1990/91 PROJECT SUMMARIES

UNIVERSITY OF CALIFORNIA, LOS ANGELES

This document summarizes four projects carried out in 1990/91 at the University of California, Los Angeles, under NASA/USRA sponsorship. One of the projects described is a mission design, the three others include the design and construction of space-related hardware and testing.

MANNED VOYAGE TO MARS WITH PERIODIC REFUELING FROM ELECTRICALLY PROPELLED TANKERS

Twenty-four UCLA students, in groups of four, participated in a mission design for a manned expedition to Mars based on the concept of midcourse refueling from electrically propelled tankers launched ahead of the manned mission. The study was conducted during the 1991 spring term.

Some of the student groups opted for non-nuclear propulsion of the manned ship, based on LOX and LH_2; others opted for one based on nuclear-thermal propulsion. The first option is illustrated below.

Electric thrusters, such as the already well-developed ion engines of the electron bombardment type, can have a very large specific impulse, but, for realistic levels of electric power, they have low thrust resulting in very long travel times. In this mission analysis, it is proposed to combine their advantage (high I_{sp}) with the advantage of chemical propulsion (high thrust) by midcourse refueling of the chemically propelled, manned ship from electrically propelled, unmanned tankers.

The tankers, which would be orbiting for periods of from three to eight years, would be launched a corresponding number of years before the launch of the manned ship. In addition to their own propellant (e.g., liquid argon), the tankers would carry a much larger quantity of LOX and LH_2 for transfer to the manned ship. In the present proposal, the tankers' electric power would be provided by a 2- to 5-MW_e nuclear reactor with, for instance, a potassium Rankine-cycle power converter. Boiloff of the cryogenic propellants would be recondensed by sorbent pumps using the reactor's waste heat.

Refueling the manned ship n times is equivalent to an $(n + 1)$-fold increase in I_{sp}. Because of the very high I_{sp} of the tankers, the total mass that must be assembled in LEO is greatly reduced.

A second feature that may or may not be applied to such a mission is to produce all the LOX, even that for the initial fueling, either from lunar soil or, alternatively, from the martian atmosphere. In the latter case, the tankers would start from LEO with only hydrogen, land on Mars, autonomously manufacture the LOX, ascend to a low altitude orbit about Mars by expending a relatively minor amount of LOX and LH_2, and return to LEO (or to orbital matching for a midcourse rendezvous with the manned ship). The advantage of this scenario is that the difference in total energy (gravitational plus kinetic) per unit mass is 3.3×10^7 m^2/s^2 for ascending from the ground to LEO (assumed here and in what follows as the Space Station *Freedom* altitude) vs. only 6.6×10^6 m^2/s^2 for the ascent from the Martian surface to a low Mars orbit (assumed as 200 km altitude). Because of the high I_{sp} of the tankers, the transport from Mars vicinity to Earth vicinity is sufficiently efficient in propellant usage to reduce by a major factor the total mass that must be brought up to LEO.

Ordinarily, low thrust, electrically propelled spacecraft are intended to apply thrust parallel to the instantaneous flight path. This results in a spiral path, nearly circular at all times, about the astronomical body. However, such a path will not allow rendezvous with a manned ship that is on an efficient, short-travel-time trajectory. In this case, it is advantageous to put the entire burden for the needed matching of position and velocity on the tankers. This can only be accomplished by applying thrust at an angle to the tankers' flight path (except at the periapsis where the angle is zero). For the trajectories of interest, it can be shown that, as a consequence of thrusting obliquely to the flight path, roughly half of the ΔV is lost.

Figure 1 illustrates a mission that provides for 58 days stay-over on Mars, with a 204-day round-trip time. As indicated in Fig. 1, the manned ship trajectory is at first depressed below the Earth's orbit, thereby gaining enough speed to compensate for the lower mean angular velocity of Mars. The velocities indicated are those pertaining to the heliocentric, nonrotating reference frame.

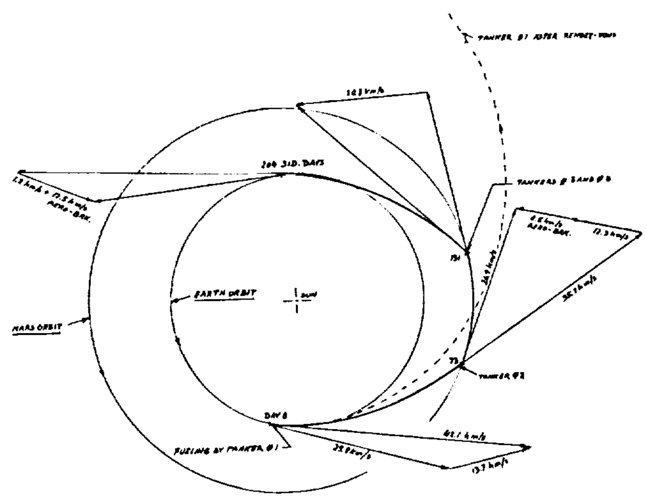

Fig. 1. 200-day mission to Mars, with refueling from electrically propelled tankers

Five tankers are required (four, if LOX is transported from the ground to LEO). The first load of propellant is sufficient for Earth escape on a hyperbolic trajectory. The next fueling takes place in close proximity to Earth.

A mass of 35,000 kg for the Command/Service Module of the manned ship is assumed (the Mars Lander and equipment left on Mars are assumed to be already in low Mars orbit, transported there by one of the cargo ships). Propulsion of the manned ship is assumed to be by LOX/LH_2 with a vacuum I_{sp} of 480 s (based on a mixture ratio of 7.0 and a large extended exit cone). Aerobraking into the Mars atmosphere with $\Delta V = 8.5$ km/s, and into the Earth's atmosphere upon return with $\Delta V = 12.5$ km/s has been assumed. The ΔV requirements for the several flight events and masses for the manned ship ascent to Mars are listed in the Table 1. (Similar values also apply to the return trip.)

For the tankers, an I_{sp} of 16,000 s and an electric-to-ion energy efficiency of 0.80 is assumed. These parameters are achievable with current technology. The nuclear reactor power plant, including power conversion, radiator, and the minimal shielding required for an unmanned vehicle, is sized at 4.2 MW_e. A mass-to-power ratio of 8 kg/kW is assumed. Typical operating times are three to eight years, depending on the rendezvous location and velocity. This results in a combined powerplant and ion thruster propellant mass of 68,000 kg, or about 20% of the total tanker mass (including both LH_2 and LOX). The thrust is 40 N.

In LEO, the mass of each tanker (with LH_2, but without LOX, which is assumed in this study to be autonomously manufactured on the Moon or on Mars) is given in Table 2. The total mass in LEO per mission is 600 metric tons for five tankers and the Command/Service Module.

Table 1. ΔV Requirements.

ΔV km/s	LOX/LH_2 Mass, metric tons		Total Mass, metric tons		Event
	initial	final	initial	final	
3.5	247	99	282	134	Earth escape at parabolic speed
6.3	99	0	134	35	tanks are emptied
0	0	247	35	282	refueling
7.4	247	24	28	259	Earth escape at 13.7 km/s (solar ref.)
2.5	24	0	59	35	tanks are emptied
0	0	247	35	282	refueling
9.8	247	0	282	35	retro thrust at Mars

Table 2. Tanker Masss.

Component	Mass,metric tons
LH_2	27
Ion thruster propellant	34
Powerplant	34
Structure, etc	18
	total 113

A drawback of midcourse refueling by electrically propelled tankers is the narrow launch window for the manned ship. So as not to miss the rendezvous in case of launch delay, the tankers must be provided with some chemical propulsion capability, comparable to the shuttle OMS engines. Thus, to provide for a launch window of six days, an additional ΔV capability of about 5% would be needed.

Mission safety can be increased by providing for more than the minimum number of tankers, with the goal of making possible a safe abort during any part of the mission, even though one particular refueling may have been missed. For subsequent missions, it may be possible to use tankers that served as backup in earlier ones. For example, the zero-thrust orbit of the tanker intended for the first refueling is shown in Fig. 1 (dashed line). Its orbital period about the Sun is 2.7 years.

A considerably lower LEO mass than the 600 metric tons estimated could be achieved if the stay-over time on Mars is reduced to, say, 15 days.

GROUND SPEED OF BALLOONS ON MARS

In an international mission to be launched to Mars in 1996, Soviet-built balloons in the planet's atmosphere will carry scientific instruments in a gondola and also in a drag rope (the Snake). During the martian night, when the balloons are cold and descend, the Snake will be in contact with the ground. Instruments, such as a gamma ray spectrometer and an instrument to measure ground speed, have been proposed for incorporation into the snake.

In cooperation with the NASA Jet Propulsion Laboratory and the Planetary Society in Pasadena, California, a student-designed prototype instrument and drag rope have been built. The instrument is intended to measure the approximate speed with which the rope slides over the terrain. When traveling at several meters per second over rough, rocky, and sandy surfaces, the rope will be bouncing and flexing with high accelerations. In addition to withstanding the harsh martian environment, the rope must not snag between rocks or other formations; thus, no instrumentation can be allowed to protrude from the rope's smooth surface.

The device designed and built by students at UCLA is similar in concept to one designed by students at the University of Utah, who are also participants in the USRA Advanced Design Program.

The principle of operation is illustrated in Fig. 2. Inside the guide rope are two pairs of accelerometers, separated by a distance along the guide rope of approximately 2 m. In each pair, the accelerometers are oriented orthogonally to each other and orthogonal to the length of the guide rope. When sliding over and around rocks, there will be a time lag in the voltage outputs from the rear accelerometers with the corresponding outputs from the accelerometers in front. The speed of the guide rope over the ground can then be inferred from forming the convolution integral of the accelerations

$$C(t,\lambda) = \int_{\tau=t-T}^{t} a_1(\tau a_2(\tau - \lambda)d\tau$$

where a_1 is the voltage from one of the rear accelerometers, a_2 the voltage from the corresponding front accelerometer, λ the (unknown) lag time, and T the "window time" (the time interval for which the data are kept in memory by the computer). The best approximation to the true time lag is found by computing the centroid, or weighted mean, of $C(t,\lambda)$ as a function of the time t. The speed of the guide rope is found by dividing the known accelerometer distance by the time lag.

To check the theory of operation of the instrument, the students developed computer programs that calculated the best value for the guide rope speed for assumed ground profiles. These included one or two "rocks," represented as Gaussians, with a superimposed ground roughness, represented by a pseudorandom function.

At the time of writing, field tests with the student designed device are in progress on the UCLA campus (Fig. 3).

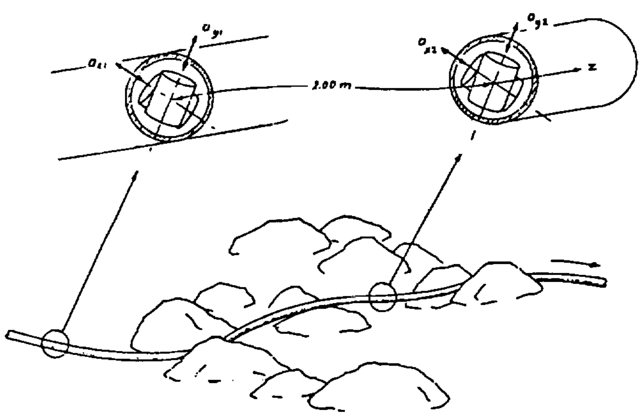

Fig. 2. Drag rope speed measurement by accelerometers

TRIPOD LUNAR LANDING STRUCTURE

In this project, the students were given the task of designing, building, and testing a simple tripod truss of their own design, which was to simulate on a small scale a lunar landing structure.

The specifications were as follows. The tripod had to support a vertical load (the space vehicle) at three strong-points forming a horizontal equilateral triangle with 15-in sides. At its base, there were to be again three strong-points (the landing pads) forming a horizontal equilateral triangle with 36.6-in sides. The height of the tripod was prescribed as 18 in.

The students were encouraged to invent their own truss structure that would connect the top three strong-points to the lower three. They were given the choice of two sizes of low-cost, seamless tubing (1020 carbon steel, 30,000 psi yield strength, 48,000 psi ultimate strength, Young's modulus 29,100,000 psi). One size of tubing had a diameter of 1/2 in, with 0.035-in wall thickness; the other had a 3/4-in diameter with the same wall thickness. The total weight of the structure could not exceed 7.50 lb.

A great variety of structures, one of them shown in Fig. 4, were proposed by the students. They were then asked to make all necessary calculations to predict the maximum load at yield that would be sustained by their design. After completion of the designs and calculations, the students cut and fitted the steel tubing. (Welding was done by a professional welder.) Each structure was tested on a simple testing machine made from an existing hydraulic press and a carefully calibrated load cell.

The entries were judged on two points: (1) the ratio of the maximum sustained load (at yield) to weight; and (2) the percentage of the difference between the maximum load as calculated by the student and the actual one measured on the testing machine.

The structural failures encountered by the students' designs were of several different types: They included Euler buckling, thin wall local buckling, exceedance of the yield strength in a tension member, and overall structural instability (violation of the rules governing the rigidity of a truss). The largest load sustained by any of the structures was 4360 lb.

Fig. 3. Field test of "Snake."

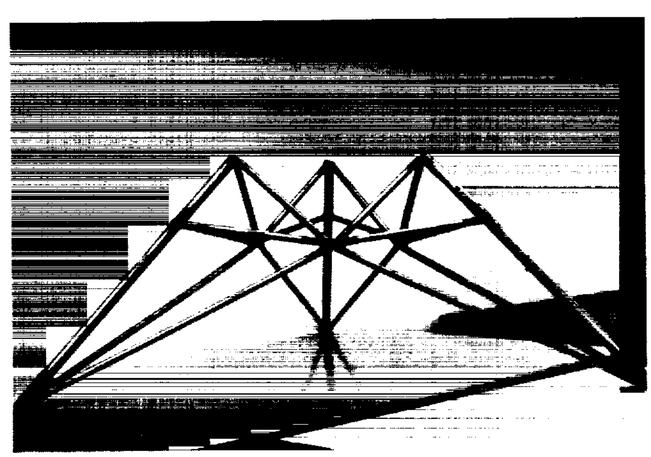

Fig. 4. One of the tripod landing structures designed, built, and tested by the students.

GRAPPLING/DOCKING DEVICE ACCOMMODATING LARGE LATERAL AND ANGULAR MISALIGNMENTS

In space operations, the mating of two space vehicles is a necessary and ever more frequent maneuver. Docking mechanisms and procedures in use today work well and are usually reliable, but allow for only quite narrow margins of angular and lateral displacements. They require intensive active control, usually by a human operator.

Several methods of joining spacecraft in flight have been developed in the past. The first method employed by the U.S. space program was the relatively simple method typified by the Apollo missions. Although reliable, it required very close control and sometimes necessitated repeated attempts before a successful docking was accomplished. The second method, still in use today in the shuttle orbiter fleet, uses an articulated arm that places the female section of the capture system over the male section. The female section then locks onto the male section, completing the capture and allowing the arm to maneuver the grappled vehicle into the desired position. This method, which has proven to be very successful, is well suited for the capture of an inactive payload module by a manned vehicle that supplies the necessary control signals.

A different approach was used in the (now canceled) Orbital Maneuvering Vehicle (OMV) program, with the OMV to be guided and docked by remote control. This approach was inherently problematic because of the large speed-of-light delay between initiated control action and verification by the controller of the actual event.

In one of the USRA-sponsored design classes at the UCLA, the students developed several ideas for a new docking mechanism that would allow a large lateral misalignment (of the order of one half of the radius of the grappling space vehicle) and an angular misalignment of at least 20°.

The first of these designs involved a deployable capture cone on one of the vehicles, and a long flexible probe on the other, an approach resembling the one used in aircraft refueling.

The second design investigated consisted of a circular capture plate (the end plate of the vehicle) with a hooked lip around its edge and a collapsible tripod on the other vehicle. In the docking maneuver, the legs of the tripod are collapsed together as one or several of them strike the capture plate. The force of the impact forces the legs apart where they lock into place under the lip. The dynamics of this mechanism, however, proved to be very difficult to analyze in any detail and was beyond the scope of this class.

By comparison, the analysis of the dynamics of the cone-probe docking system proved to be relatively simple. The problem was modeled as a bending spring, telescoping outward from the probe-carrying vehicle and telescoping back into it as the docking progresses. From the spring and damping characteristics, the forces on the two vehicles were calculated as functions of the relative angular and lateral displacements and the relative velocities of the two vehicles. To simplify this first analysis, the heavier of the two vehicles was treated as fixed, while the lighter vehicle was free to translate and rotate, depending on its mass and moments of inertia.

The students then designed models of the two spacecraft to demonstrate the cone-probe docking mechanism. Capture in these models is accomplished autonomously by the activation of solenoids that grip the probe and an electric motor that retracts the probe to bring the two vehicles together in the final phase of the docking.

This design is presently being constructed by the School of Engineering machine shop and is expected to be an exhibit at the 1992 USRA Summer Conference.