Space Station Operations Analysis
using
Gemini-Titan II-Agena

III.05

GER-10866
31 OCTOBER 1962
COPY NO. 026

GOODYEAR AIRCRAFT CORPORATION
AKRON 15, OHIO

CODE 25500

GOODYEAR

GOODYEAR AIRCRAFT CORPORATION

AKRON, OHIO

Copy No. 026

SPACE STATION OPERATIONS ANALYSIS USING GEMINI-TITAN II-AGENA

E.R. SMITH

D.C. ROMICK R.A. BELFIGLIO

GER-10866 31 OCTOBER 1962

218-52 (1-60)M

Text intentionally left blank

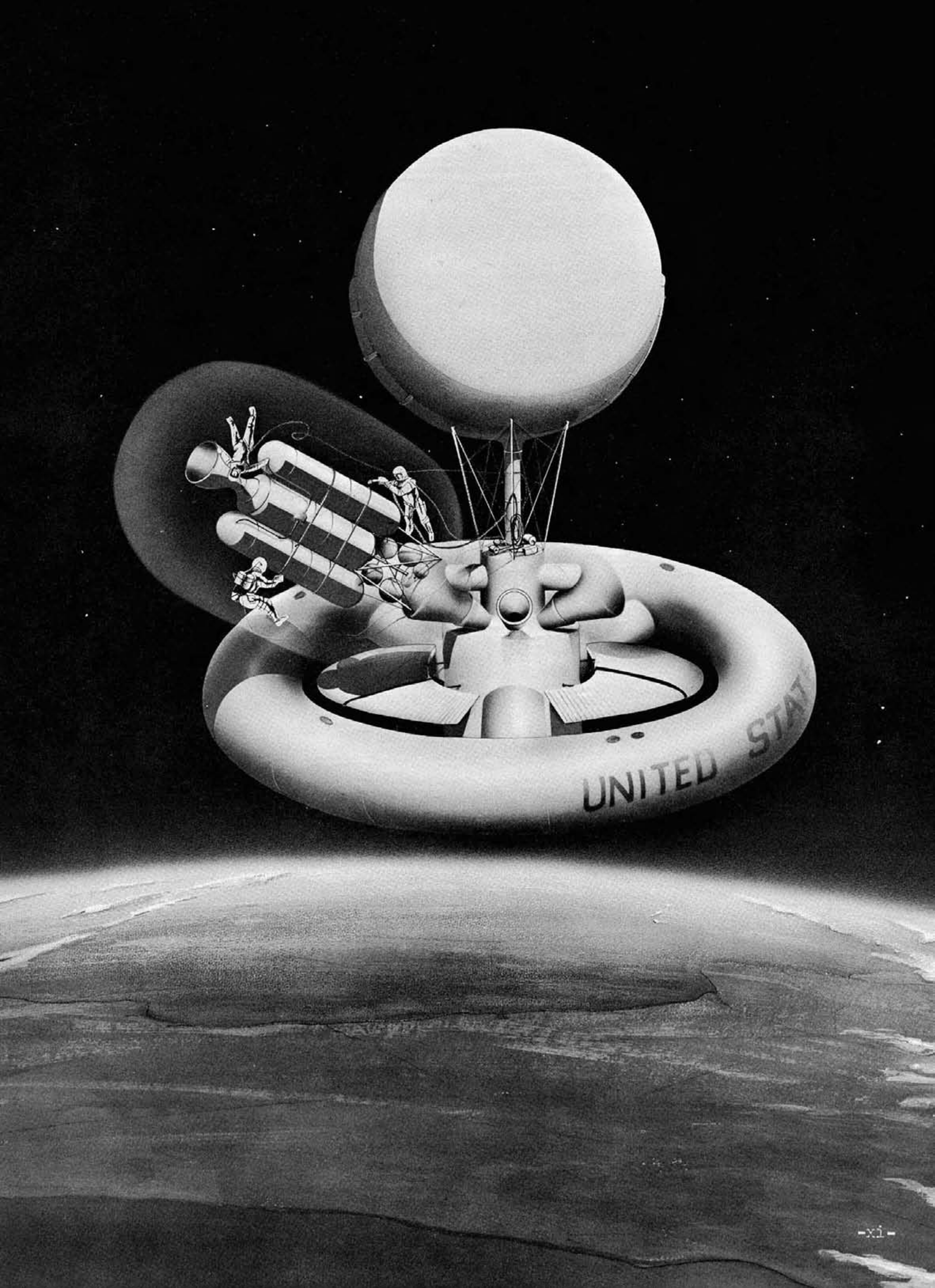
Page intentionally left blank

TABLE OF CONTENTS

LIST OF ILLUSTRATIONS

LIST OF TABLES

Page intentionally left blank


UNITED STAT

SECTION I - INTRODUCTION

1. GENERAL

Goodyear Aircraft Corporation (GAC) reports the results of an operational analysis of orbital space capabilities employing: (1) space stations, capsules, boosters and related equipment either presently available or within the current state-of-the-art; (2) orbital rendezvous; (3) orbital assembly, refueling and resupply; and (4) the concept of a space logistic system making regular and frequent operational-type flights. This study, which originally was limited to an investigation of an early, interim space station for Gemini rendezvous[1,2*], has been sponsored and supported by GAC on a continuing unfunded basis.

This analysis considers the accomplishment of many U.S. manned space tasks in the 1964 to 1966 period by using multiple ground launches and space flights of existing experimental-type equipment on a routine, repeatable, operational basis. This approach to space exploration avoids depending on the success of advanced research and long-term development programs for new vehicle systems, large launch boosters, liquid hydrogen powered upper stages, cryogenic fuel space storage, super-orbital velocity reentry techniques, and critical navigational performance. Present manned space programs are scheduled so that most equipment developed for each program will be used so seldom (e.g., the Mercury capsule - 2 suborbital and 4 orbital flights) as to never reach a stage of reliability, flexibility, economy, and usefulness that

* Superior numbers in the text refer to items in the List of References.

an operationally employed system provides. Instead, by using a continuous-repeatable (operational) task-force philosophy involving existing equipment for the initial space program, these new technological advances can be rapidly integrated into the overall logistic plans and actual operations when available or desirable to increase and enhance the logistic system's capacity, capability, reliability, and economic efficiency.

2. SOVIET SPACE PLANS

Our present national space goal is to accomplish a manned lunar landing and return in this decade, and, if possible to do so ahead of the Soviet Union. In view of the Soviet's past record, projected plans, and their own statements[3,4] (as well as their concern with the increased solar storm activities expected in the late sixties), it appears that their efforts are aimed toward a lunar landing by 1966 or before.

The cycle of solar storm activity from 1957 through the early seventies is shown in Figure 1 with late '64 early '65 as "the year of the quiet sun" or the time of minimum solar flares occurring. The period from mid '67 to early '71 will be a time of maximum solar storm activity and manned space flight during this time period may require extra protection (storm shelters) against possible large cosmic radiation from these solar flares. Thus, Soviet comments that they will make manned lunar landings by 1966 or else wait until 1972 have some technical support at least in the space "weather" aspects.

By delving deeper into Russian space plans for the early time period for a lunar landing by 1966, GAC has constructed a possible and very probable Soviet program which uses orbiting launch platforms (space stations),

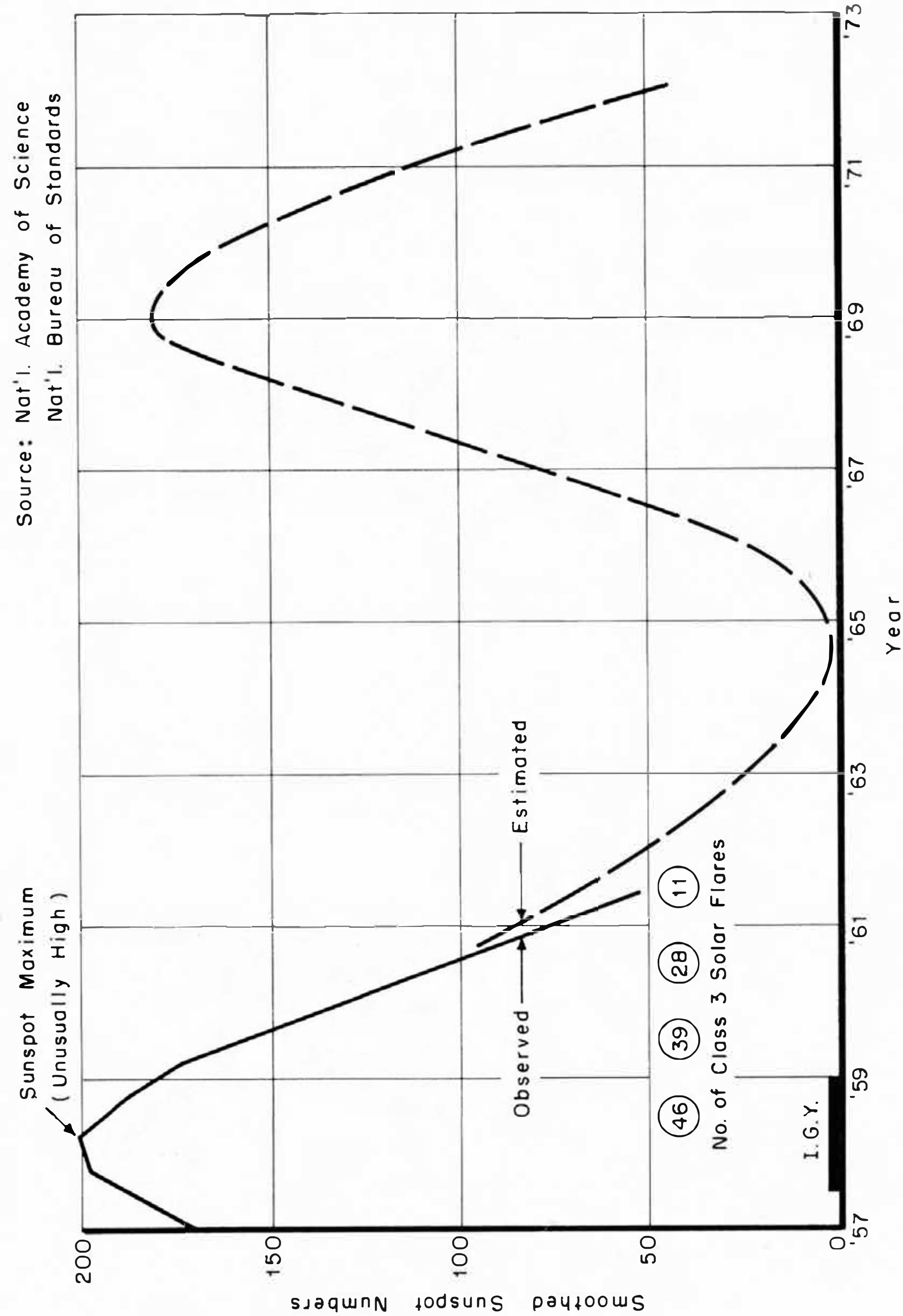


Figure 1 - Solar Storm Activity

existing boosters and space capsules, and a bit of luck that all goes well (or at least consistent with their past activities in space). The schedule for the anticipated Soviet lunar program is shown in Figure 2 along with the U.S. Apollo program which is orientated toward the same goal - landing a man on the moon, but travels a different development route. The elements of the Russian plan discussed above are not essentially new; Tsiolkovsky, the father of Soviet astronautics, and Oberth, German rocket pioneer, though along similar lines and statements of Keldysh, Shternfeld, Grigoryev, and fellow Soviet scientists substantiate such a concpet. C.L. Zakhartchenko, an American specialist on Soviet affairs, concluded that the Russians would have manned space stations in 1963-64 and make a manned flight to the moon in 1964-65[3].

A discussion of Figure 2 will show how the Russians can be expected to conduct their lunar program within these parameters (maximum benefit, minimum cost, and shortest time). By developing and performing rendezvous techniques and orbital assembly procedures in 1963 with existing equipment, the Soviet government can create a permanent manned space laboratory by late 1963 or early 1964 from modified Vostoks and rocket structures suitable as building material in space. Construction of a second space station to be used for the assembly of an interplanetary spaceship and as the launching base for such a vehicle would follow by mid 64. Continued logistic support of this station would allow the Russians to attempt circumlunar flights which

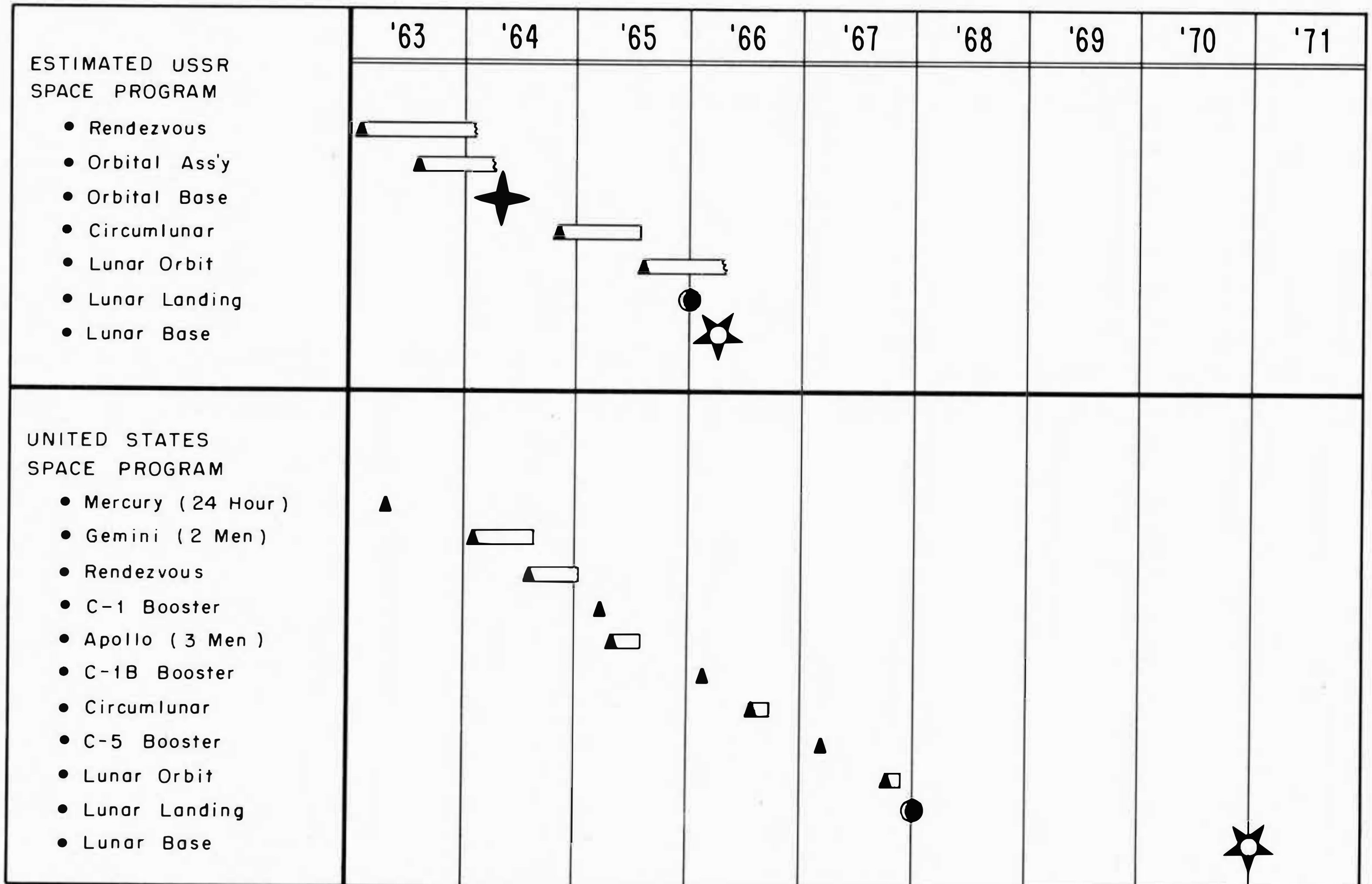


Figure 2 - USA and USSR Lunar Landing Program Milestones

originate and terminate at the orbiting launch base by late 1964 and lunar orbital flights in mid 65. The rate at which this logistic operation is continued would dictate when the Soviet cosmonauts would attempt a lunar landing but it could be as early as mid 1965 or at the same time as lunar orbital flights are accomplished. The Russians, then, will use current existing boosters, tested and proven space capsules, and present launch pads and procedures.

3. AMERICAN SPACE PLANS

In contrast, the currently planned American lunar program must develop and make operational the Saturn C-1, C-1B, and C-5 boosters; the Gemini, Apollo, and lunar excursion module (LEM) capsules; and rendezvous techniques before a lunar landing can be contemplated. In addition to these three completely new boosters, which require such new systems as cryogenically fueled upper stages; and the three new space capsules and their requirements, including exit and entrance in space techniques for the Gemini, super orbital velocity reentry capability for the Apollo, and lunar orbital rendezvous for the LEM; new Saturn launch pads and the associated checkout, handling, logistics, and transportation of these enormous missiles must be put in use. Each advancing step in the U.S. space program is dependent on the timely availability and success of a new booster and/or space capsule, (see Figure 2). The seven (7) day earth orbit and later rendezvous flights in 1964 require the man-rated Titan II and successful Gemini capsules. Apollo earth orbit flights are dependent on having man-rated C-1 boosters and Apollo capsules in 1965. Again in 1966 the requirement for Apollo circumlunar flights require still another Saturn booster, the C-1B. Finally manned lunar orbits and landings need both a new booster, the Advanced Saturn C-5, and the LEM capsule in 1967. Slippage or delays in any of these vehicles will seriously delay the entire U.S. time schedule for the lunar landing and repeated failures on some of the new systems could damage our prestige in the internationally declared "space race". Experience on the Vanguard and Mercury program amply demonstrate that this is a strong probability.

Using the parametric data discussed above as a frame of reference, GAC has studied an alternate approach based on the projected Soviet space program approach that could exploit the potential of interim space systems such as the Titan II missile, Agena boosters, and the Gemini capsule and serve as a backup system for the current Apollo-Saturn program. The following sections of this report present the results of this study.

Page intentionally left blank

SECTION II - SPACE STATIONS, CAPSULES AND BOOSTERS

1. SPACE STATIONS

The initial study of an early, interim space station to be used by Gemini astronauts after development of rendezvous techniques involved a minimum-type base capable of docking two (2) Gemini capsules, housing one to three men, having a lifetime of at least one (1) year, and supplying an accumulated 30 days of life-support. This station was a cylinder ten foot in diameter by 15 to 20 foot long containing an air lock and weighed some 4,000 to 6,500 lbs. using an Atlas-Agena or Titan II booster (depending on the station's final gross weight). Much of this work has been summarized in references 1 and 2, which are reproduced for the reader's convenience as appendices A and B, and was extended from basic work carried out in the 1959-61 period.

At this stage in the study, parameters were broadened to look at more versatile space station concepts. These parameters included: (1) partial artificial gravity, (2) an eight man crew, (3) 60 day crew duty cycle, (4) 400-500 cu. ft. volume per man, (5) docking facilities for five Gemini capsules, (6) station unmanned at launch (automatic erection), and (7) station weight compatable with Titan II booster - 7,000 lbs. maximum.

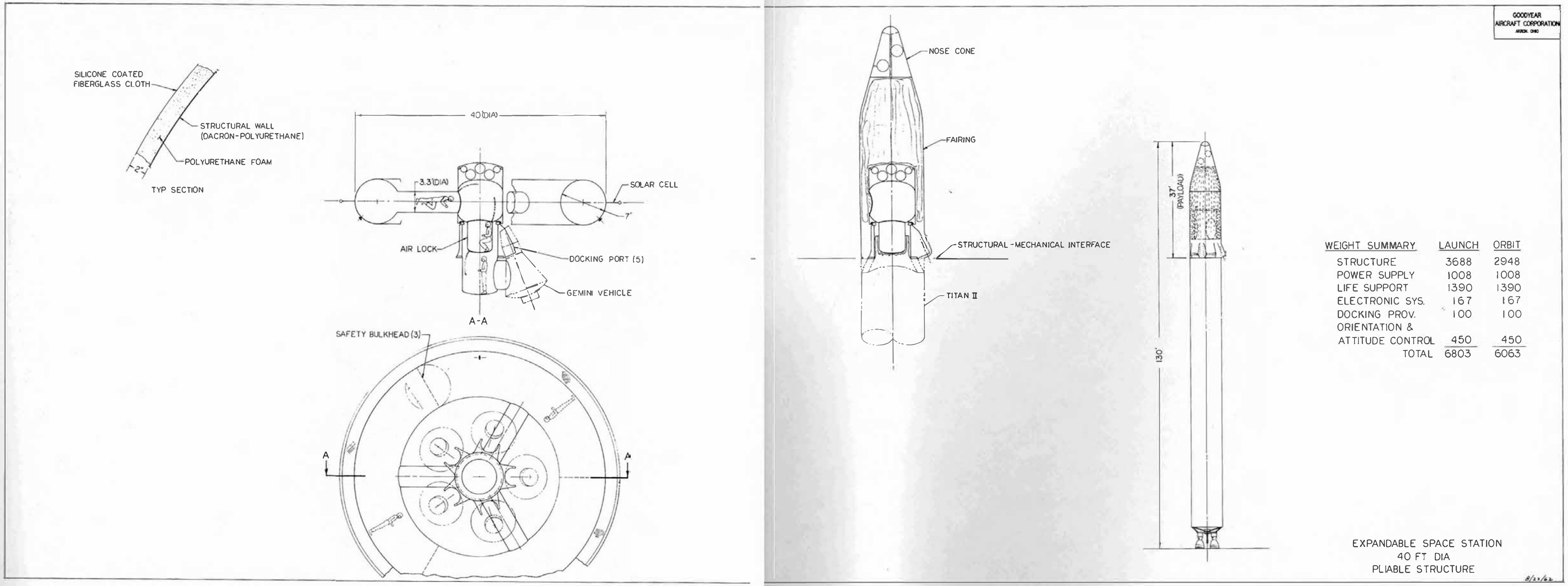
To produce artificial gravity the space station would have to be rotated but for docking and stowage of the Gemini capsules a condition of immobility and weightlessness is desirable and in many cases necessary. To meet both of these conditions, GAC examined its inflatable torus space station concept with non-rotating docking ports and air lock. The inflatable torus (rim) offers many advantages over the rigid cylinder previously discussed, such as, (1) larger volume for less structural weight, (2) ease of packaging for minimum launch volume, (3) symmetrical (equidistance) distribution of station about the center of rotation, and (4) a larger radius of revolution and consequent lower rotational velocity (rpm) for a given centrifugal force (artificial gravity). For a broader treatment of space station concept evolution, see Appendix C.

Based on previous research and development work in expandable space structures, a design for a 40 foot diameter, pliable-structure, toroidal space station compatible with Gemini and Titan II operations was prepared as shown in Figure 3. This station, consisting of a central hub, air lock, docking ports, a torus with a 7 foot diameter cross section and three access spokes from hub to torus, weighs 6,800 lbs. at launch, has docks and storage for five Gemini capsules and can be packaged within a 10 ft. dia. cylinder for mounting atop an unmanned Titan II. The unit would utilize such developed components as (1) the hub/docking unit from Gemini-Agena B, (2) fuel cell from Gemini, (3) life support system from Mercury-Gemini, and (4) communications from Mercury-Gemini. This space station is similar to an existing 30 ft. diameter working model that GAC has manufactured for use in the following research programs.

1. Developing internal furnishings, lighting and color details.
2. Developing details of expandable, foldable furniture and containers.
3. Establishing and carrying out human-factors habitability experiments.[a]

---

[a] To date, crew experiments have been performed, with the crew living up to eight hours in the station.



WEIGHT SUMMARY	LAUNCH	ORBIT
STRUCTURE	3688	2948
POWER SUPPLY	1008	1008
LIFE SUPPORT	1390	1390
ELECTRONIC SYS.	167	167
DOCKING PROV.	100	100
ORIENTATION & ATTITUDE CONTROL	450	450
TOTAL	6803	6063

Figure 3 - 40 Ft. Diameter Expandable Space Station

4. Developing packageability techniques.

5. Establishing detailed requirements and developing working models of the environmental control and air conditioning systems.

6. Developing materials, equipment and techniques for station maintenance and repair.

Figure 4 shows the 30 ft. space station and its interior furnishings for crew accomodation. Much of the habitability studies and parameters are based on extensive GAC airship and submarine experience and the realization that many aspects of space flight most closely parallel that of airship and submarine operations. Thus the 500 cu. ft. per man requirement relates directly to present undersea practise for modern long duration mission submarines.

Some of the parameters for the operation of the 40 ft. station in space include:

a. Life Support

Air consumption: 4.5 pounds/day/man

Water consumption: 9.0 pounds/day/man

Food consumption: 2.0 pounds/day/man

b. Atmosphere

Pressure: 1/2 atmosphere (7 psi)

Temperature: $75^{o} \pm 10^{o}$ F

Minimum air circulation rate: 10 cubic feet/minute/man

Composition: Carbon dioxide $\leq$ 0.5%

Relative humidity 25 - 50%

Nitrogen and oxygen in natural atmospheric ratio

Figure 4 - 30 Ft. Space Station

Carbon dioxide production: 2 pounds/day/man

c. Environment

Noise: maximum level $\leq$ 90 decibels

Radiation: maximum exposure 1.2 REM/month/man

Vibration: maximum - 10 inch at 0.1 cps frequency
1 inch at 1 cps frequency
10^{-2} inch at 10 cps frequency
10^{-3} inch at 100 cps frequency

Station air leakage: 15% of total volume/month

Rotational velocity: 4 to 5 rpm maximum, 10 fps minimum rim velocity

Fuel for adjustments, rotation and orientation: 10 pounds/day

d. Crew

Duty cycle: 8 hours work, 8 hours sleep, 8 hours rest and/or on call for work (standby).

Duty tour: 60 days

Qualifications: Gemini astronaut + experience in primary duty assignment (maintenance, operations, command, medical, etc.)

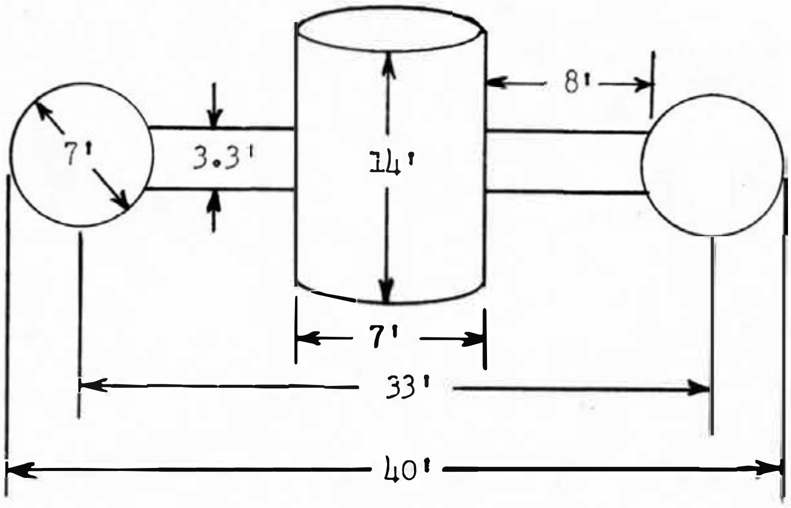
Training: crew members are cross-trained similar to submarine crews

From this data the quantities of supplies and equipment required per month for continuous operation can be calculated. These calculations include the life support at $15\frac{1}{2}$ lbs./day/man, the replacement of station air based on a 15% per month leakage, fuel for adjustment, rotation and orientation of the station at a rate of 10 lbs./day, the weight of containers for the above items, and an additional weight for mission equipment, dry goods and supplies that will enhance the overall capabilities of the space station to perform various scientific missions and operational training and experiments such as are listed in Appendix A. The calculations (on the following page) indicate that a single Titan II booster could deliver the necessary resupply package of less than 6,000 lbs. to the space station each month. Four additional Titan II flights every 60 days would be required to rotate the eight man crew if a Gemini capsule were used. Thus, as shown in Figure 5, six or seven Titan II launchings every two months would accomplish earth logistic support of the space station.

Volume of: 7 ft. dia. cross section - 40 ft. diameter torus

7 ft. dia. x 14' long cylinderical hub

3.3 ft. dia. x 8' spokes



Torus

Volume = area x perimeter

$V_T = (103.67)(38.49)$

$V_T = 3,990$ cu. ft.

$A = \pi r^2$

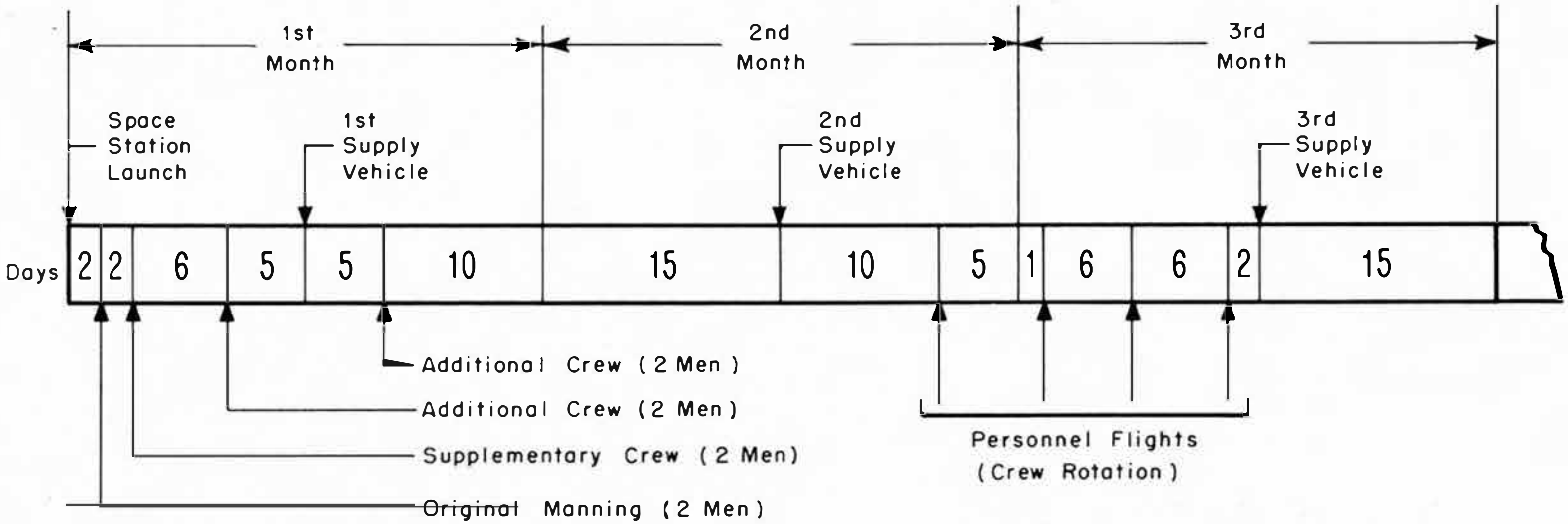
$A = (3.5)^2 \pi$

$A = 38.485$ ft.^{2}

$P = 2\pi r$

$P = 2\pi\left(\frac{33}{2}\right)$

$P = 103.67$ ft.



ASSUMPTIONS:

1. Launched unmanned vehicle — automatic deployment
2. Checkout from ground by telemetry
3. 8-Man crew
4. 4 Titan II launch pads required
5. Supply vehicle — 4000 lbs supplies, 1000 lbs equipment, 1000 lbs fuel
6. Crew duty cycle — 60 days

Figure 5 - Space Station Logistic Support Schedule

Hub $V_H = \pi r^2 h$

$V_H = (3.5)^2 (14) \pi$

$V_H = 539$ cu. ft.

Spokes $V_S = 3 (\pi r^2 h)$

$V_S = 3 \left(\frac{3.3}{2}\right)^2 (8) \pi$

$V_S = 205$ cu. ft.

Total Volume $V = V_T + V_H + V_S$

$V = 3990 + 539 + 205$

$V = 4734$ cu. ft.

Weight of Air in Space Station W = 4734 cu.ft. x 0.0382 lb./cu.ft.

W = 181 lbs.

Contents of Monthly Supply Flight to Station

Life support/month/8 men

15.5 lbs. x 30.4 days x 8 = 3770 lbs.

Fuel/month

10 lbs. x 30.4 days = 304 lbs.

Air leakage/month

181 lbs. x 15% = 30 lbs.

Sub Total 4104 lbs.

Weight for Containers 821 lbs.

Sub Total 4925 lbs.

Mission equipment and supplies 1,000 lbs./supply flight (telescopes, cameras, microscopes, power supplies, clothing, radiation shielding, etc.)

Total 5925 lbs.

2. SPACE CAPSULES

At present only four manned space capsules/vehicles are in existence or under development - Mercury, Gemini, Apollo and the X-20, Dyna Soar. Table 1 compares the major characteristics of the four vehicles. Of these, only the Mercury capsule has been developed and used for manned space flight and reentry to date. Because of the high costs and long lead time required to develop these currently approved vehicles, it is doubtful if any new manned space vehicle will be funded and developed in the next few years. The only new concept that may have a future is the Air Force's Aerospace plane and even this vehicle must wait until a lengthy Planning Study (PS) is let by the USAF and completed by the selected contractors before decisions on development go-ahead can be made.

TABLE 1 - Space Vehicles Comparison

Characteristic	Mercury	Gemini	Dyna Soar	Apollo
Crew	1	2	1	3
Orbital Endurance	$\leq$ 34 hours	14 days	?	14 days
Rendezvous	No	Yes	No	Yes
Docking Provision	No	Yes	No	Yes (?)
Space Access	No	Yes (door)	No	Yes (Air lock)
Gross Weight	4,200 lbs.	6,600 lbs.	15,000 lbs.	20,000 lbs.
Minimum Booster	Atlas-Agena	Titan II	Titan III	Saturn C-1
Reentry Velocity	Orbital (Appr. 25,000 fps)	Orbital (Appr. 25,000 fps)	Orbital (Appr. 25,000 fps)	Escape (Appr. 36,000 fps)
Landing Maneuverability	No	Some	Yes	Some

The X-20, Dyna Soar was eliminated early from consideration during this time period as a manned space vehicle for use with an orbiting space station because of three important reasons: (1) the vehicle initially weighs 15,000 lbs. thus requiring a Titan III booster for orbital injection, (2) the vehicle is unpowered (uses Titan III third stage for reentry) and thus has no orbital rendezvous capability, and (3) only one man is carried in each vehicle thus increasing the number of launchings, boosters, and vehicles required for space station operations.

The Apollo capsule was considered as a follow-on or "growth" vehicle for space station operations. Initial Apollo capsules will have a three man crew but studies are being conducted on the possibility of increasing the crew to five or six persons - a most attractive concept for space station operations and crew rotation. Early use of Apollo was considered undesirable for the following reasons: (1) its weight is 20,000 lbs. requiring a Saturn C-1 booster for earth orbital flights, (2) the capsule is equipped with a super-orbital velocity reentry heat shield and other sophisticated equipment too costly to tie up at present in simple, low altitude, earth orbital flights, and (3) Apollo capsules will not be available until 1965 for any type of manned flight and probably much later for extensive earth orbit operations. Apollo then holds much promise in the future as a six man personnel and supply carrier for space station operations and logistic support.

The Mercury capsule while enjoying success as the only U.S. manned capsule in production and by achieving actual orbital space flight has several short-comings that make it unsuitable for space station operations. The main deficiencies of the Mercury are the absence of an air lock or any other space exit, which denies the astronaut access to space while in orbit, and the lack of rendezvous and maneuvering provisions. Of less importance but still significant is the limitation on crew size - one man and orbital mission time-maximum of 34 hours.

The Gemini capsule, scheduled for manned orbital flights in 1964, will be the first manned vehicle capable of rendezvous and docking maneuvers in orbit, multi-crew operations (two men), long endurance orbital flight (up to two weeks), crew access to space via two doors (instead of an air lock) by de-compressing the internal capsule environment, and possessing some landing maneuverability during atmospheric reentry by creating lift with angle of attack changes. In addition the Gemini weighs 6,600 lbs. and can be put in orbit by an existing Titan II missile booster.

3. SPACE BOOSTERS

The Atlas missile booster has been used, in its man rated version, to put all Mercury manned space capsules into orbit. The Atlas puts slightly less than 3,000 lbs. (Mercury capsule) into a very low orbit and requires a long checkout and countdown before launching.

The Titan II, in its military ICBM version has been successfully launched by the military and is being operationally deployed in underground hardened silos at present. The Titan II can put approximately 7,000 lbs. in orbit and with storable fuels and simplified systems it should have a shorter checkout and countdown procedure. Eventually NASA and the Martin Company expect the total launch pad time of a Titan II to take less than a week for space launches. A version of the Titan II is being man-rated for Gemini launches for early 1964.

TABLE 2 - Space Booster Comparison

Characteristic	Atlas	Agena B	Titan II	Titan III	Centaur	Saturn C-1
Stages	$1\frac{1}{2}$	1	2	3	1	3
Gross Weight (lbs.)	260,000	21,000	300,000		31,000	1.5×10^6
Nominal Thrust (lbs.)	367,000	16,000	430,000	2×10^6	30,000	1.5×10^6
Propellant	Kerosene (RP)/ LOX	Red Fuming Nitric Acid/ Unsymmetrical Dimethyl Hydrazine (UDMH)	Nitrogen Tetroxide/ Mixture of Hydrazine and UDMH	Solid/ Nitrogen Tetroxide/ Hydrazine-UDMH	LOX/ Liquid Hydrogen	Kerosene (RP)/ LOX/ Liquid Hydrogen
Payload (300 mi. orbit)	2,900 lbs.	*5,000 lbs.	7,000 lbs.	25,000 lbs.	*8,500 lbs.	20,000 lbs.
Length (feet)	72	26.5	90	90	28	162
Minimum Diameter of Booster (Feet)	10	5	10	10	10	12.83

*Using Atlas Booster (Atlas-Agena and Atlas-Centaur). Titan II and III to use Agena D as additional stage and possibly Centaur.

Page intentionally left blank

SECTION III - PERFORMANCE REQUIREMENTS AND DESIGN OF AN ORBITAL BASED SPACE VEHICLE

1. GENERAL CONCEPT

After completing preliminary studies of earth orbital space station operations and missions (see Section II: 1 and Appendix A) and reviewing the anticipated Soviet lunar approach (see Section I: 2), the questions of the feasibility of employing the same techniques for the U.S. lunar landing program became apparent. Thus, the operational study continued by investigating the application of an orbiting space station as an orbital launch platform for eventual manned space excursions to the moon.

The overall concept involves frequent and regular cislunar excursion flights from the space station in a vehicle suitable for extended space flight where each flight uses and expands on the knowledge gained from the preceding flight. At first very short flights of a few miles out from the station's orbit are made to gain experience in navigation, rendezvous and docking. Then probing flights into and near the Van Allen belt are flown to determine and test the effects of radiation and the shielding required to protect men and equipment when passing through this medium. After this is accomplished, longer and longer cislunar flights out into space past the radiation belt are made ever extending the operational knowledge and experience required to successfully make such flights. Next, flights are made incorporating velocity increments to inject the vehicle into the lunar intercept plane. These flights at first are very short and similar in nature to the original space excursion flights but as this new technique is developed the flight time is constantly lengthened until all types of cislunar trajectories are a common and routine operational occurrence. At this point the experienced crews are ready to attempt circumlunar (which is a logical extension of the cislunar trajectory) and lunar orbital flights. If and when these flights are successfull, a space station identical to the launch platform in earth orbit can be put into lunar orbit. This lunar station can be manned, operated, and supplied from the earth by the logistic network already in existence and which was developed by the operational experience gained in earlier flights. From this lunar launch platform, capsules can be landed on the moon-remotely at first but with men aboard when techniques and experience are demonstrated.

2. CISLUNAR AND LUNAR TRAJECTORY CHARACTERISTICS [8]

To determine a suitable vehicle for space excursion flights, the various characteristics of the cislunar, circumlunar and lunar orbit trajectories must be calculated. The basic trajectory for all excursion flights originating from a circular earth orbit will be elliptical with its perigee (minimum) altitude, h_1, equal to the circular earth orbital height and its apogee (maximum) altitude, h_2, equal to the circular orbital altitude (h_1) plus the farthest distance, S, that the vehicle travels into space from the orbiting space station. The geometry of these trajectories is shown in Figure 6 and the required velocities (V) and periods (T) associated with these trajectories can be determined from the following equations.

At perigee, the velocity is

$$V_P = \left[\frac{\mu}{a}\left(\frac{1+e}{1-e}\right)\right]^{\frac{1}{2}} \text{ feet/sec.} \qquad (1)$$

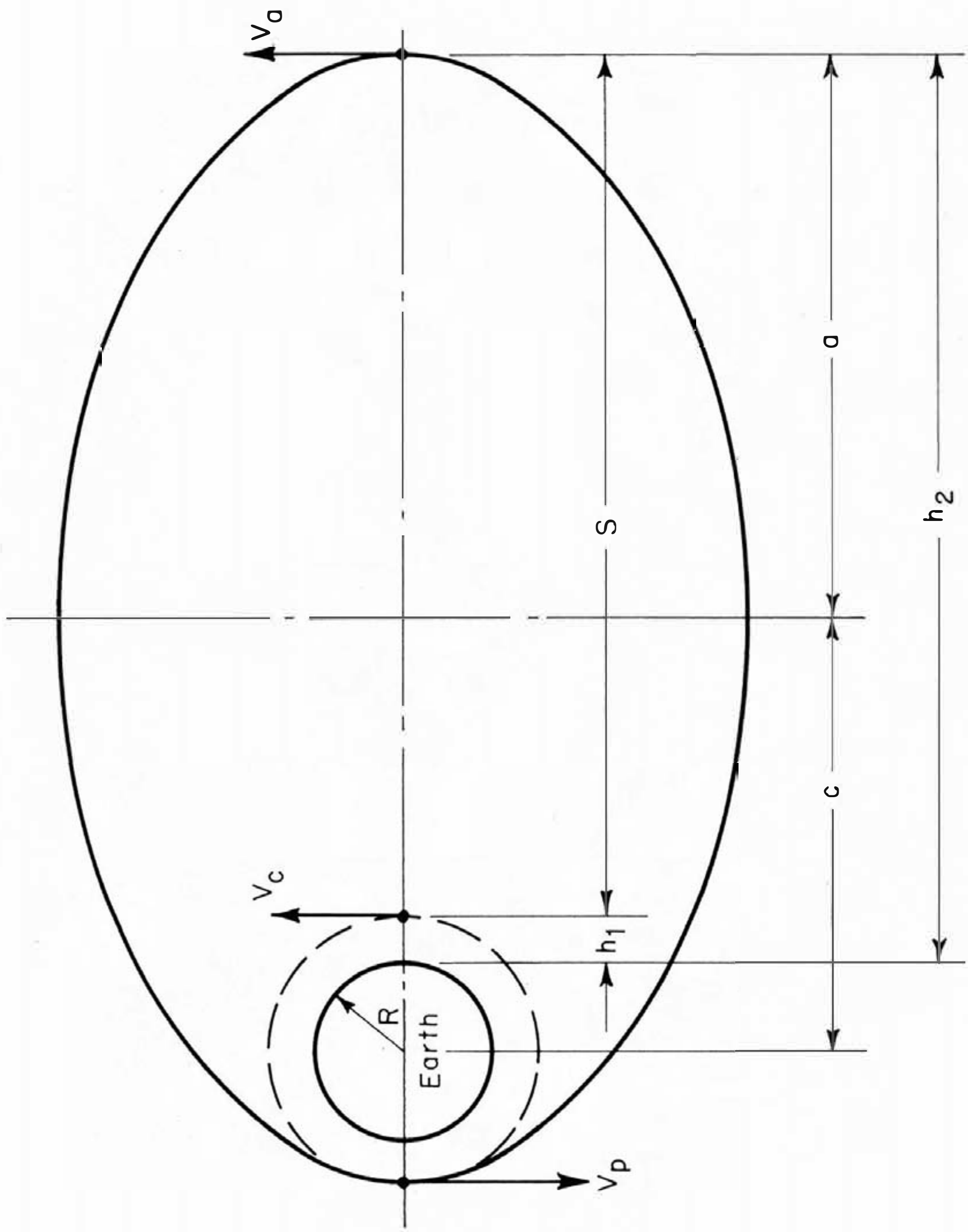


Figure 6 - Trajectory Geometry

And at apogee

$$V_a = \left[\frac{\mu}{a}\left(\frac{1-e}{1+e}\right)\right]^{\frac{1}{2}} \text{ feet/sec.} \qquad (2)$$

where μ = GM = 1.41×10^{16} ft.^{3}/sec.^{2}

a = semi-major axis or mean distance (in feet) = $\dfrac{h_1 + 2R + h_2}{2}$

e = eccentricity = c/a

and G = gravitational constant = 1.07×10^{-9} ft.^{3}/lb.-sec.^{2}

M = mass of central body (Earth) = 1.319×10^{25} lb.

R = radius of Earth

Polar: 3,950 miles

Equitorial: 3,963.5 miles

Average approximation in plane inclined 30°

to equator: 3,960 miles

C = distance from focus of ellipse to center = $\dfrac{h_2 - h_1}{2} = \dfrac{S}{2}$

And for a circular orbit where $h_1 = h_2$, $C = 0$, and $e = 0$, the velocity is

$$V_c = \left[\frac{\mu}{a}\right]^{\frac{1}{2}} = \left[\frac{1.41 \times 10^{16}}{R + h_1}\right]^{\frac{1}{2}} \text{ feet/sec.} \qquad (3)$$

Where a, R, and h_1 are in feet.

The incremental velocity (ΔV) required to go from a circular orbit of altitude h to an elliptical orbit of distance S is found by

$$\Delta V = V_p - V_c \qquad (4)$$

or

$$\Delta V = \left[\frac{\mu}{a}\left(\frac{1+e}{1-e}\right)\right]^{\frac{1}{2}} - \left[\frac{\mu}{R + h_1}\right]^{\frac{1}{2}}$$

where ΔV, the change in velocity is applied to the circular velocity, V_c, at the desired perigee space coordinate. After the complete elliptical trajectory has been travelled, an equal but opposite ΔV must be applied to reenter the original circular orbit. Thus, the total velocity increment required to go from a circular to an elliptical and back to a circular orbit is twice the initial velocity change or 2 (ΔV). If an additional velocity change is introduced to inject the vehicle into the lunar intercept plane, then this ΔV must be taken out at the proper time and place (in an equal and opposite manner) to return the vehicle to the earth orbit plane. Similarly any velocity changes used to enter a lunar orbit as a circumlunar flight approaches the moon must be applied in an equal but opposite manner to cause lunar deorbit and return to earth. Thus, except for velocity changes used for rendezvous and docking, all velocity changes made by a space vehicle operating from a launch platform in earth orbit must be double the original velocity changes if the vehicle is to return to the orbiting launch platform. This basic operating principle is one of the main differences between this approach and the Apollo program. Operating from a station, a substantial change in velocity is used to slow the space vehicle as it approaches suitable earth orbital altitudes of 100 to 500 miles (300 miles of course is the goal) from whence reentry at orbital speeds of 25,000 feet per second can be accomplished with a Mercury/Gemini type heat shield. The Apollo, on the other hand, approaches these altitudes at a lunar elliptical velocity of some 35,000 feet per second (where S = 240,000 miles) and plunges directly into the atmosphere at this high velocity which requires the development of an advanced super-orbital velocity heat shield (capable of handling approximately double the energy per lb. of vehicle weight) and precise guidance/navigation techniques for sufficiently accurate (40 mile corridor) reentry and landings. ΔV in equation (4) was calculated for various values of S, see Table 3, and the results have been plotted in Figure 7 to show the velocity increments (ΔV) required for departure from a 300 mile circular Earth orbit into a variety of elliptical orbital transfers.

The period of an elliptical orbit is

$$T = 2\pi\left(\frac{a^3}{\mu}\right)^{\frac{1}{2}} \text{ sec.} \qquad (5)$$

where a is in feet

By substituting the values for the constants π and μ in equation (5) and solving for T in minutes, then

$$T = 8.828 \times 10^{-10} (a)^{3/2} \text{ min.} \qquad (6)$$

For T in hours and a in miles, equation (6) becomes

$$T = 0.5644 \times 10^{-5} (a)^{3/2} \text{ hours} \qquad (7)$$

For a circular orbit (where $h_1 = h_2$ and hence $a = R + h_1$) the period can be calculated from equation 5, 6, or 7 after the simple substitution of $R + h_1$ is made for the distance a.

The elliptical trajectory periods (T) for cislunar flights originating from an earth orbiting launch platform of 300 mi. altitude have been computed using equation (7) for discrete values of the elliptical trajectory distances, S, and are presented in Table 4.

TABLE 3 - Velocities Associated With
Cislunar Elliptical Trajectories of Distance S

Trajectory Distances (ft.)					Dimension Less Factors			Velocity, fps			
S	Perigee h_1	Apogee h_2	c	a	e	$\frac{1+e}{1-e}$	$\frac{1-e}{1+e}$	V_c	V_a	V_p	ΔV
15	300	315	7.5	4267.5	0.0018	1.0035	0.9965	25,037	24,960	25,055	18
30	300	330	15	4275	0.0035	1.0070	0.9930	25,037	24,980	25,090	53
100	300	400	50	4310	0.0116	1.0234	0.9771	25,037	24,603	25,180	143
200	300	500	100	4360	0.0229	1.0469	0.9551	25,037	24,130	25,320	283
250	300	550	125	4385	0.0285	1.0587	0.9446	25,037	23,985	25,420	383
330	300	630	165	4425	0.0373	1.0775	0.9281	25,037	23,665	25,500	463
350	300	650	175	4435	0.0395	1.0822	0.9240	25,037	23,585	25,520	483
400	300	700	200	4460	0.0448	1.0938	0.9142	25,037	23,400	25,585	548
1,000	300	1,300	500	4760	0.1050	1.2347	0.8099	25,037	21,330	26,320	1,283
3,000	300	3,300	1,500	5760	0.2604	1.7042	0.5868	25,037	16,450	28,110	3,073
5,000	300	5,300	2,500	6760	0.3698	2.1736	0.4600	25,037	13,500	29,305	4,268
10,000	300	10,300	5,000	9260	0.5400	3.3473	0.2987	25,037	9,280	31,070	6,033
20,000	300	20,300	10,000	14260	0.7013	5.6947	0.1756	25,037	5,682	32,655	7,618
40,000	300	40,300	20,000	24260	0.8244	10.3895	0.0963	25,037	3,253	33,810	8,773
75,000	300	75,300	37,500	41760	0.8980	18.6078	0.0537	25,037	1,853	34,450	9,413
100,000	300	100,300	50,000	54260	0.9215	24.4710	0.0409	25,037	1,420	34,700	9,663
150,000	300	150,300	75,000	79260	0.9463	36.2090	0.0276	25,037	965	34,920	9,883
200,000	300	200,300	100,000	104260	0.9591	47.9476	0.0208	25,037	730	35,050	10,013
240,000	300	240,300	120,000	124260	0.9657	57.3260	0.0174	25,037	612	35,097	10,060

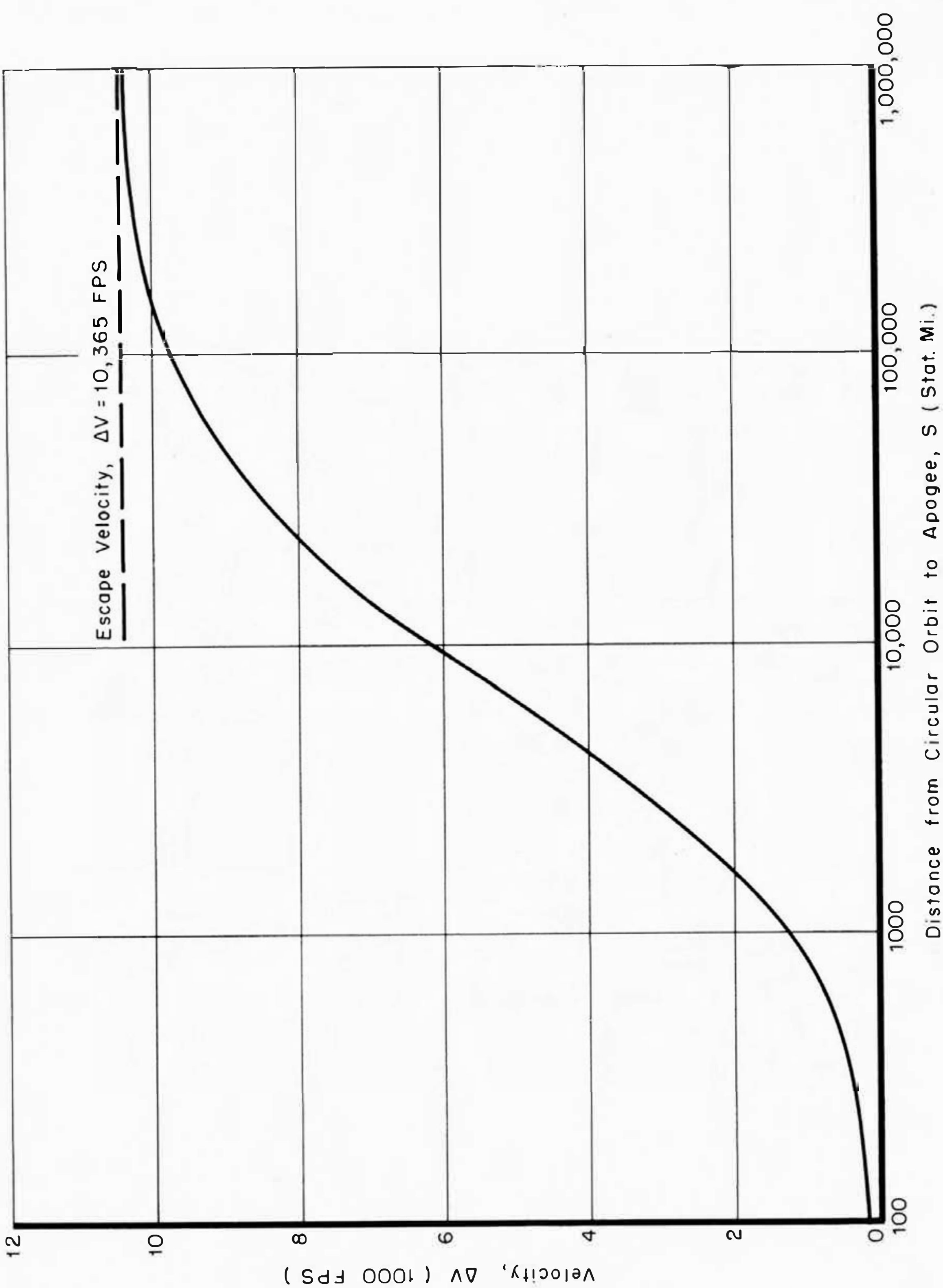


Figure 7 - Required Velocity Increment for Departure from 300 Mile Circular Earth Orbit

TABLE 4 - Orbital Period (T) for Cislunar Elliptical Trajectories

S (Miles)	a (Miles)	a^3	$(a^3)^{\frac{1}{2}}$	Period, T (hours)
15	4,267.5	7.77×10^{10}	2.79×10^5	1.57
30	4,275	7.81×10^{10}	2.80×10^5	1.58
100	4,310	8.01×10^{10}	2.83×10^5	1.60
200	4,360	8.29×10^{10}	2.88×10^5	1.63
250	4,385	8.43×10^{10}	2.90×10^5	1.64
300	4,410	8.58×10^{10}	2.93×10^5	1.65
330	4,425	8.66×10^{10}	2.94×10^5	1.66
350	4,435	8.72×10^{10}	2.95×10^5	1.67
40,000	24,260	14.35×10^{12}	3.79×10^6	21.39
75,000	41,760	0.73×10^{14}	8.55×10^6	48.26
100,000	54,260	1.60×10^{14}	1.27×10^7	71.68
150,000	79,260	4.98×10^{14}	2.23×10^7	125.86
200,000	104,260	11.3×10^{14}	3.35×10^7	189.07
240,000	124,260	19.2×10^{14}	4.38×10^7	246.64

where $T = 0.5644 \times 10^{-5} (a)^{3/2}$ hours

and $h_1 = 300$ miles = Earth orbital altitude

$$h_2 = h_1 + S$$

$$a = \frac{h_1 + 2R + h_2}{2}$$

R = 3960 miles = average radius of Earth in plane inclined 30° to equator

77-10(1-53)M

It is well to note at this point that all of the above equations are valid only for two-body solutions in which an infinitesimal orbiting body such as a satellite, capsule, etc. is influenced and/or attracted by the finite Earth's mass (where the orbiting vehicle and the Earth are the two bodies). For orbits in which the vehicle approaches the lunar gravitational influence (less than 20,000 miles of the moon's surface), the same equations for velocity and period can be used in a two-body solution (the moon and the vehicle) by modifying the constant μ to reflect the lunar mass (0.0123 of Earth's mass) and using the moon's radius of 1080 miles. More refined solutions involving the three-body problem (Earth, Moon, and the orbital vehicle) or the four-body problem (Earth, Moon, Sun, and orbiting vehicle) are possible by expressing the perturbations of the finite bodies (Earth, Moon, and Sun) on the infinitesimal body (orbiting vehicle) in integral form. These solutions are much more complex and only refine the relative magnitudes of the perturbations established by the two-body problem. By considering the moon enclosed in a sphere of influence in which only the Moon's gravitational field is used within this sphere and only the Earth's field outside of it, satisfactory results from equations (1) through (7) can be obtained for the purposes of this analysis. The more complex trajectory interfaces have been studied extensively and computed by Rand and others, and results are available in the literature for a broad range of the lunar transfer trajectories.[6,7]

The calculations for injection into and out of lunar orbit are as follows:

$G = 1.068 \times 10^{-9}$ ft.^{3}/lb.-sec.^{2}

M_M = lunar mass = 0.0123 earth's mass = 1.622×10^{23} lb.

Lunar radius = 1080 miles

Lunar orbital altitude = 1,000 miles

Velocity of 1000 Mile Circular Lunar Orbit

$V_c = \sqrt{\frac{\mu}{a}}$ where a = 1080 + 1000 = 2080 miles

$\mu_M = GM_M = 1.7343 \times 10^{14}$ ft.^{3}/sec.^{2}

$$V_c = \sqrt{\frac{1.7343 \times 10^{14}}{2080\ (5280)}}$$

$$V_c = \sqrt{15.791 \times 10^6}$$

V_c = 3974 fps

Velocity of Vehicle Approaching the Moon at 1,000 Mile Altitude

$V^2 = V_\infty^2 + V_e^2$ where V_e = escape velocity = $\sqrt{2}\, V_c$

$V_e = \sqrt{2}\ (3974)$

V_e = 5619 fps

V_∞ = moon velocity = $\frac{D}{T} = \frac{2\pi R}{28 \text{ days}}$

$$V_\infty = \frac{2\pi\,(240{,}000)\ (5280)}{28\ (24)\ (3600)} \text{ fps}$$

V_∞ = 3291.2 fps

$$V = \sqrt{(3291)^2 + (5619)^2}$$

$$V = \sqrt{42.4085 \times 10^6}$$

V = 6512 fps

Injection Velocity for Lunar Orbit

$$\Delta V = V - V_c$$

$$\Delta V = 6512 - 3974$$

$$\Delta V = 2538 \text{ fps}$$

3. LUNAR LAUNCH WINDOWS

For excursion flights to the moon from the earth or an earth orbiting launch platform, the launch window represents the physical property of time in which the geometrical positions of the space vehicle, the moon and the earth are such that the space vehicle can be launched along a trajectory or orbital path that will intercept the moon (or earth) by passing from its original trajectory plane to a lunar (or earth) intercept plane.[5] For operation from an orbital station the width or total time and spacing of the launch windows are determined by the geometrical relationship of the space station - earth - moon system, the precession of the lunar and space station planes, and the additional propulsion energy available for transfer trajectory corrections.

The earth's equator is inclined 23.5° to the Ecliptic Plane (the earth's orbit or path around the sun) while the lunar plane (the moon's orbit or path around the earth) is inclined 5° to the Ecliptic Plane. Thus the moon's inclination to the equator varies between a maximum value of 28.5° (23.5° + 5°) and a minimum of 18.5° ($23.5^\circ - 5^\circ$) during an 18.6 year period due to lunar plane precession. Because of the perturbational effect of the earth's oblateness, the space station orbital plane precesses about the earth's polar axis in a direction opposite to the satellite motion and for orbits of approximately 30° inclination and 300 mile altitude this precession rate is somewhere between 6° and 7° per day. Despite the continuously changing differences between the lunar plane and plane of the station due to precession, it is possible to make required plane changes that will increase the launch window width at the expense of relatively small added velocity increments (Δ V) during certain flight departure times from the precessing station plane. These times, at which the required velocity changes are reasonably low, are what we have defined as the overall launch window. The basic or minimum launch window (no Δ V added) consist of two types.

1. Those occurring when the precession brings the station orbital plane into approximate coincidence with the lunar plane. (These windows are larger than type 2.)
2. Those occurring when the moon in its orbit will cross the intersection of the two planes (line of the nodes) at the time that the vehicle arrives in the moon's vicinity. (These are shorter windows but occur more frequently than type 1.)

These two types of windows follow each other in succession to give the total window openings available per precession period as shown in Figure 8.[a] The precession period of Figure 8 has been established as a two lunar month (54.6 days) repeating cycle by selection of orbital inclination (30°), orbital altitude (300 miles) and lunar-orbit inclination (18.5°). The launch windows have been widened by a velocity increment (Δ V) of 1200 fps. These launch windows are presented diagrammatically with the various phases of the lunar month in Figure 9 with the end of the second month coinciding with the

[a] Taken from work of Reference 6.

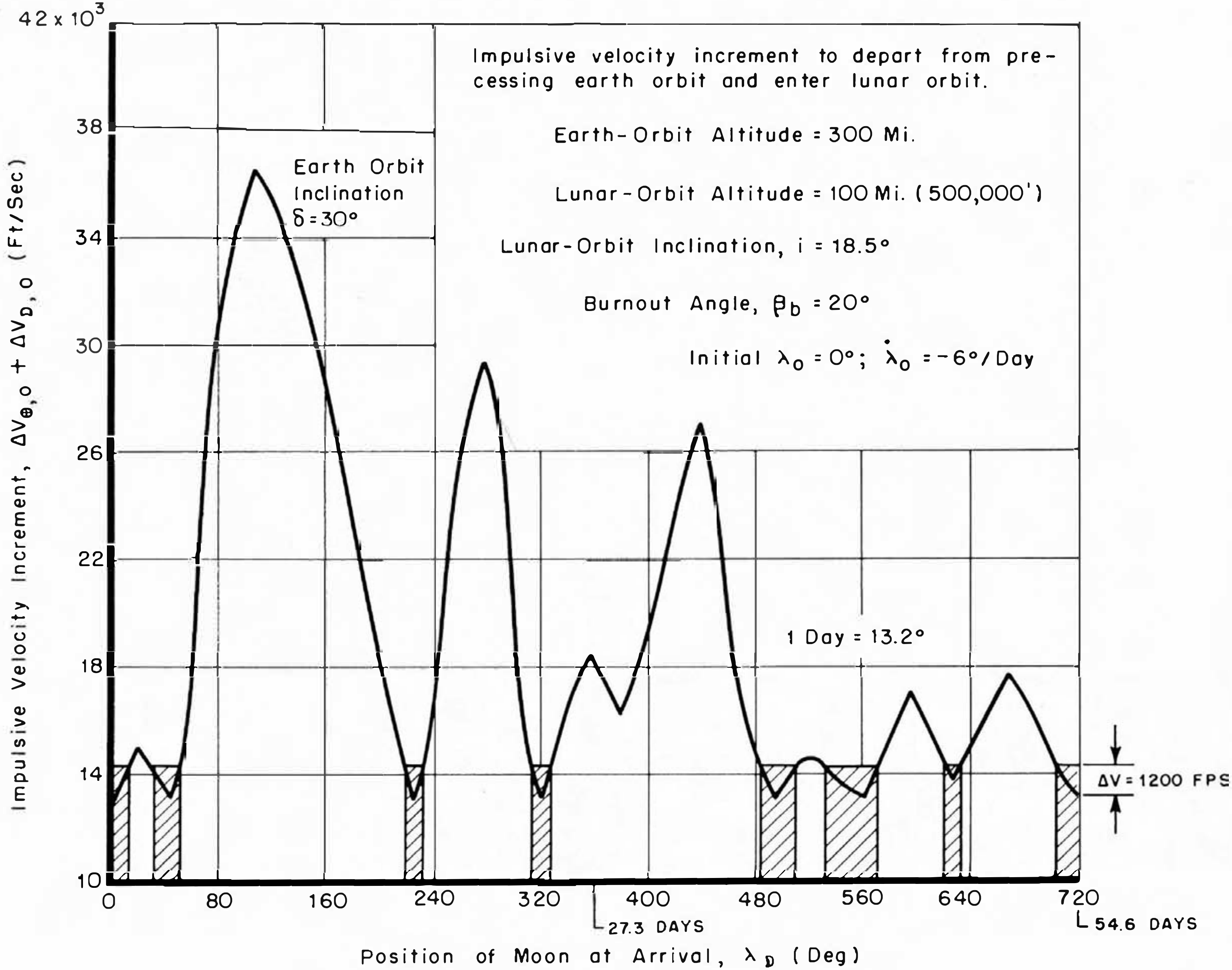


Figure 8 - Lunar Launch Windows with ΔV of 1200 fps

start of the first lunar month in a repeating two month cycle. Thus the velocity requirement for entering the lunar intercept plane will vary from zero to 1200 fps for the vehicle considered in this analysis.

4. RENDEZVOUS AND DOCKING

Many analyses and much data have been generated in the field of transfer orbits, rendezvous, docking and their related maneuvers. This analysis will not belabor this area but instead will discuss the general requirements of rendezvous that are applicable to this study. For efficient rendezvous operations going into earth orbits, the target vehicle (space station) must be in a rendezvous-compatible orbit where the satellite ground traces are synchronized with the rotational period of the earth. This synchronization is obtained by properly selecting the period of the space station in a circular orbit and the orbital inclination with respect to the launch site latitude. To meet this requirement, the earth space station in this analysis will have a period of 96 minutes at approximately 300 miles altitude in about a 30° inclined orbit and will complete exactly 15 orbits around the earth per day. To keep the space station in this orbit and to maintain the period of the satellite, it is necessary to provide station-keeping and attitude control capabilities, i.e., the periodic application of small amounts of thrust (velocity impulse) to counter the decay or drift of the orbit or variations in the station's attitude due to various perturbations or disturbing forces acting on the space station such as: (1) atomspheric drag, (2) radiation pressure, (3) charge drag, (4) magnetic field effects, and (5) gravitational attraction of the earth. A supply of fuel sufficient for accomplishing this station-keeping will be ferried to the station each

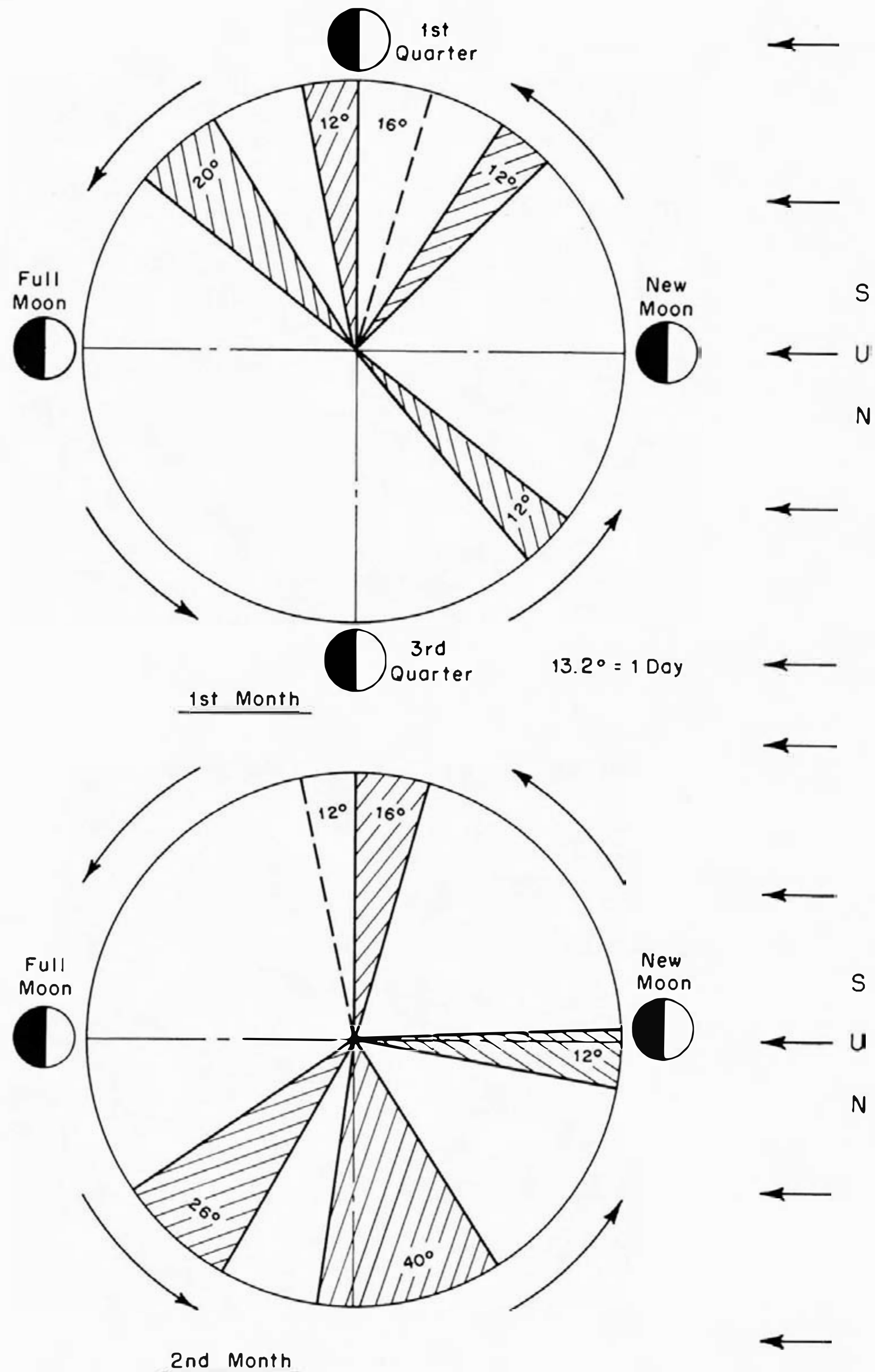


Figure 9 - Lunar Launch Windows During the Lunar Month

month during the regular, normal, monthly resupply operation, (see page 15).

The actual rendezvous operation (whether between an earth ascent vehicle or an orbiting vehicle and the station) consists of three phases: (1) parking and transfer orbit, (2) rough closure and (3) terminal or docking maneuvers. The procedure for successful rendezvous is to change the orbital elements of two space satellites, as necessary, so that they will match in position and velocity at a known future time.

To accomplish these objectives and to reduce waiting time for favorable target constellations, it is advantageous to place the maneuvering vehicle (interceptor) in an intermediate higher or lower coplanar orbit (the parking orbit). In the case of a higher intermediate or parking orbit, the station or target catches up with the intercepting satellite because of its higher velocity and shorter orbit circumference, while for a lower parking orbit, the maneuverable satellite overtakes the target from the rear. At the proper time, a velocity change has to be made in the orbital velocity of the interceptor satellite to initiate the rendezvous maneuver for a transfer orbit that will place the satellite at the same altitude as the station. This velocity increment (Δ V) is negative if the parking orbit is higher and positive if the orbit is lower than the target orbit. After reaching the target orbit altitude another velocity change has to be applied to the maneuvering vehicle so that the velocities between the two vehicles are roughly equal. This second velocity increment (Δ V) is again negative if the interceptor descended from a higher parking orbit and is positive if the vehicle ascended from a lower orbit. The total velocity change required to accomplish a 100 mile transfer orbit (where the target is at 300 mile altitude and the interceptor is at 200 or 400 mile altitude) is approximately 286 fps and each change of ΔV = 143 fps. For a 200 mile transfer the total ΔV is about 566 fps. Thus, for short excursions into space from a 300 mile earth orbit and for earth to orbit flights a velocity increment of 300 feet per second is considered satisfactory. For longer distance space excursions 600 fps appears adequate to bring the vehicles in close proximity of each other (on the order of 20 miles or less).

The second phase of rendezvous, or rough closure, in which coarse errors in altitude, position or velocity between the two vehicles are corrected probably will require velocity increments in the order of 50 fps. This velocity change will allow position or altitude corrections of up to 15 miles in case the transfer maneuver causes such errors so that the vehicles will end up with a separation of a mile or two and a relative velocity of a few feet per second for the third and final phase of rendezvous.

The terminal or docking maneuvers to mate the two vehicles within one complete orbital revolution or less will probably require an incremental velocity varying between 5 and 20 fps. This terminal correction should place the two vehicles within a couple of feet of each other with a relative velocity approximating 0.01 fps or essentially zero and permit the actual coupling.

Thus, the complete rendezvous maneuver should require for short space excursions and earth ascent flights a total velocity increment of 370 fps (300 + 50 + 20) and for longer space excursions ΔV will be approximately 670 fps (600 + 50 + 20). Most rendezvous studies mention results on the order of 500 - 1000 fps to accomplish rendezvous. For this analysis the 370 was rounded to 500 fps while the 670 was increased to 1000 fps so that adequate rendezvous capabilities would be available in the space excursion vehicle at anytime with an ample reserve.

5. SPACE EXCURSION VEHICLE

The operational technique of making cislunar and lunar space flights from an orbital launch platform requires the devising of an orbital-launched space excursion vehicle that has a staging capability of delivering the total velocity increment (Δ V) required for the space flight to lunar orbit and return. This total Δ V from the preceding calculations is 28,846 feet per second as shown in Table 5.

Table 5 - Incremental Velocity for Excursion From Earth Orbit to Lunar Orbit and Return

Maneuver	Purpose	Δ V (fps)
Ejection out of earth orbit	Translunar space passage	10,060
Injection through launch window	Gain lunar intercept plane	1,200
Injection into circular orbit	Lunar orbital flight	2,538
Lunar orbit rendezvous	Dock to lunar space station	250
Ejection out of lunar orbit	Transearth space passage	2,538
Injection through launch window	Gain earth orbit plane	1,200
Injection into circular orbit	Earth orbital flight	10,060
Earth orbit rendezvous	Dock to earth space station	1,000
		28,846

All other space excursions in cislunar or circumlunar flight will have a lower total ΔV requirement than this lunar orbital mission, and the vehicle should also perform these missions satisfactorily.

The space excursion vehicle will be developed by utilizing existing or under development space equipment as discussed in Section II-2 and II-3. The manned vehicles considered included the Mercury, Gemini, Apollo, and Dyna Soar; and the Gemini capsule was selected for the first generation excursion vehicle because of its versatility and availability, as listed in Section II-2. Likewise, the Titan II booster was selected as the earth launch vehicle because of its compatibility to the Gemini capsule (see Section II-3). Thus, the booster for space propulsion is the most important item yet to be selected for the excursion vehicle. Boosters that can be used for this purpose include the Agena B or D, Centaur, Titan II second stage, and Saturn C-1 second stage (S-IV). Table 6 lists some of the characteristics of these boosters.

Table 6 - Space Propulsion Boosters

Booster	Thrust (lbs.)	Propellant Type	Propellant Quantity (lbs.)	Empty Wt. (lbs.)	Availability
Agena B	16,000	RFNA/UDMH	13,500	2,500	1960
Centaur	30,000	LOX/Liquid H_2	-	-	1963-64
Titan II 2nd Stage	100,000	Nitrogen Tetroxide/ UDMH and Hydrazine	74,000	7,500	1962
Saturn S-IV	90,000	LOX/Liquid H_2	100,000	18,000	1964-65

The Saturn S-IV stage may be too large for efficient staging (mass ratios) with a 6,000 lb. Gemini payload and if it could be staged properly, the Saturn would be a second generation type booster (perhaps with larger payload, such as Apollo). Saturn C-1 flight testing and Apollo earth-orbit flights would also delay the availability of these stages for anyother use by several years. Likewise, the Centaur program appears to have suffered major delays, technical difficulties and program slippage and prior commitments would put this booster into a later time availability for a program such as the one considered here. Both these boosters involve cryogenic fuels and orbital refueling and storage of liquid oxygen and liquid hydrogen could be significant technical problems. Thus only the Agena and second stage of the Titan II are examined in this analysis. The relationship between velocity and vehicle (booster) performance is:

$$\Delta V = I_{sp}\, g \,\log R \tag{8}$$

where I_{sp} = specific impulse (seconds)

g = acceleration due to gravity = 32.22 ft./sec.

R = mass ratio = $\frac{W_o}{W_e}$

W_o = original weight - before fuel consumption (lbs.)

W_e = empty weight - after fuel consumption (lbs.)

For the Agena B - Gemini vehicle, the initial velocity available from the internal Agena fuel supply is:

$$\Delta V = (300)\,(32.22)\,\log \frac{22500}{9000}$$

where $I_{sp} = 300$ (vac)

$W_o = W_e + \text{fuel} = 9{,}000 \text{ lbs.} + 13{,}500 = 22{,}500 \text{ lbs.}$

W_e = Wt. of Agena + Gemini + Support Structure

$W_e = 2{,}500 + 6{,}000 + 500 = 9{,}000 \text{ lbs.}$

and $\Delta V = 8{,}850$ fps

To increase the incremental velocity ΔV, it is necessary to add additional fuel to the vehicle by attaching fuel tanks to the vehicle. By this staging arrangement various velocities can be obtained but as the gross weight of each stage increases the law of diminishing return takes hold and large quantities of fuel give rather small increases of velocity. This statement will be apparent in the following calculations.

To get the proper staging (mass ratio) weights various combinations of fuel tanks were tried with the basic Agena B-Gemini. They included tanks containing 15,000; 20,000; and 25,000 lbs. of fuel respectively. Many other combinations were examined but the best results were obtained from fuel tanks each weighing 800 lbs. with a 25,000 lb. fuel supply when used in sets of two for axial symmetry. Rapid calculations can be made when equation (8) is put in the form

$$\frac{W_o}{W_e} = e^{\frac{\Delta V}{9666}}$$

and solving ΔV for each stage (set of fuel tanks).

	Original or Final Stage (Agena B-Gemini)	1st Added Stage	2nd Added Stage	3rd Added Stage	4th Added Stage
Initial W_e (lbs.)	9,000	22,500	74,100	125,700	177,300
Weight of 2 Fuel Tanks (lbs.)	0	1,600	1,600	1,600	1,600
Total W_e (lbs.)	9,000	24,100	75,700	127,300	178,900
Fuel (lbs.)	13,500	50,000	50,000	50,000	50,000
W_o (lbs.)	22,500	74,100	125,700	177,300	228,900
W_o/W_e	2.50	3.08	1.66	1.39	1.28
$\Delta V/9666$ (fps)	0.917	1.125	0.507	0.33	0.247
ΔV (fps)	8,850	10,880	4,900	3,190	2,390

19,730

24,630

27,820

30,210

By adding four pair of tanks, or a total of eight fuel tanks, holding 200,000 lbs. of fuel (25,000 lbs. per tank) to the basic Agena-Gemini vehicle, a total incremental velocity of 30,210 fps can be produced. To obtain the 28,846 fps required to complete the lunar orbit and return, see Table 5, the values in the fifth stage must be calculated in reverse order as follows where W_e = 178,900 lbs. and total ΔV for the first four stages = 27,820 fps.

ΔV = 28,846 fps - 27,820 fps = 1,026 fps

$\Delta V/9666 = 0.106$ $\quad W_o/W_e = 1.114$

W_o = 198,900 $\quad$ Fuel = $W_o - W_e$ = 20,000 lbs. in 5th stage

Thus, a total of 183,500 lbs. of fuel will deliver the required 28,846 fps of velocity to take a 6,000 lb. payload from earth orbit to lunar orbit and return to earth orbit including rendezvous in both lunar and earth orbits with an Agena B-Gemini combination.

For the Titan II second stage-Gemini vehilce concept, the same calculations are repeated to determine the fuel requirements for this vehicle to obtain the same total ΔV. The initial velocity for the upper stage of Titan II is:

$$\Delta V = (316)\ (32.22)\ \log \frac{88,000}{14,000}$$

where W_e = wt. of second stage + Gemini + support structure

W_e = 7,500 + 6,000 + 500 = 14,000 lbs.

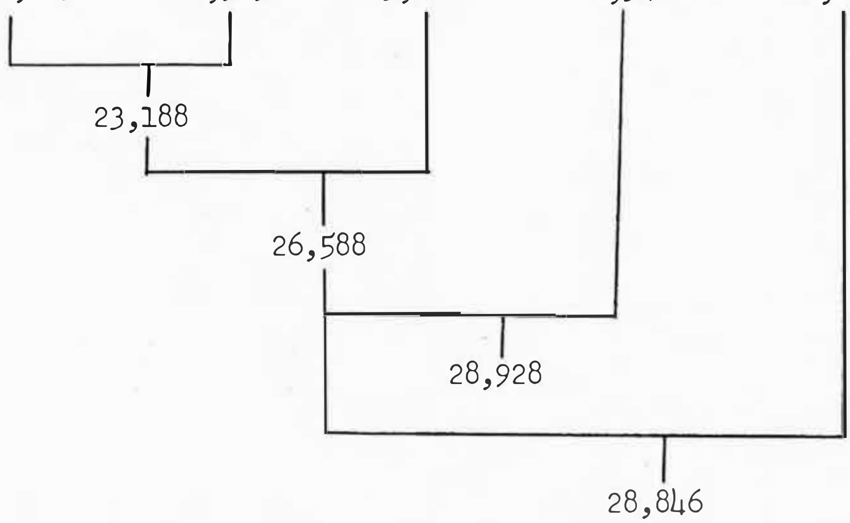
$W_o = W_e$ + fuel = 14,000 + 74,000 = 88,000 lbs.

I_{sp} = 316 (vac.)

ΔV = 18,683 fps

In staging the Gemini-Titan II second stage both 25,000 and 50,000 lb. tank combinations were calculated. Best results (less fuel to obtain a ΔV of 28,846 fps) were obtained by using the smaller 25,000 lb. tanks weighing 800 lbs. each. The calculations for each stage are as follows:

	Original or Final Stage (Titan II -Gemini)	1st Added Stage	2nd Added Stage	3rd Added Stage	3rd Added Stage for 28,846 fps
Initial W_e (lbs.)	14,000	88,000	139,600	191,200	
Weight of 2 Fuel Tanks (lbs.)	0	1,600	1,600	1,600	
Total W_e (lbs.)	14,000	89,600	141,200	192,800	
Fuel (lbs.)	74,000	50,000	50,000	50,000	45,200
W_o (lbs.)	88,000	139,600	191,200	242,800	238,000
W_o/W_e	6.285	1.558	1.354	1.259	1.248
$\Delta V/10,182$ (fps)	1.835	0.445	0.304	0.240	0.2218
ΔV (fps)	18,683	4,505	3,400	2,340	2,258



To obtain the stated total velocity of 28,846 fps requires the expenditure of 219,200 lbs. of fuel in the Gemini-Titan II combination using 25,000 lb. fuel tanks and, for the record, a total of 237,750 lbs. of fuel using 50,000 lb. tanks in the same basic Gemini-Titan II second stage concept. The Gemini-Agena vehicle used 183,500 lbs. of fuel to achieve the same Δ V and since fuel will be a major logistic item during space excursion flight operations, the difference of 35,700 lbs. of fuel for each lunar flight will be substantial over the total operational period. Thus, the Gemini-Agena concept was selected for this analysis although it is recognized that some other system combination may be optimum but would require a larger design and engineering effort than is available for this analysis. In any case, the selected system will adequately demonstrate the feasibility and potential of the space excursion concept, and affords many attractive advantages.

A preliminary design configuration of this space vehicle which is assembled in orbit at the Earth space station can now be developed from the above data. The first requirement is a support structure that will mount the Agena B to the Gemini capsule. This structure can also position and support the eight 25,000 lb. capacity auxiliary fuel tanks in circular fashion about the Agena B booster as shown in Figure 10.

The support structure consists of an adapter section which attaches the Agena B to the Gemini and a support arm structure which mount on the adapter section and attaches the auxillary fuel tanks to the basic vehicle. The weight of 500 lbs. for the adapter and support section include the following items: (1) adapter structure with attachment fittings to Gemini and Agena, (2) wiring for guidance and control, release, and auxillary tank valves, (3) guidance package installation provisions and (4) installation provisions for pressure bottles, and miscellaneous equipment. The auxillary fuel tank weight of 800 lbs. each included: (1) the weight of the tank (5' dia. x 18'), (2) attachment fittings and release, (3) valves, fittings and lines for

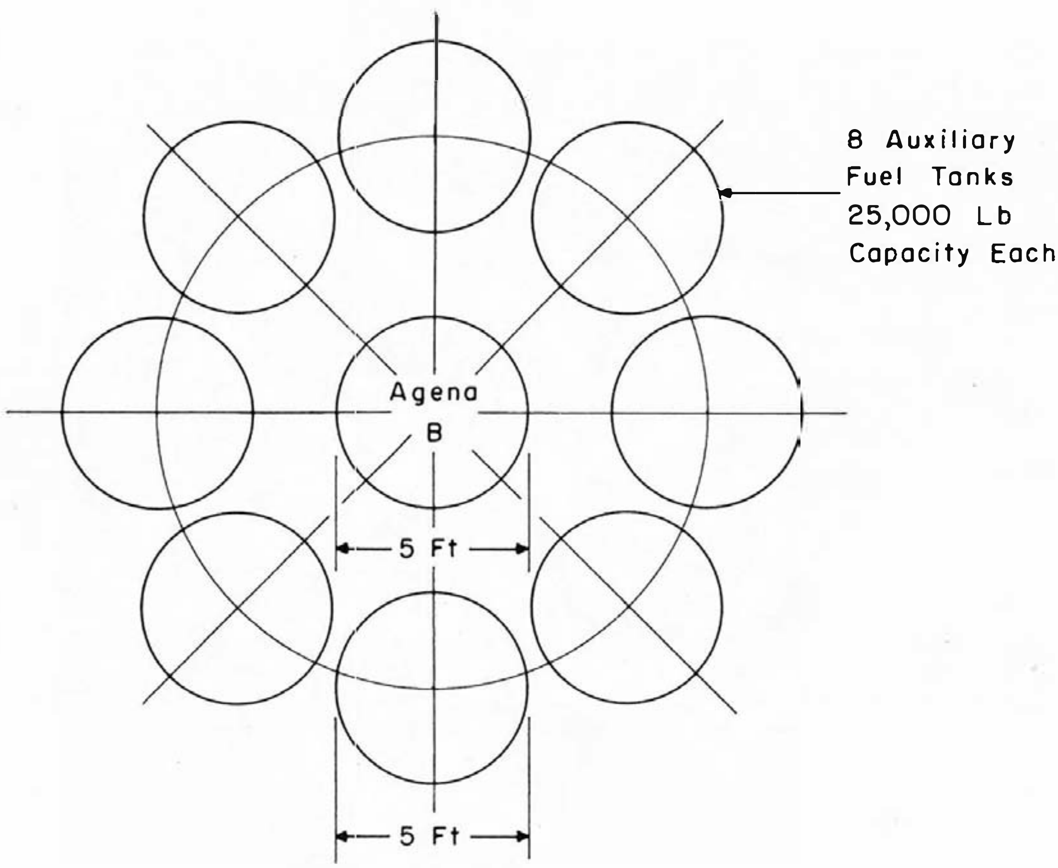


Figure 10 - Layout of Agena B and Auxillary Fuel Tanks

propellants and pressurization, (4) pressure bottles with fittings, gas, and valves and (5) fuel tank support arms which mount on the adapter section.

The Agena B booster overall dimensions are shown in Figure 11. For use in the space excursion vehicle the payload nose cone would be removed making the Agena B overall length 23 feet. The propellants used in the Agena are nitric acid (IRFNA) for the oxidizer and unsymmetrical dimethyl hydrazine (UDMH) both of which are storable and thus quite suited to space use. In addition the Agena B has a space (vacuum) restart capability.

Figure 12 illustrates the overall configuration and dimensions of the Gemini capsule. This capsule accomodates a two man crew, has two doors for exit and entrance, can be depressurized in orbit for access to space by the crew members, and can support manned space flight for 14 days with its self contained life support equipment. In addition, the Gemini will have sensors and provisions for rendezvous and docking as well as some landing maneuverability due to an offset center of gravity (cg).

Putting all the pieces of the Space Excursion Vehicle (the Gemini, Agena B, fuel tanks, adapter section, etc.) together by space assembly operations would result in a composite vehicle similar to the one shown in Figure 13. Such a space based vehicle would be capable of travelling from Earth orbit to the Moon and back. Also, if it were refueled in lunar orbit and outfitted with landing outriggers it could make a lunar landing from the lunar orbital space station but this is getting ahead of our analysis. First we must learn to walk before we can run and the walking involves the development of an operational space system based on regular and frequent space operations which will be discussed in the next section.

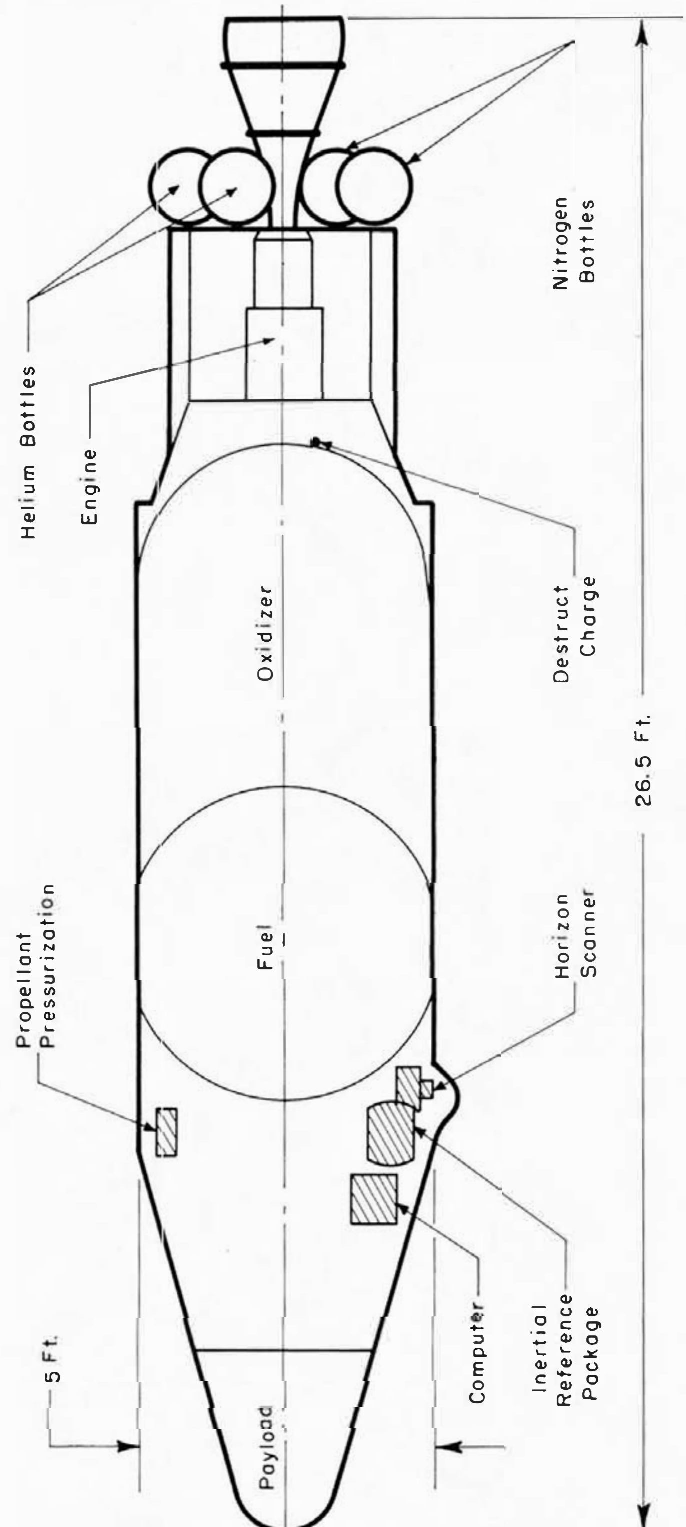


Figure 11 - Overall Dimensions of Agena B

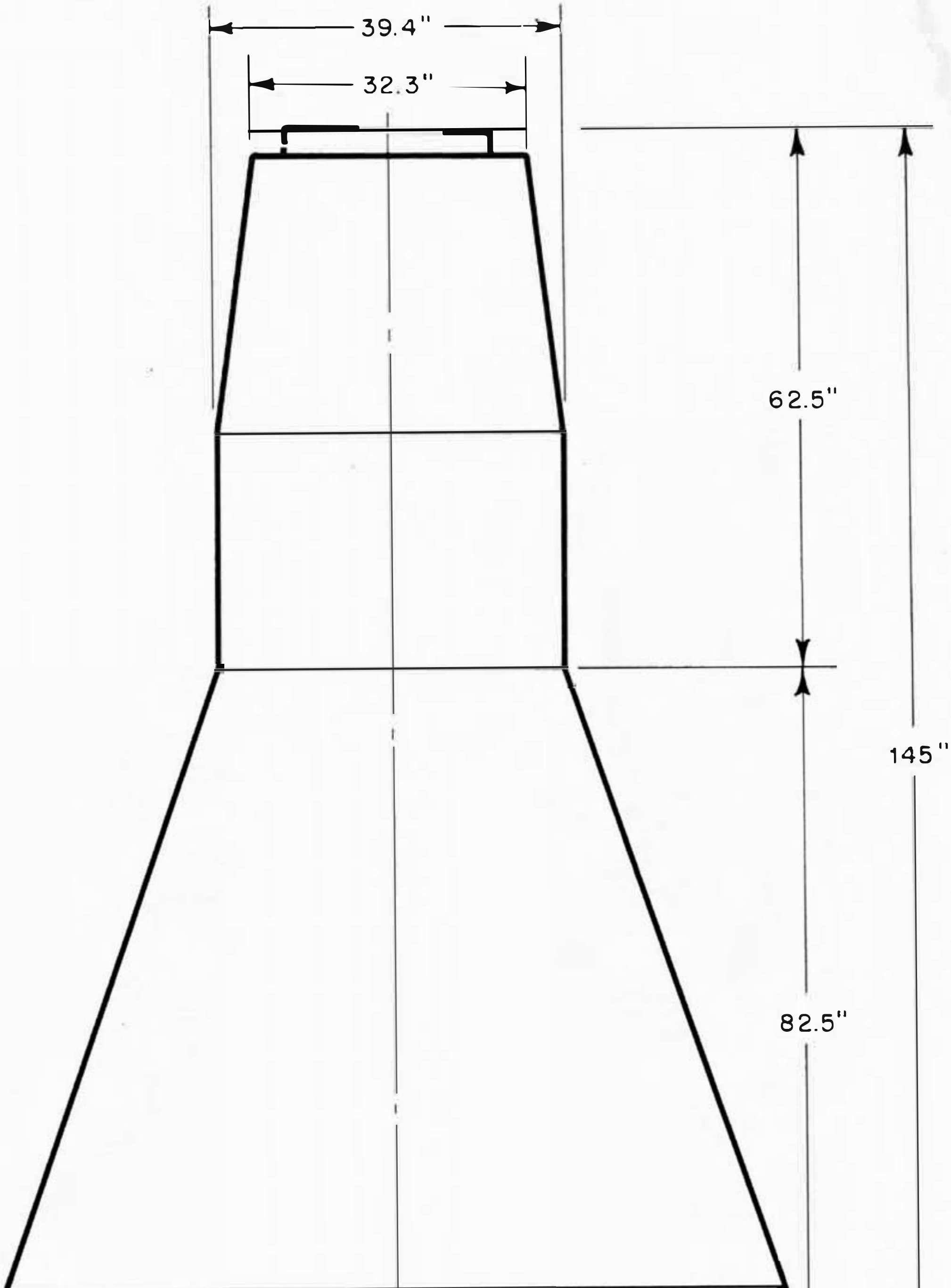


Figure 12 - Overall Dimensions of Gemini

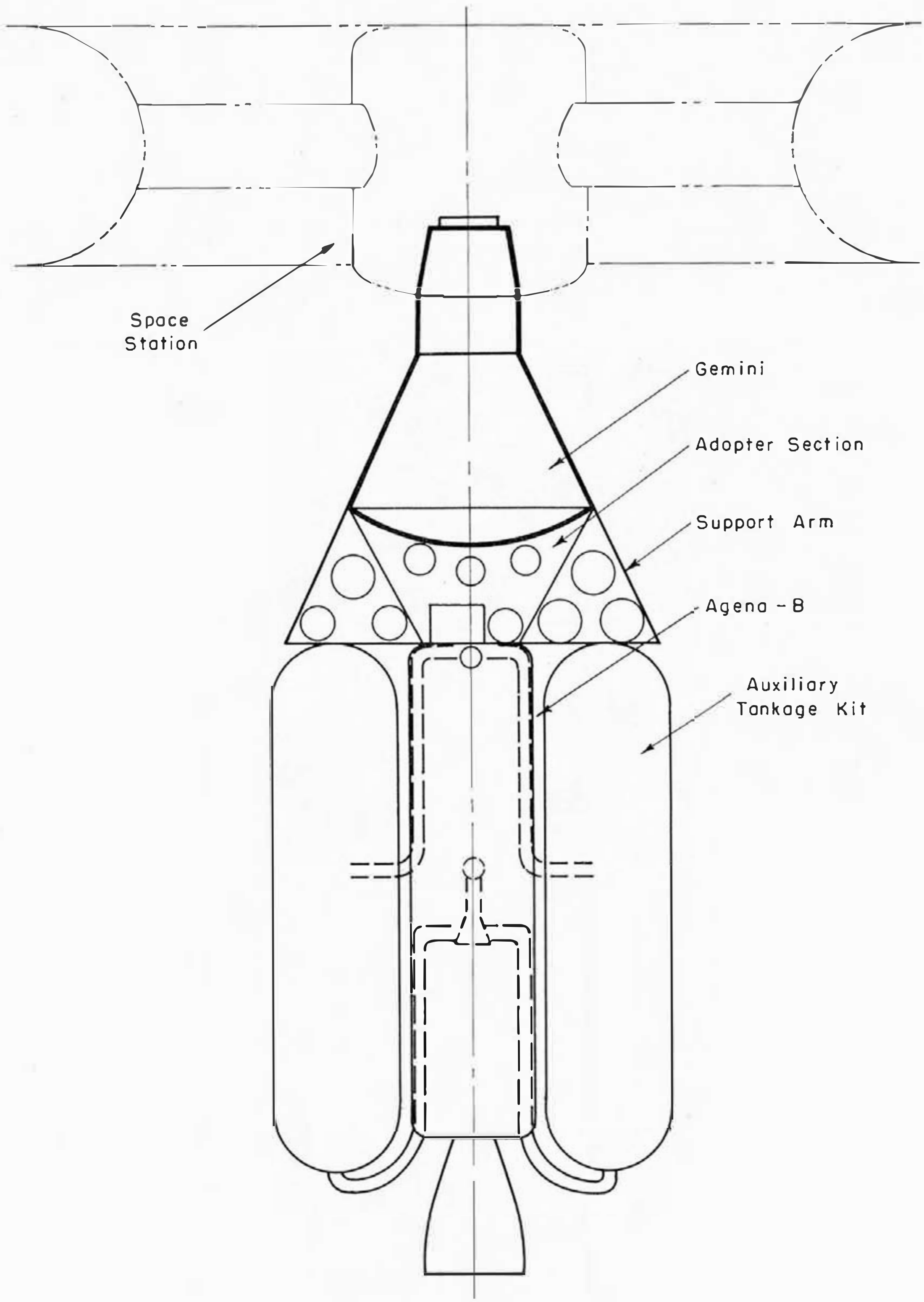


Figure 13 - Space Excursion Vehicles

Page intentionally left blank

SECTION IV - SPACE EXCURSION OPERATIONS AND LOGISTICS

1. GENERAL APPROACH

Based on the system characteristics derived in the preceeding section, the analysis can now investigate the general operational approach to determine flight schedules, equipment, facilities, logistics and costs involved in employing the space excursion system. The events of one operational cycle from an earth-launching through the entire system network to return-to-earth are summarized in the following operational sequence.

The initial space operations would begin with the launching from Cape Canaveral of the basically-equipped, unmanned space station into a 300 mile, circular orbit of 30° inclination. After orbital injection and check-out from the Mercury-Gemini tracking network, automatic deployment of the station would be initiated and systems activated remotely from a ground command station. Then, after check-out via telemetering confirmed that at least all vital systems were functioning satisfactorily, manning of the space station can be instituted two days after the original launching. Initially two men in a Gemini vehicle will rendezvous with the station and dock. After entering the station, their first task would be on-the-spot inspection, check-out and further activation of the station systems for full habitability. Two days later, a second Gemini crew would board, and subsequent manning would send a third crew into orbit six days later or 10 days after the original station launching, and the fourth and last crew would be put aboard 10 days after the third crew. Thus, the space station with its full eight man crew would be in an operational status twenty days after launch by the firing of five (5)

Titan II boosters from Cape Canaveral. Since data indicates that a maximum of fourteen (14) days will be required between the arrival of a Titan II at the launch pad and the actual firing of the missile, a total of four (4) launch pads would be required. The first four launches would be from individual launch pads but the fifth launching would be from the same pad that the space station used twenty days earlier. Within sixty days of the time that the first crew members arrived aboard the station, crew rotation would be initiated so that each crew would be replaced on or before the end of their two month, orbital tour-of-duty. In addition a monthly, unmanned resupply flight would be launched to the earth-orbiting space station containing the supplies and equipment listed on page 18.

The tentative duties of the various crew members, both primary and secondary, are listed in Table 7. Each crew member would be trained in multiple primary duties and cross-trained in additional specialties which are listed as their secondary duties. General management of the station would require a commander and deputy commander to be responsible for the administration and performance of the overall space operations. The deputy commander will also be designated the chief pilot/astronaut and space flight operations supervisor in charge of all flights of the Gemini capsule from and to the earth and the excursion vehicle into space. Reporting to the space flight operations supervisor are the two space excursion vehicle pilot/astronauts. An additional manager, the space station operations supervisor, is in charge of all space station functions including systems management, station maintenance and repair, supply, housekeeping, communication, infirmary, as well as the assembly, maintenance, repair and resupply of the space excursion vehicles. This

Table 7 - Operational Duties of Space Station Crew

No.	Title	Authority	Primary Duty	Secondary Duty
1	Station Commander	In complete command and responsible for space station and personnel	Exercise command function Docking officer (rendezvous) General station operations (8 hour watch)	Inspection of station maintenance Gemini pilot-astronaut Mechanical and electrical technician
1	Deputy Station Commander	1. In command during absence of commander. 2. In charge of space flight operations.	Assist the station commander Chief pilot - astronaut Inspection of vehicle maintenance	Docking officer (rendezvous) Relief and/or backup pilot-astronaut for space excursion vehicle flights
1	Space Station Operations Supervisor	1. In command during absence of commander and deputy commander 2. In charge of space station operations and maintenance	General station operations (8 hour watch) Communications - Radar Inspection of station maintenance Administration	Assist the station commander Gemini pilot-astronaut Mechanical and electrical technician
1	Operations Technician	None	General station operations (8 hour watch). Medic (dietician, first aid, radiation, atmosphere control)	Administration Docking officer (rendezvous) Fuel storage management
1	Mechanical Technician	None	Mechanical assembly and maintenance of vehicles and maintenance of station Fuel storage management Supply	Medic Housekeeping duties General station operations (as scheduled)
1	Electrical Technician	None	Electrical assembly and maintenance of vehicles and maintenance of station Housekeeping duties	Communications - radar Supply General station operations (as scheduled)
2	Pilot-Astronaut	Command of Space Excursion Vehicle (pilot authority only)	Gemini/Space Excursion Vehicle pilot-astronaut	Inspection of vehicle maintenance

supervisor has the three technicians who perform these duties reporting directly to him. In addition, the station operations supervisor is designated an alternate station commander in the absence of both the station commander and deputy station commander. The line organization of the space station and flight crews is shown in Figure 14.

After the station has been fully manned and declared operational, preparations for the first orbital flight from the station would begin. The space excursion vehicle is assembled, serviced and checked out and approximately one month after activation of the space station, the first short excursionary flight is made. At first short excursionary orbit and re-rendezvous flights below the Van Allen belt are scheduled based on the principle of very gradual extension of flight operational and navigation techniques from short flights just a few miles out until eventually full lunar orbiting and return flights are made. These early flights are followed by flights in and near the lower fringes of the Van Allen belt and as experience is gained by these step by step extensions of space travel, these flights are extended through the Van Allen belt, into the lunar intercept plane, and finally into circumlunar[7] and lunar orbit trajectories. After lunar orbital flight has proven successful with the space excursion vehicle, a lunar space station identical with the earth-orbiting space station would be launched from the earth station, put into lunar orbit and activated in the same manner as the earth station. This lunar space station would orbit the moon around the lunar equator at an altitude of 1,000 miles. Monthly supply flights and crew rotation at appropriate intervals would be required which enters additional flight requirements into the logistic pipeline. From this lunar space station, lunar

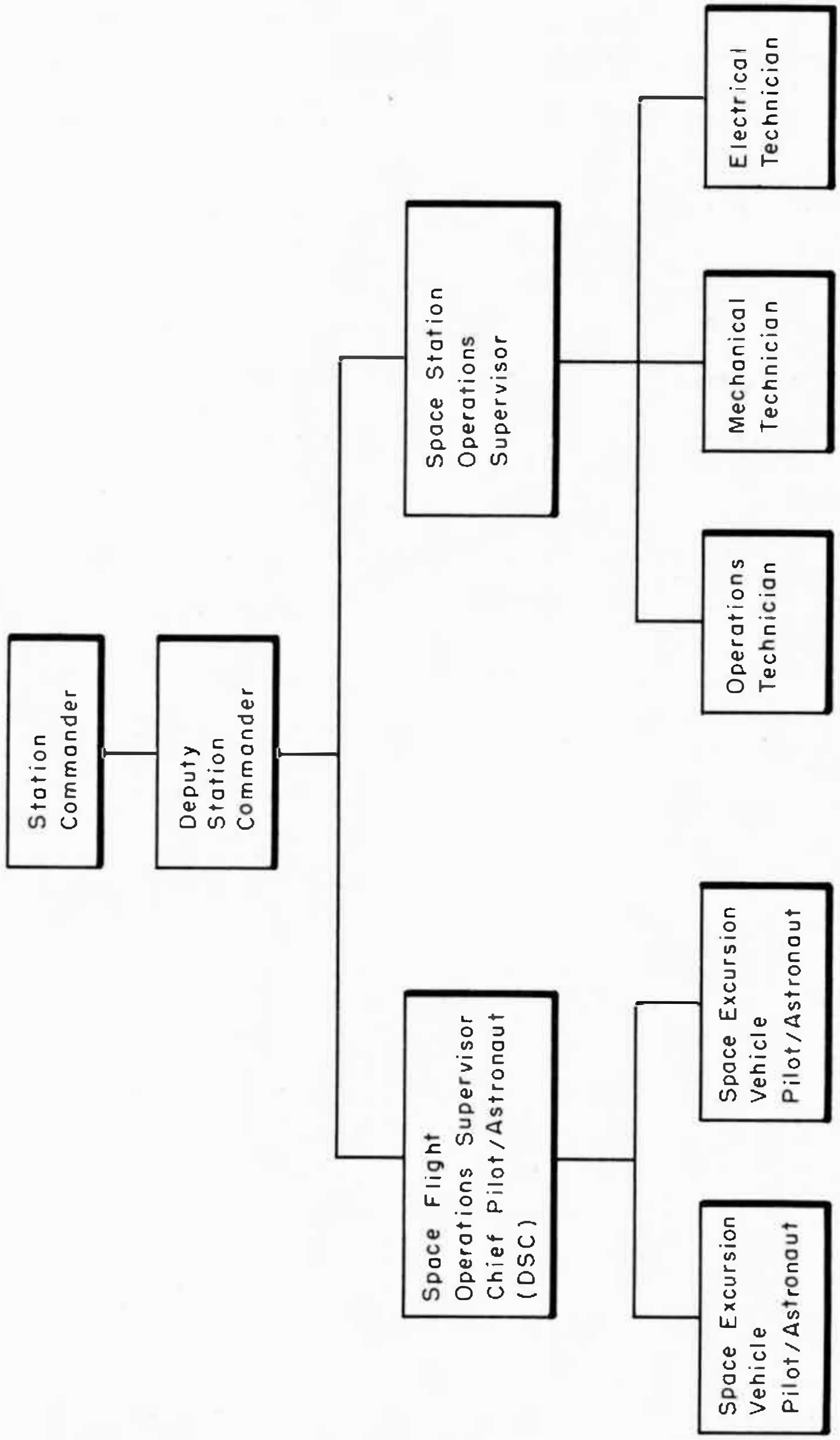


Figure 14 - Operational Space Station Organization Chart

landings and explorations can be accomplished and the logistics and techniques involved will be analyzed later in this section of the report.

Thus, all the operational flight vehicles required when all systems are operational in the space excursion concept are shown in Figure 15 for a trip from the earth to the moon and return. A lunar station crew or the excursion vehicle pilot/astronauts would launch from earth to the earth orbiting space station in a Gemini capsule boosted by a Titan II. Upon arrival at the station, they would transfer to a space excursion vehicle made up of a Gemini capsule, an Agena B and auxiliary fuel tanks with the complete vehicle containing sufficient fuel (183,000 lbs.) for a lunar orbit mission. After launch from the earth station they would travel through space till they entered a 1,000 mile lunar orbit and eventual rendezvous with the lunar space station. After serving their tour of duty (station crew) or making lunar landings (pilot/astronauts), they would again transfer to the space excursion vehicle and use the remaining fuel aboard to launch from the lunar space station, travel through space to a 300 mile earth orbit and dock at the earth station. They would then transfer to a Gemini capsule for the final leg of their journey. The return of the Gemini into the earth's atmosphere at orbital speed would be completed by a landing on the earth's surface.

The flight profiles for the space excursion missions are diagrammed in Figure 16. The lunar transfer orbit is number coded to correspond to the flight time schedule for each portion of the flight. The schedule includes the time required for the following; (1) docking and unloading from previous flight, (2) post-flight check, (3) servicing and maintenance, (4) fueling, (5) pre-flight check, (6) loading, checkout, and hold for launch window,

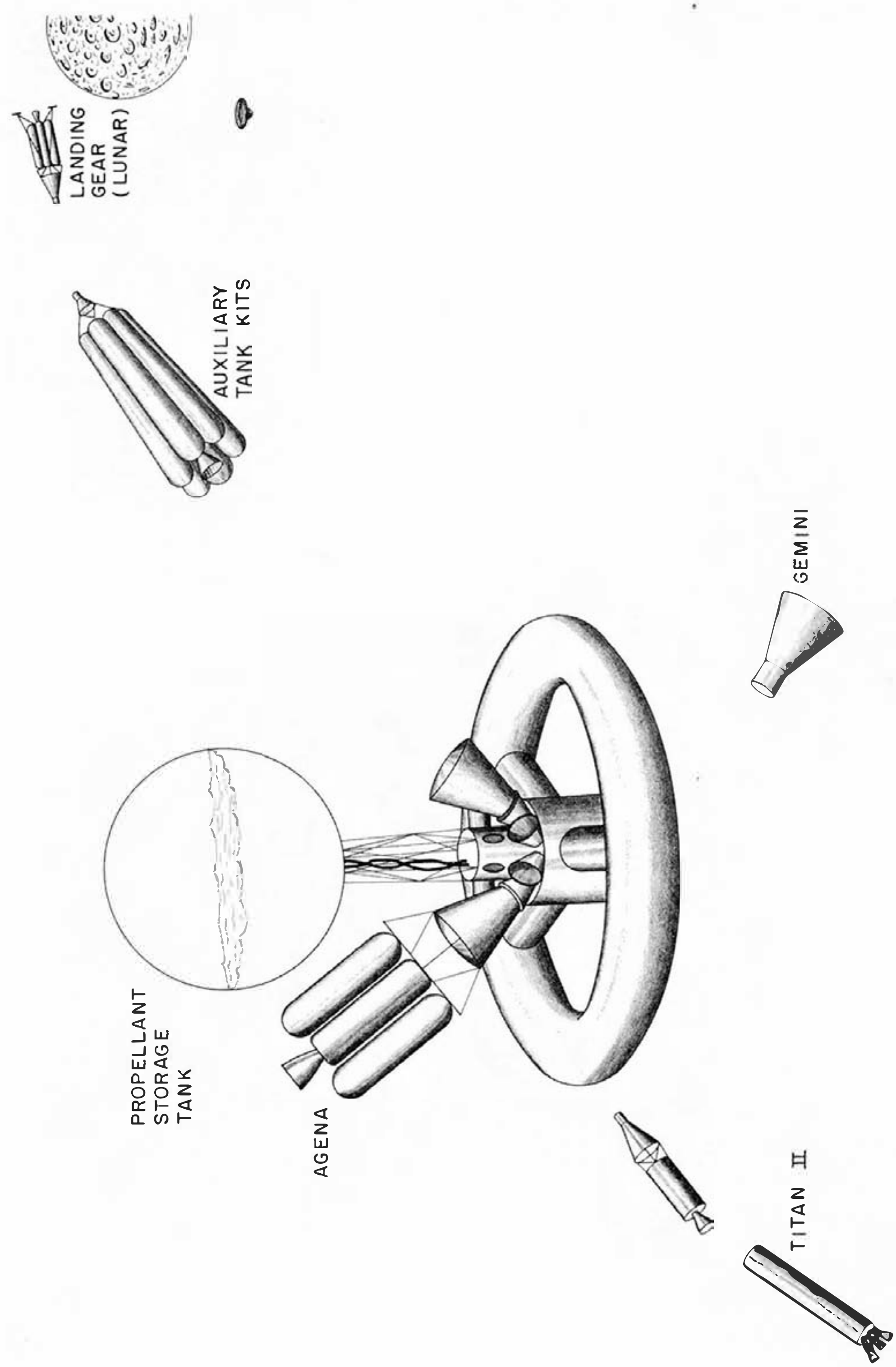


Figure 15 - Operational Flight Vehicles

Page intentionally left blank

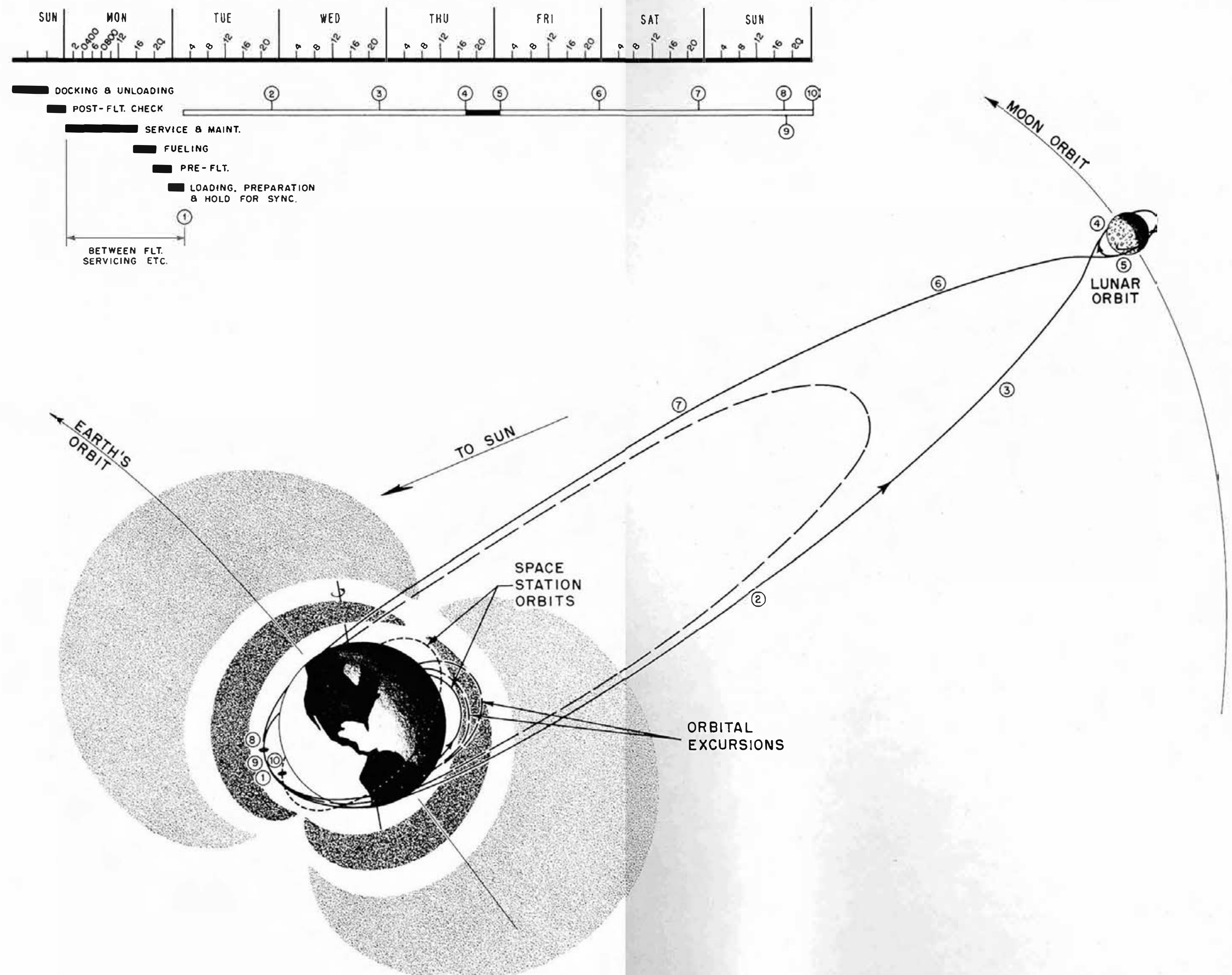


Figure 16 - Flight Profile

Page intentionally left blank

(7) launch and flight to the moon station, (8) lunar orbit, (9) launch and flight to the earth station, and (10) rendezvous, docking, and unloading. The flight schedule as shown, based on the estimated servicing cycle shown at the upper left, indicates that one lunar orbital flight per week can be accomplished by a space excursion vehicle if the time spent in lunar orbit does not exceed eight hours. For additional data on servicing and maintenance see Appendix E.

2. LUNAR OPERATIONS AND LANDINGS

To start the lunar operations, a lunar space station, identical to the earth orbiting space stations, is launched from an earth orbiting space station and put into lunar orbit. The total incremental velocity to put the station into lunar orbit is 14,048 ft./sec. (see the first four items of Table 5 - page 43). The fuel required by an Agena B to transport this packaged station from earth orbit to lunar orbit is:

Lunar station weight	=	6,800 lbs.
Agena B	=	2,500 lbs.
Support structure	=	500 lbs.
W_e	=	9,800 lbs.
Fuel (Agena B)	=	13,500 lbs.
W_o (final stage)	=	23,300 lbs.

$$\Delta V = I_{sp}\, g \log \frac{W_c}{W_e}$$

<u>ΔV for Agena B stage with 13,500 lbs. of propellant</u>

$$\frac{W_o}{W_e} = \frac{23,300}{9,800} = 2.378 \qquad \log \frac{W_o}{W_e} = 0.8671$$

77-10(1-53)M

$$\Delta V = I_{sp}\, g \log \frac{W_o}{W_e} = 9666\ (0.8671) = 8380$$

$$\Delta V = 8380 \text{ ft./sec.}$$

Propellant required in auxiliary tanks for required ΔV

$$\Delta V = 14{,}048 - 8{,}380 = 5{,}668$$

$$\Delta V = 5{,}668 \text{ ft./sec.}$$

$$\frac{W_o}{W_e} = e^{\frac{\Delta V}{I_{sp}\, g}} = e^{0.5864} = 1.793$$

$$\frac{W_o}{W_e} = 1.793 \qquad W_o = (24{,}900)\ (1.793) = 44{,}700$$

$$\text{Fuel} = W_o - W_e = 44{,}700 - 24{,}900 = 19{,}800 \text{ lbs.}$$

Fuel for orbiting lunar station from earth station launch

Total fuel = Agena fuel + Aux tank fuel

Total fuel = 13,500 + 19,800

Total fuel = 33,300 lbs.

The lunar space station is unmanned during its flight from earth orbit launch to lunar orbit injection with all navigation and thrust vectoring operated by remote control. To provide an extra velocity increment for error adjustment due to remote operation, the 250 ft./sec. ΔV normally provided for lunar orbital rendezvous has been included in the above calculations.

Once a space station begins operations in lunar orbit supply, fuel, and equipment flights are required to support the station. A 6,000 lb. unmanned payload is used for such flights and since unmanned rendezvous with the space station is required, an additional incremental velocity (ΔV) of 155 ft./sec.

is included for increased maneuvering capability. Thus the total velocity increment for these one way flights is 14,203 ft./sec. (14,048 + 155).

The calculations for the fuel required follow:

$$\text{Total } \Delta V = 14{,}203 \text{ fps}$$

$$\Delta V \text{ of Agena} = 8{,}850$$

$$\Delta V \text{ of aux tanks} = 5{,}353 \text{ fps}$$

$$W_o = W_e \; e^{\frac{\Delta V}{I_{sp} g}}$$

W_e = Agena + payload + structure + Agena fuel + aux tanks

W_e = 2,500 + 6,000 + 500 + 13,500 + 1,600

W_e = 24,100 lbs.

$$\frac{\Delta V}{I_{sp} g} = \frac{5{,}353}{9{,}666} = 0.5538 \qquad e^{0.5538} = 1.739$$

W_o = 24,100 (1.739) = 41,900

W_o = 41,900 lbs.

Fuel = $W_o - W_e$

Fuel = 41,900 - 24,100

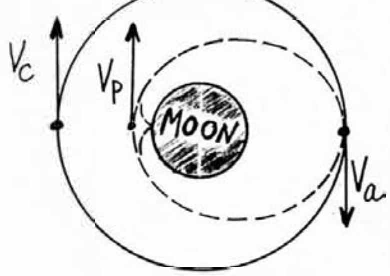
Fuel = 17,800 lbs.

Total fuel = 13,500 + 17,800

Total fuel = 31,300 lbs.

After the lunar space station is operational and has a logistic pipeline from the earth via the earth space station, lunar landing operations can be contemplated. Since the space excursion vehicle will carry the operating crews to the lunar space station it seems logical to utilize this vehicle for manned lunar landings and the unmanned supply vehicle for landing lunar caches. Both vehicles would require modification at the lunar space station to adapt them for these landings. Fuel and tanks in excess of those required for the lunar landing must be removed (and possibly the capsule heat shield) and outrigger landing gear installed. Figure 17 illustrates a manned lunar landing version of the excursion vehicle. The unmanned lunar cache would be a one way flight from the lunar orbiting station to the moon's surface. The manned excursion vehicle flights are round trip affairs with a lunar launch involved. The incremental velocities (Δ V) for landing and launch from the moon are calculated below.

Velocity for lunar landing and launch



Orbital altitude = 1,000 miles

$h_1 = 0 \quad h_2 = 1{,}000$ miles

$\mu = 1.7343 \times 10^{14}$ ft.^{3}/sec.^{2}

Radius of moon = 1,080 miles

$$a = \frac{h_1 + h_2 + 2R}{2} = 1{,}580 \text{ miles}$$

$$c = \frac{h_2 - h_1}{2} = 500 \text{ miles}$$

$$e = \frac{c}{a} = \frac{500}{1580} = .31646$$

$$\frac{1 + e}{1 - e} = 1.9259$$

$$\frac{1 - e}{1 + e} = 0.5192$$

$V_c = 3974$ fps

$$V_a = \left[\frac{\mu}{a}\left(\frac{1 - e}{1 + e}\right)\right]^{\frac{1}{2}} \qquad V_p = \left[\frac{\mu}{a}\left(\frac{1 + e}{1 - e}\right)\right]^{\frac{1}{2}}$$

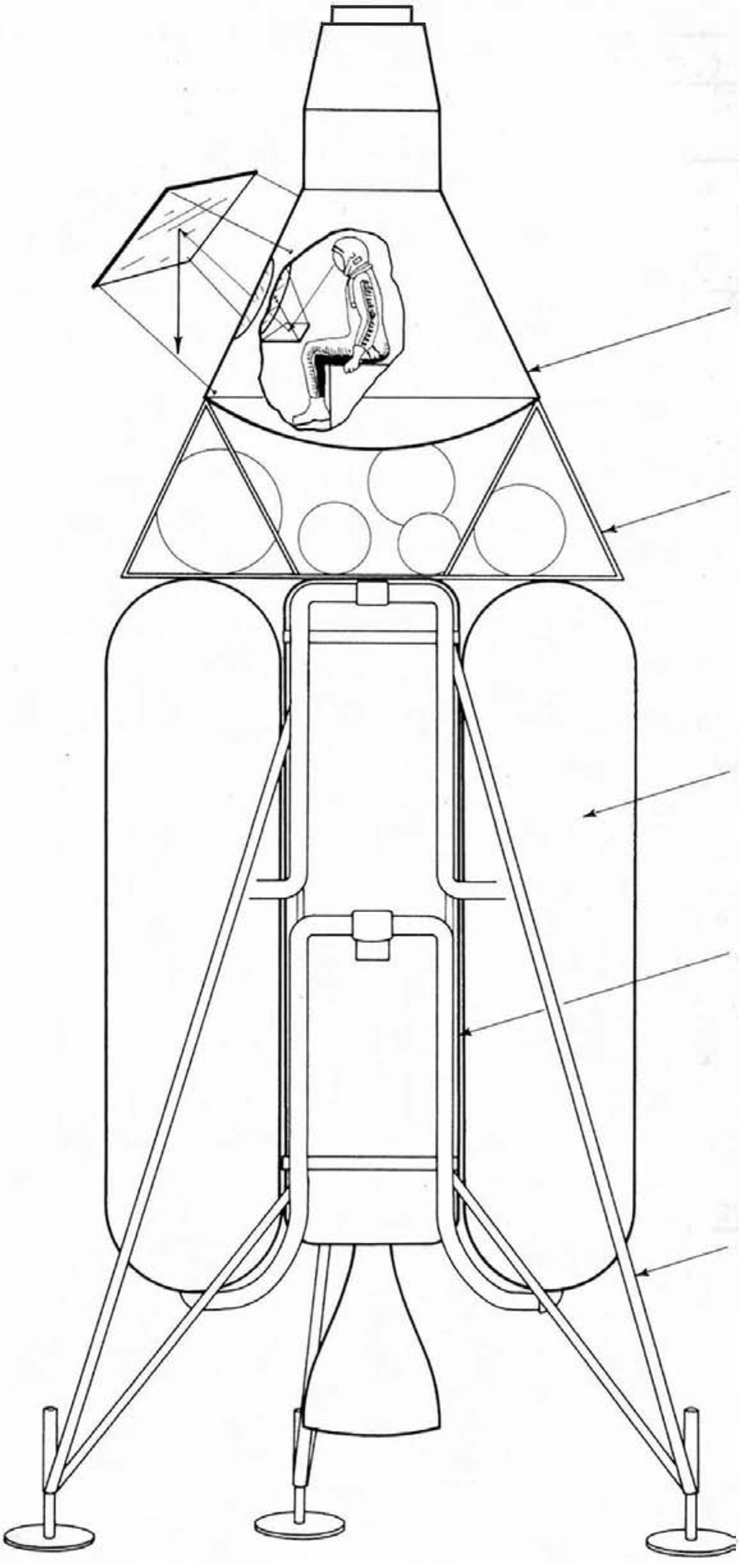


Figure 17 - Excursion Vehicle - Lunar Landing Version

$$V_a = \sqrt{\frac{1.7343 \times 10^{14} (.5192)}{1580 (5280)}} \qquad V_p = \sqrt{\frac{1.7343 \times 10^{14} (1.9259)}{1580 (5280)}}$$

$$V_a = \sqrt{\frac{0.90045 \times 10^{14}}{8.3424 \times 10^6}} \qquad V_p = \sqrt{\frac{3.34009 \times 10^{14}}{8.3424 \times 10^6}}$$

$$V_a = \sqrt{0.107936 \times 10^8} \qquad V_p = \sqrt{0.400375 \times 10^8}$$

$$V_a = 0.3285 \times 10^4 \qquad V_p = 0.6327 \times 10^4$$

$$V_a = 3285 \text{ fps} \qquad V_p = 6327 \text{ fps}$$

$\triangle V$ to land on moon

$$\triangle V = V_p + (V_c - V_a) + \sum dV$$

where $\sum dV$ is the velocity required for hovering and landing

$$\sum dV = \sum(\triangle V_h + \triangle V_l) = \int_o^{t_h} g_L dt + \sqrt{2as}$$

$$\sum(\triangle V_h + \triangle V_l) = at + \sqrt{2as}.$$

For $t_{hovering} = 20$ sec.

and $s_{landing} = 1$ mile,

Then:

$$\sum dV = \frac{32.22}{6} (20) + \sqrt{2 \left(\frac{32.22}{6}\right) (5,280)}$$

$$\sum dV = 104 + 230$$

$$\sum dV = 334$$

$$\triangle V = 6327 + (3974 - 3285) + 334$$

$$\triangle V = 7350 \text{ ft./sec.}$$

ΔV to launch from moon

$$\Delta V = V_p + (V_c - V_a) + \sum \Delta V_r$$

where $\sum \Delta V_r$ is the velocity for rendezvous

$$\Delta V = 6327 + (3974 - 3285) + 60$$

$$\Delta V = 7076 \text{ ft./sec.}$$

ΔV to land and launch

$$\text{Total } \Delta V = \Delta V_{land} + \Delta V_{launch}$$

$$\Delta V = 7350 + 7076$$

$$\Delta V = 14{,}426 \text{ ft./sec.}$$

A few additional comments on the lunar landing velocity (and fuel) requirements are in order. The calculated velocity required for these maneuvers is not particularly critical since the vehicle, in most cases, has a sufficient extra fuel capacity to supply a considerable reserve incremental velocity (Δ V). This extra fuel is practically a negligible percentage of the nominal thousands of pounds required to accomplish the mission. Proof of this is shown in the following example where the weight of fuel required is derived in terms the additional reserve velocity desired:

$$I_{sp} = \frac{\text{lbs. of thrust obtained}}{\text{lbs. of fuel used/sec.}} = \frac{Ft}{W_{fuel}}$$

or: $$W_f = \frac{Ft}{I_{sp}}$$

where: $t = \frac{v}{a}$ and $F = W_{total}$ x g's of acceleration

$$F = W_t \frac{a}{g}$$

77-10(1-53)M

When the variation in vehicle weight is sufficiently small percentage wise that it, W_t, can be considered essentially a constant, then by substituting we have:

$$W_f = \frac{\frac{W_t\, a}{g}\left(\frac{v}{a}\right)}{I_{sp}} = \frac{W_t\, v}{g I_{sp}}$$

To provide an additional 50 ft./sec. reserve for the 9000 lb. vehicle gives:

$$W_f = \frac{9000\ (50)}{32.22\ (300)} = 46.6 \text{ lbs.}$$

and for a 100 ft./sec. velocity reserve, the additional fuel requirement would only be 93.2 lbs. Thus each foot/second of velocity required or deemed necessary for safety's sake would require less than one pound of fuel.

Having calculated the velocity for lunar landing (7,350 ft./sec.) and for a landing and takeoff (14,426 ft./sec.), the basic fuel requirements for each mission can now be determined. For the one way (lunar landing) mission using the unmanned lunar cache vehicle, the requirements are:

Fuel required for landing only

Payload (lunar cache)	=	6,000 lbs.
Agena B	=	2,500 lbs.
Attaching (Support) Structure	=	500 lbs.
Outrigger Landing Gear	=	500 lbs.
W_e	=	9,500 lbs.
Fuel	=	13,500 lbs.
W_o	=	23,000 lbs.

$\Delta V = 7350$ ft./sec. and $\Delta V = I_{sp} g \log \frac{W_o}{W_e}$

$$7350 = (300)(32.22) \log \frac{W_o}{9500}$$

$$W_o = 9{,}500 \; e^{\frac{7{,}350}{9{,}666}} = 9{,}500 \; e^{0.7604}$$

$$W_o = 9{,}500 \; (2.133)$$

$$W_o = 20{,}265 \text{ lbs.}$$

$$\text{fuel} = W_o - W_e = 20{,}265 - 9{,}500$$

$$\text{fuel} = 10{,}765 \text{ lbs.}$$

This unmanned supply cache requires only the Agena B since the fuel required is less than the total Agena fuel tank capacity of 13,500 lbs.

For the round trip, manned mission using the modified excursion vehicle, the calculations to determine fuel requirements with a Δ V of 14,426 ft./sec. are:

Vehicle Weight	
Gemini	6,000 lbs.
Agena B	2,500 lbs.
Support Structure	500 lbs.
Outrigger Landing Gear	500 lbs.
W_e Agena Stage	9,500 lbs.
Agena Fuel	13,500 lbs.
W_o Agena Stage	23,000 lbs.
Two Aux Tanks	1,600 lbs.
W_e Aux Tank Stage	24,600 lbs.

77-10(1-53)M

The Agena stage Δ V, with 13,500 lbs. of propellant and lunar outrigger landing gear, is 8,100 ft./sec. The additional fuel required to get 6,326 ft./sec. (14,426 - 8,100) from the other stage is:

$$W_o = W_e \; e^{\frac{\Delta V}{I_{sp} g}} = W_e \; e^{\frac{6,326}{9,666}}$$

$$W_o = 24,600 \; e^{0.6545} = 24,600 \; (1.923)$$

$$W_o = 47,305 \text{ lbs.}$$

Fuel in aux tanks = $W_o - W_e$ = 47,305 - 24,600

$$W_f = 22,705 \text{ lbs.}$$

Total fuel = Fuel in Agena + fuel in aux tanks

$$W_{tf} = 13,500 + 22,705$$

$$W_{tf} = 36,205 \text{ lbs.}$$

The initial operational flights from the lunar space station in a 1,000 mile orbit to the moon, after the complete space excursion system is declared operational, are depicted in Figure 18. The first and second flights to the moon's surface would be unmanned lunar caches containing such things as shelters, survival gear, scientific equipment, a small amount of spare parts and tools for maintenance and repair of the manned excursion vehicle, communication gear, power supply equipment, and similar items. These flights would be beneficial in two important aspects: (1) they would prove the feasibility of soft landings with the Agena propulsion and establish the operational techniques of landing, and (2) caches (described above) would be prepositioned within a one (1) mile landing area for use by the manned crews on future flights. The next or third flight would involve a space excursion vehicle fully configured for a manned,

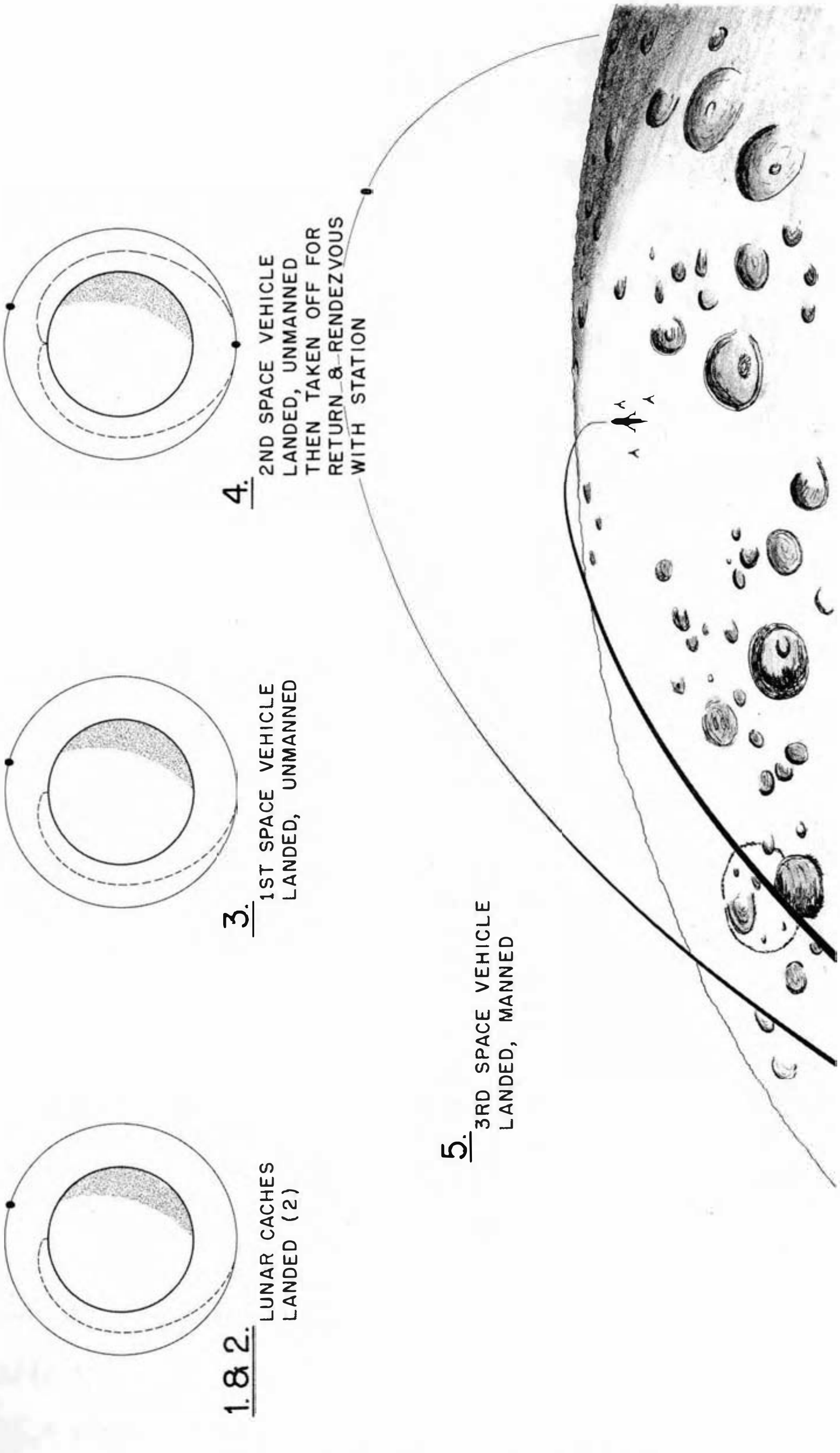


Figure 18 - Lunar Space Station Operations

round trip flight but would, in fact, be landed unmanned by remote control as were the lunar caches. This unmanned vehicle would be left on the lunar surface as an emergency rescue vehicle capable of a lunar launch if the later manned landing vehicle was damaged or in any way unusable. The fourth flight would be another unmanned excursion vehicle which will complete a lunar landing and launch back to the lunar space station. At the station it will be refueled and held in stand-by condition as an alternate rescue vehicle which could be remotely landed if needed. These four flights would provide the operational experience necessary to attempt a manned landing on the next flight with a good probability of success. Thus, the fifth flight would be a manned landing of a two man crew on the Moon's surface. After performing their assigned mission on the moon, the crew would then launch back into lunar orbit and return to the lunar space station. Because of the accumulated operational experience gained to this point by the numerous space flights and the logistic pipe line established to support these operations, a sustained program of lunar landings and exploration could be scheduled, after the first manned landing, which could lead to the establishment of a lunar base. This analysis will be restricted to the logistics and flight schedule required to accomplish the first manned lunar landing.

3. SPACE FLIGHTS AND LOGISTICS SCHEDULE

Recognizing that the pure logistics involved in this excursion concept represents a large operation, the study analyzed various flight schedules that gave a reasonable probability of mission success while keeping logistic requirements within feasible limits. For each flight schedule considered, a logistic support schedule was developed which included space fuel requirements, resupply of space stations, crew rotation, and space station and earth launch rates. A minimum (austere) and a maximum (grandiose) type schedule was compiled and trade-off studies made of the various flight schedules to determine a suitable schedule that was acceptable within the parameters of logistic capabilities, facility requirements, human factors, cost, physical limitations, and the overall time period involved in accomplishing the mission objectives (the so-called "space race").

Table 8 lists the velocity and fuel requirements, by flight number, for the selected or optimized space flight schedule from the earth orbiting space stations toward or to the moon. At first, six (6) short excursionary orbit and re-rendezvous flights below the Van Allen belt are made followed by two (2) flights into the lower fringes and then three (3) flights through the radiation belt. As experience is gained, by this slow step by step flight extension, more ambitious flights are made. The next three (3) flights repeat the previous flight profiles but add the dimension of transferring into the lunar intercept plane through the launch windows previously discussed in Section III. After this cislunar flight experience is successfully worked out, circumlunar and then lunar orbit flights are made. To provide suitable and sufficient launch windows for the lunar operations, the analysis determined that two (2) earth orbital launch platforms were required; each having the same orbital inclination (with respect to the earth's equator) and altitude but specifically different inclinations to the ecliptic in order to maintain a fixed, pre-selected phase relationship between their orbits as their orbital planes precess about the earth.

Having established the number of space stations and flights required for this

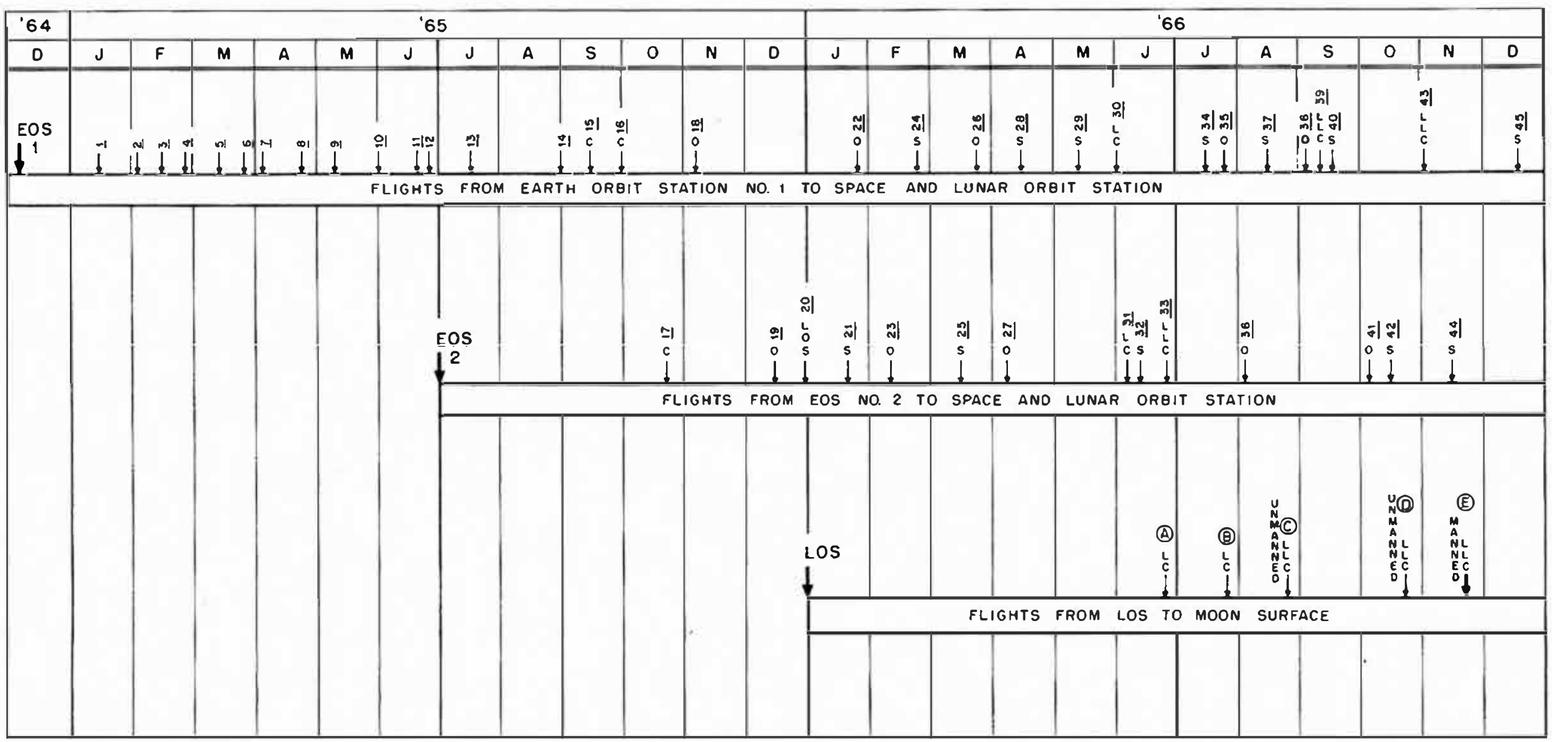
Table 8 - Velocity and Fuel Requirements for Space Excursion Program

Flight	Distance, S, From Orbit (Miles)	ΔV To Leave Earth Orbit	ΔV For Lunar Plane	ΔV For Lunar Orbit+	ΔV For Lunar Deorbit	ΔV Out of Lunar Plane	ΔV To Enter Earth Orbit	ΔV For Rendez-vous■	Total ΔV (FPS)	Time (Hours)	Fuel Required (Lbs.)	Remarks
1	15	18					18	500	536	1.58	460	
2	30	53					53	500	606	1.59	550	
3	100	143					143	600	886	1.63	805	
4	200	283					283	600	1,166	1.68	1,060	
5	250	383					383	600	1,366	1.71	1,310	
6	250	383					383	600	1,366	1.71	1,310	
											5,495	(Sub-Total)
Following Flights Enter Van Allen Belt												
7	330	463					463	700	1,626	1.76	1,570	
8	350	483					483	700	1,666	1.77	1,600	
9	40,000	8,773					8,773	1,000	18,546	21.39	60,400	
10	75,000	9,413					9,413	1,000	19,826	48.26	64,257	
11	200,000	10,013					10,013	1,000	21,026	189.07	93,023	
12	40,000	8,773	1,200			1,200	8,773	1,000	20,946	21.39	74,098	Lunar Plane
13	150,000	9,883	1,200			1,200	9,883	1,000	23,166	125.86	96,051	Lunar Plane
14	200,000	10,013	1,200			1,200	10,013	1,000	23,426	189.07	98,322	Lunar Plane
Following Flights All in Lunar Plane												
15	240,000	10,060	1,200			1,200	10,060	1,000	23,520	174.00	99,836	Circumlunar
16	240,000	10,060	1,200			1,200	10,060	1,000	23,520	174.00	99,836	Circumlunar
17*	240,000	10,060	1,200			1,200	10,060	1,000	23,520	174.00	99,836	Circumlunar
18	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	Lunar Orbit
19*	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	Lunar Orbit
20*	240,000	10,060	1,200	2,788					14,048	87	33,300	LOS (One Way)
21*	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies (One Way)
22	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	Initial LOS Crew
23*	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	LOS Crew
24	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
25*	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
26	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	LOS Crew
27*	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	LOS Crew
28	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
29	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
30	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Lunar Cache
31*	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Lunar Cache
32*	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
33*	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Lunar Landing Capsule
34	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
35	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	LOS Crew Rotation
36*	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	Crew Rotation
37	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
38	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	Crew Rotation
39	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Lunar Landing Capsule
40	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
41*	240,000	10,060	1,200	2,788	2,538	1,200	10,060	1,000	28,846	174 +	183,500	Crew Rotation
42*	240,000	10,060	1,200	2,788				155	14,203	87 +	31,300	Supplies
43	240,000	10,060	1,200	2,788				155	14,203	174 +	31,300	Lunar Landing Capsule
44*	240,000	10,060	1,200	2,788				155	14,203	174 +	31,300	Supplies
45	240,000	10,060	1,200	2,788				155	14,203	174 +	31,300	Supplies

+ ΔV for lunar orbit includes 250 ft./sec. for lunar space station rendezvous.
* Vehicles launched from EOS #2, all others from EOS #1.
■ ΔV of 500 to 1,000 ft./sec. is for earth space station rendezvous and ΔV of 155 is additional lunar station rendezvous capability for unmanned vehicles.

operation, a flight schedule for the space excursions was developed as shown in Figure 19. This schedule shows the flights to space and eventually to the lunar space station from the first earth space station (EOS #1), the launching and flights from the second earth station (EOS #2), and the flights from the lunar orbiting station (LOS) to the Moon's surface after activation of this third space station in lunar orbit. The number of each flight in Figure 19 corresponds to the same flight number listed in Table 8. It should be noted that the total space flights listed in Figure 19 are incomplete in the time period following the launching of the lunar orbital space station in that flights from the earth station to the lunar station containing the storable fuel for the lunar landing operations are not shown. Instead these "fuel only" flights which are not numbered will be listed by launch date in the logistic calculations.

With the flight schedule, fuel requirements, crew rotation, and supply requirements for the space stations and space excursion vehicles established, it is possible to calculate the logistics involved in supporting this operation. The logistics would include the launching of the three (3) space stations, the manning and crew rotation of these stations, the monthly supply flights to each station, the fuel flights delivered to each earth orbiting station, the fuel used for each space flight, the fuel shipped to the lunar station, the fuel remaining in storage at each earth space station, and the number of Titan II launches from the earth. These logistics were computed and are itemized in Table 9 by week from the start of operations. The manned moon landing is accomplished during the one hundred and fourth (104) week by which time a total of 168,000 lbs. (28 flights at 6,000 lbs. each) of fuel has been trans-



KEY:

EOS – Earth Orbiting Station

LOS – Lunar Orbiting Station

C – Circumlunar Flight

O – Lunar Orbital Flight

S – Unmanned Supply Flight

LC – Lunar Cache

LLC – Lunar Landing Capsule

Figure 19 - Space Excursion Flight Schedule

Table 9 - Logistic Schedule for Space Excursion System

Operational Week From Start	Number of Titan II Earth Launchings	Payload of Titan II's* Space Station	Crew (2 per Flt.)	Supplies 6000#/Flt.	Storable Fuel 6000#/Flt.	Space Launching From Earth Space Station Flight Number	Payload** or Vehicle	Fuel Used For Space Launch SS#1	SS#2	Fuel Balance Stored At*** SS#1	SS#2
1	-										
2	3	1 (#1)	2							0	
3	1		1							0	
4	1			1						0	
5	2		1		1					6,000 (1)	
6	2			1	1	1	SEV	460		11,540 (1)	
7	2				2					23,540 (2)	
8	2				2					35,540 (2)	
9	3		1		2	2	SEV	550		46,990 (2)	
10	3		1	1	1					52,990 (1)	
11	3		1		2	3	SEV	805		64,185 (2)	
12	3		1		2	4	SEV	1,060		75,125 (2)	
13	3				3					93,125 (3)	
14	3			1	2	5	SEV	1,310		103,815 (2)	
15	3				3					121,815 (3)	
16	4				4					145,815 (4)	
17	4				4	6	SEV	1,310		168,505 (4)	
18	4		2		2					180,505 (2)	
19	4		1	1	2	7	SEV	1,570		190,935 (2)	
20	4		1		3					208,935 (3)	
21	4				4	8	SEV	1,600		231,335 (4)	
22	4				4					255,335 (4)	
23	4			1	3	9	SEV	60,400		212,935 (3)	
24	4				4					236,935 (4)	
25	4		1		3					254,935 (3)	
26	5		1		4	10	SEV	64,257		214,678 (4)	
27	5		1	1	3					232,678 (3)	
28	5		1		4	11	SEV	93,023		163,655 (4)	
29	5				5					193,655 (5)	
30	5				5	12	SEV	74,098		149,557 (5)	
31	5				5					179,557 (5)	
32	5			1	4					203,557 (4)	
33	5				5	13	SEV	96,051		137,506 (5)	
34	5	1 (#2)	1		3					155,506 (3)	0
35	5		4		1					161,506 (1)	0
36	5		2	1	2					173,506 (2)	0
37	6				6					209,506 (6)	0
38	6		1	1	4	14	SEV	98,322		135,184 (4)	0
39	7				7					177,184 (7)	0
40	7				7					219,184 (7)	0
41	7			1	6	15	SEV	99,836		131,348 (2)	24,000 (4)
42	7				7					131,348 (0)	66,000 (7)
43	7		3	1	3					131,348 (0)	84,000 (3)
44	7		2		5	16	SEV	99,836		31,512 (0)	114,000 (5)
45	7		1	1	5					61,512 (5)	114,000 (0)
46	7		1		6	17	SEV		99,836	97,512 (6)	14,164 (0)
47	7		1	1	5					127,512 (5)	14,164 (0)
48	8				8					175,512 (8)	14,164 (0)
49	8			1	7	18	SEV	183,500		4,012 (2)	44,164 (5)
50	8				8					4,012 (0)	92,164 (8)
51	8			1	7					4,012 (0)	134,164 (7)
52	9				9					4,012 (0)	188,164 (9)
53	9			1	8					52,012 (8)	188,164 (0)
54	9				9	19	SEV		183,500	106,012 (9)	4,664 (0)
55	9			1	8					130,012 (4)	28,664 (4)
56	9	1 (LCS)			8					154,012 (4)	52,664 (4)
57	10				10	20	LSS		33,300	184,012 (5)	49,364 (5)
58	10		1	1	8					184,012 (0)	97,364 (8)
59	10		2 (1L)	1 (L)	7	21	Supplies		31,300	184,012 (0)	108,064 (7)
60	10		1	1	8	22	LSS Crew	183,500		512 (0)	156,064 (8)
61	10		3 (1L)		7					512 (0)	198,064 (7)
62	10		1	1	8	23	ISS Crew		183,500	48,512 (8)	14,564 (0)
63	10		1	1 (L)	8	24	Supplies	31,300		17,212 (0)	62,564 (8)
64	10		1	1	8					65,212 (8)	62,564 (0)
65	10				10					125,212 (10)	62,564 (0)
66	10			1	9					179,212 (9)	62,564 (0)
67	10		1 (L)	1 (L)	8	25	Supplies		31,300	227,212 (8)	31,264 (0)
68	10			1	9	26	LSS Crew	183,500		43,712 (0)	85,264 (9)
69	10		1 (L)		9					43,712 (0)	139,264 (9)
70	10				10					43,712 (0)	199,264 (10)
71	10		1	1	8	27	LSS Crew		183,500	91,712 (8)	15,764 (0)
72	10		2	1 (L)	7	28	Supplies	31,300		66,412 (1)	51,764 (6)
73	10		2	1	7					108,412 (7)	51,764 (0)
74	11		2		9					162,412 (9)	51,764 (0)
75	11		1	1	9					216,412 (9)	51,764 (0)
76	11			1 (L)	10	29	Supplies	31,300		185,112 (0)	111,764 (10)
77	11			1	10					185,112 (0)	171,764 (10)
78	11				11					245,112 (10)	177,764 (1)
79	11			1 (LC)	10 (1L)	30	Cache	31,300	31,300	213,812 (0)	200,464 (9)
						-	Fuel				
80	11			2 (1LC)	9	31	Cache		31,300	237,812 (4)	199,164 (5)
81	11			1 (L)	10 (1L)	32	Supplies	31,300	31,300	224,512 (3)	203,864 (6)
						-	Fuel				
82	11			1	10 (1L)	-	Fuel		31,300	236,512 (2)	214,564 (7)
83	11			1 (LLC)	10 (1L)	33	Land.Caps.	31,300	31,300	223,212 (3)	219,264 (6)
						-	Fuel				
84	12				12 (2L)	-	(2) Fuel	31,300	31,300	215,912 (4)	223,964 (6)
85	12			1	12 (1L)	-	Fuel		31,300	245,912 (5)	222,664 (5)
86	12		1 (L)	1 (L)	10	34	Supplies	31,300		250,612 (6)	246,664 (4)
87	12			1	11	35	LSS Crew	183,500		133,112 (11)	246,664 (0)
88	13		1 (L)		12 (1L)	-	Fuel	31,300		167,812 (11)	246,664 (0)
89	13		1	1	11	36	LSS Crew		183,500	167,812 (0)	129,164 (11)
90	13		2	1 (L)	10	37	Supplies		31,300	167,812 (0)	157,864 (10)
91	13		2 (L)	1	10 (1L)	-	Fuel	31,300		190,512 (9)	157,864 (0)
92	13		2		11 (1L)	-	Fuel		31,300	220,512 (5)	156,564 (5)
93	14		2		12	38	LSS Crew	183,500		109,012 (12)	156,564 (0)
94	14			2 (1LLC)	12	39	Land.Caps.	31,300		119,712 (7)	186,564 (5)
95	14			1 (L)	13 (1L)	40	Supplies	31,300	31,300	124,412 (6)	191,264 (6)
						-	Fuel				
96	14			1	13 (2L)	-	(2) Fuel	31,300	31,300	129,112 (6)	189,964 (5)
97	14		1 (L)		13 (2L)	-	(2) Fuel	31,300	31,300	128,112 (5)	194,664 (6)
98	14			1	13	41	LSS Crew		183,500	128,112 (0)	89,164 (13)
99	14			1 (L)	13 (3L)	42	Supplies	62,600	62,600	83,512 (3)	68,564 (7)
						-	(3) Fuel				
100	14			1	13 (3L)	-	(3) Fuel	62,600	31,300	62,912 (7)	56,264 (3)
101	14			1 (LLC)	13 (2L)	43	Land.Caps.	31,300	62,600	49,612 (3)	40,664 (8)
						-	(2) Fuel				
102	14			1	13 (3L)	-	(3) Fuel	62,600	31,300	29,012 (7)	27,364 (3)
103	14			1 (L)	13 (1L)	44	Supplies	31,300	31,300	39,712 (7)	26,064 (5)
						-	Fuel				
104	14			1	13 (1L)	-	Fuel		31,300	69,712 (5)	36,764 (7)
105	14				14					105,712 (6)	84,764 (8)
106	14		1	1	12					117,712 (2)	144,764 (10)
107	14		1	1 (L)	12	45	Supplies	31,300		140,412 (9)	162,764 (3)
108	14		2	1	11					170,412 (5)	198,764 (6)
109	14		2		12 (1L)	-	Fuel		31,300	188,412 (3)	215,464 (8)
110	14		2		12 (2L)	-	(2) Fuel	31,300	31,300	187,112 (5)	214,164 (5)

* L, LC, and LLC in parenthesis indicates payloads launched from the earth to the earth orbiting stations but whose final destination is the lunar space station and L - Lunar, LC - Lunar Cache, LLC - Lunar Landing Capsule.

** SEV - Space Excursion Vehicle
LSS - Lunar Space Station

*** Numbers in parenthesis indicate the number of fuel flights delivered to each earth space station for storage.

Page intentionally left blank

ported to the lunar space station for the manned and unmanned lunar landing flights. The fuel required for these lunar landing flights is listed below.

Two (2) unmanned lunar caches @ 10,765 lbs. each	-	21,530 lbs.
Two (2) unmanned lunar landings @ 36,205 lbs. each	-	72,410 lbs.
One (1) manned lunar landing @ 36,205 lbs.	-	36,205 lbs.
Refueling of one (1) recovered unmanned vehicle @ 36,205 lbs.	-	36,205 lbs.
TOTAL		166,350 lbs.

The largest amount of fuel stores at either earth space station is 255,335 lbs. with quantities varying widely between this maximum and zero.

Thus, a flexible fuel storage facility must be provided at each space station to allow the accumulation of Agena storable fuels (RFNA and UDMH) in the quantities listed in Table 9. Since the fuel quantities in storage vary over a large range, a variable storage tank design was considered as shown in Figure 15. This tank, in the shape of a sphere, would have a series of diaphragms to separate the oxidizer from the propellant and to keep each fuel under a positive pressure for storage as a liquid, for storage of varying quantities of fuel and for pumping operations during refueling. A storage capacity approximately 25% greater than that required (255,335 lbs.) was used in computing the diameter of the fuel storage tank.

Required storage capacity = 255,335 + 25% = 320,000 lbs.

Average density of Agena fuels (RFNA + UDMH) = 76.75 lbs./cu.ft.

Volume required = 4169.4 cu. ft.

Volume of sphere = $\frac{4}{3} \pi r^3$

$4169.4 = 4.189 \, r^3$

$r = \sqrt[3]{995.23} = 9.98$ ft.

Therefore, a spherical tank 20 ft. in diameter would supply the volume necessary for fuel storage. This tank would weigh between one and two thousand pounds and would be transported to the space station on the first supply flight for assembly and installation.

Figure 20 is a plot of the weekly launch rate of the Titan II boosters from earth launch pads as listed in Table 9. That portion of the launch rate for "fuel only" flights is plotted to show the large impact these flights have on the total logistics. The remaining logistic flights, i.e., the difference between the total weekly rate and the "fuel only" rate, consist of the supply flights and the personnel flights which transport the space crews. Obviously, the "fuel only" and supply flights do not require man-rated Titan II boosters since they are unmanned, remotely controlled flights and the total requirements for building and launching of man-rated boosters for the personnel flights are rather modest. In fact, by the time of the manned lunar landing (the 104th week) an accumulated total of 842 Titan II boosters have been expended of which only 64 are man-rated and carry personnel. These sixty-four crew launches have accomplished the original manning and eight crew rotations at earth space station #1, the manning and four crew rotations at earth space station #2, and the manning and one crew rotation at the lunar orbiting station during the two years of operations.

The existing and presently planned Titan II launch pads (three) will not be sufficient to allow the launching rate previously described and additional pads will be required for this space operation. These new launch pads (4 through 16) and the dates at which they would be required are shown in Figure 20. These calculations are based on a launch pad time (the time from arrival and

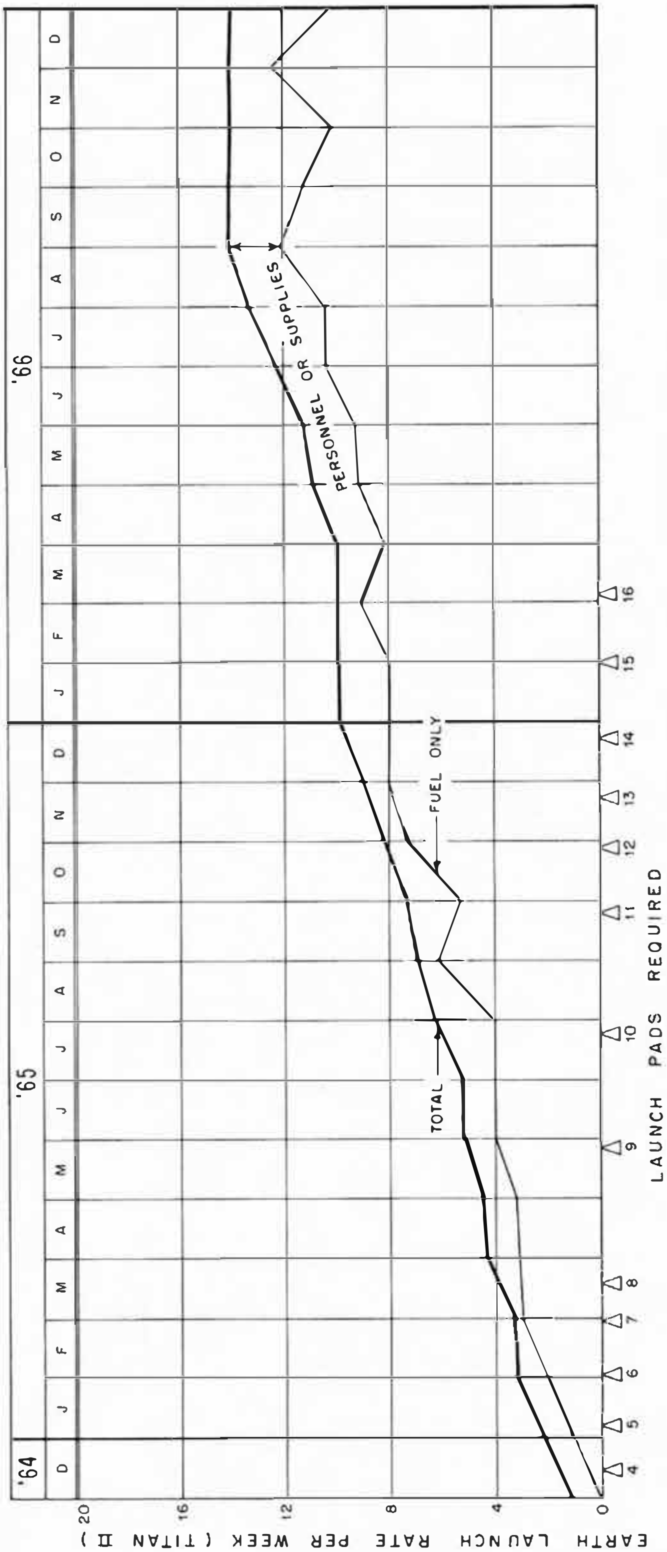


Figure 20 - Logistic Support Flight Schedule and Launch Pad Requirements

erection of the Titan II booster and its payload at the launch pad until launch) of:

(1) At the start of operations (late 1964)- 14 days

(2) After 20 weeks of operations (May 1965) - 10 days

(3) After 60 weeks (February 1966) - 7 days

(4) After 100 weeks (November 1966) - 5 days

The decrease in launch pad time for the Titan II during 1965 and 1966 is, we believe, a realistic prediction because of the simplicity deliberately designed into the Titan II as compared to the Atlas and/or Titan I, the experience and operational know-how that will be gained in military and Gemini launches with this missile in the next two years (1963 and 64), and the additional experience that would be obtained by launching 800+ boosters if the Space Excursion System were implemented.

After determining the logistic requirements, the space flight schedule, and the operational methodology as outlined in the preceeding pages, it is now possible to quantitize the various vehicles, equipment, and facilities required to implement and support the Space Excursion System. Table 10 lists these requirements which are discussed in the following pages.

The total manned Gemini flights as previously calculated total 64. Based on Mercury capsule experience, it is assumed that each Gemini capsule can be refurbished after reentry at least one time and thus a minimum of two flights from earth to space and return is used in the system requirements analysis. On this basis 32 Gemini capsules are required to perform the 64 manned earth space launches. Three additional Gemini capsules are required for the manned lunar landing vehicles which completes the total of 35 (32 + 3) listed in Table 10.

Table 10 - Vehicle, Equipment and Facility System Requirements

Quantity	Item
35	Gemini Capsule
3	Space Station
842	Titan II Booster
210	25,000# Capacity Auxiliary Fuel Tank
77	Adapter Mounting Structure
6	Outrigger Landing Gear
50	Agena Booster
27	Modified Booster (Throttleable)
2	Lunar Caches
3	Fuel Storage System for Space Station
51	Supply Carrier Module (6,000# Capacity/Module)
153 tons	Supplies (Food, Oxygen, Fuel, Equipment, etc.)
720	Fuel Carrier Module (6,000# Capacity/Module)
2,160 tons	Fuel for Space Operations (RFNA and UDMH)
3	Modified Existing Titan II Launch Pad
13	New Titan II Launch Pad

If the Gemini can be used more than two times, even less capsules would be required. This lesser requirement depending on reuse capability will be true for many of the other system requirements that will be discussed. Since, in most cases, the amount of possible reuse is unknown, the conservative or higher quantity will be listed. The three space station requirement is for two earth orbiting space stations and one lunar space station. The 842 Titan II booster requirement has been calculated and described previously in Table 9. The 210 auxiliary fuel tanks are for the space flights (72) from the earth space stations including manned excursions (19), lunar crew flights (8), lunar supply flights (17) and lunar fuel flights (28) listed in Table 9 plus the five lunar landing flights. The first eight manned excursion flights do not require auxiliary fuel tanks since their total fuel requirements are less than the Agena booster fuel capacity (13,500 lbs.) but all other space flights do require from two to eight tanks per flight. The 77 adapter sections are based on the total space flights of 77. This assumes that each adapter is used only once which is true of at least 50 flights consisting of the 17 supply, 28 fuel, and 5 lunar landing flights. The other 27 flights which are the manned space excursion and manned lunar crew flights may require as few as five adapter sections if they are continuously reused. The same considerations are applicable to the Agena booster requirement of 77 (50 regular and 27 modified) and any reuse would lower the total required. The six outrigger landing gears are for the five lunar landings with a spare gear for the recovered unmanned vehicle (flight D of Figure 19) in case of damage to the original outrigger landing gear. The two lunar caches are for flights A and B of Figure 19. The three fuel storage systems are for the three orbiting space stations and the 720 fuel carrier modules are required to supply the fuel for these storage systems as listed in Table 9. The 51 supply carrier modules are for the supply flights (also listed in Table 9). The three modified and 13 new Titan II launch pads have been discussed previously and are shown in Figure 20.

The results of this operational analysis presents a promising concept for an operational, logistically-oriented space system that depends on present state-of-the-art, existing boosters, gradual evolutionary growth of the system and its operational techniques by building steadily on past experience, and a task-force or expeditionary type operations. In addition, this concept utilizes those things which the U.S.A. does best - mass production, continuous and repeated transportation operations, characteristic mechanical aptitude and ingenuity of the average American as demonstrated by the GI's in World War II, and the ability to efficiently organize and implement large operations. It should be noted that none of the operations described in this analysis would be possible without the space capabilities becoming available as a result of perceptive planning and resolute implementation of space programs by NASA Headquarters. These programs are providing fundamental capabilities in being when needed which are absolutely essential to the Space Excursion Concept. Outstanding examples are the rendezvous decision of 1961, the Gemini capsule program, the Mercury flights, the man-in-space communication and tracking net, etc. The same is equally true of the Defense Department programs in the area of boosters (especially the Titan II and Agena), bio-astronautics, space medicine, etc. This foundation has established the opportunity for considering the early implementation of a program operating in the fashion described in this analysis.

Page intentionally left blank

SECTION V - LEAD TIME REQUIREMENTS

An analysis was made of the lead times that would be required in each area to permit availability of the necessary hardware at the times it would be needed in conformance with the operational schedules of Section IV (Figures 19 and 20). The most critical (longest lead time) items were found to be the provision for throttling of a 15,000 lb. thrust Agena engine, implementation of the space station, provision for additional Gemini capsules by the end of 1964, and for additional production of Titan boosters by the same time; although several other items also require substantial lead times. The overall results are indicated by Figure 21.

The two major items from the lead time standpoint are the Titan II boosters for the logistic support and the construction and launching of the first space station. The first item (boosters) is essentially a production scheduling and facility problem, and the second (space station) is essentially a development problem.

In order to determine the lead time requirements for the Titan II boosters needed to support the program described, it was first necessary to establish production rates and reasonable rates of build-up. For this purpose, it was estimated that a single production facility with multiple lines could reasonably produce completed units at the rate of 1 every one or two days, (or around 15 to 20 per month), after a substantial number had been built and maximum rate attained. Based on this, a maximum eventual rate of 18 per month was arbitrarily assumed to be the upper limit.

Another major consideration affecting Titan II missile availability was the need to meet military and other planned space program requirements first. Since such requirements were assumed for purpose of this analysis to have priority over this program, it was assumed that these other requirements might come to a total of

Page intentionally left blank

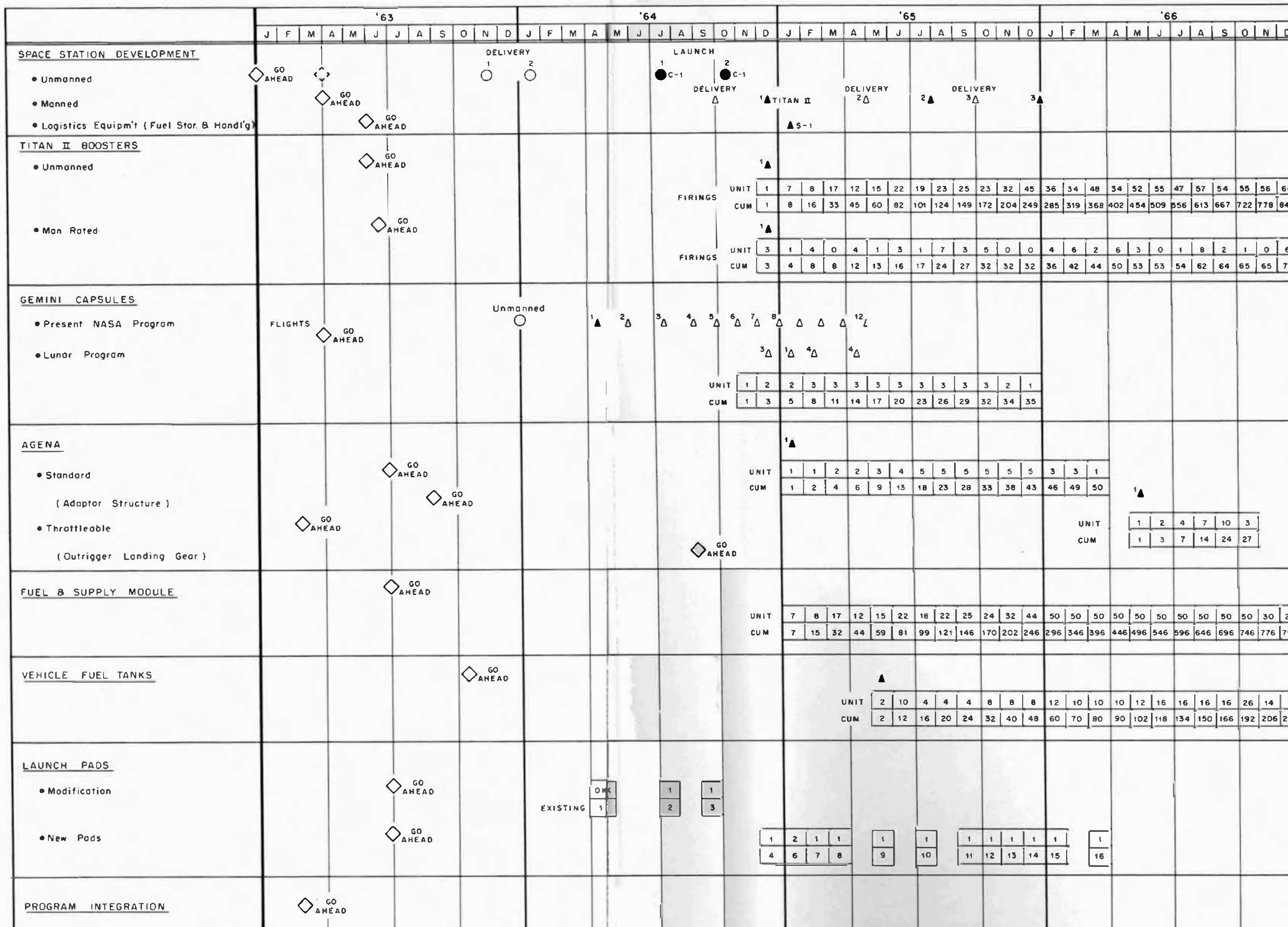


Figure 21 - Space Excursion System Lead Time Requirements

Page intentionally left blank

around 250 Titan II's. This is considerably in excess of any publicly announced plans; however, to be conservative and assure validity of the analysis, this number was assumed.

Based on these assumptions, and utilizing a normal steady build up to maximum rate for production of the 250 missiles to be delivered ahead of this program's needs, (and at a maximum rate of 15 per month), the total unit production delivery schedule shown in Figure 22 was derived. It was necessary to bring in a second source (in Fiscal '65), as indicated, to meet the total needs of this program. The resulting cumulative delivery schedule is shown by Figure 23 along with the corresponding cumulative flight requirements of this program conforming to the operational schedule of Figure 19. These are listed in Table 11.

It can be noted from Figure 23 that the resulting lead time of deliveries vs launch requirements provided at the launch site starts out at about 5 months and reduces gradually to about 2 months by about the 50th flight. After this the lead time remains at 8 to 12 weeks for the balance of the program. This is desirable in order to prevent an excessive launch site storage problem.

The overall lead time for booster production was then determined by considering the initial go-ahead requirement for the most critical of the following requirements: (1) Adjustment of present source (Martin-Denver) production plans for present facility to encompass these requirements, and authorization for same; (2) Incorporation of any changes resulting from specific requirements of this program; and (3) Establishment of a second source (probably, although not necessarily, Martin-Baltimore - which is already being established as the source of supply for man-rated Titan II's for the Gemini) to begin deliveries as indicated

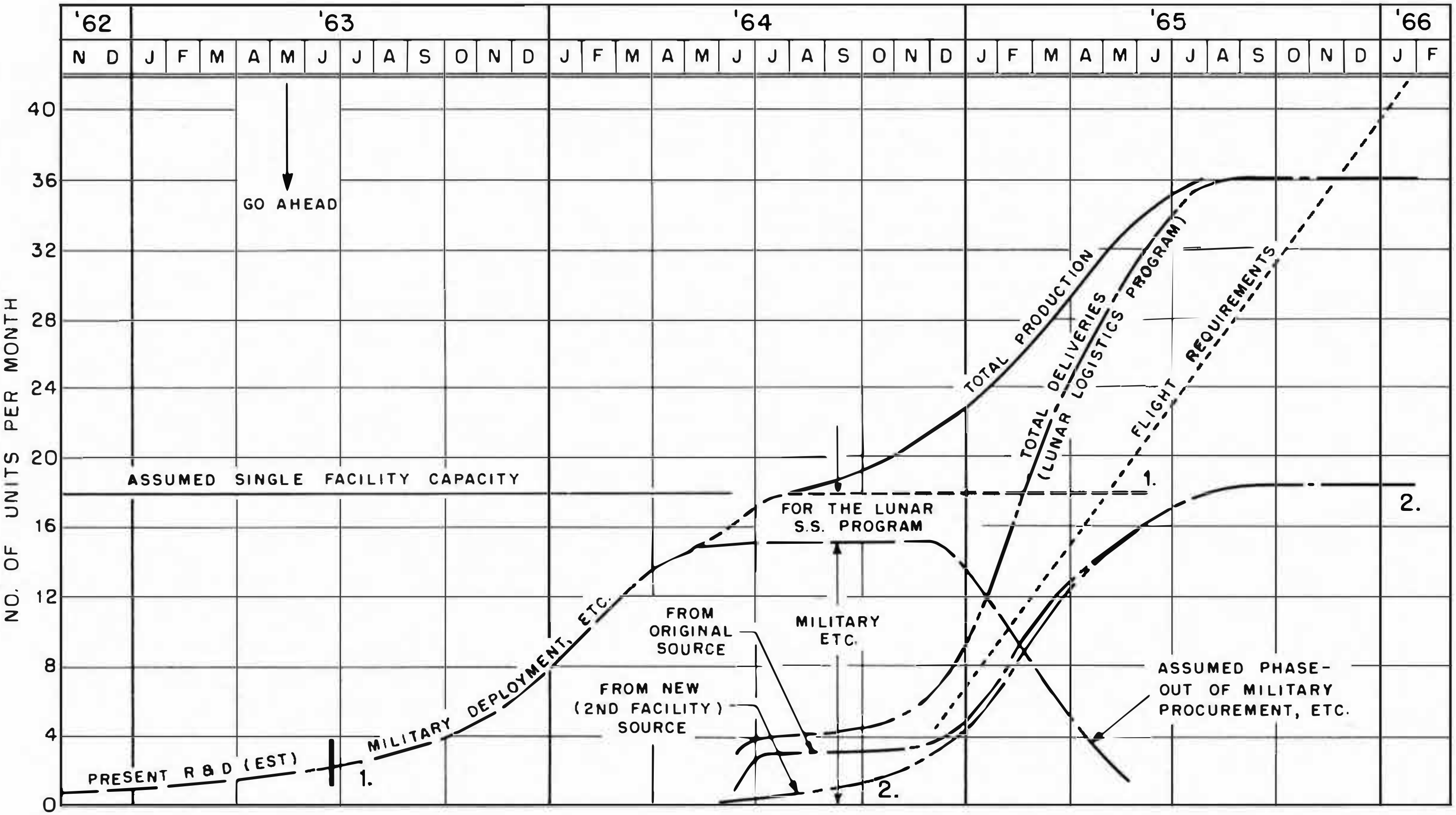


Figure 22 - Titan II Unit Production Delivery Schedule

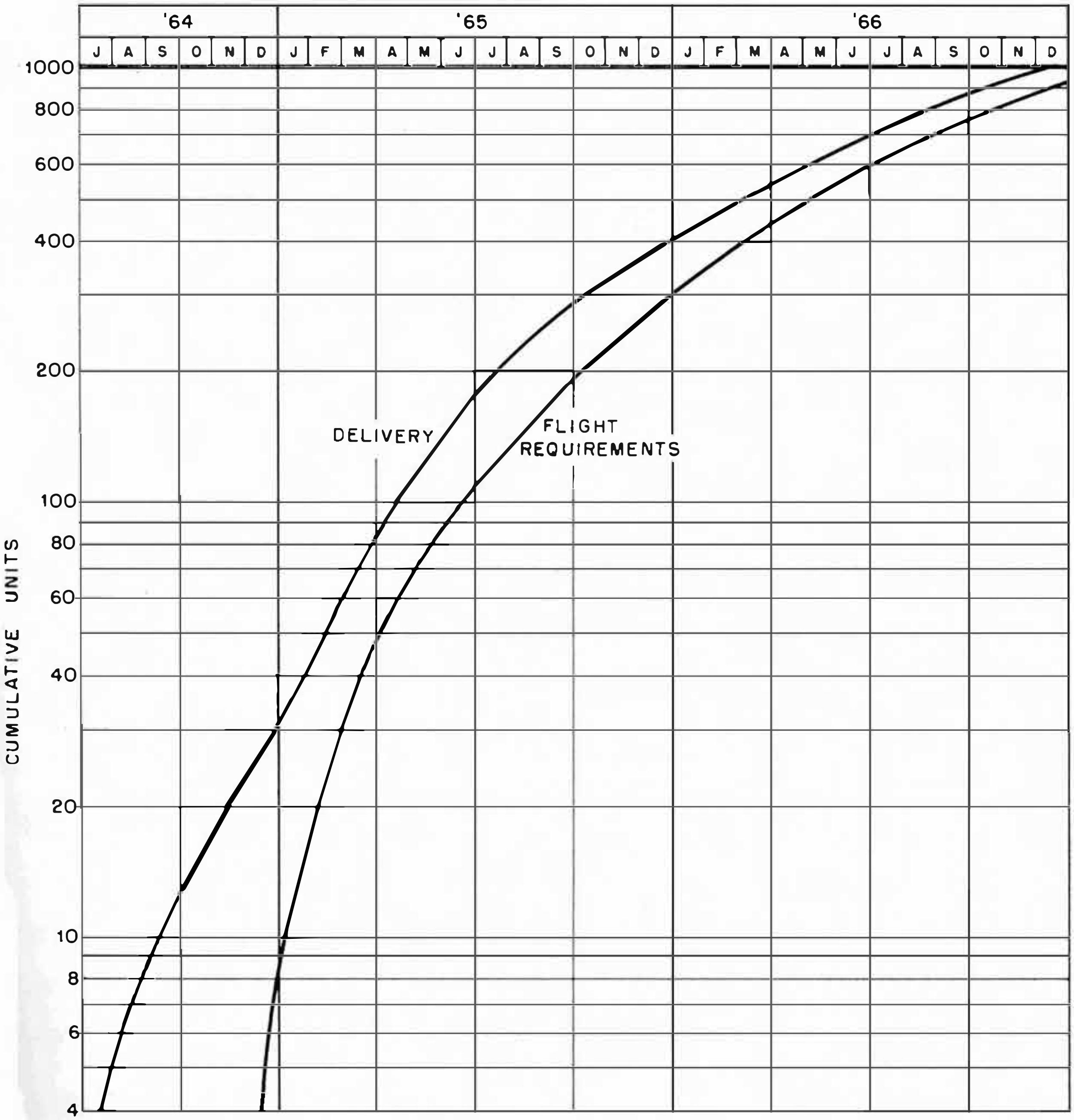


Figure 23 - Titan II Cumulative Delivery Schedule
and Flight Requirements

Table 11 - Estimated Titan II Booster Production Requirements

Date		Deliveries						
		R&D, Military and Space Programs		For Proposed Space Excursion Vehicle System			Resulting Total Titan II Requirements	
Year	Month	Unit	Cumulative	Unit (From Original Titan II Production Source)	Unit (From Second Titan II Production Source)	Cumulative	Unit	Cumulative
1961-1962	Thru October	R&D Vehicles	19					19
1962	November	1	20				1	20
	December	1	21				1	21
1963	January	1	22				1	22
	February	1	23				1	23
	March	1	24				1	24
	April	2	26				2	26
	May	2	28				2	28
	June	2	30				2	30
	July	2	32				2	32
	August	2	34				2	34
	September	3	37				3	37
	October	4	41				4	41
	November	5	46				5	46
	December	6	52				6	52
1964	January	8	60				8	60
	February	10	70				10	70
	March	12	82				12	82
	April	14	96				14	96
	May	15	111				15	111
	June	15	126				15	126
	July	15	141	3	1	4	19	145
	August	15	156	3	0	7	18	163
	September	15	171	3	1	11	19	182
	October	15	186	3	1	15	19	201
	November	15	201	3	2	20	20	221
	December	15	216	3	3	26	21	242
1965	January	13	229	5	5	36	23	265
	February	10	239	8	7	51	25	290
	March	7	246	11	10	72	28	318
	April	5	251	13	13	98	31	349
	May	4	255	14	14	126	32	381
	June	2	257	16	15	157	33	414
	July			18	18	193	36	450
	August			18	18	229	36	486
	September			18	18	265	36	522
	October			18	18	301	36	558
	November			19	18	338	37	595
	December			20	18	376	38	633
1966	January			20	19	415	39	672
	February			20	20	455	40	712
	March			20	20	495	40	752
	April			20	20	535	40	792
	May			20	20	575	40	832
	June			20	20	615	40	872
	July			20	20	655	40	912
	August			20	20	695	40	952
	September			20	20	735	40	992
	October			20	20	775	40	1032
	November			20	20	815	40	1072
	December			20	20	855	40	1112
1967	January			20	20	895	40	1152
1967	(Annual)			240	240	1335	480	1592*
1968	(Annual)			240	240	1815	480	2072*

*Future Production Potential

on Figure 22 and Table 11. The resulting initial go-ahead lead time requirement is indicated on Figure 21, Item 2, along with the unit and cumulative launching schedule for both the unmanned (Item 2a) and man-rated (Item 2b) booster firings. (The indicated go-ahead requirement is June 1963.)

The determination of lead time requirements for the space station development was based on detailed development planning schedules developed at Goodyear Aircraft Corporation, involving a number of development test launchings, some of which are shown as key milestones or program bench marks on Figure 21, Item 1a. If the earliest of the development test flights shown are to be made piggy-back on C-1's, as indicated; then the January 1 go-ahead shown is required. If they are to be made on Titan I's or II's, or some other present launch vehicle (e.g., Atlas-Agena) of 4,000 lb. or greater load capacity; then the second (April 1) go-ahead indicated is required. Much of the difference is due to differing transportation times of the boosters between factory and launch site.

For estimating the required Gemini capsule lead times, the utilization and availability concept is clearly shown on the chart of Figure 21 as Item 3. As can be seen, the concept is one of phasing in with the currently planned Gemini program after it is well advanced beyond the stage where the Mercury program is to be terminated - that is, after at least a half dozen or more manned Gemini flights are completed. This phase-in of the early flights for this program with the latter part of the planned Gemini flight program flights is represented by Items 3b and 3a, respectively, of Figure 21, with the first Gemini flight for this program occurring at the end of 1964. The corresponding Gemini capsule delivery schedule adjustment is estimated to require a 20 month lead time, for a go-ahead no later than April 1963, as indicated for Item 3b.

The lead times for the Agena B and associated equipment making up the balance of the excursion vehicle and landing vehicle do not appear to be nearly as critical, as shown on Figure 21 under Item 4, with one exception: the throttling provision for the engine (listed as Item 4b). A considerable study of this, including checks with the engine manufacturers, have indicated that 36 to 42 months lead time may be required. Taking the 42 months, for conservatism, means a go-ahead by March or April of 1963 is required. This therefore might be the second most critical lead time item on the program.

Items 5 and 6 of Figure 21 are relatively straightforward development and fabrication items which can use equipment required earlier on this or other programs; therefore the lead times are relatively non-critical, but probably still fairly rigorous, with estimated required go-ahead dates as indicated on the chart (Items 5 and 6 of Figure 21).

The launch pad lead time requirements are fairly well-established and straightforward, but also fairly rigorous, and are indicated both for new and modified pads under Item 7 on the chart of Figure 21.

Item 8, the Program Integration and Associated Requirements, is probably the most critical item of all, and as indicated, probably must start (at least for initial coordination and direction of early items) by March 1, 1963, to permit carrying out the indicated program as described by this operational analysis if the key milestones are to be met on the dates indicated.

If go-ahead authorizations cannot be achieved by the dates indicated as a result of this lead-time analysis, the results are still valid and useful - for it can be used as an elapsed time requirement indicator. For example, if any of the indicated key starting dates or critical bench marks are slipped by a certain maximum time (for example 2 months), then the estimated dates for achievement of key program goals on the operational schedule (Figure 19) can be adjusted by adding the same amount of time, and assuming an equal slippage for the entire schedule. However, if very large initial slippages are involved, this might be unfavorably modified by two effects:

(1) The present Titan II production might by then be starting phase-out operations which are not entirely reversible, therefore would probably involve some additional slippage. (The same could happen with the Gemini Capsules, although probably to a lesser extent.)

(2) A slippage of much more than a year might bring the later phases of the program into a significantly increasing portion of the solar radiation cycle, with the strong possibility of further consequent slowing of operations.

At any rate, if the maintenance of the operational schedule and accomplishment of program end goals are to be possible by the times given in Figure 19, at least partial authorization in most critical areas by early spring of 1963 appears mandatory. By early Summer of 1963 all items become critical to schedule milestones except a few, and by the Fall of 1963 (October 20 for Item g) the remainder becomes critical. (The only significant exception is the lunar landing gear outrigger, under Item 6b, which requires go-ahead by the latter part of 1964, as shown.) Slippage beyond these times will in most cases cause at least a corresponding slippage in program accomplishments, and very long delays in starting might cause even longer end date slippages.

It is interesting to note, however, that this analysis shows that significant commitments can be entered into very gradually. The go-ahead requirements for the various items are themselves staggered somewhat, and beyond this the items involving most substantial expenditures do not immediately require large irreversible commitments. For example, taking one of the largest items, the Titan boosters (Item 2a), which represents nearly half the total cost, we see that while go-ahead authorization and planning is required by shortly before mid-1963 only one tenth of the boosters required are to be delivered by mid-1965.

The overall significance of this, which can be seen by similar examination of each item on the chart of Figure 21, is that while early action is required to hold to the program objectives and dates as outlined, the rate of commitment is much more gradual; and the rate of expenditure (even counting cancellation costs that could be incurred by shifting direction at any point) is very gradual, indeed. In fact, the first billion dollars of such expenditure for this program would not be reached until sometime in 1965, with a very substantial return realized toward achievement of Apollo goals for the dollars spent to that point, regardless of the course subsequently followed. Initial commitments required at the start of the program would not exceed 2 or 3 million dollars (although larger commitments would be required soon thereafter). This situation exists to a large extent because of utilization of the products and outputs of past and present programs already going on or accomplished. Thus the lead time analysis described here together with the operational program description, all of which are largely represented by Figures 19 and 21, permits us to visualize and assess the true nature of the opportunity presented, and the associated responsibility for considering the significance of its implications.

SECTION VI - ESTIMATED TOTAL PROGRAM COSTS

The various items making up the total requirements to carry out the program described and analyzed are listed (based on Table 10 of Section IV), along with their estimated costs, in Table 12. The basis for these estimates is presented here.

By far the largest item is the booster vehicles (Titan II or equivalent) needed to ferry men and materials (fuel, supplies, and equipment) to the earth orbiting space stations. This constitutes the main leg of the logistic pipe line required for this program operation, and is estimated in terms of total launch costs per vehicle.

The unit costs of launching Titan II vehicles, exclusive of payload costs and launch pad costs are given by the progress (learning) curve shown in Figure 24 for 1 to 1,000 units. The slope used, which represents the averaged effect of those for vehicle production, servicing and checkout, and launch (including range costs), is an 83% slope. It should be noted that the total Titan curve extends back from the one shown to cover the first model or version of the Titan, now designated as Titan I, which had a much higher A-value than that shown for the first unit in this Titan II curve (of $12.2 million). However, even though a large number of Titan I's were built and launched or deployed, which gives some guide to the characteristic slope for the curve, the Titan II represents a sufficiently changed missile that a substantial step-up from the Titan I curve had to be taken for the first Titan II unit, and this constituted the A-value of this curve. (This curve of the chart relates exclusively to Titan II in its entirety).

Table 12 - Total Estimated Cost of Operations

Procurement Costs*

Reference Number	Number Required	Item	Nominal Estimated Cost
1	35	Gemini	$ 280,000,000
2	3	Space Stations (Plus Prototype - Spare)	24,000,000
3	842	Titan II Launches (Plus Abort Attrition Allowance)	2,007,000,000
4	210	Auxiliary Fuel Tanks (25,000 lb. Capacity)	25,200,000
5	77	Mounting Structures (Adapters)	3,080,000
6	6	Outrigger Landing Gear	600,000
7	45	Agenas	90,000,000
8	15	Agenas (Modified, Throttleable)	32,000,000
9	2	Lunar Caches (Plus 1 Prototype - Spare)	19,800,000
10	3	Fuel Storage Systems (For Space Stations) (Plus 1 Prototype - Spare)	800,000
11	51	Supply Carrier Modules	12,750,000
12	720	Fuel Carrier Modules	162,000,000
13	153	Tons of Supplies (Food, Oxygen, Fuel, Equipment, Etc.)	7,370,000
14	2,160	Tons of Fuel (For Space Operations from Space Stations)	4,320,000
15	3	Modified Titan Pads	2,400,000
16	13	New Titan II Launch Pads	39,000,000
		SUB TOTAL	$2,710,320,000

Development, Engineering, and Other Costs

Reference Number		Item	Nominal Estimated Cost
17		Development Costs**	$ 109,450,000
18		Salaries (For 1000 Operations and Service Personnel)	50,000,000
19		Program Management, Administration, Coordination and Miscellaneous Expenses	30,000,000
20		Program Contingencies (5%)	150,000,000
		SUB TOTAL	$ 339,450,000
		TOTAL PROGRAM COSTS	$3,049,770,000

* For individual item cost basis, see accompanying text and Table 14.
** See Table 13 and accompanying text.

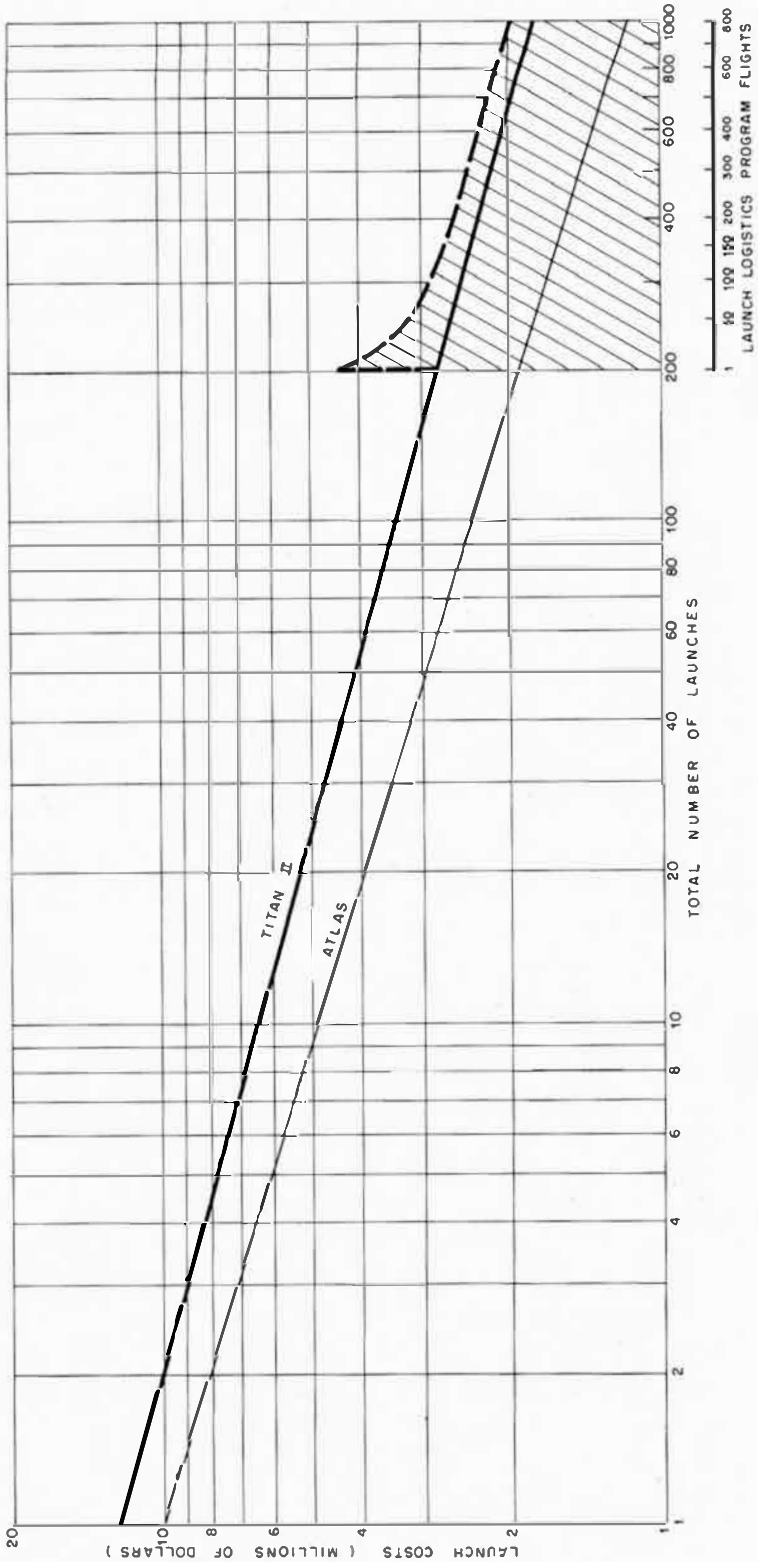


Figure 24 - Titan II Unit Launch Cost Progress Curve

Although present plans do not call for anywhere near the number of units or rates covered by this curve, if the program described herein were carried out (with over 800 units required) in addition to those presently planned and those needed for other programs (planned military deployment and training flights, plus Gemini, Titan III, etc.) the total would cover substantially the full range of units given on the curve.

The values used for estimating the costs for this program were taken to be somewhat higher than those shown on the basic Titan II progress curve of Figure 24, since it is assumed that sufficient changes might be required (even though minor) that, at least for the first 50 units or so, increased costs would be incurred. In fact, it was estimated that for some of the first few units costs might run as much as 50% higher than indicated by the over-all Titan II progress curve, and that such effects would not entirely settle out until beyond the 100th unit (as indicated by the dotted curve). This early portion is actually equivalent to a 93% starting slope, blending in well beyond the 100th unit. Data available on the Thor-Agena and other vehicles seems to substantiate this, although this program would involve far more uniform standardization of boosters and launchings than any of our programs to date.

Since the first Titan II was built some time ago (with quite a number already built and several fired), and since there are many yet to come (for R&D, evaluation, training, deployment, Gemini, Titan III, NASA and DOD space programs, etc.) before the first unit would be needed for this program (which is assumed not to start before late 1964, at the earliest), the first unit for this program was assumed to be number 201. It was also assumed that all the units needed for this program were then taken out in a solid block, for the whole 800 and more needed.

This means an assumption, for cost estimating purposes at least, that none will be needed during this interval for any other purpose. However, even though this may be somewhat unrealistic, it tends toward the conservative, or high side, so for estimating purposes it should be satisfactory.

The figure used for Titan II launch costs (of $2.007 billion) is obtained as the integral of the entire area under the curve from unit numbers 201 to 1042 using the dotted curve as the upper boundary (giving $1.957 billion) plus an added $50 million for abort attrition. It is given as Item 3 in Table 12. In regard to the allowance for abort attrition, it was estimated that for the several hundred flights involved (and starting beyond the 100th flight of the basic booster) that abort rates of between 1 in 20 and 1 in 50 could reasonably be expected (probably the former as an average near the beginning and the latter in the later portion of the program). This would represent from 17 to 42 additional boosters, and it was estimated that an additional $50 million should be allowed for this.

It is perhaps worth noting in regard to this element of the estimated program cost, that if only 2 more moon landings were required than was assumed (say 1 one way or supply landing, and 1 round trip rescue or proving flight landing with return to the lunar space station), then around 70 additional Titan II flights would be required. The corresponding cost increment would be $135 million, read by extending 70 units beyond 1042 (or to 1112) on the basic curve. However, this only represents 7% of this total cost item. This serves to demonstrate that the only way in which this item would be likely to be appreciably affected would be due to a significant basic change either in its slope or its A value, in which case a direct corresponding percentage adjustment in this entire cost item (Titan II launching) would have to be made. It should be remembered that this item does not include any part of the payload costs (except for the cost of launching it), and that all of these costs (whether fuel, supplies, equipment, vehicles - Gemini capsules, Agenas, etc. - or any other payload item) must be included in other items of the program cost estimate. Also, this item does not include the cost of the added launch pads or complexes which are needed to carry out these launchings. These costs are listed separately as Items 15 and 16.

Finally, in the case of the Titan II, most of the vehicles are straight non-man-rated Titan boosters -- only those flights for manning and rotating the space station crews are the man-rated Gemini version; and these flights (64 in number) involve less than 1 booster out of every 10 used in this operation.

Another significant cost item is the cost of the required number of Gemini vehicles, stipulated in the program described. This is given as Item 1. To estimate the cumulative average unit cost, it was assumed that presently planned and other Gemini program requirements (e.g. Air Force cooperative flights, etc.) would altogether require around 15 vehicles to be manufactured before vehicles were needed for this program. So the first unit cost read off the progress curve was for unit number 16, and it was again assumed that a solid block of 35 units were taken for this program. While, again, this may be an unrealistic assumption, either in terms of availability to (or needs of) this program, nevertheless it is on the conservative side for cost estimating purposes. Again it was also assumed that some additional costs might be incurred in the first few units for changes that might be needed or desired.

For the Agena B vehicle costs (Items 7 and 8), since this is a rather straightforward aerospace vehicle manufacturing proposition, standard methods utilizing cost data and progress curves were used to estimate this item. Again, an intact block of units was assumed.

The reduced number of total Agenas from that given in Table 10 and Figure 21, is in consideration of the 77 units given there being the top limit of the derived requirements. Due to the re-useability prospects of the Agena-powered excursion vehicles as mentioned in Section IV and elsewhere, it was felt that 77 units might be unduly conservative (55 or 56 were estimated to fulfill minimum basic needs) and that about 60 units (giving 4 or 5 for wear-out or attrition) might represent a more legitimate allowance, which is used in Items 7 and 8.

The estimated cost of the 13 new launch pads required was not refined to the extent of applying a learning curve, but was estimated at a flat rate of $3 million per pad. This was felt to be justified, on the basis that experience to date on the nature of progress curves on cost for this number of nearly identical installations of this kind is not yet sufficient to form a reliable basis for estimating. Data is available, however, on the basic construction costs for such launch pads. This cost is given as Item 16. The modification of three existing launch pads, Item 15, include the cost of making all launch pads similar for compatibility of the Titan II launch vehicles for this system - both physically and electrically.

Since the remaining items of Table 12 involve development and/or procurement (production) costs, each aspect must be considered individually. For many of these items, different methods were required for estimating the costs. Many of the items involved, unlike those discussed above, have not been manufactured or designed and tested as yet, so no actual data is available. However, most of them (tanks, payload containers, adapter structures, landing gear outriggers, etc.) are sufficiently simple and straightforward (and non-critical in terms of cost magnitude relative to the total program cost), and sufficiently similar to previously developed aerospace hardware, that standard means of deriving estimated costs should be adequate. Accordingly, basic unit costs were estimated by using previous data on a cost per pound basis for development and manufacturing of comparable types of hardware, and these used to obtain total costs for the entire item. In some cases, man-hour and material estimates of required development engineering was used and converted to equivalent costs. For a more specific treatment of these various cost items, the following additional considerations apply.

The development cost summary is given in Table 13, which includes all the major engineering cost items, such as an allowance for systems engineering and integration, operations engineering costs, and a substantial amount for unspecified contingencies. The largest development item is the space station, with the Agena modification ranking next in magnitude. All of the rest of the items are relatively simple items, primarily rather simple vehicle structural elements -- with the exception of the rendezvous guidance and control equipment for the fuel and supply carrier modules. For this, mostly Gemini-Agena rendezvous equipment would be utilized, with appropriate additions. The space station development costs were taken from development program cost estimates previously made at Goodyear Aircraft and submitted to NASA.

These development costs (with the exception of the Agena modification costs -- which were largely based on estimates of the engine and vehicle manufacturers) were arrived at mainly by the standard technique of estimating the man-hours and materials required, and allowing a reasonable average man-hour charge. For

Table 13 - Development Cost Summary

Reference Number	Item	Estimated Cost
System Elements		
A	Space Station and Lunar Cache	$ 80,000,000
B	Agena Modification (Throttling, etc.)	15,000,000
C	Excursionary Vehicle Auxiliary Tanks System	1,200,000
D	Excursion Vehicle Support Structures and Equipment	900,000
E	Space Station Storage Tank System	950,000
F	Lunar Landing Gear and Attachment	500,000
G	Fuel and Supply Modules	
	a. Structure	500,000
	b. Guidance, Propulsion, etc.	7,500,000
	SUB TOTAL	$ 91,550,000
Other Development Engineering Elements		
H	System Integration	$ 2,600,000
I	Operations Engineering Support	3,300,000
J	Contingencies	12,000,000
	SUB TOTAL	$ 17,900,000
TOTAL DEVELOPMENT COSTS		$109,450,000

example, Item G in the table is derived from an estimated requirements of 1,000,000 material dollars and 650,000 man hours of engineering, engineering shop, and technician time at an average rate of a little over $10 per hour. (These represent total costs - that is, basic costs plus material handling cost burdens, general and administrative costs, and normal contractor fee charges in addition to the direct labor and overhead costs.)

The other engineering costs elements included (Items H, I, and J) are largely straight engineering man-hour requirements, except for Item J , which is one of the major items. Item J represents a miscellaneous and contingency allowance for approximately $1 million dollars of extra materials and 1,000,000 man-hours (of which only about a third - consisting mainly of small items -- can be specifically visualized at this time).

The basis for most of the items of procurement costs are given in Table 14. In many cases, relatively good cost data, based on experience with the same or similar items were available; in other cases (e.g., Items 5 through 8) it was necessary to make estimates based on past industry experience with similar items on a cost per pound basis.

In all cases these costs were necessarily projections for activities not yet planned, and therefore subject to well recognized limitations in accuracy. The definition of numbers of units, time of requirement and availability, and in some cases the description of the items involved were therefore entirely dependent on the analysis of the postulated program presented in this report and associated preliminary design estimates and engineering analysis. No adjustments or projections were made to account for prospective changes in labor rates or other economic factors or trends.

Table 14 - Unit Production Cost Factors

Reference Number	Item	Average Unit Cost
1	Gemini Capsules (Cumulative Average for Block Assumed)	$8,000,000 each
2	40' Diameter Space Station (6,000 lb. at $1000/lb. with Furnishing and Basic Initial Equipment)	6,000,000 each
3	Agena (Cumulative Average)	2,000,000 each
4	Lunar Caches (Similar to Space Station Structure and Equipment, but Smaller, with Extra Radiators, Thermal Protection, etc.)	6,600,000 each
5	Fuel Tanks - (800 lb. Each at $150/lb.)	120,000 each
6	Tank Mounting and Vehicle Adapter Structure ($80/lb.)	40,000 each
7	Outrigger Landing Gear and Attachment ($200/lb.)	100,000 each
8	Additional Space Station Equipment (25,000 lb. at $50/lb.)	1,250,000
9	Supply Carrier Modules	250,000 each
10	Fuel Carrier Modules	225,000 each
11	Fuel Storage Supply System (Fuel Storage Tanks, Docking Equipment, Pumps, etc.	200,000 each
12	Supplies (Average, Loaded in Vehicle)	20 per lb.
13	Fuel for Agenas (Titan II Fuel Included in Launch Costs)	1 per lb.
14	Modify Existing Titan Launch Pad, Cost Per Pad	800,000 each
15	Build New Titan Pads - Average Cost Per Pad	3,000,000 each

However, despite these and other limitations and assumptions, valid data and methods were used to the extent available and within the justifiable limits of effort expended in the cost analysis to balance other aspects of the program analysis. It should also be remembered that many of the costs would be entirely different were it not for development accomplished on past programs, or programs currently nearing completion. Therefore, the overall program cost estimate derived as presented here should represent a sufficiently sound indication of the general magnitude, or "ball-park" value, of the program -- such that it can serve as a general guide for either acceptance, or for adjustment in accordance with any better data available to those wishing to use it and consider its general implications.

SECTION VII - SUMMARY & CONCLUSIONS

This study seems to reveal with significant clarity how the use of orbiting space stations as advance bases in an organized task force type of approach can add capability to achieve ambitious space objectives without undue stress and strain, and how it can thus enhance the chances for success.

As in scaling Mt. Everest or exploring the Antarctic, or any such ambitious new endeavor, it is the cumulative effect of hundreds of little details (many unexpected) that will get you if your approach is not such as to deal with them as they come. For this reason, the organized task force approach, using a series of operating bases appears highly desirable or necessary to give a more reasonable chance of success. Use of the space station as advance bases or staging points seems to give us this kind of advantage, and permits us to pursue current goals using presently available boosters. Such an approach has dominated space thinking in this area for decades. Without a space station consideration of this kind of approach becomes impossible, of course.

Therefore, the potential operational and other capabilities that can be afforded by an early initial type space station are very great indeed. Maximum use of it and resultant associated application of current hardware can greatly enhance our space capability. Also, other more specific aspects noted below appeared from the results of this analysis.

There are numerous other secondary but perhaps crucially important virtues inherent in this approach. For example, the astronauts landing on the moon would find the environment familiar, for they would have been living and working under lunar g conditions for weeks (at the space station), servicing ships in a vacuum, examining the landing site and landed equipment through binoculars and telescopes, etc. Any simple inspection or servicing adjustments needed on their vehicle (or any other operations) would be with tools and techniques that had been evolved over 2 years of operation in a similar environment. Numberous alternatives are available to choose from at every juncture as the operations expand, with continuous opportunity to change techniques in response to difficulties encountered and accumulated experience, all without development of any major new equipment items. The approach used significantly reduces operational problems and risks by progressively introducing one new operational dimension or technique at a time in each new flight series after the techniques required in the previous series have been mastered.

Various possible trade-offs were encountered in the course of the analysis, many of which (although they would show much more attractive program characteristics) were not used because of adherence to a no-risk criteria wherever the question of selection of presently available hardware versus advanced high performance hardware under development appeared. In such cases (e.g., Atlas-Centaur launch vehicles @ 8,500 lb. payload capacity, Centaur SEV propulsion @ I_{sp} = 400+) the potential higher performance hardware use was rejected in favor of existing equipments and techniques, even though they may actually become available and prove satisfactory for incorporation into operations during the program implementation.

Very favorable cost and schedule implications which appear to be inherently characteristic of this type of operation result primarily from such use of current hardware and gradual progressive initiation of the new required techniques made possible by this approach, which in turn avoids the cost, time, and uncertainties involved in developing extensive new hardware capabilities and making it work. Consideration of operations, support, lead time, etc., developed in this analysis indicate that the approach described has the inherent capability to accomplish manned landing on the moon by the end of 1966, or thereabouts - before the solar storm cycle becomes adverse.

The cost increment to support such a program is in the neighborhood of $3 billion. This gives an interesting indication of possible cost to the USSR for accomplishing lunar landing, which (allowing for their current status and lower living standards, affecting all costs through wages, etc.) should not be over the equivalent of $1½ to $2 billion.

It has been noted that continuation of the system operation beyond the point of lunar landing covered in this analysis would permit going right ahead with the lunar exploration and establishment of a lunar base. It appears that no significant new or increased effort or rate of expenditure would be required to build up a 6 to 10 man continuous operation over a period of a year or two, and continuously supporting a gradual expansion of activities, equipment, and facilities. (These considerations indicate that the Soviets also could, and probably will, do likewise at an equivalent cost to them of probably less than $1 billion annually).

From these observations, and based on the estimated USSR space budget, it does not appear that the earth orbital, lunar landing, and lunar base operations can represent a budget problem to them - there should be plenty left over for other programs. (This might prove equally true for the U.S., as well.)

In operating the space stations for this program, plenty of reserve capability should exist to handle all sorts of scientific projects and military tests (such as those listed in Appendix A) in addition to carrying out their role in the operations described herein for lunar operations. It is estimated that after operations are well established, and with a reasonable work load, each station crew could cumulatively have available approximately 5 or 6 scheduled hours and 30 non-scheduled man-hours per week to devote to such activities. Additionally, it would then be possible for scientists to visit and work at the station for 1 to 2 month periods each, to the extent of about 6 such man-months per station per year, and possibly more.

Unless specific operational goals must be met, by specified times, the rate and degree or extent of implementation of the operations described herein are entirely optional. For example, the space station can be put in orbit to serve as a Gemini rendezvous target and for only temporary occupancy (a few days at a time) at just a few million dollars, or it can be put into operation and continuously manned, without any extensive specific goals, at a total cost of less than $200 million; and for relatively little more (between $50 and $70 million), limited space operations using the station as a flight base, and extensive scientific work could be carried out. Later expansion to any part of the cislunar or lunar operations described in this analysis would still remain as a further option.

In conclusion then, the main message conveyed by this analysis is the demonstration of the extent of the operational and service capability that can be offered by the type of space station which can be launched by present boosters and propulsion units that are flying now, and therefore can be counted on to provide the required operating reliability and dependability. The flexibility and versatility afforded at low-cost by putting this type of space station into orbital operation is probably unparalleled by any other step we can undertake at this time. It is almost certainly representative of the course the Soviets will follow.

The kind of operation described in this analysis certainly possesses the inherent flexibility with which to deal with and accomodate the various problems that are certain to be encountered in any such endeavor. Such a progressive "task force" type approach is probably unparalleled in this respect. This type of operation would open up tremendous growth potential for natural follow-on utilization of more advanced vehicles and/or station improvements based on this experience as rapidly as they become available, and the attractive logistics picture shown here would improve still further as rapidly as more efficient vehicles became available for use.

Such operations would enable us to add insurance to our Apollo program objectives, and to carry out many things leading to planetary operations; and would contribute to the enhancement of our space capability in general -- all at a relatively modest cost. Therefore, it would seem to merit serious consideration as a worthwhile supplement to the other efforts making up our current space program.

Page intentionally left blank

LIST OF REFERENCES

1. WS-4024 - Memorandum on an Early Space Station for Gemini Capsule, Goodyear Aircraft Corporation, Akron, Ohio - 10 July 1962 (Enclosed as Appendix A)

2. SP-1255 - Memorandum on a Gemini Space Station, Goodyear Aircraft Corporation, Akron, Ohio - 2 August 1962 (Enclosed as Appendix B)

3. Soviet Space Programs: Organization, Plans, Goals, and International Implications, Staff Report for Committee on Aeronautical and Space Sciences, U.S. Senate, 31 May 1962

4. Veda Technika Mladezi, Vol. No. 9, 1962, Pages 305 to 307

5. SP-1362 - Determination of Orbital Launch Windows for Lunar Rendezvous, Goodyear Aircraft Corporation, Akron, Ohio - 14 September 1962 (Enclosed as Appendix D)

6. Technical Note D-866, Lunar Trajectories, Lewis Research Laboratory, NASA, Cleveland, Ohio, August 1961

7. P-1441, Circumlunar Trajectory Studies, H.A. Lieske, Rand Corporation, Santa Monica, California, - 25 June 1958

8. GER-9827 - Astronautics Handbook - Goodyear Aircraft Corporation, Akron, Ohio - 13 June 1960

Page intentionally left blank

BIBLIOGRAPHY

Note: This study is based in many respects on more than a decade of work at GAC directly related to space stations, lunar bases, and associated vehicle technology. Some of the more pertinent and significant papers and technical publications resulting from this work is listed in this bibliography. Documents are unclassified unless otherwise noted.

1. "Guidance and Control Systems for Space Ships." (A treatment of system requirements and techniques for Orbital Ascent, Rendezvous, and Lunar Landing). D. C. Romick, Goodyear Aircraft Corporation, Akron, Ohio, 27 November 1953.

2. "Preliminary Engineering Study for a Satellite Station Affording Immediate Service with Steady Evolution and Growth," D. C. Romick, Goodyear Aircraft Corporation, ARS preprint No. 274-55, American Rocket Society, 500 Fifth Ave., New York, N. Y. 21 November 1955.

3. Concept for METEOR, A Manned Earth-satellite Terminal evolving from Earth to Orbit ferry Rockets, presented at the 7th annual Congress of the IAF, Rome, Italy, by D. C. Romick, Goodyear Aircraft Corporation, Akron, Ohio, 17 September, 1956.

4. METEOR, Jr., A Preliminary Design Investigation of a Minimum sized initial vehicle system of the Meteor concept, by D. C. Romick, R. E. Knight, and Samuel Black, Goodyear Aircraft Corporation, Akron, Ohio, presented at the 8th Annual IAF Congress in Barcelona, on 7 October 1957.

5. GER-8807; Lunar Terminal Guidance, Goodyear Aircraft Corporation, Akron, Ohio, 7 May, 1958.

6. ARD-7543; Program for the Study of Environmental Cells for Crew Protection in Space Vehicles and the Mock-up Fabrication of Selected Cell, by J. C. Bell and F. M. Milhoan, Goodyear Aircraft Corporation, Akron, Ohio, 18 December, 1959.

7. GAP-9109 - SR-17527- Proposed Military Test Space Station, Goodyear Aircraft Corporation, Akron, Ohio, 15 March 1960 (CONFIDENTIAL).

8. Applications of Inflated, Expandable Structures to Orbital Flight, and reentry, by F. B. Sandgren and J. T. Harris, Goodyear Aircraft Corporation, Akron, Ohio, ARS preprint No. 1093-60; 6 April 1960.

9. GER-9766; Preliminary Development and Design of an Expandable Torus Space Station (under Contract No. NAS1-620); Goodyear Aircraft Corporation, Akron, Ohio; 22 April 1960.

10. GER-9875 - Application of Expandable Structure to Space Station Configuration, June 1960.

11. Compendium of Papers presented at the GAC Expandable Structures Symposium for the Air Force, Goodyear Aircraft Corporation, Akron, Ohio; June 21-22-1960.

12. WS-2372 - Attitude Control and Propulsion Requirements of the NASA 30 ft. diameter Inflated Torus Test Space Station - a consideration of orbital dynamic effects and the means for counteracting them, by Samuel Black, Goodyear Aircraft Corporation, Akron, Ohio, 30 June 1960.

13. WS-2446 Rendezvous Guidance Analysis by G. J. McKeel, Goodyear Aircraft Corporation, Akron, Ohio, 4 August 1960.

14. GER-9963 - Elastomer and Cord Fabric Investigations and Preliminary Design Studies Associated with an Expandable Orbital Space Station (under Contract No. NAS1-725), Goodyear Aircraft Corporation, Akron, Ohio 26 August, 1960.

15. WS-2016- Life Support System for Space Stations by F. A. Whitmire, Goodyear Aircraft Corporation, Akron, Ohio, 30 August 1960. (Note: These Technical Memorandum Reports and associated data were submitted to NASA-Langley during the summer of 1960 in support of concurrent space station work).

16. Theoretical and Experimental Evaluation of Polymeric Materials for Use in a Space Environment (Paper presented at WADD meeting on Coatings for the Aerospace Environment), by Dr. C. E. Snyder, Goodyear Tire & Rubber Company, and W. B. Cross, Goodyear Aircraft Corporation, Akron, Ohio, 9 November 1960.

17. GER-10101 - Effects of High Vacuum and Ultra-Violet Radiation on Mechanical properties of Polypropolene Films and Polyurethane Foam, 23 December 1960.

18. GER-10153 - Effects of High Vacuum and Temperature on the Creep Strength of Coated and Uncoated Nylon Yarns - 1 February 1961.

19. GAP-9671S1 - Preliminary Design Study for Proposed Expandable Space Station for 1963 Operation - a Mercury - compatible concept; Goodyear Aircraft Corp., Akron, Ohio; 10 April 1961.

20. GER-10240 - Physical Properties of Materials for Expandable Structures, Goodyear Aircraft Corporation, Akron, Ohio, 19 April 1961.

21. GER-10251 - Permanent Satellite Base and Logistics Study, Vol. IV (AFBMD TR61-64(IV)), Temporary Lunar Base, Goodyear Aircraft Corporation, Akron, Ohio; June 1961 (SECRET).

22. GER-10310 - Dynamic Analysis Applicable to GAC Expandable Space Station Program, 13 July 1961.

23. Expandable Space Stations; Implications for Prolonged Manned Space Flight -- a consideration of the effect of the human operator requirements and constraints, presented at the 3rd International Symposium on Rockets and Astronautics, Tokyo, Japan, by Dr. A. J. Cacioppo and N. L. Jeppesen, Goodyear Aircraft Corporation, Akron, Ohio; 28 August 1961.

24. GAP-1010 - Proposed Study of a Self-Erecting Manned Space Laboratory, Vol. I - Technical Outline --including definition of the main technical aspects, Goodyear Aircraft Corporation, Akron, Ohio, 11 September 1961.

25. GAP-1049 - Proposed Space Laboratory, Artificial Gravity Mechanisms, and Rendezvous Techniques for Project Apollo, Goodyear Aircraft Corporation, Akron, Ohio, 11 September, 1961 (CONFIDENTIAL).

26. GAP-1190 - Expandable Space Maintenance Hangar for MMSCV System, Goodyear Aircraft Corporation, Akron, Ohio, 30 November 1961.

27. GER-10617 (S109-9) - Structural Investigation of Torus-Cylinder Intersection, by J. D. Marketos and J. E. Houmard, 20 March 1962.

28. GER-10679 - Study of Radiation in Cis-lunar Space, 29 June 1962.

29. GAP-1204S1 - Proposal for Design Study for Lunar Survival Cache, Goodyear Aircraft Corporation, Akron, Ohio, 25 July 1962.

30. GER-10663 - Effects of Hypervelocity Particle Impact on Composite Materials for Expandable Structural Application by B. W. Reynolds, August 1962.

31. GER-10755 (S109-11) - Approximate Analysis of the Structural Stability of the C-annulus Space Station Model, by J. E. Houmard and J. D. Marketos, Goodyear Aircraft Corporation, Akron, Ohio; 18 September 1962.

32. GER-10861 - Optimum Torroidal Pressure Vessel Filament Wound Along Geodesic Lines, by James D. Marketos, Goodyear Aircraft Corporation, Akron, Ohio, 10 October 1962.

Page intentionally left blank

MEMORANDUM

10 July 1962
WS-4024

To: F.J. Stimler

Subject: Early Space Station for Gemini Capsule

I. GENERAL

The Gemini two-man space capsule program is presently orientated towards the accomplishment of two major goals:

1. Provide manned, orbital flights of a week (or more).
2. Test bed for developing and proving orbital rendezvous techniques.

Secondary objectives to be realized by the Gemini program include:

3. Development of multi-crew procedures.
4. Training of space crews.
5. Evaluation of a limited number of Apollo systems and subsystems.

Attainment of goal number 1 will furnish preliminary information concerning the effects of long term orbital flight on men and equipment. Goal number 2 will check the feasibility and develop the procedures for intercepting and mating of objects in orbit. By the successful completion of these goals, long duration space travel and rendezvous, the Gemini program will have advanced the feasibility of prolonged manned space flights to the moon and the use of orbital rendezvous techniques for accomplishing the lunar (and other) space mission.

The addition of a space station compatible with the Gemini vehicle and available at an early date (1964) is an attractive concept in that it will allow the Gemini crew a much larger area containing more equipment and power with which to accomplish their space mission than is available in the Gemini capsule. This station will also serve as the proving grounds for future large space stations already under study. An early space station could check out such Apollo systems as its air lock, environmental and life support equipment, or even a complete mock-up of the Apollo capsule.

II. MISSIONS IN SPACE

The following is a list of space missions some of which are presently being accomplished by unmanned satellites and the Mercury program while others must wait for more advanced manned vehicles and space stations.

1. Scientific

   a. Radiation
   b. Astronomical
   c. Interplanetary
   d. Space Sounding
   e. Spectrology
   f. Extra-terrestial Life
   g. Materialogical Studies
   h. Minerological Studies
   i. Lunar Exploration

2. Medical

   a. Radiation
   b. Biological
   c. Zoological
   d. Entomalogical
   e. Germicological

3. Meteorological

   a. Climatological
   b. Ionosphere
   c. Cosmic Radiation
   d. Solar Activity
   e. Stellar Activity
   f. Interplanetary Effects

4. Military

   a. Observation (Reconnaissance)
   b. Training
   c. Interplanetary Staging
   d. Retalitory Base (Orbital Bombardment)
   e. Inspection (Orbital Defense)
   f. Command Control
   g. Space Astronomy
   h. Biological Research
   i. Physiological Research
   j. Meteorological Research
   k. Communications

5. Space Medicine & Behavioral Science

a. Weightlessness
b. Confinement
c. Isolation
d. Nutrition
e. Artificial Life Support
f. Radiation
g. Acceleration
h. Vibration

6. Space Biology

a. Search for Extraterrestrial Life
b. Study of Environmental Effects

7. Biotechnology

a. Man-Machine Integration
b. Crew Performance
c. Stress Tolerance
d. Zero G
e. Artificial Gravity
f. Atmospheric Physics and Chemistry (Upper Air Research)
g. Spectroscopy
h. Fields and Particles
i. Meteorites
j. Astronomy
k. Celestial Mechanics
l. Relativity
m. Biology (physiology & psychology responses to environment, exobiology, anti-contamination)
n. Information Theory

8. Astronomy

a. Telescope (high definition photographs)
b. IR Spectograms
c. Grating Telescope (900-3000A) (IA Resolution)

III. MAN IN SPACE

1. What are the Objectives of Manned Flights?

a. Explore Moon (Lunar)

Land on, explore and eventually establish a base on the moon. The landing of men on the moon (Apollo) is the immediate goal of the United States Space Program.

b. Explore the Universe (Interplanetary)

As the art of space operations advances manned exploration of the Universe and especially Mars and Venus (our two closest planets) is planned as a "second generation" goal of the U.S. space program.

c. Conquest of the New Medium of Space

Develop systems and procedure by repeated space flights so that operations in space can be made as effective as present commercial and transportation systems on land, sea and air. (Commercial aspects of space.)

2. What are the National Goals in Space?

a. Guarantee free access and use of space to all nations.

b. Utilize space for peaceful commercial purposes.

c. Increase or uphold national prestige.

d. Enhance or expand national security by use of space.

e. Expand man's knowledge of the universe

3. What Can Man Do In Space?

a. Increase system reliability many fold by performing monitor, operation, maintenance and repair functions.

b. Replace complex automated devices by using man's judgement, ingenuity, adaptability, and decision making ability.

c. Operate and test space systems and hardware in the space environment. Monitor, record and perform various experiments and tests required to check-out equipment and procedures.

d. Control orbital vehicle operation to add flexibility and selectivity in performing space missions.

e. Utilize man's unique talents as an observer, operator, and reporter.

4. What are Present Manned Space Programs?

a. Approved

(1) X-15 Research Aircraft
(2) Dyna Soar Aircraft (15,000 lbs.)

(3) Mercury (2,900 lbs.)
(4) Gemini (6,000 lbs.)
(5) Apollo (A-29,000 lbs., B-50,000 lbs., C-215,000 lbs.)

b. Proposed

(1) Military Test Space Station
(2) Recoverable Orbital Launch System (Aerospace Plane)
(3) SLOMAR (Supply, logistics, maintenance and rescue)
(4) Global Surveillance
(5) Self Erecting Space Station (150,000 lbs.)
(6) Orbital Inspection of Space Vehicles
(7) SMART (Space Maintenance & Repair Techniques)
(8) Command and Control Post
(9) Orbital Bombardment System

5. How Can Future Space Requirements be Implemented?

a. Advance state-of-the-art with step-by-step progression.

b. Advance state-of-the-art with accelerated technical jumps.

c. Major technical breakthroughs.

6. How Can Manned Space Flights Aid and Enhance the Implementing and Development of Space Programs?

a. Future, advanced space systems will be mostly manned and man must be included in the development, testing and operation of the system hardware.

b. Man has always been the most efficient machine for conducting and controlling experiments and tests. Can use man's ability to observe, interpret, and judge results to determine: (1) if test must be repeated, (2) if test can procede to next item, or (3) if test can by-pass next item(s), thus accelerating program.

c. Man can develop operating procedures for the operational use of space systems simultaneously with the development and testing of the systems.

d. Man can inspect, adjust, maintain, and repair equipment during experiment and test periods for increased reliability and desired results.

7. Contributions of Man

a. Human intelligence, decision-making, and information processor.

b. Control and stabilization (operation) of space systems.

c. Equipment Maintenance:
(1) Detection of malfunction
(2) Diagnose cause of malfunction
(3) Repair malfunction
(4) Replace with repaired or spare unit

8. Objective of Early Space Station

a. Space Laboratory.

b. Advanced systems evaluation and qualification.

c. Training and development of operational procedures.

9. Manned Duties

a. Flight duties involving system management of propulsion (altitude control) for proper orientation, guidance, environment control, fuel and power management, and other command functions.

b. Duties involving navigation and position checks including operation of navigation equipment.

c. Duties involving communications including operation of communication equipment.

d. Duties involving scientific experiments and observations including laboratory equipment.

e. Duties involving maintenance and repair of mechanical and electronic equipment.

f. Duties involving maintenance and repair of station and other space vehicles.

g. Duties involving leadership and supervision for adequate work, rest, relaxation and sleep assignments plus health, sanitation, nutrition and morale.

h. Duties involving emergency procedures including escape and rescue.

i. Duties involving training and standard procedures development.

j. Duties involving advanced system testing, evaluation and qualification.

k. Duties involving crew rotation, rendezvous, docking and resupply.

IV. NASA AND USAF SPACE PROGRAM

1. Mercury Three-Orbit Flight (1961 to 1962)

    a. Achieve manned orbital flight and successful recovery at earliest practical date.

    b. Study man's capabilities and effects in a space environment.

2. Mercury One Day Flight (18 Orbits) (1963)

    a. Obtain experience under weightless conditions for periods up to 1 day.

    b. Determine effects of prolonged weightlessness and G-stresses after weightlessness.

3. Gemini Earth Orbit (1963 to -)

    a. Early manned rendezvous capability

        (1) Develop techniques
        (2) Assess pilot functions
        (3) Develop propulsion, guidance, and control
        (4) Develop pilot displays
        (5) Train pilots

    b. Provide long duration manned flight experience

        (1) Study effects of weightlessness
        (2) Determine physiological reactions
        (3) Determine psychological reactions
        (4) Develop performance capabilities of the crew

    c. Test bed for Apollo components

    d. Practice orbital changes

4. Apollo Earth Orbit (1965)

    a. Test and evaluation of components and systems in space environment.

    b. Crew training.

    c. Development of operational techniques

    d. Laboratory for scientific measurements and technological developments.

5. <u>Apollo Cislunar and Circumlunar (1966 - 1967)</u>

   a. Exploration of space flights at increasing distances from earth until circumlunar flight is achieved.

   b. Development of guidance and control.

   c. Re-entry from space at high velocities.

6. <u>Apollo Lunar Landing (1968)</u>

   a. Rendezvous

   b. Lunar Landing

   c. Lunar Launch

   d. Lunar Exploration

   e. Earth Re-entry

   f. Lunar Orbit

7. <u>Self-Erecting Space Station (1967)</u>

   a. Scientific Experiments

      (1) Psychological and physiological response of man to space environment (weightlessness, confinement, isolation, radiation, nutrition, etc.).
      (2) Space Biology
      (3) Space Astronomy
      (4) Space Meteorology

   b. Systems Research

      (1) Testing of space systems, subsystems, and components such as:

         Advanced life-support systems
         Structures
         Propulsion (nuclear and ion drives)
         Communications
         Telemetry
         Exotic Fuels
         Surveillance
         Attitude Control
         Guidance
         Power Supplies

(2) Develop operational procedures for space systems.
(3) Advance state-of-the-art of large space stations for multi-mission usage.

8. <u>X-15 (1960 to 1964)</u>

a. Research of upper atmosphere.

b. Research of heating and other re-entry problems.

9. <u>Dyna Soar (1964 to -)</u>

a. Follow on to X-15 research vehicle.

b. Upper atmosphere and re-entry research leading to orbital reconnaissance, bombing capability, and other military missions.

c. Aerodynamic - controlled, maneuverable recovery and landing of an orbital vehicle.

V. MISSIONS FOR GEMINI SPACE STATION

1. <u>Operational Period - Gemini Space Station</u>

a. June 1964: Gemini available; unmanned lab launch.

b. January 1966: Apollo available for lab use - if special design and manufacture can be justified.

2. <u>Specific Missions</u> (Which can not logically be accomplished by Gemini or Apollo in the early time period)

a. Systems Management

(1) Power
(a) Configure, initiate and monitor operation
(b) Repair minor malfunctions (install spares)
(c) Balance, regulate and select alternate power sources

(2) Life Support
(a) Initiate and monitor
(b) Maintain (filter change, bed recharge—)

(3) Propulsion and Fuel
(a) Operate
(b) Repair (open clogged nozzles)
(c) Resupply

(4) Communications
  (a) Channel selection
  (b) Coding
  (c) Maintenance and minor repair
  (d) Relay

(5) Telemetry
  (a) Maintain and repair - station data, animal subject data--
  (b) Selection of significant data for recording and/or transmitting

(6) Resupply
  (a) Conduct resupply maneuvers (life support, etc.)

b. Observatory

(1) Photograph, process, evaluate, rephotograph, return negatives to earth. (high resolution)

(2) Deploy, position, operate high resolution camera

(3) Repair minor malfunctions

c. Meteorology

(1) Rapid interpretation of earth weather for warning purposes.
(2) Cataloging earth weather phenomena for advanced weather prediction techniques.
(3) Evaluation of earth heat balance as controlled by cloud cover and pressure gradients.
(4) Investigate feasibility of developing techniques to modify weather or climate from orbit.
(5) Measure and compile data on "space weather" to facilitate forecasting of solar flares, etc. (deploy, monitor and observe large fields of micrometeoroid sensors).

d. Space Biology and Medicine

(1) Operate biological laboratory
  (a) Develop anti-contamination techniques
  (b) Determine norms in space vacuum and weightless state before and after solar flare and other environmental effects.
  (c) Maintain prescribed environmental conditions including equipment repairs and changes.
  (d) Develop "biological" life support systems (algae, bacteria, recycling, etc.).
  (e) Simulate lunar gravity for extended periods.
  (f) Determine feasibility of conducting first aid and emergency operations in weightless condition.

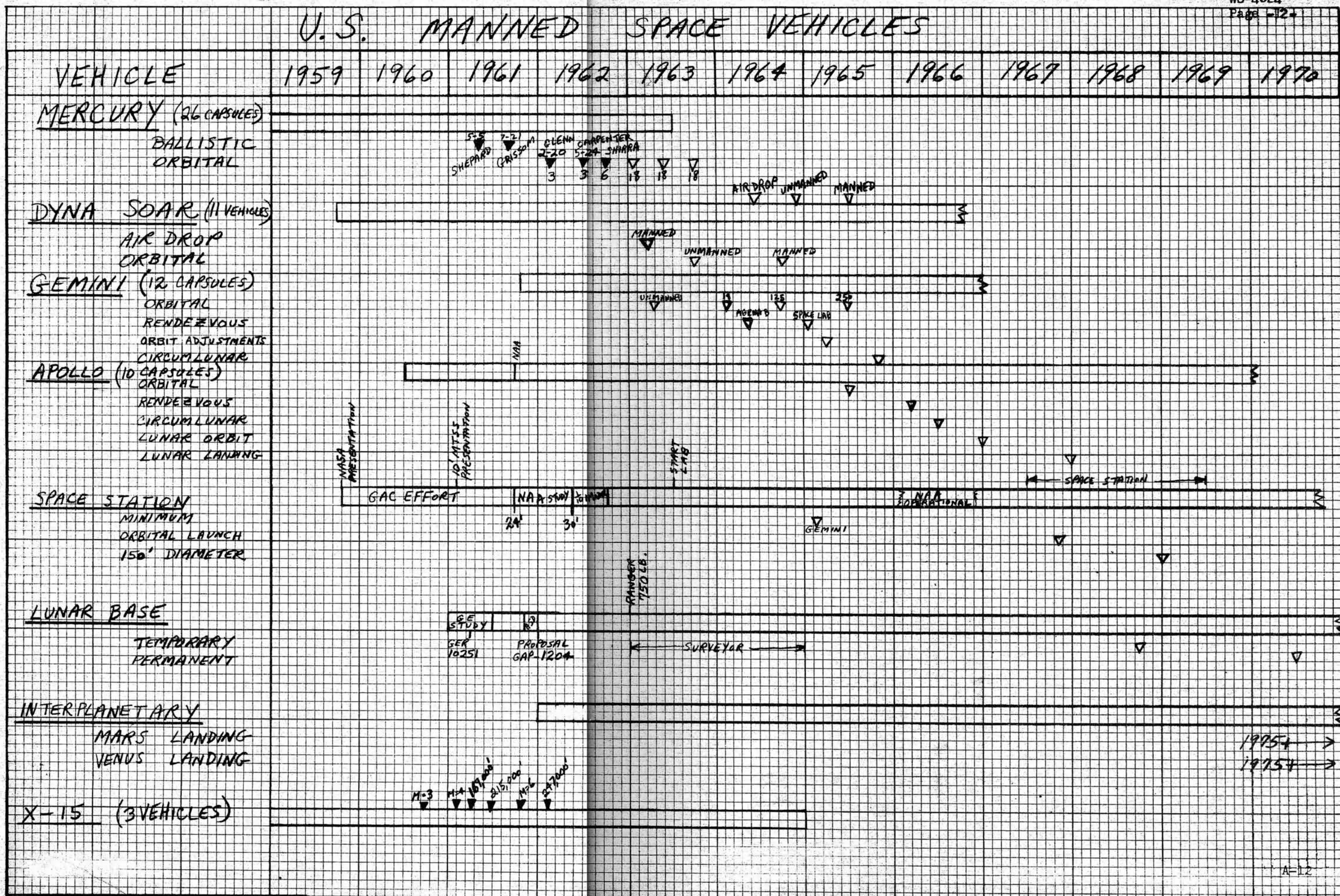
e. Advanced System Development and Testing

(1) Apollo subsystems
(2) Unmanned satellite subsystems (bread board)
(3) Large space station subsystems
(4) Advanced power and propulsion systems
(5) Check-out and calibration of piggy back satellites
(6) Advanced communication systems

f. Additional Missions and Objectives

(1) Operations under simulated lunar gravity.
(2) Observatory - Fix jammed camera, reload film, develop and evaluate, focus and aim. Lunar surface mapping, photo development, rephotographing missed areas, varying magnification, hunting out more likely landing areas. Map coast lines and geographical features of earth.
(3) Weather Station - Detecting unusual patterns, interpreting phenomena with re-look and varying looks to evaluate height of clouds. Quick warning of dangerous situations.
(4) Communication - Equipment repair and adjustment. Test and evaluate Apollo communications equipment.
(5) Micrometeoroid Field - Sensor and photograph. Deploy sensors and check/repair circuits and recording devices. Examine and photograph actual penetration areas as identified/located by sensors.
(6) Radiation studies including photographic processing of plates at normal periods and after solar flare. Evaluate materials and geometrics of shielding.
(7) Perform Apollo crew functions in simulated cabins in space environment.
(8) Biomedical - Examine animals before and after solar flare/zero gravity.
(9) Material Evaluation - Progressive measurement of prolonged exposure of materials (or devices) to space environment.
(10) Control Center - Orbital assembly and launch station.
(11) Permits use of lower reliability subsystems.
(12) Study pump cavitation and anti-slosh characteristics in zero gravity environment.
(13) Examine characteristics of food preparation for future space station kitchens.
(14) Develop self-locomotion and stabalization techniques for future space stations.
(15) Test and evaluate Apollo life support equipment.
(16) Investigate techniques and physical effects of producing artificial gravity by rotation.

Page intentionally left blank


U.S. MANNED SPACE VEHICLES
VEHICLE
1959
1960
1961
1962
1963
1964
1965
1966
1967
1968
1969
1970
MERCURY (26 CAPSULES)
BALLISTIC
ORBITAL
5-5 SHEPARD
7-21 GRISSOM
GLENN 2-20
CARPENTER 5-24
SHIRRA
3
3
6
18
18
18
DYNA SOAR (11 VEHICLES)
AIR DROP
ORBITAL
AIR DROP
UNMANNED
MANNED
MANNED
UNMANNED
MANNED
GEMINI (12 CAPSULES)
ORBITAL
RENDEZVOUS
ORBIT ADJUSTMENTS
CIRCUMLUNAR
UNMANNED
AGENA B
SPACE LAB
APOLLO (10 CAPSULES)
ORBITAL
RENDEZVOUS
CIRCUMLUNAR
LUNAR ORBIT
LUNAR LANDING
NAA
NASA PRESENTATION
10' MTSS PRESENTATION
START LAB
SPACE STATION
SPACE STATION
MINIMUM
ORBITAL LAUNCH
150' DIAMETER
GAC EFFORT
NAA STUDY
24'
30'
NAA OPERATIONAL
GEMINI
RANGER 730 LB.
LUNAR BASE
TEMPORARY
PERMANENT
GE STUDY
GER 10251
PROPOSAL GAP-1204
SURVEYOR
INTERPLANETARY
MARS LANDING
VENUS LANDING
1975+
1975+
X-15 (3 VEHICLES)
M-3
M-4
169,000'
215,000'
M-6
247,000'

Page intentionally left blank

VI. OPERATIONAL REQUIREMENTS FOR GEMINI SPACE STATION

1. Use same booster as Gemini - Titan II Missile.
2. Maximum payload of Titan II - 6,000 lbs. in 300 nm orbit.
3. Launchings from Cape Canaveral.
4. Dock two (2) Gemini capsules simultaneously.
5. Availability - 1964.
6. Growth from 1 man crew to 3 man crew.
7. At least one (1) air lock.
8. Orbital life - one (1) year (minimum).
9. Life support - 30 man days minimum.
10. Micrometeoroid protection.
11. Automatic checkout equipment for unmanned launch into orbit.
12. Supply largest volume feasible including work area, rest area, and housekeeping area.
13. Provide shirt sleeve environment.
14. Design should permit eventual assembly in space of 2 or more stations into larger space stations.

VII. DATA FOR AGENA (LOCKHEED)

	A	B
Launch Weight*	12,000 lbs.	21,000 lbs.
Wet Weight	8,500 lbs.	15,800 lbs.
Dry Weight	1,700 lbs.	2,100 lbs.
Fuel	6,800 lbs.	13,700 lbs.
Orbital Weight*	5,000 lbs.	7,100 lbs.
Dry Weight	1,700 lbs.	2,100 lbs.
Payload (30 nm orbit)*	3,300 lbs.	5,000 lbs.
Length	19.2 feet	26.5 feet
Diameter	5.0 feet	5.0 feet
Propulsion (Bell Aircraft)		
Thrust	15,000 lbs.	15,000 lbs.
Burning Time	120 secs.	240 secs.
Specific Impulse	295 secs.	295 secs.
Nozzle Ratio	20:1	45:1
Combustion Chamber Pressure	500 psi	
Engine Length	7.0 feet	
Engine Diameter	35 inches	
Engine Weight	280 lbs.	

Fuel: Unsymmetrical Dimethyl Hydrazine and Red Fuming Nitric Acid.

*These weights correspond to an Agena A or B launched by an Atlas Missile (Atlas-Agena).

VIII. REMARKS

An early available space station for the Gemini program appears to be attractive to the national space program for the following reasons:

1. Will provide early experience in space station operations. The data obtained will enhance and optimize future large space stations-notably, the self erecting space station (1967 - 1968 ea.).

2. Will provide experience in docking, boarding, and exiting from an actual space station including the use of air locks, compression and decompression of space suits, etc. which is much more advanced operational experience for the Gemini crew and the space effort than the continual rendezvousing with an empty Agena vehicle.

3. Will provide advanced environmental and laboratory test facilities for experiments necessary for the space effort that are impractical or impossible to perform on earth.

A capability such as this space station potentially possesses for performing a variety of civilian and even military experiments and missions would offer this country's space program a flexibility of approach in meeting our space goals. By developing space bases (stations) along with our space vehicles a number of options (trade-offs) are available in how we chose to use these systems, singularly or in combinations, in our conquest of space.

E R Smith
E.R. Smith
Department 460-G

ERS:hh

Page intentionally left blank

APPENDIX B - DESIGN REVIEW OF AN EARLY GEMINI SPACE STATION

MEMORANDUM

August 2, 1962
SP-1255

To: R. R. Carman
Astronautics Design
Department 453 - Plant G

Subject: Gemini Space Station

Copies to: Gibson
Klindworth
Swanson
Stimler

Introduction

This memo represents a summary of the results of a one-man month effort design review of a hard shell space station concept compatible with the Gemini program. Basic ground rules used were: Specification for Gemini Spact Station and Design Criteria, Manual Space Laboratory. Using the Titan II as a launch vehicle the station gross weight would be 6500#. On this basis the design review indicates a mission payload of approximately 2200#.

This preliminary design review is not intended to indicate optimum design configuration, system details, etc, however it does establish a feasible design concept which could be the basis for a comprehensive design study if a serious effort is initiated at GAC.

J. V. Brodecky

J. V. Brodecky
Astronautics Design
Department 453 - Plant G

JVB:lmm

A. STRUCTURE

1. Lab Structure (Inhabited Region)

Internal pressure of the station is 7 psi with a safety factor of 1.33 based on the yield strength of the basic material. Temperature of the skin $\leq$ 200° F. The basic shell skin uses a seam (or butt) weld for pressure tightness and spot welded splice straps for strength (load transfer) thereby using full tension yield strength of the basic material. Lab structure for meteoroid protection shall achieve a 95% probability of no puncture of critical elements of the structure. A double wall structure with low density filler was used, permitting a total skin thickness that is 1/3 that required for a single sheet. Materials considered for the lab shell included aluminum, stainless steel and Titanium.

The lab material for meteoroid protection indicates a choice of a .012 double wall of Titanium or an .017 double wall of aluminum. The lab material for internal pressurization indicates a choice of .025 aluminum or .011 Titanium. Keeping in mind the fact that once a shell material is selected all supporting structure welded to the lab skin must be if the parent material, Titanium was selected to achieve the lowest overall shell weight.

The lab structure resulted in a double wall construction of .012 Titanium skins separated by a 1" thick low density thermoflex insulation. Support structure includes equipment support ribs, floor and floor support ribs, 2 quadpane viewing ports, mounting flanges for adapter and jettisionable nose fairing, mounting provisions for the solar cell paddles and support frames for the air lock hatch. Total lab weight is approximately 538# (See Page 5). Typical sections through lab see Figure I, II, III. Composite station see Figure VI.

2. Adapter Section

The adapter section is a cylindrical section which joins the lab shell to the booster. Construction could be either sandwich or corrugated skin, seemingly both quite competitive. Using a 10 g boost load and S.F. of 1.35 a sandwich adapter was configured. Diameter is 10' and height approximately 8½', weight approximate 321#.

3. Nose Fairing

With a lab diameter of 10' the nose fairing will be very much similar to the centaur nose fairing. A N_2 stored gas thruster bottle separation system is used. Basic shell skin is sandwich. Weight is approximately 170#.

4. Air Locks

This is a cylindrical section 3' in diameter by approximately 6½' long. Provisions include pumps, storage tanks, pressure indicators, etc, with a reserve gas allowance for 2 complete cycles. The actual design configuration will be in conjunction with the docking and transfer provisions with the Gemini capsule. Weight estimated approximately 140#.

5. Solar Cells

For electrical requirements as given in GAP-96718l (see Page 7) approximately 284 ft^2 of solar cell surface area is required. This will be provided by the use of eight (8) solar cell mounting panels which will fold against the lab during launch and extend out during orbit (see Figure VI).

6. Propulsion and Attitude Control System

Design requirements of the system indicated a ΔV of 200'/sec for thrust vecter and approximately ΔV at 200'/sec for attitude control. With those requirements a H_2O_2 mono-prepellent system was selected. See Figure IV for schematic. An I_{sp} of 174 Sec was determined (optimum expansion and altitude $>$100,000) with (2) 500# thrust nozzles for main thrust and (2) 10# thrust nozzles for fine thrust control. t_b was 36½ sec, $\dot{\omega}$ of 2.88 lb/sec. The orientation system has a dual eight nozzle system with (8) 10# thrust nozzles and (8) .5# thrust nozzles for fine vernier control. Total weight of the system is approximately 780# with a 70# weight allowance for the ASCS system (Mercury ASCS weighs approximately 59# and TROS magnetic coil stab system about 20#).

7. Pressure System (lab)

Total lab volume approximates 525 ft^3 and the air lock approximately 50 ft^3. Leakage was directed at .05 lb/hr. (approximately ½ ft^3/hr) A 30 man day, open type life support system was used with a 10% reserve. The gas system used was stored gas. O_2 usage was based on approximately 25 ft^3/day/man. Total pressure system weight was approximately 380#.

B. SYSTEM EQUIPMENT

The system schematic upon which the environmental control system was based is shown on Figure V. The system is basically an open system except for water recovery where a semi-closed system recovers wash water from the daily water output. Only drinking water is carried. Total water and food imput/day is 6.3#. Water output per day in vapor and urine is approximately 5.2#/day of which approximately 3.98#/day is required to be recovered for sanatation needs. The system details follow the schematic is shown. Thermal control is by heat exchangers using glycol; odor removal uses an activated charcoal-filter bed; CO_2 removal by Li OH system; Illumination, Sanatation (waste storage only), ventilation system, galley, humidity control and environmental equipment are all similar in nature to zero g systems and equipment presented in most environmental control system space station proposals. Detailed weight estimates for each system were estimated with the lot. The weight summary as shown on page 5.

C. ELECTRICAL POWER SUMMARY (REFERENCE: GAP-967151)

1500 W capacity (800 W average electrical equipment load + recharge power for batteries)

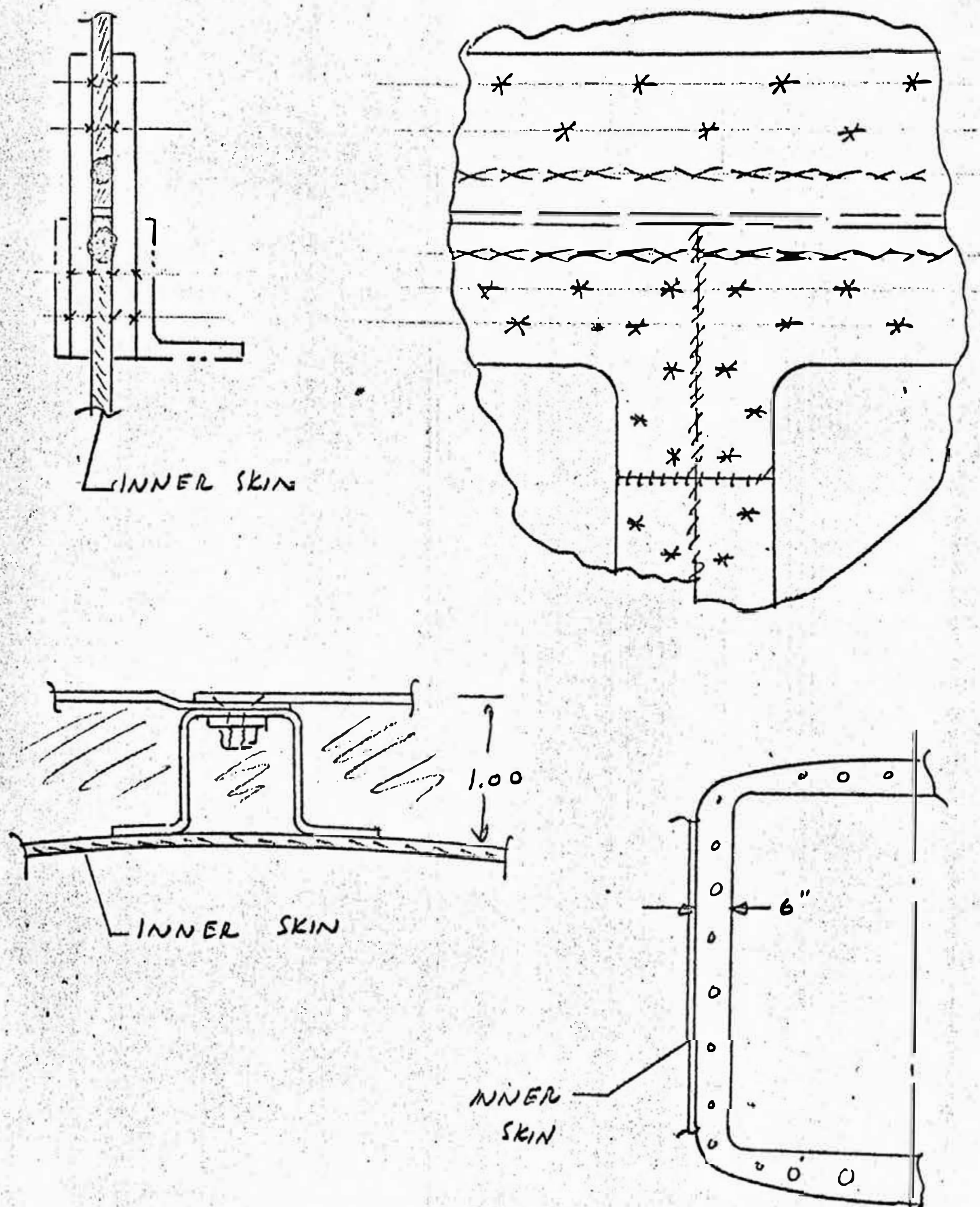
Peak electrical loads in shadow by 9000 W/HIC Silver-Cad. batteries. Silicon solar cells

Major Characteristics

AVERAGE LOAD	-
light cycle	800 W
dark cycle	400 W
Nominal Voltage	28V DC
Subsystem power	
Electronic	400W
Electrical	200W
Life Support	200W
Solar Cell	
Capacity	1.5 KW
Area	284 ft^2
Weight	242#
Efficiency	11%
Temperature	57° C
Batteries	
Silver-Cad.	9000W/Hr
Weight	262#
Charge (14 days)	225 cps

SPACE LAB MODULE WEIGHT

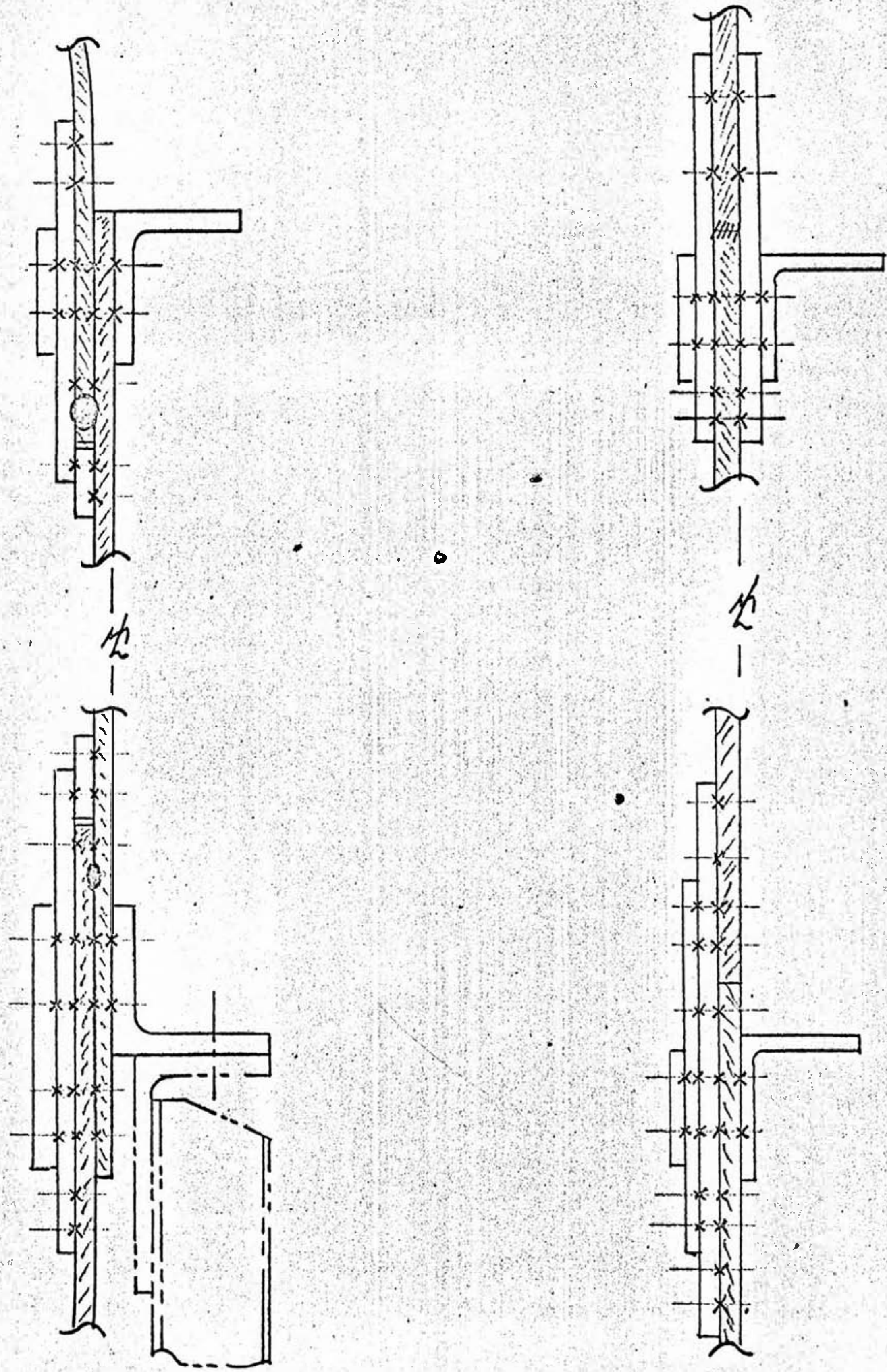
	Weight	Pounds
STRUCTURE		2106
Fairing (Jettisonable)	170	
Adapter	321	
Lab (approximately 525 ft^3 gross)	538	
Insulation	210	
Air Lock	140	
Furnishings	79	
Solar Cells	242	
Batteries	262	
Solar Cell & Battery Installation	144	
LIFE SUPPORT SYSTEM		
Food and Containers		1058
Water and Containers	209	
Gases	380	
System Equipment (Ventilation, Thermal, Galley, Environment)	130	
System Control (Humidity, Odor, CO_2, Illum., Sanitation, etc.)	298	
Emergency Equipment	41	
ELECTRONIC SYSTEM		218
Communications	5	
Telemetry	47	
Television	25	
Antenna System	10	
Tape Recorder, etc.	80	
Lab Instrumentation	51	
DOCKING SYSTEM	100	100
Motor, Gears, etc.		
ORIENTATION, GUIDANCE AND THRUST SYSTEM		780
Fuel	452	
Equipment	328	
MISSION PAYLOAD		2238
TOTAL		6500



TYP LAB SECTIONS

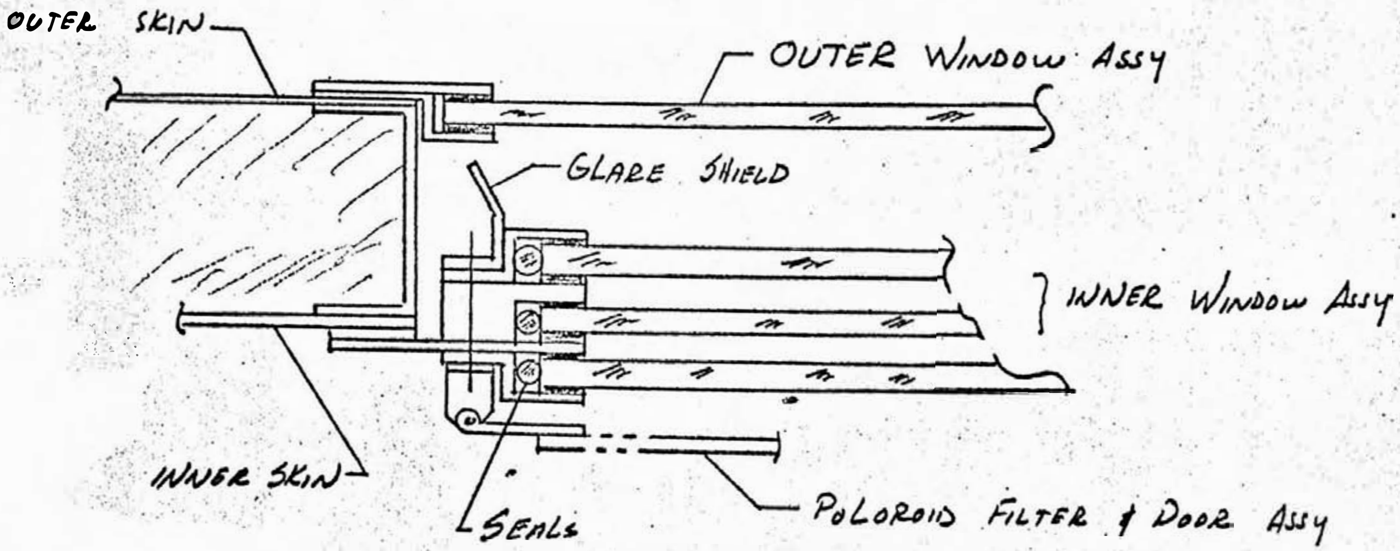
EQUIP SUPPORT RIBS

fig I

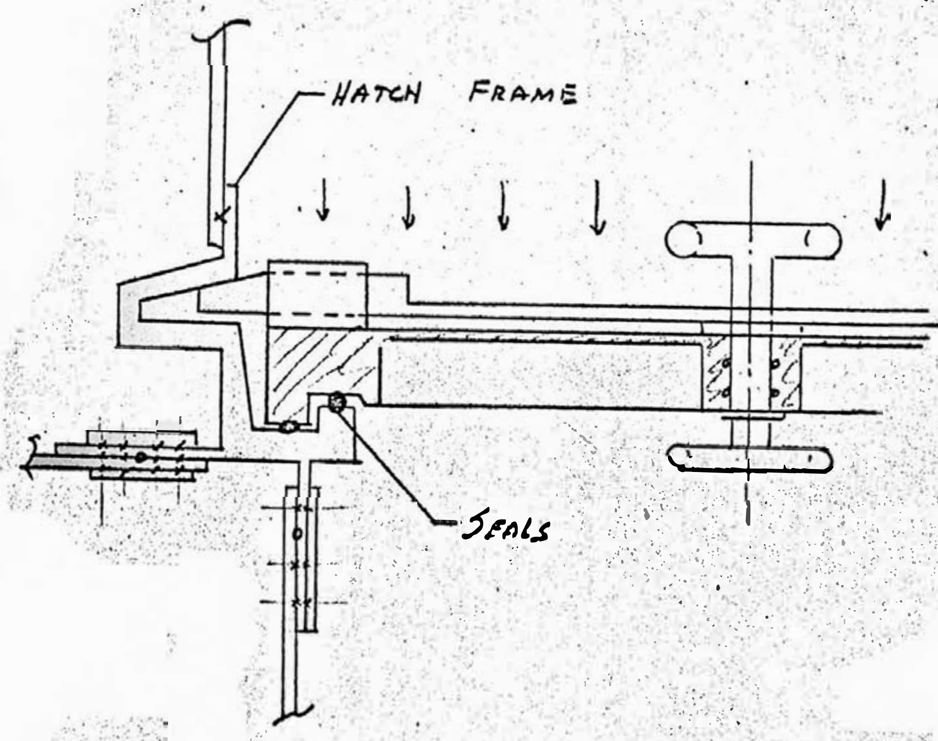


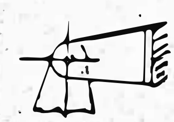
TYP ALTERNATE SECTIONS

fig II



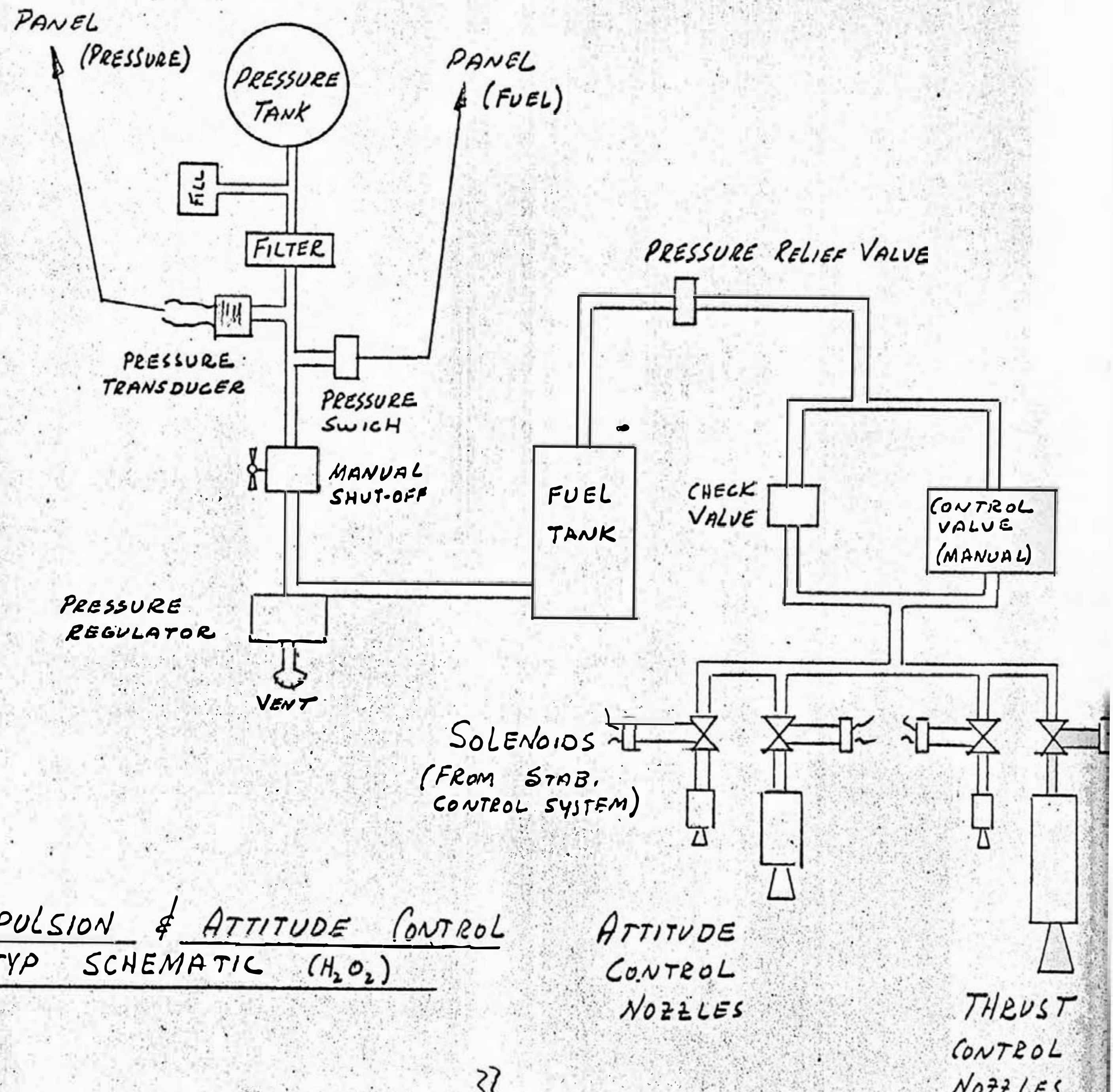
TYP WINDOW X-SECTION



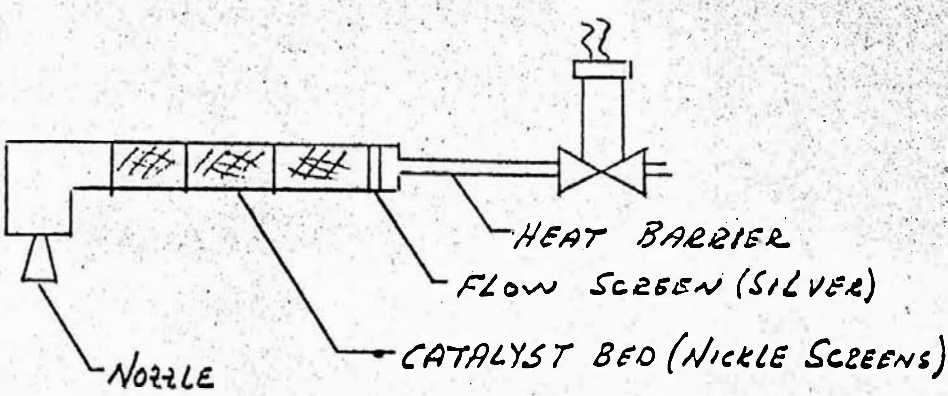


TYP SECTION - AIR LOCK DOOR

fig III

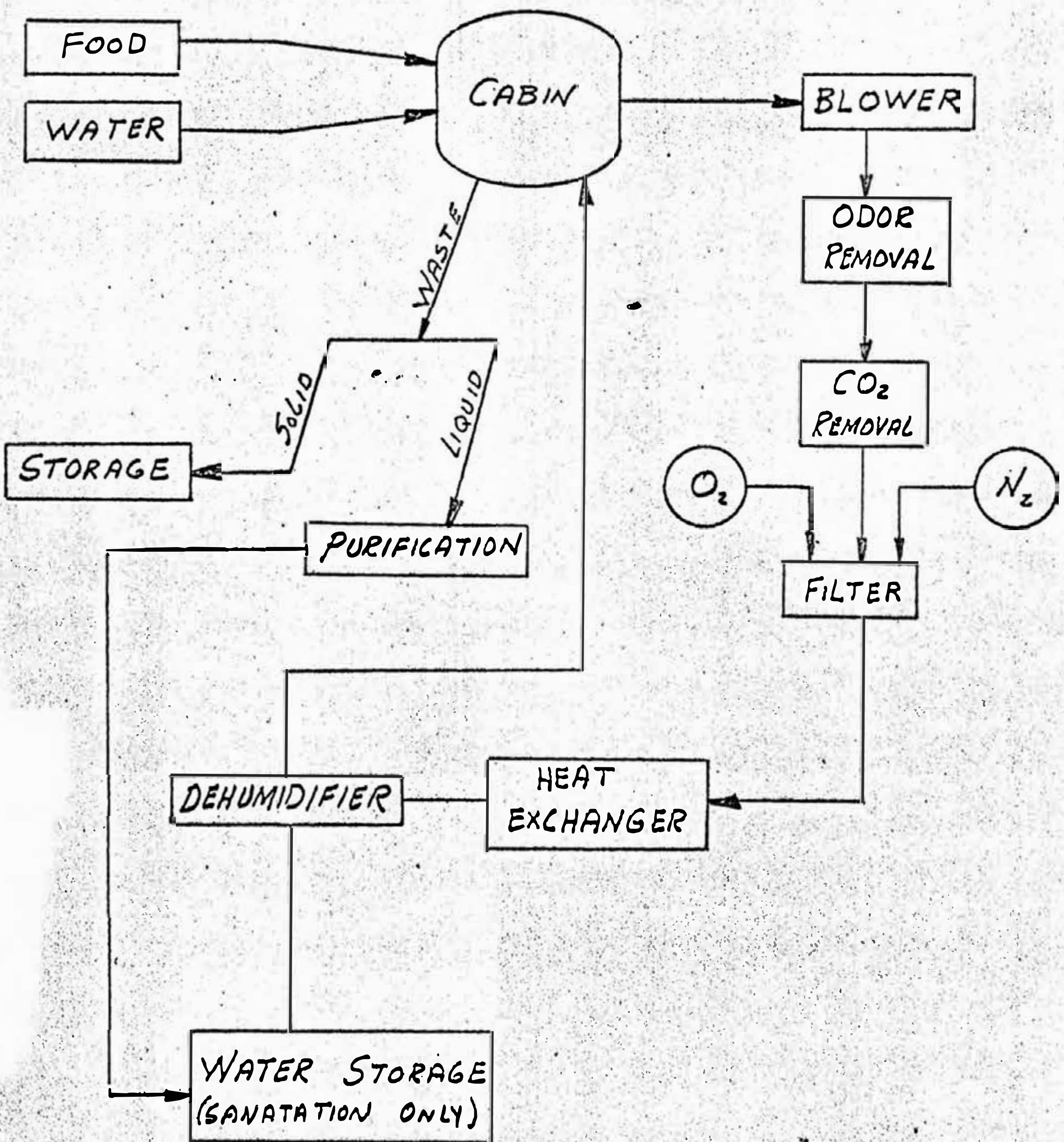


PROPULSION & ATTITUDE CONTROL
TYP SCHEMATIC (H_2O_2)



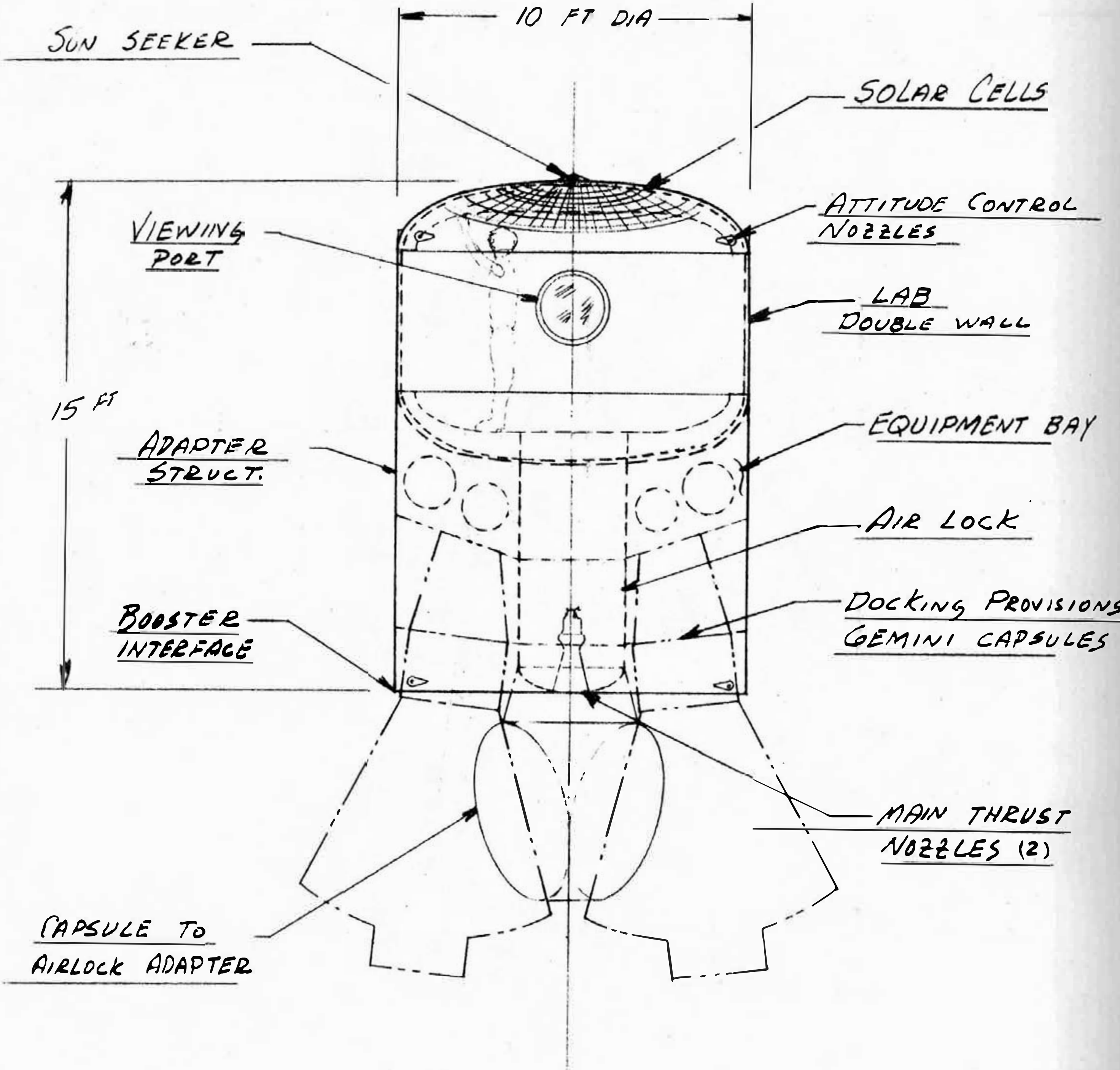
TYP THRUST CHAMBER

fig IV



TYP ENVIRONMENT SYSTEM SCHEMATIC

fig V



SPACE LAB

GEMINI RENDEZVOUS

fig VI

1/50

APPENDIX C

CONCEPTS FOR ORBITING SPACE STATIONS

Many concepts for orbiting space stations have been derived, from single spheres, to swinging dumbbells, to various toroidal station concepts. For meeting operational needs, some of these are suitable and some are not. The operations described in this report can be supported by any of the concepts which meet the fundamental requirements listed (Section II-1). The one shown was used as an example simply because it was one concept which did meet the requirements and for which the necessary definitive design data was available to permit specific interface of its operating and other characteristics with the other operations involved in this analysis. This does not mean that other concepts would not be satisfactory, or that the one used was necessarily the optimum.

In order to more clearly see what concept or concepts might represent a satisfactory or optimum approach for this type of operations, it is necessary to look at the broader overall picture of space station ideas as they have developed to date, and see what has been done with them, and what has been found out so far.

In looking objectively at the broad picture of space station considerations to date, we note first of all that for operations of this kind, involving lunar or interplanetary exploration or other enterprises, from the beginning of serious treatment of such matters the space station has commanded a key position in the thoughts and analysis given the problem -- especially by those who probed most deeply, and devoted significant portions of their lifetimes to its study. This prominent position given to space stations throughout the work of the first-half of the twentieth century (by Oberth, Pirquet, Noordung, vonBraun, and others) signified its inherent basic importance to such operations, -- and this standing has persisted throughout this period of growing understanding and attention to the fundamentals of space flight right up to the present time.

Before taking a look at the recent past and probable future activities concerning space stations, it is probably first worthwhile to take a brief analytical look at the trends of the past, so that we can consider adequately what lessons the wisdom of time has to offer in this regard. We first note that one of the early factors recognized was the absence of natural gravity, and the effect of this on the postulated occupants and concepts. From this, we observe the idea of rotation (often of a wheel-shaped arrangement) creeping in very early; and for the last several decades, every treatment of stations considered the zero-g situation, and the associated question: "to rotate, or not to rotate!" And we notice the overwhelming preponderance favoring the rotating wheel or equivalent concept. The single most prevalent trend has been the embodiment of some form of a torus, at least for interim space stations.

Another, and much more obvious, consideration from the very start was the recognition of the vacuum environment, and the corresponding need for a completely closed, artifically maintained environment, - one which was either fed by a "pipeline" of supply carriers, or one which was virtually self-sustained. Most investigators recognized and drew on the basic similarity to submarine crew environmental requirements, utilizing the growing knowledge and experience in that field to help define corresponding requirements of space vehicles and stations.

A great many other significant observations can be made from the work of these early investigators (many of which spent a major part of their lifetime working in this area), but probably the most significant has to do with the almost universal recognition of the logical need for a space station in order to carry out any activities beyond near earth orbits. It was generally concluded and accepted by these experienced investigators that for deep space, lunar and interplanetary endeavors, it only made sense to use space station terminals for these operations, unless some breakthrough beyond chemical rocket propulsion developed. And even then, Oberth and others were aware of the potential nature of far advanced propulsion concepts, such as ion propulsion, for which the first engineering concepts were treated in publications in 1954 - by Romick and Stuhlinger (independent studies with simultaneous release); and it was recognized that space terminals were even more essential for advanced vehicles using these higher performance propulsion principles and devices.

Such was the foundation laid down during the first half of the twentieth century, from which the current situation arose. For at mid-century there was developing a significant blossoming of activity in this field, to which we now turn our attention.

At this point, although we were technologically ripe for moving ahead (with not only the above theoretical background, but the practical foundation of the growing missile developments, jet aircraft, V2, captured German research and development data, Viking, Bumper, Aerobee, Space Medical research, etc), political realities of the time acted to slow the start being made. This resulted in an initial significant period of hesitation, during which all new technological ventures in the U. S. were stopped - the ballistic missile program and the space flight program were canceled (in 1948) and the thermonuclear program held in abeyance, to be resurrected in hesitant succession during the first half of the new decade (1950-1955). But the onrush of knowledge and technology could not be held back. The USSR was moving steadily ahead on all these fronts, and the surge of technical activity erupting in the USA from about 1948 on is clearly shown by the attached chronology. Resulting publications of findings were beginning to come out in England, Germany, and the U.S. during 1949, 1950, and 1951; and by 1952 this developed into a surge that continued to expand right up to the present time. That this consistent attention to space station concepts kept such steady representative pace with all the associated growing space technologies shows the compelling power of the idea, and the wide recognition of its key place and essential nature in space flight development.

During this period, the idea of the rotating torus remained in the forefront, and the idea of using flexible coated fabric structure that could be packaged for launching and erected to full volume in orbit by simple inflation with the artificial atmosphere required gained increasing attention, - starting with vonBraun, who made the first widespread presentation of the idea. At Goodyear Aircraft (which was probably in the best position to know the problems and potentials, and how to develop a dependable structure of this type) this approach was being studied, along with light weight rigid metal concepts for broad-based comparison purposes. Early publications by Goodyear Aircraft treated mainly the latter, with orbital assembly, in order to give a generalized long range picture of potential future space operations involving stations (see chronology). But, in the meantime, Goodyear was carrying out studies and research to determine the true potential of the inflatable structure approach for toroidal space stations and other applications - with particular interest centered on its potential for expediting and enhancing the initial steps in space exploration utilizing the space station advantages.

At the heart of this endeavor was the determination and development of suitable characteristics for such materials and structures in space, and this necessitated a considerable research program which was reaching substantial proportions by the time of the true dawn of the space age (1957-58). As soon as NASA was formed, Goodyear joined forces with them in this effort, in order to pool our R&D results and objectives.

This resulted in a much expanded program of research and development both at NASA (mostly NASA-Langley) and at Goodyear. Some of this was contracted, but a major share was in-house supported, both at NASA and at Goodyear, with extensive co-ordination of results. Other companies were also beginning to carry on some studies and research in this direction. The overall result was a considerable further expansion of activity, as well as equally significant advancement in both understanding and in capability to provide suitable structures of this kind. (These unfolding steps in the developing picture are detailed in the chronology appearing at the end of this discussion).

This space station work ultimately included much work in fabricating and testing of actual structures and the structural materials. This included space environment tests of all kinds (vacuum, solar radiation, particle and e.m. radiation, meteor penetration, vacuum deployment, etc) and involved construction of a variety of scaled models and full sized structures of various kinds in order to explore a host of problems (including many not even known to exist) to be certain that the many practical, unavoidable problems were uncovered and resolved. This, along with the other paralleling space development work and programs going on simultaneously has brought us to the present situation, which may be described as follows:

THE PRESENT SITUATION:

Currently, we seem to have generated two main questions, which are the direct result of the activities described above, that concern our space planners and everyone else involved. These are:

1. Is it desirable or necessary to start with a good, adequate interim station concept; or should we go direct to a more advanced, larger scale version?

2. Should they be flexible inflated type structures, or rigid metal (or plastic, or both) type structures (along with the question of orbital erection or assembly)?

To a considerable degree these two questions are inter-related, as both are mutual trade-offs in nature against the inescapable boundaries formed by the characteristics and capabilities of available launch vehicles, and the corresponding time-schedule to be followed. It seems that these inter-related questions and considerations have forced those concerned generally into two camps:

1. The first camp, believing that flexible coated filament or fabric type structures are unsuitable, finds that a satisfactory alternative rigid structure (without resorting to extensive untried orbital rendezvous and assembly) is sufficiently large and heavy (with necessary equipment, etc. included) that a Saturn C-5 vehicle is required for launch; therefore, it cannot be used before the 1967-70 period (or perhaps beyond); and

2. The second camp, relying on the results of these R&D test programs to show that suitable inflatable type structures can be employed, believes that a suitable fully functional useful station of this type can be built of such size and weight that it can be launched with available Titan II vehicles, and serviced with Gemini; and that it should therefore be developed and put into use immediately, thereby paving the way both for greatly expanded space operational capability and for natural evolution to better future stations.

This latter approach, if feasible, would make possible their use by late 1964 or 65, thus giving an indicated 3 or 4 year lead over the other, or large rigidized, approach. There is also a third viewpoint, which holds that neither of the above is correct, and that a small, minimum experimental unit (such as a simple cylindrical vessel) is needed first and should therefore be built and launched as soon as possible. This might not provide any gravity effect, or might utilize the swinging counter-balance (final booster stage) for this - essentially the Krafft Ehricke "outpost" concept (See chronology). Of course, there is also the Russians, who (though we certainly don't know for certain) give every indication of a strong belief in (and actual plans for) going directly ahead with orbital rendezvous and assembly to provide a suitable space platform in the next couple of years from which to carry on their lunar landing and exploration program.

We can see that the whole question posed by the above viewpoints seem to hinge on the suitability of fabric type structures. (Rendezvous cannot be the question, since it is required in every case.) These differences appear to arise from a lack of information (imagined or real) on the characteristics of fabric type structures in the space environment, with most emphasis perhaps on their meteoroid penetration resistance characteristics. It was an appreciation of the fundamental importance of these characteristics which prompted Goodyear several years ago to embark on an extensive research program to answer these questions. The results to date are as follows:

1. Micrometeoroid Penetration Resistance

   The results of the test program show that flexible type structures have equal or superior resistance to that of rigid structural materials of equivalent weight. This is what theoretical consideration of the molecular physical mechanism of micrometeoroid penetration had led us to expect, but the problem is complex, and testing was an absolute necessity. There has been a considerable temptation, because of these complexities, for many people concerned who did not have such results available, to resort to more superficial examination and conclusions, and establish unjustifiably firm opinions as a consequence. At any rate the tests indicate that the GAC developed bumper and structure performance should be quite satisfactory in this regard.*

2. Solar Radiation Resistance

   The solar radiation research and test program for flexible materials has been going on for many years, and satisfactory results have been obtained for practically every application pursued. This works out particularly good for the bumper-structure combination designed for the space station application, and is especially advantageous for thermal balance.

3. Particle and Electromagnetic Radiation Resistance

   There are two distinct aspects to this type of radiation effects: a) The effect on the long term material stability and structural integrity, and b) the effect on occupants or equipment inside. With regard to the first aspect Goodyear began a research and testing program many years ago when the radiation facility was installed at the Goodyear Research Center, and as a consequence have found that this problem can be handled quite satisfactorily. With regard to the second aspect (effect on occupants, etc) it has been found by research at Goodyear and elsewhere that these non-metallic structures are superior both with regard to capture characteristics and secondary (scattering) characteristics, (especially the latter), although the importance of this superiority should not be over-rated. Neither type structure (metallic or non-metallic) affords sufficient protection for intense solar flare conditions, so in any case, separate localized protection must be provided. However, for space station operation below the Van Allen

*See GER 10663; Goodyear Aircraft Corporation, Akron, Ohio; Aug. 1962 (unclassified)

3. (continued)
belt, where tolerable, long-term cumulative effect is the main consideration, then the favorable scattering characteristics which the non-metallic structure provides is of considerable importance.

4. <u>Structural Integrity and Reliability</u>

There are two aspects to the structural integrity and reliability question also. The first involves design for assurance of adequate strength throughout its lifetime, and the second involves quality control adequate to insure that this will be realized. Unfortunately neither of these can be considered in simple straight-forward terms, for non-metallic, coated fabric type structural engineering and fabrication is partly a science and partly an art. It is like many other products where high standards of reliability and safety are essential,—automobile tires, for example. There are quality tires available for purchase which will not go flat from puncture or blowout, and which can be depended upon to give high performance and satisfactory service - their manufacturers guarantee this, and it has been found that people are justified in entrusting their lives to the quality of such tires. Yet it would not be possible to write a text-book containing what is necessary to acquire and train people in a short period of time so that a previously inexperienced intelligent group of people could be set up to develop and produce such tires in a reasonable length of time without the guidance of an experienced group. To some extent this is true of many things, but it is certainly fully true of the type of space structures we are considering. This has caused much uncertainty on the part of people not experienced with such structures, who are understandably reluctant to accept them as suitable, regardless of the evidence that can be offered. But this same type of inexperience does not prevent them from using tires, or others from using airships, etc., as far as their consideration of structural integrity and reliability is concerned, even though they could not themselves build them. The key factor seems to be that there is an industry in existence which has demonstrated its capability to produce such products with assurance of required high quality. At any rate, the situation with regard to such structures for space stations has been found to be as follows:

First, with regard to design for the required maintainable strength; this proved relatively straight-forward to resolve, for the required basic structural characteristics for application to torroidal space stations lie (approximately mid-way) between those of tires and those of airships, in almost every important regard. It was necessary therefore, to conduct an extensive research and testing program to achieve and demonstrate the properties required to permit the proper engineering and design of the proper structures. This has been done and the results obtained have been found to be satisfactory.

With regard to the quality control requirements, these are not as straightforward. The materials are completely dependable if properly fabricated, but the fabrication processes are quite sensitive to a number of environmental factors and to subtleties in the fabrication techniques, composition of ingredients, etc. Additionally, the means for inspection without destruction are limited, and for this application, the product dependability requirements are extremely high. But these things are all also true of airship structures, as one extremely pertinent example. Regardless of difficulties, the entire huge envelopes and all their attaching parts and details have to collectively possess and maintain a structural adequacy such that, for all of the many hundreds of airships that saw service (and throughout their service lifetime), they could be depended on not to fail in any manner during flight that would endanger the ship or its occupants, and depended on to the same degree that prevails for any other aircraft. That such standards for these types of structures can be, and were, met has been amply demonstrated in the case of the airships, and the same standards must be held for space station structures, for very similar reasons.

There are two approaches being followed to insure obtaining such standards of quality and dependability. One is the provision of the full system and techniques of quality control that have been acquired to date from all past experience, particularly that in airships where the same high standards are required and have been successfully proven and demonstrated. The second is a continuous program of extending the capabilities of inspection. This is being steadily improved, as more research is applied to this problem. It should be remembered that, while the airship quality control proved quite satisfactory, without the benefit of most of the recent improvement in inspection capabilities, our goal remains to obtain a 100% flaw detection capability - a goal that is seldom if ever quite reached in any field. However, work is and will be continued in this direction. Recently new techniques have been developed by GAC and applied to the filament-wound plastic rocket cases for Polaris and Minuteman which represent a significant step forward, and can be applied to space station structures. But regardless of where such improvement efforts take us, it is absolutely necessary that, with the methods at hand at any given time, satisfactory quality control must be applied to meet the rigorous objectives and requirements involved - and the ability to do so consistently for these types of structures has been fully demonstrated, and the capability to do so thereby established.

There are other points of comparison but they are less critical, - the ones treated above are the most critical, and probably the most widely misunderstood. But the importance of these things, and the simple facts they portray, cannot be over-emphasized, - for the differing chains of conclusions set off by lack of information and consequent wrong assumption (even regarding just one of these aspects, such as micrometeoroid penetration resistance) often leads in widely differing directions, - and this probably accounts mostly for the differing position taken by the various camps as described above. Therefore, such information is of key importance in making it possible to obtain agreement in judging correctly what is the right course to follow for maximum achievement and assurance of success with a minimum effort and at the earliest time.

In the meantime, activities over the past year have been stepped up in regard to other concepts and approaches with particular emphasis on a large segmented torus design of erectable rigid design. This approach, usually considered with a total diameter of 150 ft., would be a later, more sophisticated capability; and involves hinging of segments so they can be packaged folded parallel to the longitudinal axis of a large launch vehicle such as the C-5, and then unfolded in orbit. This permits all equipment to be included in final installed position at launch, but gives rise to sealing, mechanical, and other problems which are currently under investigation. Other concepts are also being examined, and much has been published regarding this work (e.g., see Sept. 1962, Astronautics magazine).

In the context of this picture, the activities of several aerospace companies and government agencies pertinent to space station designs, requirements, and plans are continuously increasing. Much of the basic equipment requirements for space stations are meanwhile being advanced toward fulfillment by the requirements of other programs with similar needs, such as Gemini, Apollo, Dynasoar, etc. One of the most fundamental requirements for space station use is rendezvous capability, which is a main target of the Gemini program and is in turn moving along under the impetus of Apollo requirements for the same thing. (The target date for this is 1964.)

So many of the necessary ingredients for achieving an actual operating space station continue, as of now (late 1962), to move forward, probably driven by the underlying basic logic and importance of space stations to significant progress and accomplishment in space flight.

This then, is the general history of space stations up to the present time, with a consistent place and a consistent share in the total history of all space flight development throughout its course up to this point. It is doubtful that this trend will change.

A LOOK AHEAD

No history, however, is complete without making some logical projection into the future based on the trends and picture established by that history. It seems logical therefore to take a brief look ahead, and see what is indicated.

In looking ahead, one thing appears certain,--and that is that the growing attention to space stations will continue until one is placed in operation. From that point on, their use will probably grow rapidly, limited only by how fast suitable improved operational earth-to-orbit ferry vehicles are developed to support them. This in turn will pace the rate in which their expanded usage for support of lunar, interplanetary, and other operations can be developed. The biggest question immediately before us, therefore, is what course will be taken to move from where we are to the point of putting the first stations up and into operation, and when this will take place.

We can probably set two boundaries to this path at least, - the one representing the most direct and rapid advance toward earliest possible accomplishment, on the one hand; and on the other, the most cautious, slow-moving path that is likely to be followed, and the longest time it is likely to take without overlooking the basic impetus and attention the space station idea has consistently commanded thus far.

For the shortest pathway to a useful station, taking into full account what has been done thus far, and what remains to be done, it would appear that the quickest way to get a useful station up for supporting space operations, and one which has provisions for all the minimum features that might be required, would be to build an inflatable torus station, so that a station meeting the presumed minimum requirements could still be launched with present boosters. Since the basic research work for such a station has been done (as outlined above), and since full-sized stations of this type have been built for ground testing, and since one has been designed taking these results into account and providing the necessary minimum operational features to work with Gemini support, with a launch weight of 6,600 lbs. (total); it therefore appears that such a station could be launched with a booster now flying - namely, the Titan II - and supported by Gemini and Titan II; and that preliminary test flights could be made using either piggy-back space on the Saturn C-1 R&D flights or on Atlas-Agena B's or Tital I's or II's. This prospect has been examined, and might proceed by the following over-lapping steps:

1. Initiate stepped up extension of testswith present GAC built stations, and, (simultaneously)

2. Build a 40 ft. ground test structure, and

3. Start development of a 40' flight test and demonstration torus station for launch on any of above listed boosters which are available.

4. Institute a system and subsystem design for first operating (manned) station.

5. Develop and (ground) test two manned stations.

6. Launch first manned station (unmanned at launch), deploy, activate, and check out systems from the ground.

7. Launch Gemini to rendezvous with station. Crew docks, boards, and checks station.

8. Second Gemini is launched, crew boards, and station is placed in full operation.

9. Station is supported by Titan II supply flights. (Added equipment for expanding the scope of operations is also carried up in this manner).

10. Additional crew and crew rotation provided by Gemini flights. (The Gemini vehicles also constitute continuous standby escape provision for entire crew. This station can accommodate up to 10 persons efficiently.)

An analysis of this approach has indicated that if it were authorized and vigorously pursued, the earliest date at which it could be in operation would be late 1964 or early 1965. In other words, even with the research that has been done, the structural development work, and the building and testing of the two stations that has already been done, it would still require approximately two years (plus any delay in starting the program or further slippage in the Gemini schedule) to develop, test, and launch a man-rated station of this type and put it in operation.

It is possible that by going to the simple cylinder space laboratory approach recommended by some that this could be accomplished a little earlier, but this is doubtful, for the following reasons:

1. For the flexible torus, with its present status it is quite possible that the structure would not be the limiting factor, as far as schedule is concerned.

2. The smaller cylinder would impose some problems (of less space, stability, etc.) not involved to the same degree in the torus approach, which would also have to be accommodated.

3. It is likely that sub-system procurement, integration, test, and installation would be the critical time path for either approach.

4. Even if none of these factors prevailed, it is doubtful that the Gemini would be ready for its part before this time (and it represents the first rendezvous capability).

The Russians, of course, by going directly to rendezvous with their Vostoks and orbital assembly may well reach this point earlier, but only because they laid their plans in this direction earlier than the U.S., and are therefore further advanced at this time (have launch pads available for successive controlled launches for rendezvous, etc.)

So it appears that the lower time boundary, or the earliest possible time at which the U.S. could place a useful orbital station in operation if vigorously pursued, is right around the end of 1964 or beginning of 1965.

Now, considering the other reasonable boundary, or probable upper limit (latest)time) for U.S. achievement of its first orbital station in operation, the following conservative course might be followed:

Preoccupation with Apollo problems and associated budget problems might restrict NASA space station activities to the area of studies for the next year or two, and military space charter restrictions might limit the Air Force to a similar scope. If so, some continuing work would be carried out by industry, and by the end of two years, a point should be reached where either the Air Force, or NASA, or the two together in a joint cooperative program, would move into a limited experimental hardware stage. It is likely that one or two test launch models might be procured for launch within two years from then, or in late 1966, perhaps. Then it should be possible to develop a manned station for launch in about 3 years, which would be late 1969 or in early 1970.

If the various concerns regarding flexible inflated fabric type structures mentioned earlier caused them to be rejected for this purpose, then a rigid operationally useful concept might well be pursued which required a Saturn C-5 for launch, such as currently favored by NASA as represented by the self-erecting concept on which North American Aviation worked under NASA contract. If so, the C-5 launch vehicle availability might dictate a similar pace to that indicated in the preceeding paragraph, since it appears doubtful that one could be diverted until after the Apollo mission moon landing and return had been achieved, and this may be in 1968 or 69. Therefore, 1969 would be the most likely time when a space station could first be launched with a C-5.

Thus, it appears that the first U.S. space station launch and start of operation will occur sometime during the second half of this decade - about the first of 1965 at the earliest, and by early 1970 at the latest. It seems somewhat unlikely that either extreme will actually be followed, but rather that some middle course in between is more probable. For example, while the flexible structure approach would permit the earliest launch, and the available facts would seem to justify its use, it is likely that some time must pass before people become familiar with these facts or become willing to accept them and authorize a program for developing and launching such a station. Thus, the start would most likely be 6 to 18 months away. If a test and demonstration could be made, it would probably assure starting by that time. Then, unless priority and urgency were given the program (which would be a considerable departure from present attitude) another 2 to 2-1/2 years would probably be required before launch and operation. This would make it come some time in late'65 or early 1966 as the most likely time for such a station to be placed in operation if such an implementation route is followed.

Thus, for the first (U.S.) space station in operation, if the flexible, expandable, inflatable type is used, early 1965 is possible, and sometime in 1966 much more likely in the light of present circumstances. If a rigid type is to be developed, it will probably be nearer 1969. However, the trends evident in the space station history to date strongly suggest that pressures for providing a station as soon as practical will grow, and may well constitute a considerable motivating force toward acceptance of the flexible structure approach so that a station can be launched at least by 1966. In the meantime, Soviet activities may intensify the tendency toward this course.

If this is indeed decided upon and accomplished in this way, then its availability can be of substantial benefit to the Apollo program as a significant backup capability to enhance training and developing operating techniques in space, and could help the program considerably by helping to more quickly resolve and overcome the unexpected difficulties that are almost certain to be encountered. Beyond this it can later serve as a space terminal and thus up-grade the logistic efficiency and flexibility of Apollo follow-on lunar operations.

Thus its operation by about 1966 gives a space station a strong prospect of rendering a high degree of service in support of our national programs. It can be used for many other purposes, including, after the initial Apollo goals are reached, a considerable enhancement of the start of interplanetary exploration. So in this way the service it would render could be of significant value in enabling us to be moving swiftly and successfully ahead in space endeavors by the end of this decade.

The most outstanding fact in all this is that, laying prejudices aside, the purely engineering information available for consideration strongly indicates that if we were to embark on a course of action aimed at earliest attainment of a truly operational initial space station capability in being utilizing the inflatable torus approach (as represented by the design concepts presented herein) then

1) it should work satisfactorily, and

2) it is the only way that offers substantial assurance of affording required service this early.

This is simply because it is the only technically verified approach which gives the required capacity, capability, and simplicity that can be launched with present boosters.

OVERALL SPACE STATION CHRONOLOGY

1897 - Kurd Lasswitz (Germany) - First idea and need for a space station outlined as the key to space travel.

1903 - Konstantin Tsiolkovsky (Russia) - Develops technical basis which makes space stations possible - the multi-staged rocket concept.

1923 - Hermann Oberth (Transylvania) - First sketchy theory of a station in space

1928 - Guido von Pirquet (Vienna) - Expands theory of space station to utilization for interplanetary launch.

1928 - Captain Potoznik (under the nom de plume of H Noordung) - works out concept for synchronous triple station, using a spinning wheel configuration.

1928-30-

Space flight organizations formed in Germany, England, U.S., U.S.S.R.

Germany: VfR (Verein für Raumschiffahrt - Society for Space Travel)

Russia: GIRD (Gruppa Isutcheniya Reaktivnovo Dvisheniya - Group for Investigation of Reaction Motion)

England: BIS (British Interplanetary Society)

U.S.: AIS (American Interplanetary Society, later changed to ARS, American Rocket Society)

1929 - Oberth's enlarged book shows space station construction principles

1930-1940 -

These societies publish technical journals including many articles and papers by competent investigators. Many space station concepts and configurations appear, several centering on the idea of rotation to provide synthetic gravity.

1946 - Several significant publications on space flight, including rocket vehicles, orbital flight, vehicles, space stations, lunar flight, etc.

- Willy Ley - Missiles, Rockets and Space Travel (new edition, significantly expanded)

- Rand (actually the Douglas A/C R&D forerunner group) space ship preliminary design study.

1946 - 1952 -
Significant space flight studies carried out by people in Army group at Fort Bliss - Huntsville (vonBraun); Bell A/C (Krafft Ehricke and Kurt Stehling), Goodyear A/C (Romick and Knight), Martin Co (John DeNike and Elliott Felt), and G. E. (Bob Havilland).

1949 - Rotating space station concept with stationary docking provision and analysis of system and subsystem requirements (Ross and Smith)

1951 - Generalized toroidal (bicycle wheel) space station concept presented by vonBraun (in Space Medicine, University of Illinois Press).

1951 - Segmented ring space station concept - H H Koelle

1952 - Inflatable fabric toroidal space station concept and utilization potential published by vonBraun, in Das Marsprojekt (published in Germany, later in U.S. in 1953) and in Collier's article and later book edition by Cornelius Ryan in 1952.

1952 - Early concepts of integrated Ferry Rocket system and Space Station developed at GAC, using building block space station concept with concentric toroids for gravity wheel.

1953 - Treatment of principles of orbital assembly, as required for space station or space launch - in book on Space Travel by Ken Gatland and Anthony Kunesch.

1953 - Space Vehicle Guidance and Control and Rendezvous principles published and presented at National ARS meeting Space Flight Symposium (Romick)

1954 - Preliminary design analysis of Ferry Rocket Vehicle System published by GAC (Romick, Knight, vanPelt)

1954 - First of several space station concept presentations by Krafft Ehricke (Convair) - simplified 4-man station leading toward later "outpost" concept

1955 - Preliminary design analysis of Space Station concept with building block expansion provisions, including concentric toroids for gravity wheel published by GAC (Romick)

1956 - Publication of integrated space station and Ferry Rocket concept (called Meteor-for Manned Earth-satellite Terminal evolving from Earth to Orbit ferry Rockets) by GAC and presented at Rome (Romick)

1957 - Preliminary design study for initial minimum-sized integrated Ferry Rocket and Space Station system published by GAC and presented in Barcelona (concurrent with Sputnik I launch) - (Romick, Knight, Black)

1957-1958

GAC review studies to get space flight going; generate UTV (which became Scout) and MSF Phase I (which became Mercury) concepts, and made wide presentations, including MSF Phase II (Meteorite aerospacecraft carrier) as Ferry Rocket (Yellow Peril) and space stations (to Air Force, and NASA,—when formed).

1958 - Lockheed space systems study published, including provisions for toroidal space station, to be assembled with help of "astrotug".

1958 - Atlas concept for orbital system with forerunner of Centaur - cable connected separate bodies could be rotated about mutual CG for artificial gravity effect - Krafft Ehricke (Convair)

1959 - Initial space station research contracts with NASA awarded to GAC.

Decision to build 30 foot GAC station (by Astronautics - Romick, Peterson - in November).

GAC carried out integrated overall space station system study, and enlarged concepts for other orbital space operational elements, such as work cells, transfer units, repair cells, etc.

12-man station concept by Boeing for utilizing orbiting booster elements as components for orbital laboratory.

1959 to present - NASA carries out in-house and funded space station studies and research program (mainly at NASA-Langley)

1959 - Expanded "outpost" concept, showing growth potential to larger stations by Krafft Ehricke, Convair.

1959 to present - Air Force carries out SR studies and related considerations of Military Test Space Station concepts, potential growth and utilization (currently known as MODS).

1960 - GAC decision to build 30 foot space station approved.

GAC proposed 1-man Mercury space station to NASA-Langley.

Martin Company carries out various orbital operations study, including orbital hangar for maintenance.

Manned Space Station Symposium, jointly sponsored by IAS, NASA, and RAND Corp., held in Los Angeles April 20-22. Extensive treatment of space station technology, concepts, techniques, and general considerations (Report published).

GAC 30-foot space station construction started later in year.

1961 - GAC builds space station dynamic model and thermal balance model for NASA program, steps up contracted and in-house R&D work on space station structures and materials.

GAC builds 2 full-scale space stations: NASA 24 foot C-annulus and completion of GAC 30 foot torus.

Northrop space station launch platform study completed in 1961, published in 1962.

Bid on 100 foot (later became 150') erectable space station study for NASA (awarded to NAA).

1962 - GAC develops its station further, equips 30 foot station

24 foot under test at NASA; exhibit and promotion of 30 foot station

Development (by GAC) of space station operational concept, task force approach to Apollo and other national goals, and capabilities with presentations to NASA-Langley, NASA-Lewis, and NASA-Headquarters (D Brainerd Holmes)

NASA Space Station Office set up at MSC, Houston

Detail publications on space station, with emphasis on NASA (Berglund) erectable torus concept, much of it resulting from the work done by North American Aviation under contract to NASA.

GAC conducts operational and design analysis of integrated operational system utilizing space stations using the latest outputs of its research, engineering, and testing program, and derives corresponding test, development, and operational plans, schedules, cost and economic analyses.

Note: For the last 7 or 8 years, literature and activities of Soviets strongly indicate paralleling studies and developments toward orbiting platforms - with possibilities they are now further advanced toward implementation than the U. S. Indications seem to favor orbital assembly method.

MEMORANDUM

14 September 1962
SP-1362

To: W. N. Brewer, Manager
Astronautics Systems
Department 454-Plant G

From: Paul Hlebak
Astronautics Systems
Department 454-Plant G

Subject: Determination of Orbital Launch Windows for Lunar Rendezvous

Enclosures:
1) Figures 1 - 4
2) References
3) Nomenclature
4) Equations

For the lunar rendezvous mission, the orbital "launch or escape windows" are durations of time during which launches from an earth-satellite orbit are possible with a given propulsion system capability. The escape window width indicates the total time that orbital plane of the satellite (space station) and the lunar plane are in a proper geometrical position for the launching of the probe or vehicle.

Launch from orbit requires a consideration of many variables, both geometric and dynamic. In order to determine orbital launch windows, it is necessary to have a clear understanding and visualization of the 3-dimensional geometrical relationships of the space station-earth-moon system. Figure 1 shows the angular relationships for the satellite and lunar planes and the vehicle's trajectory. The coplanar geometry for the translunar flight trajectory of the vehicle is illustrated in Figure 2.

Because of the perturbational effect of the earth's oblateness, the satellite orbital plane precesses about the earth's polar axis in an opposite direction to the satellite motion. The rate of orbital precession (or regression of the line of nodes) for earth orbit is given by

$$P = \frac{10.05}{\left(\frac{a}{R_\oplus}\right)^{7/2}} \cos i \qquad \text{deg./day} \qquad \text{(Eq. 1)}$$

where i is inclination to the equator,

a the orbit semi-major axis

$R_\oplus$ the earth's radius

which is derived analytically from Potential Theory.

The moon travels in a slightly elliptical orbit of eccentricity $e = 0.055$ and hence for preliminary mission analysis the moon's orbit may be assumed to be circular. The plane of the lunar orbit is inclined by 5° to the plane of the earth's orbit around the sun (Ecliptic Plane), which is inclined by 23.5° to the earth's equator. The period of rotation or true month is 27.32 days, giving an average angular rate of rotation about the earth of 13.2 degrees per day. The period of precession of the moon's orbit is 18.6 years, during which time the inclination to the equator varies between a maximum value of 28.5° ($23.5^\circ + 5^\circ$) and a minimum of 18.5° ($23.5^\circ - 5^\circ$), as shown in Figure 3

For the rendezvous mission, minimum energy transfers can be initiated whenever the precessing orbit is orientated so that it will contain the moon at the time of arrival of the probe. That is, an optimum time for launching from orbit occurs when the resulting transfer trajectory will permit a rendezvous with moon at the line of nodes or intersection of the satellite and lunar planes.

In the event of a non-optimum launch time, additional propulsive energy above the minimum ΔV, must be expended in adjusting the launch plane angle or firing azimuth so that the resulting transfer trajectory will intersect the moon. This operation of varying the launch azimugh is commonly called "dog-legging" and it results in a rotation of the trajectory plane about a line joining the earth's center and the launch point. The amount of additional propulsion energy (impulsive velocity increment, ΔV) available for dog-legging determines the launch window limits, which indicate the tolerance on launch time deviations from the optimum nodal firing time.

Assuming impulsive thrust, the ΔV required to depart from the satellite orbit on a coast trajectory to the moon is given by

$$\Delta V_{\oplus,o} = \sqrt{V_b^2 + V_c^2 - 2V_bV_c \cos \Delta a \cos B_b} \quad \text{(Eq. 2)}$$

where V_c is the circular orbital velocity,

V_b the burnout velocity

B_b the initial flight path angle

Δa the "dog-legging" angle

The equations for computing Δ a and V_b are given in Enclosure 4.

Additional ΔV applied as retrothrust is required for the establishment of a circular orbit around the moon, as the vehicle always possesses hyperbolic velocity relative to the moon that must be reduced to circular velocity. Enclosure 4 contains the complete equations for computing the $\Delta V_{\mathfrak{D},o}$ retrothrust, which depends largely upon the burnout velocity (V_b) and lunar parking orbit altitude and to a lesser extent upon the burnout altitude (h_b), initial flight path angle (B_b), and the angle (u) between the trajectory plane and lunar plane.

For a non-precessing orbit (polar or equatorial), minimum energy transfers are possible twice a month. In case of a precessing orbit whose period of precession is about twice the moon's orbital period, the most favorable launch opportunities will occur six times in a 2-month period at irregular intervals of time.

This is illustrated in Figure 4 (From Ref. 1) which shows the variation with time of propulsion requirements (impulsive ΔV) for departing from a 300 N.M., 30^o-inclined satellite and establishing an 82 N.M. lunar orbit. In interpreting this curve, it should be noted that, although the B_b is 20^o, the ΔV - values given do not include the velocity increment required to deflect the V_b upward thru the initial flight angle of 20^o. For this representative example, the moon is initially at its descending equatorial node, which is taken as the zero longitude reference (See Fig. 1).

The minima on the curve represent the times when the moon is at a node of the lunar and orbital planes. These minima occur at irregular intervals due to the fact that the angular rate of rotation of the line of nodes in the lunar plane varies throughout the 2-month precession cycle. Determination of these times of minimal ΔV requires a trial-and-error iterative solution of a transcendental equation defining the spherical trigonometric relationships, which is given in Enclosure 4.

In Fig. 4, the width and frequency of the launch windows for a 1000 fps dog-legging capability (indicated by heavy dark lines) are shown to vary markedly with time. The maximum and minimum widths are approximately 2 1/2 days and 1 day respectively, and the maximum and minimum time lapses between the launch windows are 8 days and 1 1/2 day, respectively.

It should be noted that the launch window pattern is not periodic unless the period of precession of the satellite orbit is an integral multiple of the lunar orbital period. Thus, a 30^{o}-inclined circular orbit of 284 N.M. altitude would have its launch window pattern recurring identically every two months, since its precession rate of 6.6 deg./day is one-half of the moon's orbital angular rate (13.2 deg./day).

As mentioned in Reference 1, the ΔV values in Figure 4 were computed for the most favorable launch position (λ_b) on the precessing orbit. For each position of the moon at rendezvous (λ_D), an optimum earth orbit-lunar orbit coast trajectory is selected by specifying the launch or burn-out longitude (λ_D), velocity (V_b), and azimuth (a_b) so that the total ΔV requirement is a minimum. In place of V_b, the transit time (t) or trajectory central angle (θ) may be specified.

Included in Reference 1 is a catalog of earth-moon trajectories (excluding multi-body effects) in the form of parametric curves giving travel angle (θ) and burn-out azimuth (a_b) as functions of lunar position (λ_D) and burn-out longitude (λ_b) at 10-degree intervals of burn-out latitude, for lunar orbit inclinations of 18.5^o and 28.5^o.

For preliminary mission analysis, it is feasible to compute orbital launch windows with the aid of these curves and a slide rule, for the case of non-precessing orbits. If precession is taken into account, the procedure (indicated in Enclosure 1 - Part A) is more involved and requires a high-speed digital computer.

A simpler and more practical method of computing launch windows is to fix the transit time T, thereby, fixing the travel angle θ and burn-out velocity V_b, simultaneously. The procedure (indicated in Enclosure 4) also involves iteration, requiring a digital computer.

For logistic considerations, it is desirable to determine launch windows for fixed transit times, as the time of travel determines the weight of life support systems, the amount of evaporation of cryogenic propellants, etc.

Therefore, it is recommended that an orbital launch window determination program be written for the IBM 1410 Digital Computer, by the fixed-transit-time method indicated in Enclosure 4, Part B.

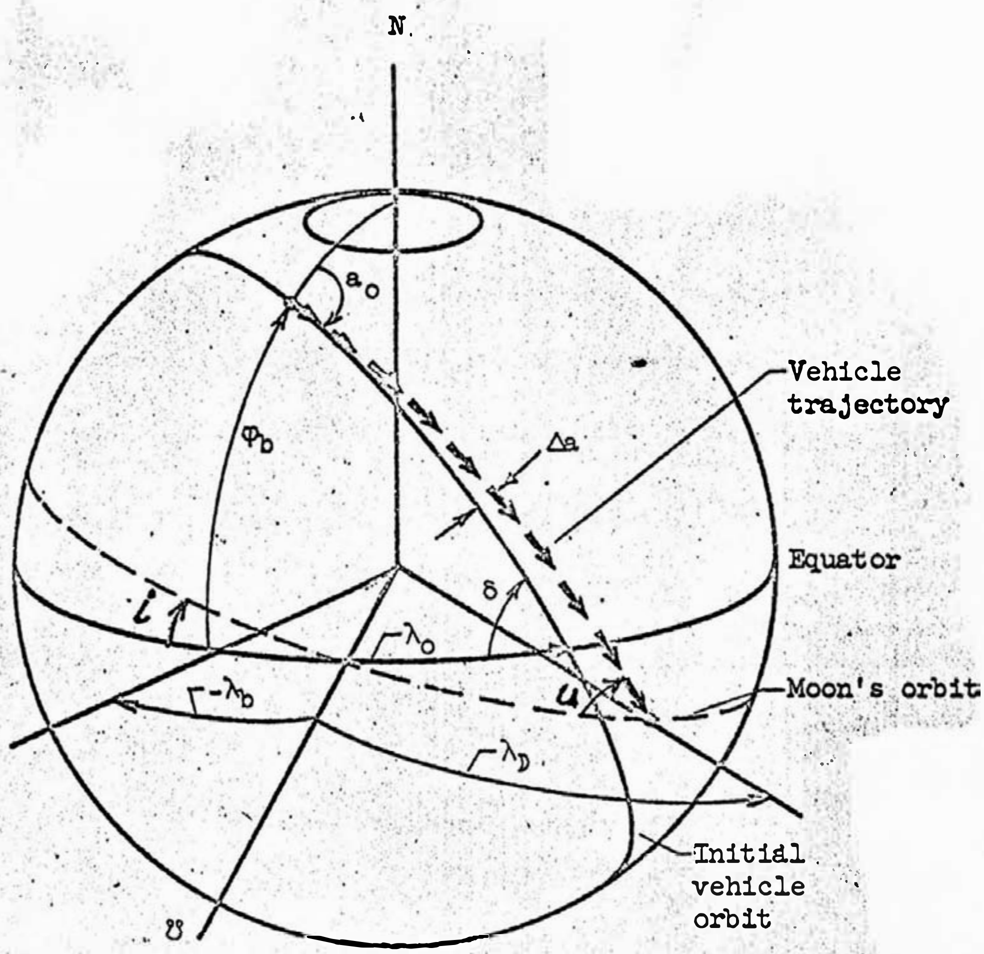
Such a program would be required in order to determine what configuration of space station orbits (of 30-degree inclination and 284 nautical mile altitude) would give a combined orbital launch window distribution as continuous as possible for a given propulsion system capability. In solving this problem, it is necessary to compute a ΔV distribution curve as Figure 4, for each space station orbit with initial equatorial descending node in the range, $0° \leq \lambda_{0,I} < 180°$ for a given initial position of the moon.

Paul Hlebak

Paul Hlebak
Astronautics Systems
Department 454-Plant G

PH/emg

cc: D. C. Romick
E. R. Smith
R. H. Emmons
C. J. Peirce
J. L. Altekruse
S. J. Pipitone
F. H. Kierstead
D. E. Campbell
N. L. Jeppesen
F. R. Gross
F. J. Stimler
J. Singer
F. A. Pake



a_o Azimuth of vehicle orbit at departure point
δ Inclination of vehicle orbit about Earth
λ_b Longitude of burnout point from ☋ (in equatorial plane)
λ_o Longitude of vehicle orbit's descending equatorial node from ☋
λ_D Longitude of impact point from ☋ (in Moon's plane)
φ_b Latitude of burnout point
☋ Moon's descending equatorial node
Δa Doglegging Angle

(b) Departure from vehicle orbit about Earth.

Figure 1. Angular relations for lunar trajectories.

ω = Angular Velocity of Moon

T = Total Time of Flight

r, θ = Nonrotating Polar Coordinates

v_b = Burnout Velocity

β_b = Burnout Flight Path Angle

r_E = Radius of Earth

h_b = Burnout Altitude

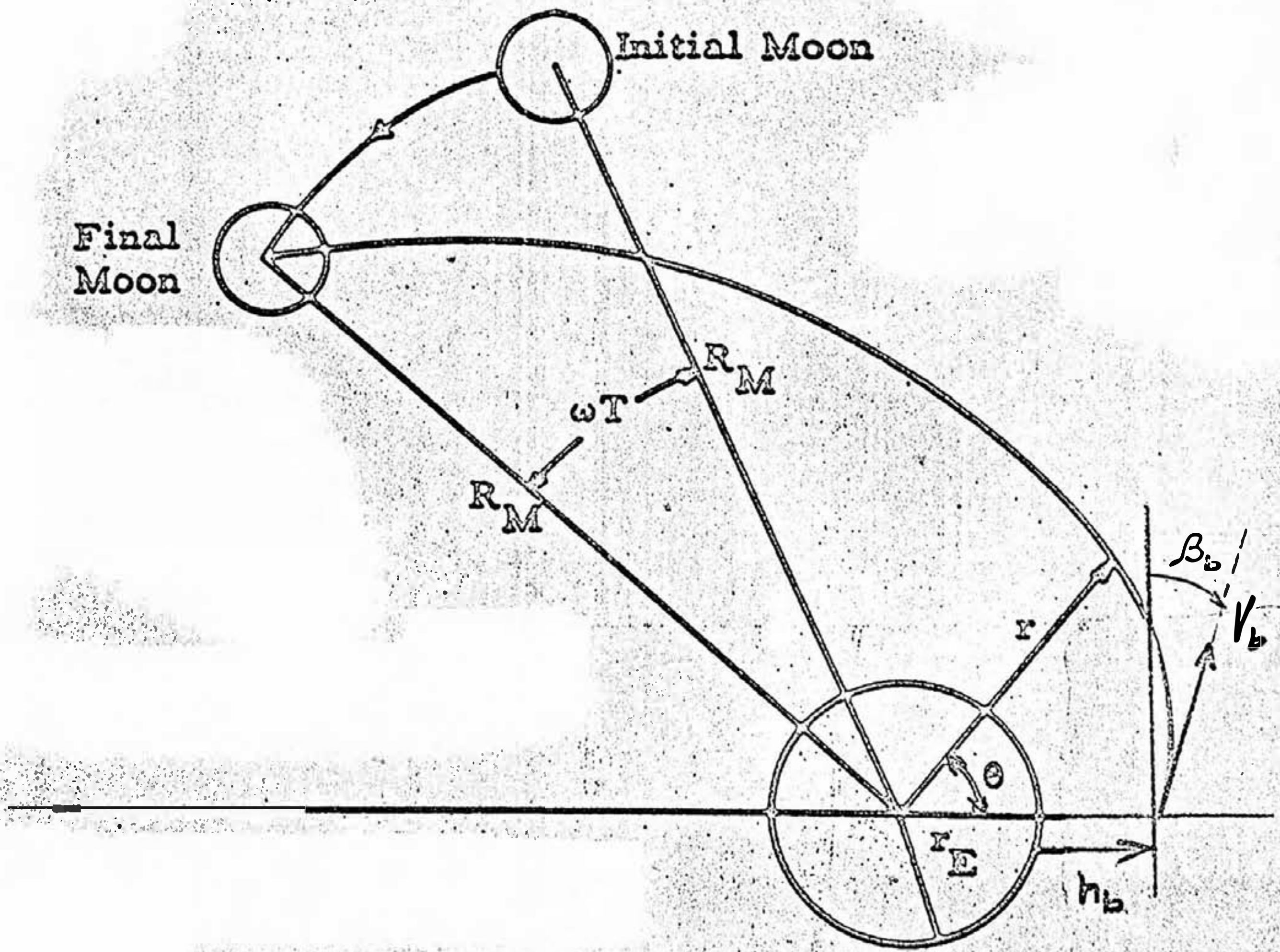


Figure 2. Planar Geometry.

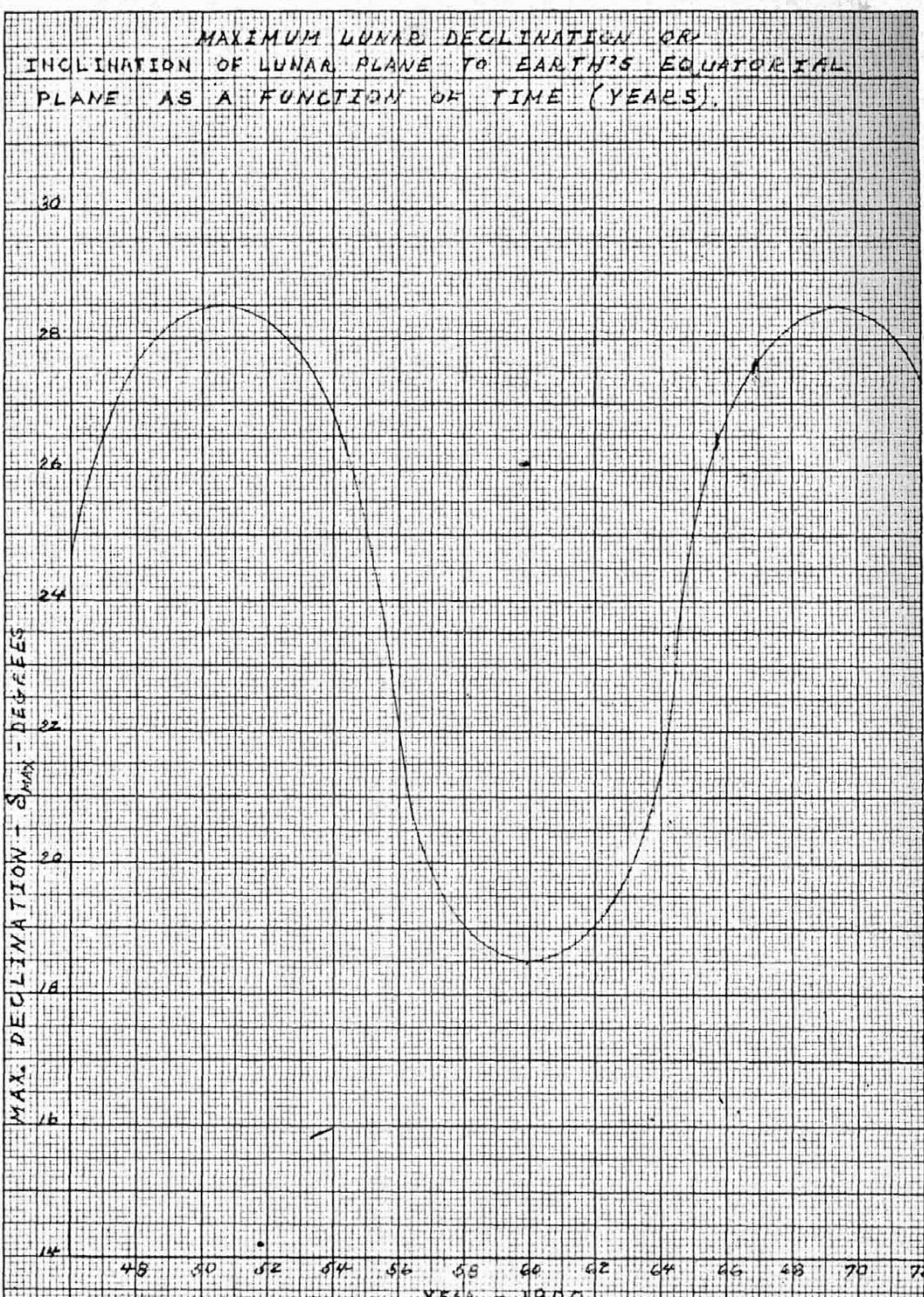

MAXIMUM LUNAR DECLINATION OR
INCLINATION OF LUNAR PLANE TO EARTH'S EQUATORIAL
PLANE AS A FUNCTION OF TIME (YEARS).
30
28
26
24
22
20
18
16
14
MAX. DECLINATION - δ_{MAX} - DEGREES
48
50
52
54
56
58
60
62
64
66
68
70
72
YEAR - 1900

FIGURE 3.

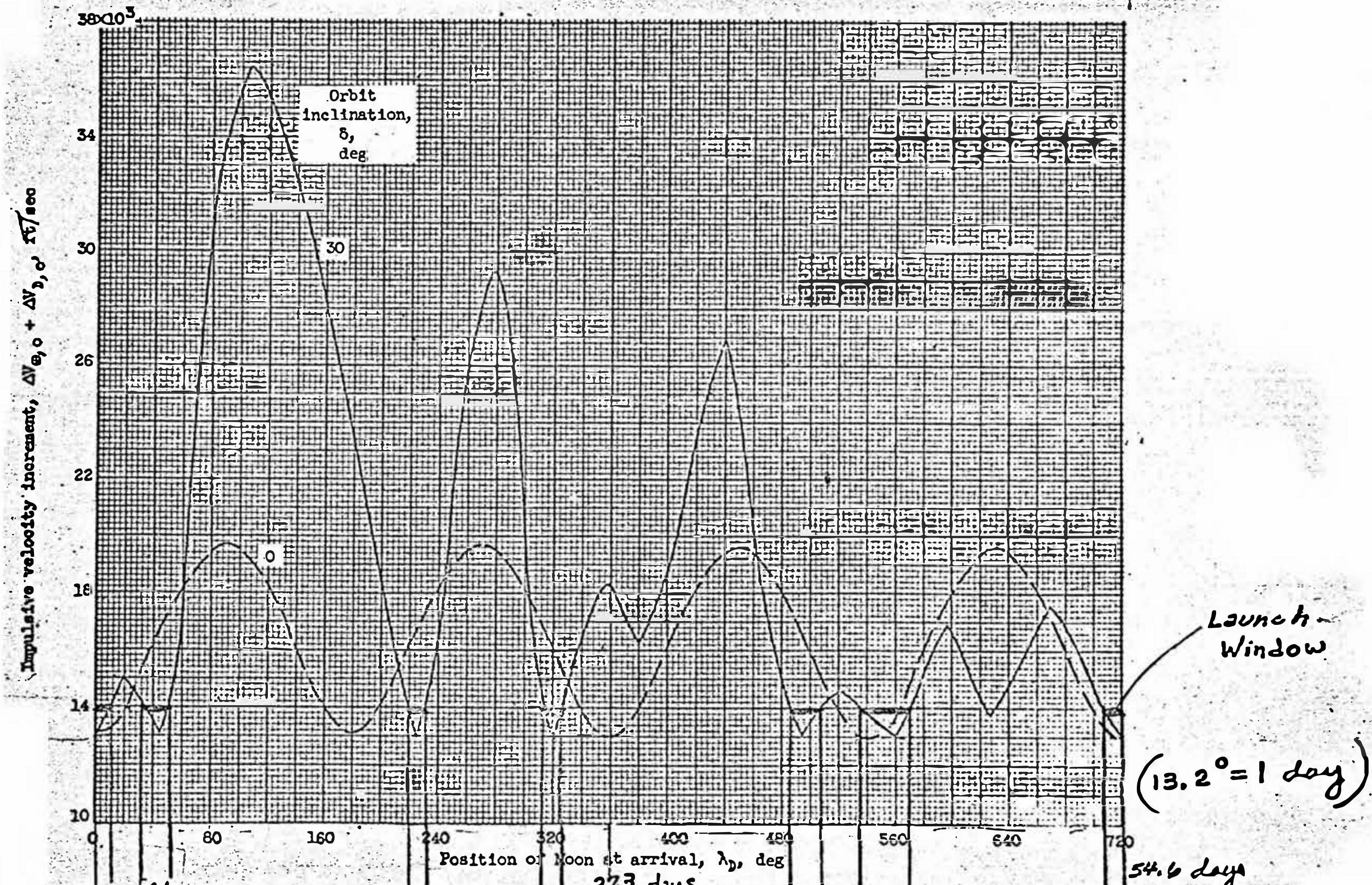


Figure 24 - Impulsive velocity increment to depart from precessing Earth orbit and enter lunar orbit. Earth-orbit altitude, 300 International nautical miles; lunar-orbit altitude, 500,000 feet (82 Int. naut. miles); lunar-orbit inclination, i, 18.5°; burnout angle, β_b, 20°; initial λ_o, 0°; $\dot{\lambda}_o$, -6°/day.

Launches made from the Optimum orbital position (φ_b, λ_b)

"Launch Windows" are shown for a "dog-Legging" ΔV = 1,000 fps.

REFERENCES

1. Weber, R. J., Pauson, W. M., Burley, R. R.:
Lunar Trajectories NASA TN D-866

2. Mickelevait, A. B. and Booton, R. C., Jr.
"Analytical and Numerical Studies of Three-Dimensional Trajectories to the Moon" - Journ. of Aerospace Sciences Aug. 1960 Vol. 27 No. 8

3. Alexander, G.
"Marshall Intensifies Rendezvous Studies"
Aviation Week, March 19, 1962

4. Houbolt, J. C.
"Problems and Potentialities of Space Rendezvous"
Astronautica Acta, Vol. VII, 1962

5. Peterson, N. V.
"Orbital Assembly and Launch for Lunar Operations"
IAS Paper No. 62-81

6. Anon.:
"Lunar Probe Flight - Intro. to Flight Geometry and Accuracy Problems"
ABMA DA-TN-58-58 (1958)

7. Anon.:
"Lunar Probe Trajectories - Characteristics and Tolerances for Coplanar Flight"
ABMA DA-TM 26-59 (1959)

8. "A Dynamic Analysis and Preliminary Design of Guidance for Lunar Vehicles"
Aeronutronic Publication C-590, Interim Report - September 1959

9. Buckheim, R. W.
"Motion of a Small Body in Earth-Moon Space"
RM-1726 - Rand Corp., June 4, 1956

10. Weber, R. J. and Pauson, W. M.
Some Thrust and Trajectory Considerations for Lunar Landings.
NASA TN D-134, 1959

11. Moeckel, W. E.
Departure Trajectories for Interplanetary Vehicle
NASA TN D-80, 1959

12. Michael, W. H. and Tolson, R. H.
Three-Dimensional Lunar Mission Studies
NASA Memo 6-29-59L, 1959

13. Huss, C. R., Homer, H. A., Mayer, J. P.
Parameter Study of Insertion Conditions for Lunar Missions Including Various Trajectory Considerations
NASA TR R-122, 1961

14. Clement, G. H.
The Moon Rocket
P-833 Rand Corp., 1956

15. Ehricke, K. A.
Some Basic Aspects of Operation in Cislunar and Lunar Space
Convair-Astronautics Report

16. Lieske, H. A.
Lunar Trajectory Studies
P-1293. Rand Corp., 1958

17. Buchheim, R. W.
Lunar Flight Trajectories
P-1268, Rand Corp., 1958

18. Mickelwait, A. B.
Lunar Missions - Launch to Rendezvous
ARS Paper No. 2456-62 (1962)

19. Clarke, V. C.
Design of Lunar and Interplanetary Ascent Trajectories
J.P.L. TR 32-30 (1960)

20. Tolson, R. H.
Effects of Some Typical Geometrical Constraints on Lunar Trajectories
NASA TN D-938 (1961)

21. Woolston, D. S.
Declination, Radial Distance, and Phases of the Moon for the Years 1961 to 1971 for use in Trajectory Considerations
NASA TN D-911 (1961)

NOMENCLATURE

a	azimuth, deg. east from north
h	altitude
i	inclination of Moon's plane to Earth's equatorial plane, deg.
r	radius
t	time
u	angle between vehicle's trajectory plane and lunar plane
V	velocity
$\bar{V}$	dimensionless velocity, V/Vc
ΔV	characteristic-velocity increment
α	angle between velocity vectors of moon and vehicle
β	Angle between vehicle velocity vector and local horizontal
δ	inclination of vehicle's orbital plane to equatorial plane
θ	in-plane travel angle, deg.
λ	angular distance from descending lunar node, deg.
μ	universal gravitational constant times mass of celestial body
ρ	radius ratio
φ	latitude, deg.
$\mho$	Moon's descending equatorial node

Subscripts:

I	initial
b	burn-out (i.e., start of coast trajectory)
c	circular
L	launch
M	Moon
O	orbit
P	perigee
V	vehicle
$\oplus$	Earth
$\rightmoon$	Moon at time of rendezvous

Sept. 14, 1962
SP-1362
Enclosure 4-A

In order to determine the $\Delta V_{\oplus,0}$ for the most favorable launch point (within a 10-degree interval of latitude Φ_b) for a 30-deg. inclined precessing orbit, the following procedure must be repeated 12 times.

Given: Φ_b, λ_D, $\lambda_{O,I}$, h_b, β_b, i, δ.

Solve for the transit time t by iterating equations 1-7, thereby, simultaneously obtaining burn-out longitude λ_b, travel angle θ, and burn-out velocity V_b.

1. $\lambda_{M,L} = \lambda_D - \dot{\lambda}_D t$

2. $\lambda_{O,L} = \left(\dfrac{\dot{\lambda}_O}{\dot{\lambda}_D}\right) \lambda_{M,L} + \lambda_{O,I}$

3. $\lambda_b = \lambda_{O,L} - \sin^{-1}(\tan \Phi_b \operatorname{ctn} \delta)$

4. $\gamma = \tan^{-1}\left(\dfrac{\tan \lambda_b}{\cos i}\right)$

5. $$\theta = \cos^{-1}\left\{\cos(\lambda_D - \gamma)\cos\left[\Phi_b + \sin^{-1}(\sin\gamma \sin i)\right] + \sin(\lambda_D - \gamma)\sin\left[\Phi_b + \sin^{-1}(\sin\gamma \sin i)\right]\sqrt{1 - \left(\frac{\sin\lambda_b}{\sin\gamma}\right)^2}\right\}$$

6. V_b is computed from the following equation by an iteration method, after transforming it

to a more suitable form.

6. $$\theta = \cos^{-1}\left[\frac{\bar{V}_b^2\cos^2\beta_b - \rho}{\rho\sqrt{1+(\bar{V}_b^2-2)\bar{V}_b^2\cos^2\beta_b}}\right] - \cos^{-1}\left[\frac{\bar{V}_b^2\cos^2\beta_b - 1}{1+(\bar{V}_b^2-2)\bar{V}_b^2\cos^2\beta_b}\right]$$

where $\rho = \dfrac{r_D}{r_b}$; $\bar{V}_b = \dfrac{V_b}{V_c} = \dfrac{V_b}{\sqrt{\mu/r_b}}$

Compute transit time t from:

7. (a) If $\bar{V}_b^2 < 2$

$$t = \frac{r_b}{V_c}\left\{\frac{\bar{V}_b\sin\beta_b - A}{2-\bar{V}_b^2} - \frac{1}{(2-\bar{V}_b^2)^{3/2}}\left[\sin^{-1}\frac{(2-\bar{V}_b^2)\rho-1}{1-(2-\bar{V}_b^2)\bar{V}_b^2\cos^2\beta} - \sin^{-1}\frac{1-\bar{V}_b^2}{1-(2-\bar{V}_b^2)\bar{V}_b^2\cos^2\beta_b}\right]\right\}$$

(b) If $\bar{V}_b^2 = 2$

$$t = \frac{r_b}{V_c}\left[\frac{1}{3}(\rho+2\cos^2\beta_b)\sqrt{2(\rho-\cos^2\beta_b)} - \frac{1}{3}(1+2\cos^2\beta_b)\sqrt{2(1-\cos^2\beta_b)}\right]$$

(c) If $\bar{V}_b^2 > 2$

$$t = \frac{r_b}{V_c}\left\{\frac{A-\bar{V}_b\sin\beta_b}{\bar{V}_b^2-2} - \frac{1}{(\bar{V}_b^2-2)^{3/2}}\ln\sigma\right\}$$

where $$\sigma = \frac{(\bar{V}_b^2-2)\rho + 1 + A\sqrt{\bar{V}_b^2-2}}{\bar{V}_b^2-1+\bar{V}_b\sin\beta_b\sqrt{\bar{V}_b^2-2}}$$

and $$A = \sqrt{\rho(\bar{V}_b^2-2) + 2\rho - \bar{V}_b^2\cos^2\beta}$$

8. $$\Delta a = \sin^{-1}\left[\frac{\sin\lambda_b\sin(\lambda_D-\gamma)}{\sin\gamma\sin\theta}\right] - \sin^{-1}\left[\frac{\cos\delta}{\cos\phi_b}\right]$$

9. $$\Delta V_{\oplus,0} = \sqrt{V_b^2 + V_c^2 - 2V_bV_c\cos\beta_b\cos\Delta a}$$

If a fixed transit time t is specified the suggested computational procedure is as follows:

Given: t, λ_D, $\lambda_{O,I}$, h_b, β_b, i, δ,

Θ_b and V_b may be obtained from parametric curves in REF. 2 or computed from the preceding eqs. 6 and 7.

Compute:

1. $$\lambda_{M,L} = \lambda_D - \dot{\lambda}_D t$$

2. $$\lambda_{O,L} = \left(\frac{\dot{\lambda}_O}{\dot{\lambda}_D}\right)\lambda_{M,L} + \lambda_{O,I}$$

3. Compute λ_b by iteration from
$$\lambda_b = \tan^{-1}(\cos i \tan \lambda_D) - \sigma$$
$$\sigma = \cos^{-1}\left\{\frac{\cos\theta\sqrt{\operatorname{ctn}^2\delta + \sin^2(\lambda_{O,L}-\lambda_b)} - \sin(\lambda_{O,L}-\lambda_b)\sin\lambda_D \sin i}{\operatorname{ctn}\delta\sqrt{1-\sin^2\lambda_D \sin^2 i}}\right.$$

4. $$\varphi_b = \tan^{-1}\left[\frac{\sin(\lambda_{O,L}-\lambda_b)}{\operatorname{ctn}\delta}\right]$$

5. $$a_o = \sin^{-1}\left(\frac{\cos\delta}{\cos\varphi_b}\right)$$

6. $$a_b = \sin^{-1}\left[\frac{\sin\lambda_b \sin(\lambda_D - \gamma)}{\sin\gamma \sin\Theta}\right].$$

7. $$\Delta a = a_b - a_o$$

8. $$\Delta V_{\oplus,o} = \sqrt{V_b^2 + V_c^2 - 2V_b V_c(\cos\beta_b)\cos\Delta a}$$

The $\Delta V_{D,0}$ retrothrust for establishing a lunar parking orbit is computed as follows:

1. $\bar{V}_{V/\oplus}^2 = \bar{V}_b^2 - 2\left(1 - \frac{1}{\rho}\right)$

2. $\cos B_{V/\oplus} = \dfrac{\bar{V}_b \cos B_b}{\rho \; \bar{V}_{V/\oplus}}$

3. $u = \sin^{-1}\left\{ \dfrac{\sin \lambda_b \sin\left[\varphi_b + \sin^{-1}(\sin\delta \sin i)\right]}{\sin\delta \sin\theta} \right.$

4. $\alpha = \cos^{-1}(\cos u \cos B_{V/\oplus})$

5. $V_{V/D} = \sqrt{(V_{V/\oplus}\cos\alpha - V_D)^2 + (V_{V/\oplus}\sin\alpha)^2}$

6. $V_P = \sqrt{V_{V/D}^2 + 2\mu_D/r_P}$

7. $\Delta V_{D,0} = V_P - \sqrt{\mu_D/r_P}$.

The time of arrival of a ~~moon~~ at a node may be computed by iteration from the equations: (transit time (t) fixed)

1. $\lambda_D = \lambda_{M,L} + \dot{\lambda}_D t$

2. $\lambda_{0,L} = \left(\dfrac{\dot{\lambda}_0}{\dot{\lambda}_D}\right)\lambda_{M,L}$

3. $\cot \lambda_s = \dfrac{\cot\delta \sin i - \cot(\lambda_{0,L} + \lambda_{0,I})\cos i}{\sin(\lambda_{0,L} + \lambda_{0,I})}$

APPENDIX E

SPACE EXCURSION VEHICLE SERVICING AND MAINTENANCE OPERATIONS CONCEPT

The general concept envisioned for the excursion vehicle servicing and maintenance at the space station, under which the operations involving such requirements were configured, would correspond generally in character and scope to what is considered launch site or field maintenance and servicing operations for rocket launch vehicles, or flight line type servicing and maintenance operations for aircraft. That is, it would be limited to minor repairs, replacements, component repair, all sorts of inspection and check-outs, and general servicing operations such as fueling, adjustments, equipment tests and calibrations, etc.

Beyond this the general servicing and maintenance philosophy, which would apply to the station and its equipment as well as to the vehicles, would be one of progressive maintenance integrated with the regular servicing operations, plus specific required maintenance and repair as such requirements occur. This would be carried out by the mechanics and others specified in the manning chart (Section IV of GER 10866), with appropriate assistance by other station personnel, with the facilities, tools, equipment, parts and materials specifically provided for this purpose. Such items are listed in Table E-1.

Approximately 14 feet of station section would be allocated for such equipment and activities, and for storage of tools, parts and materials. For servicing or maintenance operations required on the outside of the station or vehicle, minor ones could be carried on without the aid of protective cover, - whereas

TABLE E-1

SERVICING AND MAINTENANCE EQUIPMENT AND SUPPLIES

Vehicle Servicing Equipment (added after operating in Orbit)				3050
Fuel Handling			1500	
Storage Tank, mount, controls		1225		
Lines, hose, dry (ball) connectors, meters		75		
Valves, pumps, air supply, regulators, gauges		200		
Vehicle Handling			1000	
Docking, air lock, controls equip.		250		
Entrance hatches, connector tunnels vehicle enclosures		750		
Guidance			550	
Optical sighting, radar measuring equipment		200		
Communications, command control		150		
Docking officer station, displays controls		100		
Station coordinate monitoring equip.		100		
Maintenance & Repair Equipment				2225
Equipment and Facilities			1255	
Tools, workbench, accessories, (vise, grinder, etc.)		180		
Mechanical equipment maintenance equip., machine tools		800		
Hand drills, - drill press arbor press	200			
Lathe, milling attachment	250			
Welding, riveting equip.	150			
Sheet metal working tools, shears, small brake	200			
Electronic repair facilities		150		
Meters, generators, oscilloscope, testers	120			
Soldering equip. & tools	30			
Work benches, cabinets, etc.		125		
Materials and Supplies			970	
Spare parts		350		
Materials stock		250		
Miscellaneous supplies and equipment		370		
Hydraulic fluid, lubricants, paints sealing compounds, etc.	120			
Air compressor, motors, etc.	150			
Wire, tubing, etc.	100			
TOTAL WEIGHT				5275

more extensive ones could utilize flexible work cell or "hangar" enclosure concepts such as that shown in the frontispiece in phantom. Such enclosure is erected by bringing it over the vehicle, and "zippering" in place to a flap edge provided around the docking collar and the entrance tunnel, and along the remaining closure edge. Then the electrical plug, air line, and inter-com cable would be plugged in to connectors on the side of the air lock - docking unit, and inflation pressure applied.

Studies made of such hangar set-up[a] for the SLOMAR concept have indicated a pressure of $3\frac{1}{2}$ psi should be about optimum. This would permit back-pack operations without requiring suit pressurization. On the inside, simple lighting, electrical outlets (for power tools, and partial vehicle power during servicing), and compressed air outlets should be provided. The power and air would be supplied from the station systems through the above-mentioned connectors. Use of the "hangars" or work cells would be limited to such operations as justified its erection, although with a little practice the erection should be quite simple to perform under weightless conditions.

The equipment and supplies for these operations, as listed in Table E-1, would be brought up in properly selected sequence as portions of the loads on the first 2 or 3 supply rockets, and would be supplemented from that point on as experience indicated. The initial material and spare parts stock (for both vehicle and station maintenance) would be determined on the usual basis for remote base operation -- emphasis on those items subject to failure that are

---

[a]See GAP-1190, Goodyear Aircraft Corporation, Akron, Ohio, November 30, 1961 (Unclassified)

required to maintain essential functions that cannot be readily repaired or duplicated with the facilities at hand (as listed in Table E-1). One only of each item would normally be stocked, since stock replacement items (after use) could readily be brought up (they would usually weigh much less than 50 pounds) on any of the regular flights.

No detailed study of these particular maintenance operations has been made - however, the general considerationsset forth here were the basis of such elements in the servicing schedule of Figure 16 of GER-10866, and considerations of similar operations at remote isolated bases in a severe environment (e.g., aircraft and base equipment service and maintenance at arctic bases) were used as a guide. The most significant characterisitic of the operations considered here is that they will grow gradually under the operational concept described herein, thus affording maximum opportunity for accumulating experience to facilitate improving efficiency and ease in carrying them out. Thus rigorous and critical requirements are not as likely to be encountered before months of accumulating experience and capability to deal with them have passed, with a resulting acquisition of gradually increasing proficiency and skill that should match the slowly growing requirements.

GOODYEAR AIRCRAFT CORPORATION