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GEOSEL TRANSPORT SYSTEM (G.T.S.)

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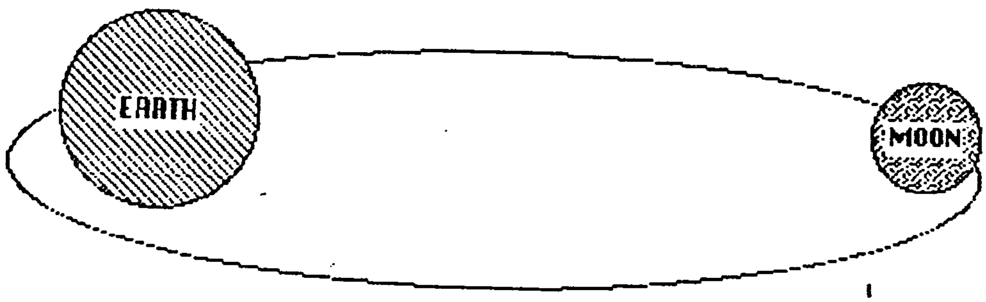

ABSTRACT

In order to meet the requirement of logistically supporting planned lunar activities a reusable three component transport system that moves various payloads from lower earth orbit to the moon's surface is proposed. The system consists of cargo containers, a trans-orbital tug that transports four payload containers from an earth parking orbit to a similar orbit about the moon, and a lander that brings the containers to the lunar surface one at a time.

The first system requirement was shuttle compatibility. Since the containers are to be brought into orbit by the STS, their maximum weight and size as well as mounting pins were fixed by the shuttle. No further design aspects of the container were considered outside of these three parameters.

The second component of the system, the tug, is the heart of the system and was the focus of the design effort. The energy needed to transfer four containers to lunar orbit caused a considerable change from the original concept of the tug. The high energy requirement dictated that the tug fuel tanks be prohibitively large. The re-design of the spacecraft showed that by staging the vehicle the tankage problem could be solved. The propulsion and tankage

requirements, the structural demands, and a detailed weight estimation were the major design aspects given the most detail in the tug design. Other tug components, such as the crew and power module, attitude control system, and communication systems were taken from either already proposed spacecraft design, such as the one for the mars mission (ref. 2), or from already existing design such as the space shuttle. The two major components taken directly from the space shuttle are the four SSME's used to propel the tug and the shuttle's remote manipulator arm that will be used to transfer the payloads from the tug to the lander while in lunar orbit.

As far as the lander is concerned, time constraints did not allow for more than a preliminary design overview. Initial weight, thrust and tankage estimates were made. From this, a drawing was made of a possible configuration for the lander.

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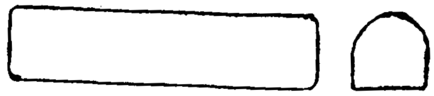
CHAPTER 1 - CONCEPT FORMULATION

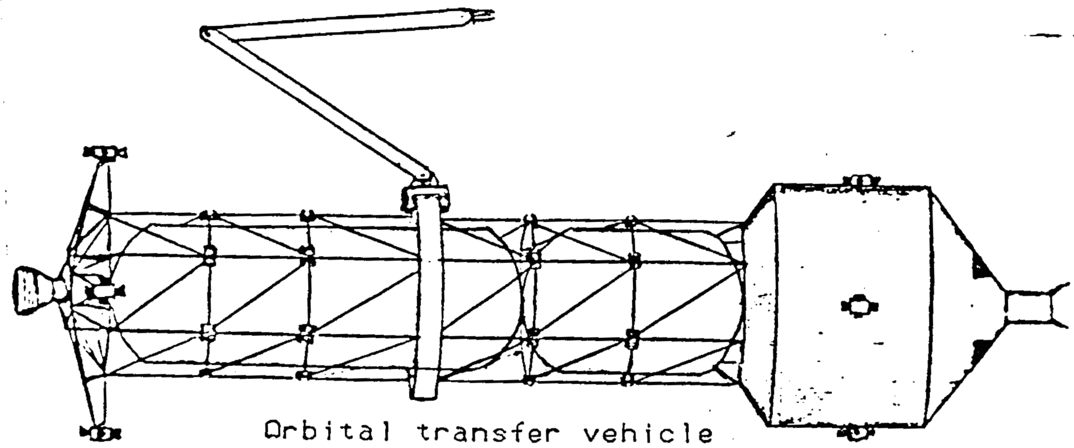
Currently NASA is considering the possibility of the next step beyond the space station - a permanently manned lunar base. Such a base would require logistical support as well as a means of initial transportation to the moon. For a base of any size and duration a substantial transportation capability would be necessary. The space shuttle now allows western countries to carry payloads into lower earth orbit, but no further. There is therefore a need for a lunar transportation system, similar to the shuttle.

It is proposed here that a system for supporting a permanent base on the moon is quite feasible and can be a reality in relatively a short period of time. Such a system would be largely based upon the space shuttle and many of the systems integral to it, allowing for use of proven systems with minimal additional design.

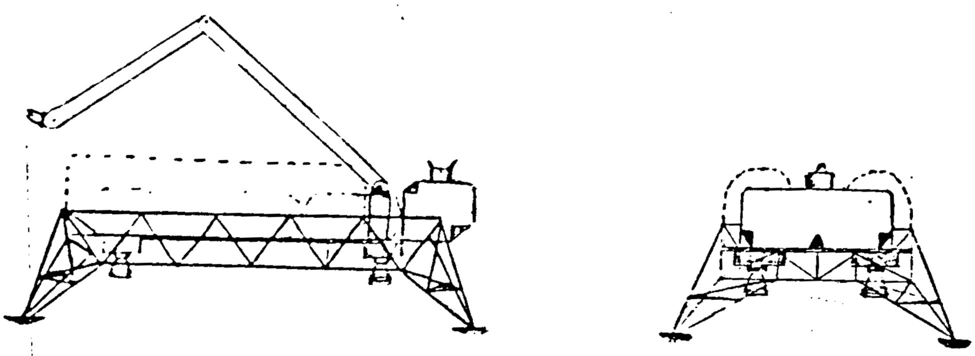
The system that is proposed would be made up of essentially three components: a trans-orbital vehicle, a lunar lander and a generic cargo container (Fig. 1-1). The simplified mission profile would begin with a low earth orbit rendez-vous between the trans-orbital vehicle (or tug) and the space shuttle or its derivatives. Next would follow the flight to the moon, and a rendez-vous there with the lunar lander (Fig 1-2). At each rendez-vous payloads would be exchanged. The lander would then independently fly to the surface of the moon to unload its cargo. Cargo could then be returned to the earth in reverse order.



Container built to
Shuttle specs



Orbital transfer vehicle
without the two annular rings
of containers attached



Lunar landing vehicle
(the two container positions
are shown by the dashed lines)

fig.1.1

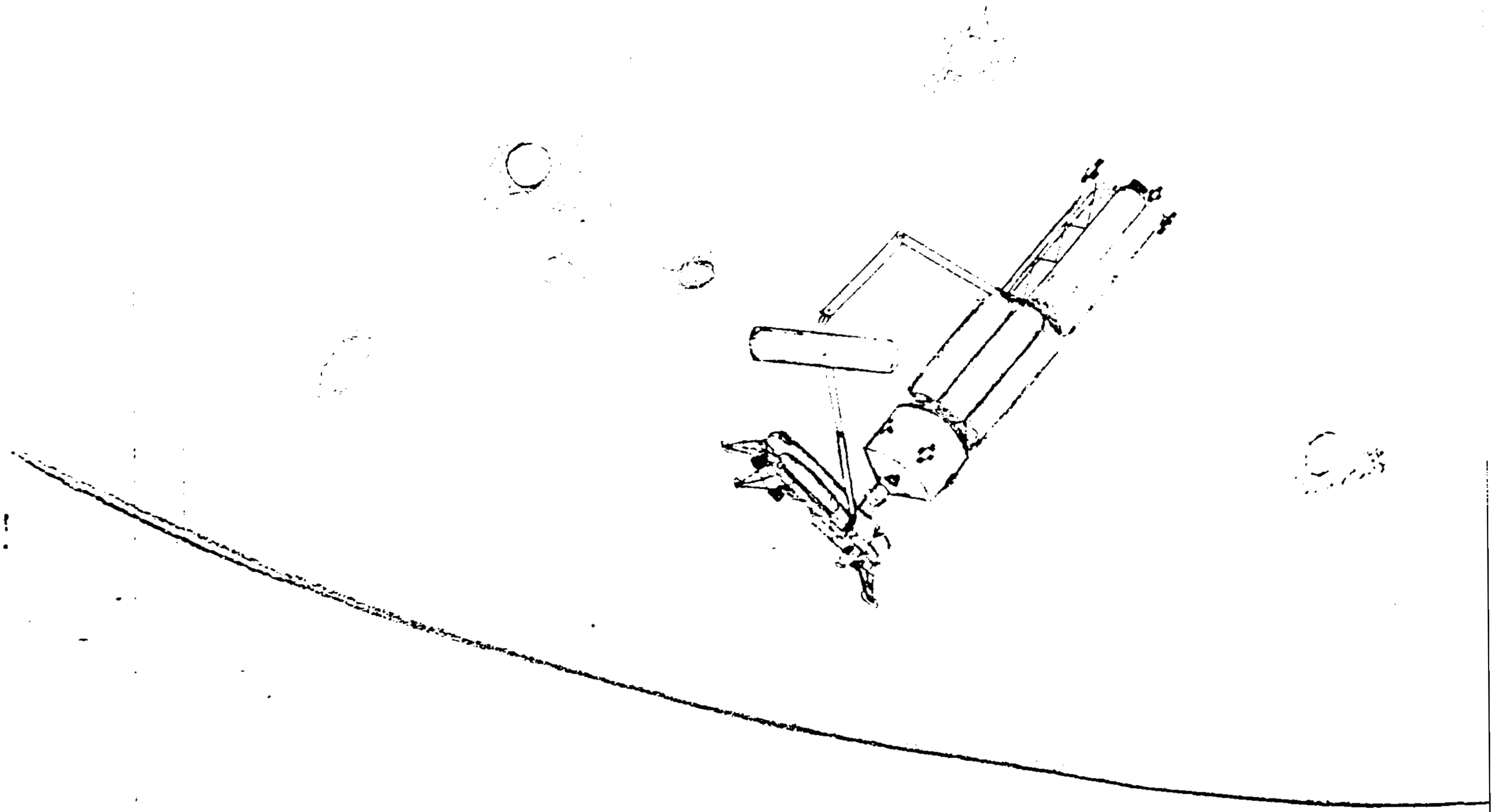


fig. 1.2

CHAPTER 2 - PRELIMINARY DESIGN

2.1 BASIC ASSUMPTIONS

As a starting point in the development of the space tug and lunar lander system, the following general system assumptions are made: first, the construction and assembly of the three components is complete and the system is fully operational, functioning in the intended environments. Next, the altitude of the earth orbit will be set at a specific distance in order to interact with the space station and other low earth orbit facilities. Also, all orbit calculations will be made ideal by considering only point masses, a smooth spherical lunar surface, and the presence of no orbital perturbations. Finally, the transport vehicles will be manned as well as reusable.

2.2 SYSTEM COMPONENTS

The first system component consists of the container, analogous to a box car on a train. No design consideration beyond the size, weight, and the means of attaching them to the transport vehicles will be made. The size and weight will be determined, based on Space Shuttle compatibility.

The next component, designated the tug for design purposes, will carry the containers from earth to lunar orbit and back. The aspects to be determined will include the type of propulsion and fuel tankage requirements, the attitude control system, crew requirements, structural materials, and structural design. Along with these aspects size, weight, and possibly cost estimation will be made.

The final component, the lunar lander, will transport the containers between the lunar surface and the parking orbit of the tug. The design considerations of the lander will be the same as those of the tug. The additional assumption that the landing site characteristics will not affect the lander's capabilities will also be made.

2.3 COMPONENT INTERFACE

In addition to these specific aspects of the individual components, the way in which the two vehicles will rendez-vous and transfer the containers will be examined. Some sort of remotely manipulated arm will be used to facilitate this requirement.

With these assumptions and basic design requirements made, a simplified design approach is intended for all components of the system. The final report, which has completed all tasks proposed should make an excellent starting point for actual detailed design and implementation of an earth to moon transportation system.

CHAPTER 3 - ENGINEERING DEVELOPMENT

3.1 THE ORBITS

As with every spacecraft design, a sequence of steps in a specified order must take place to ensure all requirements have been met. The choice of an orbit for the space tug and lander system was the first step. From the choice of orbit the energy needed and mission lifetime was established. There were three parameters that drove the choice of the outbound orbit: energy requirements, the error tolerance at the pereselenium altitude, and the time of flight. The altitude tolerance was arbitrarily set to fall within a two sigma disperson or to within 98% of the target value. Once this was fixed, possible solutions were examined, weighing time of flight verses the energy required to accomplish the mission.

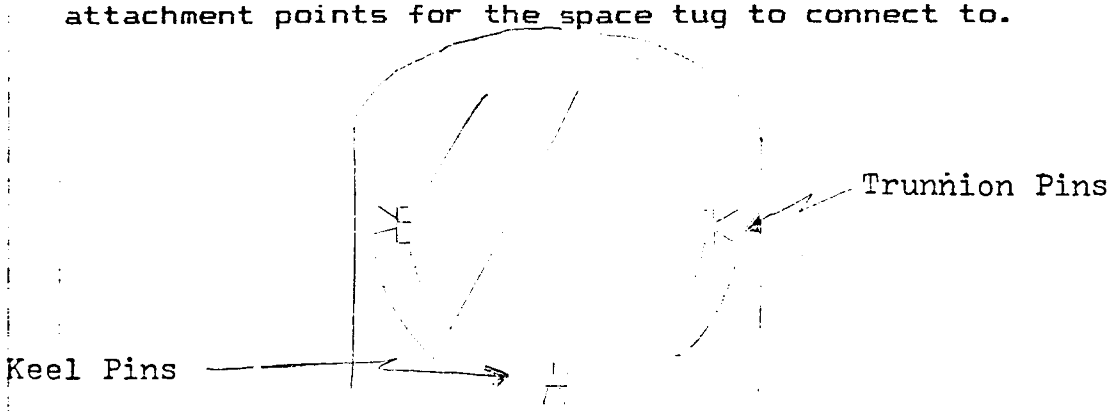
From the large number of possible orbits that fell within the lunar altitude requirements, two were chosen as fulfilling all three requirements. The first solution provided the best mission time for the least amount of energy expended. A time of flight of 2.7 days for a modest delta V of 4.167km/s was obtainable. However, this delta V placed a severe load on the propulsion system and eventually the fuel tankage requirements became too restrictive.

Because of the fuel requirements, it became necessary to look into the orbit that allowed the minimum expenditure of energy. This orbit reduced the delta V required to 3.935 km/s, yet provided an acceptable 4.7 day time of flight. This was the orbit chosen for the earth to moon transfer.

Since the time needed to transfer from the lunar parking orbit to the surface of the moon was less than an hour, the energy required was the only driving factor in the choice of the lander's orbit. The minimum energy transfer orbit is a Hohman transfer as was used to an altitude of twice the highest lunar mountain peak or five kilometers. At this point the lander will transition to a highly elliptical orbit for the final descent to the surface of the moon. The orbit eccentricity was fixed at .96 in order to provide an acceptable vertical to horizontal velocity ratio at touch down on the moon. The final energy requirement placed upon the lander was to provide for an abort orbit. With this requirement, the delta V needed was 3.124 km/s and a time of flight of .95 hours.

3.2 THE PAYLOAD CONTAINER

With the requirement that payload containers be shuttle compatible, a cylindrical container 60 feet in length and 14 feet in diameter was developed. As shown in figure 3-1 a payload clearance envelope of 15 by 60 feet in the STS's cargo bay permits sufficient room for a container to fit inside. The placement of keel and trunnion pins on the containers, as shown below, provides compatibility with the shuttle cargo bay's keel and trunnion tracks and are effectively used as attachment points for the space tug to connect to.



3.3 PROPULSION CONSIDERATIONS FOR THE SPACE TUG

After deciding upon a mission and a basic structure, a propulsion system was needed. Several choices were available for this system, or would be in the near future. These were solid and liquid propellent chemical rockets and non-chemical rockets such as nuclear heated rockets, electrostatic rockets, and electrothermal thrusters.

Electrothermal thrusters, which are also called arc jets, are simple and operate with a single propellant substance and electric current. These are limited, however, by the onset of high dissociation losses and thermal losses to the nozzle walls. Exhaust velocities are limited presently to 17,000 m/s and as a result, they are good for missions with small velocity changes, but are not suitable to this mission.

Electrostatic rockets offer the advantage of having exhaust streams that can be steered using electric fields. The exhaust stream can also be contained and shaped using electric fields which would greatly reduce the problem of solid-surface erosion. This device is limited in high energy devices by the difficulty of producing adequate thrust per area. This is because the required large area of the emitter restricts the beam flow rate by interfering with itself. The device also requires heavy propellants such as cesium or mercury for higher thrusts. Electrostatic rockets are not able to produce the power required for the mission at an acceptable power-to-mass ratio.

Nuclear rockets are fairly efficient and can use a variety of propellants. They operate by passing the propellent through heat exchanger passages within a nuclear reactor and then releasing them through a nozzle. The exhaust temperature is much lower than that of chemical rocket because it is limited by the structural temperature limits within the reactor itself. The most efficient propellant is, therefore, one that can give the maximum possible specific enthalpy for this limiting temperature. Specific enthalpy is nearly proportional to the inverse of molecular weight, so molecular hydrogen is the best choice, but other propellants can be used if necessary. Using hydrogen, the exhaust velocity can be made as much as twice that of chemical rockets. Although it requires massive shielding, this device is a very viable option for the GEOSEL mission.

Solid propellant rockets have good thrust-to-weight ratios and it is currently possible to design these rockets to produce the velocity changes necessary for the GEOSEL mission. The difficulties arise when the mission profile is considered. The mission requires a throttled device and one that can be shut down and restarted. This is not possible now and, although it may be in the future, solid rockets were not considered for this mission.

The remaining option is the liquid-propellant rocket. Of these, the best choice is the liquid oxygen-liquid hydrogen combination. A liquid propellant rocket system allows the propellant to be brought up to the spacecraft in portions, and is also available in "off the shelf" designs now. With a throttleable engine, the problem of high burn-out acceleration is avoided, and the thrust can be tailored to the payload requirements. The problem with this rocket is that a first stage system requires very large fuel volumes.

Both the liquid oxygen - liquid hydrogen and the nuclear heated rocket systems are viable options for the space tug. The nuclear rocket must be shielded and is very bulky itself. The LOx - LH2 rocket requires large fuel tank sizes for a first stage. Each system weighed against the other has its advantage and disadvantages and either could fulfill the mission requirements, but because of the added difficulties with the nuclear system, the liquid - propellant rocket was chosen.

3.4 SPACE TUG DEVELOPMENT

Once the orbits had been established the actual design of the space tug was begun. Given the mission specifications, the first design consideration was to make a space tug that would be single stage and capable of delivering a sufficient number of payloads to make shuttling from lower earth orbit to the moon's orbit a feasable method of supporting various lunar activities. Figure 1-1 was used as a starting point in the tug's design. In considering the space shuttle's capability of carrying a payload of 65,000 pounds, four containers were chosen as the total payload of the space tug. This amount of payload is more than sufficient to support lunar mining, manufacturing, or colonization of a large scale.

The process of designing the space tug involved an iterative process that required the design specification of a number of components. Specifically, the spacecraft's propulsion, propulsion tankage, structure, and any other aspects affecting weight and sizing needed to be considered. As a starting point in the iteration process, a main structure was chosen that consisted of a truss that would surround the propulsion tankage and also support containers attached to the exterior of the truss (Fig. 3-2). A crew module and a power module would attach to this structure at the forward end of the truss. The power module would fasten to the aft end of the crew module and the power module would attach directly to the truss framework (Fig. 3-3).

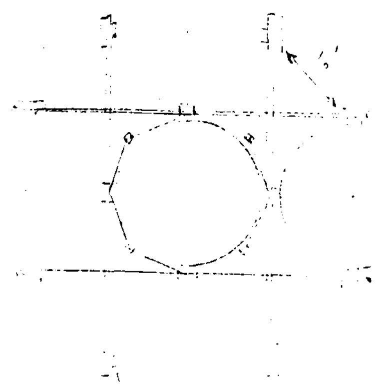
Space shuttle derived engines were used for the tug's propulsion system (Fig. 3-4). Single oxygen and hydrogen tanks would be used to house the propellents. Attitude control thrusters would be placed at the fore and aft ends of the tug to provide for maneuvering capability. These thrusters would also be shuttle-based design (Fig. 3-5).

The shape for the hydrogen and oxygen tanks was chosen to be cylindrical with hemispherical ends. This shape was chosen because it is well suited to take longitudinal forces and can be easily fitted into a space frame structure. A spherical tank would also have been possible, but for the volumes of fuel required, the radius of the tank would be very large. The mission also requires that the payloads experience most of their forces along the longiutudinal axis, and be easily aaccessible for transfer. A spherical tank would make this difficult, while it is easily done with a cylindrical tank. The hemispherical ends simply make the tanks stronger and more structurally sound than right circular tanks would be.

With four containers surrounding a truss structure, a remote manipulator arm (Fig. 3-6) was chosen that would be able to transfer, one at a time, each container. Using only one manipulator arm to move payloads attached on essentialy a circular structure required that the manipulator arm be placed on a circular track where it could move around the structure so that all four containers could be accessible.

Using a total payload weight of 260,000 pounds and estimating the structural weight, a computer program was used to determine the total fuel required to give the necessary change in velocity. The thrust was also determined at the same time, and this was used to set the number of engines required. This number was then used to re-estimate the total structural weight and to repeat the process. The final number of engines was chosen to be two shuttle main engines with two smaller shuttle-derived engines to make up the difference in required thrust.

While the total weight of the space tug was being re-estimated, based on very rough estimates of the tug's structural weight, an initial design of the tug's structure was made that included actual dimensions. An octogonal truss structure was chosen with a length of 60 feet and a diameter of 14 feet. The length of 60 feet was decided on to minimize size and weight but still allow for complete support of the containers. The 14 foot diameter of the truss, the same diameter as the containers, again minimized structural weight and sizing but also allowed for ample spacing between the four containers. A spacing of six feet between the containers was calculated, allowing room for the remote manipulator arm to maneuver.



In order to reduce structural weight as much as possible, T-6061 aluminum was chosen as the material for all truss member components. Aluminum, as compared with such other materials as titanium and graphite, is also less expensive to use. The compressive yield stress of aluminum is 35,000 psi (Ref. 15). With a total of eight longitudinal stringers that form the truss structure, initial structural stress estimates permitted reasonable cross sections for these members. Finally, thermal expansion of the members was not critical because the spacecraft is not affected by small changes in member lengths.

In order to simplify construction of the truss framework, common member and joints were chosen when possible. Specifically, the framework would consist of common longitudinal, lateral, and diagonal members. Members and joints that would not be part of the common elements would be the ones that support the fuel tanks and also support the containers. For further simplification, only the member's cross-sectional area would be considered in the design process. Cross-sectional geometry did not need to be specified due to the assumption that the truss consisted of two force members.

For safety purposes and to meet mission requirements, a three man crew would be necessary. Using a volume per man of 650 cubic feet, allowing room for communications, consummables, electronic systems, and including the power module as an initial size estimate, a module-type structure 20 feet in length and 15 feet in diameter would attach to the front of the truss framework.

Having produced an initial design of the tug, with a number of further design considerations still to be made, the propulsion requirements, including fuel tankage considerations, that were being calculated during the design process, produced results that did not permit the number of engines originally considered to produce enough thrust. The volume of fuel required also could not be contained by the structure.

The volume of fuel required would have needed several smaller tanks to contain it and still withstand the design accelerations. This would have caused unnecessary complexity in the supporting structure. This became a key problem in the design process and required new ideas in solving it. To reduce the fuel amount, two options were considered. The first of these simply involved reducing the number of payloads to two. This did save a good deal of volume and weight, but it would still not fit within the first structure. The second option was to keep the four payloads and to stage the tug. This option allowed the design already completed to be used for the second stage of the tug. The fuel savings realized through staging reduced the size and projected cost of the mission. This second method was chosen as the design to be used.

Staging offers many advantages over non-staging systems. The most obvious of these is the fuel savings due to not having to carry the extra weight through all maneuvers. This method also allows for mission flexibility by using different size lower stages for different payloads and mission profiles. To further save weight, the booster stage could utilize the engines of the second stage during its burn and supply them from its own tanks in a manner similar to that used with the space shuttle and its external tank.

The booster stage would use the same truss structure as the payload stage to simplify construction, and would use the same basic propulsion system. The booster would require an additional truss structure to connect it to the second stage during the portion of the mission that would require the two to operate as one unit. The booster would separate from the second stage to allow it to go into orbit around the moon, while the booster returned to earth via a free-return orbit and resumed orbiting the earth near the space station altitude. The second stage would then rendezvous with the landing vehicle in lunar orbit and return to earth on its own.

To further save on fuel, the original transfer orbit, which was chosen on the basis of time in transit, was traded for a Hohmann transfer. This did not cause excessive time delays (less than 2 days) and saved around half a kilometer per second in velocity changes. For the return orbit, the same transfer was used, except that a small deviation would be made at the moon in order to lower the perigee altitude. This would be done to allow aerobraking to be used. A ballute system with controllable drag was decided upon as the braking system. The ballute drag would be controlled by changing the volume contained by the ballute while it was deployed. This would save considerable amounts of fuel and weight in both the booster and the second stage.

The engineering development of the space tug continued at this point with the design of the container supports. Having decided that the containers would be constructed with external pins that would provide compatibility with the space shuttle's cradling system, a container support frame was designed that would attach the container, by means of its trunnion and keel pins to the truss structure. With the assumption that all containers would be 60 feet long and 14 feet in diameter, having the pins on each side of the container six feet inward of the ends, and requiring only mechanical support, a cross frame, I-beam type structure was developed (Fig. 3-7). There would be welding of two of these frames to make a corner piece. This corner piece would connect at four points on the truss framework. At the ends where the containers would attach, a swivel type clamp was considered that would grasp around the container's keel and trunnion pins. An I-beam type cross section was chosen in order to produce a light weight, rigid structure (Fig. 3-8).

The power for the spacecraft would be provided by a nuclear power plant. Nuclear power generation was chosen because of the mission characteristics: extended operating times, repeated accelerations, and operating in the shadows of both the moon and the earth. The nuclear generator allows longterm operation without replenishing its fuel, and frees the spacecraft from having to seek the sun or carry enough extra fuel to burn for power. Solar arrays would not be capable of withstanding the acclerations necessary nor would they work well when repeatedly exposed to the high energy particles in the Van Allen belts of the earth.

The command, guidance and control systems would be taken from existing system developments such as the shuttle. All engine burns in orbit would require a small "priming" burn by a pressure fed system such as the attitude control system to get the fuel in the main tanks to gather at the aft end for collection into the main engine feed system. Without the small burn, the fuel would be free floating within the tank and would not be able to be pumped.

3.5 LUNAR LANDER DEVELOPMENT

Due to time constraints nearing the end of the semester, the lunar lander was not fully developed into a final design with all parameters fully defined. The preliminary development of the lunar lander design was driven by two primary factors - the payload specifications and the delta V requirements. The payload specifications were set by the decision to make the containers completely shuttle compatible, meaning that the their maximum weight, length and diameter were fixed. Also, this decision set the means of mounting a single container on the lander, requiring the use of the same arrangement and type of fittings found in the shuttle bay.

The orbit chosen for the lunar rendez-vous is a 100 km circular orbit. This is the highest altitude the lander will have to attain, as it will meet the space tug there. From that orbit the lander will effect a 57 minute descent to the surface. The total delta V required for a trip to the surface and a full abort capability is 3.17 km/s.

With these two factors in mind, a weight estimate was made. With a maximum payload of 65,000 pounds, a structure of 20,000 pounds, and an original estimate of fuel weight at 20,000 pounds the total weight was estimated at 105,000 pounds. Considering that the gravitation of the moon is approximately one sixth that of the earth, it was decided to stress the craft as little as possible when thrusting so as to keep the structural requirements for strength down. A total thrust of 25,000 pounds was, therefore, selected. Considering that a single shuttle OMS engine develops 6000 pounds thrust, four of these were chosen (Fig. 3-9). The fuel weight was determined to be 94,000 pounds, somewhat higher than expected. However, a propulsion expert at RCA Astro Division in Princeton, New Jersey was consulted and felt that number was reasonable. It is assumed that advances will be made in the nozzles to allow for long (more than 40 minutes) continuous burns over a long service life. It is also assumed that the OMS engine will be re-designed for throttleability. Four such improved engines are to be attached by a pin joint at the throat of each engine (Ref. 10).

The tanking was determined to be at least 187,000 pounds of H2 and N2O4, (Appendix 10). These fuels are stored, by convention, at 260 psi (Ref. 8). Their density mised is .0419 lb/ft^3, requiring a total tankage of 2,600 ft^3. This is separtated into four tanks, each 650 ft^3. They will be cylindrical with hemispherical ends, app;roximately 20.5 ft long and 6.5 ft in diameter. They will be hung laterally below the main structure of the lander. The program to determine tanking is included in Appendix C-3.

The base for the lander's structure is the keel. This is similar to the keel in the space shuttle, and will accept shuttle compatible payloads. This will be the strongest member of the lander. The g-loading of the keel (see Appendix 10) and the entire lander for the flight profile is included in Appendix C-3.

The method of attaching payloads to the lander will be a set of cradles that will accept the mounting pins used in the shuttle bay. These cradles may be fixed in position or may be hinged at their base to allow a roll-on, roll-off capability (Fig. 3-10). When ladning at improved landing sites the payloads will be lifted off the lander using a crane. The payloads will be transferred to and from the lander in orbit by the remote manipulator arm on the space tug.

A cockpit at the bow of the craft will be open to space, and the pilots will merely strap into a seat for the flights up and down. Extra oxygen will be provided that the pilot may tap into so as to not deplete his own reserves. The role of the pilot in the lander will be the same as in the Apollo LEM; the ship will be flown down by computer and the pilot will take over or aid as needed.

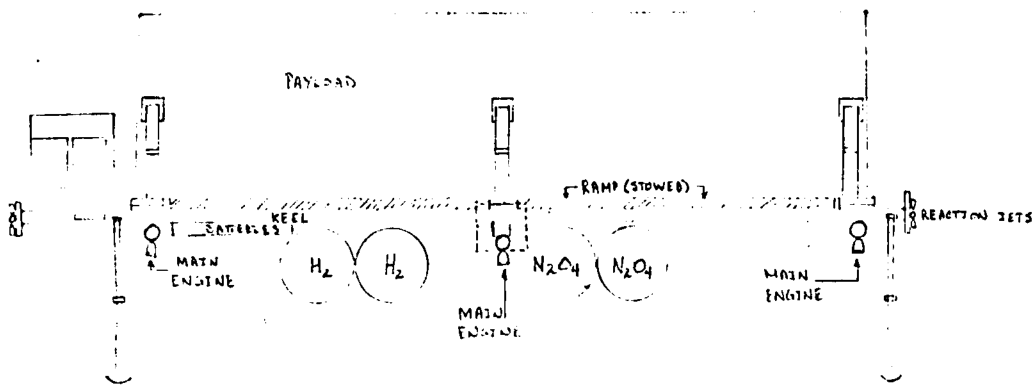
Landing gear for the lander will have knees that will enable it to first absorb the shock of landing and then later allow the vehicle to squat down. Four struts with two at each end of the lander will suffice (Fig. 3-10).

Plane changes will be accomplished by strap-on boosters that will only require the lander to provide attitude control.

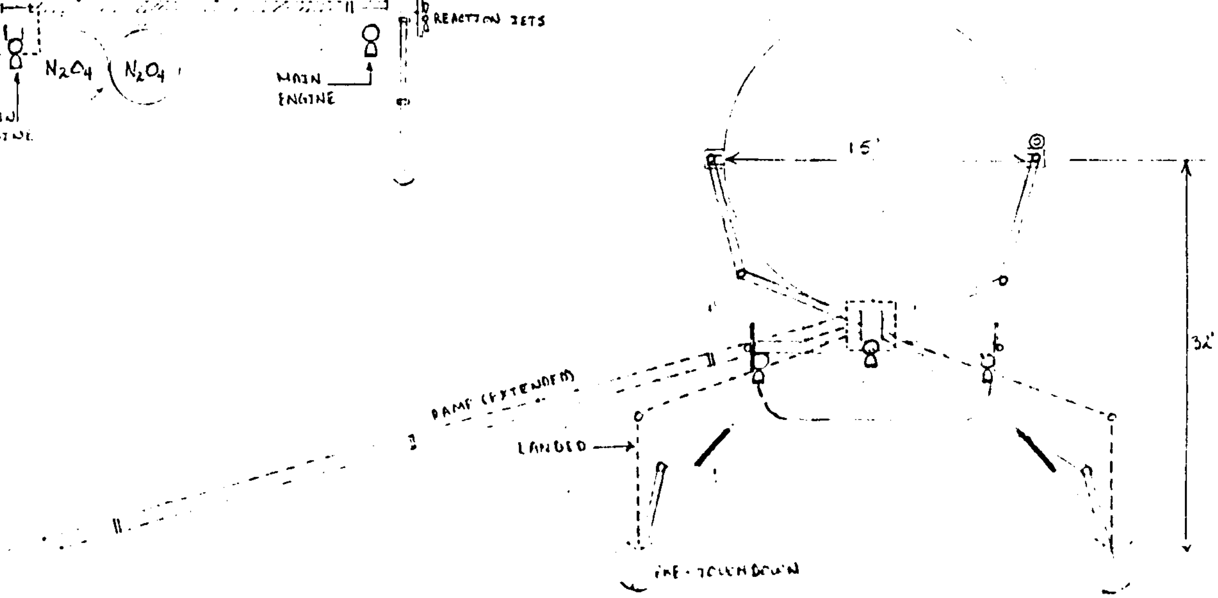
Power for the control systems of the lander will be provided by batteries hung under the keel. They will be able to provide enough power for a round trip of five hours. Once the lander is safely on the surface or in a stable orbit, the pilot will manually exten a non-articulated solar panel to recharge the batteries. A power requirement of not more than 1 kW was estimated by Dr. Pieper of NASA Goddard.

The engineering development of the lander is the extent to which it will be designed. Further design steps will not be taken due to time restraints and the difficulties experienced in designing the space tug.

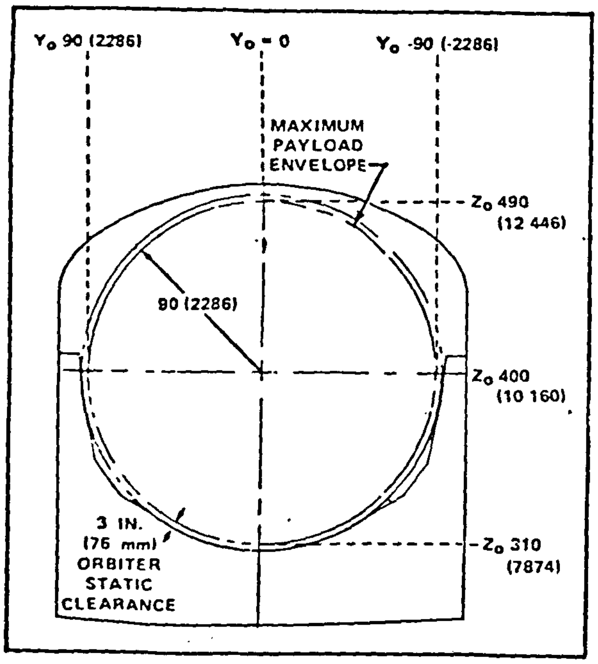
LUNAR LANDER SIDE VIEW



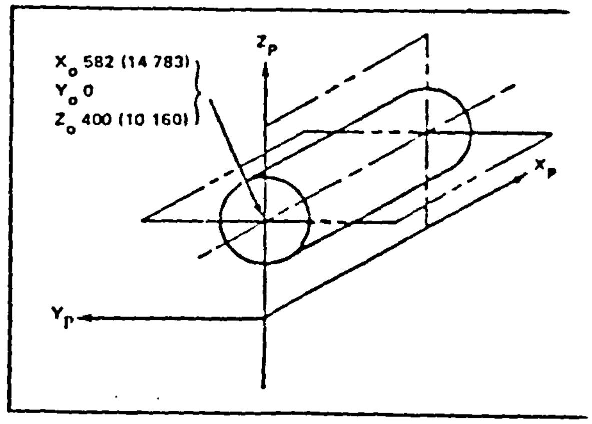
LUNAR LANDER REAR VIEW



The payload clearance envelope in the Orbiter cargo bay measures 15 by 60 feet (4572 by 18 288 millimeters). This volume is the maximum allowable payload dynamic envelope, including payload deflections. In addition, a nominal 3-inch (76-millimeter) clearance between the payload er lope and the Orbiter structure is provided to ŗ vent Orbiter deflection interference between Orbiter and the payload envelope.



View of payload envelope looking aft.



Payload coordinates, showing relationship to Orbiter statio on each axis.

PAYLOAD CLEARANCE ENVELOPE

fig. 3.1.1
(ref.12)

PAYLOAD RETENTION - UPPER LONGERON

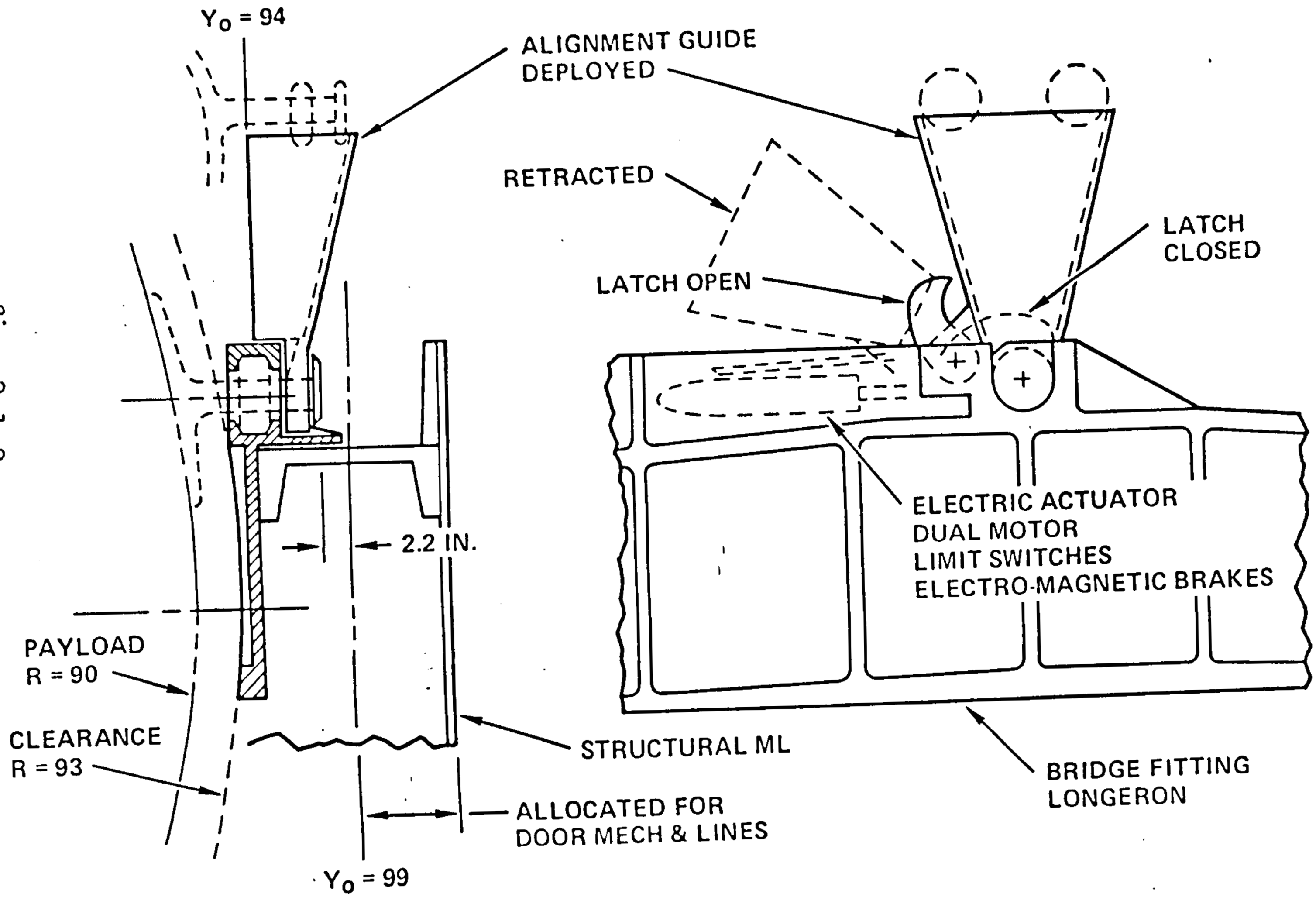


fig. 3.1.2
(ref.12)

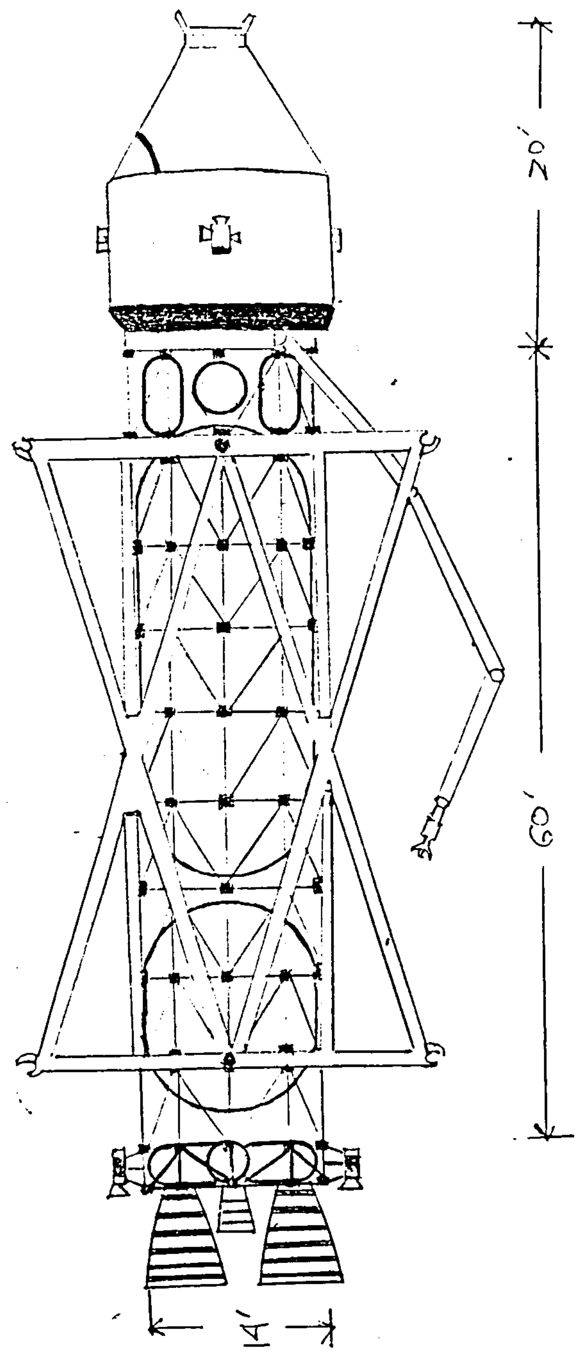


fig. 3.3

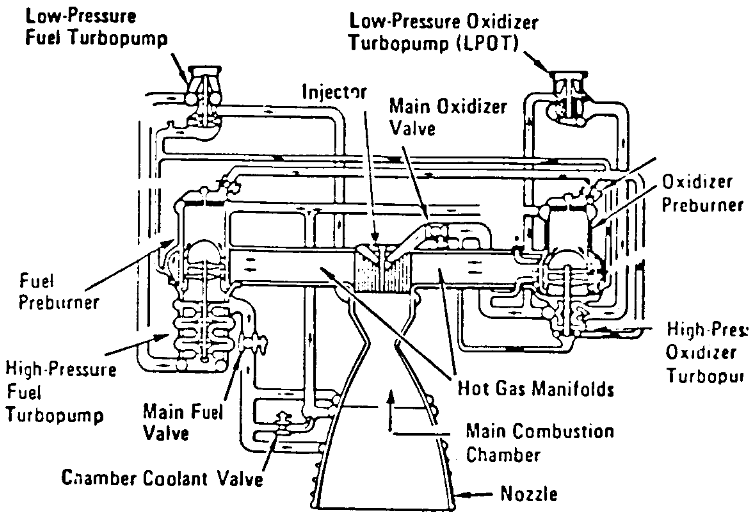


fig.3.4(a)
(ref.12)

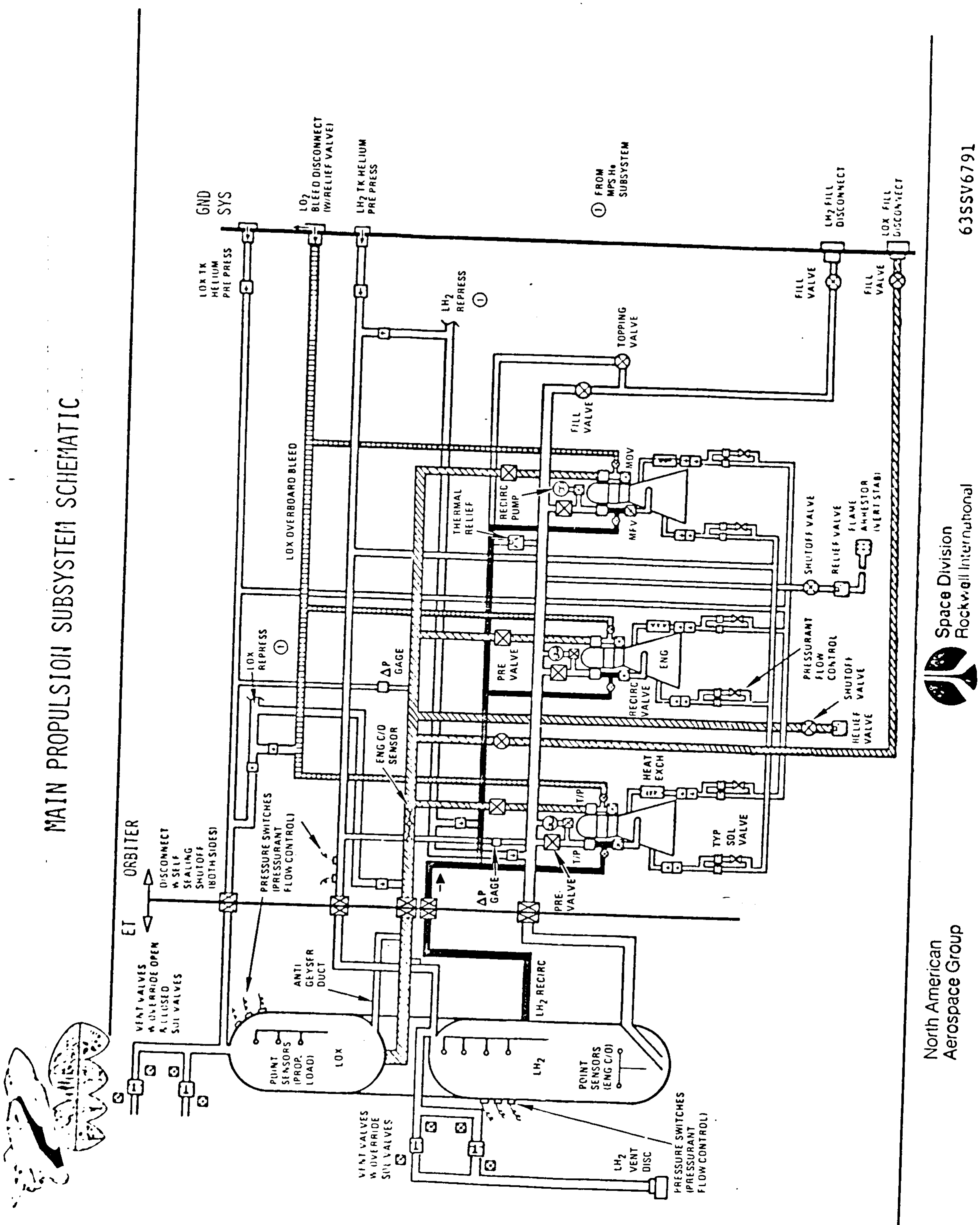


fig. 3.4(b)

(ref.12)

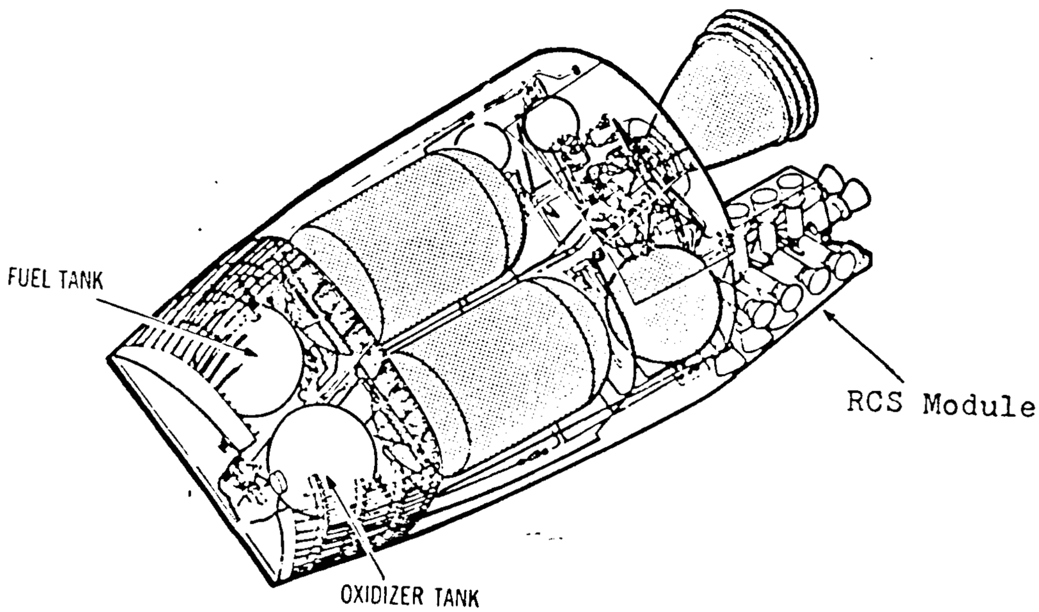
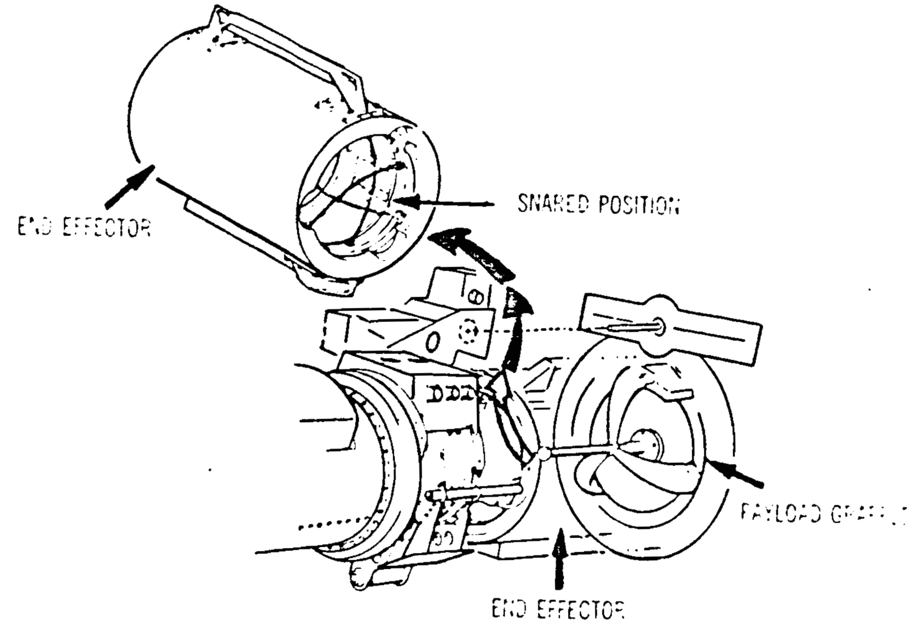
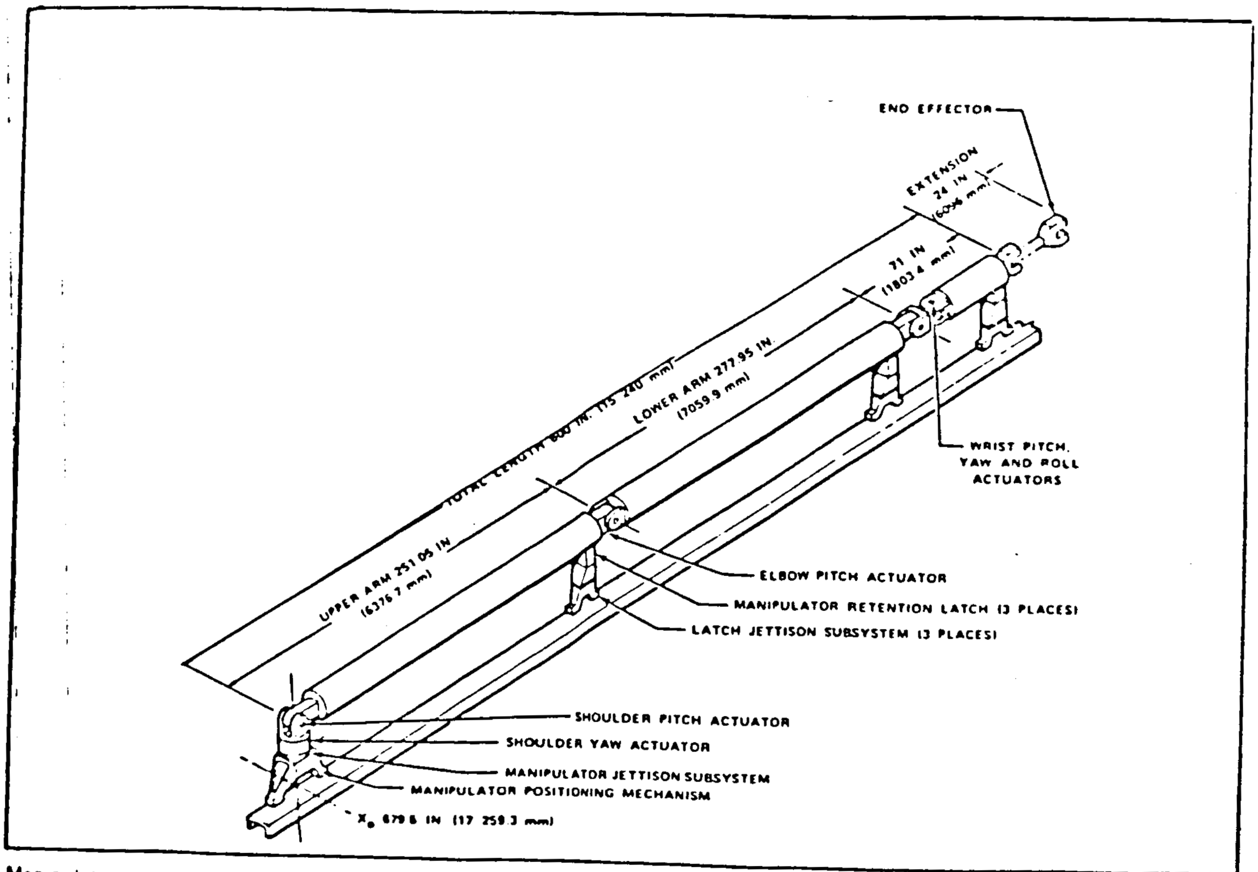


fig. 3.5
(ref.3)

MANIPULATOR ARM ASSEMBLY





Manipulator arm assembly

fig. 3.6

(ref.12)

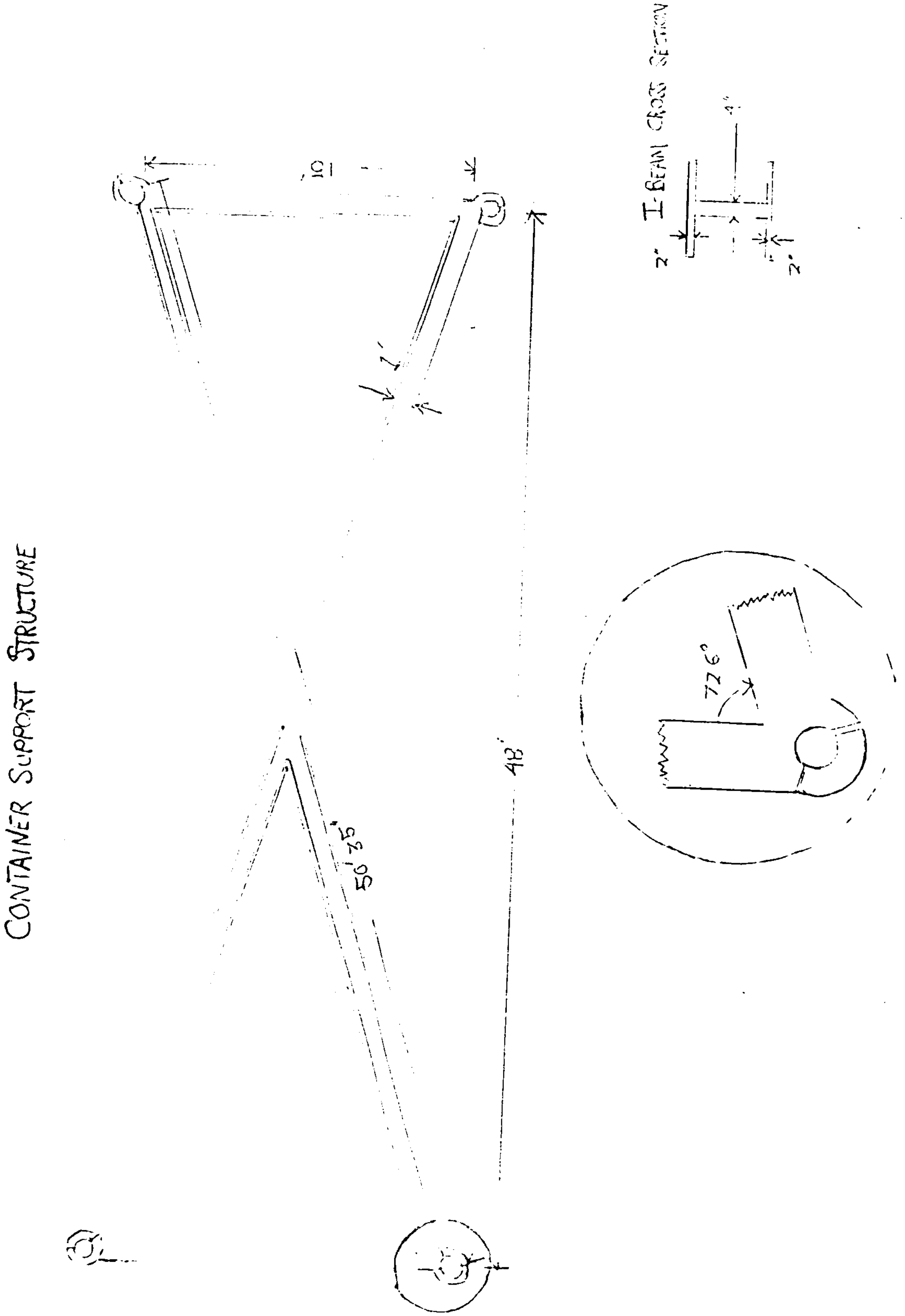
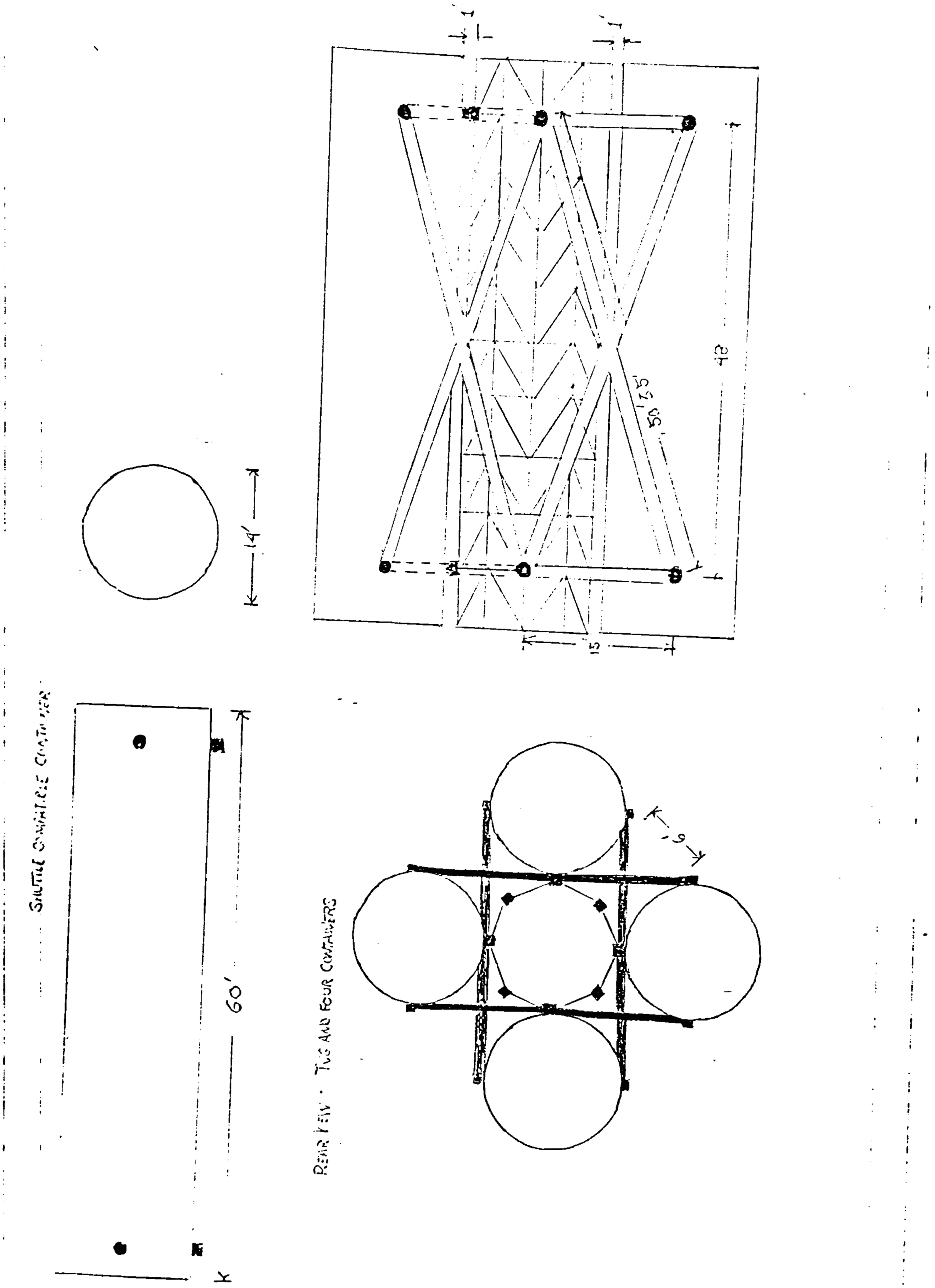
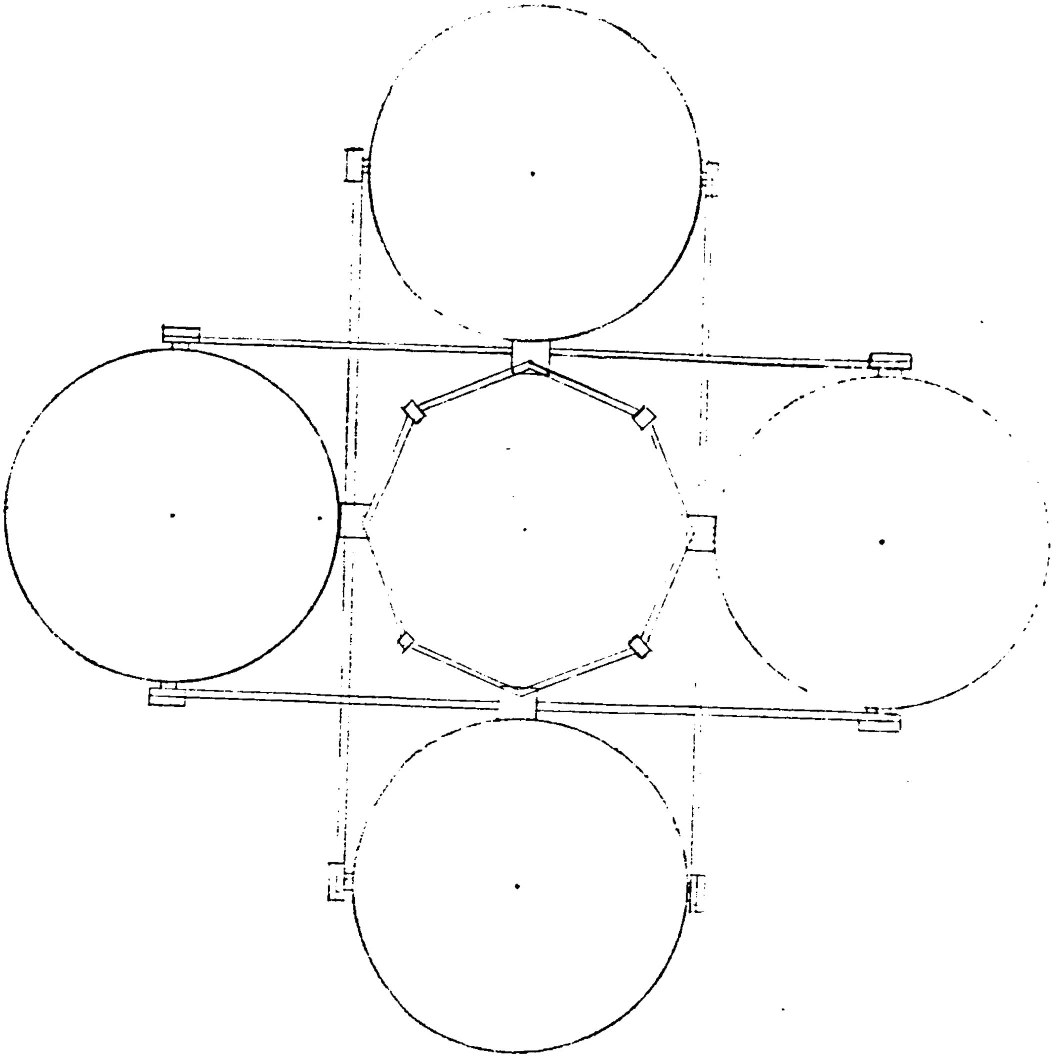

CONTAINER SUPPORT STRUCTURE
10'
1'
48'
72.6°
I-BEAM CROSS SECTION
2"
4"
2"

fig. 3.7 (a)



SHUTTLE COMPATIBLE CONTAINER
14'
60'
REAR VIEW · TUG AND FOUR CONTAINERS
6'

fig. 3.7(b)



SPACECRAFT CROSS SECTION
WITH FOUR PAYLOADS

fig. 3.8(a)

LEGEND
$\frac{1}{8}'' = 1'$

ORBITAL MANEUVERING ENGINES

Orbital Maneuvering System

Two **orbital maneuvering system** (OMS) engines, mounted in external pods on each side of the aft fuselage, power the Orbiter during orbital insertion and deorbit. Aditionally, the OMS provides thrust for large orbital changes.

Each engine has a thrust of 6,000 pounds (26,700 newtons). The propellants are monomethyl hydrazine (the fuel) and nitrogen tetroxide (the oxidizer). Helium gas forces the propellants from their tanks and into the engines. Propellants for each engine are contained in their respective pods. However, there is a cross-feed system to transfer propellants from one pod to the other if needed.

The OMS engine is designed to last a hundred missions. It is 77 inches (2 meters) long and weighs 260 pounds (118 kilograms). The engine is gimbaled in pitch and yaw.

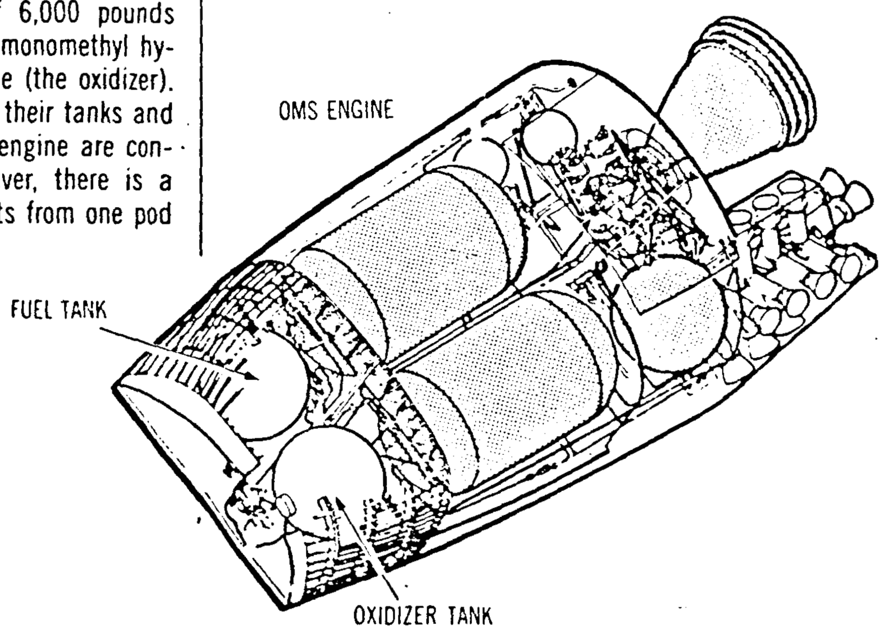


fig. 3.9
(ref.3)

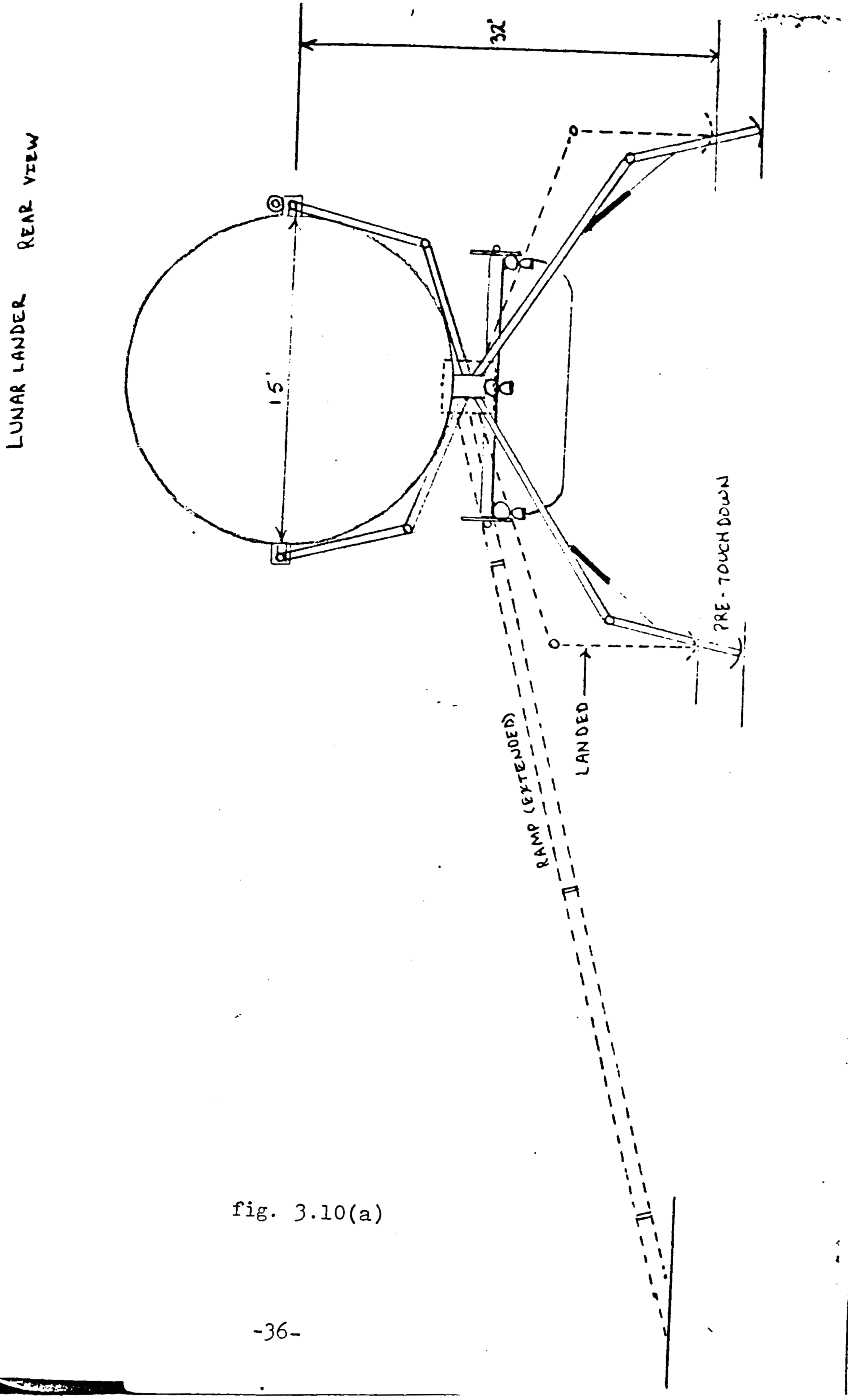

LUNAR LANDER
REAR VIEW
32'
15'
PRE-TOUCHDOWN
LANDED
RAMP (EXTENDED)

fig. 3.10(a)

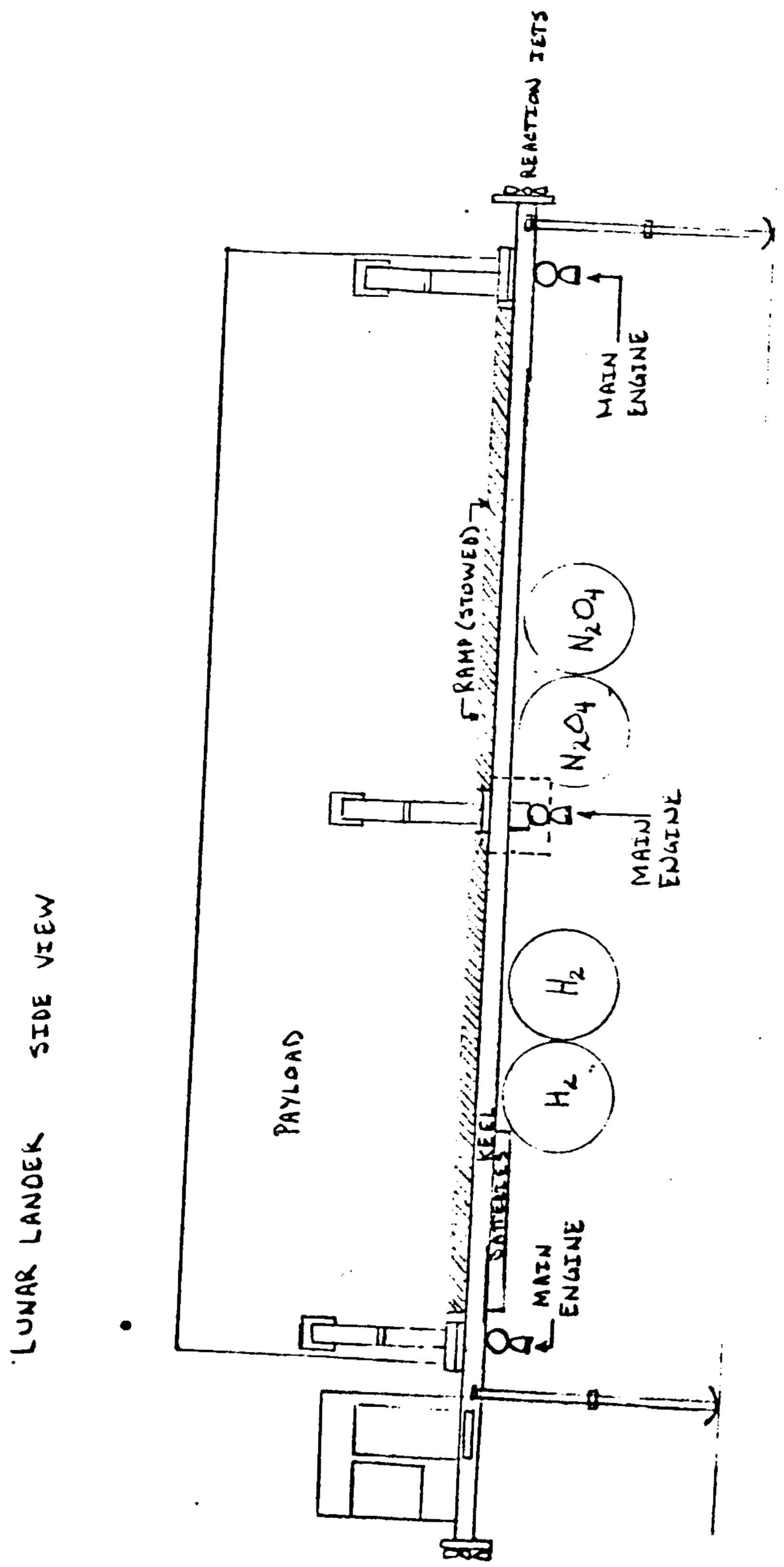

LUNAR LANDER SIDE VIEW
PAYLOAD
REACTION JETS
MAIN ENGINE
RAMP (STOWED)
N_2O_4
N_2O_4
MAIN ENGINE
H_2
H_2
KEEL
BATTERIES
MAIN ENGINE

fig. 3.10(b)

CHAPTER 4 - ORBITS

4.1 The Earth Moon System

In solving the lunar flight mechanics problem the vehicle will be considered as a point mass in earth-moon space. The flight environment will be described by an idealized model. The earth and moon will be thought of as spherical bodies with the gravitational fields of point masses. They will be considered to be isolated in space and revolving in circles around their common center of mass. The entire system can be specified by the following parameters:

$r_{\oplus}$ = mean radius of the earth
= 6378 km

r_M = mean radius of the moon
= 1738 km

ω_m = angular velocity of the system
= 2.649 X 10 rad/sec

$u_{\oplus}$ = gravitational constant of the earth
= 3.986 X 10 km /sec

u_M = gravitational constant of the moon
= 4.887 X 10 km /sec

R_M = distant between the centers of the earth and moon
= 384,400 km

R_S = distance from the center of the earth to the lunar sphere of influence
= 66,300 km

V_M = velocity of the moon relative to the earth
= 1.018 km/sec

The modeled parameters cannot match observed values exactly for the model does not include all factors that contribute to the real situation. The quantities included are known to degrees of precision that are adequate for our application.

4.2 Neglected Factors

The factors that are not considered in the earth moon system include the gravitational field of the sun, the oblateness of the earth, the eccentricity of the moon's orbit and the inclination of the moon's orbit to the earth's equatorial plane. In addition to these exclusions from the model itself, no other outside forces such as solar pressure or meteoroid disturbances of the vehicle will be considered.

4.3 The Geocentric Departure Orbit

Figure 1 shows the geometry of the departure orbit. The four quantities that specify the orbit are

r_o – the orbital radius at perigee
v_o – the orbital velocity at perigee
ϕ_o – the heading angle
γ_o – the phase angle at departure.

In order to avoid difficulties in determining the correct phase angle at departure a more convenient set of independent variables is

$$r_o \, , \; v_o \, , \; \phi_o \;\&\; \lambda_1$$

where λ_1 is the angle that specifies the point at which the geocentric orbit intercepts the lunar sphere of influence. The perigee radius is fixed by the altitude of the parking orbit around the earth. This altitude was chosen to be 500 km so that the tug could interface with proposed space stations. Since the initial burn needed to affect the transfer orbit is to be done at perigee the heading angle will be zero. Different values of the remaining two variables, v and λ were used in a computerized iterative process whose output was values of v and λ , along with the rest of the parameters needed to specify an orbit that meet a predetermined periselenium altitude error tolerence of 2%.

Given the four independent variables r , v , ϕ and λ_1 , the departure orbit can be determined as follows.

The energy and angular momentum of the orbit can be determined from

$$E = (v_o^2/2) - (\mu_\oplus/r_o)$$
$$h = r_o v_o \cos \phi_o$$

From the law of cosines, the radius at the arrival at the lunar sphere of influence, r , is

$$r_1 = \sqrt{R_M^2 + R_S^2 - 2R_M R_S \cos \lambda_1}$$

The speed and heading angle can be found using conservation of energy and momentum:

$$v_1 = \sqrt{2(E + \mu_\oplus/r_1)}$$

$$\cos \phi_1 = h/(r_1 v_1)$$

The heading angle will fall between 0 and 90 degrees since it will be assumed that the arrival at the sphere of influence occurs prior to apogee of the geocentric departure orbit.

The phase angle can be determined from geometry where:

$$\sin \gamma_1 = (R_S/r_1) \sin \lambda_1$$

It is now possible to determine the time of flight (T.O.F.) from injection to arrival at the lunar sphere of influence. The values for the parameter, eccentricity and the semi-major axis, must be determined first from

$$p = h^2/\mu_\oplus$$

$$a = -\mu_\oplus/2E$$

$$e = \sqrt{1-(p/a)}$$

Then the eccentric anomaly can be determined from

$$\psi = \cos^{-1}[(p-r_1)/r_1 e]$$

Finally, since the injection occured at perigee the T.O.F. equals the time since perigee or

$$T.O.F. = tp = \sqrt{a^3/u_\oplus}\,[\psi - e\sin\psi]$$

During this time the moon has moved through an angle

(T.O.F.), where ω_m is the angular velocity of the moon in its orbit. The phase angle at departure can then be found from

$$\gamma_0 = \psi - \gamma_1 - \omega_m(T.O.F.)$$

4.4 Conditions at the Patch Point

It is now possible to determine the trajectory inside the moon's sphere of influence where it will be assumed that only the moon's gravity will be acting on the spacecraft. The first thing that must be done is to determine the spacecraft's velocity with respect to the moon. Figure 2 shows the geometry of the situation at arrival. The subscript 2 indicates the initial conditions relative to the moon's center. Therefore the selenocentric radius, r , is

$$r_2 = R_S$$

The velocity of the spacecraft is determined from the law of cosines as follows

$$v_2 = \sqrt{v_1^2 + v_M^2 - 2v_1 v_M \cos(\phi_1 - \gamma_1)}$$

The direction of the initial selenocentric velocity relative to the moon's center, ϵ, is

$$\epsilon = \sin^{-1}[(v_M/v_2)\cos\lambda_1 - (v_1/v_2)\cos(\lambda_1 + \gamma_1 - \phi_1)]$$

4.5 The Selenocentric Arrival Orbit

The initial selenocentric conditions, r, v and ϵ are now known which makes it possible to compute the conditions at pereselenium. The energy and momentum relative to the moon are given by

$$E = (v_2^2/2) - (u_M/r_2) \qquad h = r_2 v_2 \sin\epsilon$$

The parameter and eccentricity of the selenocentric orbit can be found from

$$p = h^2/u_M \qquad e = \sqrt{1 + (2Eh^2/u_M^2)}$$

The periselenium conditions are obtained from

$$r_p = p/(1+e) \qquad v = \sqrt{2(E + u_M/r_p)}$$

If, at this point, the periselenium conditions were unacceptable, then the value of v_o, λ_1, or both were adjusted to give a satisfactory orbit.

The final orbit was chosen such that it required the minimum energy and provided a selenocentric altitude error of less than 2%. The parameters of the final orbit are

RELATIVE TO THE EARTH	RELATIVE TO THE MOON
v_o = 10.671 km/s	v_p = 2.508 km/s
r_o = 6878 km	r_p = 1836.39 km
ϕ_c = 0	e = 1.3627
γ_o = 109.44 degrees	p = 4338.85 km
a_1 = 195820 km	
p_1 = 13514.4 km	
r_1 = 375933 km	
λ_1 = 77.73 degrees	
γ_1 = .17 degrees	

4.6 The Lunar Descent Orbit

Figure 3 depicts the geometry of the descent orbit. The first orbital manuever used is known as a Hohmann transfer and corresponds to a minimum energy solution. To calculate the velocity change requirement the semi-major axis of the transfer orbit must be evaluated according to the equation

$$a = (1/2)(r_a + r_p)$$

Once this is accomplished the velocities at apoapsis and periapsis can be determined from

$$v_a = \sqrt{2u_{\mu}[(1/r_a)-(1/2a)]}$$
$$v_p = \sqrt{2u_{\mu}[(1/r_p)-(1/2a)]}$$

Knowing the circular parking orbit velocity, it is now possible to calculate the velocity change need to place the vehicle into the transfer orbit using

$$\Delta v_1 = v_{PARK} - v_a$$

The Hohmann transfer will only bring the lander to an altitude of 5 km above the surface. At this point it becomes necessary to enter a more highly elliptical orbit to facilitate lander touch down. The point in the orbit where the maneuver will take place is at periapses of the Hohmann transfer and the apoapsis of the final descent orbit. With the eccentricity of the final orbit set, the following parameters must be calculated in order to determine the velocity needed at the maneuver point that will allow the vehicle to enter the new orbit.

$$p = r_a(1-e)$$

$$a = p/(1-e)$$

$$v_a = \sqrt{2u_{\mu}[(1/r_a)-(1/a2)]}$$

Subtracting the Hohmann transfer periapsis velocity from the final descent orbit apoapsis velocity yields the velocity change required to complete the manuever.

$$\Delta v_2 = v_p - v_a$$

The final velocity change is simply that needed to stop the lander on the lunar surface. The velocity at the surface is

$$v_S = \sqrt{2\mu_M[(1/r_M)-(1/2a)]}$$

$$\Delta v_3 = v_S - 0$$

The total velocity change needed to descend from the parking orbit to the lunar surface is then the summation of the individual changes.

$$\Delta v_{DESCENT} = \Delta v_1 + \Delta v_2 + \Delta v_3 .$$

4.7 Abort Orbits

Because both the tug and the lander are manned it is necessary to provide an abort capability. The worst possible case, as considered from an energy standpoint, was chosen as the design point. For the tug the location in its orbit that would require the most energy to correct for any trajectory error would be immediately after the first burn which places the tug into its transfer orbit. At this point, if it was decided that the mission should be aborted, the only energy the tug has left to affect the transition to the abort orbit is that energy originally intended for the circularization of its orbit at the moon. If this energy is spent to abort the mission it must be determined if an acceptable orbit can be obtained. First the new velocity after the abort burn has been made must be determined by

$$v_A = v_p - \Delta v_2$$

From this the semi-major axis of the new orbit and the orbital period can be calculated according to

$$a = 1/2[(1/r_c) - (v_A^2/2\mu_\oplus)]^{-1}$$

$$T = (2\pi a^{3/2})/\mu_\oplus^{1/2}$$

In this case the period of the abort orbit is 7.789 hours which is acceptable.

For the lander the abort capability must allow for a return to the 100 km parking orbit if for some reason the decision to land has been changed after the lander's transfer into the final elliptical descent at an altitude of 5 km. The energy required to return the lander to its parking orbit is the sum of the burns used to place the spacecraft into the final descent phase or

$$\Delta v_A = \Delta v_2 + \Delta v_i = 1.3 \text{ km/s}$$

With the orbit capability the total energy required to be carried by the lander is

$$\Delta v_{TOT} = \Delta v_{DESCENT} + \Delta v_{ABORT}$$
$$= 3.124 \text{ km/s}$$

4.8 The Return Orbit

For the tugs return orbit an energy saving pass through the earth's atmosphere will be used. Because of the complicated calculations of both aspects of this orbit only a very superficial look into the energy saved by the aerobraking maneuver will be made. A one pass orbit will be used with a perigee altitude of 200 km. The average value for the density of the sensible atmosphere along the tugs flight path as well as an average velocity per unit mass can be used to calculate the drag force per unit mass from

$$F_D = K X v^2$$

where K is 1/2 X the coeffecient of drag X the reference area. For a preliminary inquiry into the drag effect K is taken to be 500 m^2. From the equation

$$F = Ma$$
$$F/M = dv/dt$$
$$dv = (F/M)dt$$
$$dv = F_D\,dt$$

and the fact that the drag force will be considered to be constant the delta V applied to the vehicle during the pass is

$$\Delta v_{ATM} = F_D \Delta t$$

The delta V was assumed to be applied at the perigee point. Therefore the new perigee velocity is

$$v_{p_{NEW}} = v_{p_i} - \Delta v_{ATM}$$

and the semi-major axis is now

$$a = (1/2)[(1/r_p) - (v_{p_{NEW}}^2/2\mu_{\oplus})]^{-1}$$

From this, the velocity at the altitude of the parking orbit is

$$v = \sqrt{2\mu_{\oplus}[(1/r)-(1/2a)]}$$

The delta V needed to enter the parking orbit is

$$\Delta v = v - v_{PARK}$$

This rough estimate of the energy that can be saved over the outbound energy requirements is approximately 5%.

OUTBOUND ORBIT

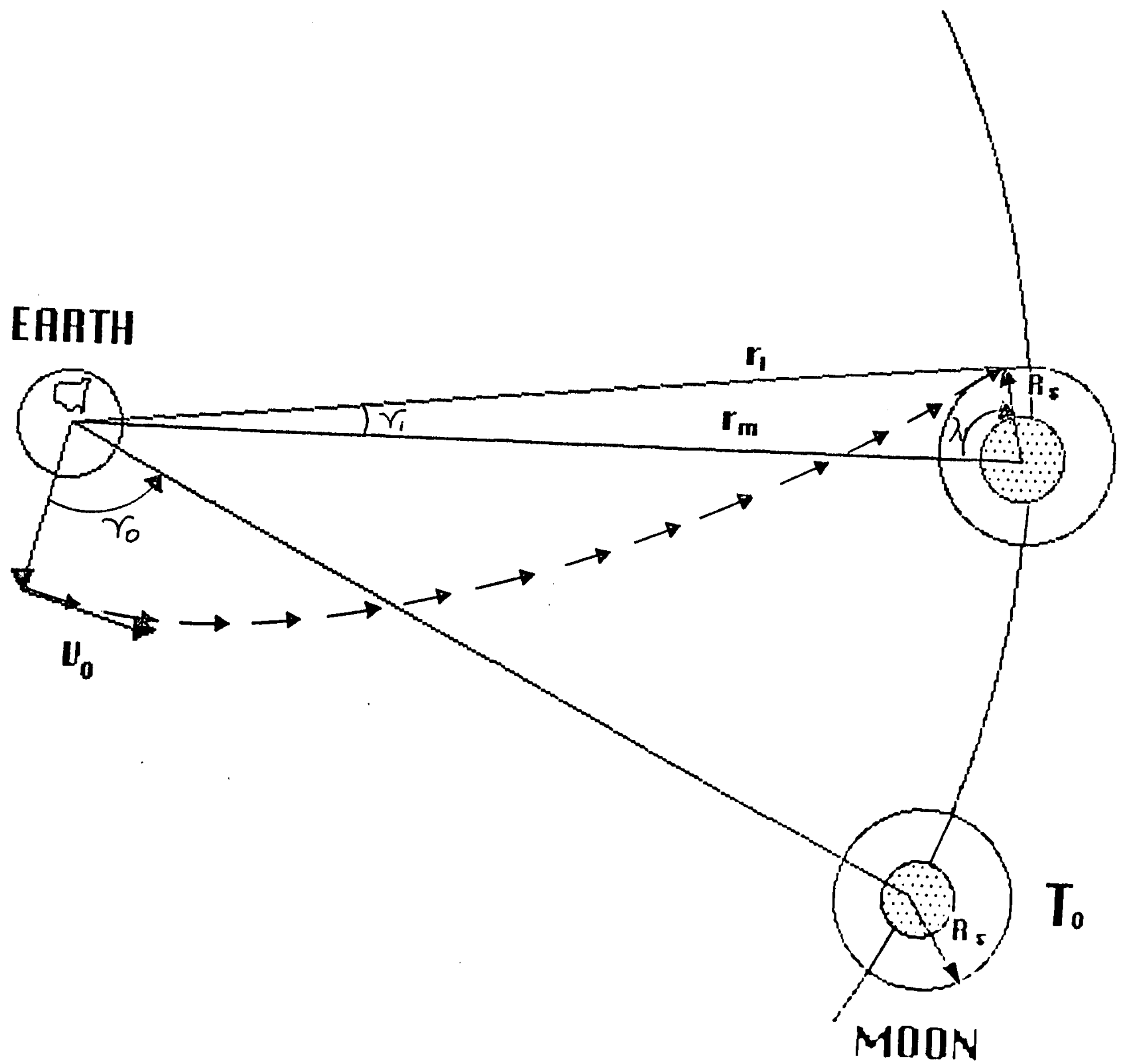
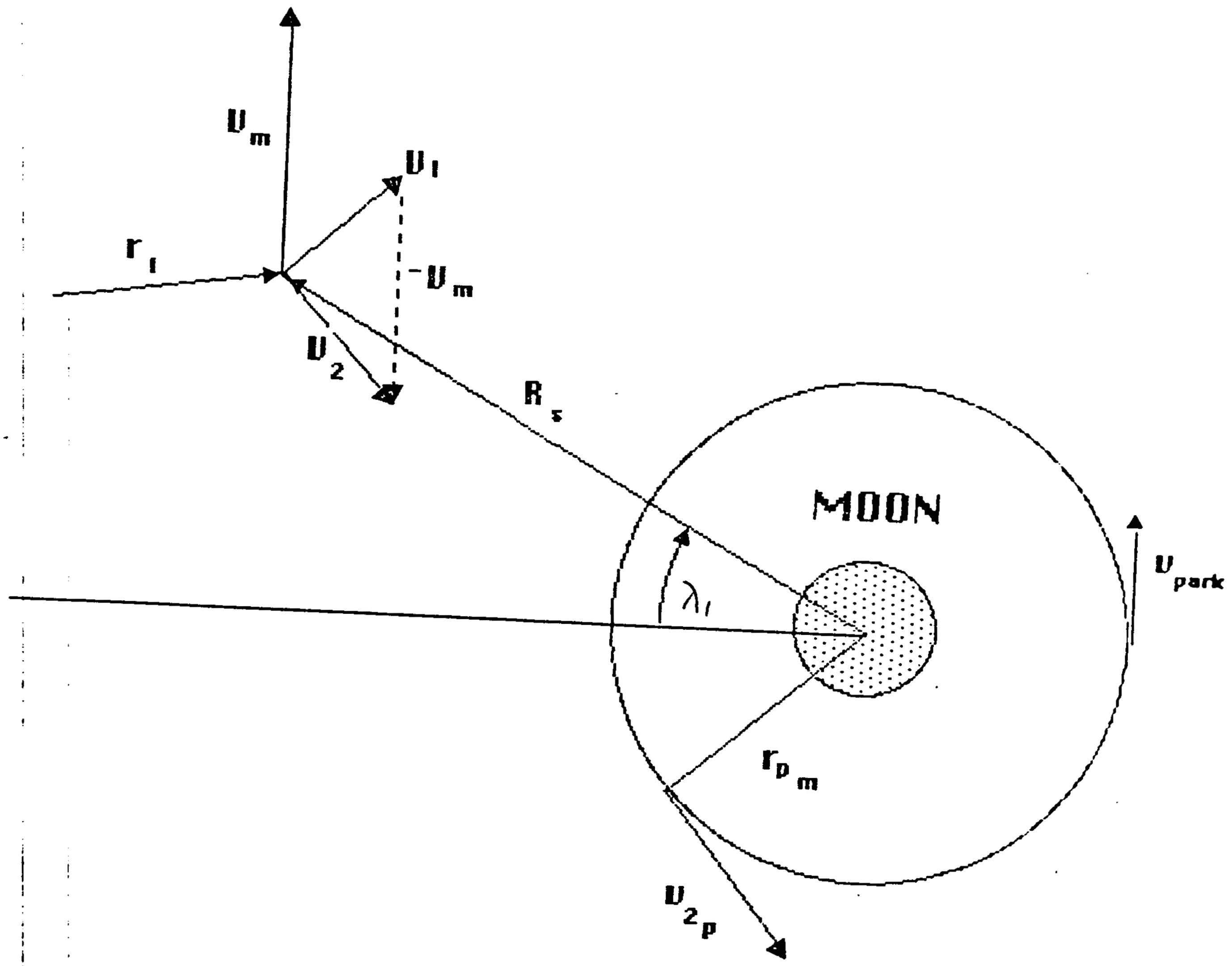
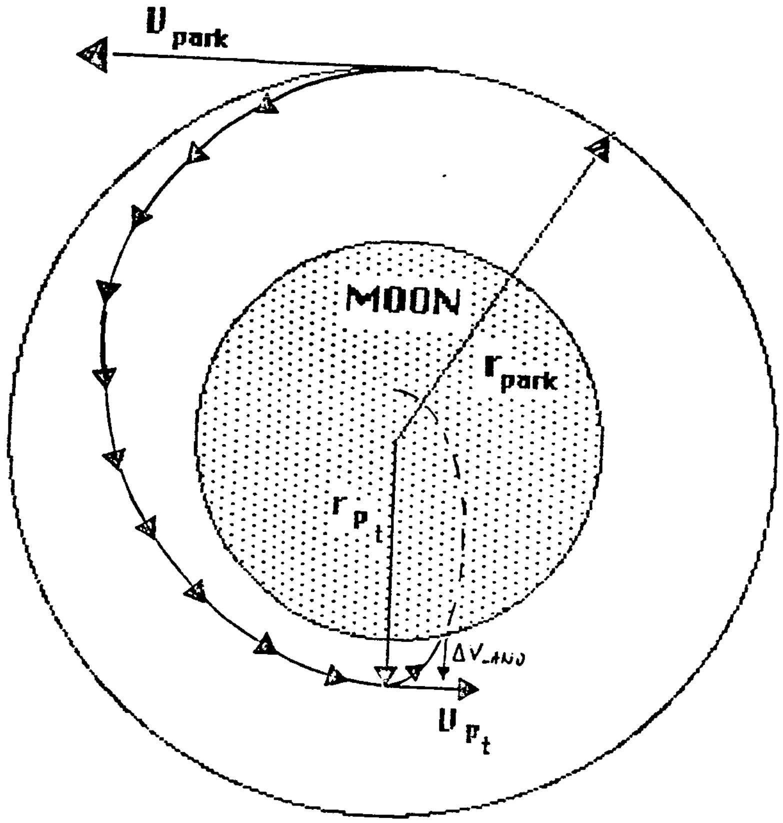


FIG-1
(ref.1)



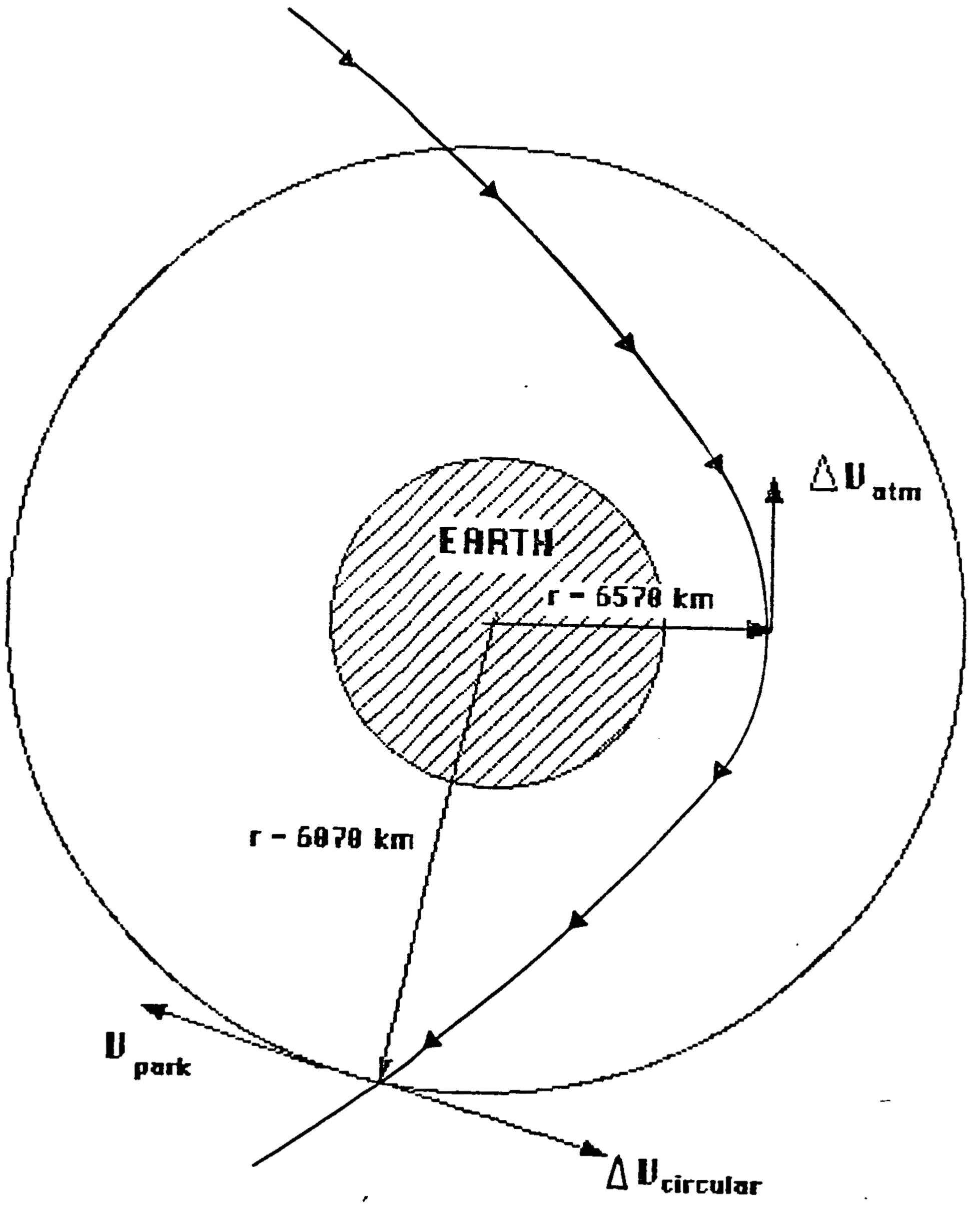
PATCH POINT

FIG-2
(ref.1)



DESCENT ORBIT

FIG-3
(ref.1)



RETURN ORBIT

FIG-4
(ref.1)

Chapter 5 - SPACE TUG PROPULSION

5.1 OVERVIEW

For the space tug propulsion, the Space Shuttle main engine was chosen because it is the most efficient high thrust engine currently available. Specific performance information was acquired via a phone conversation with Jim Sander of the Marshal Space Center. This information was then used as a constraint on the design characteristics.

The other mission and design constraints included orbital requirements, time of flight desired, burn time limitations and structural limits. The orbital requirements and time of flight were covered in chapter 4, so they will not be covered here except in mentioning how they affect the design.

5.2 PROPULSION ASSUMPTIONS

The volume of fuel required was calculated using a computer program which numerically solved a recursive formula (Appendix C-2). This program was based upon several assumptions which are good estimations of the actual response characteristics. The first of these was the assumption that the engines used responded instantaneously to throttle, ignition, and cutoff commands. No lag time was considered for simplicity's sake and because specific data about this was not available. On the actual spacecraft, change in velocity could be determined by integrating the output of accelerometers, so this does not cause a problem, but it does affect the actual burn times slightly. Along the same lines, the engine thrust was assumed to be constantly updated by the computers to maintain a constant-acceleration burn.

The most important assumptions are also the ones with the largest chance for error. Both of these have to do with the engine's mass flow to thrust produced characteristics. The first assumption of this type was that the response is linear throughout the throttling range of seventy to one hundred nine percent of the design thrust. According to professor Saarlas of the United States Naval Academy Aerospace Division, this is a fairly good estimation if two points on the response curve are known. The only point available, however, was the mass flow rate at one hundred percent thrust. The real mass flow response would be nearly proportional to the thrust percent, but not necessarily exactly proportional. The most fallible assumption made, then, was that the mass flow rate remained proportional to the thrust (i.e. at seventy percent thrust, seventy percent of the mass flow at one hundred percent thrust would be observed).

5.3 CALCULATION AND ITERATION

The orbital considerations gave the required velocity changes used as inputs to the program, which along with initial mass estimates were used to determine the initial tank volumes. These were then used to re-estimate the structural mass and iterate again. The initial concept of a single stage system required extremely large tank volumes, and large structural masses. By changing to a staged vehicle, with both stages capable of aerobraking for recapture into a space station orbit, over ten thousand pounds of fuel were saved, and the tank sizes became more manageable.

The thrusts required in some parts of the mission made it also necessary to have smaller engines aboard both the tug and the booster stages. These engines were assumed to have the same mass flow to thrust ratio as the main engines and need to operate at fourteen to twenty six percent of the main engine thrust. Their design will be left to future development.

CHAPTER 6 - SPACE TUG STRUCTURE AND SYSTEMS REQUIREMENTS

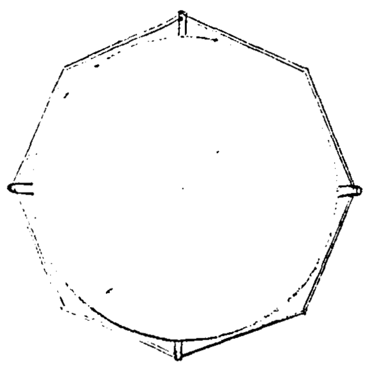
6.1 TRUSS FRAMEWORK DESIGN

While the final design of the propulsion system took place, detailed analysis of the structural requirements was also being performed. As the central component of the space tug structure, the truss framework was analyzed for maximum forces in its members (Appendix 2). The maximum forces were observed to take place in the aft longitudinal members of the truss. Having made an original estimate of the tug's total weight as 420,000 pounds, an extra 20,000 pounds was added to make a total weight of 440,000 pounds. Having originally estimated the truss structure at 10,000 pounds, the added 20,000 pounds would compensate for a possible underestimate so that the truss members were certain to be designed for maximum forces.

Having determined the forces in the longitudinal members, a minimum cross section was calculated based on the compressive yield stress of aluminum and providing a factor of safety of 1.5 (Appendix 3). With a design cross-sectional area of 7.1 in^2, all members subject to a force equal to or less than the longitudinal members would have the same cross-sectional area. This would simplify construction of the truss framework in space, and also allow for that much more factor of safety in the members subjected to less force. Furthermore, this design is reasonable since the cross-sectional area is not large to begin with.

6.2 FUEL TANK DESIGN

The final design of the hydrogen and oxygen tanks was restricted to tank diameters of no greater than 13 feet and a combined overall length of 60 feet. Having established the fuel requirements for the two SSME's and the two shuttle derivatives, both tanks were designed to have a diameter of 12.5 feet and a combined length of 47.3 feet. These dimensions fit well within the truss framework providing room for adequate spacing between the two tanks as well as allowing room for fuel tankage for the attitude control system and main engine piping (Fig. 6-1). With the truss having a diameter of 14 feet and the tanks having a diameter of 12.5 feet, the spacing between the truss and the tanks is only 9 inches on each side. The small amount of spacing allows for a more sturdy attachment of the fuel tank struts to the truss members as shown.



As calculated in appendix 4, the forces at the tank attachment points and in the strut members permits the design of a reasonable cross-sectional area (1.6 in^2 or greater) for the members. The tank structure is assumed to be similar in design as the space shuttle external tank (ET), a tank made primarily out of aluminum. The stress limitations for the attachment points on the tanks are, therefore, the same as the stress limitations for the strut members. The stress experienced at the attachment points on the space tug's truss framework is not analyzed since the tank struts attach at the joint connection plates. The force applied, therefore, is in the longitudinal direction of the tug's truss members, producing a force on the members well within the limits of the force the truss framework was designed to withstand.

6.3 PAYLOAD CONTAINER SUPPORT DESIGN

All of the loads on the payload supports were restricted to the longitudinal forces in the aft direction. The structure itself was analyzed as a two-force member truss. This does not hold precisely true, but this is a conservative approximation in that it yields slightly higher stresses on the members than will be actually experienced. This produced a small safety factor which was multiplied by a factor of 1.5 for determination of the cross-sectional area.

Each support was assumed to be supported equally at six points. The location of the two container keel pins placed one third of the force directly on the space frame of the tug itself. The other two-thirds of the force was shared by the other four trunnion pins (See appendix 5). Each support frame (Fig-6.3) supported four pins, one at each corner. The resulting force was used to calculate the cross-sectional area of the support truss and to check the stress on the space frame at the points where the support truss connected to it. A cross-sectional area of 1.64 in^2, which included the safety factor, was required. This yielded an individual truss mass of 277 pounds and a total mass of all four trusses of 1108 pounds.

6.4 BOOSTER SUPPORT DESIGN

The booster support was chosen to consist of four frames mounted on the booster with detachable pin connections attaching them to the space frame of the tug. The four frames were placed assymmetrically on the structure so that they could cross without touching and allow movement when attaching and detaching from the tug (Fig. 6-2). The four frames had to be capable of supporting both longitudinal forces and moments caused by the thrust of the booster and tug engines not being through the center of gravity. The force in the longitudinal direction is the total mass (the mass of the structure and the total fuel mass) multiplied by the acceleration (2 g's). The forces caused by the offset thrust are solved for using four simultaneous equations (Appendix 6) and then transferred into the plane of each truss by solving for the angle between vertical and the truss plane. Each individual truss had different forces, so the final required cross sections were different for each member. These can be found in table 1 of appendix 6. The total mass of the four frames is 2,666 pounds mass. The lateral stringers at the points of attachment for the middle two frames require different cross sections than the rest of the space frame. These are listed in Table 2 of appendix 6.

6.5 WEIGHT ESTIMATE REFINEMENT

A refinement of the weight estimation of the space tug to a more accurate figure was accomplished by first summing all of the truss framework component weights (Appendix 7). As shown in figure 3-2 the truss members and the joint connection elements were the only components to be considered for a weight estimation of the framework. With a cross-sectional area of the truss members of 7.1 in^2, the joint connection plate was designed with a square side length of 9 inches and a thickness of 4 inches. Only one type of plate element would be used throughout the structure, furthering design simplicity. With the dimensions of the plate element given, the truss members would be inserted into the plate to the point where their ends would be as close to the center part of the plate as possible.

In determining a weight for the space tug's hydrogen and oxygen tank structures, existing data on the space shuttle's ET was used. With the weights estimated being small relative to other components of the space tug, the method of comparing the size of the tug's propellent tanks with the shuttle ET is accurate for design purposes (appendix 8).

Approximating the weight of the crew module was based on the weight estimation for a mars mission space vehicle having a crew of four on board (Ref. 2) (Table in appendix 9). By accounting for the difference in crew size, duration of the mission, and mission requirements a weight estimation of 7,000 pounds was made for the crew module alone.

By comparing, as before, the differences in mission requirements and mission duration of the mars mission space vehicle with that of the space tug, estimates for the weight of the attitude control system, batteries, nuclear power module, communicatons equipment, airlock, and thermal protection were made.

6.6 CREW MODULE

The design of the crew module entailed only giving the exterior dimensions to provide room inside for a crew of 3, their personal gear, consummables required, spacecraft instrumentation, and communications. The size of the crew module was based on assuming that each crew member requires 650 cubic feet of living space and then adding extra volume for the other items mentioned above (Ref. 10). Also, the diameter of the crew module was retricted from the start to 15 feet so that the truss framework of the tug could be attached to it as close to its strongest section as possible (Fig 3-3).

6.7 ATTITUDE CONTROL SYSTEM

With the importance of effective attitude control for the space tug, a high weight estimate was made for the attitude control system since many large thrusters would be required on the tug to ensure proper maneuvering during docking and other high maneuvering points in the orbits. The system has a forward and aft set of thrusters, both modularized for ease in servicing and simplifying logistical problems. Each module contains identical components which will include not only the thrusters but also the bipropellent tanks. These modules are based upon the space shuttle's attitude control thrusters (Ref. 12).

6.8 THERMAL PROTECTION

The crew module's external structure and the truss framework are made of T-6061 aluminum. In order to perform aerobraking of the space tug and permit the tug's structure to withstand high heat tolerances, thermal protection is required for the tug's entire external structure. The thermal protecton will be accomplished in two phases - one active and one passive. The passive phase will include different heat shields, depending on the heating taking place. Areas subjected to intensive heating will have a coating of a reinforced carbon carbon material, similiar to the tiles on the Space Shuttle. This will be particularly important during the aerobraking pass through the earth's atmosphere. For areas subject to less severe heat, high temperature reusable insulation will be used. The active systems will include various heaters, radiators, boiler coolers, and other heat exchangers that will be integrated with the environmental control and life support systems (Ref. 12).

6.9 ENERGY STORAGE

The electrical power system consists of the equipment and reactants required to generate and store energy. The storage requirements will be met by nickel-cadmium batteries whose purpose will be to provide the capability for power subsystem reset and restart. To compliment this, three silver-zinc batteries will be used to meet flight instrumentation contingency requirements (Ref. 12).

6.10 POWER GENERATION

Power generation will be provided by a nuclear reactor. This reactor will supply the energy required to drive three separate DC generators. Each generator will be tied to the load through an independent DC bus (Ref. 12). For design purposes, the power generation module was assumed to be part of the crew module. Specifically, the power generation module would be located at the very aft section of the crew module.

6.11 COMMUNICATIONS

The communication and tracking subsystem will provide RF communication and tracking links. The hardware will consist of various transponders for tracking, telemetry, commands, and voice transmission, as well as their required antennas. Audio processing and distribution equipment will also be provided.

6.12 AIR LOCK

The airlock provides the necessary support for extra vehicular activity (EVA) by allowing access to and from the space tug and lunar lander. The small cylindrical chamber will allow for depressurizing and repressurizing without affecting the entire crew module.

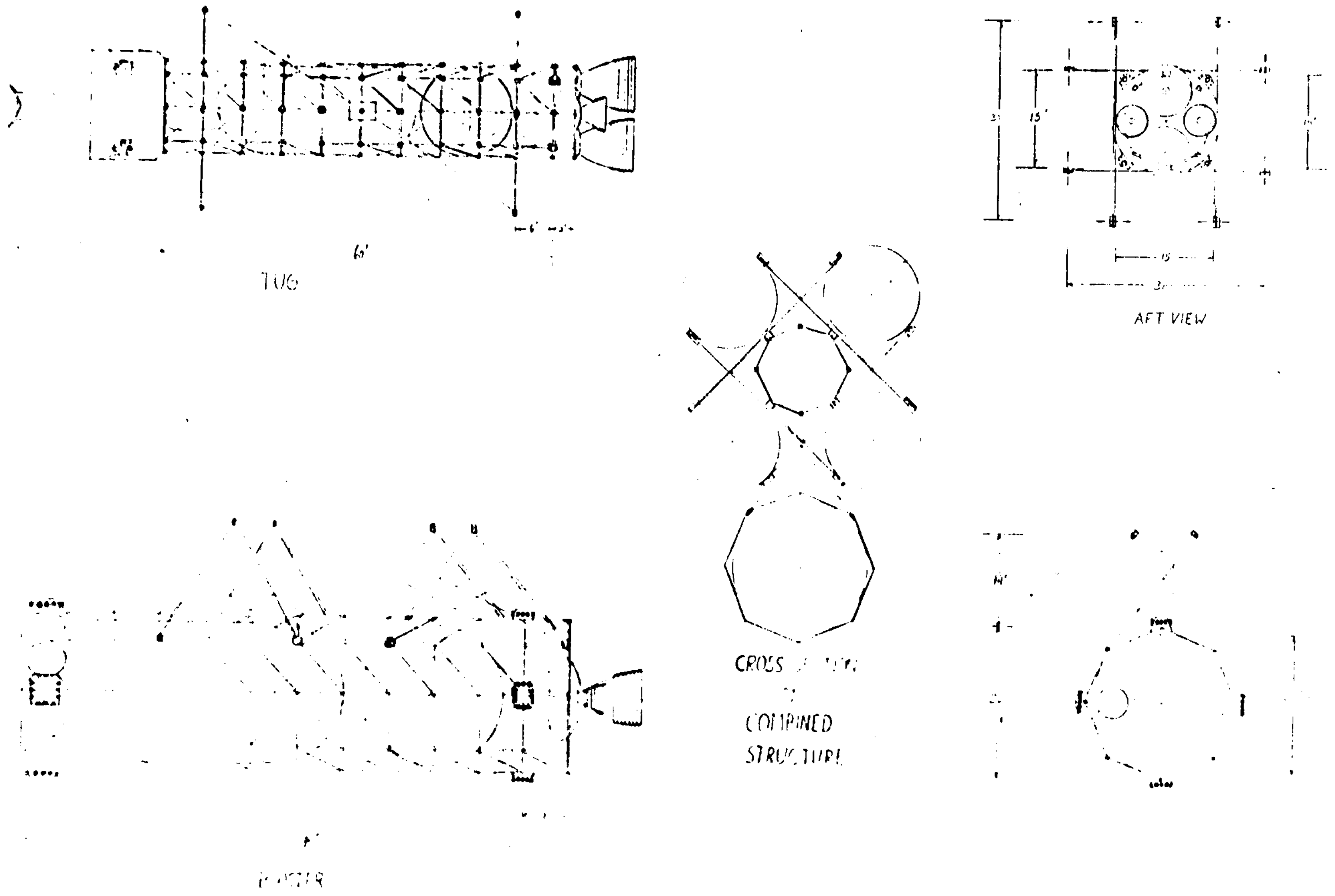
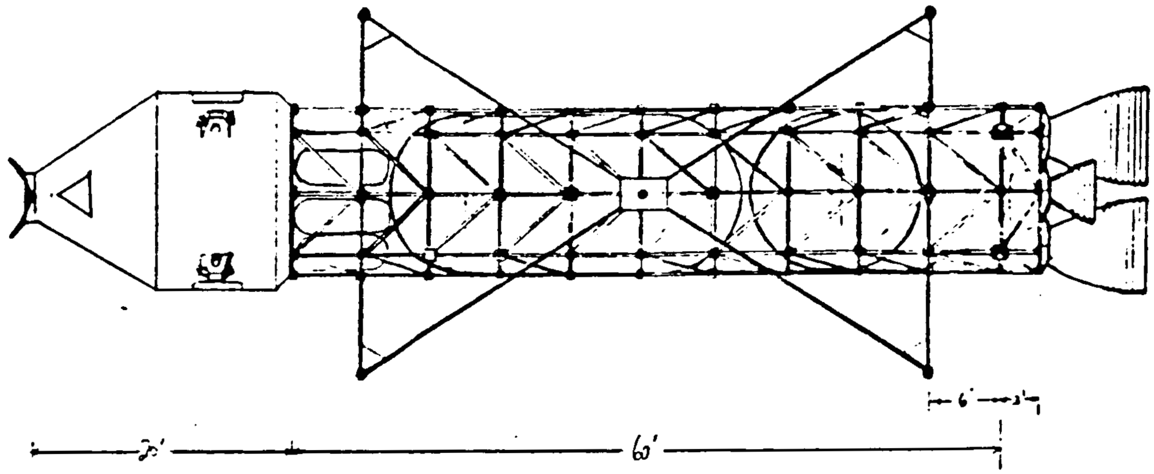

TUG
AFT VIEW
CROSS
COMBINED
STRUCTURE

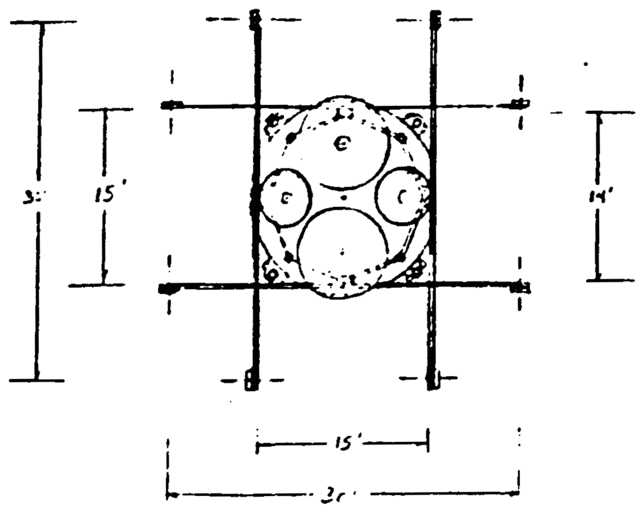
Table 6-1

SPACE TUG WEIGHT ESTIMATION

4 Containers	– 65,000 lbs each	–	260,000 lbs
Truss Structure	– Members and Joint Connection plates	–	15,000 lbs
Oxygen Tank (empty)		–	3,250 lbs
Hydrogen Tank (empty)		–	7,200 lbs
Oxygen propellent		–	92,500 lbs
Hydrogen propellent		–	15,500 lbs
Crew Module		–	7,000 lbs
Crew of 3, personal gear + consummables		–	1,500 lbs
4 Container Support Structures	–270 lbs each	–	1,100 lbs
Batteries		–	900 lbs
Communications		–	400 lbs
Attitude Control System		–	2,500 lbs
Thermal Protection		–	500 lbs
Power Module (Nuclear)		–	7,300 lbs
2 SSME's –	6875 lbs each	–	13,750 lbs
2 Shuttle derived engines –	2000 lbs each	–	4,000 lbs
Air Lock		–	1,500 lbs
	TOTAL WEIGHT	–	433,000 lbs

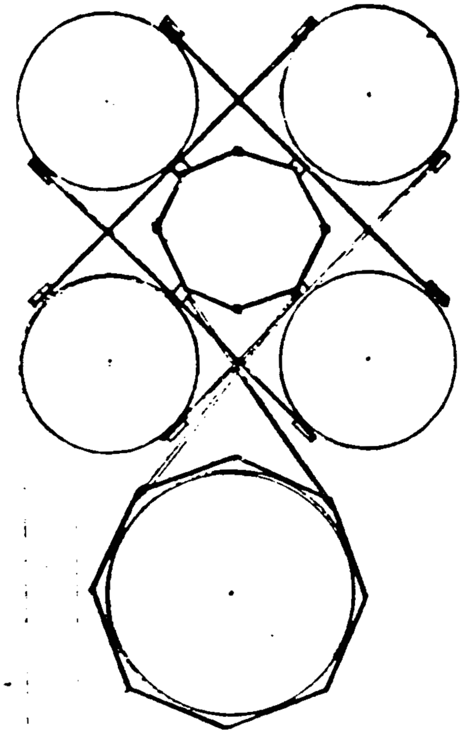


TUG

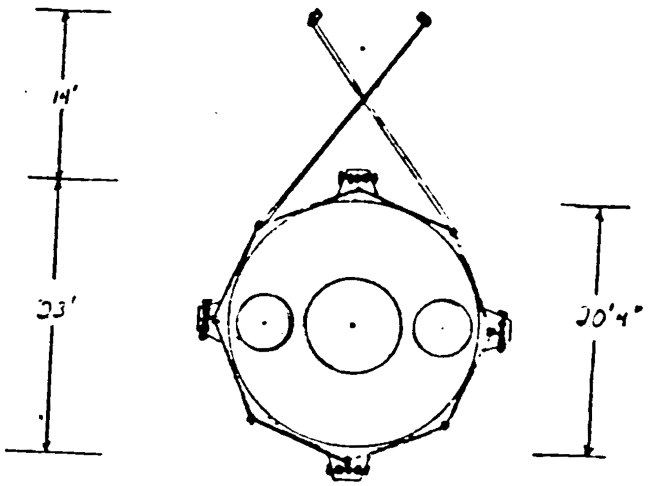


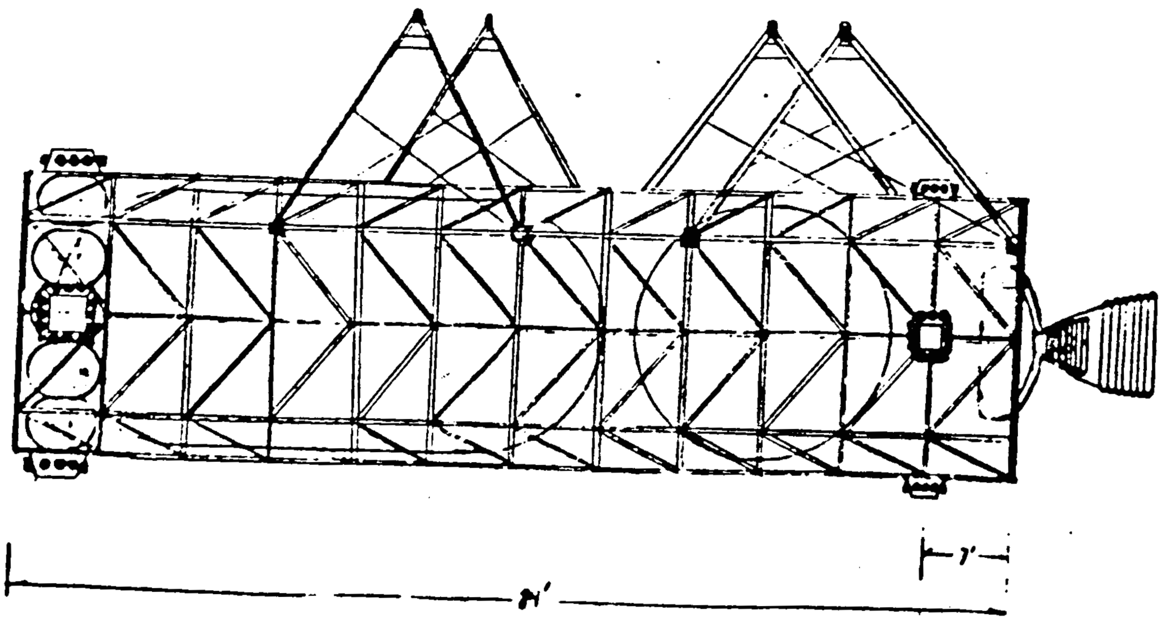
AFT VIEW

fig. 6.1



CROSS SECTION
of
COMBINED
STRUCTURE





BOOSTER

fig. 6.2

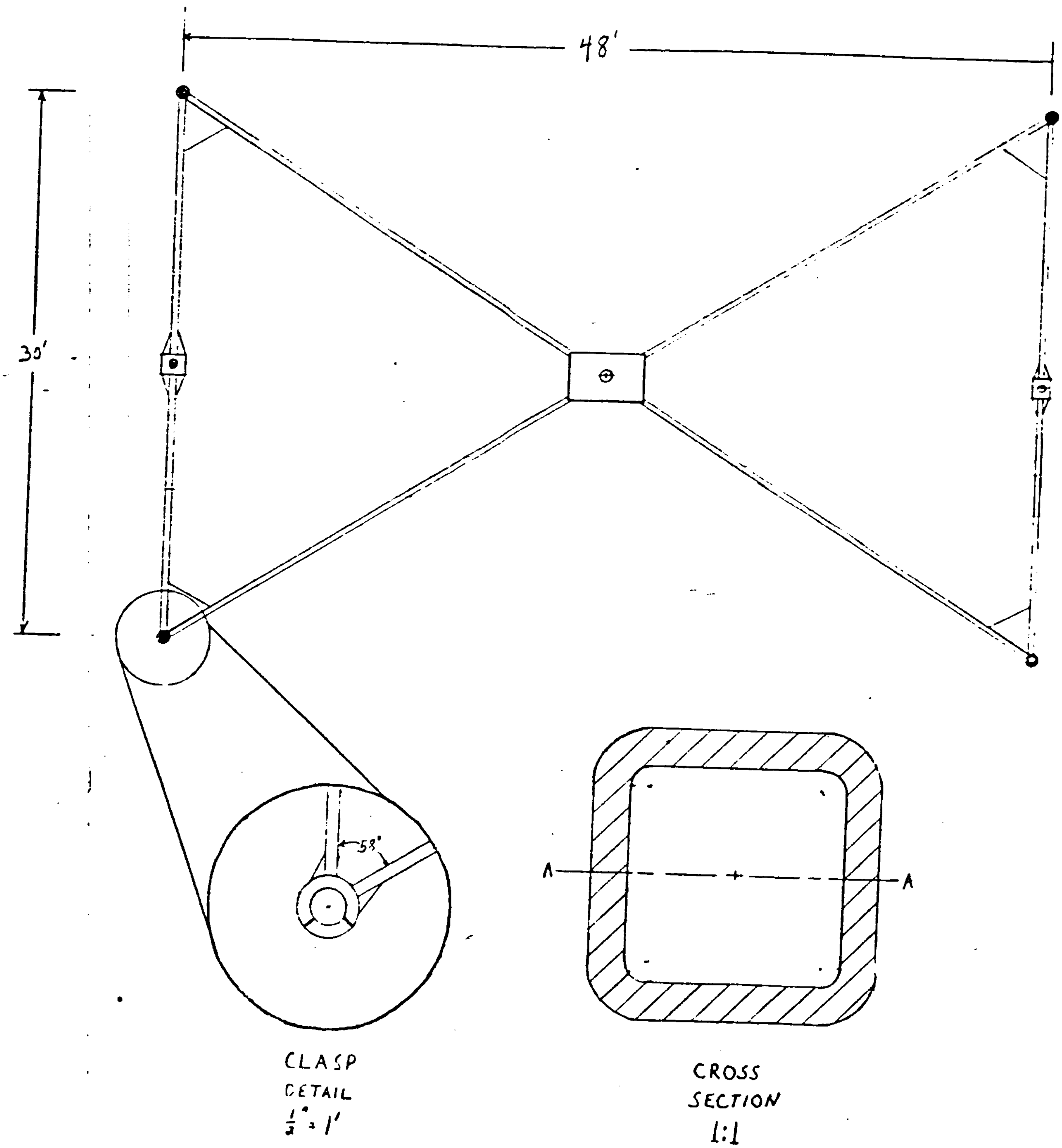


fig. 6-3

CHAPTER 7 - CONCLUSIONS AND RECOMMENDATIONS

Due to time constraints, an in-depth analysis of both vehicles was not possible. Because the tug was considered the principle aspect of the project and work in the area of lunar landers had already been accomplished it was decided that only a preliminary design of the lander would be warranted. Further development of the project should include a detailed look into the lander's subsystems as well as specifying in more detail propulsion and structural design requirements.

Although the tug's propulsive and structural design was carried out in some detail and the feasability of the project verified, there are considerable portions of its development that were beyond the scope of this research. A black box approach to subsystem design was used and therefore allows for further research. It is recommended that a more precise determination of the tug's power requirements along with life support and communication needs be made.

Because of the technique employed to determine the return orbit, only a rough inquiry was made into the trajectory and energy of the aerobraking maneuver. In order to go beyond this a more sophisticated model must be developed. This will allow a more accurate and detailed assessment of the energy that will be saved and the thermal and structural requirements that will be placed on the tug during this portion of its flight.

This project should be viewed as a preliminary look into the overall feasibility of a lunar transport system to logistically support proposed lunar activities. All calculations and subsystem design rely heavily on current technology, especially that of the STS, and demonstrate the capability of meeting the logistical support problem successfully.

APPENDIX 1

ORBIT CALCULATIONS

OUTBOUND ORBIT

$V_{o_\oplus} = 7.613$ km/s
$r_{o_\oplus} = 6878$ km
$e_{o_\oplus} = 0$

$e_1 = .9649$
$Vp_1 = 10.671$ km/s
$a_1 = 195820$ km
$p_1 = 13514.4$ km
$r_1 = 375933$ km
$\lambda_1 = 1.3566$ rad
$\gamma_1 = 3.0075\times10^{-3}$ rad

$e_2 = 1.3627$
$Vp_2 = 2.508$ km/s
$rp_2 = 1836.39$ km
$p_2 = 4338.85$
$r_2 = 66{,}300$ km

$V_{o_M} = 1.631$ km/s
$r_{o_M} = 1838$ km

$$\Psi_1 = \cos^{-1}\frac{a_1 - r_1}{(a_1)(e_1)} = \cos^{-1}\frac{(195820 - 375933)}{(195820)(.9649)} = 162.41^0 = 2.8346 \text{ rad}$$

$$tp_1 = \sqrt{a^3/\mu_\oplus}\left[\Psi_1 - e(\sin\Psi_1)\right] = \sqrt{\frac{195820^3}{3.986\times10^5}}\left[(2.8346 - .9649(\sin 2.8346)\right] = 349{,}032.68 \text{ sec}$$

$tp_1 = 4.04$ days

$$a_2 = p_2/\ 1-e_2^2 = 4338.85/(1 - 1.3627)^2 = -5{,}063.12 \text{ km}$$

$$H_1 = \cosh^{-1}\frac{(a_2 - r_2)}{(a_2)(e_2)} = \cosh^{-1}\frac{(-5{,}063.12 - 66{,}300)}{(-5063.12)(1.3627)} = 3.0271$$

$$tp_2 = \sqrt{-a^3/\mu_M}\left[e \sinh H - H\right] = \sqrt{\frac{(5{,}063.12)^3}{4887}}\left[1.3627\sinh(3.0271) - 3.0271\right] = 56{,}697.07 \text{ sec}$$

$tp_2 = .66$ days

T.O.F. $= tp_1 + tp_2 = (4.04 + .66)$ days $=$ 4.70 days

$$\Delta V_1 = V_{p_1} - V_{o_\oplus} = (10.671 - 7.613) \text{ km/s} = 3.058$$
$$\Delta V_2 = V_{p_2} - V_{o_M} = (2.508 - 1.631) \text{ km/s} = .877$$

$\Delta V_{tot} = \Delta V_1 + \Delta V_2 =$ 3.935 km/s

$$\gamma_0 = \Psi_1 - \Psi_2 - \gamma_1 - w_m(tp_1) = 2.8346 \text{ rad} - 0 - 3.0075\times10^{-3} - 2.64\times10^{-6}\ (349{,}032.68) = 1.910 \text{ rad}$$

$\gamma_o = 109.44$

OUTBOUND ABORT ORBIT

$$Vp_A = Vp_1 - \Delta V_2 = (10.671 - .877) = 9.794 \text{ km/s} \qquad aA = 1/2\left[\frac{1}{1/r - \frac{Vpa^2}{2}}\right]$$

$$aA = 19,946.6 \text{ km}$$

$$\Upsilon = \frac{2\pi a^{3/2}}{\mu_{\oplus}^{1/2}} = \frac{2\pi(19946.6)^{3/2}}{(3.986 \times 10^5)^{1/2}} = 28,035.91 \text{ sec} = \underline{.324 \text{ days}}$$

$$r_{a_A} = 2a_A - r_{o_{\oplus}} = 2(19946.6) - (6878) = \underline{33,015.2 \text{ km}}$$

LANDER ORBIT

$V_{O_M} = 1.631$ km/s
$r_{O_M} = 1838$ km

TRANSFER TO 5 km ALTITUDE

rp = 1743 km   ra = 1838   a = 1790.5

$$Va_T = \sqrt{2\mu_M(1/ra - 1/2a)} = \sqrt{2(4887)(1/1838 - 1/2(1790.5))} = 1.609 \text{ km/s}$$

$$Vp_T = \sqrt{2(4887)(1/1743 - 1/2(1790.5))} = 1.696 \text{ km/s}$$

$$\Delta V_1 = V_{O_M} - Va_T = (1.631 - 1.609) = \underline{.022 \text{ km/s}}$$

Final elliptical descent to surface ($e_D = .96$)

$$ra_D = P_D/(1-e_D) \qquad P_D = ra_D(1-e_D) = 1743(1-.96) = 69.72 \text{ km}$$

$$a_D = p_D/(1-e_D^2) = 69.72/(1-.96^2) = 889.29 \text{ km}$$

$$Va_D = \sqrt{2(4887)(1/1743 - 1/2(889.29))} = .335 \text{ km/s}$$

$$\Delta V_2 = Vp_T - Va_D = (1.696 - .335) = 1.361 \text{ km/s}$$

Actual Landing $r = R_M = 1738$ km

$$Vsurface = \sqrt{2(4887)(1/1738 - 1/2(889.29))} = .358 \text{ km/s}$$

$$\Delta V_3 = Vsurface - Vlanding = (.358 - 0) = .358 \text{ km/s}$$

$$\Delta V_T = \Delta V_1 + \Delta V_2 + \Delta V_3 = (.022 \text{ km/s} + 1.361 \text{ km/s} + .358 \text{ km/s})$$

$$\Delta \underline{V_T = 1.741 \text{ km/s}}$$

LANDER ABORT ORBIT

Return to parking orbit after second descent burn ($Vs/c = Va_D$)

$\Delta V_1 a = Vp_T - Va_D = \Delta V_2 = 1.361$ km/s
$\Delta V_2 a = Vo_M - Va_T = \Delta V_1 = .022$ km/s

$\Delta Vtot = \Delta V_T + \Delta V_{1A} + \Delta V_{2A} = (1.741 + 1.361 + .022)$ km/s

ΔVtot = 3.124 km/s

TIME OF FLIGHT

Transfer to 5 km altitude

$$T = 1/2\,\Upsilon = 1/2\left[\frac{2\ a\text{^}3/2}{\mu_M\text{^}1/2}\right] = \frac{1790.5\text{^}3/2}{4887\text{^}1/2} = 3404.79s \times \frac{Hr}{3600\ s}$$

$$= \underline{.946\ Hr}$$

Final Descent

$$\Psi = \cos-1\ (a_D - r_D)/ae_D$$

$$\Psi_1 = \cos-1\ \frac{889.29 - 1743}{(889.29)(.96)} = 180 = 3.1416\ Rad$$

$$\Psi_2 = \cos-1\ \frac{889.29 - 1738}{889.29(.96)} = 173.79 = 3.033\ Rad$$

$$T_D = \sqrt{a\text{^}3/\mu_M}\left[\Psi_1 - \Psi_2 - e(\sin\Psi_1 - \sin\Psi_2)\right]$$

$$T_D = \sqrt{889.29\text{^}3/4887}\left[3.1416 - 3.033 - .96(\sin 3.1416 - \sin 3.033)\right]$$

$$T_D = 1.803s \times \frac{Hr}{3600\ s} = 5 \times 10\text{^}{-4}\ Hr$$

$$Ttot = T_T + T_D = \underline{.9465\ Hr}$$

RETURN ORBIT TO ALTITUDE OF 200 KM AT PERIGEE

Vp_1 = 10.917 km/s	e_2 = 1.4127
e_1 = .9668	r_2 = 66300 km
a_1 = 198238 km	P_2 = 4436.46 km
r_1 = 357716 km	

$$\Psi_1 = \cos-1\ (a_1 - r_1)/ae_1 = \cos-1\ \frac{198238 - 357716}{(198238)(.9668)} = 146.315^\circ$$

$\psi_1 = 2.5537$ Rad

$$tp = \sqrt{a^3/\mu_\oplus}\left[\psi_1 - e\sin\psi_1\right] = \sqrt{\frac{198238^3}{3.986\times10^5}}\left[2.5537 - .9668 \sin 2.5537\right]$$

$= 282{,}047.63$ sec

<u>tp = 3.26 days</u>

$a_2 = P_2/(1 - e_2^2) = (4436.46 \text{ km})/(1 - 1.4127^2) = -4455.5$ km

$$H_1 = \cosh^{-1}\frac{(a_2 - r_2)}{(a_2)(e_2)} = \cosh^{-1}\frac{(-4455.5 - 66300)}{(-4455.5)(1.4127)} = 3.1108$$

$tp = \sqrt{-a^3/\mu_M}\left[e\sinh H - H\right] = 54{,}060.95$ s $= .63$ days

T.O.F. = (3.26 + .63) days = <u>3.89 Days</u>

<u>RETURN ORBIT WITH BRAKING</u>

$V = \sqrt{2\mu(1/r - 1/2a)}$ $\quad$ Vp = 10.917 at altitude of 200 km

$V_{ALT=400} = \sqrt{2(3.986 \times 10^5)\ (1/6778 - 1/198238)} = 10.752$

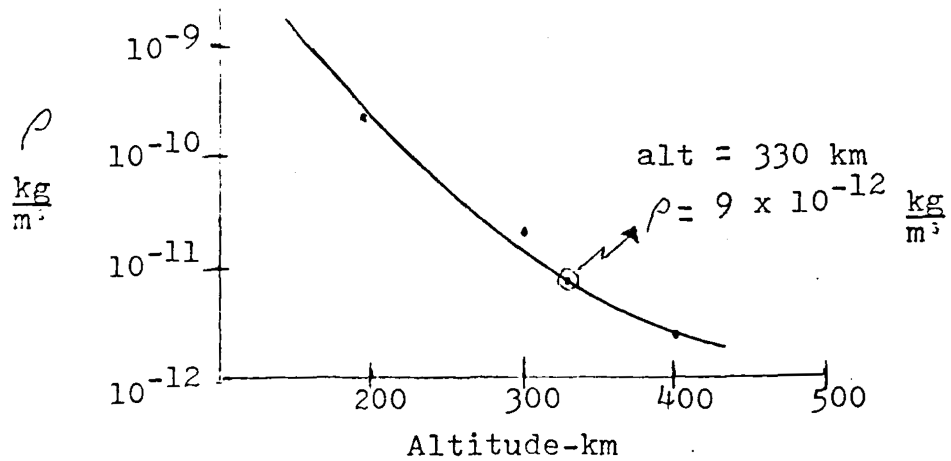
Vavg = (2(10.752) + 10.917)/3 = 10.807 km/sec

$$\cos\psi = \frac{198238 - 6778}{(198238)(.9668)} \quad \psi = 2.595 = .0423 \text{ rad}$$

$tp = \sqrt{198238^3/3.986\times10^5}\left[.0423 - .9668 \sin 2.595\right] = 211.654$ s

$\Delta T = 2tp = 423.307$ s

Alt avg = (2(400) + 200)/3 = <u>330 km</u>



$K = 1/2 \times C \times A = 500\ M^2 \qquad V = 10.807 \times 10^3\ M/S$

$F_D = K \times V^2 = (9.0 \times 10^{-12}\ kg/M^3)(10.807 \times 10^3)^2\ M^2/s^2\ (500\ M^2)$

$F_D = .5256$

$F = MA \qquad F/M = dV/dt \qquad dV = F/M\ dt \qquad \Delta V = F/M\ \Delta t = F_D\ \Delta T$

$\Delta V = (.1613)(423.307) = .223\ km/s$

$V_p = 10.694 = \sqrt{2\mu_\oplus(1/r - 1/2a)}$

$2\left[(1/6578 - 10.694^2/2(3.986 \times 10^5))\right] = 1/a \qquad a = 58358.3\ km$

$V_{ALT=500KM} = \sqrt{2(3.986 \times 10^5)(1/6878 - 1/2(58358.3))} = 10.444\ km/s$

$V_{park} = 7.613\ km/s$

$V = (10.444 - 7.613) = 2.831\ km/s$

$V = (2.532 - 1.631) = .901\ km/s$

$V = 3.732\ km/s$

$V_{saved} = (3.935 - 3.732) = .203\ km/s = 5.2\%$

APPENDIX 8

WEIGHT ESTIMATION OF SPACE TUG HYDROGEN AND OXYGEN TANKS

Both the hydrogen and oxygen tanks are cylindrical with hemispherical ends

Using the existing information of the space shuttle external tank (ET)

Total surface area: 13,614 ft^2
Total Weight: 76,500 lbs (Ref. x)

and the equation for the surface area of the space tug's tanks

4 R + 2 RL = TOTAL SURFACE AREA

the following computations were made:

OXYGEN: Length(L) = 2.170 ft
Radius(R) = 6.25 ft

Total Surface Area = 576.09 ft^2

Set up proportion with space shuttle ET information

$$576.09 \text{ ft}^2 \times \frac{76{,}500 \text{ lbs}}{13{,}614 \text{ ft}^2} = 3237 \text{ lbs}$$

WEIGHT ESTIMATE OF OXYGEN TANK = 3250 LBS

HYDROGEN: Length(L) = 20.156 ft
Radius(R) = 6.25 ft

Total Surface Area = 1282.4 ft^2

$$1282.4 \text{ ft}^2 \times \frac{76{,}500 \text{ lbs}}{13{,}614 \text{ ft}^2} = 7{,}206 \text{ lbs}$$

WEIGHT ESTIMATE OF HYDROGEN TANK = 7,200 LBS and using the yield stress of Aluminum at 35,000 psi

Allowable stress = 35,000 psi/1.5 = 23,333 psi

Minimum Cross Section of Strut Members = 36,744 lbs/23,333 psi

= 1.6 in^2

... VEHICLE

Crew Size = 4
Maximum Mission Duration (days) = 1100

Habitat Module			7850 kg
Two Spacelab Short Modules		6350	
Support Equipment and Furnishings		1500	
Experiment Module			4540
Spacelab Short Module		3800	
Cruise Science		700	
Communications		40	
Logistics Module			1870
Consumables	18700		
Containment Structure		1870	
Power Module			1490
AACS		450	
Structure and Thermal Protection		200	
Power Conditioning and Cabling		340	
Solar Arrays and Radiator		500	
EVA Station			1406
MMU		310	
MMU Storage and Recharge		300	
N2 Propellant		236	
Airlock		400	
EVA Suits		160	
Module Adapters and Connectors			886
Contingency (10% of dry mass)			1804
Subtotal (w/o consumables)			19846
Consumables			18700
Total at Launch			38546 kg

MARS RETURN VEHICLE

Crew Size = 4
Return Mission Duration (days) = 864
Samples (kg) = 400

Habitat Module			7850 kg
Two Spacelab Short Modules		6350	
Support Equipment and Furnishings		1500	
Experiment Module			4625
Spacelab Short Module		3800	
Cruise Experiments		725	
Communications and HGA		100	
Logistics Module			1469
Consumables	14688		
Containment Structure		1469	
Power Module			1490
AACS		450	
Structure and Thermal Protection		200	
Power Conditioning and Cabling		340	
Solar Arrays and Radiator		500	
EVA Station			1486
MMU		310	
MMU Storage and Recharge		300	
N2 Propellant		236	
Airlock		400	
EVA Suits		160	
Emergency Escape Spheres		80	
Aerobrake Return System			4432
Crew Capsule		3100	
Consumables		51	
Propellant		154	
Tankage and Engine		87	
Shield		1041	
Contingency (10% of dry mass)			2135
Subtotal (w/o consumables)			23487
Consumables			14688
Total at Launch			38175 kg

ORBITER/RENDEZVOUS VEHICLE

Crew Size = 3
Flight Time (days) = 30

Habitat Module			3240 kg
Structure		3000	
EVA		240	
Ascent Capsule			2750
Ascent Propulsion System			13218
Surface Equipment			2850
Rovers		800	
ISPP		1000	
Science		900	
Misc.		150	
Power Module			3300
Nuclear Reactor		1650	
Power Distribution and Cabling		550	
Lander Structure			5180
)escent Propulsion			2696
ontingency (10% of dry mass)			1963
ubtotal (w/o consumables)			35196
nsumables			383
robrake			16875
(Aerobrake Contingency - 10%)			1687
tal at Launch			54141
and Total Outbound at Launch			112953
Interplanetary Vehicle		38546	
Orbiter/Rendezvous Vehicle		19545	
Lander Vehicle		54141	
Crew and Personal Gear		720	
d Total Return at Mars Departure			39295 kg
Return Vehicle		38175	
Crew and Personal Gear		720	
amples		400	

MARS LANDER VEHICLE

Crew Size = 1
Flight Time (days) = 30

Habitat/Experiment/Logistics Module			4600 kg
Spacelab Short Module		3800	
Experiments		200	
Support Equipment and Furnishings		500	
Communications and HGA		100	
Logistics Support Structure			50
Power Module			1180
AACS		200	
Structure and Thermal Protection		100	
Power Conditioning and Cabling		230	
Solar Arrays and Radiator		260	
Batteries		390	
EVA Station			963
MMU		155	
MMU Support Station		150	
N2 Propellant		118	
Airlock		400	
EVA Suit		80	
Emergency Escape Spheres		60	
Departure Vehicle			10293
Crew Capsule		3100	
Consumables		85	
Propulsion Syst. (propellant, engine, & tanks)		6769	
Adapter		338	
Circularization Propulsion System			1202
Contingency (10% of dry mass)			1130
Consumables			128
Total at Launch			19545 kg

Appendix 9

Appendix 10

1. Weight Estimate

payload	- 65000 lb
engines (4 * 296 lb)	- 1184 lb
structure	- 20000 lb
total dry weight	86184 lb roughly= 85000 lb

2. Engine Selection

Considering the gravity of the moon = 1/5 earth gravity, choose to accellerate at 1/5 earth g.

weight of ship + fuel(guess) = 85000 + 40000 = 125000

125000 lb / 5 =25000 lb

Knowing that the OMS engine thrust = 6000 lb,

select four OMS engines

6000 lb * 4 = 24000 lb

3. Fuel Estimate

Total mass of ship at start of flight will be

mass of ship = dry weight + original mass of fuel

= 85000 lb + MF

or = 85000 lb + fuel flow rate * burn time

For the OMS engine, fuel burned in one second is equal to 11.9 lb N2O4 and 7.21 lb H2, or

FFR = 19.11 lb/sec

therefore, total fuel onboard > 19.11 * burn time.

These criteria are interatively solved by computer in

Appendices L2 and L3. L2 assumes a required delta V of 4 km/sec, and L3 assumes a required delta V of 3.17 km/sec. The solution for L3 is 187000lb with a burn time of 2440.75 seconds and an average acceleration of 4.33 ft/sec^2.

4. G-Loading

The g-loading of the structure in flight is found by simply applying F=ma, or a=F/m where

m = mass of ship + original mass of fuel - FFR * time

= 85000 lb + 187000 lb - 19.11 lb/sec * t

and F = thrust = 24000 lb

This was solved for in a second- by-second format by computer in the program in Appendix L4.1. The g-profiles in ft/sec^2 and in earth g's are in Appendices L4.2 and L4.3.

APPENDIX C-1

```
[illegible]

[illegible]

[illegible] REM R0 = RADIUS OF [illegible]
[illegible] REM MU1 = GRAVITATIONAL [illegible]
[illegible] REM MU2 = GRAVITATIONAL [illegible]
[illegible] REM RS = DISTANCE TO THE [illegible] SPHERE OF INFLUENCE
[illegible] REM VM = VELOCITY OF THE [illegible]
[illegible] REM E = ENERGY
[illegible] REM ECC = ECCENTRICITY
[illegible] REM EP = THE DIRECTION OF THE SELENOCENTRIC VELOCITY [illegible]
[illegible] REM TH = HEADING ANGLE
190 REM GA = PHASE ANGLE AT DEPARTURE
200 REM LAM = THE ANGLE WHERE THE ORBIT CROSSES THE LUNAR SPHERE OF INFLUENCE
210
[illegible]
[illegible] PRINT "INPUT ERROR"
[illegible] INPUT ERROR
[illegible] -
[illegible] FOR V0 = 10.[illegible] TO [illegible] STEP [illegible]
[illegible]     LET R0 = 6378
[illegible]     LET MU1 = 3.986E5
[illegible]     LET MU2 = [illegible]
[illegible]     LET RS = [illegible]
[illegible]     LET VM = [illegible]
[illegible]     REM TH = THETA
[illegible]
[illegible]     FOR LAM = [illegible] TO [illegible] STEP [illegible]
[illegible]         LET R1 = [illegible]*COS(LAM)))
[illegible]         LET [illegible] = [illegible]
[illegible]         LET E1 = [illegible]

390         IF [illegible]+(E1+MU1/R1)<0
400         THEN GOTO [illegible]  REM NO SOLUTION BECAUSE ENERGY<0
410         IFEND
420
430         LET V1 = SQR(2*(E1-MU1/R1))
440         LET P1 = [illegible]/MU1
450         LET A = [illegible](2*E1)
460         LET ECC1 = SQR(1-P1/A)
470
[illegible]         LET X = [illegible]/57.3
[illegible]         LET Y = (RS/R1)/57.3*SIN(LAM)
[illegible]
[illegible]         LET TH1 = ATN(SQR(1-X*X)/X)
[illegible]         LET GA1 = ATN(Y/SQR(1-Y*Y))

[illegible]         LET [illegible] = [illegible]
[illegible]         LET [illegible] = SQR([illegible]*VM*COS(TH1-GA1))

[illegible]         LET [illegible] = [illegible]-(V1/V2)*COS(LAM-GA1-TH1)

[illegible]         LET [illegible]
[illegible]         LET [illegible]
[illegible]         LET [illegible]
[illegible]         LET [illegible]
[illegible]         LET [illegible]
[illegible]         LET [illegible]
[illegible]         LET [illegible]
```

```
[illegible]
[illegible]          PRINT "VI=";VI
[illegible]          PRINT "LAM=";LAM
[illegible]          PRINT "R1=";R1
[illegible]          PRINT "P1=";P1
[illegible]          PRINT "a1=";A
[illegible]          PRINT "EDDEN=";EDD1
[illegible]          PRINT "RP=";RP1
[illegible]          PRINT " P=";VP2
[illegible]          PRINT "P2=";P2
[illegible]          PRINT "EDDEN2=";EDD2
[illegible]          PRINT "GAMMA=";GA1
[illegible]          PRINT
[illegible]      IFEND
[illegible]
340      NEXT LAM
350  NEXT [illegible]
360  END

Ready
```

RUN OutBound Orbit Solutions

```
ORBIT1     24 Feb 86  21:20

INPUT ERROR
? 2
V0= 10.671
LAM= 1.3566
R1= 375738.
P1= 13514.4
a1= 195820.
ECCEN= .964876
RP= 1836.39
VP= 2.50751
P2= 4338.85
ECCEN2= 1.3627
GAMMA= 3.00753e-3

V0= 10.683
LAM= .9942
R1= 346763.
P1= 13544.8
a1= 224028.
ECCEN= .969298
RP= 1838.66
VP= 2.59076
P2= 4643.88
ECCEN2= 1.52569
GAMMA= 2.60171e-3

V0= 10.695
LAM= .7112
R1= 347743.
P1= 13575.3
a1= 261765.
ECCEN= .973725
RP= 1837.88
VP= 2.63904
P2= 4813.77
ECCEN2= 1.61919
GAMMA= 2.62942e-3

Resource increment exceeded; type "EXPLAIN RESOURCE INCREMENT".
Do you wish to continue? YES

V0= 10.703
LAM= .7378
R1= 333296.
P1= 13595.6
a1= 274922.
ECCEN= .976679
RP= 1839.7[illegible]
VP= 2.6[illegible]
P2= 4[illegible]
ECCEN2= 1.7[illegible]
GAMMA= [illegible]

V0= [illegible]
LAM= [illegible]
```

RETURN TO AN ALTITUDE OF 200 KM

```
[illegible]    11 Mar 86  [illegible]

[illegible] ERROR


Resource increment exceeded: type "EXPLAIN RESOURCE INCREMENT".
Do you wish to continue? YES
V0= 10.917
LAM= 1.0761
R1= 357716.
P1= 12987.7
a1= 178238.
ECCEN= .966818
RP= 1838.77
VP= 3.53228
P2= 4436.46
ECCEN2= 1.41278
GAMMA= 2.84632e-3

V0= 10.925
LAM= .9265
R1= 348633
P1= 12956.7
a1= 217106.
ECCEN= .969701
RP= 1839.04
VP= 3.57747
P2= 4597.57
ECCEN2= 1.49999
GAMMA= 2.65352e-3

V0= 10.927
LAM= 1.0455
R1= 355808.
P1= 12961.4
a1= 222401.
ECCEN= .970423
RP= 1838.07
VP= 3.57754
P2= 4592.95
ECCEN2= 1.49879
GAMMA= 2.8135e-3

V0= 10.932
LAM= .9483
R1= 344173.
P1= 12973.3
a1= [illegible]
ECCEN= .972227
RP= [illegible]
VP= [illegible]
P2= [illegible]
ECCEN2= [illegible]
GAMMA= [illegible]

V0= [illegible]
LAM= [illegible]
R1= [illegible]
```

APPENDIX C-2

```
100 ! FUEL REQUIREMENT ANALYSIS FOR THE BOOSTER AND TUG STAGES
110 ! OF THE GEOSEL TRANSPORT SYSTEM
120 !
130 !  This program is based upon several assumptions that yeild fairly
140 !  good results.  The first of these is that the SSME operation is
150 !  linear over the throttling range of 70 to 100% thrust.  This assumptio
160 !  is fairly accurate according to a discussion with Professor Saarlas
170 !  of the Naval Academy Aerospace Department.  The only problem with it
180 !  is that the mass flow at 70% thrust is not necessarally 70% of the
190 !  mass flow at 100% thrust.  No data was available about the mass flow
200 !  at this point, so it was assumed that the mass flow percentage was the
210 !  same as the thrust percentage.  The second assumption was that small
220 !  engines will be designed to have thrust and mass flow characteristics
230 !  proportional to those of the SSME.  For design purposes, the engines
240 !  were assumed to be computer controlled with constant updating, and
250 !  that there were instantaneous responses to change of throttle commands
260 !
270 !
280 DIM M(5000),TH(5000),DM(5000)
290 OPEN #F:"MTFILE"
300 SCRATCH #F
310 PRINT "WHAT IS THE EMPTY, UNLOADED STRUCTURE MASS (Kg)";
320 INPUT MI
330 PRINT
340 PRINT "WHAT IS THE MASS OF ADDITIONAL STAGES AND THEIR FUEL ";
350 PRINT "THAT IS BEING CARRIED DURING THE BURN (Kg)";
360 INPUT PL
370 PRINT
380 PRINT "HOW MANY PAYLOAD CONTAINERS WILL BE CARRIED BY THE ";
390 PRINT "TOTAL STRUCTURE";
400 INPUT NL
410 PRINT
420 PRINT "WHAT IS THE DESIRED DELTA-V IN Km/s";
430 INPUT DV
440 PRINT
450 PRINT "WHAT IS THE MASS OF FUEL THAT IS DESIRED TO REMAIN ";
460 PRINT "AFTER THE BURN IS COMPLETED (Kg)";
470 INPUT FI
480 PRINT
490 LET DV=DV*1000
500 PRINT "WHAT IS THE DESIRED ACCELERATION IN g'S";
510 INPUT G
520 PRINT
530 LET A=G*9.807                          ! a IN m/s
540 LET TH100 = 2090760                    ! THRUST OF ONE ENGINE @ 100%
550 LET DM100 = 467.2049                   ! TOTAL MASS FLOW IN Kg/s @ 100%
Ready
```

```
560 LET TR=DM100/TH100
570 LET TB=DV/A                                 ! REQUIRED BURN TIME IN SECONDS
580 LET M(1) = MI + 29484 * NL + FI + PL        ! TOTAL MASS AFTER DELTA-V BURN
590 LET DM(1)=TR*A*M(1)                         ! MASS FLOW IN Kg/S
600 LET TH(1)=A*M(1)                            ! THRUST REQUIRED AT TIME ONE
610 LET DT=TB/400                               ! TIME INCREMENT USED
620 PRINT "BURN TIME =";TB
630 PRINT "DT =";DT
640 LET SDT= INT(1/DT+.5)                       ! AVOIDS PROBLEMS IN MATRICES
650 PRINT "SCALED DT =";SDT
660 PRINT
670 FOR T = 2 TO 1+(TB+DT)*SDT STEP 1
680     LET M(t)= M(t-1) + DM(t-1)*DT
690     LET TH(t) = A*M(t)
700     LET DM(t) = TR*TH(t)
710     PRINT #F:(TB-(t-1)/SDT),M(t)-M(1)
720 NEXT t
730 PRINT "TIME-0","THRUST","FUEL MASS","MASS FLOW","% OPERATION"
740 FOR T=1 TO TB*SDT+6 STEP 5*SDT
750     PRINT (t-1)/SDT,TH(t),M(t)-M(1),DM(t),DM(t)/DM100*100
760 NEXT T
770 PRINT TB,TH(TB*SDT+1),M(TB*SDT+1)-M(1),DM(TB*SDT+1),DM(TB*SDT+1)/DM100*10
780 LET MF=M(TB*SDT+1)-M(1)+FI
790 LET MH=MF*1/7                               ! TOTAL MASS OF HYDROGEN
800 LET MO=MF*6/7                               ! TOTAL MASS OF OXYGEN
810 LET VH=MH/70                                ! TOTAL VOLUME OF LH2
820 LET VO=MO/1149                              ! TOTAL VOLUME OF LOX
830 PRINT
840 PRINT "VOLUME OF H2 IS"; VH ;"CUBIC METERS"
850 PRINT "VOLUME OF O2 IS"; VO ;"CUBIC METERS"
860 PRINT "TOTAL FUEL MASS IS"; MF ;"Kg"
870 END
Ready
```

FUEL 01 May 88 21:[illegible]

WHAT IS THE EMPTY, UNLOADED STRUCTURE MASS (Kg)? 11340

WHAT IS THE MASS OF ADDITIONAL STAGES AND THEIR FUEL
THAT IS BEING CARRIED DURING THE BURN (Kg)? 56502

HOW MANY PAYLOAD CONTAINERS WILL BE CARRIED BY THE TOTAL STRUCTURE? 4

WHAT IS THE DESIRED DELTA-V IN Km/s? 3.059

WHAT IS THE MASS OF FUEL THAT IS DESIRED TO REMAIN
AFTER THE BURN IS COMPLETED (Kg)? 1369

WHAT IS THE DESIRED ACCELERATION IN g'S? 2

BURN TIME = 155.909
DT = .389773
SCALED DT = 3

TIME-0	THRUST	FUEL MASS	MASS FLOW	% [illegible]
0	3.6707e+6	0	820.281	175.[illegible]
5	3.7859e+6	4853.51	841.524	180.121
10	3.88259e+6	9832.89	863.359	184.[illegible]
15	3.98378e+6	14941.4	885.749	189.[illegible]
20	4.08558e+6	20192.4	908.72	194.[illegible]
25	4.17202e+6	25559.3	932.287	199.54[illegible]
30	4.28022e+6	31075.7	956.465	204.721
35	4.39122e+6	36735.1	981.271	210.03
40	4.50511e+6	42541.3	1006.72	215.477
45	4.62194e+6	48498.1	1032.83	221.065
50	4.74181e+6	54609.4	1059.61	226.798
55	4.86478e+6	60879.1	1087.09	232.68
60	4.99095e+6	67311.5	1115.29	238.715
65	5.12039e+6	73910.7	1144.21	244.905
70	5.25318e+6	80681.	1173.88	251.257
75	5.38942e+6	87626.9	1204.33	257.773
80	5.52919e+6	94752.9	1235.56	264.456
85	5.67258e+6	102064.	1267.6	271.317
90	5.81968e+6	109564.	1300.48	278.353
95	5.97062e+6	117259.	1334.21	285.572
100	6.12547e+6	125154.	1368.81	292.978
105	6.28433e+6	133253.	1404.31	300.576
110	6.4473e+6	141562.	1440.73	308.371
115	6.61451e+6	150087.	1478.09	316.369
120	6.78605e+6	158833.	1516.42	324.573
125	6.96204e+6	167806.	1555.75	332.991
130	7.1426e+6	177011.	1596.1	341.627
135	7.32784e+6	186455.	1637.49	350.487
140	7.51788e+6	196144.	1679.96	359.576
145	7.71285e+6	206085.	1723.53	368.902
150	7.91287e+6	216283.	1768.22	378.469
155	8.11809e+6	226746.	1814.08	388.284
155.909	8.15977e+6	228870.	1823.4	390.277

VOLUME OF H2 IS 469.979 CUBIC METERS
VOLUME OF O2 IS 171.759 CUBIC METERS
TOTAL FUEL MASS IS 230239. Kg

Ready

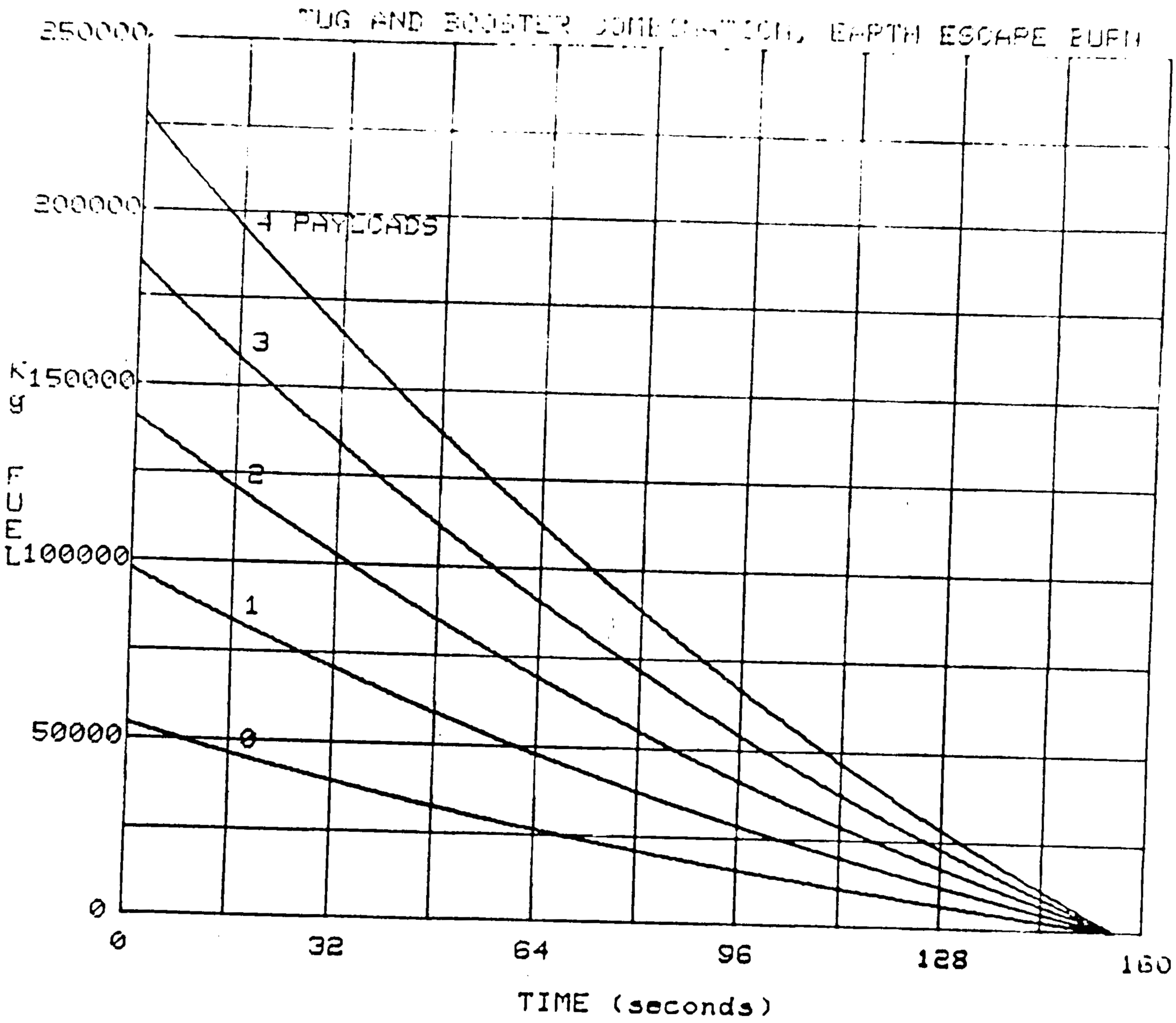
```
RUN
FUEL     30 Apr 86  23:08

WHAT IS THE EMPTY, UNLOADED STRUCTURE MASS (Kg)? 11340

WHAT IS THE MASS OF ADDITIONAL STAGES AND THEIR FUEL
THAT IS BEING CARRIED DURING THE BURN (Kg)? 0

HOW MANY PAYLOAD CONTAINERS WILL BE CARRIED BY THE TOTAL STRUCTURE? 0

WHAT IS THE DESIRED DELTA-V IN Km/s? .5

WHAT IS THE MASS OF FUEL THAT IS DESIRED TO REMAIN
AFTER THE BURN IS COMPLETED (Kg)? 0

WHAT IS THE DESIRED ACCELERATION IN g'S? 3

BURN TIME = 16.9947
DT = 4.24867e-2
SCALED DT = 24

TIME-0         THRUST         FUEL MASS      MASS FLOW      % OPERATION
 0              333634.        0              74.5545        15.9576
 5              345005.        386.496        77.0955        16.5014
 10             356764.        786.164        79.7231        17.0636
 15             368923.        1199.45        82.4402        17.6454
 16.9947        373902.        1368.69        83.5529        17.8826

VOLUME OF H2 IS 2.79324 CUBIC METERS
VOLUME OF O2 IS 1.02103 CUBIC METERS
TOTAL FUEL MASS IS 1368.69 Kg

Ready
```

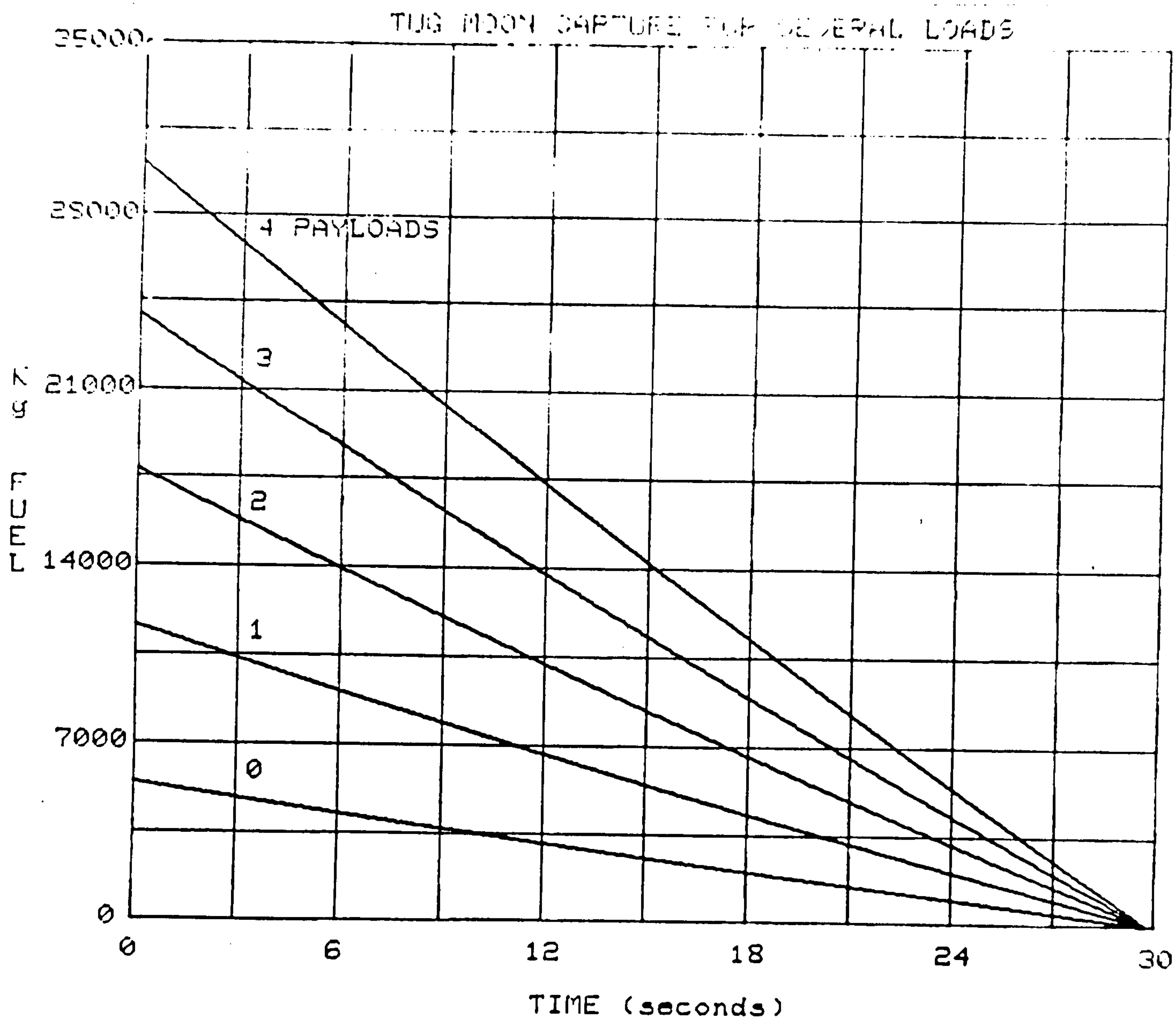
```
RUN

FUEL     30 Apr 86  23:19

WHAT IS THE EMPTY, UNLOADED STRUCTURE MASS (Kg)? 26310

WHAT IS THE MASS OF ADDITIONAL STAGES AND THEIR FUEL
THAT IS BEING CARRIED DURING THE BURN (Kg)? 0

HOW MANY PAYLOAD CONTAINERS WILL BE CARRIED BY THE TOTAL STRUCTURE? 4

WHAT IS THE DESIRED DELTA-V IN Km/s? 1.490\\01

WHAT IS THE MASS OF FUEL THAT IS DESIRED TO REMAIN
AFTER THE BURN IS COMPLETED (Kg)? 0

WHAT IS THE DESIRED ACCELERATION IN g'S? 3

BURN TIME = 47.619
DT = .119048
SCALED DT = 8
```

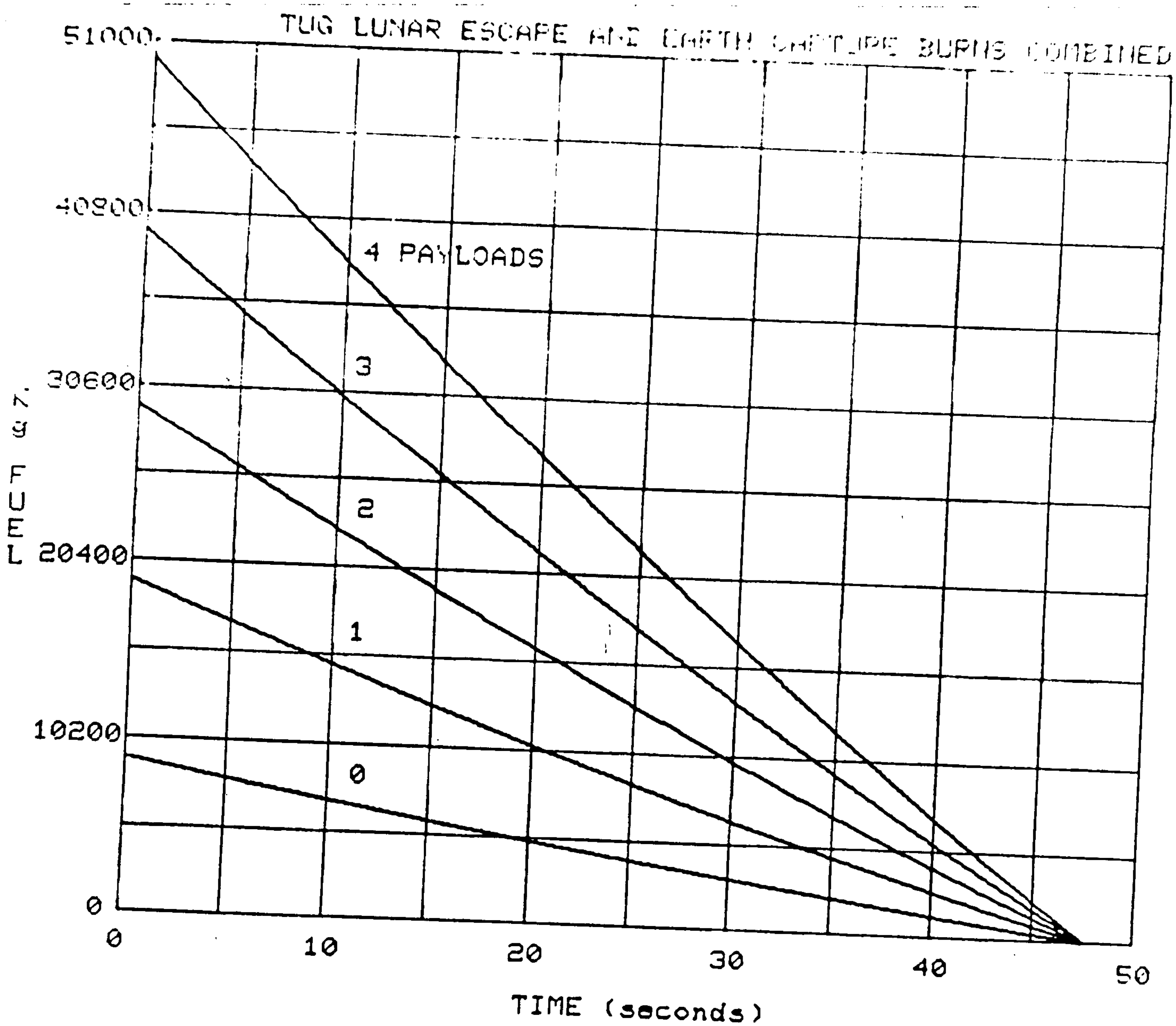
TIME-0	THRUST	FUEL MASS	MASS FLOW	% OPERATION
0	4.24386e+6	0	948.341	202.982
5	4.37877e+6	4585.52	978.488	209.434
10	4.51797e+6	9316.81	1009.59	216.092
15	4.6616e+6	14198.5	1041.69	222.962
20	4.80979e+6	19235.4	1074.8	230.05
25	4.96269e+6	24432.4	1108.97	237.363
30	5.12045e+6	29794.6	1144.22	244.909
35	5.28323e+6	35327.3	1180.6	252.694
40	5.45118e+6	41035.9	1218.13	260.727
45	5.62447e+6	46925.9	1256.85	269.015
47.619	5.71764e+6	50092.7	1277.67	273.472

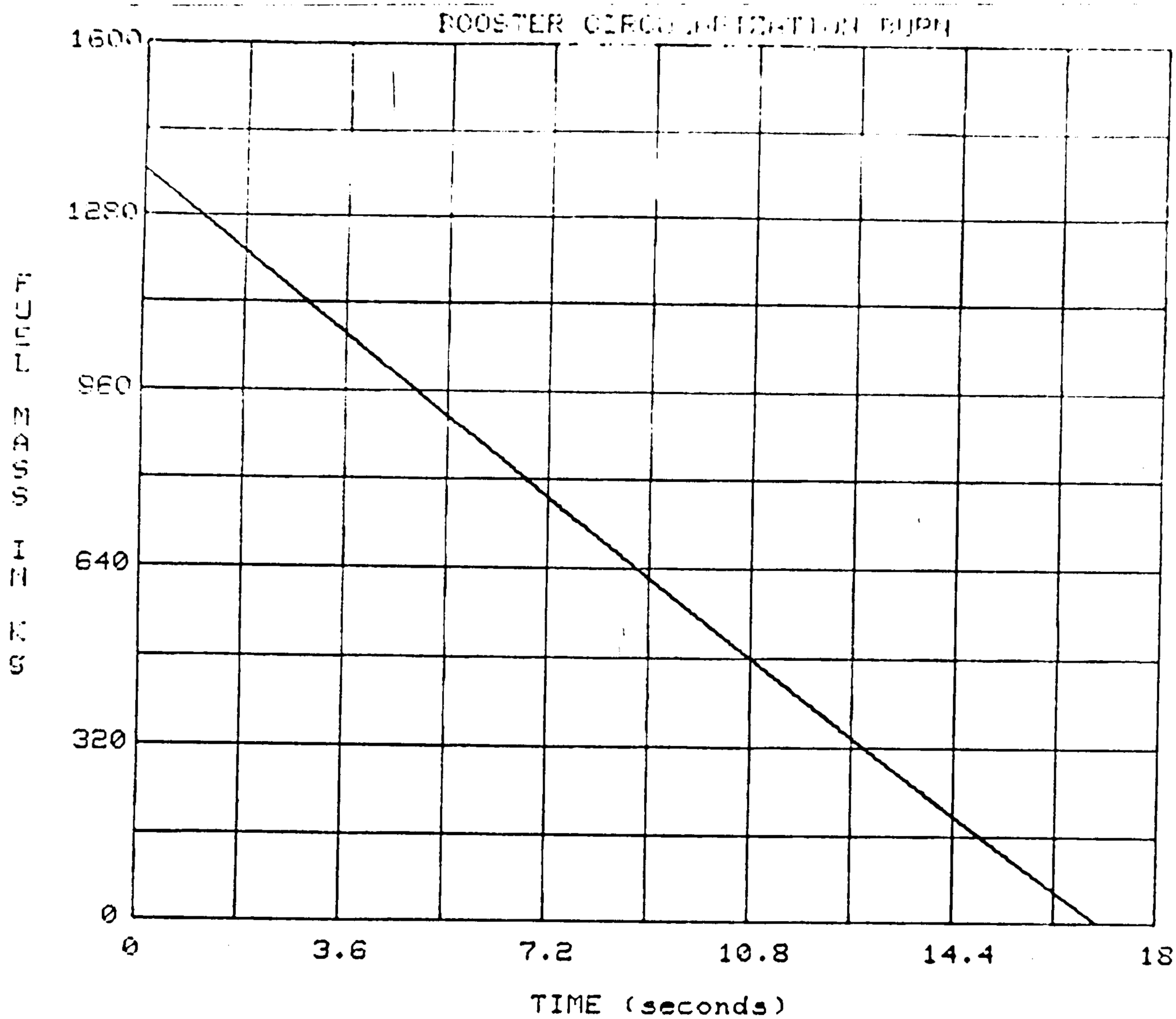
```
VOLUME OF H2 IS 102.23 CUBIC METERS
VOLUME OF O2 IS 37.3687 CUBIC METERS
TOTAL FUEL MASS IS 50092.7 Kg

Ready
```

```
RUN
FUEL     30 Apr 86  23:32

WHAT IS THE EMPTY, UNLOADED STRUCTURE MASS (Kg)? 26310

WHAT IS THE MASS OF ADDITIONAL STAGES AND THEIR FUEL
THAT IS BEING CARRIED DURING THE BURN (Kg)? 0

HOW MANY PAYLOAD CONTAINERS WILL BE CARRIED BY THE TOTAL STRUCTURE? 4

WHAT IS THE DESIRED DELTA-V IN Km/s? .877

WHAT IS THE MASS OF FUEL THAT IS DESIRED TO REMAIN
AFTER THE BURN IS COMPLETED (Kg)? 0

WHAT IS THE DESIRED ACCELERATION IN g'S? 3

BURN TIME = 29.8086
DT = 7.45216e-2
SCALED DT = 13

TIME-0          THRUST          FUEL MASS       MASS FLOW       % OPERATION
 0              4.24386e+6       0               948.341         202.982
 5              4381153          4666.44         979.02          209.548
 10             4.52289e+6       9483.84         1010.69         216.327
 15             4.6692e+6        14457.1         1043.39         223.326
 20             4.82026e+6       19591.2         1077.14         230.55
 25             4976193          24891.5         1111.99         238.009
 29.8086        5.13215e+6       30192.2         1146.84         245.468

VOLUME OF H2 IS 61.6167 CUBIC METERS
VOLUME OF O2 IS 22.5231 CUBIC METERS
TOTAL FUEL MASS IS 30192.2 Kg

Ready
```


TUG AND BOOSTER COMBINATION, EARTH ESCAPE BURN
4 PAYLOADS
3
2
1
0
Kg FUEL
250000
200000
150000
100000
50000
0
0
32
64
96
128
160
TIME (seconds)


TUG MOON CAPTURE
LOADS
4 PAYLOADS
3
2
1
0
KG FUEL
21000
14000
7000
0
0
6
12
18
24
30
TIME (seconds)


TUG LUNAR ESCAPE AND EARTH CAPTURE BURNS COMBINED
51000
40800
30600
20400
10200
0
FUEL
4 PAYLOADS
3
2
1
0
0
10
20
30
40
50
TIME (seconds)


BOOSTER CIRC
1600
1280
960
640
320
0
FUEL MASS IN KG
0
3.6
7.2
10.8
14.4
18
TIME (seconds)

APPENDIX C-3

```
[illegible]
LET [illegible]
LET [illegible]=[illegible]
LET [illegible]=[illegible]
[illegible] NAME "[illegible]"
FOR [illegible]F=[illegible] TO [illegible] STEP [illegible]
LET [illegible]=M[illegible]
LET [illegible]=[illegible]*32.2[illegible]
LET T=[illegible]
PRINT #1;MF;",";T
IF [illegible]F>[illegible] THEN 1100
[illegible]0 NEXT MF
[illegible]0 PRINT"NO SOLN"
1[illegible]00 GOTO 1110
[illegible] PRINT "SOLN IS TIME=";T;" AND MFUEL=";MF;" AND ACCELLERATION= [illegible]
[illegible] END

[illegible]

[illegible]

[illegible]

SOLN IS TIME= [illegible]  AND MFUEL= [illegible]000  AND ACCELLERATION= [illegible]

[illegible]

[illegible]
```

```
LINER1      30 Apr 8-  21:02

10 LET D=10E6
20 LET MS=35000
25 LET TH=24000
26 OPEN #1: NAME "TANKACE"
30  FOR MF=30000 TO 400000 STEP 1000
40  LET M=MS+(MF/2)
50  LET A=TH*32.2/M
60  LET T=D/A
65 PRINT #1:MF;",";T
70  IF MF>76.44*T THEN 1100
80 NEXT MF
90 PRINT"NO SOLN"
100 GOTO 1110
1100 PRINT "SOLN IS TIME=";T;" AND MFUEL=";MF;" AND ACCELLERATION=";A
1110 END

Ready

RUN

LINER1      30 Apr 80  21:02

SOLN IS TIME= 2441.75  AND MFUEL= 187000  AND ACCELLERATION = 4.3294

Ready
```

APPENDIX C-3

```
ADDL1     [illegible]

10 LET MS=55000
20 LET MF=137000
25 LET TH=24000
26 OPEN #1: NAME "GPROFILE"
30 FOR T=0 TO 2155 STEP 1
40  LET M=MS+(MF-T*72.44)
50  LET A=TH*32.2/M
60  PRINT #1:A;",";T
70 NEXT T
80 PRINT "DONE"
90 END

Ready

RUN

ADDL1     [illegible]

DONE

Ready

ADDL1     [illegible]

10 LET MS=55000
20 LET MF=137000
25 LET TH=24000
26 OPEN #1: NAME "GPROFILE"
30 FOR T=0 TO 2155 STEP 1
40  LET M=MS+(MF-T*72.44)
50  LET A=TH*32.2/M
60  PRINT #1:A;",";T
70 NEXT T
80 PRINT "DONE"
90 END

Ready
```

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