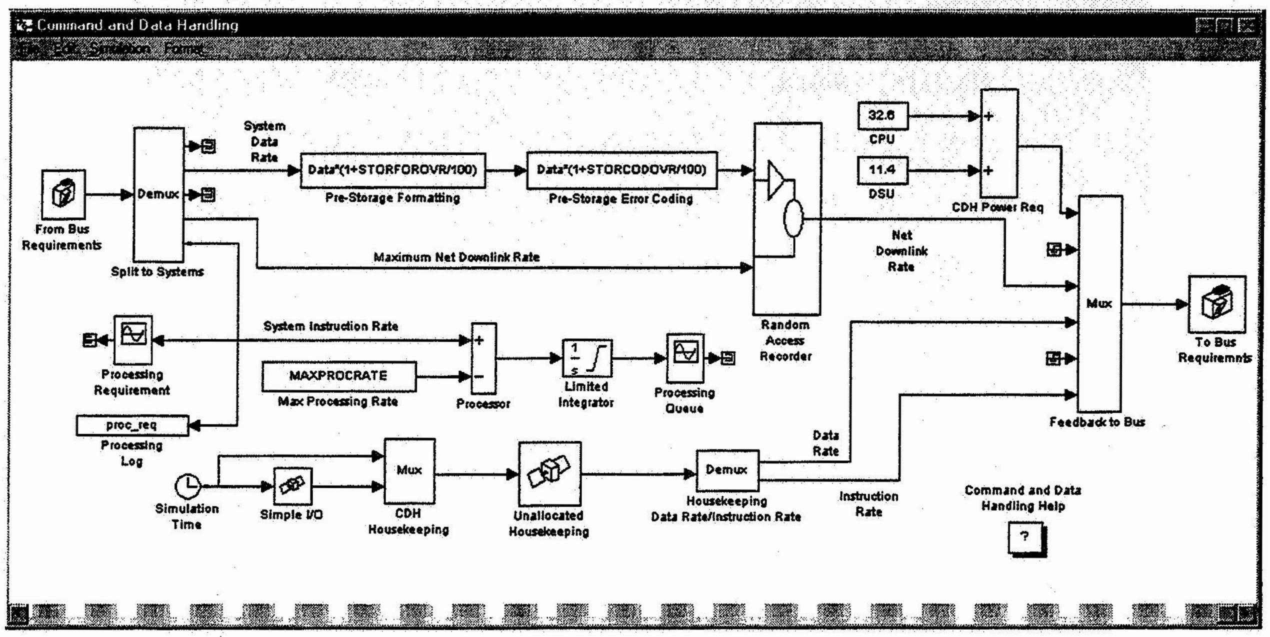

Figure 17: Command and Data Handling Subsystem

The sensor-sampling rate indicates how many times the spacecraft sensors need to be sampled per second. The data and instruction rates needed to support this function for the CDH subsystem are calculated based on this rate and how many analog and discrete sensors are in this subsystem. The unallocated data and instruction rates are the rates needed to support this function in the subsystems whose models do not calculate them. The time stamp size indicates the minimum number of bits needed to identify the time at which payload and housekeeping data was taken.

The inputs to this model are: the spacecraft data rate requirement, the spacecraft instruction rate requirement, and the maximum net downlink rate from the CT subsystem. The spacecraft data rate requirement is the rate at which payload and housekeeping data is generated. This data has to be stored until it can be downlinked.

The spacecraft instruction rate requirement is the number of instructions per second that the processor needs to execute to meet the requirements of the spacecraft. The rate at which the processor is executing instructions is subtracted from this input. The resulting rate is integrated to calculate the number of instructions queued.

The maximum net downlink rate is the maximum rate at which the CT subsystem can accept data for downlinking. This rate is zero when the spacecraft is not in contact with a ground station.

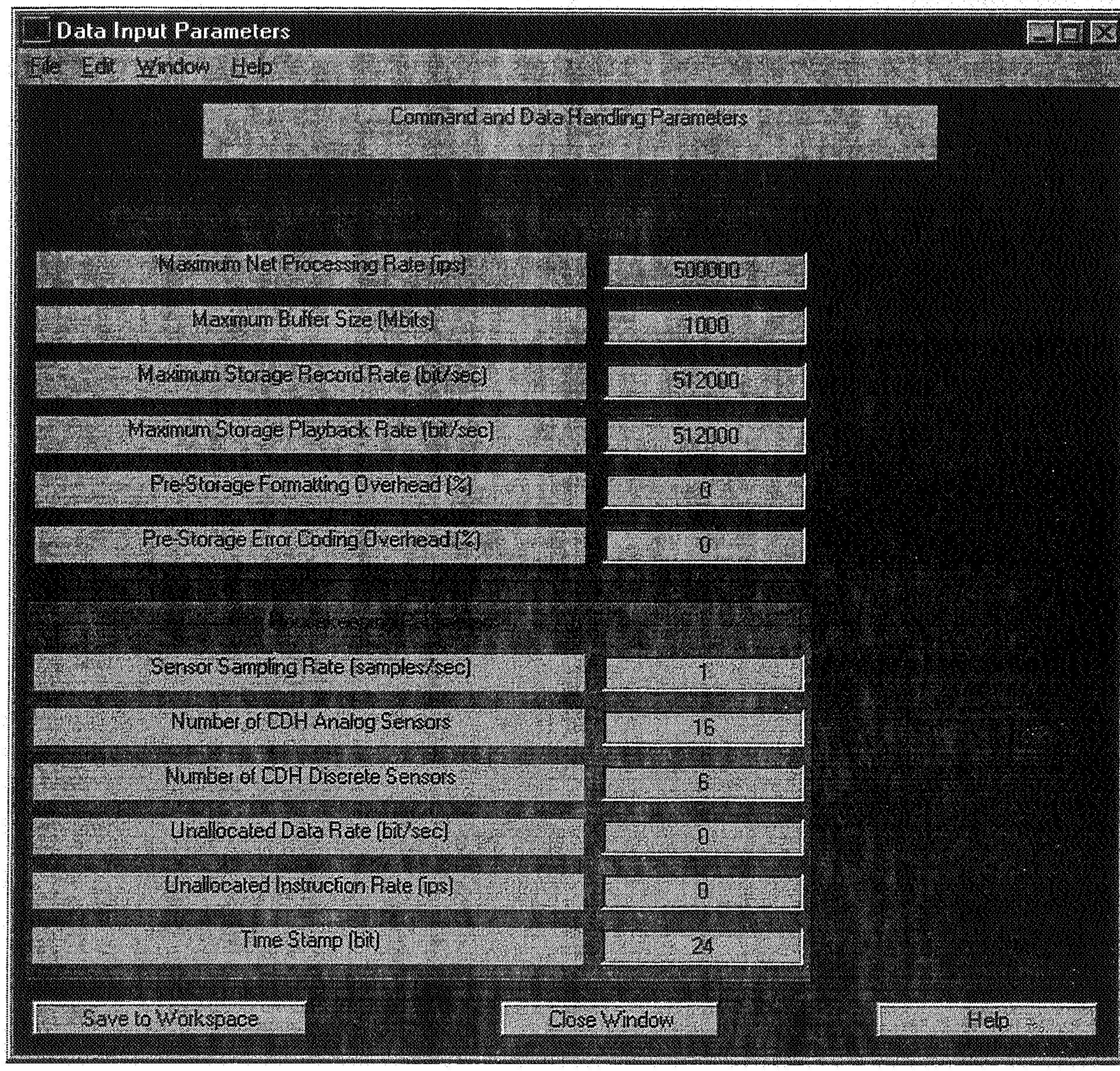

Figure 18: Command and Data Handling Parameters

This model has four outputs. Three of these are the contributions from this model to the spacecraft power requirement, the spacecraft data rate requirement, and the spacecraft instruction rate requirement. The fourth is the net downlink rate, which is the maximum rate at which the CDH subsystem can send data for downlinking. This model has predefined plots of time versus: data stored, spacecraft data rate requirement, data lost, spacecraft processing requirement, and processing queue.

RESULTS AND FUTURE WORK

On a 133 MHz Pentium laptop with Windows 95™ and 40 MB of RAM, SPASIM runs 26.67 times faster than real time. Six of SPASIM's 26 predefined plots are shown in the next six figures. Figure 19 shows the spacecraft's ground track. Figure 20 shows the slant range to a ground station. The data stored, which decreases when the spacecraft goes over a ground station, is shown in Figure 21. The power requirements, which increase when the spacecraft transmits

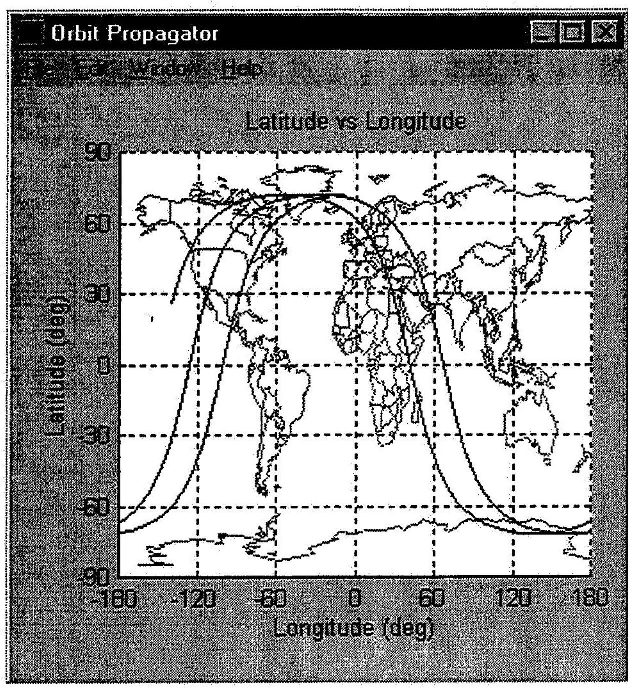

Figure 19: Orbit Track

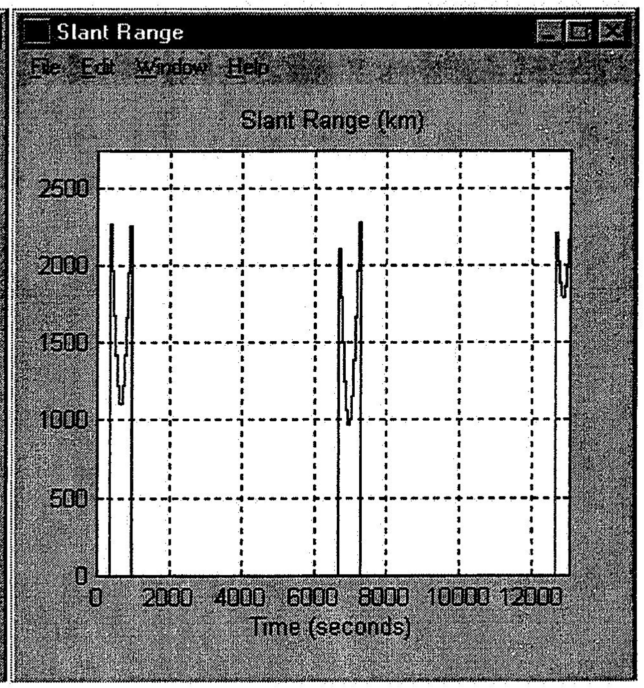

Figure 20: Slant Range

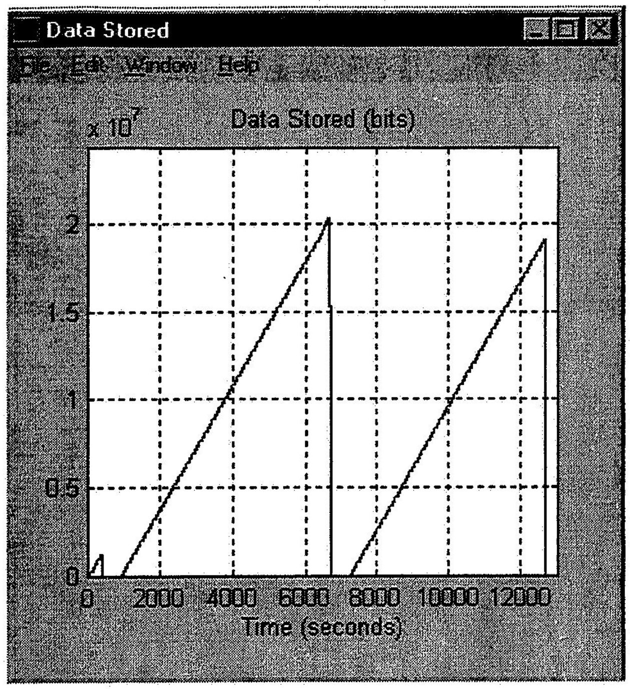

Figure 21: Data Stored

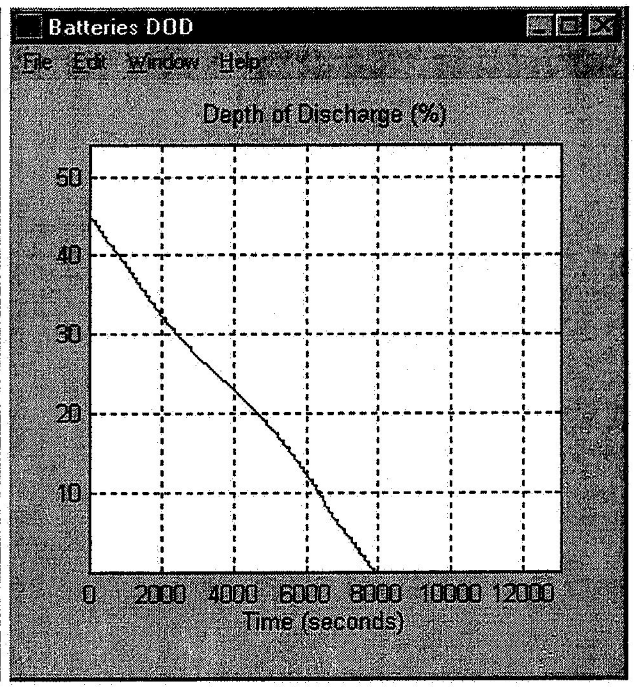

Figure 23: Depth of Discharge

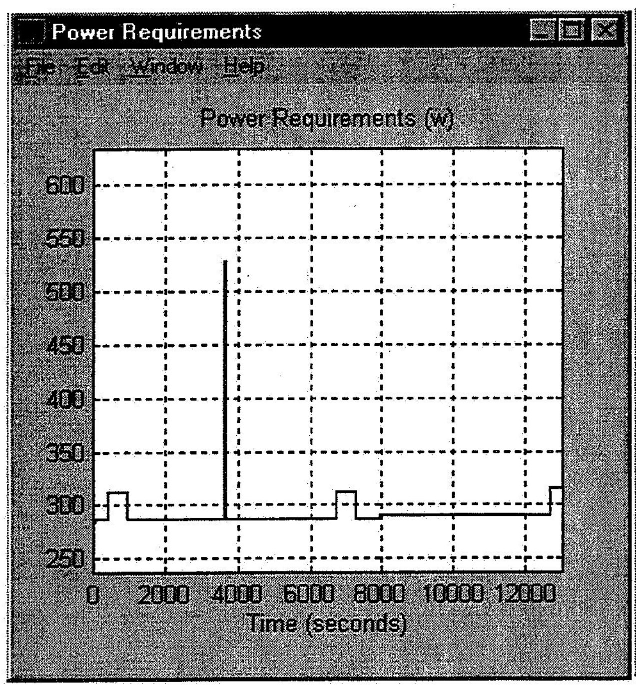

Figure 22: Power Requirement

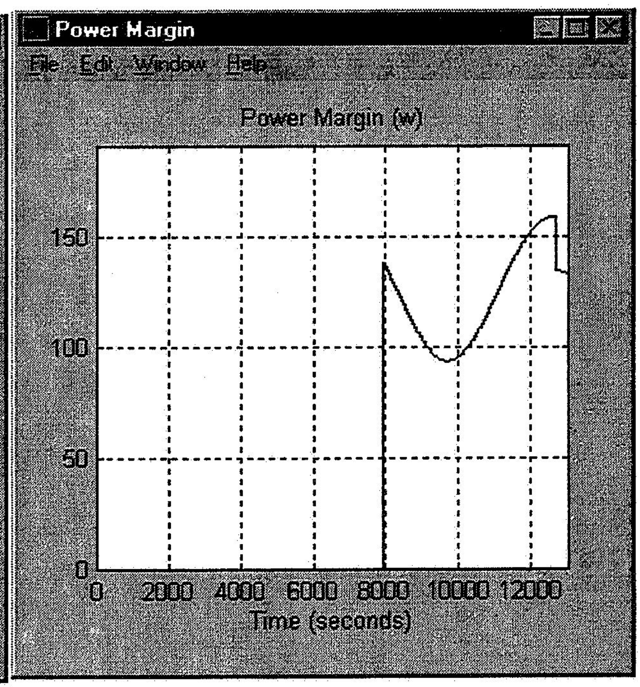

Figure 24: Power Margin

data, are shown in Figure 22. Figure 23 shows the DOD of the battery. The power margin, which is positive when the battery is fully charged, is shown in Figure 24.

SPASIM is now in beta testing at other NASA centers and within the U.S. aerospace industry. It is going to be validated further by applying it to as broad a range of spacecraft as possible and by comparing operational telemetry data with SPASIM predictions. It is also going to be expanded to crewed spacecraft like the space station and to planetary spacecraft.

SUMMARY

SPASIM can be used to validate spacecraft design and sizing estimates by performing an integrated time simulation of the spacecraft. This identifies resource bottlenecks or inadequacies resulting from simplified assumptions. Since SPASIM is a time based simulation, discrete events and duty cycles can be modeled and their resulting impacts can be assessed across all of the spacecraft. Failure modes and operational contingencies can be evaluated allowing the analyst to plan operations (what-if scenarios) and optimize the spacecraft performance for a range of mission scenarios. The SPASIM interface allows the analyst to easily change system functional architectures via block diagrams and to easily update performance characteristics of system components with parameter input menus. By changing specific parameters in a model, the user can assess the impacts of using different technologies.

SPASIM has been validated using several spacecraft designs that were at least at the Critical Design Review level. The user and programmer guide, including figures, is available on line as a hypertext document. This is an easy-to-use and expandable tool which is based on MATLAB® and SIMULINK®. It runs on Silicon Graphics Inc. workstations and personal computers with Windows 95™ or NT™.

ACKNOWLEDGEMENTS

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Ferebee, M.J., Jr.; P.A. Troutman; and D.W. Monell. 1997. "Satellite Systems Design/Simulation Environment: A Systems Approach to Pre-Phase A Design." In *Proceedings of the 35th Aerospace Sciences Meeting and Exhibit* (Reno, NV, Jan.6-9). AIAA, Reston, VA.

ACTIVE DISPENSER AS A POSSIBILITY TO EVOLVE LAUNCH VEHICLE CAPABILITIES

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ABS ONLY

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Daimler-Benz Aerospace Space Infrastructure,
Department Market Development Small Launch Vehicles

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Current trends on the satellite market show a rapidly growing LEO /MEO satellite constellations potential from the present time to the year 2000 and well beyond. On the other hand, it can be now predicted that the launch capacity will not be sufficient to serve the planned launch demand. In this context, the availability of a medium size launch vehicle that can fit the above purposes becomes more and more important.

Daimler-Benz Aerospace, Space Infrastructure Division (Dasa) in Bremen has initiated several studies that are aimed at finding such a launch vehicle that would be able to meet launch demand predicted for the nearest future, beginning from the year 2000. Evidently, the existing Russian/Ukrainian launch vehicles or their proven components are of great interest taking into account advantages of their flight proven reliability and cost effectiveness. However, the now available features of the existing launch vehicles may not be sufficient if applied to currently predicted satellite market.

One of the effective possibilities for increasing the launch vehicle capabilities in particular with respect to multiple launches (that are frequently desirable for constellations) was offered by GKB Yuzhnoye of Dniepropetrovsk, Ukraine and investigated in more details under one of Dasa/Yuzhnoye studies. An application of the "active" dispenser was considered that would allow the injection of multiple payloads into orbits with different orbital parameters and would provide for controlled deorbiting of the launch vehicle upper stage. This kind of a dispenser can also be specified as an additional (kick) stage, and can perform up to 14 ignitions. Such a provision can be effectively applied to launching small constellation satellites.

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CONCENTRATION DIFFUSION EFFECT ON HEAT AND MASS EXCHANGE OF A PARTICLE EVAPORATING IN THE ARCJET PLASMATRON

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Abstract

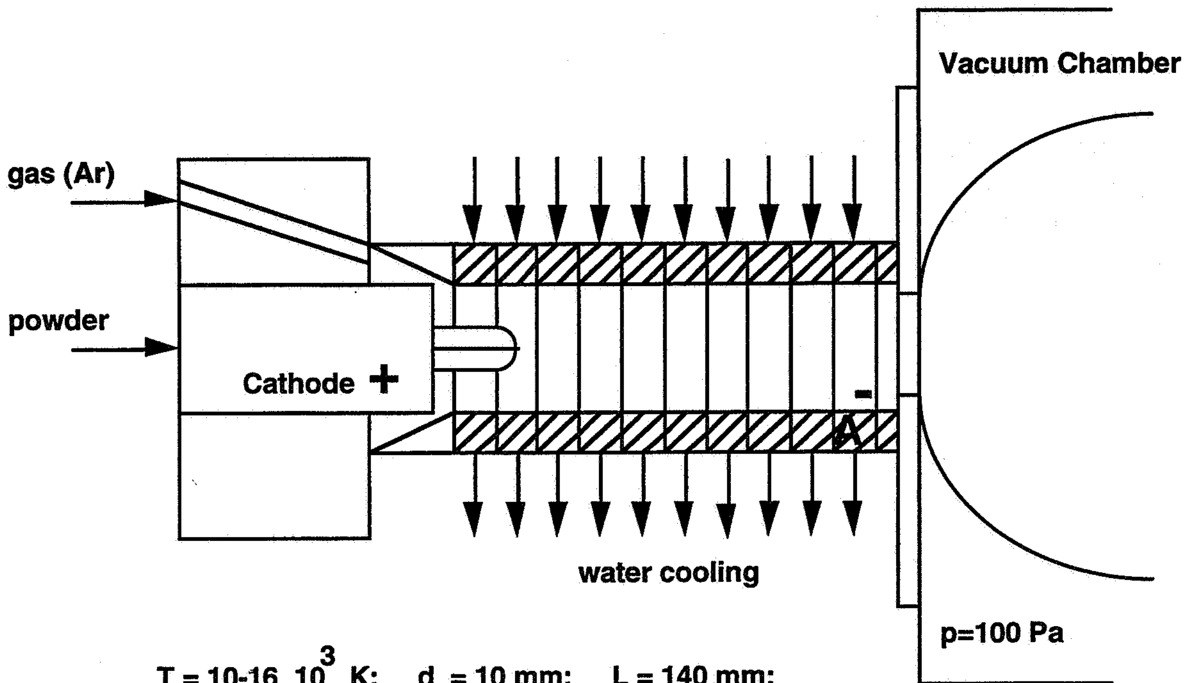
Investigation of phenomena in two-phase low-temperature plasma flows is of practical importance for various technologies such as plasma-assisted coating deposition. The effect of industrial application of this technology is growing making the research in this field one of the necessary components of technological progress. One of the most important phenomena in such two-phase plasma flows is evaporation of dispersal particles fed into the plasma flow. This paper investigates the particular effect of concentration diffusion on heat and mass transfer of particles evaporating in the electric arcjet plasmatron at low pressure.

Background

Two-phase plasma flows are of great interest, especially in cases when they are used for certain technological processes, e.g. plasma coating deposition. New requirements of these technologies make it necessary to further study the processes and phenomena occurring in the two-phase plasma flows at both macro- and micro-levels. There are known several plasma coating deposition techniques which may be split into two main groups: deposition at the atmospheric pressure, and vacuum deposition.

The former technique provides a very good coating thickness rate (about 10-100 μm/s), whereas the quality and chemical composition of such coatings remain at a relatively low level. On the other hand, vacuum coating deposition techniques (e.g. CVD, vapor deposition or magnetron sputtering) produce coatings of exceptional quality, approaching that of the initial material, but their efficiency in terms of the coating thickness rate is very low - about 0.11-7.8 nm/s or reaching 300 nm/s in some cases.

An alternative to these two methods may be the so-called low-pressure plasma deposition (LPPD) technique which implies use of an electric arcjet plasmatron with plasma flow exhausting into a vacuum chamber (p~100 Pa). This technique ensures a medium coating thickness rate (about 0.1-0.5 μm/s) which is still remarkably higher than that of vacuum techniques. On the other hand, LPPD coatings have better quality than those obtained with atmospheric methods and may approach the quality of the vacuum-deposited coatings. This is why studying LPPD and two-phase plasma flows in such conditions seems to be of particular interest. At the same time, it is possible to greatly enhance the LPPD coating quality by eliminating the liquid droplet phase and using only the vapor of the material to form the coating. Since experimental studies are quite expensive (because of the large variety of plasmatron operating modes and parameters to be investigated), the use of numerical simulation of such two-phase plasma flows would be of great importance. It would allow for variability of the parameters while requiring relatively little time and labor. The goal of this study was to investigate the particular effect of concentration diffusion on heat and mass transfer of particles evaporating in the electric arcjet plasmatron at low pressure. The schematic of the plasmatron used for this study is presented in Fig. 1.



$T = 10\text{-}16\ 10^3$ K; $d_c = 10$ mm; $L = 140$ mm;

$p_{in} = 10^4$ Pa; $D = 20$ mm; $I = 400\text{-}1200$ A;

Fig. 1. Schematic of an arcjet plasmatron with a two-phase plasma flow

The study was both computational and experimental. The computational results were compared to the experimental data obtained also by the authors.

Mathematical Modeling

The mathematical modeling was split into two parts: modeling of the plasma flow as such, and modeling of the dispersal phase (particles).

The plasma-generating gas flowing inside the plasmatron channel of constant diameter was modeled using the complete Navier-Stokes equation system [4], taking into account the plasma thermal non-equilibrium. The model was verified, its results were compared to the experimental data showing good correspondence [1, 4].

The dispersal phase modeling consisted of the three main components: impulse, heat and mass-exchange. The first approximation has been done for a single particle in the plasma flow. For the impulse exchange the impulse conservation equation has been used:

$$F_p d\tau = \overline{m}_p du\,; \quad F_p = \frac{\rho_g u_{rel}^2}{2} \cdot \pi \cdot r_p^2 \cdot C_D = f(\mathrm{Re}) \qquad (1)$$

where: F_p - the drag force acting on the particle, ρ_p - the particle density, u_{rel} - relative velocity of the particle ($u_{rel} = u_{gas}$-$u_{particle}$), r_p - radius of the particle, C_D - the drag coefficient, and $\overline{m}_p$ - is the mean particle mass between the two neighbouring states during the evaporation process.

The drag coefficient was calculated by the well-known formula [6]:

$$C_D = 24\mathrm{Re}^{-1}\ (1+0.15\mathrm{Re}^{0.687}) \qquad (2)$$

The heat exchange is governed by the Nu number, which is determined by the formula:

$$\mathrm{Nu} = 2 + 0.06\mathrm{Re}^{0.5}\cdot \mathrm{Pr}^{0.33} \qquad (3)$$

The mass exchange is very tightly interdependent with the heat exchange. This mechanism must also take into account concentration diffusion of the vapor from the particle surface into the surrounding vacuum [2]. The Maxwell's formula is used to describe the concentration diffusion [3,5]:

$$I = 4\pi D r_p (C_p - C_\infty) \qquad (4)$$

where: D - is the diffusion coefficient, C_p and C_∞ - is the concentration of the particle material vapor on the particle surface and infinitely far from it, respectively, and I - the particle evaporation rate. Thus, there must be two basic modes of particle evaporation: evaporation into the vacuum ($C_\infty = 0$) and that into the vapor environment of already vaporized material of the particle ($C_\infty \geq 0$). The vapor concentration on the particle surface depends on its temperature at a given pressure. Typically, the surface area change-rate is described by the Sreznevsky's formula and does not depend on the particle size [5]:

$$\frac{dS}{d\tau} = 8\pi D \cdot (C_p - C_\infty) \qquad (5)$$

Vapor concentration on the particle surface is proportional to the surface temperature. It has been found [5] that, when a particle evaporates into a vacuum or into a low pressure environment which is the case for low-pressure plasma coating deposition, the concentration diffusion phenomenon results in lower particle surface temperature and extended evaporation distance.

In the case when the dispersal mass flow rate is significant (exceeding 20% of the gas mass flow rate [7]), the dispersal phase does affect the gas parameters (temperature, velocity, electric conductivity, etc.), so another approach must be used to model such a two-phase flow.

Computation Results

The importance of taking into account the concentration diffusion when analyzing the heat and mass balance of a particle in plasma flow is emphasized by the chart in Fig. 2. The calculation has been done for a Cu particle of 10 μm in diameter and plasmatron thermal conditions at 600 A discharge current, 1 g/s mass flow rate of gas (Ar), diameter of the channel - 20 mm, pressure in the plasmatron channel $p=10^4$ Pa, and the channel length - 140 mm. The particle is considered to be moving along the channel axis.

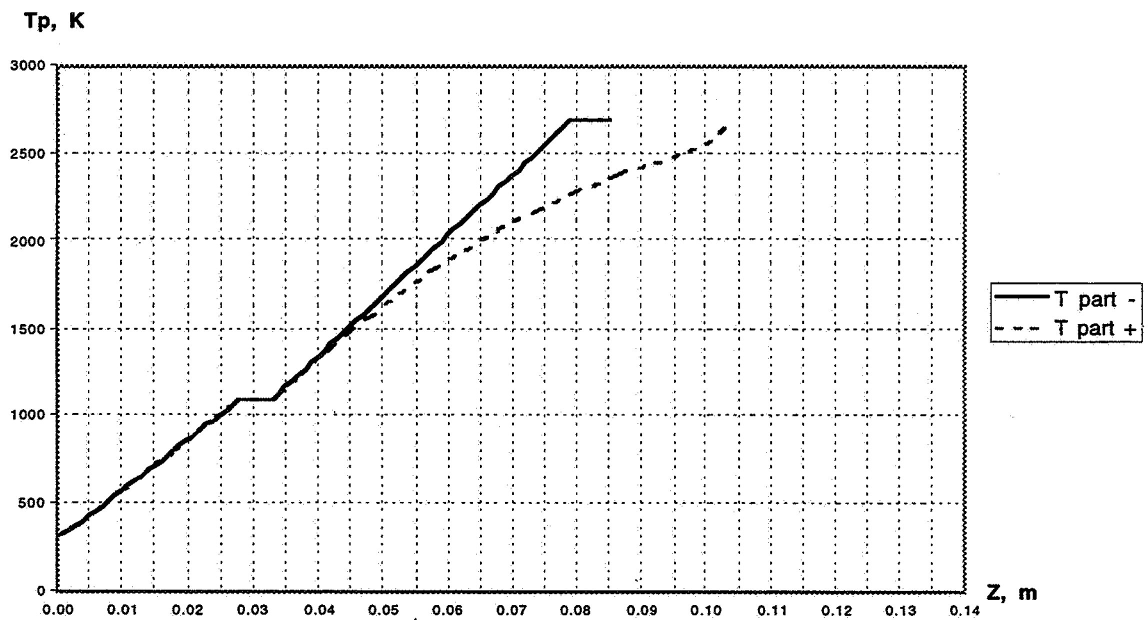


Fig. 2 . Computation results: particle evaporation with (dashed line) and without (solid line) concentration diffusion taken into account

By varying the power of the plasmatron and powder mass flow rate, and keeping all the other parameters constant, it is possible to build a set of curves showing: evaporation distance (l) vs. plasmatron power (N) (Fig. 3) which allows to determine the evaporation distance for the mass flow rate of the given dispersal material.

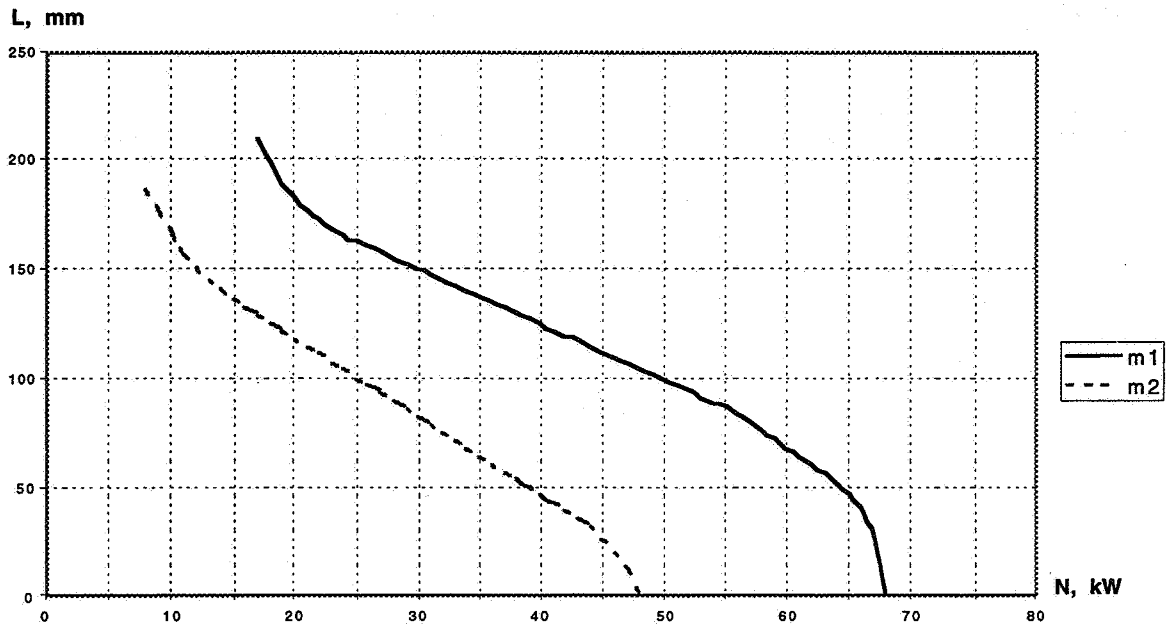


Fig. 3. Evaporation distances for small mass flow rates (dashed line - 0.01 g/s, solid line - 0.1 g/s) of 10-μm mesh powder vs. the discharge power applied

Experimental Study

The experimental part of the study included the deposition of samples onto the glass substrates, followed by microphotometry of the specimen to determine the percentage of the liquid phase in the flow.

The experiment was carried out for the electric current values I = 400, 600, 800, 1000 and 1200 A, and a plasmatron length of 140 mm. Cu-powder of 100, 60 and 40 μm mesh was fed into the device through the cathode. The powder mass flow rate constituted 0.1 of the gas mass flow rate. Therefore, according to the previous research [7], it is possible to assume that the dispersal phase does not affect the plasma flow parameters. The pressure at the plasmatron inlet was 10^4 Pa, and that in the vacuum chamber was 93 Pa. Glass substrates were moved into the supersonic part of the plasma jet at the distance of 120 mm from the plasmatron cut for about 3 seconds each. It is assumed that the supersonic flow is uniform there and plasma flow is "frozen" which means that there is no interaction between the particles (vapor atoms) and plasma. Thus, the parameters the flow had at the cut of the channel remain the same until the flow reaches the glass substrate. The mass of the glass plates was measured before and after the experiment, and the mass of the deposited material was calculated. The liquid particle diameter was determined by measuring the crater diameters on the glass using the microphotometer MF-2.

The patterns on the glass samples are shown on Fig. 4. It has been found out that, in order to fully evaporate the 40-60 μm grains of Cu powder the discharge current of the plasmatron must be not lower than 1000 A, and to evaporate the particles of 100 μm in diameter, the discharge current must be over 1200 A at the given plasmatron length of 140 mm.

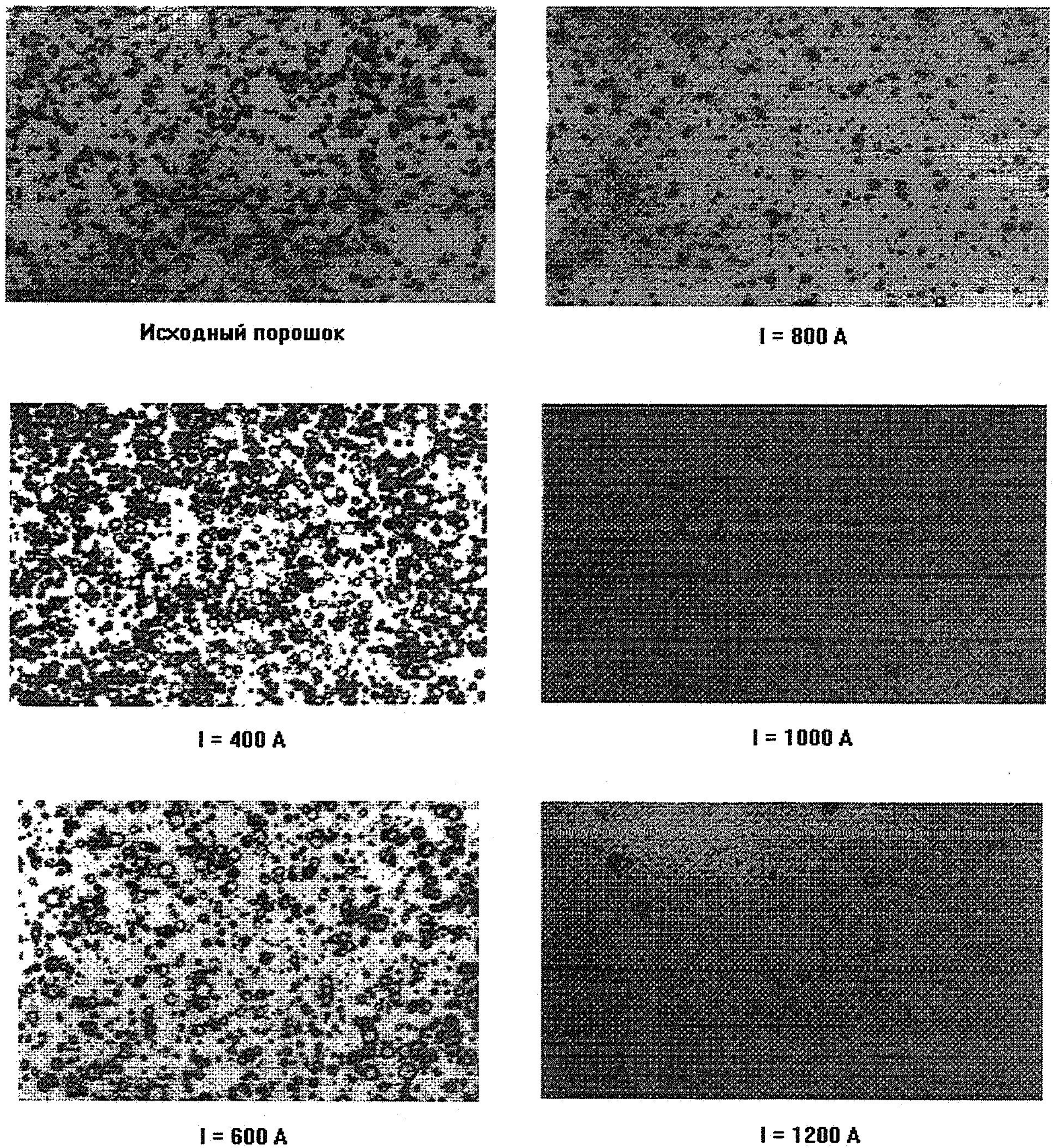


Fig. 4. Patterns of the initial powder and Cu-deposited glass samples for I = 400, 600, 800, 1000 and 1200 A

An interesting comparison of the calculation results and the experimental data on the χ vs. I chart is shown in Fig. 5. $\chi(\%)$ - is the portion of unevaporated mass of the dispersal phase, and I - is the discharge current. The data comparison uses the same particle diameter (100 μm), gas mass flow rate (1 g/s), and plasmatron length (L = 140 mm). The powder mass flow rate was 0.1 g/s.

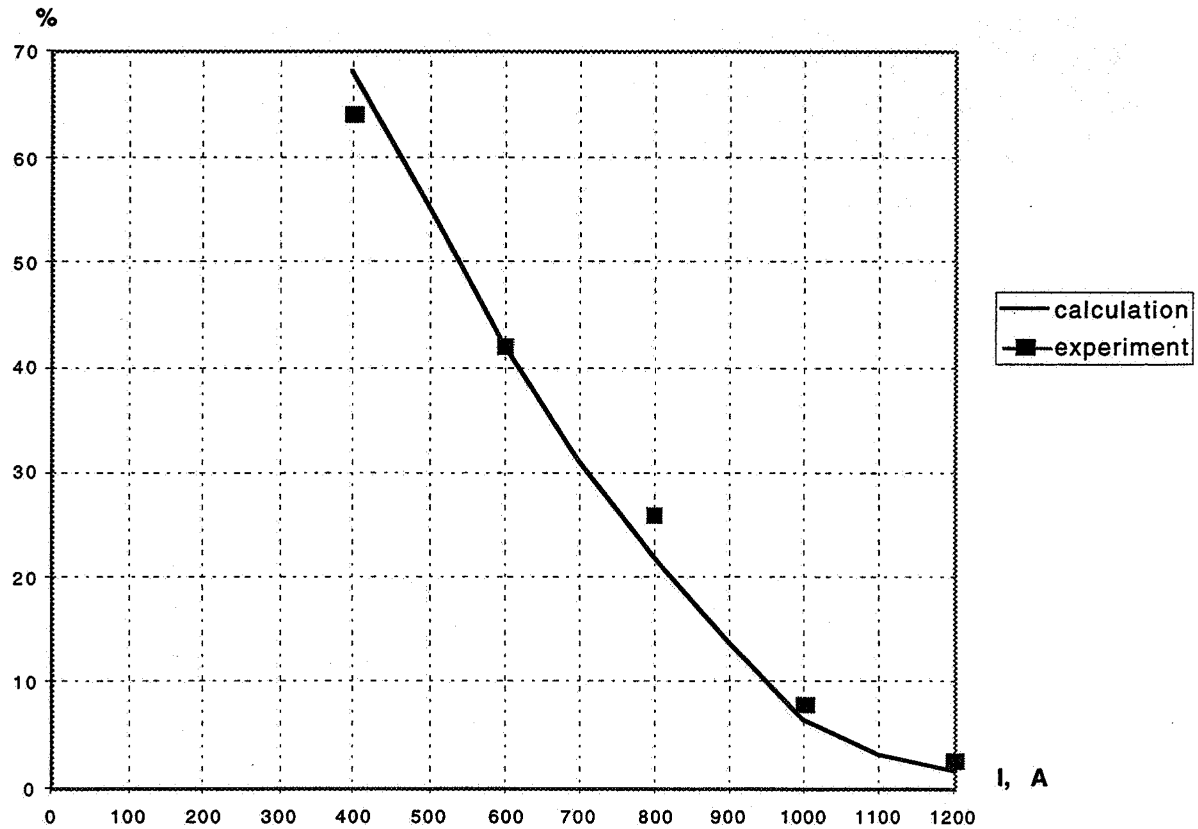


Fig. 5. Unevaporated portion of the powder mass flow rate χ vs. the discharge current

Conclusions

The particle evaporation process in the plasma flow generated by the electric arcjet plasmatron has been studied and the concentration diffusion mechanism has been taken into account. As a result, there has been developed a methodology for a two-phase plasma flow mathematical simulation requiring minimum of effort and time and provides a relatively good accuracy. Such plasma flows are used for technological purposes, for example, low-pressure plasma coating deposition. The computation results obtained showed a good correspondence with the experimental data. Thus, the methodology and the Turbo-Pascal computer program developed may be used for diagnostics of complex two-phase plasma systems, and to determine evaporation distance as well as the required power of the plasmatron necessary to fully evaporate the specified mass flow rate of the dispersal material fed into the device. It has been shown that the role of the concentration diffusion mechanism in the process of particle evaporation in plasma under low pressure conditions is critical and must be taken into consideration when dealing with two-phase plasma flows.

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STARTING AN "ADOPT-A-KID" SPACE CAMP

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This paper presents an overview of a process that any committed individual can use to establish a space outreach program for kids, in their local community. Members of the Houston space community successfully piloted such an outreach program in 1995. Ninety-four of Houston's underprivileged, inner-city kids were financially "adopted" by members of the space community and the general public, and brought to NASA's visitors center, for a tremendous evening of learning and space exploration. This paper outlines the human and physical elements that made the program a success, and challenges members of the world-wide space community to repeat such an outreach program with needy kids in their area of the world.

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The X PRIZE Competition

Peter H. Diamandis
Chairman/President, X PRIZE Foundation

Abstract

The X PRIZE is an international competition and education effort under development for the practical demonstration of the first reusable craft capable of taking three people to the edge of space (62 miles) and back. The X PRIZE is modeled after the great aviation prizes of the 1920's and 1930's. A particular example is the Orteig Prize of $25,000 which resulted in Charles Lindbergh's landmark flight from New York to Paris in 1927. This flight transformed aviation from the category of "stunts" to the beginnings of the multi-billion dollar aviation industry which has knit the world together.

The X PRIZE Foundation is a 501(c)(3) non-profit educational foundation which was created for the purpose of developing the prize and educating the public about the benefits public spaceflight.

Since its inception two years ago the X PRIZE has obtained the endorsement and support of numerous organizations and individuals including NASA, the Lindbergh Family, the Explorers Club, the Association of Space Explorers (ASE, the world membership of Astronauts and Cosmonauts).

No one today doubts the social and economic benefits which have resulted from the aviation industry which knits our planet together. Yet, within the memory of many, aviation was considered a foolhardy activity, suitable only for scientist sand daredevils. A series of prizes served as catalysts which dramatically altered public perceptions of aviation. Today, we are on the threshold of a similar change in perception about commercial spaceflight, beginning with the use of near-Earth space for rapid suborbital flight between locations on our planet's surface.

It is our belief that the X PRIZE can catalyze a set of technological and perceptual changes which will result in future benefits of equal magnitude to our present aviation industry.

The X PRIZE is now based at the St. Louis Science Center and is conducting a series of educational programs in addition to organizing the actual competition.

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These proceedings cover the sixth alumni conference of the International Space University, coordinated by the ISU U.S. Alumni Organization, which was held at Rice University in Houston, Texas, on July 11, 1997. The alumni conference gives graduates of the International Space University's interdisciplinary, international, and intercultural program a forum in which they may present and exchange technical ideas, and keep abreast of the wide variety of work in which the ever-growing body of alumni is engaged. The diversity that is characteristic of ISU is reflected in the subject matter of the papers published in this proceedings.

This proceedings preserves the order of the alumni presentations given at the 1997 ISU Alumni Conference. As in previous years, a special effort was made to solicit papers with a strong connection to the two ISU 197 Summer Session Program design projects: (1) Transfer of Technology, Spin-Offs, Spin-Ins; and (2) Strategies for the Exploration of Mars. Papers in the remaining ten sessions cover the departmental areas traditional to the ISU summer session program.

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