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THE SPACE MISSION PLANNING CHART

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* SUMMARY

The Space Mission Planning Chart provides a
5 simple method for rapidly determining vehicle re-
quirements for planetary missions likely to be per-
formed through the early 1980's. While there is a
general familiarity with the type and size of boost-
12 ers required for lunar operations, the propulsion
means required for planetary operations are not so
well known. The Space Mission Planning Chart makes
booster and propulsion requirements for manned plan-
18 etary operations readily comprehensible. No factor
has more effect on the overall size and total con-
figuration of space exploration systems than the
38 means of propulsion used, and this chart provides a
simple graphical means of viewing the interrelation-
ship between propulsion capability (specific im-
pulse), mission velocity requirements, and payload
* requirements for a variety of manned space missions
within the solar system. Through use of this chart,
the effects of a change in any of these variables
43 can be quickly assayed.

INTRODUCTION

Determining vehicle requirements for manned
47 missions involves the interrelating of propulsion
capability and velocity requirements with payload
demands. Through the use of the "Space Mission
54 Planning Chart,"* this highly complex operation can
be simplified considerably. An early version of the
chart is shown in Figure 1.

60 The chart is made up in five sections, the most
important of which are the Nomograph (bottom, cen-
ter) and the I_{sp} Protractor. The graph relates
booster velocity capability to payload weight capa-
71 bility in such a manner that propulsion requirements
for any space mission can be determined quickly and
easily. With the Nomograph, a graphical solution
can be found to problems as basic as an Earth-orbit
76 or as complex as a round trip reconnaissance to the
planet Saturn.

81 The Nomograph can be used in other ways as
well. For example, the launch weight of the Russian
booster used to place the Vostok manned capsule into
90 orbit can be determined graphically with this de-
vice, using as a starting point data readily avail-
able from newspapers. Such a problem is worked out
in detail here, as are two other space mission exer-
93 cises. In summary, the Nomograph, together with the
I_{sp} Protractor, permit the determining of vehicle
requirements for any mission likely to be performed
within the next 20 years.

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U.S. Gross National Product (1960)

The Bar Graph (Figure 2) at the upper left of the Space Mission Planning Chart, while not directly applicable to space mission vehicle requirements, is, nevertheless, of great interest to all taxpayers. It compares the amount of money spent annually in the United States for all sorts of goods and services to the number of dollars spent for space activities. For instance, in 1960, for every dollar spent by Americans to probe the depths of outer space, five dollars were spent on amusements, six dollars for schools, and eight dollars for liquor and tobacco. These figures show all too clearly the relatively slow pace of America's Space Program at the start of the 1960 decade. The pace is now quickening, and the projection of current trends indicates that Americans will be spending on the order of 5 billion dollars a year for space activities by 1970 -- roughly five times the 1960 amount.

Solar System Chart

The Solar System Chart (top, right in Figure 1) has been aligned so that it can be used in conjunction with the Nomograph directly below. For example, the velocity requirement for an orbital reconnaissance of the planet Saturn and the return to Earth is 400,000 ft/sec. Saturn has been positioned on the chart so that it is directly above the 400,000 ft/sec line on the X-axis. The other planets (excepting Uranus, Neptune, and Pluto) have been positioned similarly directly above the velocity requirements for similar round-trip reconnaissance missions. Also included on the chart are the duration of each planet's year, distance from the sun, number of moons, surface gravity, and escape velocity. Velocity requirements and mission durations have been tabulated in Table I for a multitude of missions.

Possible U.S. Boosters

Figure 3 shows an enlarged view of proposed boosters which appear in the lower left-hand corner of Figure 1. Payload capabilities have been shown for each booster for selected planetary and lunar missions assuming the use of O_2 - H_2 propulsion in the booster upper stages and in the spacecraft propulsion modules.

Separate mention should be made of the I_{sp} Protractor which has been constructed in such a manner that the relationship of payload weight versus velocity change can be determined for any given propellant or type of propulsion system. Or, working backwards, if it is desired to know the impulse that was or is to be developed, a sloped line on the graph can be compared with the I_{sp} Protractor in order that the effective specific impulse can be determined.

Manned Spacecraft

Manned spacecraft (Figure 4) proposed for trips to Mars and to Jupiter are shown at the lower right of the chart (Figure 1) and are drawn to the same scale as the boosters on the left side of the chart. A four-engine jet airliner has been drawn to the same scale to give some idea of spacecraft size.

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* Large multi-colored copies of a revised and updated version of the Space Mission Planning Chart suitable for hanging may be obtained from the Aerojet-General Corporation, Azusa, California.

Nomograph

Description of Coordinates On the Nomograph (Figure 5) payload weights are shown on the Y-axis and range from 1K to 100M pounds (1,000 to 100,000,000 pounds). Velocity requirements are shown along the X-axis. The maximum velocity indicated is 400,000 ft/sec; sufficient for a round-trip reconnaissance to Saturn.

In actual practice, the velocities for planetary missions will vary with the ever-changing positions of the planets. The type of rocket or electric propulsion used will also influence mission velocity values. For missions to Jupiter or to Saturn, actual velocities may differ from velocities indicated on the chart by as much as 10,000 to 20,000 ft/sec, or even more. In addition, the manned spacecraft weights will very likely not be exactly those indicated on the chart. In any event, it is felt that the spacecraft weights shown on the chart are reasonable and will serve the intended purpose of the chart.

The four heavy approximately horizontal lines running from left to right indicate nominal weights of manned spacecraft at the end of any particular space reconnaissance mission. For instance, the ultimate payload weight of a 45-man reconnaissance mission to Jupiter is 600K pounds (Pt. A). The weights indicated by these four lines would be analogous to the weight of the Mercury capsule just before retro-firing when terminating orbital operations: 3K pounds in the case of the Mercury capsule.

Reconnaissance Mission to Jupiter To understand how the Nomograph is used, assume that 45 men are to be sent on a round-trip orbital reconnaissance mission to Jupiter. Assume also that electric propulsion with a specific impulse of 6,000 sec will be used in going to and from Jupiter. The problem: to determine the weight which must be delivered into earth orbit to begin such a mission.

First locate the pertinent vertical line from the Planet Jupiter marked, "Jupiter Orbit and Return" (250,000 ft/sec). Trace down this line until it intersects the line marked, "Final Weight, 45-man Space Mission". The intersection of these two lines, labelled Pt. A for convenience, establishes one point from which to find the solution.

The second point lies somewhere along the vertical line marked, "Earth-Orbit." But where? Since it is known that a specific impulse of 6,000 sec is involved, determination of Point B is a simple matter. It is necessary only to refer to the I_{sp} Protractor, set a ruling device parallel to the slope for 6,000 sec, and, using this same slope, draw a line from Point A up to the left until it intersects the "Earth-Orbit" vertical. The point of intersection (Point B) is found to have a Y-axis value of 5M pounds. The weight, then, that must be put into earth-orbit at the start of such a 45-man round trip reconnaissance mission to Jupiter is 5,000,000 (5M) pounds.

Mars Landing and Return Mission Space missions involving planetary landings are only slightly more complicated than orbital reconnaissance missions. Take, for example, a proposed small-scale manned mission to Mars, using nuclear propulsion. Assume that 5M pounds can be placed into orbit to start such a mission and that in transferring in both directions between Earth orbit and Mars orbit nuclear propulsion operating with a specific impulse of 950 seconds is to be used. Assume also that, after establishing an orbit about Mars, a three-man landing craft will be detached to descend to the surface and later return to the mother ship. Propulsion for the lander will be Oxygen-Hydrogen operating with a specific impulse of 425 seconds. The problem is to determine how many men can be carried on the mission. A better understanding of the problem can be obtained if this problem is worked in several parts. First, the chart will be used to determine the number of men who can make the journey if no planetary landing is involved. Next, the number of men will be determined when a 3-man Mars landing is included in the mission.

First to be determined is the velocity change that must be produced by the nuclear rockets; this is, simply enough, the difference between 30,000 ft/sec, Earth orbit velocity, and 98,000 ft/sec, the total velocity required for a Mars reconnaissance. The difference is 68,000 ft/sec, and it is reasonable to assume that half of this, 34,000 ft/sec, must be developed by the nuclear propulsion system during each direction of the interplanetary transfer maneuvers.

Starting from Pt. B (5M-lb Earth orbit) and using the I_{sp} Protractor, a line should be drawn with a slope of 950 secs down to the right to Pt. C (64,000 ft/sec). This shows the weight that can be delivered into Martian orbit to be 900K lbs. If no lander is involved, the line can be continued with the same slope to Pt. D at 98,000 ft/sec, where the weight returned to Earth proves to be 150K lbs. Inspection of the chart shows that this is representative of a 10-man spacecraft.

How many men will make up the Mars Reconnaissance Mission if a 3-Man Landing Mission is involved? The solution to the first part of this problem, that of determining spacecraft arrival weight at Mars (Pt. C) is identical to the solution for the Mars Reconnaissance Mission which was just worked. After determining Pt. C, the solution to the rest of the problem is somewhat different in that the mass of our spacecraft will be lightened by the total mass of the 3-Man Lander. It is assumed here that, upon completion of the landing excursion, the landing craft will be left in Mars orbit and not carried back to Earth.

The next step, then, is to determine the weight of the lander at the time of detachment from the mother ship. This will be treated as a separate problem. From the graph, the weight of a 3-Man lander is determined to 12K lbs. In the statement of the problem it was said that the I_{sp} for the lander's propulsion system would be 425 sec. All that is needed now to find the initial weight of the lander is a value for the characteristic velocity of that Landing Mission. In this instance, we will consider that some 1,000 to 2,000 ft/sec will be required to retro from orbit into the atmosphere where aerodynamic drag then slows the craft to a safe landing velocity. Escape velocity for Mars is obtained from the chart and is 16,700 ft/sec and orbital velocity is 0.707 x 16,700 = 11,800 ft/sec. Considering then the retro ΔV and the climb against gravity through the Martian atmosphere when returning to orbit, we will rather arbitrarily assume that the velocity change that must be produced by the Lander's propulsion system is 133 percent of orbital velocity (16,000 ft/sec). Velocity requirements for landing missions to other planets and the larger moons have been tabulated in Table II.

In the lower right-hand corner of the graph a secondary Lander ΔV scale intended especially for the solution of Lander weights has been provided. In this instance, the final weight is known, having been found on the graph to be 12K lbs, and all other information required to determine the Lander's initial weight is available. From Pt. E (16,000 ft/sec and 12K lbs weight) a line is drawn sloping upward to the left with a slope of 425 secs to zero velocity (on the Lander ΔV scale). At this point, Pt. F, the initial weight of the Lander is found to be 60K lbs.

At the start of the return trip to Earth, the mother spacecraft will be lighter by 60K lbs, since the 3-Man Lander will have been jettisoned. This jettisoned amount is subtracted from the 900K lbs Mars arrival weight and 840K lbs (Pt. G) is found to be the lightoff weight at the start of the return to Earth.

From Pt. G a line with a slope of 950 seconds is drawn down to the right until it intersects the vertical line, Mars Reconnaissance Mission, at 98,000 ft/sec. This intersection is indicated as Point H.

Pt. H is then inspected to see if it lies above or below the line marked, "Final Weight - 9-Man Space Mission." Since Pt. H is found to lie slightly above the line, it is shown that 9 men could be carried on the Mars Reconnaissance and Landing Mission just described.

One interesting thought occurs. In this problem, a nuclear propulsion system was used as the sole means of producing the velocity changes during the interplanetary transfer maneuvers. But if, instead, an ablation-type heat shield could somehow be used in decelerating to orbital velocity upon return to Earth, the mass loss of the heat shield would be considerably less than the mass expended by rocket propulsion in producing the same deceleration/velocity change. Since in both cases mass is expended to produce a velocity change, it should therefore be possible, in the case of any ablation-type heat shield, to relate the "mass ratio"* and the corresponding velocity changes brought about by aerodynamic deceleration to obtain an "equivalent I_{sp}" for such a heat shield. Such calculations have been made and the results are indicated on the I_{sp} Protractor for orbital, lunar, and hyperbolic reentries.

Assuming ablation means can be used to produce the 25,000-ft/sec velocity change, Pt. I is found by drawing a vertical line at 73,000 ft/sec (98,000 minus 25,000) and marking the point where this line intersects line GH. Point J is found by using the Equivalent I_{sp} Ablation slope in drawing the line from Pt. I to the "Mars Orbit and Return" vertical line -- 98,000 ft/sec. From the location of Pt. J it can be seen that if ablation means had been used, 25 to 30 men could have been taken on this mission to Mars rather than a mere 9 men.

The substitution of electric power for nuclear power is also of interest. From the slopes of the lines involved, it is clear that a vehicle of Sea Dragon size will be required for the Mars mission if nuclear power were used. But if electric propulsion (6,000 sec I_{sp}) were to be substituted for nuclear power, a single Nova launch would suffice for this same Mars mission. To confirm this, use the 6,000-sec slope in drawing a line from Pt. H to the 64,000 ft vertical. Then, move up 60K lbs - the weight of Mars Lander. From this point, draw a line of 6,000 sec slope until the "Earth-Orbit" vertical is reached. This will establish Pt. K at 400K lbs. Since the chart at lower left indicates Nova can put 400K lbs into Earth-Orbit, a single Nova could be used to start an electrically-propelled 9-Man spacecraft on a Mars Reconnaissance and Landing Mission.

Launch Weight of Russian Vostok Booster What is the launch weight of the Russian Vostok booster; the launch vehicle that placed Gagarin and Titov into orbit? Performance data available on this booster is very limited, but by piecing together items appearing in the newspapers and making use of the I_{sp} Protractor, a reasonable weight estimate can be made. The following facts are known:

(1) The orbiting (30,000 ft/sec) manned spacecraft had a gross weight of about 11K lbs (Pt. L).

(2) The Russian Venus probe (44,000 ft/sec) assumed to have been launched by the same basic booster, weighed about 1.4K lbs (Pt. M).

Connect Pt. L with Pt. M and by comparing the slope of this line with the I_{sp} Protractor, its slope is found to be 300 sec. Inasmuch as 300 sec is substantially the impulse obtained in expanding storables, solids, or LOX-kerosene propellants into a vacuum, it would appear that the Venus probe launch did not use the higher-impulse oxygen-hydrogen propellant combination in the upper stages. It follows then that if the Russians did not have high-impulse (O_2 - H_2) propulsion systems for these smaller upper stages, it is not likely they would have them for the much larger lower stages. With their lower-stage propulsion performance now established (range 240-300 sec), the mean value (270 sec) can be used in the final computation. This involves the drawing of a line from Pt. L to the zero vertical velocity (the earth launching point, referred to as Pt. N), using a 270-sec slope. The value of Pt.N is seen to be between 800K - 900K, establishing the launching weight of the Russian Vostok booster as approximately 850,000 lbs. From the chart at the lower left, it can be seen that this launching weight is about the same as that of our C-1 Saturn. The Vostok booster has been operational since 1960. Our C-1 Saturn, on the other hand, is not scheduled to become operational until 1964.

CONCLUSIONS

The concept presented here of the I_{sp} Protractor clearly indicates the need for development of

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*Here for a deceleration produced by an ablation-type heat shield the "mass ratio" = $\dfrac{\text{initial reentry weight}}{\text{initial reentry weight minus heat-shield weight loss}}$

high-specific impulse propulsion systems. The use of air-breathing lower stages should be of value. For a jet engine (the specific impulse is 10,000 to 15,000 seconds when computed on the basis of fuel flow rate). The atomic ramjet, another possibility, would have a specific impulse which theoretically approaches infinity but which, in practice, would be several hundred-thousand seconds.

Until operational high specific impulse propulsion devices are developed, manned exploration of the planets and their moons will not be very practicable. It is clear also that extensive colonization of other planets cannot be accomplished until means are devised to generate a reaction thrust without the necessity of expelling mass. Anti-gravity would accomplish this, a discovery that should be man's by the end of the 20th century.

TABLE I

Selected Ballistic, Lunar, and Planetary Missions

	Nominal Velocity* (ft/sec)	Nominal Duration
Intermediate Range Ballistic Missile (IRBM)	18,000	15 minutes
Intercontinental Ballistic Missile	27,500	30 minutes
Earth Orbit, 300-n-mi. Orbit	30,000	90 minutes
Earth Escape, Circumlunar	40,000	5 days
Lunar Landing, One Way	50,000	2-1/2 days
Lunar Landing, Earth-Return	60,000	1 - 2 weeks
Mars Cargo Mission, One Way	64,000	9 months
Mars Reconnaissance Mission (1-Year Duration, No Landing, No Martian Orbit)	85,000	12 months
Close Solar Probe, One-Way Instrumented	90,000	4-1/2 months
Mars Reconnaissance, Planetary Orbit and Return to Earth	98,000	2-1/2 years
Venus Reconnaissance, Planetary Orbit and Return to Earth	104,000	1-1/4 years
Solar System Escape, Ecliptic	120,000	
Mercury Reconnaissance, Planetary Orbit and Return to Earth	150,000	1-1/2 years
Solar System Escape, 45° out of Ecliptic	245,000	
Jupiter Reconnaissance, Planetary Orbit and Return to Earth**	250,000	3-2/3 years
Saturn Reconnaissance, Planetary Orbit and Return to Earth**	400,000	4-1/2 years

* These tabulated values have been increased to include aerodynamic and gravity losses, which are assumed to be 4,000 ft/sec for attaining an Earth orbit.

** Landing expeditions will no doubt be attempted to the moons of these largest planets.

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TABLE II

NOMINAL VELOCITY REQUIREMENTS FOR LANDING MISSIONS
DESCENT FROM ORBIT TO SURFACE AND RETURN TO ORBIT*

Planets	
Mercury	19,000 ft/sec
Venus	30,000 ft/sec
Earth	32,000 ft/sec
Mars	15,000 ft/sec
Jupiter, Saturn	Velocity Requirements Exceed Technological Capabilities In The Immediate Future
Earth's Moon	10,600 ft/sec
Moons of Jupiter	
I IO	11,000 ft/sec
II Europa	9,000 ft/sec
III Ganymede	12,000 ft/sec
IV Callisto	10,600 ft/sec
Moons of Saturn	
Rhea	3,000 ft/sec
Titan**	8,000 ft/sec
Moon of Neptune	
Triton	8,000 ft/sec

* For bodies without atmospheres landing mission total ΔV assumed equal to 133% V_{escape} for body.

** For bodies having atmospheres landing mission total ΔV assumed equal to 90% V_{escape} for body.

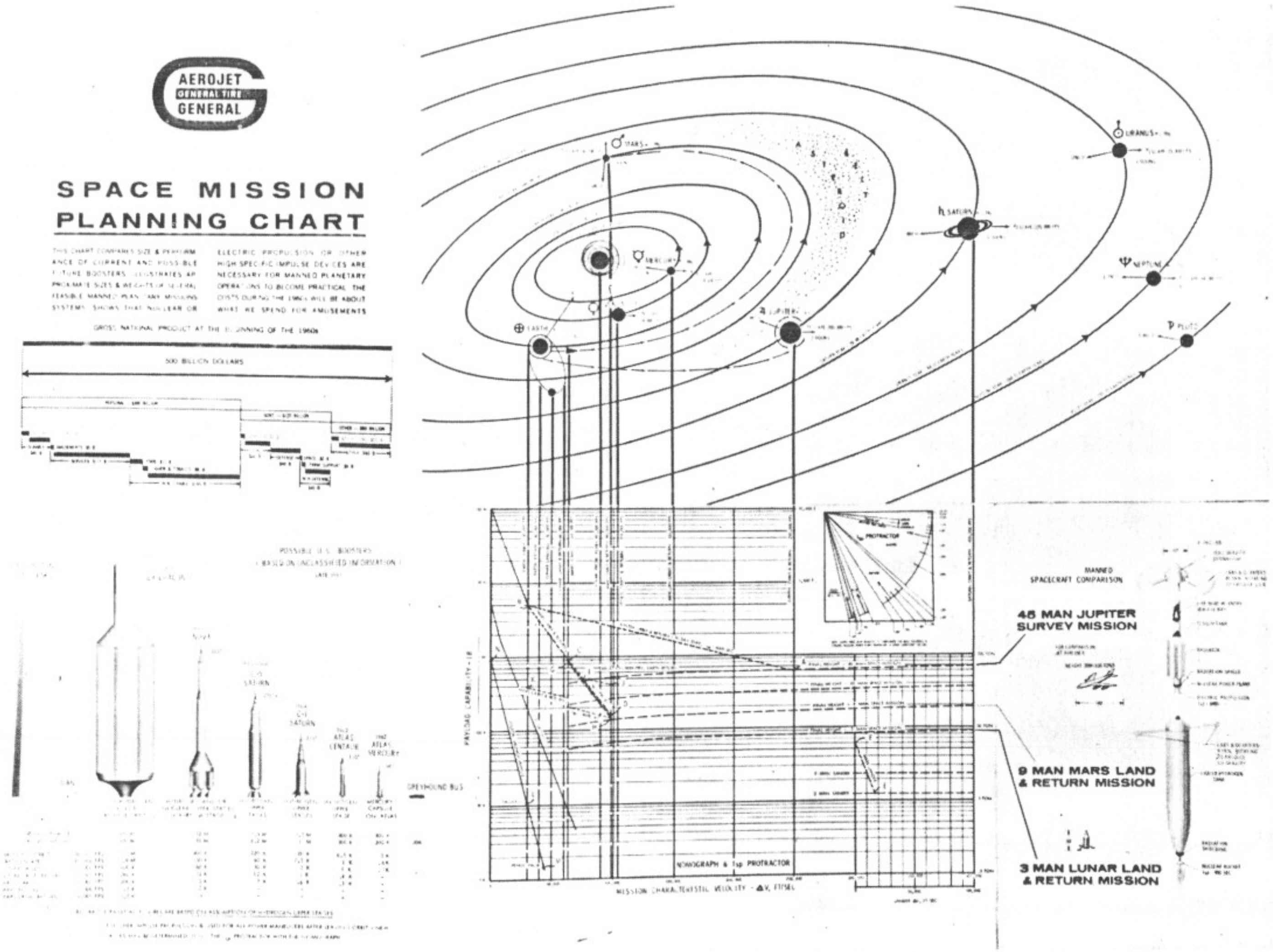
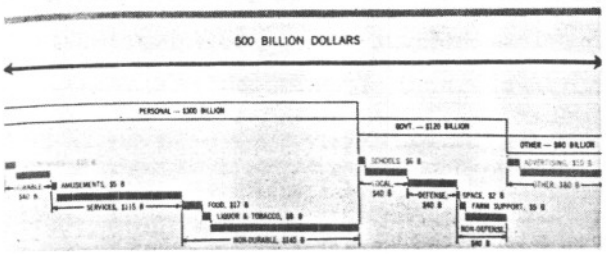
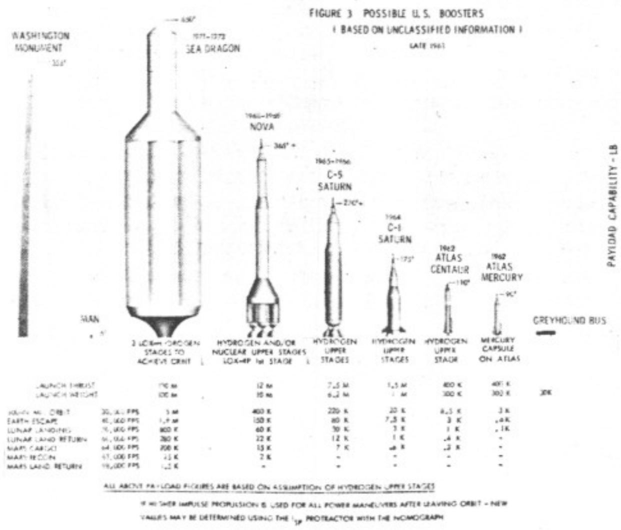
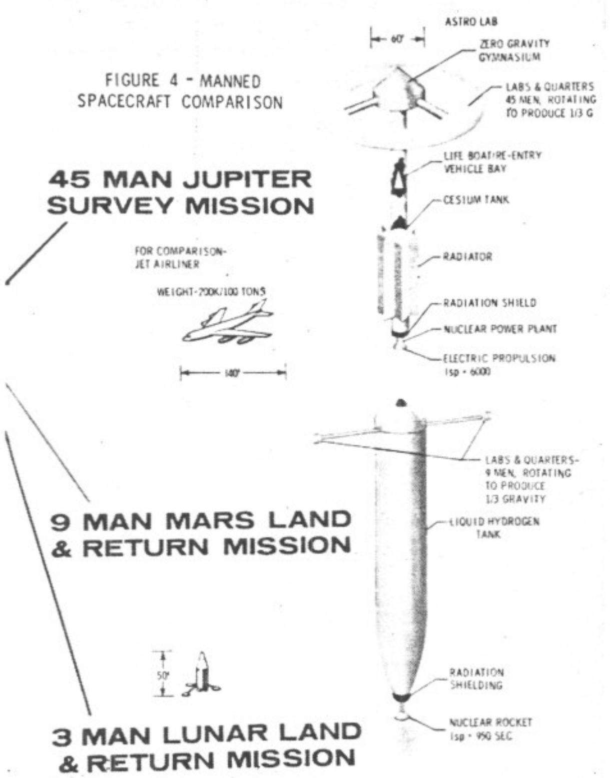


Figure 1 Early Version of Space Mission Planning Chart

FIGURE 2 - GROSS NATIONAL PRODUCT 1960







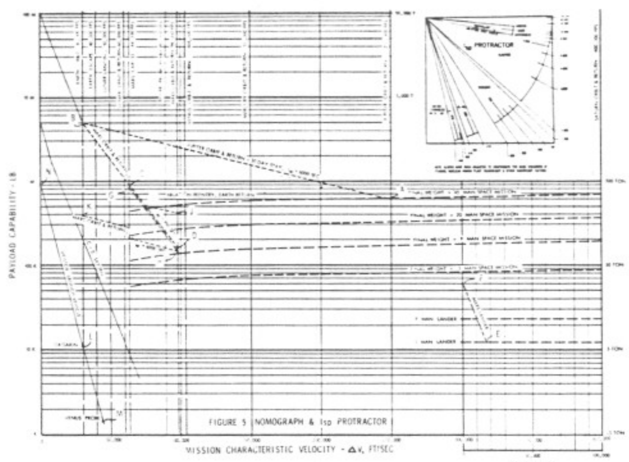


Figure 1 Early Version of Space Mission Planning Chart