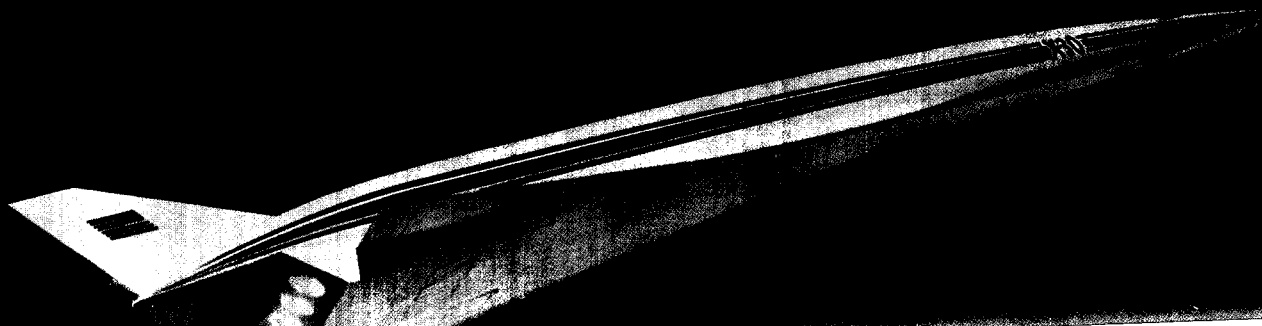


# NASA/USRA

## University Advanced Design Program

### 4th Annual Summer Conference

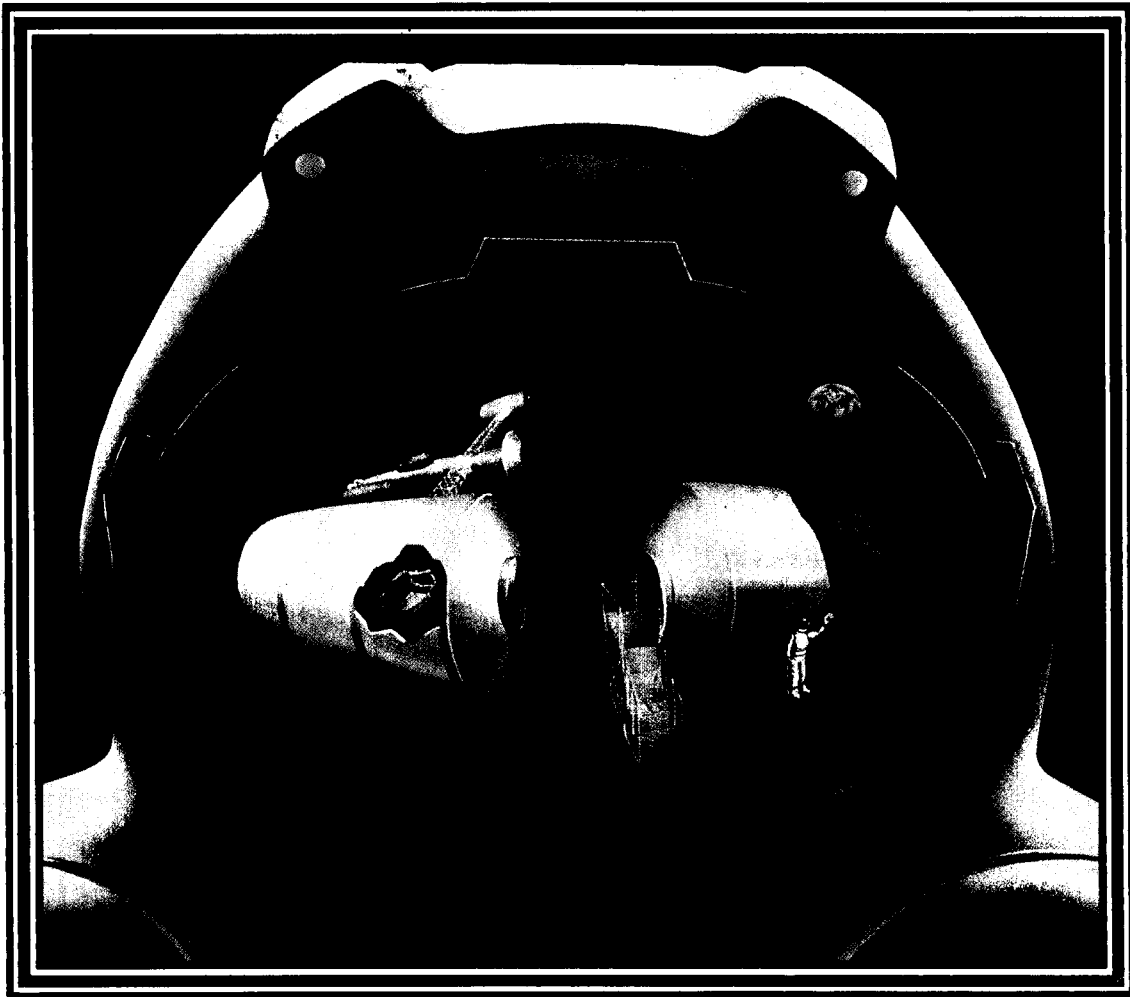


(NASA-TM-101750) AGENDA OF THE FOURTH ANNUAL SUMMER CONFERENCE, NASA/USRA UNIVERSITY ADVANCED DESIGN PROGRAM (NASA. John F. Kennedy Space Center) 105 pCSCI 22A

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NATIONAL AERONAUTICS & SPACE ADMINISTRATION  
UNIVERSITIES SPACE RESEARCH ASSOCIATION



Kennedy Space Center

June 13-17, 1988

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**AGENDA**  
**FOURTH ANNUAL SUMMER CONFERENCE**  
**NASA/USRA UNIVERSITY ADVANCED DESIGN PROGRAM**

**HOWARD JOHNSON PLAZA-HOTEL**  
**COCOA BEACH, FLORIDA**

**JUNE 13-17, 1988**

*Cosponsor: American Institute of Aeronautics and Astronautics (AIAA)*

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## *DEDICATION*

*The Advanced Design Program owes its success to the individual effort of a great many people from NASA, the universities, and USRA. Two of these individuals, Stan Sadin and Larry Spencer of NASA's Office of Aeronautics and Space Technology, will be leaving the program after this conference to pursue other activities. More than anyone else, they have helped shape the program and we will miss their wise counsel. Stan was responsible for conceiving the program in the first place and his guidance over the past three and one-half years has been a significant factor in its growth. Larry has exerted a similar influence in the Aeronautics portion of the program from the beginning of that activity two and one-half years ago. We wish them the best in their new endeavors and respectfully dedicate this conference to them.*

**FOURTH ANNUAL SUMMER CONFERENCE**

*Cosponsor: American Institute of Aeronautics and Astronautics (AIAA)*

The NASA/University Advanced Design Program is operated by the Universities Space Research Association (USRA) under grants from NASA Headquarters (NGT 21-002-080 and NGT 80001). Inquiries regarding the program may be directed to the Deputy Director:

Ms. Carol Hopf  
USRA--Suite 530  
2525 Bay Area Blvd.  
Houston, Texas 77058  
(713) 480-5939



**WORKSHOP ASSIGNMENTS**

**Planetary Surfaces and Operations (A):**

Prairie View A&M University  
Texas A&M University  
University of Wisconsin  
University of Illinois

**Planetary Surfaces and Operations (B):**

Florida A&M University/Florida State University  
Georgia Institute of Technology  
University of Houston

**Orbiting Facilities (A):**

University of Arizona  
University of Maryland  
University of North Dakota

**Orbiting Facilities (B):**

University of Central Florida  
Utah State University  
Worcester Polytechnic Institute

**Transfer Vehicles (A):**

University of Colorado  
Massachusetts Institute of Technology  
United States Naval Academy  
University of Virginia

**Transfer Vehicles (B):**

Auburn University  
University of Michigan  
Virginia Polytechnic Institute

**Biological Systems:**

Clemson University  
University of Florida  
Old Dominion University  
University of Puerto Rico

**Launch Systems:**

Florida Institute of Technology  
University of Texas  
University of Washington

**Aeronautics:**

University of California, Los Angeles  
California Polytechnic State University  
California State Polytechnic University, Pomona  
Case Western Reserve University  
University of Kansas  
Ohio State University  
Purdue University  
Rensselaer Polytechnic Institute

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*Agenda*

**MONDAY, JUNE 13 - *Challenger Main Ballroom***

7:00 - 10:00            **Informal Reception**

**TUESDAY, JUNE 14**

7:30 - 8:30            **Breakfast - *Poolside***

***Challenger Main Ballroom***

9:00                    **Opening Remarks:**  
                         **John R. Sevier, Director**  
                                 **Advanced Design Program, USRA**  
                         **Paul J. Coleman, Jr., President**  
                                 **USRA**  
                         **Larry Spencer, Advanced Technology Program Manager**  
                                 **NASA Headquarters**

10:00                    **Charge to Workshops:**  
                         **John Alred, Technical Assistant, Advanced Programs Office**  
                                 **Johnson Space Center**

10:15                    **Break**

**SESSION I - *Challenger Main Ballroom***

**AERONAUTICS**

10:30                    **Purdue University - *High Speed Civil Transport***

11:30                    **University of Kansas - *Transpacific High Speed Civil Transport***

12:30                    **Lunch - *Poolside***

1:30                    **Ohio State University - *High Speed Passenger***  
                                 ***Transportation System***

2:30                    **California Polytechnic State University, San Luis Obispo -**  
                                 ***V/STOL Aircraft Concepts***

3:30                    **Break**

4:00                    **University of California, Los Angeles - *Hypersonic Vehicle***  
                                 ***Design; Advanced Aeronautical Design***  
                                 ***(Hypersonic Drone)***

5:00                    **California State Polytechnic University, Pomona -**  
                                 ***Planform Effects on High Speed Vehicles***

7:00 - 9:00            **Beach Pizza Party - *Poolside***

**SESSION II**

- 10:30                    **SPACE WORKSHOP DISCUSSIONS**
- Planetary Surfaces & Operations (A) - *Apollo***
- Planetary Surfaces & Operations (B) - *Boardroom***
- Orbiting Facilities (A) - *Saturn***
- Orbiting Facilities (B) - *Mercury***
- Transfer Vehicles (A) - *Atlas***
- Transfer Vehicles (B) - *Skylab***
- Biological Systems - *Plaza Tower Room 401***
- Launch Systems - *Gemini***
- 12:30                    **Lunch - *Poolside***
- 1:30                      **Space Workshop Discussions (cont.)**
- 5:00                      **Space Workshop Report Planning Session**
- 7:00 - 9:00            **Beach Pizza Party - *Poolside***



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*Agenda*

**WEDNESDAY, JUNE 15**

7:30 - 8:30            Breakfast - *Poolside*

**SESSION I - Enterprise/Columbia**

**AERONAUTICS**

8:30                    Rensselaer Polytechnic Institute - *The Apollo  
Lightcraft Project*

9:30                    Case Western Reserve University - *Hypersonic Transport  
Aircraft for Commercial Trans-Pacific Flight*

10:30                  Break

10:45                  Aeronautics Summary

11:45                  Lunch - *Poolside*

**SPACE**

1:00                    Utah State University - *The Lunar Orbital Prospector (LOP)*

2:00                    Georgia Institute of Technology - *Construction Equipment for  
Lunar Base Site; Preparation and Excavation*

3:00                    Break (Free time/workshops)

7:00                    Dinner- *Challenger Main Ballroom*

8:00                    Featured Speaker

**SESSION II - Atlantis/Discovery**

8:30                    University of Florida - *Bioregenerative System for Growing  
Higher Plants in Space*

9:30                    Texas A&M University - *Lunar & Martian Power Plants;  
Nuclear Power Systems on Space Stations; Lunar Base;  
Deep Space Scientific Probe*

10:30                  Break

10:45                  Aeronautics Summary

11:45                  Lunch - *Poolside*

- 1:00                    **University of Michigan - *Personnel Transportation Systems  
Between Earth and Mars***
- 2:00                    **Virginia Polytechnic Institute - *Space-based Laser Powered  
Orbital Transfer Vehicle at Earth and Mars***
- 3:00                    **Break (Free time/workshops)**
- 7:00                    **Dinner- *Challenger Main Ballroom***
- 8:00                    **Featured Speaker**

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*Agenda*

**THURSDAY, JUNE 16**

7:30 - 8:30            *Breakfast - Poolside*

**SESSION I - Enterprise/Columbia**

8:00                    *University of Colorado - Cislunar Space Infrastructure*

9:00                    *University of Texas - Bootstrap Lunar Base; Convertible Lower Lunar Lander Stage; Mars Mission Crew Transfer Vehicle; Earth Orbiting Transportation Node; Lunar Shack; Lunar Navigation System; Lunar Radiation Protective Structures*

10:00                  *Break*

10:15                  *Auburn University - Unmanned Multiple Exploratory Probe System for Mars Observation*

11:15                  *Clemson University - Zirconia Cell for Mars Oxygen Processor; Habitat and Manufacturing Facilities*

12:15                  *Lunch - Poolside*

1:15                    *University of Illinois - Manned Mars Aircraft and Delivery System Design*

2:15                    *University of Maryland - Space Station Automation and Robotics*

3:15                    *Break*

3:30 - 5:15            *Workshop Reports - Challenger Main Ballroom*

7:30 - 9:30            *Industry Roundtable - Challenger Main Ballroom*

**SESSION II - Atlantis/Discovery**

8:00                    *Old Dominion University - Mars Oxygen Production System*

9:00                    *United States Naval Academy - Unmanned Star Probe*

10:00                  *Break*

10:15                  *University of Central Florida - Space Station Integrated Refuse Management System*

11:15                  *University of North Dakota - Variable Gravity Research Facility*

12:15                  *Lunch - Poolside*

- 1:15                    **Massachusetts Institute of Technology - *A Mixed Fleet of Manned Space Vehicles for Solar System Exploration***
- 2:15                    **University of Washington - *Ram Accelerator Direct Launch Mass Launch System***
- 3:15                    **Break**
- 3:30 - 5:15            **Workshop Reports - *Challenger Main Ballroom***
- 7:30 - 9:30            **Industry Roundtable - *Challenger Main Ballroom***



### **The Program:**

The NASA/USRA University Advanced Design Program is a unique national program that brings together NASA engineers and students and faculty from U.S. engineering schools by integrating current and future NASA space/aeronautics engineering design projects into the university engineering design curriculum. The Advanced Space Design Program was conceived in the fall of 1984 as a pilot project to foster engineering design education in the universities and to supplement NASA's in house efforts in advanced planning for space and aeronautics design. ("Advanced" was defined as being post Space Station Initial Operating Configuration.) Nine universities and five NASA centers participated in the first year of the pilot project. Close cooperation between the NASA centers and the universities, the careful selection of design topics, and the unbridled enthusiasm of the students resulted in a successful first year and the decision to extend the experiment to a second year. Nineteen universities (including the original nine) and eight NASA centers were involved in the second year's effort. The same year saw the formation of the Advanced Design Program for Aeronautics. The Aeronautics Program has eight universities and three NASA research centers (Lewis, Ames and Langley) participating. Expanding to thirty-four universities studying space and aeronautics and eight NASA centers in 1988, the Advanced Design Program continues to mature and strengthen.

The study topics include potential space and aeronautics projects which could be undertaken during a 20-30 year period beginning with the Space Station Initial Operating Configuration (IOC)

scheduled for the early to mid-1990's. This includes system design studies for both manned and unmanned endeavors; e.g., lunar launch and landing facilities and operations, variable artificial gravity facility for the Space Station, manned Mars aircraft and delivery system, long term space habitat, construction equipment for lunar base, Mars oxygen production system, trans-Pacific high speed civil transport, V/STOL aircraft concepts, etc.

The NASA/USRA University Advanced Design Program continues to be an effective mechanism for integrating the educational objectives of the university community with the advanced engineering design efforts at the NASA centers.

### **The Conference:**

The summer conference provides the opportunity for the universities to report on the results of their design projects. The oral reports are made by students (typically 3-4 per university) representing the design team (as many as 65 students) who worked on the project during the academic year. The presentations often evoke considerable discussion, particularly among schools who may have taken somewhat different approaches to similar design topics.

Workshops comprise the other major element of the summer conference. Organized along technology lines, the workshops will be an opportunity for students, professors and NASA representatives to discuss this year's projects, how they relate to one another, and the technology needed to accommodate the advanced designs. It is also an opportunity to critically assess the program and to make recommendations for improvements.

**HYPERSONIC DRONE VEHICLE DESIGN:  
A MULTIDISCIPLINARY EXPERIENCE**

**UNIVERSITY OF CALIFORNIA, LOS ANGELES**

UCLA's Advanced Aeronautic Design group focused their efforts on design problems of an unmanned hypersonic vehicle. It is felt that a scaled hypersonic drone is necessary to bridge the gap between present theory on hypersonics and the future reality of the National Aerospace Plane (NASP) for two reasons: (1) to fulfill a need for experimental data in the hypersonic regime, and (2) to provide a testbed for the scramjet engine which is to be the primary mode of propulsion for the NASP.

The group concentrated on three areas of great concern to NASP design: propulsion, thermal management, and flight systems. Problem solving in these areas was directed towards design of the drone with the idea that the same design techniques could be applied to the NASP.

A seventy degree swept double-delta wing configuration, developed in the 70's at NASA Langley, was chosen as the aerodynamic and geometric model for the drone. This vehicle would be air-launched from a B-1 at Mach 0.8 and 48,000 feet, rocket boosted by two internal engines to Mach 10 and 100,000 feet, and allowed to cruise under power of the scramjet engine until burnout. The vehicle would then return to base for an unpowered landing. Preliminary energy calculations based upon the flight requirements give the drone a gross launch weight of 134,000 lb. and an overall length of 85 feet.

The efforts of the Propulsion Group have been directed towards developing the tools to analyze the effects of aircraft geometry and free stream conditions on scramjet performance. Specifically, the group has been con-

cerned with the shock structure, boundary layer growth, and inlet geometry on the underbody of a Mach 10 drone research vehicle, as well as with the impact of these parameters on the efficiency and thrust of the scramjets.

Through researching NASA documents, it was found that engine airframe integration is very important, particularly for the forebody compression and afterbody expansion. Also, boundary layer growth and control were determined to be important parameters affecting scramjet performance.

A lack of empirical data regarding scramjets, however, has led the group to explore both preexisting computer programs, as well as creating others as needed, as a means to model free stream flow and engine performance. Application of each program is shown in Figure 1. Briefly, the computer codes allow the calculation of the shock structure and boundary layer growth along the fore- and afterbodies of the vehicle, as well as the calculation of the inlet geometry and combustor performance. Using the programs in conjunction with one another, nose-to-tail analysis of various parameters affecting scramjet performance such as ramp angle, boundary layer suction, surface cooling, and inlet geometry can be performed. A change in these parameters can be analyzed faster by using the programs, and the merits of various design changes appraised. The group also uses SCRAM, which is a 1-dimensional program used to give broad estimates of scramjet performance, to make a first estimate of the impact of the designs on engine per-

formance and to check the results obtained using the other programs.

For the first analysis of the configuration, SEAGULL and STUB were used to calculate the shock structure on the forebody, and EDDYBL was used to calculate the boundary layer properties. The boundary layer was found to begin transition forty feet from the nose of the aircraft, and was unacceptably thick entering the engine. Boundary layer diversion (as opposed to boundary layer suction) was discarded as an option due to the large cooling required on the sharp leading edge of the splitter plate. Boundary layer suction at the skin was also investigated as a means of preventing transition.

Overall, the efficiency and thrust delivered by the design were calculated from momentum considerations (Figure 1).

The phenomenon of aerodynamic heating is first experienced on reentry vehicles where heating rates on the order of tens of  $\text{Btu/ft}^2\text{s}$  are encountered for relatively short periods of time. In spite of high heating rates, the short time interval that the vehicles spend in the critical regime allows the problem to be managed with passive systems such as ablative surfaces or heat sink structures.

Air breathing vehicles, while generally experiencing less intense heating rates, tend to be in the critical regimes for much longer periods of time. In this case, the aircraft skin does not possess sufficient thermal capacity to act as a heat sink. In addition, requirements for reusability,

and turnaround time, rule out ablative surfaces.

With these considerations the aim of the Thermal Management Group is to design and compare active cooling systems that meet the requirements of the hypersonic drone and its flight envelope. To keep the problem manageable, heating data was drawn from a single flight condition of Mach 10 at 100,000 feet. The active cooling system is designed to cool the nose of the aircraft aft to where aerodynamic heating is not nearly as severe. The nose is configured as a 10 degree half-angle cone capped with a 1 inch radius hemisphere. Coolant impinges on the back of the hemisphere and this is channeled through tubes aft along the skin as shown in the figure below.

ACSES, or Actively Cooled Structure Evaluation System, is a Fortran code written by the group that models the system for a given flight condition and system variables. The purpose of ACSES is to allow the operator to vary such parameters as tube diameter, coolant mass flow rate, and tube spacing and subsequently to study the effects on maximum skin temperature, total heat load, and coolant pressure drop. In analyzing the impingement cooling manifold, the code uses Livingood's correlation for the Nusselt number of the jet and determines the final temperature of the coolant exiting the manifold.

The tube system was analyzed using the fin concept and divided into sections resembling thin-walled circular shells. The temperature at the midpoint between two tubes could not exceed the maximum allowable wall



temperature. In the code, a provision was provided to branch each tube (at a specified distance) into two additional tubes to further increase the heat transfer and reduce the midline temperature of the skin.

The first coolant considered was liquid hydrogen. The other coolants which are applicable, Therminol 60 and aqueous glycol (30%), require the addition of a heat exchanger. A heat exchanger design code along with ACSES were used to design systems with these coolants. Graphs were generated showing the relation of the number of tubes to the skin maximum temperature at each station for a constant mass flow rate (Figure 2).

The techniques and tools developed this year by the Thermal Management Group will allow analysis and initial design aircraft configuration.

The Flight Systems Group has been involved in conceptual sensor design to meet the needs of the Thermal Management and Propulsion groups. Devices were designed to measure: outer skin friction shock-wave structure, engine temperatures, flow velocity, flow constituent concentration, and pressure. Factors that made the design challenging were extremely high temperatures, pressures, flow velocities, numerous shock waves, and turbulence.

Concepts from such diverse fields as: quantum mechanics, solid-state physics, fiber optics, and fluid dynamics were employed. A number of analysis devices have been designed. The flush Air Data System measures angle of attack, velocity, and angle of slideslip of the aircraft. This sensor consists of four sets of

small pressure orifices imbedded around the nose, aligned along its vertical and horizontal diameters.

The Platinum Resistance Thermometer is a sensor that uses a temperature sensitive platinum resistor in a bridge circuit in a current-compensated configuration.

A pressure sensor uses a Michelson-type interferometric technique to measure the displacement of a material under the load of outside pressure by embedding mirror-like materials inside the detector, a 5cm by 5cm by 3cm block of graphite transparent to laser beams, and processing the information about displacement using digital electronics to continuously monitor the pressure.

The MOSFET Hydrogen Sensor measures hydrogen concentration using a semiconductor with a palladium surface. Placed in contact with the hydrogen flow, accurate real-time measurements are produced.

A Fiber Optic Pressure Sensor is a sensor that consists of an optical fiber wrapped around an elastic drum that is allowed to vibrate and bend in response to pressure. A known intensity laser beam is injected into the fiber and the output intensity is a function of the pressure the drum is exposed to, thereby allowing accurate pressure measurement.

The Coherent Anti-Stokes Spectroscopy (C.A.R.S.) analyzes flow species concentration, velocity, and temperature in the combustor. Two laser beams of specific frequencies are optically wave-mixed to produce a single beam. When this beam is directed at specific molecules, they

respond by emitting a frequency shifted spectrum that is characteristic of the state they are in. This emitted light is spectroscopically analyzed to determine flow conditions.

The Dual-Laser Skin Friction Interferometer is a sensor that utilizes a

nonintrusive, double laserbeam oil viscosity balance technique. Two out-of-phase laser beams, reflected off an oil film surface, interfere constructively and destructively to generate light fringe spectra. The skin friction may be obtained by analyzing these spectra.

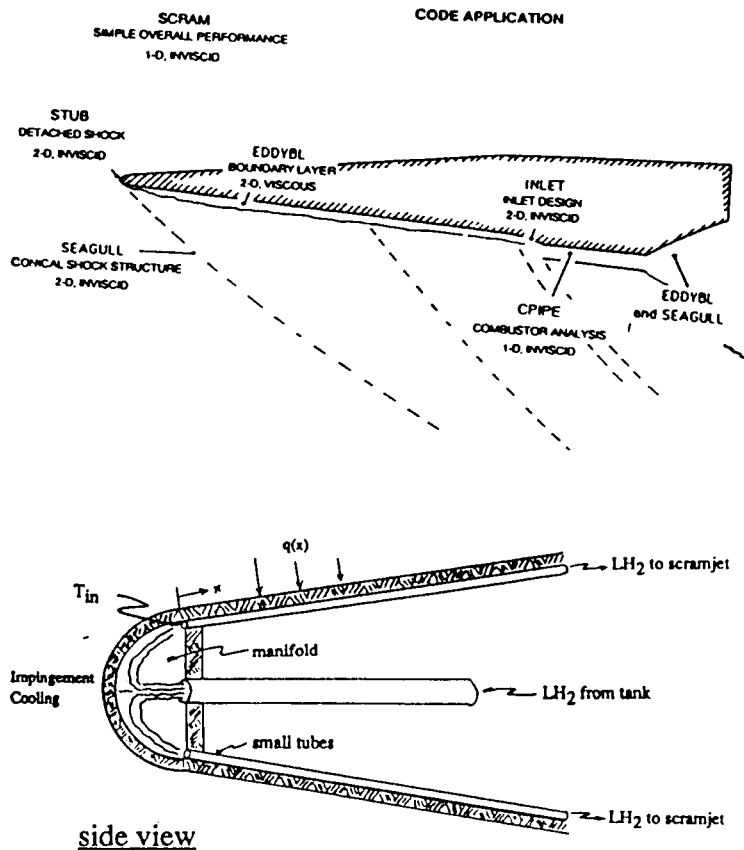


Figure 1 - Application of CFD Existing Computer Programs

side view

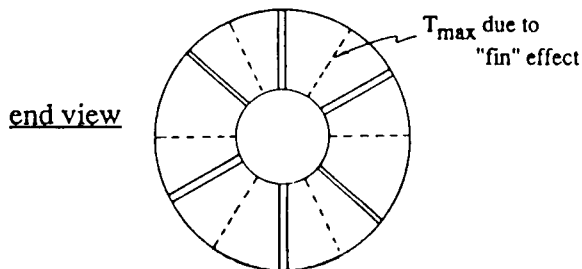


Figure 2 - Combined Impingement and Actively Cooled Panel Concept for the Drone Nose

- Maximum skin temperature is midway between coolant tubes
- Tubes welded to skin chosen to simplify fabrication

**AIR TRANSPORTATION SYSTEMS  
FOR THE CALIFORNIA CORRIDOR OF 2010**

**CALIFORNIA POLYTECHNIC STATE UNIVERSITY**

In 1986 NASA and USRA identified Cal Poly as one of seven "Centers of Aircraft Design Education," and accepted a proposal from Cal Poly to conduct a three-year study of the potential for Lighter-Than-Air (LTA), Vertical Take-Off and Landing (VTOL), and Short Take-Off and Landing (STOL) aircraft concepts for air transportation within the California corridor. The project emphasized configurations that are both innovative and unconventional in design for use in the 2010 time period.

The topic of LTA/VTOL/STOL aircraft was selected because it is consistent with the mission of the NASA Ames Research Center and is a broad topic that succeeding classes at Cal Poly can continue to iterate and refine to produce meaningful results for NASA.

The students have been encouraged to be innovative in order to identify key technology areas that can be developed into future research programs. The aircraft/air transportation systems currently being developed are:

- VTOL concepts
- warm cycle rotary wing aircraft
- joined wing, variable diameter tilt rotor

- STOL concepts
  - blown flap flying wing aircraft
  - upper surface blown cantilever wing aircraft
  - upper surface blown delta wing aircraft
- Air Transportation Systems - incorporates a VTOL aircraft which serves as a 'taxi' to and from a continuously moving platform
  - helipsoid airship/twin turbo-prop, quad lift fan cantilever wing aircraft
  - modified deltoid airship/advancing blade concept helicopter
  - magnetic levitation train/joined wing, variable diameter tilt rotor
- Air transportation needs in the California corridor include the following:
  - commuter air service
  - executive air transport
  - emergency medical service
  - public service
  - offshore oil support
  - freight/package service

Along with studying the technical issues normally involved in any aircraft design problem, the topics of safety, noise, public acceptance, and economic viability in commercial operations are also addressed.

## **STUDY ON THE PLANFORM EFFECTS ON HIGH SPEED VEHICLES**

### **CALIFORNIA STATE POLYTECHNIC UNIVERSITY, POMONA**

An overview of the high speed civil transport concept is presented. With information supplied by NASA concerning Boeing and McDonnell Douglas Aircraft studies, a request for a high speed aircraft design with a common mission profile was developed for use as a base for a planform effects study. The four configurations investigated include the blended-wing body, joined wing, oblique wing, and waverider concepts. These configurations were developed into complete aircraft designs. These designs were then used to study the planform effects along these four configurations.

In 1986 NASA Langley Research Center asked Boeing and Douglas Aircraft companies to assess the technological and economic feasibility of a high speed civilian aircraft. Two years later, both companies came to the conclusion that the international market justifies the need for a high speed transport. The speeds considered by the two companies were in the Mach 2.2 to Mach 5 range. Because the effective flight time for a transcontinental range does not decrease much above Mach 6, the California State Polytechnic University, Pomona study was limited to this range.

As a result of the studies done by Boeing and Douglas, California State Polytechnic University, Pomona (Cal Poly, Pomona) chose to evaluate the planform effects on high speed aircraft in the Mach 3 to 6 range. In order to accomplish this, twenty students were put into groups of five with the task of designing a high speed civil transport, given the same request for proposal and mission profile. After all of the configurations

are completed at the conceptual level, trade studies between the configurations will be made to determine the planform effects.

The class consisted of three two unit courses for a total of six quarter units. The first quarter was dedicated to researching the problems associated with high speed aircraft design. There were two speakers from NASA Ames Research Center during this quarter.

The second quarter was to develop the request for proposal, mission profile, and four possible configurations. The request for proposal and the mission profile were based on the Boeing and Douglas studies. The four configurations to be investigated were chosen with the concurrence of George Kidwell at NASA Ames. The four configurations are:

1. The Blended - Wing - Body Concept
2. The Joined Wing Concept
3. The Oblique Wing Concept
4. The Waverider (Caret) Concept

Each group developed the initial sizing of the aircraft during this second quarter. The Joined Wing configuration team visited Dr. Julian Wolkovitch who pioneered the joined wing concept. The design team discussed the joined wing concept and their particular configuration with Dr. Wolkovitch.

The third quarter was dedicated to refining their final reports and developing presentation skills. During this quarter, Cal Poly, Pomona joined UCLA and Cal Poly, San Luis Obispo at NASA Ames-Dryden Flight Research Facility for a mid-year con-

ference. At this conference, the students presented their configuration studies and gave critiques. At the end of the year, the students present their configuration studies to invited guests from industry who critique their designs. The final aircraft designs consist of a final configuration with all aerodynamic, propulsion, and structural analysis complete.

Noise and air pollution, as well as cost and systems layouts are also included in the design analysis. When the final four configurations are completed, analysis and trade studies are performed in order to reach conclusions regarding the planform effects of high speed aircraft.

The request for proposal was developed by the students using the results of the Boeing and Douglas studies. The request for proposal is for a complete aircraft design that meets the following criteria.

1. The aircraft must have an overall range of 6500 nautical miles in order to travel transcontinental distances, e.g., Los Angeles to Tokyo.

2. The design cruise speed range shall be between Mach 3 and Mach 6.

3. The aircraft must be able to accommodate 250 passengers at 170 pounds each with an additional 30 pounds of baggage for each person. This is equivalent to 50,000 pounds of payload.

4. The aircraft must be able to operate from existing airports.

5. Turn-around time is to be minimized in order to decrease operating cost. Goal: 1 hour.

6. Noise pollution is to be minimized. Goal: 1 psf overpressure.

7. Consideration will be given to the effects on the ozone layer. The designs and operating conditions are to minimize ozone depletion.

8. The aircraft must be designed to meet all FAR requirements.

9. The aircraft must be designed with 1988 technology.

The mission profile that was developed for the configurations is shown in figure 1. It is a ferry mission with an overall range of 6500 nautical miles. Travel distance is associated with climb, cruise and descent legs. Although some current commercial mission profiles do not include climb and descent in the overall range calculations, the high speed civil transport is operating at such a high altitude that the climb and descent legs have significant ranges.

The four configurations were chosen because of their unique designs. The blended-wing-body is the configuration with which most people are familiar. Similar high speed aircraft such as the SR-71 use this type of configuration and thus it represents current aircraft design. The joined wing was chosen for its structural stability and ability to minimize wing weight. The oblique wing configuration was chosen for its performance characteristics at both low and high speed ranges. Finally, the waverider configuration was chosen for its high speed performance characteristics. After these configurations were completed, trade studies were performed on the planform effects.

The four configurations were designed to meet the request for proposal in order to perform the planform effects study. Each of the configurations uses the same mission profile, uses the same type of fuel, and carries the same amount of payload.

With these four configurations, computer codes were used to make trade studies and conceptualize each design. With these designs, aircraft performance and operating costs can be compared to see the advantages and disadvantages of each design.

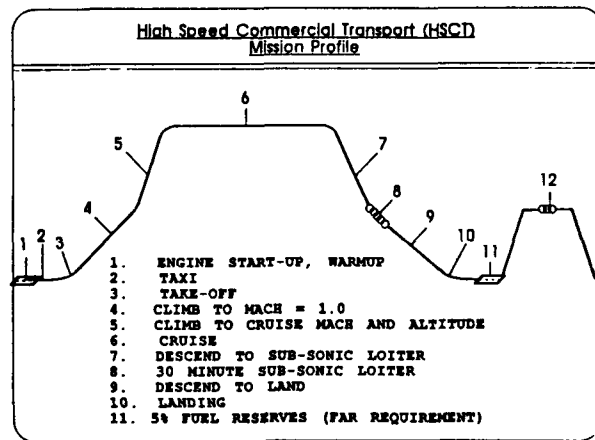


Figure 1

**A PRELIMINARY COMPONENT ANALYSIS  
OF A MACH 7 HYPERSONIC VEHICLE**

**CASE WESTERN RESERVE UNIVERSITY**

A preliminary study of a hypersonic cruise vehicle's basic components uses a comparison of time, fuel and cooling requirements in the Mach range of 5 to 10 to pick operating conditions. Mach 7 is determined, through analysis, to be the optimum cruise speed for an 8000 mile trip from the United States to Japan. The higher cruise speed allows a two-hour flight time and can make the entire business trip as short as one day long. An alti-

tude of one hundred thousand feet is chosen because the density of air in that range is low enough to provide sufficient lift for the vehicle and at the same time, have a low enough pressure so as to not produce an excessive heat flux at the cruise speeds. In addition, a sonic boom constraint, although not computed but referred to, seems to be sufficiently dissipated at this altitude.

## PRELIMINARY DESIGN OF TWO TRANSPACIFIC HIGH SPEED CIVIL TRANSPORTS

### THE UNIVERSITY OF KANSAS

Two high speed civil transport (HSCT) design concepts are presented. Both transports are designed for a 5500 nm. range with 300 passengers. The use of conventional hydrocarbon fuels is emphasized to reduce the amount of change required in current airport facilities. Advanced aluminums are used in the designs when possible to reduce material and production costs over more "exotic" materials. Methods to reduce the airport noise, community noise, and fly-over noise are incorporated into the designs. In addition, requirements set forth by the Federal Aviation Regulations (FAR's) have been addressed.

#### Mission Profile

The mission profile for both configurations is presented in Figure 1 with a brief description of each mission leg given in Table 1. The design mission represents a typical Los Angeles to Tokyo flight.

#### Mach 2.5 Transport

The first HSCT design concept is a Mach 2.5 joined-wing single fuselage transport. A side and top view of this configuration is shown in Figure 2. Table 2 contains some basic information about the airplane. High temperature aluminum is used in the primary structure to keep the production costs low.

The joined-wing is incorporated into the Mach 2.5 transport design for two reasons:

1. To reduce the wing structural weight.
2. Improve low speed trimmed lift capabilities.

Four variable cycle engines are located on the wing in pairs. A large

flap is located on the inboard portion of the wing for take-off and landing. The landing gear consists of a four post main gear, and a two wheel nose gear. The nose of the airplane droops 12.5° during landing to improve pilot visibility. Figure 3 shows the nose/crew station layout.

#### Mach 4.0 Transport

The second configuration is a Mach 4.0, twin fuselage, variable sweep wing transport. A side and top view of the airplane is shown in Figure 4. Some basic information about the airplane is contained in Table 3. The primary structure of the airplane is made of titanium. High temperature jet fuel (JP-7) is used to reduce the facility changes that would be required for a cryogenic fuel.

The Mach 4.0 transport incorporates a variable sweep wing to improve low speed lift capabilities. This helps reduce landing and take-off speeds, and increases the initial take-off climb rate.

Six turbojet engines are podded in pairs under the wing. To eliminate nose gear F.O.D. problems, an unusual landing gear arrangement was designed. The gear layout uses six main landing gear struts, with no nose gear. The six struts are located under the fuselages, as shown in Figure 4. All six struts are steerable, to allow for easy ground maneuvering and crosswind landings. Fluid is metered between the front and rear struts to assist in take-off rotation.

The twin fuselage sections are blended with the main wing. Twin passenger sections allow the fuselage fineness ratio to be minimized. The



flight deck is located in the left fuselage, with first class seats occupying the right fuselage nose section.

Operational considerations, such as turn-around time and required airport facilities, are addressed for both con-

figurations. The noise generated by both configurations (fly-over and airport/community) is compared with current and proposed FAR 36 requirements. An estimate of the production and operating costs for both configurations were addressed.

Table 1. - Mission Profile For A Transpacific Transport

- 1) Start-up, Taxi, Take-off
- 2) Climb to 10,000 ft at 250 kts
- 3) Accelerate to M=.95, Climb to 45,000 ft
- 4) Subsonic Cruise (M=.95, 150 nm)
- 5) Climb and Accelerate to Cruise Conditions
- 6) Cruise a) M=2.5, 65,000 ft  
b) M=4.0, 80,000 ft
- 7) Decelerate to M=.95, Descend to 45,000 ft
- 8) Subsonic Cruise (M=.95, 150 nm)
- 9) 30 minute Loiter
- 10) Descend to 1,500 ft (Start Reserve Mission)
- 11) Climb to 45,000 ft
- 12) Subsonic Cruise ( M=.95, 300 nm)
- 13) Descend to 1,500 ft
- 14) Landing, Taxi, Shutdown

Mission Range: 5,500 nm  
Reserve Range: 300 nm

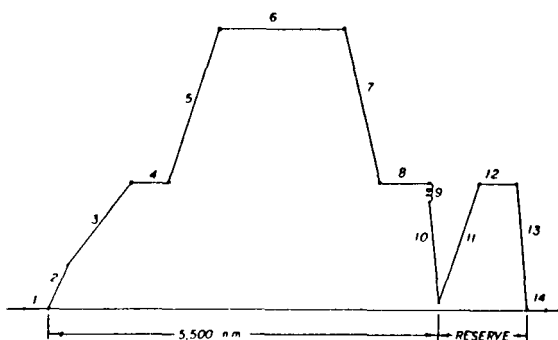


Figure 1. - Transpacific Transport Mission Profile

Table 2 - Mach 2.5 Transport Characteristic Data

	Front Wing	Rear Wing	Vertical Tail
Area (ft <sup>2</sup> )	7226	1400	829
Span (ft)	118	86	23
Aspect Ratio	1.9	5.3	0.6
L.E. Sweep	70°	-70°	70°
	Fuselage		Overall
Length (ft)	300		308
Max Height (ft)	15.5		42
Max Width (ft)	14.0		118
-----			
T.O. Weight	618,000 lbs		
Empty Weight	278,000 lbs		
Fuel Weight	262,750 lbs		
Payload Weight	64,500 lbs		
T.O. Thrust	247,200 lbs		
Wing Loading	85 lb/ft <sup>2</sup>		
Thrust/Weight	0.40 lb/lb		

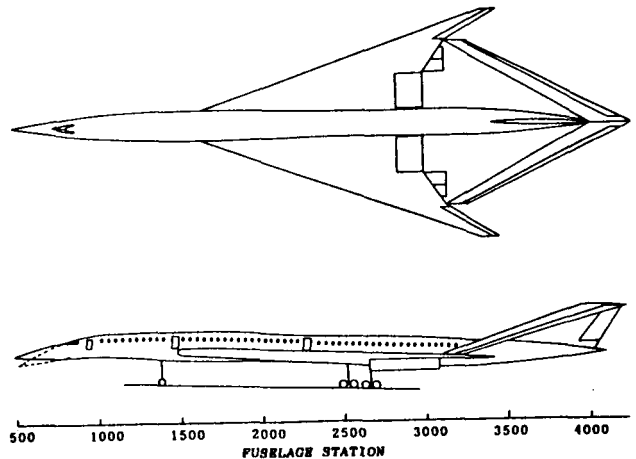


Figure 2 - Mach 2.5 Transport Drawing

Table 3 - Mach 4.0 Transport  
Characteristic Data

	Wing		Vertical
	Take-off	Cruise	Tails
Area (ft <sup>2</sup> )	12,940	12,920	2x257
Span (ft)	130	280	10.7
Aspect Ratio	1.3	6.0	0.4
L.E. Sweep	78°	13°	67°
	Fuselage	Overall	
Length (ft)	246	316	
Max Height (ft)	10.3	35	
Max Width (ft)	10.3	296	
-----			
T.O. Weight	1,155,800 lbs		
Empty Weight	531,460 lbs		
Fuel Weight	660,000 lbs		
Payload Weight	64,500 lbs		
T.O Thrust	520,000 lbs		
Wing Loading	89 lb/ft <sup>2</sup>		
Thrust/Weight	0.45 lb/lb		

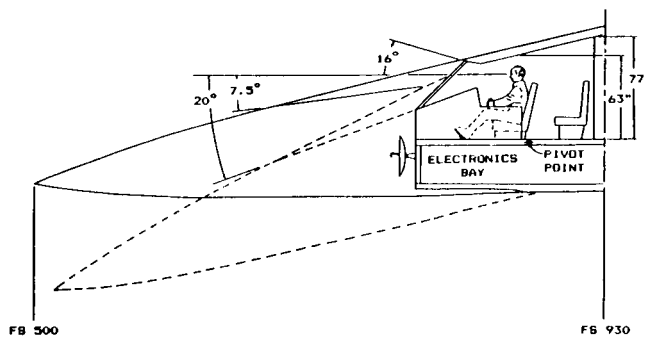


Figure 3 - Mach 2.5 Transport Nose/  
Crew Station Layout

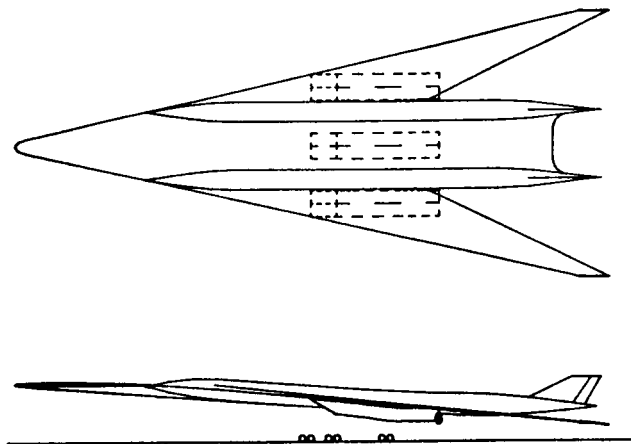


Figure 4 - Mach 4.0 Transport Drawing

## HIGH SPEED TRANSPACIFIC PASSENGER FLIGHT

### THE OHIO STATE UNIVERSITY

The Advanced Aeronautics Design program at The Ohio State University is an elective sequence of three courses taught to senior Aeronautical and Astronautical Engineering (AEE) students throughout their senior year. A one hour seminar on flight vehicle design is open to all AAE seniors in the Fall Quarter; it is followed by a four credit hour design laboratory in Winter and a three credit hour design laboratory in Spring.

The purpose of the seminar is to introduce the students to the design process, with guest speakers from NASA and industry sharing their design experiences with the class. As the seminar progresses, the design task for the year is defined, so the students can elect to follow the advanced program or to take the single required four credit design laboratory. Should they elect the advanced design sequence, they must commit to take the two design courses.

During the Winter Quarter, the students meet formally Tuesday and Thursday afternoons. Design teams are formed, group leaders chosen and specialities assigned -- e.g., aerodynamics, propulsion, performance analysis, stability and control, etc. These design teams are free to organize as they see fit. Each class meeting typically consists of a formal lecture for one hour; then the teams break up to follow individual assignments -- in the library or at the computer, for example.

The design task selected for the 1987-1988 year was based on high speed flight to serve travelers around the Pacific rim. A controversy has developed within the commercial aircraft industry as to the appropriate

speed regime for such travel. Boeing Airplane Company proposed this travel be made at Mach numbers less than three, while McDonnell-Douglas favors flight above Mach five. Since the design of a passenger transport to operate at these speeds and range is a real world problem of national as well as technical interest, the requirements for the students were guided by this design task. The specifications to be satisfied are listed below:

#### Design Specifications for Commercial Transport

Passenger Capacity	250 +
Range	6500 m
Takeoff & Landing	
Distance	<15000 ft
Cruise Speed	3<M<5

These broad requirements allowed the students much latitude to develop their design concepts. The four design teams formed during the Winter Quarter consisted of two teams examining the  $M = 3$  regime and two teams working on  $M = 5$  aircraft.

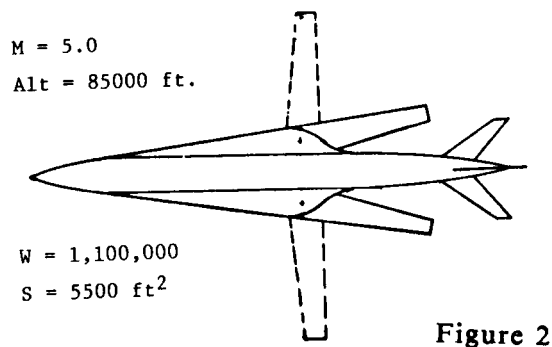
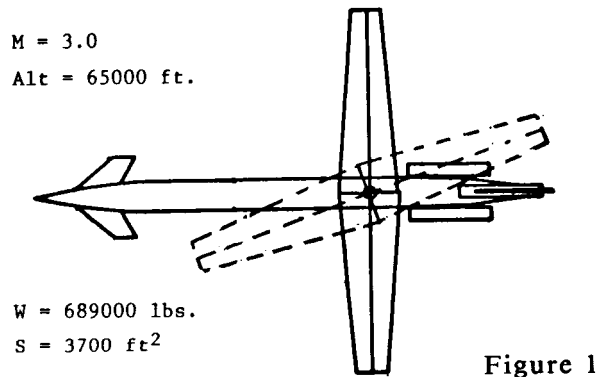
The designs progressed during the Winter Quarter, assisted by lectures from NASA and industry personnel. The General Electric Engine Group was especially helpful with performance estimates for supersonic engines operating from  $M = 2$  to  $M = 6$ , with the capability of burning JP, methane, and hydrogen fuel. Computer codes used during the design process included weight estimating and engine performance codes provided by NASA Lewis Research Center and the Harris Wave Drag code obtained from NASA Langley Research Center. The latter program was especially helpful in minimizing

the supersonic wave drag; this was the first time this complex code had ever been used at Ohio State University, now possible because of the extended design period of this program.

The aircraft designs developed by the class are illustrated below, showing four different configurations: two variable geometry aircraft (one for  $M = 3$  (Figure 1) and the other,  $M = 5$ ) (Figure 2) and two fixed wing, blended fuselage designs (Figures 3 and 4). Similarly, the engine types varied from supersonic flow compressor engines, turbo-ramjet, turbo by-pass and one aircraft with a

conventional jet plus subsonic combustion ramjet propulsion system.

These designs were refined during the Spring Quarter, with additional problems examined, including costs, environmental impact and operational considerations. Subsonic models were built to test the aerodynamic performance of the configurations in the Ohio State University 3' x 5' Subsonic Wind Tunnel, and two models were built to measure the heating rates of the hypersonic designs in the Ohio State University 12" diameter Hypersonic Wind Tunnel.



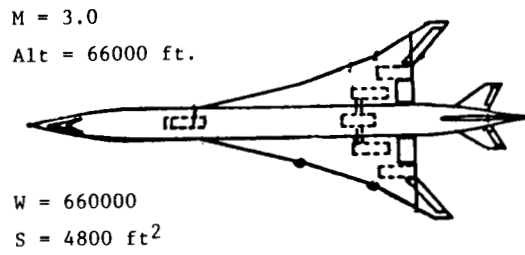


Figure 3

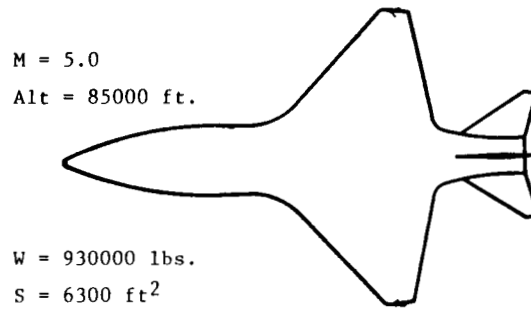


Figure 4

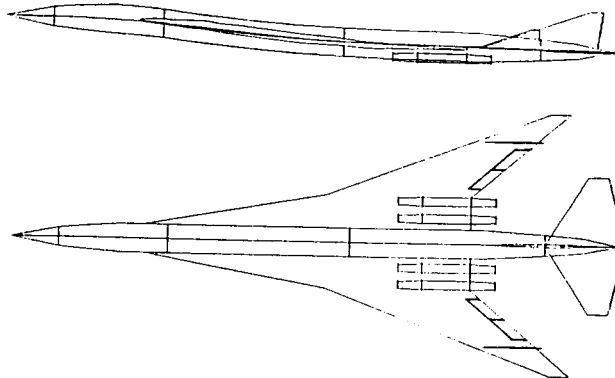
**AIRCRAFT INTEGRATED DESIGN AND ANALYSIS - A CLASSROOM EXPERIENCE**

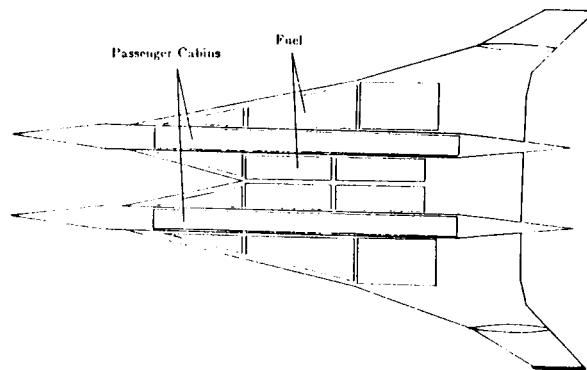
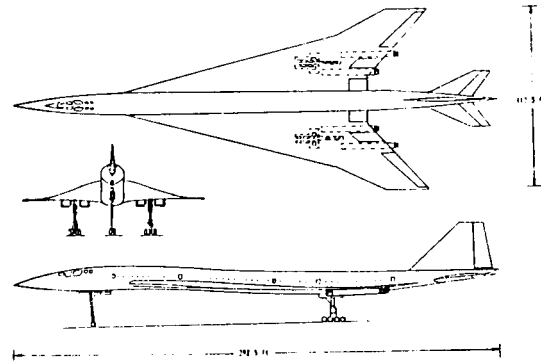
**PURDUE UNIVERSITY**

The USRA sponsored design course is the capstone course required of all senior undergraduates in the School of Aeronautics and Astronautics at Purdue University. During the past year the first steps of a long evolutionary process were taken to change the content and expectations of this course. These changes are the result of the availability of advanced computational capabilities and sophisticated electronic media availability at Purdue. Both the long range objectives and this year's experience using the High Speed Commercial Transport (HSCT) design, the AIAA Long Duration Aircraft design, and an RPV design proposal as project objectives are described. The central goal of these efforts is to provide a user-friendly, computer software-based environment to supplement traditional design course methodology. The Purdue University Computer Center, the Engineering Computer Network and stand-alone personal computers are being used for this development. This year's accomplishments center primarily on

aerodynamics software obtained from NASA/Langley and its integration into the classroom. Word processor capability for oral and written work and computer graphics were also blended into the course. A total of 10 HSCT designs were generated, ranging from twin-fuselage aircraft, forward swept wing aircraft to the more traditional delta and double-delta wing aircraft. Four Long Duration Aircraft designs were also submitted, together with one RPV design tailored for photograph surveillance.

Supporting these activities were three video satellite lectures beamed from NASA/Langely to Purdue. These lectures covered diverse areas such as an overview of HSCT design, supersonic aircraft stability and control, and optimization of aircraft performance. Plans for next year's effort will be reviewed, including dedicated computer work station utilization, remote satellite lectures, and university/industrial cooperative efforts.





**THE APOLLO LIGHTCRAFT PROJECT**  
**RENSSELAER POLYTECHNIC INSTITUTE**

The detailed design of a small beam-powered transatmospheric vehicle, "The Apollo Lightcraft," was selected as the project for the design course. The vehicle has a lift-off gross weight of about six (6) metric tons and the capability to transport 500 kg of payload (five people plus spacesuits) to low Earth orbit. Beam power was limited to 10 gigawatts.

The principal goal of this project is to reduce the low Earth orbit payload delivery cost by at least three orders of magnitude below the space shuttle orbiter -- in the post 2020 era.

The completely reusable, single-stage-to-orbit, shuttlecraft will take off and land vertically, and have a reentry heat shield integrated with its lower surface much like the Apollo command module. At the appropriate points along the launch trajectory, the combined cycle propulsion system will transition through three or four air breathing modes, and finally a pure rocket mode for orbital insertion.

As with any revolutionary flight vehicle, engine development must proceed first. Hence, the objective for the Spring semester propulsion course was to design and perform a detailed theoretical analysis on an advanced combined-cycle engine suitable for the Apollo Lightcraft. The analysis indicated that three air breathing cycles will be adequate for the mission, and that the ramjet cycle is unnecessary.

The theoretical analysis of this combined cycle engine is now complete and the acceleration performance along representative orbital trajectories has been simulated. Average vehicle acceleration is approxi-

mately 4 to 5 Earth g's. Transition between engine modes occurs at Mach 3, Mach 11, and Mach >25.

Beam power can be reduced to 2.5 billion watts without sacrificing vehicle performance. The liquid hydrogen propellant requirement is typically 300 kg, or roughly five percent of the vehicle lift-off weight for delivery to a one hundred nautical mile orbit. The total beam energy requirement is 520 GW-sec. for this boost mission.

The propellant cost at current bulk liquid hydrogen rates is \$975. The energy cost is \$2,455, assuming present wholesale hydroelectric power rates. The total round trip cost is only \$3,430, or \$686 per person. This represents a payload delivery cost of \$3.11 per pound which is a factor of one thousand below the Space Transportation System.

Most recently, the Apollo Lightcraft concept was subjected to a more detailed investigation during the 1987 Fall semester Trans-atmospheric Vehicle Design Course. The class was divided into smaller design groups, each composed of 3 to 5 students, to study in a more concentrated manner:

- a. Aerodynamics of reentry
- b. Propulsion developments
- c. Structural dynamics
- d. Thermal analysis
- e. Flight control systems
- f. Optimal trajectory analysis
- g. Flight crew systems
- h. Power beaming system architecture.

The Apollo Lightcraft concept survived this critical examination in-



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The Apollo Lightcraft concept survived this critical examination in-

tact, and no technical showstoppers were uncovered.

A new teaching assistant will spend the entire summer at NASA Lewis

Research Center gathering technical reports and computer programs of relevance to Trans-atmospheric Design in preparation for the next 1988 Fall Design Course.

**AUTONOMOUS SPACE PROCESSOR FOR ORBITAL DEBRIS REMOVAL  
AND  
FLAME AUGMENTATION ADDITIVES IN SCRAMJETS FOR THE NASP**

**UNIVERSITY OF ARIZONA**

Approximately eighty percent of the effort associated with the NASA/USRA sponsored project at the University of Arizona was devoted to the development of a novel engineering concept for in situ processing of orbital debris utilizing the various resources available in LEO; the other twenty percent was devoted to innovative additives for anchoring of supersonic combustion zones that find direct use in the National Aerospace Plane (NASP) that is expected to use SCRAMJETS. The seriousness of the orbital debris problem is described and proposed solutions from literature are outlined. The engineering design is briefly mentioned, emphasizing the positive aspects of the space environment that should be used in an economical approach. The aspects of microgravity, vacuum, and abundant solar energy are mentioned. A quantitative computer animation is developed to fix ideas with three specific "dead" spacecraft identified for an initial cleanup mission. The solar processor remote arm manipulators, and gradual processing of orbiting space junk are described. Construction of the necessary hardware is decreased and the operations and actual processing of simulated debris parts (aluminum, for now) are demonstrated. For the NASP task, construction of a new design for measuring the radiation from the key free radicals (as enhanced by the additives) is described. Immediate (1988) and long-range (through 1992) plans are shown to clearly indicate the full engineering design strategy in the light of the national space program thrusts.

Orbital debris is a growing hazard affecting our space program efforts.

Human activities in space have generated significant quantities of debris that are posing a major threat to continued explorations. Many of these problems have been lucidly described by Kessler (1, 2). Although it is not within the scope of this USRA effort to make a thorough study of the hazard aspect of this debris, it is worthwhile to recall that a direct collision is only one aspect of the hazard. In fact, such a hit did indeed occur in the Columbia flight (3) causing extensive window damage. The energy transferred from these hypervelocity impacts has the ability to vaporize even metals. Occasional reentry of some of the larger junk pieces have caused anxiety to the people on Earth. The Skylab crash in Australia and the more recent Russian craft incident provide two such examples. The United States Space Command has made a good compilation of these junk pieces in orbit and several large pieces are evident. Even astronomers, who argued for telescopes in LEO in search of the pristine sky, away from the polluted terrestrial sites, have been painfully surprised to see their sky view crisscrossed by orbital junk that ruined their heavenly search (4). If these past encounters with junk are considered bad, the future looks much worse. Systematic projections have shown that by 1990, the probability of collisions is almost double what it is today. Many of the more recent activities in LEO are not of the type to alleviate this problem either.

Taking a more constructive point of view, a concept for the utilization of the resources in LEO to clear the junk was presented by Ramohalli in 1986 (5). The idea was to use solar energy to provide power to process usable junk materials while deorbiting the

larger, non-useful pieces. Recalling that many of the "junk" pieces are actually good high-tech materials, the concept attempted to use some of the more "unspoiled" materials to add to the solar collectors, for example. Finally, mostly limited by the propulsion requirements, this autonomous spacecraft (that has grown to a good size) would self-process to a compact mass, to be retrieved by the shuttle; or reenter the Earth's atmosphere to burn up, or splashdown harmlessly. A quantitative computer animation showed the feasibility (6). One point of some importance must be made here. Some have argued that these larger pieces are not really the problem, but the innumerable smaller ones are. Future spacecraft can always avoid the larger junk, but the smaller ones are the more unpredictable, untrackable ones. Hence, the argument goes, we are solving the wrong problem. We disagree. It is certainly true that the large, well-characterized orbiting junk seems to pose a smaller threat to our space operations at this time. But left alone, even a single collision can multiply the numbers enormously; then, we will have lost the earlier opportunity to clear a large compact mass. The recent operations in space have been shown to create an incredibly large number of small particles with a single collision.

This analysis presents the first step in a well planned engineering design activity that will eventually lead to the development of technologies to clear orbital junk. At the time of this report the plan has evolved, the first prototype has been designed, the first hardware has been built and the feasibility of processing with solar energy has been demonstrated. In the NASP contribution, we have proven

the design of an apparatus for measuring the enhanced radiation associated with the flame stability enhancement proved earlier (7). This paper ends with a summary of future activities.

The authors are grateful to Mr. Jack Sevier, Carol Hopf and the entire USRA staff for support of this engineering design project. The participating students are: Donald Barnett, Bryan Cindrich, Steve DiVarco, Catherine Dodd, Velda Dykehouse, Reid Greenberg, Jim Matson, Ruzila Mohdkhirhadi, James Poon, Xenophon Xenophontos (Spring 1988), and R. Scott Reid (Summer 1988); David Campbell's summer work is funded directly by USRA at JPL.

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**THE UNMANNED MULTIPLE EXPLORATORY PROBE SYSTEM (MEPS)  
FOR MARS OBSERVATION**

**AUBURN UNIVERSITY**

The unmanned Multiple Exploratory Probe System (MEPS) is designed to test various systems and mission techniques under consideration for future manned Mars missions. The elements of the modular vehicle are transported to Low Earth Orbit by Heavy Lift Launch Vehicles or the Space Shuttle with the vehicle being assembled near the Space Station. Orbital Transfer Vehicles move the MEPS to the ecliptic plane to prepare for the trans-Mars insertion maneuver.

The propulsion requirements are achieved by two stages. The first stage provides most of the thrust necessary to accelerate the vehicle to velocities corresponding to the desired Earth-Mars transfer orbit; this stage is jettisoned in an elliptic recovery orbit about the Earth. The second stage provides the remainder of the interplanetary transfer momentum change, is used to place the MEPS into an elliptic orbit about Mars, and aids in circularizing the observation orbit.

Another technology to be tested by the MEPS is aero-braking at Mars.

Although the Earth's atmosphere can be used to prove technology, future Mars missions using this process will require detailed knowledge of aero-braking in the Martian atmosphere.

Surface and atmospheric samples are collected by two rover systems, one deployed at the equator and the other at the North Pole. The polar lander is separated from MEPS during the Earth-Mars transfer. This will allow mid-curve corrections to be made to reduce the fuel requirements due to large inclination changes. The equatorial lander is deployed from the vehicle after orbit circularization is complete. Onboard surface laboratories perform biologic and organic tests and transmit the data to Earth.

An additional system of the MEPS is a satellite which is deployed into an observation orbit above the main vehicle. Information concerning the weather, surface formations, atmospheric density and temperature, and the surrounding space environment is obtained and relayed to Earth.

**THE SPACE STATION INTEGRATED  
REFUSE MANAGEMENT SYSTEM**

**UNIVERSITY OF CENTRAL FLORIDA**

The design and development of an Integrated Refuse Management System for the proposed International Space Station was performed by the University of Central Florida through cooperation with Kennedy Space Center. The primary goal of the yearlong study was to make use of any existing potential energy or material properties that refuse may possess. The secondary goal was based on the complete removal or disposal of those products that could not, in any way, benefit astronaut needs aboard the Space Station. The National Aeronautics and Space Administration's design of a continuous living and experimental habitat in space has spawned the need for a highly efficient and effective refuse management system capable of managing most of the forty thousand pounds produced annually. To satisfy this need, the following four integrable systems have been researched and developed:

1. Collection and Transfer.
2. Recycle and Reuse.
3. Advanced Disposal.
4. Propulsion Assist in Disposal.

For the purposes of this study, refuse is defined as all materials requiring disposal and includes both biologically active and inactive materials. It does not include metabolic/bodily wastes.

The design of a Space Station subsystem capable of collecting and transporting refuse from its generation site to its disposal and/or recycling site was accomplished. Refuse canister transport, receptacle designs, storage systems, and power supply were among the topics researched. Materials research warranted the use of high density polyethylene bags and cylin-

drical polypropylene canisters for refuse containment. A "bank shuttle" network, similar to those used in commercial bank applications, was recommended for canister transport exterior to a Space Station module or node. A select storage design consists of an exterior rack unit to house excess refuse generated from any of the proposed multi-disposal site arrangements. Size reduction was determined to be most effective with the use of a compaction technique capable of simultaneously removing nearly all liquids and gases while packaging takes place. System decontamination was researched in detail. General sanitization, airborne, and surface contaminant control were addressed. A combination of room arrangement, microbiological filtration, and application of germicidal vapors and gases was employed for an optimum solution. Focus was also placed on inventory control which incorporated the use of both color coding and bar coding to maximize simplicity and automation, respectively.

Several methods of recycling or reusing refuse in the space environment were researched. The optimal solution was determined to be the method of pyrolysis. Pyrolysis is described as "the destructive distillation of a carbonaceous material in the presence of heat and the absence of oxygen." The objective of producing a technically self-supporting recycle/reuse system led to the design of the Pyrolysis Processing Facility. The facility is comprised of 1) refuse size reduction, 2) pyrolysis reactor design, and 3) power generation. An optimal solution for the design consists of a counter-rotating, self-cleaning shredder coincident with a cyclonic entrained-flow pyrolysis reactor, and a

hybrid power generating system. The combination of an electrostatic parametric generator coupled with a heat pipe Rankine cycle supply power to the shredder and reactor was proposed. Extensive research has indicated all components of the Pyrolysis Processing Facility show great promise for space applications.

The objective of removing refuse from the Space Station environment, subsequent to recycling, was fulfilled with the design of a jettison vehicle. Design goals included safely containing refuse while also insuring prompt destruction of the vehicle and its contents upon atmospheric reentry. The vehicle to undertake such a mission is a rigid, aluminum alloy cylinder which will be launched via an expendable rocket. The vehicle will be assembled and mated with its propulsion unit on Earth. It will then be placed in low Earth orbit, be retrieved by an orbital maneuvering vehicle, and placed into its desired location on the Space Station: dimensions include a 4.5 feet diameter with a 3.5 feet length. The interior features pigeonhole storage racks that will accommodate six canisters of compacted refuse. Studies of worst

case scenarios have indicated the need for a maximum of ten canisters annually. In addition to vehicle design, debris casualty risks and the environmental effects associated with atmospheric reentry were investigated.

A number of jettison vehicle launch scenarios were analyzed. Selection of a proper disposal site and the development of a system to propel the vehicle to that site were completed. Reentry into the Earth's atmosphere for the purpose of refuse incineration was determined to be the most attractive solution. Interfacing a Morton Thiokol "Star 17" expendable rocket to the jettison vehicle will provide the propulsion/disposal system. The Titan 3 Commercial rocket will transport the system to the orbiting Space Station. Once filled, an orbital maneuvering vehicle will remove the assembly out of close proximity of the Space Station, initiate spin with proper attitude, and return to the Space Station. The launch of the "Star 17" rocket, which incorporates orbital mechanics and guidance controls, will deliver the refuse payload into the upper atmosphere completing destruction within one low Earth orbit.

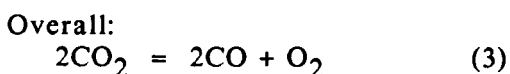
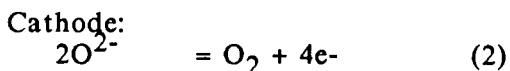
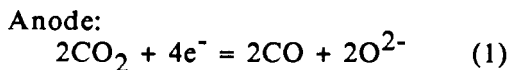
## A ZIRCONIA SOLID ELECTROLYTE CELL FOR THE MARTIAN ATMOSPHERE

### CLEMSON UNIVERSITY

The Martian atmosphere, because of low pressure and the lack of oxygen, is incapable of sustaining human life. The pressure is only 0.006 to 0.009 atmospheres and the composition is 95% CO<sub>2</sub>. The oxygen content is only 0.13 mol% O<sub>2</sub>. If it were possible to extract the O<sub>2</sub> from the CO<sub>2</sub>, and to increase the pressure of the O<sub>2</sub>, the problems of sustaining human life would disappear.

Of the several options available, the most attractive method of decomposing CO<sub>2</sub> appears to be a process employing a solid ceramic electrolyte. Oxygen at low pressures would be electrolytically pumped across the electrolyte membrane to a higher pressure.

This process is accomplished by applying a voltage between electrodes deposited on each side of the membrane and causing the oxygen at the surface of the negative electrode to become oxygen anions (O<sup>2-</sup>). These anions then migrate toward the positive electrode. When the oxygen anions reach the positive electrode, they lose the two extra electrons and become neutral atoms. Two neutral atoms then pair up to become an oxygen molecule. The overall reactions for this cell may be summarized as follows:



The rate of CO<sub>2</sub> conversion for a ZrO<sub>2</sub> membrane is optimized by de-

veloping membranes with a maximum surface area and minimum thickness. The membranes must also be free of pores and uniform in thickness. Further, it is important that the membranes be arranged in a manner which optimizes the amount of surface area for a given volume of space. The task for each design team was to design a membrane configuration and propose a method for forming the membrane system.

The class was divided into five design teams. Each team proposed a configuration and a means of fabrication. The fabrication methods used ranged from extrusion to sol-gel processing. All teams used ZrO<sub>2</sub> stabilized with 8 mol% Y<sub>2</sub>O<sub>3</sub> after first establishing that this material formed the most satisfactory solid electrolyte.

Slip casting was chosen by two of the teams. This method consists of pouring a water suspension of stabilized zirconia powders into a porous mold (usually made from plaster). The mold absorbs some of the water causing the portion of the slip next to the mold to become semi-rigid. The remaining slip is poured out and the ceramic is allowed to dry further. Mold design became quite critical with this project. The design these teams proposed to fabricate consisted of an array of adjoining tall pyramids arranged in a plate. The pyramid array was subsequently placed in an enclosure.

Extrusion of long shapes was tried by another team. This group was unable to actually construct a device because of the lack of equipment for doing so. They did, however, establish the necessary parameters for extruding a nonplastic material such as zirconia.



The shape proposed by this team consisted of a long hexagonal or square tube, divided into many small chambers (as in a honeycomb). Triangular or square chambers were found to be best since each wall could participate as an active membrane in decomposing  $\text{CO}_2$ .

Tape casting flat plates was also attempted. These plates were "glued" together into the final configuration. Tape casting is a process which uses a thick liquid suspension. This suspension is spread on a flat surface and distributed across this surface with a "doctor blade" spreader. Binding agents are added to this suspension to give the suspension added strength when it dries. This group demonstrated that bonding these plates together with a ceramic "glue" may hold potential.

A possible problem with forming these plates may be the porosity of

the fired plates. The porosity is difficult to eliminate unless extraordinary steps are taken. A possible solution to this is to make the plates using a sol-gel formation technique. Monolithic plates of  $\text{ZrO}_2$  with no pores might possibly be formed if the particle size is extremely small or if an oxide gel is first formed. This process requires very controlled drying conditions. This group formed several gels and established the problem areas that will be present if this fabrication method is adopted.

Since very little literature is available on manufacturing solid electrolytes and relatively little is known about the fabrication and firing of stabilized zirconia bodies, much of the time was spent by the class performing research in the laboratory trying to establish satisfactory manufacturing parameters. The groundwork laid this year will be used next year to further advance this project.

**THE DEVELOPMENT OF  
A CISLUNAR SPACE INFRASTRUCTURE**

**UNIVERSITY OF COLORADO**

Objective

The primary objective of the University of Colorado Advanced Mission Design Program is to define the evolution of a near-Earth space infrastructure that includes a permanently manned, self-sustaining base on the lunar surface and to provide a preliminary design of the key components and/or systems of that infrastructure. The implementation of this design effort should be carried out with the idea that the base itself is not an end but an important step in a larger plan to expand man's capabilities in science and technology. For instance, the presence of a cislunar space infrastructure would greatly facilitate the staging of future planetary missions. This paper will provide a sound rationale and a detailed scenario in support of the infrastructure design.

Methodology

The evolution of the proposed infrastructure is illustrated in Figure 1. This preliminary scenario timeline consists of specific mission target dates and is subject to change depending on both funding and available technologies. The timeline is also subject to an initial delay depending upon the date on which the program is actually initiated. The infrastructure development program is checked periodically by means of breakpoints. These breakpoints are used to evaluate the progress and success of the program. Additionally, they will establish a series of driving questions that will have a bearing on the future of the program.

The methods by which the goals of the program will be achieved are

dependent upon the technologies available throughout the system's developmental period. Some of the more important technologies which will drive the system are:

Life Sciences and Life Support  
Transportation Systems  
Space Construction Techniques  
Automation/Robotics  
Power Generation  
Space Suits

Four assumptions were made in establishing the scenario: 1) a reliable transportation system exists between Earth and low Earth orbit (LEO); 2) there is a functional LEO space station; 3) a heavy lift launch vehicle exists; and 4) on-orbit construction capabilities exist.

Rationale

The University of Colorado Advanced Mission Design Program proposes a near-Earth space infrastructure consisting of a lunar base, a manned LI space station, and a fleet of associated transportation vehicles. A near-Earth space infrastructure has the potential to bring great economic and scientific returns, as well as less tangible benefits such as increased nationalism and greater stature in world politics. It is expected that a near-Earth space infrastructure would be beneficial to the future of the United States by providing a degree of economic return similar to that attained by the Apollo program. A United States presence in Earth orbits would advance research in areas such as Earth studies, material processing and variable gravity experiments, as well as astronomy and solar studies. Although the direct benefits of a space initiative are difficult to assess,

Unmanned vehicles consist of two basic types. Small orbital maneuvering vehicles will shuttle between the L1 station and GEO to retrieve and redeploy satellites which need servicing. A nuclear electric ion propulsion vehicle will transfer cargo and propellants from LEO to L1 or to lunar orbit. These electric propulsive orbital transfer vehicles would take approximately one year for a round trip.

#### Infrastructure Development

The lunar base configuration was designed considering maximum crew safety, ease of construction, modularity, adaptability to different missions, durability, and cost. The driving factors in determining the size of the base were required habitation area, experimental and research activities, and the volume of CELSS.

The lunar base design was developed in logical and incremental advances from the initialization of remote sensing site selection devices to the ultimate goal of a self-sustaining lunar base. Lunar surface development occurs in four distinct phases. The development of the L1 station parallels the progress of the lunar base.

Phase I, Remote Sensing and Site Selection, will begin in 1994. This phase will encompass a variety of remote sensing exploration missions, as well as one manned mission to two sites. This phase will determine the location of the initial outpost.

Phase II, Outpost Development, consists of establishing a man-tended outpost on the Moon for 10 to 14 day missions. The purpose of these missions will be to perform science ex-

periments and to research lunar processing capabilities. Concurrent with the lunar development is a modest space station located at L1 to serve as a transportation and staging node.

The lunar base will be powered by a 100 kW solar array with fuel cells as backup. Other hardware staged to this point include a habitation module, multi-purpose lunar surface vehicles and a sandbagging device to bag regolith for shielding purposes. Radiation protection will be provided by a regolith "tent" supported by a truss structure (Figure 2).

The success of Phase II activities will provide the means to assess the practicality of continued lunar development. Key considerations are: lunar oxygen production rates, CELSS efficiency, and the demonstrated potential for further scientific and economic gains.

Phase III, Permanently Manned Base, begins the development of large-scale processing activities which should lead to lower space program operational costs by providing water, propellants, and other materials from lunar resources. A nuclear power plant will provide the energy requirements for this processing. Two more habitation modules, increased CELSS capability and closure, and a larger crew on the lunar surface are required to support the expanded lunar operations. A far side radio observatory will be constructed during this phase.

Phase IV, Self-sustaining Base, results from the continued success of Phase III operations. Integrating more systems, new processes and better technologies (such as CELSS) into the

the United States must plan ahead as an investment in the technological future.

### High Earth Orbit Activities

A high Earth orbit platform is needed to support the in-orbit activities and services which are necessary to promote safe and efficient space exploration. For safety, flexibility and cost efficiency, a platform will be placed in a halo orbit about Libration Point 1. The L1 station will act as a staging point between LEO and the Moon, thus providing a more cost efficient transportation infrastructure. Activities such as cargo and fuel storage, refueling, satellite servicing, geosynchronous orbit cleanup, and a transportation control center provide justification for an L1 station. This location is ideal for staging future planetary missions.

### Lunar Base Activities

The moon will provide a focal point for advanced astronomy research, for planetary studies, and for possible economic returns from processing of lunar resources.

Advancements in sciences will be enabled through the use of lunar-based observatories to study the universe, while a better understanding of the solar system will be aided through selenological studies conducted on the Moon. Lunar-based astronomy is not shielded by an atmosphere. All wavelengths of electromagnetic radiation (x-ray, infrared, gamma ray, and ultraviolet) will reach the lunar surface. Additionally, radio-interference from Earth can be shielded by the Moon, making radio telescopes located

on the far side more effective.

To reduce the dependence on Earth-based supplies, water for life support and lunar-derived oxygen and fuels for propulsion purposes are chosen as key lunar products. Carbothermal reduction is a candidate process to extract oxygen and water. Mining and materials processing may provide the lunar base with raw materials to be used in future expansions of lunar activities. A Controlled Ecological Life Support System (CELSS) will fulfill the requirements for long term human space missions by reducing the total mass required to be transported from Earth. Additionally, the lunar CELSS will provide the "seed" to start up other life support systems, such as those on an advanced L1 station or a manned Mars mission.

### Transportation System

The transportation system will support the wide range of transportation needs of the space program. Modular manned orbital transfer vehicles will utilize aerobraking technology and cryogenic liquid oxygen/hydrogen-based propulsion to transfer crews and time-sensitive cargo through cislunar space. Primary destinations include: LEO, lunar orbit, geosynchronous Earth orbit (GEO) and L1. Additional manned transportation will be provided by two types of lunar landers. One lander is used to bring modest equipment and small crews from L1 to the lunar surface. Cargo and larger crew transportation requirements will be satisfied by a larger lander which travels between lunar orbit and lunar surface.

base promote greater self-reliance. These benefits will pertain to the entire infrastructure. An advanced L1 station becomes justified as lunar activities and capabilities are expanded. The advanced L1 station will rotate to generate artificial gravity and will serve as the staging site for manned Mars missions.

Thus, the University of Colorado Advanced Mission design Program has defined the evolution of a cislunar space infrastructure and designed many of the infrastructure components. Technological advances, scientific gains, economic returns, greater political stature and national pride justify the development of such an infrastructure.

Specific activities will take place through the use of an L1 space station, lunar habitation, and an exten-

sive transportation system. The development and construction of the infrastructure are phased such that specific breakpoint criteria must be satisfied before the next phase can begin. By advancing logically and methodically, a solid foundation is established upon which a credible and successful program may be built.

Development of the infrastructure begins when the program is accepted around 1990 and research and design of technologies, hardware, and programmatic considerations are accelerated. This scenario continues through 2021, when the lunar base becomes self-sufficient. At this point, the open-ended nature of this design, the cislunar space infrastructure, becomes an important stepping stone for more aggressive projects, such as a manned Mars mission.

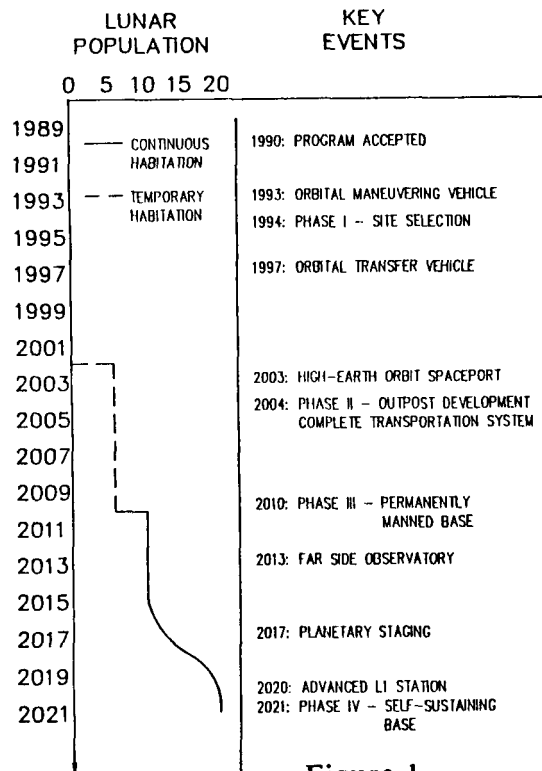


Figure 1

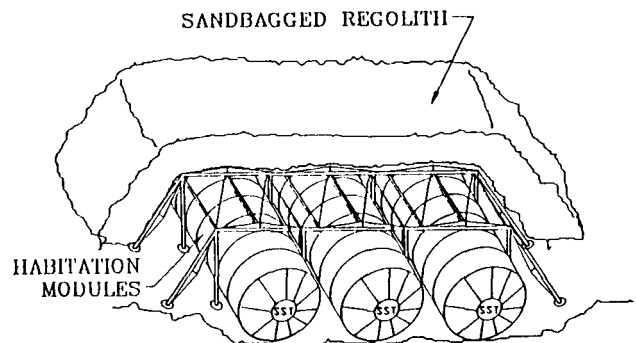


Figure 2

## **BIOREGENERATIVE SYSTEMS FOR GROWING HIGHER PLANTS IN SPACE**

### **UNIVERSITY OF FLORIDA**

The overall goal of the project is to design unique systems and components for growing higher plants in microgravity during long term space missions. The plants will be used for food and atmospheric regeneration. This project is being conducted in cooperation with personnel of the NASA Controlled Ecological Life Support System (CELSS) project at Kennedy Space Center.

The goals of the first semester were to obtain a comprehensive overview of the requirements of the integrated system, to identify promising design topics, to perform preliminary design on promising tasks and to determine the practicality of various candidate systems and components for detail design and prototype fabrication efforts. The areas selected for indepth study and preliminary design efforts were: (1) automatic seeding and tissue culture, (2) adaptable geometry growth chambers, (3) plant health monitoring systems (including the areas of leaf finding, non-destructive tissue analysis and leaf diagnosis, and sensors), (4) Expert systems for automated, intelligent sensor interpretation, intervention and control, and (5) food preparation and waste processing.

After study and preliminary design, the candidate design tasks were ranked for appropriateness on the basis of potential importance, compatibility with class background and interest, availability of necessary resources, and likelihood of successful design and prototype fabrication. The top ranked projects chosen for detailed design during the second semester were:

1. automated seed preparation, manipulation and placement in microgravity,
2. automated sensing and interpretation of plant health, and
3. automated food processing, particularly milling.

Three projects were then formed, design criteria were refined and detail design and prototype fabrication efforts initiated.

The seeder group sought a system which would automatically separate seeds individually from bulk storage, move them to a desired location and plant them with precise spacing. Capabilities for use of either dry or wet, pre-germinated seeds were desired. Avoiding damage to fragile pre-germinated seeds was a particularly difficult problem. The seeder group produced three working prototypes. These were a dry seeder operating on air pressure gradients, a liquid-medium seeder relying on water pressure gradients, and an electrostatically-driven seed separation system. Each of these seeder prototypes is capable of planting wheat seeds with uniform spacing in a tabular seed cassette, which could then be placed in a growth chamber for cultivation.

The plant health monitoring group sought a reliable early warning system for detecting health problems across many different plants due to a variety of causes. They explored a wide range of plant attributes and associated sensing technologies and chose chlorophyll level and absorption spectroscopy as the most promising combination. The group then con-

structed a prototype spectroscopic analyzer that measures plant chlorophyll levels. This analyzer is completely computer controlled and has a limited capability to interpret the data obtained and to make predictions of the state of health of the plant.

The goals of the food processing group were to design and fabricate a device which could completely mill wheat into flour in microgravity without leaking the flour produced into the atmosphere. The group produced a closed containment milling device in which a high speed rotating shaft with retractable blades is

inserted into a sealed container of wheat. The device uses the air flow induced by aerodynamically configured blades to control the distribution of the wheat. Flour leakage during operation and shaft withdrawal is controlled by a combination of sliding disks and tight fitting compliant gaskets.

Overall, these projects were highly successful in that they produced novel and feasible designs for components essential to the growth and utilization of higher plants for food and atmosphere regeneration during long term space missions.

## DESIGN OF A LUNAR TRANSPORTATION SYSTEM

FLORIDA A&M UNIVERSITY/  
FLORIDA STATE UNIVERSITY

The purpose of the project on lunar transportation systems was to investigate the transportation requirements on the lunar surface. These requirements arise from the need to transport personnel and materials during the establishment of the first generation lunar base and subsequent expansion and enhancement. In the early years of habitation, lunar bases serve as facilities for scientific research, economic exploitation, and establishment of human presence on the moon. As a part of the requirement in an introductory course in engineering design, the lunar surface transportation problem, which typifies a real world problem, was specified as a project to be addressed. Real world problems are typically ill-defined and unstructured; there may be insufficient information to solve them, nor a unique solution for them. Students are encouraged to break the associated complex problems into many sub-problems, use rational decision making procedures in selecting from different alternatives, perform analyses, and synthesize results while communicating effectively among themselves. Three different topics were addressed on the general issue of lunar surface transportation:

1. Redesign of Earth-based flying machines on the moon.
2. Movement of material on, or inside, the surface of the moon.
3. Conceptual design of surface-based or flying vehicles on the moon.

These projects were chosen to enable the students to have a clear understanding of the geophysical and economical factors inherent with the lunar transportation problem and to identify possible solutions at a con-

ceptual level. Also, the projects were intended to emphasize innovative ideas and to foster sustained enthusiasm in the space program. Attention was focused on specific designs to be pursued during subsequent efforts in advanced courses. Objectives in the project included:

1. To minimize the construction materials and fuel to be supplied from Earth and maximize the use of lunar material.
2. To utilize novel materials and lightweight structures.
3. To utilize new manufacturing methods and technology such as magnetically levitated, or superconducting materials.
4. To develop innovative concepts effectively utilizing the exotic lunar conditions viz, high thermal gradients, lack of atmosphere and wind forces, lower gravity, etc.

The designs are conceptual in nature but cater to long and short distance movement of personnel and materials. These designs are to be applicable in the time frame of the first and second half of the 21st century as the lunar base continues to grow.

When the redesign of Earth-based flying machines was considered, the design of a rocket-type machine incorporating a catapult takeoff and additional boost capable of sustaining flight and increasing the range was investigated. Many of the flying machines on Earth were discarded because an atmosphere is needed for propulsion and lift. But many other concepts can be directly adapted or improved taking the lunar conditions, especially no drag forces, into account.



A transportation system called the Lunar Turtle Module (LTM) is contemplated:

1. To provide a flying mode of transportation for personnel and materials.
2. To minimize the use of propellants.
3. To minimize weight and materials used.
4. To maximize life by protecting the vehicle from adverse conditions present in the lunar environment.
5. To provide means of maintenance and maximum reliability and safety.

The LTM is a versatile transportation vehicle which operates above the surface of the moon. This proposed vehicle would operate using vectored thrust produced by hydrogen-oxygen rockets mounted on the vehicle. Forward motion can be obtained by vectorizing the thrust to produce the desired motion. Rotations can be obtained by using small thruster rockets mounted to the sides of the module. The cockpit will be designed to allow maximum comfort, visibility, and vehicle control. The cargo hold is to have a multiple task function allowing the transportation of cargo, people, or both, simply by using quick release mechanisms.

To address the problem of moving material on the lunar surface, one design solution utilizes a vehicle powered by conventional batteries. Another design involves the use of a nuclear or fuel cell power source in a multi-purpose lunar transportation vehicle capable of transporting personnel and materials. This track-type vehicle uses a Gleasman steering mechanism and drive system. This

system uses two standard differentials, worm and spider gears, and contains a minimum number of parts. A power differential allows the vehicle to move in a straight line. The steering differential utilizes (or employs) a gear and is powered by an auxiliary motor. The vehicle is designed in a sectional format. The first section of the vehicle is designed to contain only the drive, steering, and power supply. It will accommodate two occupants in full space suits and carry some minor equipment. The rear section will be equipped with an automatic coupling system in order to attach to various mission-dependent trailers.

A common but innovative design, called the Lunar Lift, involves a cable and pulley network for hauling materials and men and is powered by a thermoelectric module buried under the lunar surface. The thermal module is a thermoelectric device that utilizes a temperature differential to generate power. The underlying design of a thermo-module is based on the Seebeck effect. The Seebeck effect states that when two dissimilar conductors are connected and the junctions are maintained at different temperatures, an EMF can be observed in the circuit. In the lunar lift system, a thermo-module/heat-pipe bank will supply the charge for the batteries that will run the electric motor. Such a power generation system can also be used elsewhere. Design of an underground superconducting rail system using a repulsive magnetic force for levitation is also suggested. Use of high temperature superconducting material in conjunction with highly powerful magnetic material can be an effective answer to the problem of lunar surface transport.

However, the problem of control is still a major issue.

A possible design of a lunar surface-based vehicle is a van with a self-sustaining oxygen supply with a range of 120 km and a payload of 3600 kg. This vehicle is of square boxed construction, taking advantage of a lack of atmosphere and no drag. The box is made of Aluminum-Kevlar-

Aluminum composite material. The wheels are magnetically levitated and are of adjustable height to compensate for rough terrain. Another design is a fast moving surface-based vehicle suspended above a trackway by the use of superconducting magnets. The power requirements for such vehicles are considerably lower than those on Earth because of reduced weight and no wind loads.

**FACILITIES AND OPERATIONS FOR  
LUNAR LANDING AND LAUNCH**

**FLORIDA INSTITUTE OF TECHNOLOGY**

A preliminary definition of a lunar landing and launch facility (LLLF or Complex 39L) has been formulated. A phase III lunar base is considered (Ref. 1,2). Without specifying specific lunar base scenarios, three traffic levels are envisioned: 6, 12, and 24 landings/launches per year. A single multipurpose vehicle for the lunar module is assumed. The design and specification of the vehicle and of the lunar base are outside the scope of this study. The scope of this study is graphically illustrated by the system diagram in Figure 1. Here, major functions or facilities are represented by blocks in the system diagram. The dashed line represents the boundary of Complex 39L; and thus, the scope of this study. Based upon this diagram, nine major design items are considered:

1. Landing/launch site considerations
2. Structure, shelter, safety, and environmental needs
3. Landing/launch guidance, communications, and computing needs
4. Lunar module surface transport system
5. Heavy cargo unloading/loading systems
6. Personnel unloading/loading systems
7. Propellant unloading/loading systems
8. Vehicle storage
9. Maintenance, repair, test, and check-out requirements.

Initially, a general, conceptual description of each of these items is discussed. Then, preliminary sizes, capacities, and/or other relevant design data for some of these items are identified.

A Design Matrix for the lunar landing and launch facility has been developed and is shown in Figure 2. The matrix is composed of the nine major design items as rows. The specification, mass, power, and manpower of the facility are the columns of the matrix. Three main resource requirement areas are recognized: Construction, Operation, and Maintenance. The Construction area represents the resources that will be required during the construction phase of the landing and launch facility. The Operation area represents the resource requirements for the "steady-state" operation of the facility, while the Maintenance area represents the facility maintenance requirements. A matrix element is "checked-off" when the design work corresponding to that element has been completed.

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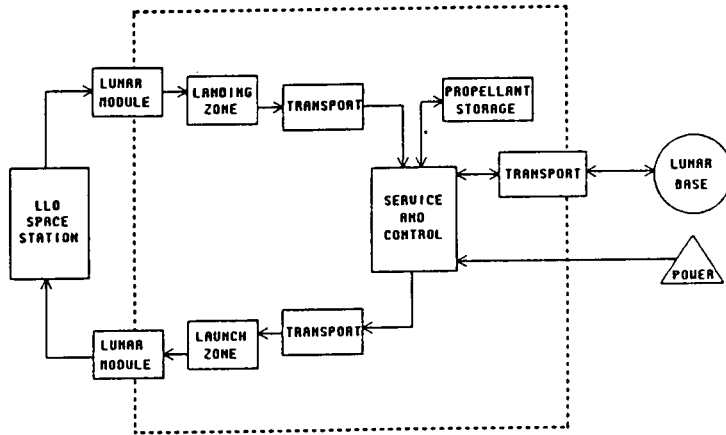


Figure 1 - Complex 39L Systems Diagram

	CONSTRUCTION			OPERATION			MAINTENANCE		
	SPECIFICATION	MASS	POWER	SPECIFICATION	MASS	POWER	SPECIFICATION	MASS	POWER
LANDING / LAUNCH SITE CONSIDERATIONS									
SHELTER, STRUCTURE, SAFETY, ENVIRONMENTAL NEEDS									
LANDING / LAUNCH GUIDANCE, COMMUNICATIONS, COMPUTING NEEDS									
LUNAR MODULE SURFACE TRANSPORT SYSTEM									
HEAVY CARGO UNLOADING / LOADING SYSTEMS									
PERSONNEL UNLOADING / LOADING SYSTEMS									
PROPELLANT UNLOADING / LOADING SYSTEMS									
VEHICLE STORAGE									
MAINTENANCE, REPAIR, TEST & CHECK-OUT REQUIREMENTS									

Figure 2 - Complex 39L Design Matrix

**THE SKITTER PROJECT**  
**GEORGIA INSTITUTE OF TECHNOLOGY**

The Georgia Institute of Technology has continued to develop its expertise in the field of lunar surface preparation and construction equipment. The approximate era of lunar exploration and the initial phase of outpost habitation are being addressed.

Work has continued on the analysis and design of SKITTER, a three-legged mobile platform. Dynamic and kinematic models have been developed and incorporated into a three-dimensional computer model that accurately models the behavior of SKITTER. Not only does this computer model graphically demonstrate its range of motion, but it also performs force and velocity analysis needed for actuator sizing. Designs of innovative structures to minimize weight but still provide the necessary strength have also been developed.

Implements to attach to SKITTER have also been developed. These include a robotic arm, an automated regolith movement system to bury the

common modules, drill rigs, and foot designs for the three legs.

While the centerpiece of this work has been SKITTER, the research has not been limited to walking machines. Wheeled vehicles that have the ability to be self-righting are one of the technologies of interest for the movement of large quantities of soil.

The ability to drill into the lunar soil is of prime importance in the construction of a lunar base. Techniques to remove the cuttings from the hold without the use of fluids have been developed and experimentally verified with mathematical models.

To provide the astronauts with the ability to perform manual tasks on the moon's surface, tools have been designed that will allow the operator to choose, from a variety of end effectors, the tool that will allow him to complete a variety of maintenance tasks.

**THREE SYNERGISTIC STUDIES:  
A MANNED LUNAR OUTPOST, A MANNED MARS EXPLORER,  
AND AN ANTARCTIC PLANETARY TESTBED**

**UNIVERSITY OF HOUSTON**

The students at the University of Houston College of Architecture undertook three synergistic studies. These studies included a Manned Lunar Outpost, a Manned Mars Explorer Mission, and an Antarctic Planetary Testbed which would provide the necessary facilities for testing proposed missions to the Moon, Mars, and beyond. Research was based on existing or near-term technology.

Manned Lunar Outpost

The Manned Lunar Outpost (MLO) is proposed as the initial permanent base for manned activities on the Moon. The study concentrated on identifying the equipment, support systems, and initial base configuration necessary to accomplish the various science, industrial, and exploration activities envisioned. The mission planning included the goal of low dependence on Earth-based goods, evolutionary self-sufficiency of the MLO, incorporating facilities such as agriculture and aquaculture, and eventual support of the Space Station.

The primary concepts of the MLO were using hard modules for habitation areas with inflatable interconnect nodes; creating a flexible, modular transportation system; designing a multi-functional vehicle; and using an overhead radiation protection system.

The transportation system, dubbed the Lunar Mobile Surface Transport System (LMSTS), carries the hard modules to the surface of the Moon and provides a method to move them to the desired location through the use of interchangeable pallets. The avionics pallets are exchanged with wheel and hitch pallets, transforming

the LMSTS (Figure 1) into a "tractor-trailer" to be used with the Multi-Functional Vehicle (MFV).

The modules are placed under the Regolith Support Structure (RSS) which provides a stable environment and radiation protection for the entire base. The overhead structure was chosen over simply burying the modules to provide information on the advantages and disadvantages of this type of system. The advantages include providing easy access to the exterior of the modules, providing a protected area for vehicles and equipment used in EVA, and creating an area of constant temperature. Disadvantages include a need for prefabrication of structural components and subsequent delivery, and the need to develop a conveyor system to lift the regolith into place. The research discusses all components in depth, including the pre-construction and construction phases of the initial MLO. Study team members included: Nathan Moore, Thomas Polette, and Larry Toups, with Nilanjan Bhattacharya consulting.

Manned Mars Explorer

The Manned Mars Explorer study had two primary objectives: to develop a mission scenario to deliver a crew of six to the vicinity of Mars and to design a transportation system to accomplish this mission.

The mission scenario was developed around the concentration of Phobos as the primary destination with the assumption the spacecraft would be built in LEO and outfitted with a crew of six. Using an opposition class Venus inbound swingby trajectory, the MPV would travel to the vicinity

of Mars in approximately 300 days, where it would stay for 60 days before the return leg to LEO of approximately 210 days.

The 60-day exploration period in the vicinity of Mars would consist of sending a crew of three to the surface of Mars for one week. The crew would then return to the MPV and spend the remainder of the time ferrying between the MPV and Phobos. During this period, the astronauts would perform scientific research on resource utilization on Phobos and remote sensing of Mars.

The design encompassed several considerations, including: all-chemical propulsion vs. nuclear-electric propulsion, the issue of reliability vs. redundancy, the need for artificial gravity vs. zero gravity, and the use of as yet undeveloped technologies such as large scale aerobrakes.

The primary components of the transportation system included a Manned Planetary Vehicle (MPV), a Crew Command Module, and an Interplanetary Cargo Transport for a later, split mission scenario.

After a detailed comparison of all-chemical vs. nuclear-electric propulsion for the MPV (Figure 2) was conducted, an all-chemical mission was chosen as the baseline. The MPV was designed to create one g of artificial gravity for crew health and safety. To accomplish this task, a tether system was developed which resists twisting through a unique spreader system. The tether would be deployed during trans-Mars coast and trans-Earth coast trajectory phases, and reeled in during propulsive maneuvers.

Due to the long and dangerous nature of this mission, reliability of vehicle components was established as a driving force in design. This was shown through the design of a Multifunctional Crew Command Vehicle (CCV) which would house the crew during all propulsive maneuvers, and ferry the crew between the MPV, Mars, and Phobos. Study team members included: Sean Nolan, Deborah Neubek, and C. J. Baxman.

#### Antarctic Planetary Testbed

The Antarctic Planetary Testbed was conceived as a project to complement crew preparation and construction technology development for both a lunar and Mars base. The primary focus of the research was the development of a program which defined the needs of a planetary testbed. Through the research of appropriate analogs, these needs were defined as: a master test and operational plan, selection of a transportation system, development of a construction system, and design of a module Figure 3 which responds to the program requirements.

The project's proposed Dry Valley site was chosen because of its close similarity to the harsh Martian environment as well as the barrenness of the lunar landscape. The analogs also extend to the realms of international cooperation, scientific research, and "launch windows" due to the unpredictable nature of the weather.

The master plan consists of 16 circulation modules, two habitation and laboratory/work modules, and four support modules. An inflatable structure is envisioned for vehicle storage

and agricultural facilities. An observation/control tower is also provided.

The transportation system, construction system, and design of the modules were interrelated in their development. Due to the limitations of delivery vehicles, the Sikorsky "Skycrane" was chosen. Therefore, a module was developed which fit within the helicopter's limitations while still closely following Space Station module dimensions and ratios. A module was designed which became the basic "building block" of the base. It consists of a compact core with two telescoping utility compartments on

the top and bottom. Once these utility compartments are deployed, the circulation core is exposed and Space Station derived racks are attached to the sides. The modules are then leveled by six hydraulic jacks.

In addition to the development of the master plan and design of basic module types, a conceptual analysis was done for support features such as power generation systems, waste management systems, and crew habitation space planning. Study team members included: Mashid (ShiShi) Ahmadi, Alejandro Bottelli, Fernando Brave, and Muhammad A. Siddiqui.

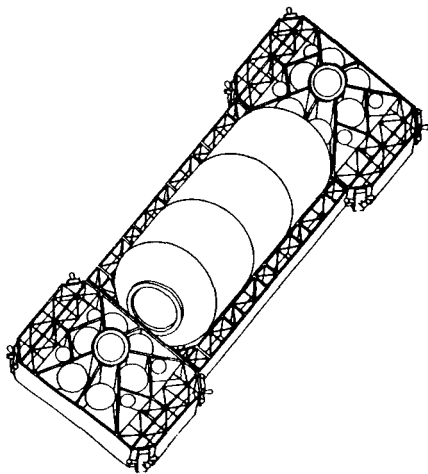


Figure 1 LMSTS with Avionics Pallets and Module

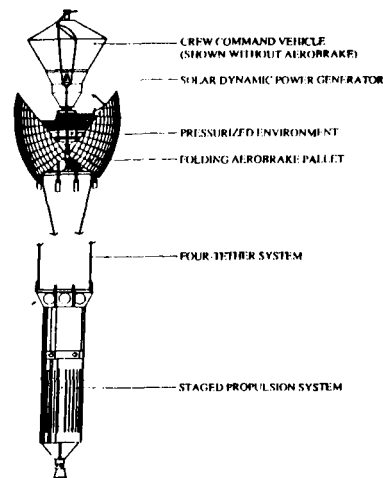


Figure 2 Manned Planetary Vehicle Components

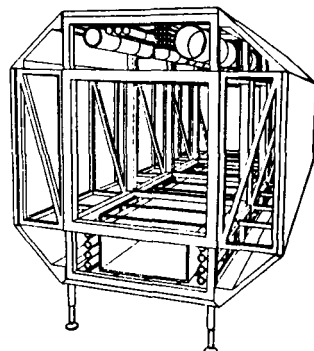


Figure 3 Module after Deployment and Leveling



## MARSPLANE

### UNIVERSITY OF ILLINOIS

The University of Illinois at Urbana-Champaign combined its senior-level aircraft and spacecraft design courses by addressing the design and delivery of a Marsplane. This offered a unique opportunity for systems engineering and integration to occur between the two groups. The aircraft group consisted of 56 students divided into eight groups, each addressing a major technical area. Similarly, the 50-member spacecraft section was divided into seven groups.

#### Marsplane

The general specifications for the Marsplane design were:

1. Payload: 1200 N.; two suited astronauts with life support systems,
2. Airfield Performance: Operations from prepared airstrips no longer than 1.0 km,
3. Cruise Performance: An unrefueled endurance of 8.0 hrs. at a cruise altitude of 1500 m, and
4. Crew Rescue: Scenario for the rescue of the crew of a down and disabled Marsplane.

For the purposes of the design, it was assumed that the mission occurs in the 1995 to 2020 time frame, that facilities on the Martian surface are available for assembling and servicing the Marsplane, and that all necessary operational facilities, materials, and supplies (e.g., fuel) are available.

A further objective of the study was to examine the impact on the Mars-

plane design of the decade of technological progress since the unmanned Marsplane design study of the 1970's.

The major technical areas of responsibility for the Marsplane design group members were:

1. Aerodynamics
2. Performance
3. Power and Propulsion
4. Stability and Control
5. Structures and Materials
6. Surface Operations
7. Weights and Balance

The designs were driven primarily by the low density of the Martian atmosphere (leading to low wing loadings) and the need for low propulsion system weight (including the weight of fuel). The take-off and landing specification proved to be the most difficult to satisfy. These factors, coupled with the need for rescue capability, drove several designs to include VTOL capabilities. With eight design groups, a wide variety of design approaches were developed. In summary they include:

1. Wing Configuration. Conventional, canard, joined wing, sail wing.
2. Propulsion System (not including take-off and landing). All were propeller systems driven by either fuel cells, solar cells, rockets, or a Stirling engine.
3. Take-off and Landing. CTOL, including rocket assist and powered carts; VTOL, rotors and rockets.
4. Materials. All designs relied heavily on composites to control weights.

The design results can be summarized by the following table of ranges of important parameters.

Maximum gross weight	4600 to 7500 N.
Wing loading	19 to 80 N/m <sup>2</sup>
Maximum installed power	12 to 62 kW
Power loading	79 to 625 N/kW
Operating empty weight	2850 to 4620 N.
Payload fraction	0.16 to 0.25
Fuel fraction	0.0 to 0.33
Propulsion system fraction	0.10 to 0.28

Spacecraft Delivery System

Seven transport spacecraft system design groups conducted sizing trade studies, selected subsystem components, investigated spacecraft dynamics and control, and performed trajectory analyses for the mission.

The mission was to be carried out by two primary components: the payload reentry system and instrument bus carrying scientific instruments for remote sensing of the planet's surface. The instrument bus was to remain in orbit after separation from the reentry system.

Enclosed in a sealed capsule and decelerated into orbit by an advanced aerobrake, the aircraft awaits an opportune moment for descent to the surface. Remote sensing instruments determine if the predesignated landing site is suitable before committing the reentry system containing the aircraft to its final journey to the surface. Afterwards, the orbiting instrument bus was to act as a relay satellite supporting the aircraft in its operations.

Actual packaging requirements specifying volume and mass were dictated by the aircraft groups. Several additional mission requirements were also specified for the transport system. A partial list is given.

1. Subsystems identified for the purposes of system integration:

- a. Aerobrake (including orbit capture, reentry, and detachment)
- b. Structure (including materials, design, thermal control)
- c. Power and Propulsion
- d. Attitude and Articulation Control
- e. Command and Data Control
- f. Science and Radio Relay Instrumentation
- g. Mission Management, Planning, and Costing.

2. The spacecraft's components and payload will be delivered to orbit in the cargo bay of the Space Shuttle and be assembled on-orbit at the space station spacecraft assembly and repair facility. The extent of shuttle support should be identified and minimized.

3. The spacecraft will be able to be retrieved by a remote manipulation device on the space station or space shuttle.

4. Nothing in the spacecraft's design should preclude it from performing several possible missions, carrying vastly different payloads to different destinations.

5. The spacecraft will have a design lifetime of four years, but nothing in its design should preclude it from exceeding this lifetime.

6. The vehicle will use the latest advances in artificial intelligence where applicable to enhance mission reliability and reduce mission costs.

7. The design will stress simplicity, reliability, and low cost.

8. For cost estimating and overall planning, it will be assumed that four space delivery systems will be built. Three will be flight ready, while the fourth will be retained for use in an integrated ground test system.

Several different trajectory options were explored for the mission. While Venus flybys were examined, requirements for a minimum  $\Delta V$  were best met by a direct ascent for the optimistic launch window between 2005 and 2010. The use of an aerobrake at the Mars terminal end of the trajectory lowered total  $\Delta V$  requirements to approximately 4.5 km/sec. A one year time of flight was typical for these trajectories.

Total up mass (wet delivery vehicle and spacecraft) varied from 15800 kg to 28800 kg. High and low thrust propulsion options were considered for the mission. The firm aerobrake requirement was shown to be a disadvantage for the low thrust options. Using a spiral trajectory out from earth and into an orbit around Mars, the low thrust options did not accrue the mass reduction advantage normally achieved by all chemical systems incorporating an aerobrake. The extra weight of the aerobrake could have been traded against a lower up mass and lengthened mission time.

Terminal orbit selection was varied among the design groups. While some groups sought to provide maximum global remote sensing coverage of the planet's surface, others sought to provide maximum global remote sensing coverage of the planet's surface, and still others sought to maximize contact time with their aircraft once it was in flight. Taking advantage of the fact that three spacecraft would carry their respective cargoes to Mars and then be available for operations, one group selected a sun-synchronous orbit for two of the spacecraft, guaranteeing global coverage as well as continuous contact with the aircraft.

Analysis showed that a thermal stagnation point exists approximately two aerobrake shield diameters behind the aerobrake. Accordingly, the aircraft payloads always had to be packaged to fit within the resulting protected conical volume behind the aerobrake. In most instances the aerobrake was reused for reentry protection. Parachutes and landing rockets were sized to provide a soft touchdown on the surface.

The Marsplane and Delivery System project demonstrated the technical feasibility of transporting an atmospheric flight system to Mars, assembling it there, and using it to carry out manned aerial reconnaissance of the Martian surface. Given certain enabling technologies now under development (e.g., low mass propulsion and structural systems, in situ fuels, etc.), a manned Marsplane appears to be a viable tool for use in the future exploration of Mars.

**SPACE STATION AUTOMATION  
AND ROBOTICS**

**UNIVERSITY OF MARYLAND**

A group of 20 electrical engineering students studied the problem of designing a dual-armed, free-flying, space robot. The students participated in a number of aspects of the design problem, including: spacecraft bus selection, battery power system sizing, attitude control system sizing, novel articulation concepts (highly redundant manipulators), novel robotic devices (dextrous grippers), dynamic modeling of free-flyers, etc. The students also enhanced an existing graphical animation test-bed to explore motion planning problems for the free-flyer. Figure 1 is representative of the current animation.

A subgroup of eight students undertook the ambitious task of designing and building a planar dual-armed

free-flyer (DAFF) to be floated on an air table in order to study engineering issues of robotic manipulation in zero gravity. The current design for the DAFF includes two planar two-link arms with grippers, remote drives located on the base body, a MIZAR single board computer for on-board intelligence, gas jets and a momentum wheel for attitude control, and digital wireless communication for command telemetry. The electromechanical hardware has been fabricated, and motor drive circuits have been designed and tested. The fabrication work will continue through the summer.

The course has been under the supervision of Professor P. S. Krishnaprasad.

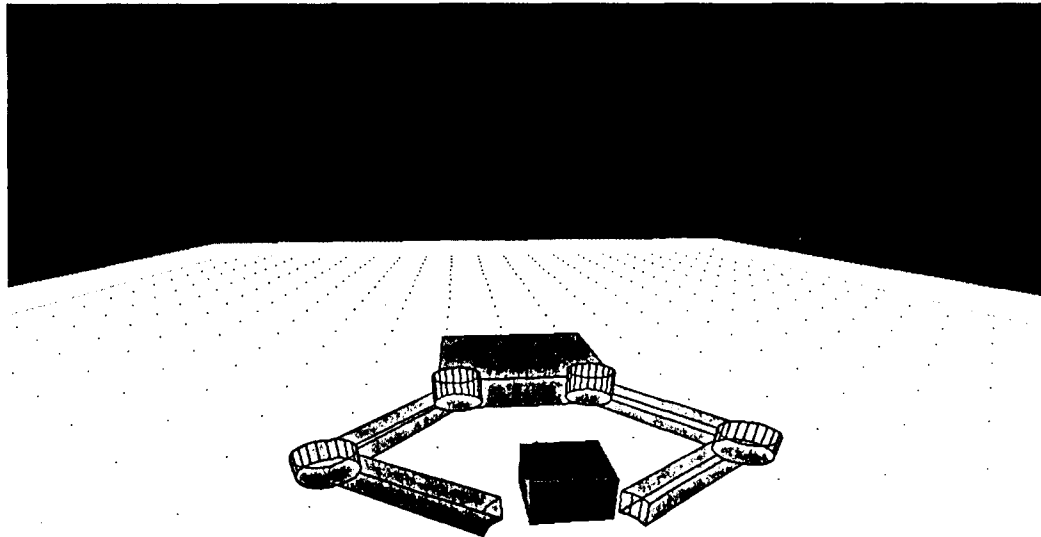


Figure 1

**A DECADE OF MANNED SOLAR SYSTEM EXPLORATION:  
THE MISSIONS OF DISCOVERY, CALYPSO, AND ENTERPRISE**

**MASSACHUSETTS INSTITUTE OF TECHNOLOGY**

To stimulate the engineering assessment of potential far-term missions for the space program, the assignment for the MIT Space Systems Engineering Class of 1988 was given as follows:

The goal is to design a system which will transport people on an (unspecified) mission in the solar system. Each of three design teams will be given a mission and vehicle name (based on famous ocean and space vessels), a departure date from Earth orbit, a technology readiness date (limiting the degree of advanced technology which may be assumed for the vehicle), and a total mission time. The objective is to design a system and mission that will extend the amount of useful knowledge that may be obtained from these exploratory missions, and that will result in exploring the most interesting real estate in the solar system. The mission groups are:

*Discovery.* A ten-year mission, leaving Earth in 2001, using technology available in 1990.

*Calypso.* A seven-year mission, leaving Earth in 2005, using technology available in 1998.

*Enterprise.* A five-year mission, leaving Earth in 2010; technology readiness date 2005.

Mission design must take into account actual planetary positions in calculating launch windows, mission scheduling, etc. Missions must carry and plan for effective utilization of humans, who must be returned safely to Earth at the end of the mission. Mission design elements will include orbital mechanics, space vehicle

design, mission science, details of landing craft, probes, and other payloads, and estimates of useful exploratory results from each stop/flyby of the mission. All mission designs will reflect sound engineering fundamentals, with reasoned extrapolation of future technological capabilities.

Space Systems Engineering is one of two restricted electives for Seniors in the MIT Department of Aeronautics and Astronautics. There were 72 students, divided evenly among the three missions, enrolled in the one-semester class in 1988.

The initial part of the study consisted of developing a work breakdown structure, which included all potential options for each of the necessary mission systems. Of obvious criticality in this particular mission were propulsion and power, life support (particularly radiation protection for the crew), and trajectory analysis/mission planning.

Detailed investigations of potential propulsion systems were performed, including conceptual systems such as gas-core nuclear reactors and anti-proton propulsion. The final assessment was that these advanced systems, however promising, would not be reliably available by 2005, and all three missions elected to use variants of ion or magnetoplasmadynamic (MPD) thrusters. The large power requirements of these systems dictated the use of multiple nuclear reactors, and a concurrent requirement for substantial waste heat radiation.

The primary problem in life support was found to be the protection of the crew from radiation. All three

extrapolations of current technologies. The final class report details (in approximately 1200 pages) the engineering and programmatic results of the mission designs and analyses; an overview of the design details are presented in Table 1. Estimates of costs for each mission were of the

order of one hundred billion (1988) dollars, which illustrates the scope of these missions, but the conclusion of the MIT study team was that each of these missions was technically feasible, as well as scientifically valuable.

**Table 1**  
**Overview of Mission Designs**

Mission	<i>Discovery</i>	<i>Calypso</i>	<i>Enterprise</i>
Scientific Objectives	Saturn Mars Venus	Asteroids (Ruth, Spiridona, Safara, Vesta)	Saturn Mars
Mission Duration	12.2 years	7.9 years	10.5 years
Crew Size	21	15	35
Habitable Volume	3117 m <sup>3</sup>	2800 m <sup>3</sup>	12,000 m <sup>3</sup>
Shielding	Passive: 20 gm/cm <sup>2</sup>	Passive: 530 gm/cm <sup>2</sup> (propellant)	Plasma shield
Total Dosage	200 rem	67 rem	50 rem
Total Mass	1.0x10 <sup>7</sup> kg	3.0x10 <sup>7</sup> kg	2.2x10 <sup>7</sup> kg
Propulsion System	Ion	Magnetoplasmadynamic (MPD)	Magnetoplasmadynamic (MPD)
Propellents	Mercury	Argon	Lithium
Total Power	116 MWe (9 reactors)	336 MWe (42 reactors)	408 MWe (50 reactors)
Waste Heat Radiated	188 MW thermal	646 MW thermal	410 MW thermal
Radiator Type	Fluid Tube (conventional)	Rotating bubble membrane	Liquid droplet sheet
Program Cost	\$168 B	\$115 B	\$170B

**CAMELOT II:  
PERSONNEL TRANSPORT BETWEEN EARTH AND MARS**

**UNIVERSITY OF MICHIGAN**

**Design Requirements**

A summary of the second phase of a design study of a cycling spacecraft for regular and frequent personnel transport between Earth and a Mars base is presented. The first phase study, called project CAMELOT, was a mission analysis and configuration design and was completed by the Space Systems Design class at the University of Michigan in the Winter term 1987. In the second phase the configuration and component design have been developed beyond the initial configuration study by the Space Systems Design class in the Winter term 1988. Analysis has led to numerous design modifications and improvements, as well as provision for more detailed system definition.

The initial study examined the mission objectives, functions, and requirements for the spacecraft. The mission of the spacecraft is to provide frequent and regular transportation of personnel between Earth and a Mars base. Seventeen engineers and scientists are required every few years to replace existing crew of the Mars base. The primary objective of Project Camelot is to transport these 17 people to Mars in the shortest possible time that will allow for regularity and frequency of such transfer. The Mars base is assumed to be well developed with a large support infrastructure consisting of supply tankers, fueling stations, a Phobos mining station and adequate facilities for communications.

Consideration of the trajectories required and the functions that the transportation system would perform led to an initial configuration whose main features were a large rotating

torus, a non-rotating boom, two docking ports, a microgravity research facility, and three solar dynamic collection clusters. As a result of the requirements and subsequent design to meet these requirements, the initial design team coined the acronym CAMELOT -- Circulating Autonomous Mars-Earth Luxury Orbital Transport. In the current study no changes were made in the mission objectives or orbital mechanics of the mission, but several key changes have been made in the spacecraft systems and layout. Figure 1 shows the spacecraft design after the second phase.

**Nominal Trajectory**

The nominal trajectory was calculated using several standard simplifications, namely:

1. Earth and Mars are in concentric, co-planar, circular orbits around the sun.
2. Gravity effects of Mars are ignored.
3. The synodic period of Earth and Mars is 2.135 years.

These assumptions result in an "up-escalator" orbit that has a period of 2.135 years, exactly equal to the synodic period, with a short leg transfer time between Earth and Mars of only 4.5 months and a long leg transfer time between Mars and Earth of 21 months. By equating the period of the up-escalator orbit with the synodic period of the two planets, the spacecraft should encounter Earth and Mars in the same relative positions each orbit. The issue is complicated, however, by the fact that the Earth-Mars alignment, while repeating every synodic period relative to each other, does not repeat itself in an in-

ertial reference frame. The Earth-Mars alignment occurs 48.7 degrees further around the Sun each orbital period. This advance in the positions of the planets requires that the semi-major axis of the escalator orbit also be rotated by 48.7 degrees in order for the encounters to occur on a regular basis. With an Earth flyby altitude of 1000 km, a rotation of 43.7 degrees is achieved using gravitational assist. This is almost the entire rotation required and a small impulsive burn ( $\Delta V$ ) near the aphelion of the trajectory provides the remaining 5 degrees of rotation.

#### Mission Overview

After the spacecraft has been assembled in low Earth orbit (LEO) and inserted into the up-escalator trajectory, two "taxi" (orbital transfer vehicles) will depart from a LEO space station with 17 passengers (the three crewmembers will already be on board) and rendezvous with the spacecraft. The two taxis will berth and remain attached until five days prior to flyby of Mars. The 17 passengers will then disembark via the taxis to replace 17 people currently serving their tours of duty on the Mars base. The three crew members will remain on board for station keeping and maintenance purposes until the next encounter with Earth.

At the same time two taxis will depart the Phobos spaceport with the rotated Mars base personnel for eventual rendezvous with the spacecraft. The taxis will be required to perform propulsive burns that will allow each taxi to escape planetary gravity along hyperbolic trajectories that will intercept the spacecraft up-escalator trajectory. These two taxis will remain

with spacecraft until the next planetary encounter. Earth encounter and taxi transfer will occur in a similar manner. The Earth taxi base will most likely be stationed in a low Earth orbit, possibly a space station.

#### Detailed Design Studies

Analysis has led to numerous design modifications and improvements, as well as to the provision of a more detailed system definition. Major components of the second phase study are discussed.

#### Propulsion

The designs of a liquid hydrogen/oxygen rocket system for orbital insertion and change and of thrusters for spin-up of the torus, for maintenance of a constant rotation rate, and for altitude and direction control were addressed. Consideration was given to engine lifetime and fuel storage problems.

#### Docking

The design of a two module docking facility -- the Docking and Operations Capsule (DOC) and the Cargo Acquisition Bay (CAB) -- was undertaken. The DOC contains the two berthing ports and the operations and control for activities involving personnel and cargo transfer. The CAB serves for all cargo transfer and as a multi-purpose space platform for various activities including repair and maintenance.

#### Power Systems

Design of the power systems to provide electrical power for the vehicle was investigated. A solar dynamic



power system consists of an energy source subsystem of 16 solar collectors, a power conversion subsystem of Stirling heat engines which convert thermal to electrical energy using alternators, and a radiator subsystem for rejecting waste heat. The reserve power system consists of hydrogen/oxygen regenerative fuel cells.

### Interface

A breakthrough detailed design of an interface between the rotating and non-rotating portions of the spacecraft consisting of concentric cylinders was identified. An outer cylinder is attached to the torus and rotates with it while the inner cylinder is attached to the non-rotating portions of the spacecraft. Friction in the bearings between the cylinders is overcome by electrical motors. The interface also serves for load transfer between the boom, where thrust is applied in orbital maneuvers, for personnel and cargo transfer between the two parts, and for power and data transfer.

### Elevator

A design of an elevator for moving personnel and cargo from the nonrotating interface section at the hub of the rotating torus along a spoke of the torus to the outer ring and back was pursued. The elevator is designed to compensate for the disorienting and uncomfortable effects of the various velocities and accelerations of the spacecraft, the elevator, and the rotating torus.

### Radiation Protection

The design of a magnetic radiation shield using four super conducting

cables was addressed. The cables exert tremendous forces when fully charged; nevertheless, using a magnetic shield has the potential for a much lower mass for the spacecraft than passive shielding.

### Habitat

Requirements for the living and working areas for the crew were identified and conceptualized. The torus is divided into 11 modules of varying sizes with airlock-type doors. Each module is integrated into the overall ventilation, water supply, waste disposal and electrical systems. An innovative mass balancing system consisting of shifting water to compensate for movement of people and objects within the torus was designed.

### Truss

A truss structure within which the various components are attached was designed. The truss carries the main loads from orbital insertion and change. The solar collectors also are attached to truss arms.

### Assembly

Identification of the components and process for assembly of the spacecraft in low Earth orbit was additional task addressed in this study. A breakdown of the spacecraft into manageable units and a launch sequence were determined. Several techniques and tools were designed to allow a small crew inhabiting two modules to efficiently complete the assembly.

Finite element stress and overall dynamic analysis has verified much of the system and indicates the validity of the design. Computer simula-

tion of many moving components, including torus rotation, elevator and interface operation, and attitude control mechanisms further validate and support the various systems.

References

1. Project CAMELOT: A Cycling Transportation System in Support of a Manned Mars Base. University of Michigan Space Systems Design Class

1986-87.

2. Pioneering the Space Frontier: The Report of the National Commission on Space. Bantam Books. May 1986.

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4. Circulating Transportation Orbits between Earth and Mars. A. Friedlander, J. Niehoff, D. Byrnes and J. Longuski. AIAA Paper 86-2009.

Castle Specifications Legend

Mass in Orbit	2.5 Million Kg
Orbital Period	2.135 years
Inhabitants	3 Crew, 17 Passengers
Torus Cross Section	7.5 x 4 m
Torus Radius	35 m
Torus Rotation Rate	3.22 RPM
Artificial Gravity	0.4 g
Distance From Hub to	
Main Engines	70 m
Main Engines	9 Rocketdyne
Main Engines	OTV Engines
Thrust per Engine	67,000 N
Propellant Tank Radius	4 m
Solar Collector Radius	9.4 m

Castle Isometric View

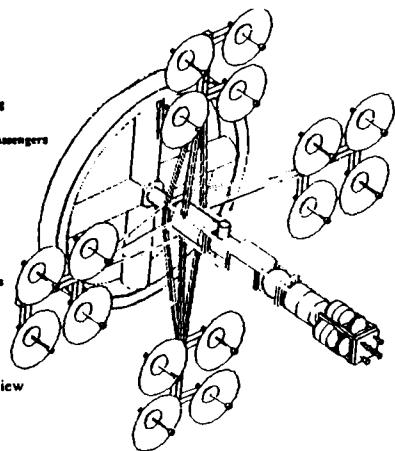


Figure 1: Design Specifications

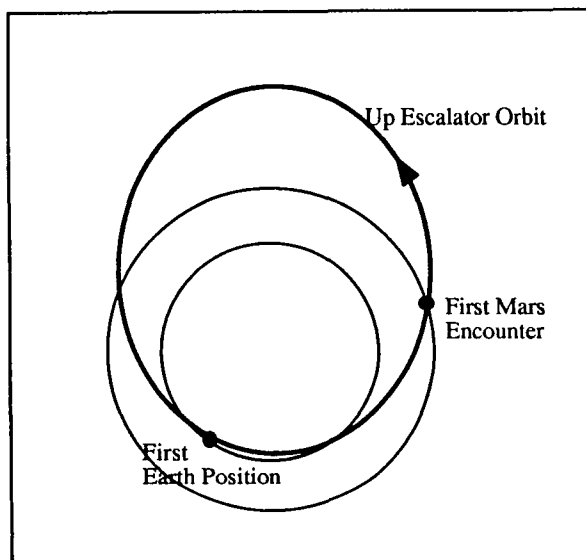


Figure 2: Up Escalator Orbit

**VARIABLE-GRAVITY RESEARCH FACILITY  
CONCEPTUALIZATION AND DESIGN STUDY**

**UNIVERSITY OF NORTH DAKOTA**

Background

The Variable-Gravity Research Facility (VGRF) is a manned low Earth orbiting station which is designed to study the effects of different levels of gravity on human physiology. The VGRF is a simple design which makes use of existing equipment to keep costs as low as possible. This approach is intended to make the project affordable in a tightly controlled congressionally constrained budget period.

The United States and the Soviet Union have a modest amount of data on human physiology under zero gravity (0 g) conditions. It is clear that the physiological changes which occur when adapting to 0 g are not appropriate for returning to 1 g. In order to be physiologically fit to return to 1 g from 0 g, it is necessary to spend hours every day exercising. Furthermore, it is not clear that all physiological systems do readapt well in reasonable time periods after extended time in 0 g. The VGRF will provide data on the physiological adaptation to gravity levels between 0 and 1 g.

Proposal

We propose three six-month missions at differing levels of gravity (0.255, 0.39, and 0.64 g) with crews of three individuals. These experiments will be done in order to obtain the first approximation of the shape of the curves for the physiological responses to different levels of gravity. The resulting data will permit preliminary evaluation of the appropriateness of artificial gravity as a countermeasure for the deconditioning which occurs in 0 g.

Conceptual Design

The basic design of the VGRF (Figure 1) is a habitation module (HM) attached by flexible Kevlar tethers to a counterweight. This structure can be spun while in orbit to provide artificial gravity. One of the cheapest available counterweights is an external tank (ET) from the space transportation system. The HM will be modified from the Space Station (SS) HM. After the three six-month missions are completed, it can be reconfigured and used as the SS HM or used for those gravity applications such as a lunar base or manned Mars mission. The HM by itself lacks some of the equipment needed to make the system habitable so an SS node (node 2) will be required as well. In addition, as in the SS, a crew emergency return vehicle (CERV) will be needed for safety purposes. The VGRF cost can be substantially reduced by using the HM, node, and CERV from the SS design.

Several minor modifications are required in the SS HM in order to make it suitable for use in a gravity field: A) The sleeping compartments must be rearranged to permit sleeping horizontally under gravity. B) Racks to support equipment modules of the HM while they are being maintained will have to be taken to orbit since the system will now be under gravity. C) Due to the rotation, the power requirements must be handled differently than on the SS. A 15 kW fuel cell system using residual fuel from the ET seems to be most appropriate. D) Radiators will need to be mounted on the side of the HM instead of at a remote location as on the SS truss. E) Antennae must be mounted so that they can swivel with the VGRF's ro-

tation. F) Tethers will be used to hold the HM and ET together. These can be attached to the HM and ET at the points employed for moving these structures under 1 g while on Earth. G) Small rocket motors for spin up will be installed on the HM and ET. H) Despinning will be accomplished by severing the tethers at the HM. The successive missions will each use new tethers and ETs.

Tether Stability Test Equipment

At a more practical level, test equipment has been built to evaluate

questions related to stability in the tether system. A rotating arm has been designed and constructed which will release a model HM-ET tethered system and photograph it while it is in free fall. Disturbances can be introduced into the HM model to permit examination of their propagation in the tethers. To complement the experimental work, a substantial start has been made on a computer model of tether variations. The combination of a physical model and a computer model will be ideal for evaluating various designs.

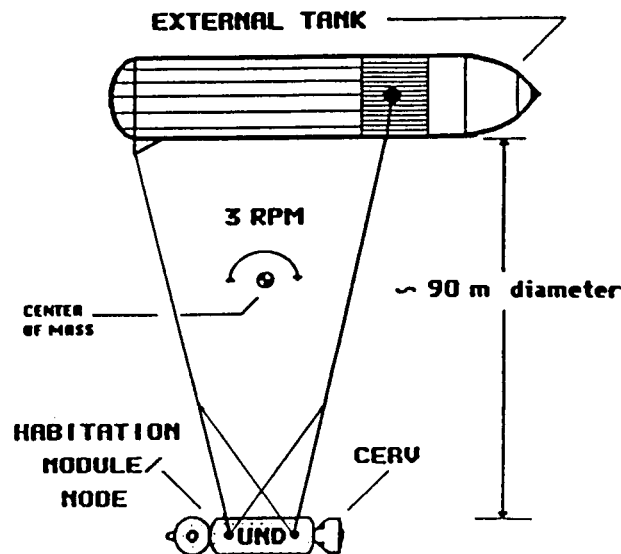


Figure 1

## MARS OXYGEN PRODUCTION SYSTEM

### OLD DOMINION UNIVERSITY

In situ production of oxygen is an essential requirement for future manned missions to Mars, but can also be an important building block in earlier missions. The Viking Lander data provided measurements of the composition of Martian atmosphere. Previous studies have shown that carbon dioxide, which makes up more than 95 percent of that atmosphere, can be dissociated thermally to produce oxygen. In order for oxygen production to become an accepted option for Mars missions, it is important to demonstrate that systems can be built which operate reliably, with minimal human attention, for at least one year. The project reported here is sponsored jointly by USRA and The Planetary Society and is directed toward demonstrating the feasibility of a Mars oxygen production system.

The demonstration effort consists of custom-mixed, bottled gas containing the five primary constituents of Mars atmosphere in their proper proportions, and is metered through a flow network which is operated at absolute

pressures ranging between 50 and 100 mb. The simulated atmosphere flows into an oven where it is heated to 10000K and introduced into a stabilized zirconia cell. The oxygen produced by dissociation of carbon dioxide is separated from the stream by electrochemically pumping it across the zirconia cell membrane. The oxygen and exhaust gas flows are measured accurately before they enter the vacuum tank prior to venting.

Sampling of the simulated Mars gas can be accomplished at pressures above one atmosphere. However, both the oxygen and exhaust gases must be collected at sub-atmospheric pressures. Design problems associated with the sampling system were addressed. In addition, evaluation and testing of hardware and software required for data acquisition, temperature, pressure and flow rate measurement and control, and data display have been accomplished. Oxygen production is planned to be demonstrated in May 1988.

**CONCEPTUAL DESIGN OF A WATER TREATMENT SYSTEM TO SUPPORT  
A MANNED MARS BASE**

**PRAIRIE VIEW A&M UNIVERSITY**

The objective for the NASA/USRA design team for Prairie View A&M University has been the conceptual design of a system for treating a supply of subsurface water, in an amount sufficient to satisfy the needs of biological consumption and general use by a Mars base. As a part of the effort, it is also necessary to provide for the manufacture of any chemical materials which are required by the water treatment system, and/or manufacturing of materials, and life support biosphere.

The specific design objective has been set at a total supply of 10,000 gal/day of water. Water is to be made available at three different levels of treatment: 1,000 gal/day of potable water, 1,000 gal/day of demineralized water, and 8,000 gal/day for general domestic use and manufacturing. Since no details are known about a potential underground supply of water, the water was assumed to be similar to a representative source of underground water in Texas.

The water treatment system involves

a number of physical and chemical steps. Since all chemicals must be produced from local raw materials, physical processes are used wherever possible. The following sequence of treatment steps is present in the system: sand filtration, ultrafiltration, reverse osmosis, disinfection, ion exchange, and an activated sludge process to remove biological waste. Sand filtration and ultrafiltration remove suspended solids; reverse osmosis and ion exchange remove dissolved solids. Ultrafiltration and reverse osmosis are both membrane separation processes; however, the membranes have different physical and chemical characteristics. Ion exchange involves reaction between mobile ions at specific sites on the resin surface and ions of similar charge in solution. Periodic regeneration is required for this treatment. Chemicals for the regeneration of the ion exchange resin are derived from the concentrated salt stream which is rejected from the reverse osmosis process. Effluent from the waste treatment is directed to growth of plants for food.

**1987-88 DESIGN PROJECTS**

**THE UNIVERSITY OF TEXAS AT AUSTIN**

Students in Aerospace Engineering and Mechanical Engineering at The University of Texas at Austin completed eight separate design projects under the sponsorship of the NASA/USRA Advanced Space Design Program. The designs executed by student teams were:

Fall 1987

Bootstrap Lunar Base - ASE  
Convertible Lower Lunar Lander Stage - ME

Spring 1988

First Mars Mission Crew Transfer Vehicle - ASE  
Earth Orbiting Transportation Node - ASE  
Lunar Construction Shack Vehicle - ASE  
Ozone Depletion Arresting Systems Assessment - ASE  
Lunar Surface Navigation System - ME  
Lunar Base Radiation Protective Structures - ME

The design objectives, a summary of the results, and selected comments are given concerning each of the projects. Due to the number of projects completed and the space limitations for this project summary, the reader should refer to the project reports themselves and the more extensive summaries found in the conference summary for additional information.

Bootstrap Lunar Base

A design for bootstrap lunar base was specified which required a minimal total payload to be delivered to the

moon in the process of setting up a preliminary base. The specifications required that there be maximum utility of every item delivered to the surface of the moon and that every item be reusable and/or "transformable" (capable of being used for another purpose). The design team divided its efforts into three areas: lander fleet requirements definition, lander design development, and lunar surface operations definition. The lander fleet activity defined the trajectory, navigation, and communications requirements for the lander fleet. The lander design activity centered on the definition of preliminary specifications and designs for candidate lander vehicles, concentrating on lower stage designs which provide resources for use in establishing the bootstrap base. The surface operations activity centered on the definition of a bootstrap base development scenario in which the bootstrap base becomes the cornerstone for the development of a permanent lunar base.

Convertible Lander Lower Stage

A three person team of ME students designed a lower stage for a lunar lander vehicle which was capable of being converted into a shaft head and elevator for a vertical access shaft for a buried lunar base facility. The design specifications were the same as those given to the ASE group designing candidate lower stages for lunar landers -- namely that no items transported to the lunar surface could be of the one-time use variety. The elevator provided compression and decompression to and from the lunar surface and the habitat.

### Fast Mars Mission Crew Transfer Vehicle

This design consists of trajectories and vehicle specifications for the crew transfer vehicle required in the split mission to Mars. The purpose of the fast crew transfer is to minimize the total radiation dosage and the effects of microgravity on the crew. The vehicle flies to Mars in 150 days, is serviced at the equipment barge already in Mars orbit, waits 40 days in the vicinity of Mars, and then returns to Earth in 150 days. The 40 days are spent exploring Mars. In case of a mission abort prior to entering Martian orbit (and being refueled), the crew transfer vehicle would be capable of a low energy (slow) return to Earth. An abort after Mars orbit would require refueling prior to return to Earth. In the proposed scenario, all propulsive velocity changes are made using chemical propellants.

The vehicle itself does not employ artificial gravity and the life support system is designed for a nominal mission duration of one year with reserves for a slow return abort. The vehicle employs four crew modules similar to the proposed Space Station Common Modules plus a heavily shielded command module. The shielding is designed to protect the crew in case of a major solar radiation event.

### Earth Orbiting Transportation Node

This design outlines the components of an Earth orbiting transportation node which supports the establishment and maintenance of a permanent lunar base during its first decade. The study includes a comparison of

two lunar base traffic models and assumes the existence of orbital transfer vehicles (OTV's), orbital maneuvering vehicles (OMV's) and heavy lift launch vehicles (HLLV's). The transportation node, called GATEWAY, is distinct from the Space Station, and is designed to be a high activity "noisy" environment unsuitable for microgravity experiments.

GATEWAY is characterized by a delta truss design and has facilities for a dedicated crew of four plus temporary accommodations for transient personnel. Propellant storage sufficient to meet the fuel and oxidizer requirements of the OTV's is provided. Provision is made for assembly of OTV payload stacks within a hangar area within the facility. Remote manipulators and other facilities necessary for OTV and OMV maintenance are included in the design.

Several candidate launch sites and orbital inclinations were studied for use by GATEWAY and its associated vehicle fleet. A circular orbit at an altitude of 240 to 260 nautical miles and an inclination of 28.5 degrees, supported by launches out of KSC was chosen. Analyses of the number of launch windows available per year between GATEWAY and various lunar orbits and lunar orbiting facilities are included. It is concluded that unless a lunar orbiting facility (MOONPORT) is in a lunar equatorial orbit, it will only be occasionally accessible from GATEWAY, and might as well not exist. It seems much easier to reach most points on the lunar surface directly from GATEWAY than to go through a seldom accessible lunar orbiting transportation node.



### Lunar Construction Shack Vehicle

The lunar shack vehicle is a lunar lander which provides an initial construction crew of eight with a habitat and radiation protection during the initial phases of lunar base construction. The lunar shack vehicle carries supplies for an initial construction period of three weeks and can be resupplied. The crew and additional supplies are brought to the surface via a crew ferry vehicle.

It is assumed that a nuclear power plant has been landed prior to the arrival of the lunar shack and that it takes about one week to activate the nuclear power plant. During the first week, power is obtained from fuel cells in the lunar shack. The fuel cells also serve as a backup power supply for the shack vehicle. The descent engines and the fuel cells share a LOX-LH2 fuel system.

Several design features deserve comment. First, the thermal control system of the vehicle features a Water Reservoir Heat Exchanger consisting of an insulated water reservoir which acts as a heat sink during the lunar day and as a heat source during the lunar night. Provision is made to radiate heat from the reservoir during the lunar night if and when necessary. Second, crew radiation protection is provided by partially burying the shack vehicle. The center of the cylindrical vehicle is buried with the ends, which contain airlocks, remaining uncovered. Slumping of the regolith over the airlocks is prevented by collars at the ends of the buried segments. Finally, interior layouts for the chosen design and an alternate design are provided.

### Ozone Depletion Arresting Systems Assessment

This study is an initial inquiry into (a) the nature of the mechanisms which are causing the depletion of ozone at high altitudes over the south pole (and to a smaller extent over the remainder of the Earth), (b) the possibility of slowing or stopping the ozone depletion process either by accelerating the formation of ozone or by removing substances responsible for ozone depletion from the atmosphere, and (c) the preliminary characterization of vehicles and/or mechanical systems which might be useful in combatting the ozone depletion problem. The study resulted in more question than answers, but this was expected. The primary motivation of this study was to focus the immediate attention of engineers on this problem because of its possible extreme consequences. It was felt that early thought into possible solution mechanisms, even with a lack of complete understanding of the underlying causes, would be beneficial.

Important questions identified during the study are: (1) What are the effects of the decrease in atmospheric ozone and at what stage does the problem become critical? (2) How can we accelerate the natural processes which "wash" out of the atmosphere substances that attack ozone? (3) Can we shift the reaction equilibrium away from the ozone destruction by introducing additional substances into the atmosphere without causing other harmful effects?, (4) Can we filter ozone destroying substances from the atmosphere, either near the surface or at altitude?, and (5) What facilities

and/or vehicles will be necessary to implement a solution.

#### Lunar Surface Navigation System

This study investigates a navigation system to be used on the lunar surface by astronauts to locate their positions when beyond the line of sight of the base facility. The navigation system is to be deployed for use in the time period of the early development of the moon's surface operations. The navigation system must be usable by an astronaut while in full pressure suit. The study provides an analysis of four navigational methods for adaptation to the lunar surface: celestial, satellite, inertial and radio-beacon. The study revealed that a spread-spectrum radio-beacon navigation system is best suited for the design needs. The report develops the scenario necessary for initial operations of a lunar base, and provides alternatives which will allow the original system to be expanded as lunar surface activities expand. A celestial navigation system is suggested as a back-up in case of power failure of the spread-spectrum radio-beacon navigational system.

#### Lunar Base Radiation Protective Structures - ME

This study analyzes alternatives and proposes a design that will provide astronauts protection from lunar radiation and micrometeorite hazards. The protective structure is designed for use on a manned lunar research outpost. Similar to the other studies performed by The University of Texas this year, emphasis was placed on increased use of material found on the lunar surface. Any material delivered to the lunar surface is designed for continuous use or is to be transformable. The study investigated both above the surface and below the surface alternatives. The most potentially promising solution was the use of existing lunar lava tubes to provide radiation protection. The design team concluded, however, that insufficient information currently is available to design the necessary structures using the lava tubes. As a result, the team designed a structure consisting of a cylindrical module buried in lunar regolith.

## **FIVE DESIGN PROJECTS**

### **TEXAS A&M UNIVERSITY**

Five designs, under the Advanced Program Space Project grant by NASA/USRA, were accomplished during the Spring Semester, 1988. Two design projects were done by Aerospace Engineering students, while Nuclear Engineering students undertook three projects.

#### Aerospace Engineering

1. **A Lunar Tunnel-Boring Machine.** A need exists for obtaining a safe and habitable lunar base that is free from the hazards of radiation, temperature changes, and micrometeorites. A device for excavating lunar material, and simultaneously generating living space in the subseleanean environment to meet the aforementioned requirements, has been researched and conceptualized. Preliminary investigations indicate that a device using a mechanical head to shear its way through the lunar material while creating a rigid ceramic-like lining meets design constraints while utilizing existing technology. The Lunar Tunneler is totally automated and guided by a laser communication system. The potential exists for the excavated lunar material to be used in conjunction with a surface mining process for the purpose of the extraction of oxygen and other elements. Experiments in lunar material excavation and further research into the concept of a mechanical Lunar Tunneler are suggested. Figure 1 illustrates this concept.

2. **A Subseleanean Melting Head Tunneler.** The placement of base facilities in subsurface tunnels created as a result of subsurface mining is described as an alternative to a base on the lunar surface. Placement of the

base facilities and operations in subseleanean tunnels will allow personnel to live and work free from the problems of radiation and temperature variations. A conceptual design was performed for a tunneling device applicable to a lunar base to assess the feasibility of the concept. The resultant design was a tunneler which would melt through the lunar material leaving behind glass-lined tunnels for later development. The tunneler uses a nuclear generator which supplies the energy to thermally melt the regolith about its cone-shaped head. Melted regolith is excavated through intakes in the head and transferred to a truck which hauls it to the surface. The tunnel walls are solidified to provide support lining by using an active cooling system. Also addressed in this study are the rationale for a subseleanean tunneler and the tunneler configuration and subsystems, as well as the reasoning behind the resulting design. A conceptual drawing of this design is shown in Figure 2.

#### Nuclear Engineering

1. **Mars Sample Return Power Supply.** The purpose of this design is to provide a power supply for a vehicle which will be able to operate on Mars for a time period of five to ten years. The vehicle will be used for sample and data collection on the Martian surface. This design is based on the assumption that the vehicle will be unmanned. Also, there will not be any means by which components could be repaired or replaced while on the Martian surface. A consequence of this constraint is that all equipment will be forced to meet a high standard of reliability and, if possible, redundancy.

Power will be supplied to the vehicle by means of a General Purpose Heat source capable of producing 7 kW of minimal thermal power. The heat generated from the General Purpose Heat Source will be transferred to a Stirling engine via "hot side" heat pipes. The Stirling engine will then convert this heat into 2 kW electrical power. "Cold side" heat pipes will be used to carry away waste heat from the cold side of the Stirling engine. This heat will then be released to the Martian atmosphere via radiators connected to the end of the "cold side" heat pipes.

2. Nuclear Electric Propulsion Turbine Driven Uranium Nitride Energy (NEPTUNE) Source. The NEPTUNE source consists of a boiling potassium space reactor system providing electrical power for habitation and propulsion. The system is designed for a 20 year life to accommodate a manned, roundtrip mission to the outer planets. Primary emphasis of the design is placed on the reactor and propulsion systems in order to provide a framework for a future mission of this type. The selection of the reactor system and propulsion components was based on the power requirements for the propulsion system due to the fact that it represents the bulk of the power consumption. The entire system is designed to meet the mission lifetime requirements with a high degree of reliability. The work presented is a baseline design capable of fulfilling the mission requirements. Recommendations are made throughout for possible future work to further detail the system.

3. The MPR-300 Reactor System for use in Mars Applications. A pebble-bed nuclear space reactor and supporting equipment were designed to supply energy for a Mars mission propulsion system, as well as for additional space vehicle power. Results are shown below.

Reactor Parameters

Thermal Output	1 MWt
Electrical Output	300 kWe
Power Density	3.1 W/cm <sup>3</sup>
Power Production	
Thermal Flux	91%
Fast Flux	9%

Fuel Parameters

UN Fuel Mass	145.7 kg
Fuel Enrichment	8.0%
Mass of U-235	10.8 kg
Critical mass	
at startup	7.6 kg
Fuel burnup	3.2 kg
Specific Power	6.86 kW/kg UN

System Thermodynamic States

State	Temperature(K)	Pressure(MPa)
1	850	6.58
2	677	3.76
3	428	3.63
4	500	6.90

System Energy Balance

$Q_{in}$	=	1038 kW
$Q_{out}$	=	738 kW
$W_T$	=	513 kW
$W_C$	=	213 kW
$h$	=	28.9%

Propulsion		Mass	
MPD Thrusters	13 units	Propulsion	2350 kg
Total Craft Mass	8000 kg	Reactor system and shield	1029 kg
Total Thrust	3.6N	Rotating machinery	565 kg
Propellant	NH <sub>3</sub>	Ducting and structure	700 kg
Mass	1571 kg	Heat pipes	2130 kg
Flight Time	242 days	Miscellaneous	480 kg
MPD Power	296 kW		
Specific Impulse	6140 sec		
		<b>Total Mass</b>	<b>7254 kg</b>

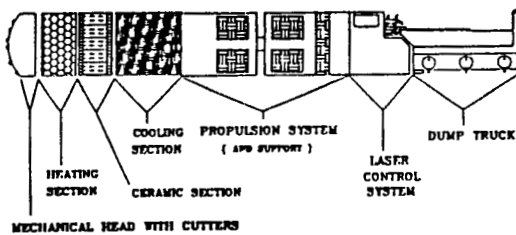


Figure 1 - Lunar Tunnel-Boring Machine

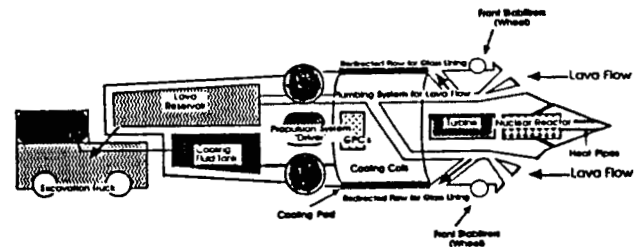


Figure 2- Subseleanean Tunneler Melting Head Design

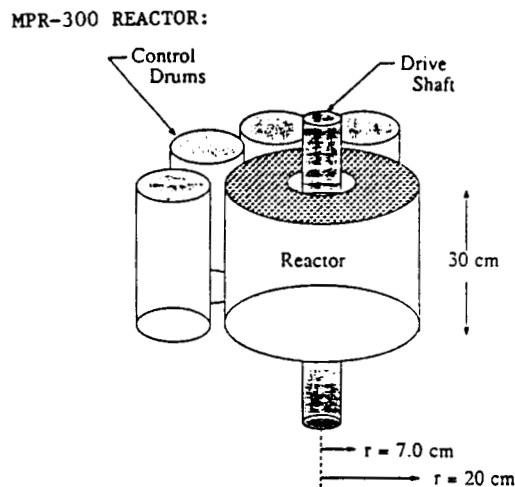


Figure 3 - Conceptual Reactor Design

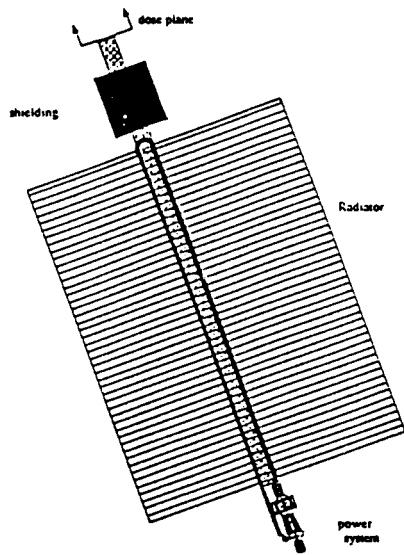


Figure 4 - Space Craft Configuration

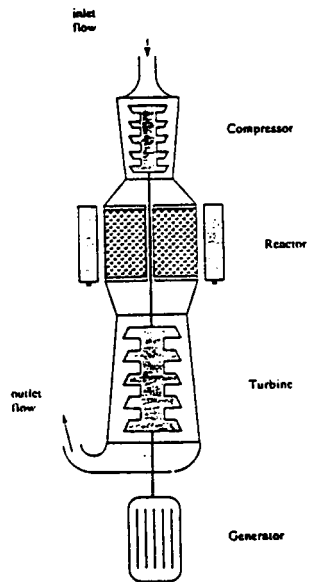


Figure 5 - Power System

## PROBE TO ALPHA CENTAURI

### U.S. NAVAL ACADEMY

This project details a design for an unmanned probe to Alpha Centauri with a planned launch in the early 21st century and 100-year travel time. The work was based upon the requirement stated in the National Commission on Space Report, Pioneering the Space Frontier. One of the components of the Commission's solar and space physics plan is "A long-life, high-velocity spacecraft to be sent out of the Solar System on a trajectory to the nearest star." A large amount of research, development, test, and evaluation is included to allow the emergence of several key and enabling technologies. This will also ensure that the required space operations infrastructure will be in place and operational.

Using purely current technologies, such a mission would be impossible. The main areas requiring advancement are propulsion, power, and command/data processing. The solutions envisioned in this report are: pulsed fusion microexplosion propulsion with 1,000,000 sec specific impulse, a space-rated 300 kilowatt fission reactor with a ten-year active life for power in the target system (as well as for starting the fusion drive), and advanced, artificially intelligent high-speed computers to provide adequate command and control functions.

The spacecraft will be assembled using modular components at the space station. The probe will be boosted to a heliocentric orbit inclined minus 61 degrees to the ecliptic, then to the interstellar trajectory with chemical upper stages. After using the fusion propulsion system to reach the Alpha Centauri system, the probe will enter an orbit

about Beta Centauri.

During transit, vehicle status information and data on the interstellar medium will be returned at low rates. Once in the target system, data will be transmitted at high rates using lasers operating in a hole of the frequency spectrum of the three stars there. Instruments will include packages for studying the stars in the system, any planets that are present, and astrometry.

The probe will consist of a large, truss-like structure that will support four main components: the instrument package, the fission reactor, the fusion drive system, and the fuel tanks for the drive. The fusion drive and fission reactor will be placed at the rear of the truss and be separated from the rest of the spacecraft by thermal and radiation shielding. The tanks will be cylindrical in shape and attached to the sides of the truss (forward of the nuclear systems) in a hexagonal arrangement. Instruments will be positioned forward of the fuel. Particle shields will be installed at both ends of the probe for protection from dust at the high velocities the probe will reach. Fuel tanks will be jettisoned as emptied to enhance the performance of the spacecraft; the fusion drive and particle shields will be discarded upon arrival at the target system.

Excessive power produced by the fusion drive will be used by spacecraft systems so that a dedicated power generation system will not be needed in transit. The fission reactor will provide power for the probe as it orbits about Beta Centauri.

## THE LUNAR ORBITING PROSPECTOR

### UTAH STATE UNIVERSITY

One of the primary rationales for establishing a manned lunar presence is the possibility of utilizing the Moon's resources. The Moon is known to be abundant in oxygen and various metals. Given the known resource potential of only a few explored lunar sites, the possibility of large deposits of these resources and other undiscovered resources elsewhere on the Moon seems highly likely. As such, a continued search for lunar resources and exploration on a global scale in conjunction with a manned, lunar base will aid in full utilization of the Moon's resource potential.

A remote sensing orbital mission, such as the planned Lunar Geoscience Orbiter (LGO), is a necessary precursor to the development of a manned, lunar base. However, the need for a mission of this nature does not end with the establishment of the base. Long-term observation of the Moon, a continued search for lunar resources with new techniques, and continued lunar science studies are paramount to understanding the Moon and fully benefitting from its total resource potential.

#### The Lunar Orbiting Prospector Concept

The Lunar Orbiting Prospector (LOP) is a lunar orbiting platform whose primary mission is to prospect and explore the Moon from orbit in support of early establishment of human presence on the Lunar surface and in site resource utilization efforts. Using LGO as a baseline, the LOP is designed to direct the next generation of lunar exploration in conjunction with an early manned base.

The LOP concept is divided into two

distinct parts; an orbiting platform carrying the remote sensing instruments and a servicing vehicle deployed from the lunar surface. The orbiting platform contains its own propulsion system for orbital maneuvering and remains in lunar orbit indefinitely. The Platform mounts modular remote sensing instruments packages, communications modules, and power systems for ease of repair and replacement. The servicing vehicle, called the Generic Transfer Vehicle (GTV), launches from a lunar base site, rendezvous with the platform for servicing, and returns to the base site. The GTV serves as a "lunar truck" to deliver refueled propulsion modules, and provides servicing of the platform in the event of failure.

The primary purpose of the orbiting platform is to map the chemical and mineralogical composition of the Moon. Remote sensing instruments mounted on the platform probe the lunar surface and subsurface with electromagnetic energy. The data returned from the instruments will give an indication of the mineral and chemical species present and an indication of the subsurface geological structure of the Moon. Through careful processing and examination of this data, lunar resource distribution on a global scale can be determined.

The ability to replace, repair, and upgrade remote sensing instruments is a critical attribute of the LGO. The concept of modularity permits the orbiting platform to be upgraded and modified as needed. The spacecraft can be repaired, refueled, and its instrument packages upgraded to perform nearly an required remote sensing task. Highly specialized lunar remote sensing missions can be per-



formed and/or small lunar orbit experiments flown without tailoring and deploying an entire spacecraft for each application. The value of this concept has been proven in Earth applications by shuttle-based experiments.

### Mission Description

The LOP mission is divided into three primary phases: transport from Earth to low lunar orbit (LLO), operation in lunar orbit, and platform servicing in lunar orbit. Transport in the platform from Earth can be accomplished by a vehicle with a 1000 Kg Earth escape payload capability. This is within the capabilities of the Titan IV rocket or a Space Shuttle/IUS upper stage combination.

After delivery to low lunar orbit, nominal LOP orbital operation commences. Here, the platform alters its orbit to attain the desired surface viewing, and the orbit can be changed periodically as needed. After completion of the initial remote sensing mission, more ambitious and/or complicated prospecting and exploration missions can be contemplated.

### System Description

Figure 1 shows the platform configuration. The overall goal of the configuration is to allow the platform to grow and adapt to new and different science and exploration needs. The base structure of the spacecraft serves to house the propulsion module and mount the required power, control and communications subsystems. The sides of the spacecraft provide 24 instrument drawers for mounting required subsystems and sensor payloads.

Communications is provided by four phased array medium gain antennas mounted on the sides of the spacecraft. These antennae are electronically steered to track relay satellites located at Earth-Moon LaGrange points L1 and L2. The primary remote sensing instrument module is mounted opposite the propulsion module in the nadir pointing direction. This position gives the instruments the required nadir and anti-nadir viewing while providing for unrestricted expansion away from undesired spacecraft thermal and magnetic interference.

### Orbital Considerations

Mission objectives require orbits stable enough to permit orbital maintenance with a reasonable amount of maneuvering but low enough for good instrument resolution. Due to the Moon's anomolous gravity field, stability of lunar orbits appears to be directly a function of altitude and inclination. Experience from earlier lunar flights and known lunar gravitational harmonics indicates that some orbital inclinations and some low altitude orbits are unstable.

### Orbital Strategies

In determining the relationship between surface coverage and operational constraints it was found that a variety of orbits is required to meet. These orbits may vary in altitude, eccentricity, and inclination. Orbital parameters dictate the type of surface coverage obtained and by varying orbital inclination, eccentricity, and periapsis, various remote sensing mission requirements can be satisfied.

Remote Sensing Concepts

Many variations of remote sensing instruments can be flown on the LOP as a result of its modularity. Instruments mentioned represent only a small subset of the possibilities.

Reflectance Spectroscopy

Sunlight reflected from the lunar surface contains absorption lines that are characteristics of minerals present. Mineralogical mapping by this method, combined with data from other instruments, can lead to a much improved understanding of the Moon's surface properties and lunar crust history.

Gamma Ray Spectroscopy

Measuring the Moon's natural gamma ray emissions is recognized as a powerful means for measuring the surface elemental abundances. Gamma ray spectrometers were flown on Ranger 3, 4, and 5 missions and on Apollo 15, 16, and 17 command modules.

Raman Spectroscopy

Incident light can interact with the molecules by either absorption or scattering. If a photon interacts elastically with the molecule, Rayleigh scattering occurs. If a photon interacts inelastically with the molecule, Raman scattering occurs. The distinguishing characteristic of the Raman effect is the shift in frequency that occurs between the exciting energy and the scattered energy. This frequency difference, called the Raman shift, is directly characteristic of the molecule and is generally independent of the laser excitation frequency.

By measuring the Moon's Raman spectra and correlating this with complimentary reflectance spectra, accurate information on mineral and chemical presence and concentration can be obtained. This has become a powerful laboratory technique and has great potential in lunar remote sensing applications.

Propulsion

Since the primary operating expense of the LOP will be propellant cost, this cost can be substantially reduced through the use of lunar derived fuels which do not require costly transport out of Earth's gravity well.

Since the Moon is primarily composed of oxygen, it is an obvious choice for a bipropellant oxidizer and can be extracted from the lunar minerals by many chemical and thermal processes. The critical element of a lunar derived propellant is thus the fuel. Hydrogen is an excellent fuel when used with oxygen but its lunar concentrations are small. Other lunar derived propellant possibilities include, but are not limited to, Silane (SiH<sub>4</sub>/O<sub>2</sub>), AlCa/O<sub>2</sub>, Al/O<sub>2</sub>, AlCaMg/O<sub>2</sub>, Ca/O<sub>2</sub>, and lunar soil/O<sub>2</sub>.

A comparison of lunar derived propellants revealed the following:

	Al/LOX	LH <sub>2</sub> /LOX	LSiH <sub>4</sub> /LOX
Oxid. Mass	1	0.70	0.77
Oxid. Vol.	1	0.70	0.78
Fuel Mass	1	0.78	1.12
Fuel Vol.	1	29.84	4.47
Prod. Time	1	10.86	9.57

Liquid  $H_2/O_2$  gives the best performance of the propellants but production of hydrogen from lunar soils is time intensive and quantitatively

limiting. Silane performs well but production is also time intensive. Aluminum and oxygen thus show the most promise for lunar-derived fuels.

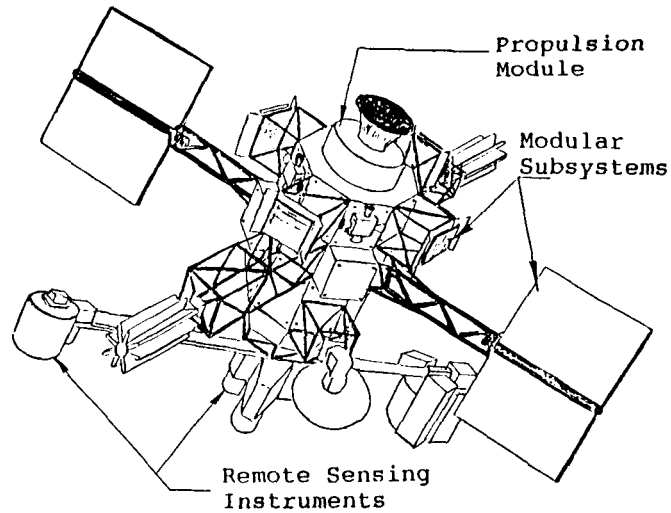


Figure 1 - Configuration

## **A COMPARISON OF TWO PROPULSION MODES FOR AN ORBITAL TRANSFER VEHICLE**

### **UNIVERSITY OF VIRGINIA**

With the advent of the manned space station, the United States now requires a spacecraft to be based at the space station with the ability to deploy, recover, and repair satellites quickly and economically. Such a craft would prolong and enhance the life and performance of many satellites.

As part of the undergraduate design course at the University of Virginia, two groups were set up to create proposals for manned orbital transfer vehicles (OTV). Before beginning the design process the following set of criteria were established as a minimum for the spacecraft:

1. The OTV will be based at the space station.
2. The spacecraft will have the capability of supporting 3 people for a mission lasting no longer than 14 days.
3. The spacecraft will be able to perform multiple missions between low Earth orbit (LEO) and geosynchronous orbit (GEO) with a minimum amount of servicing.
4. The OTV will carry a payload of at least 15,000 pounds between LEO and GEO.
5. The spacecraft will be capable of supporting extra-vehicular activities.

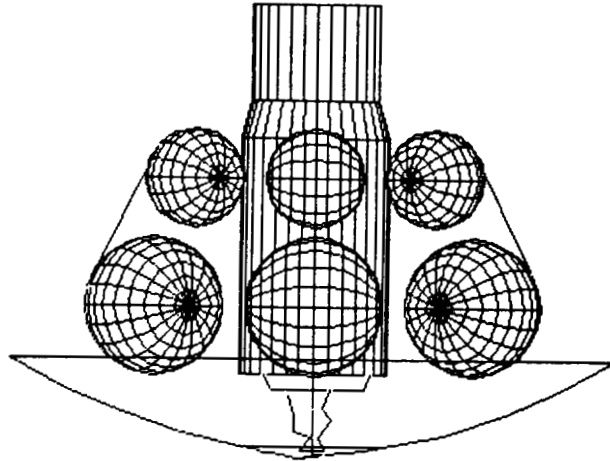
With these criteria, the two groups created different vehicle designs that could meet this need.

One group created Project ORION. The goal of the group was to design a vehicle that would meet the criteria

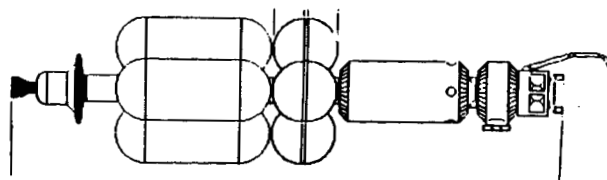
by using a state-of-the-art chemical propulsion system. Performance of the vehicle is enhanced by use of an aero-brake to accelerate the spacecraft on its return to the space station from GEO (Figure 1).

The other group, known as MOVERS, chose to use a high-thrust nuclear propulsion system. Depending on the mass of the payload, the system developed by MOVERS has been shown to result in propellant savings of up to 50-60% as compared to traditional chemical engines, assuming that aerobraking is not used. Environmental analysis also supported the belief that this propulsion system could be used safely in low Earth orbit. However, handling of the nuclear-propelled OTV, when it is docked at the space station, could be a difficult operation. Also, the use of a nuclear propulsion system in near Earth orbit could raise many national and international concerns, and would most likely rule out aerobraking (Figure 2).

The propulsion systems of both spacecraft proved to be the major design constraint. Nevertheless, the designs are very similar in other aspects, with modularity being a key aspect of both designs. This permits each craft to be capable of meeting missions which are more demanding than the one specified in the design criteria. Comfort and safety of the crew were also important design constraints. It was concluded that the OTV with nuclear propulsion could be lighter than one with chemical propulsion performing equivalent maneuvers, and would use less fuel, an expensive commodity when it has to be supplied from the space station.



**Fig. 1 - The ORION chemically-propelled OTV.**



**Fig. 2 - The MOVERS nuclear-propelled OTV.**

## SPACE-BASED LASER-POWERED ORBITAL TRANSFER VEHICLE

### VIRGINIA POLYTECHNIC INSTITUTE AND STATE UNIVERSITY

This project focused on the development of conceptual designs of space-based laser-powered orbital transfer vehicles (LOTV) capable of transporting 16,000 kg of cargo between LEO and either GEO or Low Lunar Orbit (LLO). The work focused mostly on the LEO/GEO missions which assumed that the power to the LOTV is beamed by a single 32 megawatt solar-pumped iodide laser (1.315  $\mu\text{m}$ ) having a transmitting aperture of 30 m and orbiting the Earth at an altitude of one Earth radius with a  $0^\circ$  - inclination. The laser beam is captured by the LOTV primary mirror which redirects the beam into the secondary optical train consisting of three mirrors which refocus the beam into the thrust chamber. The laser engine selected for the LOTV uses hydrogen propellant and an array of window lenses which create seven discrete laser-sustained plasmas resulting in the thermal conversion efficiency (laser-to-jet) of at least 50%. To keep the vehicle oriented with the primary mirror pointed directly at the transmitting mirror of the laser, the vehicle uses three double-gimbaled, control moment gyroscopes and a turntable which can rotate the optical train independently of the vehicle. These systems are controlled by the pointing and tracking system, which along with the reaction control system, interfaces with the guidance, navigation, and control system.

Both all-propulsive and aerobraked LOTV configurations were analyzed and developed. The all-propulsive vehicle uses a rigid primary mirror and its engine produces a thrust of 2000 N at a specific impulse of 1500 sec. The mirror has multi-layer dielectric coatings providing better-than-99% reflectivity at a wavelength

of 1.315  $\mu\text{m}$ . Three options corresponding to three different return payload masses were considered: 16,000 kg, 5,000 kg (standard option) and 1,600 kg. All three options carry 16,000 kg on the LEO-bound leg. Analysis of the laser beam transmission, orbital maneuvers and strategies, and structural requirements indicated the following main characteristics of the all-propulsive standard option LOTV operated between the  $28.5^\circ$  - inclination LEO and the equatorial GEO: a primary mirror diameter of 11.5 m, a dry vehicle mass of 3855 kg, 14,000 kg. of propellant needed for a round trip, and an overall initial mass of 33,855 kg. For the LEO-to-Geo portion of the trip, the payload ratio,  $\frac{m_{\text{payload}}}{(m_{\text{propellant}} + m_{\text{dry vehicle}})}$ , is 1.19 and the trip time is about 6 days. These characteristics indicate greatly improved payload mass fraction capability of the LOTV when compared with the chemical OTV, at a cost of longer trip duration which is perfectly acceptable for cargo transfer.

The aerobraked LOTV version uses an inflatable ballute aeroshield of 20 m diameter and a retractable wrapped-rib primary mirror which is folded for the duration of the aerobraking maneuver. The standard option of the vehicle has a dry mass of 4953 kg and requires about 12,000 kg of the propellant for a round trip.

An operational life cycle cost analysis was performed assuming a 20-year long useful operational life of the LOTV, mission frequency of five or ten per year and major overhauls once every 20 or 30 missions for the aerobraked or all-propulsive configurations, respectively. The analysis showed that the aerobraked configu-

ration has an economic advantage over the all-propulsive one as long as the cost of launching the fuel to LEO is higher than about \$500/kg in current dollars.

An initial study of the aerobraked laser-powered transportation between LEO and LLO assumed that in addition to the solar pumped laser and

two relays orbiting the Earth, a nuclear-powered laser is placed on the Moon's surface. The lunar LOTV uses a 30 m - diameter wrap-rib primary mirror and a 33 m - diameter inflatable ballute aerobrake. For trans-lunar injection, the laser propulsion is supplemented by LH<sub>2</sub>/LOX chemical thrusters which are part of the reaction control system.

## DESIGN OF A RAM ACCELERATOR MASS LAUNCH SYSTEM

### UNIVERSITY OF WASHINGTON

The viability of any large scale permanent space structure relies on the capability of being able to launch mass easily and efficiently into orbit. Present systems for doing so are complex and costly. Therefore, emphasis must be placed on the development of a new cost-effective launch system. The ram accelerator mass launch system has recently been proposed to greatly reduce the costs of placing less acceleration-sensitive payloads into low earth orbit.

The ram accelerator, conceived and experimentally demonstrated at the University of Washington, is a chemically propelled, impulsive, mass launch system capable of efficiently accelerating relatively large masses from velocities of 0.7 km/sec to 9+ km/sec. The principles of propulsion are based upon those of a conventional supersonic air-breathing ramjet; however, the device operates in a somewhat different manner. The payload carrying vehicle resembles the centerbody of the ramjet and accelerates through a stationary tube which acts as the outer cowling. The tube is filled with premixed gaseous fuel and oxidizer mixtures that burn in the vicinity of the vehicle's base, producing a thrust which accelerates the vehicle down the tube. The requirements for placing a 2000 kg vehicle into a 500 km circular orbit with a minimum amount of onboard rocket propellant for orbital maneuvers are examined. The goal is to achieve a 50% payload mass fraction.

As a result of structural considerations, the launch tube inner diameter was fixed at 1.0 m. In-tube propulsive requirements and vehicle structural constraints resulted in a vehicle diameter of 0.76 m, a total length of

7.5 m, and a nose cone half-angle of 7.8. An acceleration of 1000 g's was chosen as the upper limit due to structural considerations on the vehicle and tube wall. An ablating nose cone constructed from carbon-carbon composite was chosen as the thermal protection mechanism for atmospheric transit. The remainder of the vehicle is constructed from titanium alloys and/or composites.

To achieve the desired launch velocity, two modes of in-tube propulsion involving ramjet cycles are required. The two modes that have been investigated are a thermally-choked subsonic combustion mode (Fig. 1a) and a mode which utilizes a stabilized, oblique detonation wave for combustion (Fig. 1b).

As with a ramjet, an initial velocity is required to start the propulsive process. This initial velocity is provided by firing the vehicle into the launch tube using a combustion-driven gas gun. The gas gun accelerates the 2000 kg projectile to a velocity of 0.7 km/s from a standing start using a stoichiometric methane-air mixture.

The subsonic combustion mode (Fig 1a) utilizes thermally choked combustion to accelerate the vehicle from 0.7 km/s to 2.5 km/s. The composition of the pressurized gas mixture is chosen such that the vehicle Mach number is sufficient to ensure that the flow remains supersonic through the throat of the diffuser. The nose cone angle is designed to ensure that the oblique shock system in the diffuser does not initiate combustion. A normal shock is located downstream of the diffuser throat and is stabilized by the heat release which chokes the flow down-



stream of the vehicle. The flow behind the normal shock then expands subsonically to the aft end of the vehicle where combustion begins and proceeds to thermal choking. After reaching a velocity of 2.5 km/s, a transition is made to the oblique detonation mode.

The oblique detonation combustion mode, which accelerates the vehicle from 2.5 km/s to 9+ km/s, requires a strong oblique shock wave to raise the propellant temperature high enough for combustion to occur (Fig. 1b). This is accomplished by means of a small protuberance located on the vehicle to initiate an oblique detonation wave. This mode requires that the nose cone half-angle be small enough to ensure that the bow and reflected bow oblique shock waves in the diffuser do not ignite the propellant.

The analyses for both modes of propulsion were performed using one and two-dimensional gasdynamics to model the flow around the body. The performance parameters of the ram accelerator can be described by the ballistic efficiency (ratio of the rate of change of vehicle kinetic energy to the rate of chemical energy expenditure) and thrust pressure ratio (ratio of effective thrust pressure to peak cycle pressure). These parameters are kept near optimum levels by varying the propellant mixture along the length of the launch tube. Nine different propellant mixtures were utilized in achieving the desired velocity range, each at a fill pressure of 33 atm and a fill temperature of 300<sup>o</sup> K. In this manner the acceleration of the projectile can be kept near the design limit of 1000 g's, at an average ballistic efficiency of 24% and an average

thrust pressure ratio of 14%. Based on the analysis, the required total length of the stationary launch tube is 5.0 km.

Once the necessary launch velocity of 9 km/sec has been obtained, the vehicle must traverse the atmosphere. At such high hypersonic velocities, the nose of the vehicle experiences severe stagnation temperatures, on the order of 9,000<sup>o</sup> - 13,000<sup>o</sup> K, even after considering the effects of equilibrium dissociation across the bow shock wave and in the boundary layer. A carbon-carbon ablating nose cone was chosen to protect the vehicle from the atmospheric heating. Since the drag of the vehicle, being primarily pressure drag, is affected almost exclusively by body shape, the blunting nose cone shape due to ablation increases the drag, thereby increasing the velocity loss.

All analyses were performed between altitudes of 4 km (launch altitude determined by geographical considerations) and 40 km, with launch angles ranging for 16<sup>o</sup> to 30<sup>o</sup>. An altitude of 40 km was assumed to be the edge of the sensible atmosphere due to the low density, the onset of slip flow, and the fact that aerodynamic heating becomes negligible. The 16<sup>o</sup> to 30<sup>o</sup> launch angle constraint is the consequence of parameters specified by orbital mechanics considerations. Results indicate that a launch angle of ~20<sup>o</sup> is most beneficial in regards to meeting design goals. At this angle, the vehicle retains 85% of its original launch velocity and suffers an ablative mass loss of only 1.3%. The drag coefficient increases from 0.058 at launch to 0.11 during the atmospheric transit.

Vehicle instability was investigated by using small perturbation theory. The effect of changing atmospheric density on the magnitude of instability is also of primary concern. The vehicle appears to be highly unstable at launch for any atmospheric disturbance. Angular accelerations on the order of  $164 \text{ rad/s}^2$  at the 4 km launch altitude and  $16 \text{ rad/s}^2$  at 40 km altitude were estimated. Spinning the vehicle or stability augmentation devices, such as control surfaces, to provide the needed stability will clearly be required.

Once beyond the atmosphere, an onboard propulsion system is needed to perform orbital maneuvers to place the vehicle into the desired low Earth orbit. A parking orbit 50 km below the final orbit has been proposed to increase the flexibility of the system. Reducing the onboard propulsion system mass to a minimum results in an increase of the payload mass ratio. With this goal in mind, the feasibility of a multi-step orbital maneuver scheme involving aerobraking was studied (Fig. 2). The proposed set of maneuvers imparts a velocity at the initial ballistic apogee such that the vehicle is placed into an elliptical orbit with a perigee within the earth's upper atmosphere (phase 3a on Fig. 2). An aerobraking maneuver (phase 3b), which uses atmospheric drag to slow the projectile, is then performed to reduce the second apogee altitude to the desired parking orbit altitude.

The total velocity change needed from the onboard propellant at the optimum launch angle of  $20^\circ$  and equatorial launch is approximately 750 m/s. At this angle, the multi-step orbital maneuver with aerobraking allows a 40% savings in the AV

required of the onboard propulsion system as compared to a similar orbital maneuver without aerobraking, thus decreasing considerably the amount of onboard propellant needed.

The orbital requirements of the payload vehicle dictate the necessary velocity change and thrust performance for the onboard propulsion system. The maximum of the mass fraction of payload limits the size and mass specifications of the propellant system. An operable system to meet these requirements using current technology has been narrowed down to a chemical propellant rocket which employs the liquid propellants monomethylhydrazine and nitrogen tetroxide. Solid propellant rockets were also investigated and subsequently discarded since they did not meet the necessary weight and performance requirements.

The onboard propulsion system operates most effectively using a pressurized propellant-delivery configuration, which employs a gas generator and diaphragm equipped tanks. This setup can be expanded to incorporate the necessary attitude control thrusters, operating from the same propellant system. A nozzle designed to fit within the nose cone or tail of the projectile employs an exit diameter of approximately half of the vehicle diameter. This nozzle configuration provides the necessary thrust levels and gives an acceptable specific impulse value for the propellants used. The entire onboard propulsion system, including propellants, constitutes one quarter of the proposed mass of the vehicle, and a slightly greater volume fraction.

Preliminary analysis on the structural design (Fig. 3) of the ram accelerator payload vehicle and launch tube wall were performed using simplified models to determine stress components and required shell thickness. Various structural configurations and their corresponding masses for the vehicle were first examined using composites and titanium alloys. These results then served as a starting point for a more detailed finite element analysis. The pressure distributions on the vehicle used in the structural analysis were obtained from computer simulations of the two in-tube launch modes of the ram accelerator. The payload vehicle must structurally withstand a maximum pressure on the order of 1500 atm (160 MPa) and temperatures on the order of  $3000^{\circ}$  -  $4000^{\circ}$  K for

brief periods ( $\sim 1$  second). The effect of placement of the internal components (such as payload, onboard propulsion system, guidance and controls) on such factors as center of gravity and moments of inertia was analyzed. It was found that using a factor of safety of 1.4, the inert structural mass is approximately 600 kg.

Throughout the design, great diligence was taken to utilize existing technologies in order to determine the feasibility of the ram accelerator as a mass launcher. It has been demonstrated that there is no technological barrier that forbids the development and implementation of this novel concept.

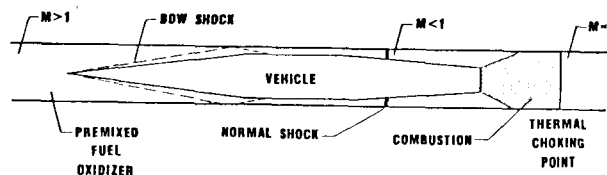


FIG. 1a SUBSONIC COMBUSTION MODE

VEHICLE ACTS AS RAMJET CENTERBODY  
TUBE ACTS AS OUTER COWLING

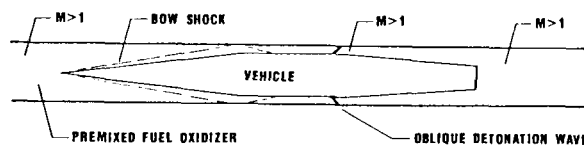


FIG. 1b OBLIQUE DETONATION MODE

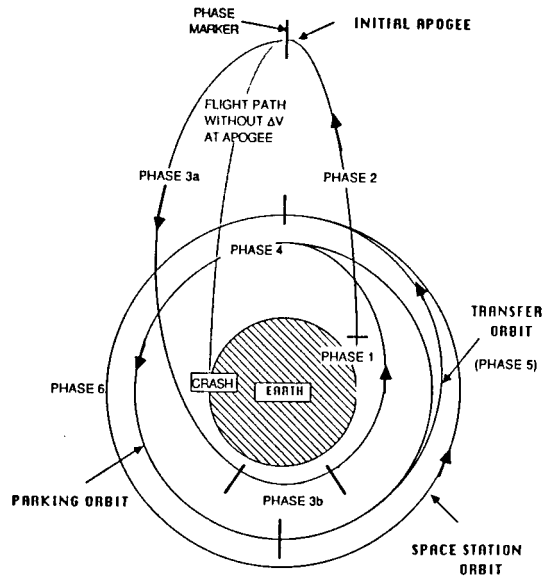


FIG. 2 PROPOSED TRAJECTORY OF THE RAM ACCELERATOR VEHICLE.

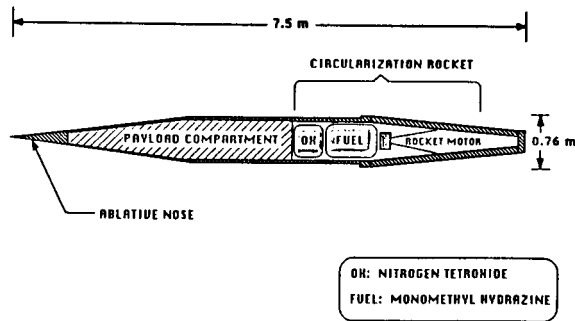


FIGURE 3: CONFIGURATION FOR A GRAPHITE/EPOXY RAM ACCELERATOR VEHICLE

**MARS ROVER/SAMPLE RETURN AND MARS EXOBIOLOGY  
PENETRATOR MISSIONS**

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This report details the work done at the University of Wisconsin-Madison by the Engineering Mechanics senior design teams for the Mars Rover/Sample Return (MRSR) and Mars Exobiology penetrator missions. The work on the rover mission has been divided into four sections: rover mobility, core drilling, a core handling device, and a sample return canister. The rover design is a three-cab, eight-wheel configuration propelled by eight independently-mounted sinusoidal harmonic drives. Figure 1 depicts one of the cabs. The cabs are connected by three degree-of-freedom articulated joints. Particular attention in the design of these joints was given to the capability of having all wheels follow in the same path. In addition to engineering analysis, a detailed one-tenth scale model of the entire rover was produced. A coring drill is attached to the front of the rover and was designed with low power, weight, and low complexity as key design criteria. The drill is capable of a minimum drilling depth of one meter and up to a maximum of ten meters, depending on soil conditions. This is the first component of an entirely integrated sample acquisition handling and storage system. In addition, a core handling device was designed that prepares core drill sam-

ples for scientific analysis. The device removes the outer skin (~10% dia.) of the core and submits samples to various onboard instruments. One unique aspect of the design is that it preserves up to 75% of the core cross sections for placement in the sample return canister. The return canister was designed with an entirely passive thermal control system. The device itself is that of a three-track helical storage arrangement. In addition, the design of the return canister was a major driver for the sample return risk reduction mission scenario outlined in this report.

For the Exobiology penetrator mission, a device was designed to retrieve a pristine sample from outside the altered area around the penetrator body. The area directly around the penetrator is altered when the penetrator descends from its orbiter release and impacts the surface at high speeds (Figure 2). The device drills through the altered area until it reaches a distance of three cm from the outer surface of the penetrator body. The rock material is moved through the drilling core until the desired sample is obtained. Once in the penetrator body the sample is analyzed by a number of scientific instruments.

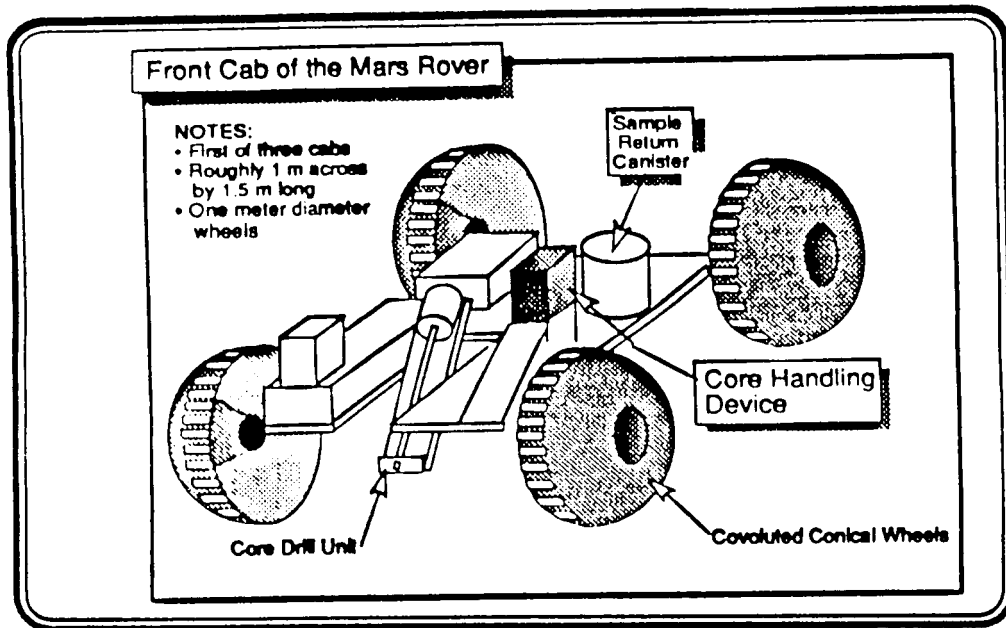


Figure 1

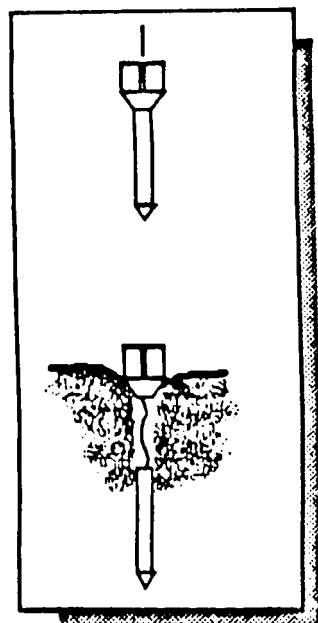


Figure 2

## FIRE SAFETY DESIGN CONSIDERATIONS FOR ADVANCED SPACE VEHICLES

### WORCESTER POLYTECHNIC INSTITUTE

Fire safety will continue to be an important concern aboard spacecraft used to perform missions well into the 21st century. Reduction of risk through prevention, control, extinguishment, and recovery will be essential to maximize mission success. In situations such as prolonged space flight, or habitation, where a fire incident can cause disruption or possible mission failure, designs which minimize the chance of fire should be emphasized.

Ignition and combustion in microgravity are significantly different than on Earth because flow due to buoyancy is not present. Given that there will be potential ignition sources in spacecraft environments, fire safety designs require that the energy necessary to cause ignition of a given material in a low gravity environment be known. It has been proposed that during the ignition process, the absence of buoyancy characteristic of a microgravity environment will result in the accumulation of a layer of pyrolysis products (fuel vapors) at the fuel surface. Further, the absence of gravity will reduce or eliminate the normal air circulation patterns seen in normal Earth gravity. It is this circulation that permits the mixing of air with pyrolysis products to form the correct fuel to air ratio necessary for ignition. This circulation also cools the pyrolysis products by entrainment and affects the ignition time. The confinement of the fuel vapor and the lack of circulation may result in less energy required for ignition, while the reduced availability of air may have the opposite effect.

In order to develop safe designs with respect to fire protection, the ignition

process in space needs to be compared to the ignition process on Earth and correlations formed. An experiment to study microgravity combustion (Fig. 1) is being readied for flight onboard the space shuttle as part of an experiment package in NASA's Get Away Special Program. By conducting identical experiments on the ground and comparing results with data generated by the flight experiment, a better understanding of the ignition process in microgravity is possible. This will eventually lead to safer spacecraft designs.

To study the ignition process, an experiment was designed and built as shown in Figure 1. A high intensity lamp connected to a battery was used as a radiant heat source to ignite a paper sample. The paper used was alpha cellulose, chosen for its uniform burning characteristics and obtained from the National Bureau of Standards. A radiometer located flush with the sample measures radiant energy flux as a function of time (Fig. 2). The curve is integrated over time from  $t_0$  (lamp on) to  $t_1$  (ignition), yielding the total energy required for ignition. Thermocouples were used to measure temperatures at select locations during the experiment. The power to the lamp was terminated once a flame was detected by a flame ionization sensor.

In an attempt to predict the energy required for ignition in space, ground-based experiments are being performed with the apparatus in various orientations (Fig. 3). The orientation displayed in Figure 3a minimizes the effects of buoyancy since the fuel vapor has nowhere to escape and will remain in the vicinity of the sample. This is expected to be similar

to the process in space. Alternately, the configuration in Figure 3b maximizes gravitational effects and these results will be used for comparison purposes.

design effort will aid in explaining the differences in combustion on Earth and in space. With this new information safer space vehicles and habitats may be possible.

It is hoped that the findings from this

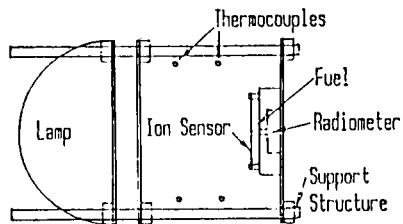


Fig. 1 Experiment Schematic.

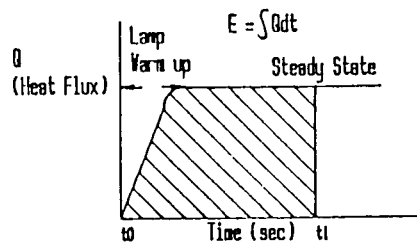
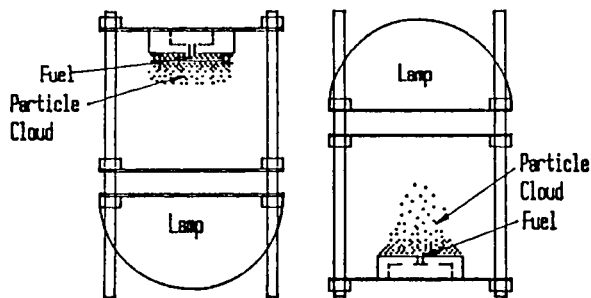


Fig. 2 Flux vs. time during experiment.



a) minimum effect      b) maximum effect

Fig. 3 Orientations to vary effects of gravity for on ground testing.



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