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TR-59-0000-27881

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PRELIMINARY CONSIDERATIONS FOR MILITARY SPACE SYSTEMS

VOLUME I

ADVANCED WEAPON SYSTEMS STUDY
PART II

DOWNGRADED AT 12 YEAR INTERVALS; NOT AUTOMATICALLY DECLASSIFIED. DOD DIR 5200.10

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SPACE TECHNOLOGY LABORATORIES, INC.
P.O. BOX 95001
Los Angeles 45, California

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TR-59-0000-27881

Preliminary Considerations for Military Space Systems

Volume I

Work Completed August 1958

Prepared for the Air Force Ballistic Missile Division

Headquarters Air Research and Development Command

Under Contract AF 04(647)-127, Call No. 8.

Note: This report was first written in December 1957 and is presented here essentially as originally written. Only recently (October 1959) was it prepared for publication.

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SPACE TECHNOLOGY LABORATORIES, INC.
P.O. Box 95001
Los Angeles 45, California

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I. AIR FORCE SPACE MISSION REQUIREMENTS

A. Introduction

The basic Air Force space missions are indicated in Table 1. The primary one, at the present time is "Reconnaissance", both visual and electromagnetic. The second application for satellite vehicles will be "Communications", where satellite vehicles can serve as transponders for multiple military purposes. The third requirement, "Manned Space Flight", is listed as a means of supporting the Reconnaissance and Communications missions mentioned previously, and is a necessary step in the development of a future space superiority capability. The fourth area lists "Technical Development and Experimental Support", and covers the general areas of basic and exploratory research involving space flight where the Air Force presently has the unique capability to carry on research, and in the future will have military need of the knowledge to be obtained by such space explorations. A more detailed consideration of each area in Table 1 is appropriate here.

Table 1. Air Force Space Mission Requirements

1. Reconnaissance
2. Communications
3. Manned Space Flight
4. Technical Development and Experimental Support

B. Reconnaissance

The obvious application of satellites is for reconnaissance, just as it was the initial obvious application of the airplane. The current Air Force 117-L program is the first step in this direction, directed at continuously obtaining visual surveillance, and, in later versions, providing electromagnetic surveillance on a continuous basis of any desired area in the world.

Visual reconnaissance, of course, can best be conducted from low altitude satellites. Besides the direct military value obtained from surveys of ground installations, the weather data obtainable from such continuous mapping of world cloud patterns should prove of inestimable value from both a military and civil standpoint.

Primary requirements in the reconnaissance area involve establishing an initial limited capability as soon as possible, followed by a later, higher quality system. Special emphasis needs to be placed on satellite recovery systems, especially information recovery systems, the means of operating at lower altitudes for better resolution, and on the immense problems of relaying and properly using the large quantities of data obtained.

Another aspect of reconnaissance is the early warning capability that could be obtained with high altitude satellites. The promising possibility is the stationary, or 24-hour satellite which goes around the earth once per 24-hours, or at the same rate as the earth rotates. Satellites of this type, if placed in suitable spots could see all of the USSR and, using an infrared detection scheme, could ascertain whenever a large rocket was launched. Further, it might, with suitable adaptations, provide tracking information, indicate velocities gained, trajectories, and even predict impact points. The use of infrared and visual techniques would be perfectly feasible because such a satellite would be operating above the atmosphere where clouds and such interference would be non-existent. Adequate information on location, observation, tracking, and warning of such launchings could certainly be obtained with an automatic system, but the significance of such early warning data, and the potential complexity of the equipment, indicate that a manned capability existing for such a mission is highly desirable. Manned monitoring would then permit a check on any rocket launching of sufficient duration to be an ICBM, or even an IRBM. This would make possible some 25 to 30 minutes warning of ICBM launchings, as compared to the 15 minutes promised by presently proposed large ground radars. Such a space station could also provide data to aid the anti-ICBM problem, since the conditions at burn-out determine precisely the impact position of an ICBM. Hence, measurement of, and transmission of, this information to a defending area would greatly facilitate separation of the true nose cone from possible decoys. While adequate tracking would require at least two satellite stations using an infrared means, it would appear that, since the base line or separation between the two satellites could be very large, accurate impact prediction is inherently possible. In view of the need for better early warning, the very great difficulties involved in detecting differences between decoys and true nose cones after missile burn-out, and the impracticability of obtaining reconnaissance data by other means, it would appear that intensive development of both low and high altitude satellites for reconnaissance missions is of the greatest importance.

C. Communications

A major military problem, especially for the Air Force which deals with rapidly moving vehicles, is that of world-wide communications. It is of the utmost importance that reliable communications be available at all times to a widely distributed system of Air Force bases throughout the world, and to the numerous aircraft which are in the air at any particular time. While the emphasis in this plan is on space missions, it must not be forgotten that with the development of ICBM's and equivalent deterrent systems, the potential enemy will likely turn to limited or small scale wars to achieve his objectives, and that the Air Force must therefore be ready with aircraft systems capable of handling the limited war problem. A primary requirement for both limited and general war is adequate world-wide communications. At the present time such communications are quite unsatisfactory, multiple frequency bands are required, atmospheric difficulties cause serious interference, and jamming is usually not difficult. Investigations*, which will not be detailed here, have indicated that suitable radio relay stations placed in 24-hour satellites could provide reliable, jam-free communications between the United States and all other parts of the world, and particularly between the United States and all aircraft or space vehicles in flight. Much of the complication of low altitude reconnaissance satellites could be eliminated by suitable high altitude satellite relay stations. Problems of recalling or diverting a Strategic Air Command mission once launched, might be handled through communications with such a satellite. Conceivably even an ICBM mission could be set up for destruction prior to re-entry, if desired. Intercontinental television, including television from aircraft or space vehicles in flight on a real time basis would be available. With the addition of suitable electronic scrambling, noise and directional techniques already under development, effective world-wide visual and verbal communication could become a reality. The space stations would also be of enormous assistance to aircraft navigation. Intensive effort is indicated here to develop the vehicles required for the communication satellites, providing the guidance system to operate them effectively, and finally to equip them with the payloads that will make possible the achievement of the military objectives.

D. Manned Space Flight

At the present time no military space mission can be said to have been defined which absolutely requires the presence of a man for its fulfillment. In general, the missions outlined previously under reconnaissance and communications can, at least in theory, be remotely controlled and automatically operated. However, as one considers more advanced

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* Advanced Military Satellites, GM-TR-59-0000-00604.
AFDAP Advanced Weapons Systems Study, Part III

reconnaissance and communications missions which will doubtlessly have to be conducted in the future, it becomes increasingly apparent that the presence of men to repair and maintain the equipment, to observe and monitor questionable results, and to make available the human judgment factor as close as possible to the data source, will become increasingly desirable as our space capability grows. Thus, essentially, while all missions can in theory be done by machines alone, it appears that the time will come when maximum military effectiveness in space missions will probably be achievable only with manned vehicles. It must be realized that our current view of the utility of space vehicles and, particularly of manned space flight, is necessarily extremely narrow and limited. It may, by analogy, be compared with the concept of the usefulness of the airplane as a military vehicle in 1907 when Orville Wright made his first demonstrations at Fort Myers, Virginia. At that time it was conceived that the airplane would have great military value for reconnaissance, but it was doubted that it would have any other uses. While bombing was considered, what were then believed to be payload limitations, precluded its general recognition as a major aircraft application. The concept of an air battle was generally conceived to be ridiculous, with the result that aircraft entered World War I completely unarmed. As history has shown, this condition lasted for only a few days before weapons appeared on aircraft of both sides. It is, of course, debatable as to the validity of the analogy between the airplane and the space vehicle, but the many aspects of similarity during the initial period of man's venture into space would indicate that we should anticipate a need for manned space flight, and should expect new and unforeseen military exploitations of our space capability to be invented. While developments in automation may tend to reduce the number of men required for space flight, it must be recognized that automation so far has been most useful when we know exactly what we want to do, when we want to repeat the same operations many times, and when nothing unexpected is anticipated. The Air Force must always be prepared to deal with the unexpected and, while it should be planned to use automation to the maximum extent possible in dealing with the routine and expected occurrences, the flexibility of the man may still be the most advantageous way of handling highly intermittent random or unanticipated operations. In particular, in considering manned space flight there exists some small possibility that establishment of retaliatory launching stations on high altitude satellites might be desirable. Such stations, if placed at 100,000 miles altitude, would be at least 24 hours removed from any earth launched means of destruction. Hence, if they carried small ballistic missiles, they would in effect present a launching base system that could not be destroyed without the launchings against them giving 24 hours' warning of the imminent attack. Thus, while ICBM's can complete their missions in 30 minutes and conceivably, if a sufficient number were launched nearly simultaneously, they might destroy all United States launching sites in this time, the high altitude satellite vehicles would still be in place 23 hours later, and could launch retaliatory weapons. An enemy therefore would be unable to condense his war effort into a single half hour, even theoretically. He would be forced either to launch an attack against the satellites 24 hours before he launched missiles against the United States, thereby giving away his intent, or he would be faced with accepting the consequences of the retaliatory action of the space vehicles. It must be emphasized here that this discussion deals with high altitude retaliatory satellites which give an additional expansion of the time dimension to a ballistic missile war. The low altitude offensive satellite has been set aside in planning this program as inferior in most respects to the straight ICBM.

The early accomplishment of manned space flight would, of course, have tremendous military prestige value, and would assist in re-establishing United States technological leadership. While the value in the "hot" or active war sense of this accomplishment in the immediate future might be minimal, its value in achieving superiority in the cold war is unquestioned.

In view of the probable eventual military importance of a manned space flight capability, it is urgent that the current programs directed at solving the problems of manned space flight be expanded, and that the development of vehicles and hardware to make possible extended manned space missions be expedited.

E. Technical Development and Experimental Support

In order to accomplish the basic military space missions of the next decade and to be suitably prepared for the next level of developments, it is essential that the Air Force have an adequate program of basic research, research applications, and technical development in space technology. This activity must have an adequate experimental program to support the basic technical developments. As a result of the ICBM and IRBM developments, the Air Force is equipped with the basic booster vehicles to make possible the initial phases of reconnaissance, communications, and manned space flight. In addition, these vehicles can be adapted to explorations of the moon, of the nearer planets, and of the sun. The present military value of such explorations of our planetary system, except from a prestige standpoint is not clear. The scientific value, however, is enormous. Any time new scientific information is acquired it quickly turns out to have a military value, usually long before it has significant value to civilian life. Hence, the Air Force will have the vehicles and technology to make the initial explorations of our planetary system, and it is probable that the information initially gathered will be of maximum use to the Air Force and the military. Hence, the Air Force should devote a significant portion of its space effort to exploring our solar system and obtaining basic data on the moon, Mars, Venus, Mercury, and the sun. The military prestige of such accomplishments in the cold war would alone justify this effort, but the unforeseen military applications that would arise from the information obtained from the explorations and from the developments required to carry them out would assist greatly in maintaining our continued military superiority. Hence, an intensive program of space technology development, including interplanetary explorations, should be carried on by the Air Force during the next decade.

F. Technical Approach to Astronautics Development

From the technical standpoint an astronautics development plan must treat four areas:

1) The vehicles for traveling to and from space;

2) The guidance systems for controlling the vehicles;

3) The payloads carried in the vehicle for either military or scientific missions;

4) The plans for using the vehicles.

Using basic sub-system components from these areas, systems can be assembled to handle a wide variety of potential Air Force space missions. In many respects, the Air Force is well on its way towards having these space systems. The missiles presently in the Air Force ballistic missile programs can provide the basic boosters for most of the space missions. All that is needed in the vehicle area, as will become apparent from a later section, is the development of new upper stages and minor adaptations of the booster stages. In the guidance area, the ballistic missile guidance program can provide systems which, in general, can be adapted to the launch phase of space missions, and which will provide components for the later phases of space operation. However, in the cases of mid-course guidance, recovery and re-entry guidance, satellite positioning guidance, manned flight guidance, and similar areas, complete new sub-system developments will be required. Of particular significance in this area will be the requirement for attitude control systems which can operate for many months, and for other auxiliaries, such as stabilization equipment, power supplies, orbital tracking systems, and satellite positioning systems, all of which will require new developments.

In the payload area, very major developments are required. While some development with respect to the low altitude satellites has been conducted under the 117-L program, and limited thinking has been done, as indicated under the section on Mission Requirements, on other applications, very little really has been defined in the past in the way of the developments needed to make the Air Force space vehicles useful military systems. In this respect, the present program on visual and electromagnetic reconnaissance techniques for low altitude satellites needs to be greatly expanded. Extensive programs should be established for communications systems and early warning systems to operate from high altitude satellites, and possibly for retaliatory missiles to operate from space launchers. The requirements for, and possibilities of, the anti-satellite-satellites need investigation, and it can be expected that numerous other applications requiring development of special space payloads will be invented when the vehicle systems become available. In addition to the payloads for the immediate military applications, development of payloads for scientific missions must be undertaken. Payloads for explorations of the moon, Mars, Venus and the sun must be developed, including means of transmitting the data in suitable form back to earth. Many types of measurements will have to be made with very little weight for instrumentation, and transmitted effectively with low powers over great distances. Payloads are not, as mentioned, discussed in this report. Dates are indicated for what are expected to be the earliest applications. Formulation of plans for payloads for other applications can be expected as the basic vehicle and guidance capability comes into existence.

II. ASTRONAUTICS VEHICLE DEVELOPMENT PLANS

A. Equivalent Velocity Requirements for Space Vehicles

Figure 1 shows the equivalent velocity required for various space missions. By "equivalent velocity" is meant the velocity that would be achieved by the payload if the entire vehicle were used to carry the payload to as high a velocity as possible. In reviewing the figures it will be noted that a 5,000-mile ICBM must reach approximately 22,000 ft./sec., while the most severe mission listed on the charts is a soft landing on the moon and return to earth, which requires 53,000 ft./sec., or a little over twice as much as an ICBM. Impact on the moon, flight around the moon and return to earth, or flight to Mars or Venus, require velocities of only thirty-five to forty thousand feet per second, or a little more than 50% increase over an ICBM. To reach an ICBM capability, the Air Force has had to increase flight speeds from the Mach 2 to 3 range of air-breathing vehicles and early air-launched rockets to approximately Mach 23, a gain in velocity of about one order of magnitude. To perform the most difficult space mission listed in Figure 1 requires only double the ICBM velocity and, hence, is not nearly the development problem from the vehicle standpoint it might initially appear to be. As will be seen from a later discussion, this relative easing of the development problem is true only if the space capability is derived by the use of the Air Force ballistic missiles as boosters of new upper stages, and if the Air Force ballistic missile technology is fully applied to the development of the astronautics capability.

B. Space Capabilities of Existing Air Force Boosters

Formulation of a vehicle plan for astronautics begins with the study of what can be done using the existing Air Force ballistic missiles adapted to space applications.* The three basic vehicles considered for this application are the

* Many of the vehicles discussed in this section are analyzed in detail in Appendix A.

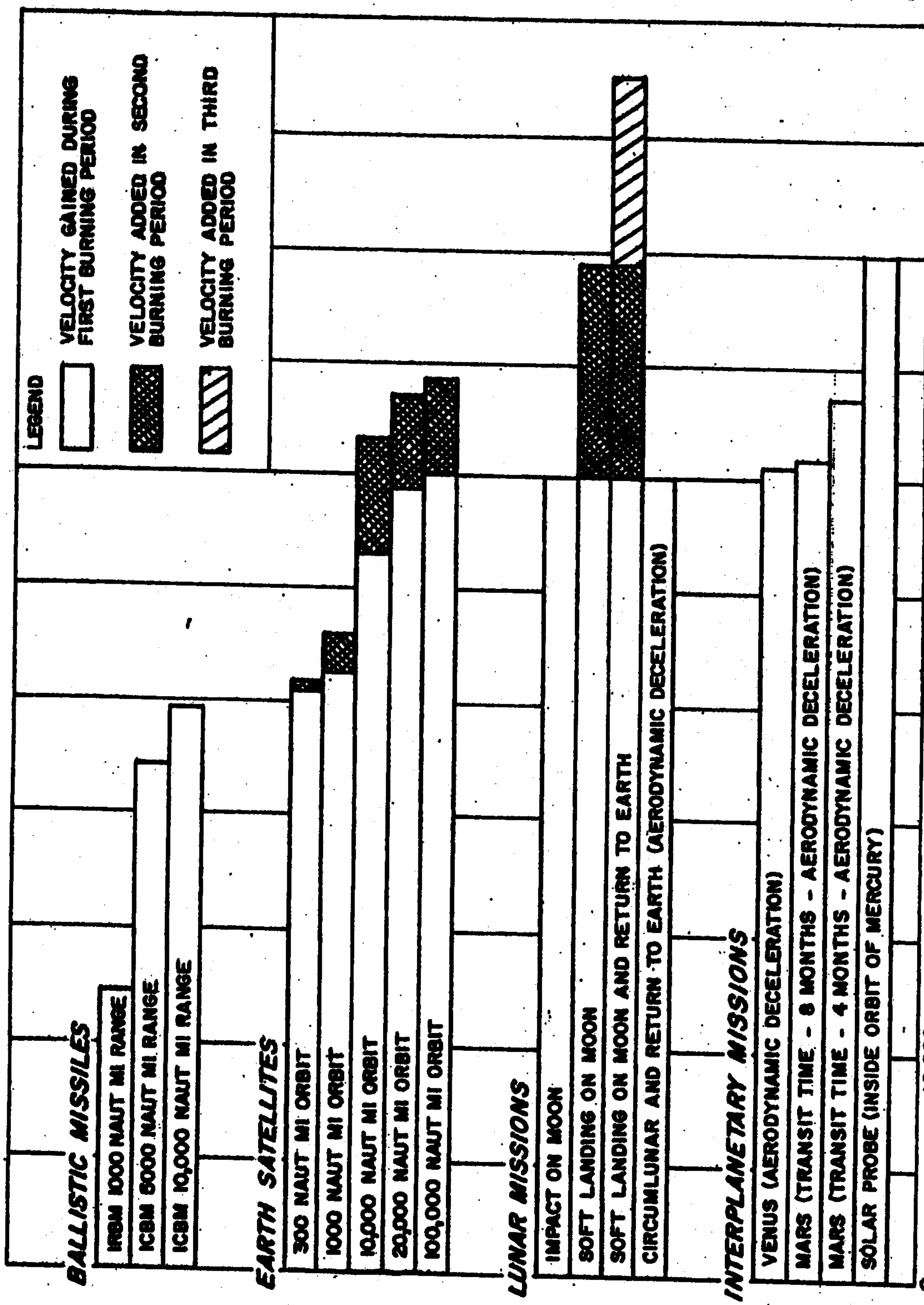
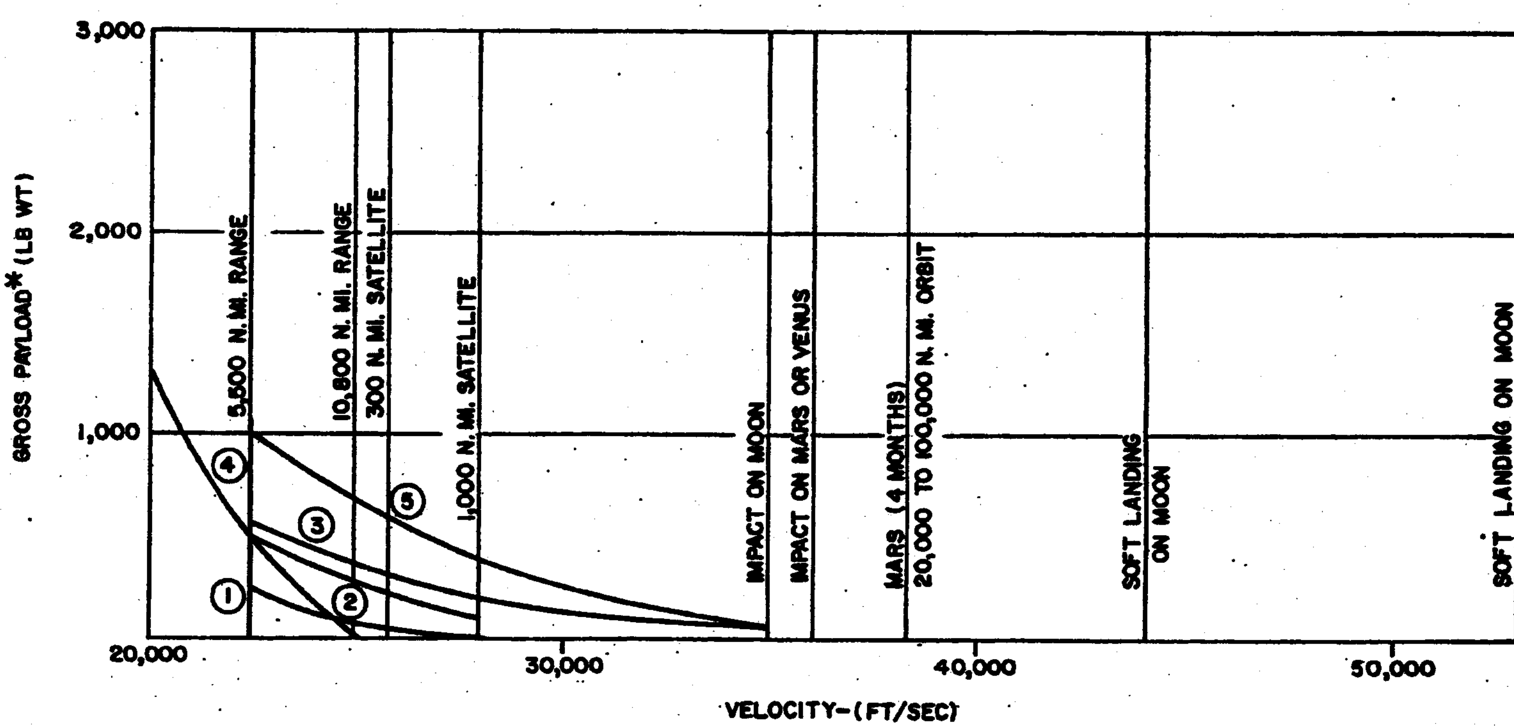


Figure 1. Equivalent Velocity Required for Various Missions.

Thor, the Atlas and the Titan. Each has been studied with added stages and with modifications of its existing stages in an attempt to cover the major spectrum of space potential for these vehicles. The studies for each vehicle have included combinations with additional stages of gradually improving performance capability consistent with availability of the basic vehicle, the development time for added stages and refinements, and with optimization of the total vehicle system. The study of Thor capabilities is perhaps most representative of the range covered, and the results of the study are indicated by Figures 2 and 3, Figure 2 showing the Thor capabilities with added stages of solid propellant, while Figure 3 indicates the capability obtained with liquid propellant added stages. Nine additional stage combinations have been treated, and the payload versus equivalent velocity determined, from which the applicability to various possible missions can be readily evaluated. The added stages considered range from a single Vanguard third stage solid rocket of the type that exists today, through a 15,000 lb. gross weight pressurized fluorine** stage that would require nearly 20 months for development. A similar chart is shown in Figure 4 for the Atlas. Structural limitations presently preclude more than 9,300 lbs. in a stage on top of the Atlas and, hence, the study was limited to this weight. This limit could be raised most efficiently by changing the tank shape from its present tapered nose to a cylindrical nose, which would require new tooling and design modifications. Alternatively, the allowable upper stage weight could be increased by thickening the forward skins and providing for higher pressurization of the LOX tank.

** Fluorine has been used in this report as a basic high performance oxidizer to be used with various fuels. Other high performance propellant combinations have been evaluated. Some were not considered satisfactory for theoretical reasons and others, including the extremely high performance H_2-O_2 combination, were unsatisfactory for practical reasons. Therefore, if substantial engineering advances are made, some combinations may become much more attractive.

	1ST STAGE	2ND STAGE	3RD STAGE
1	THOR	1 VANGUARD-3RD STAGE	—
2	THOR	4 VANGUARD-3RD STAGE	—
3	THOR	4 VANGUARD-3RD STAGE	1 VANGUARD-3RD STAGE
4	THOR	SERGEANT	—
5	THOR	XM-34	1 VANGUARD-3RD STAGE



*GROSS PAYLOAD INCLUDES ALL FIXED EQUIPMENT, GUIDANCE, ETC, BUT DOES NOT INCLUDE THE BARE STRUCTURE OF THE FINAL STAGE

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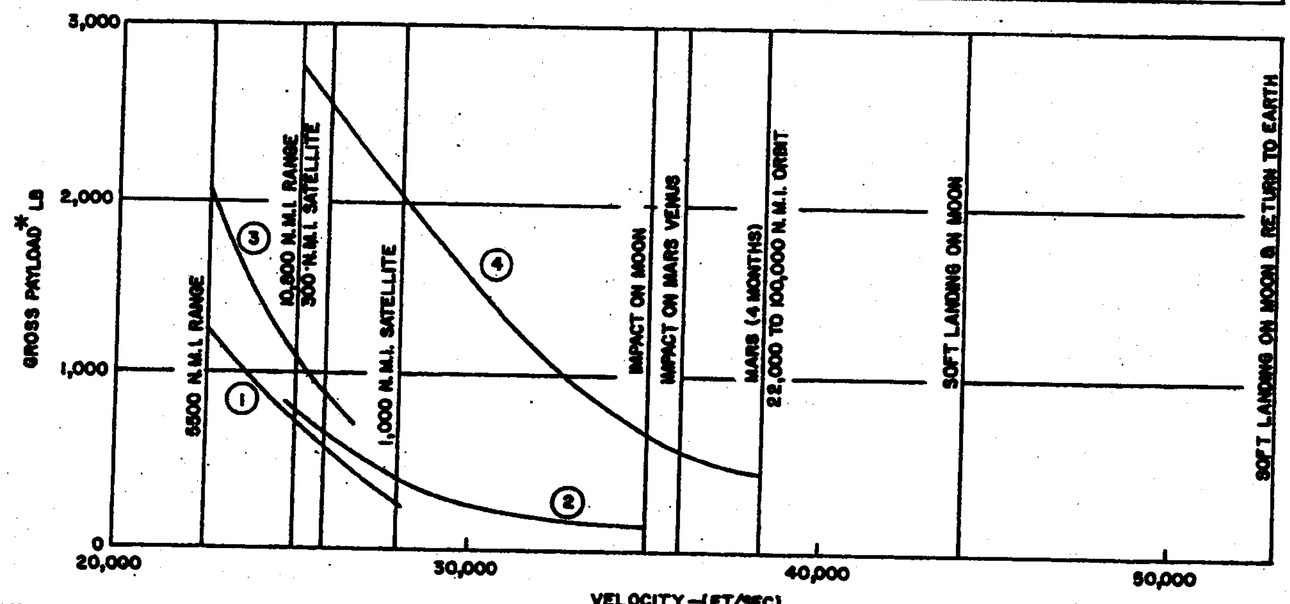
Figure 2. Thor Capability with Added Solid Propellant Stages.

[redacted]

	1 ST STAGE	2 ND STAGE	3 RD STAGE
1	THOR	VANGUARD 2 ND STAGE	—
2	THOR 175,000 LB THRUST	VANGUARD 2 ND STAGE	VANGUARD SOLID 3 RD STAGE
3	THOR 175,000 LB ENGINE REDUCE FUEL TANK MILLING	15,000 LB STORABLE YLR-81 ENGINE	—
4	THOR 175,000 LB ENGINE REDUCE FUEL TANK MILLING	15,000 LB F_2-NH_3 WEIGHT 15,000 LB	—



*GROSS PAYLOAD INCLUDES ALL FIXED EQUIPMENT, GUIDANCE, ETC, BUT DOES NOT INCLUDE THE BARE STRUCTURE OF THE FINAL STAGE

Figure 3. Thor Capability with Liquid Propellant Second Stages.

[redacted]

LEGEND	
1	4300 LB VANGUARD 2ND STAGE "AS IS"
2	9300 LB STAGE VANGUARD 2ND STAGE ENGINE
3	9300 LB STAGE YLR-81 ENGINE 15,000 LB THRUST
4	9300 LB F_2-NH_3 STAGE 3-10,000 LB THRUST 1200 LB STORABLE STAGE 4-1000 LB THRUST
5	SAME AS CASE D WITH ADDED 600 LB STORABLE STAGE 5-600 LB THRUST

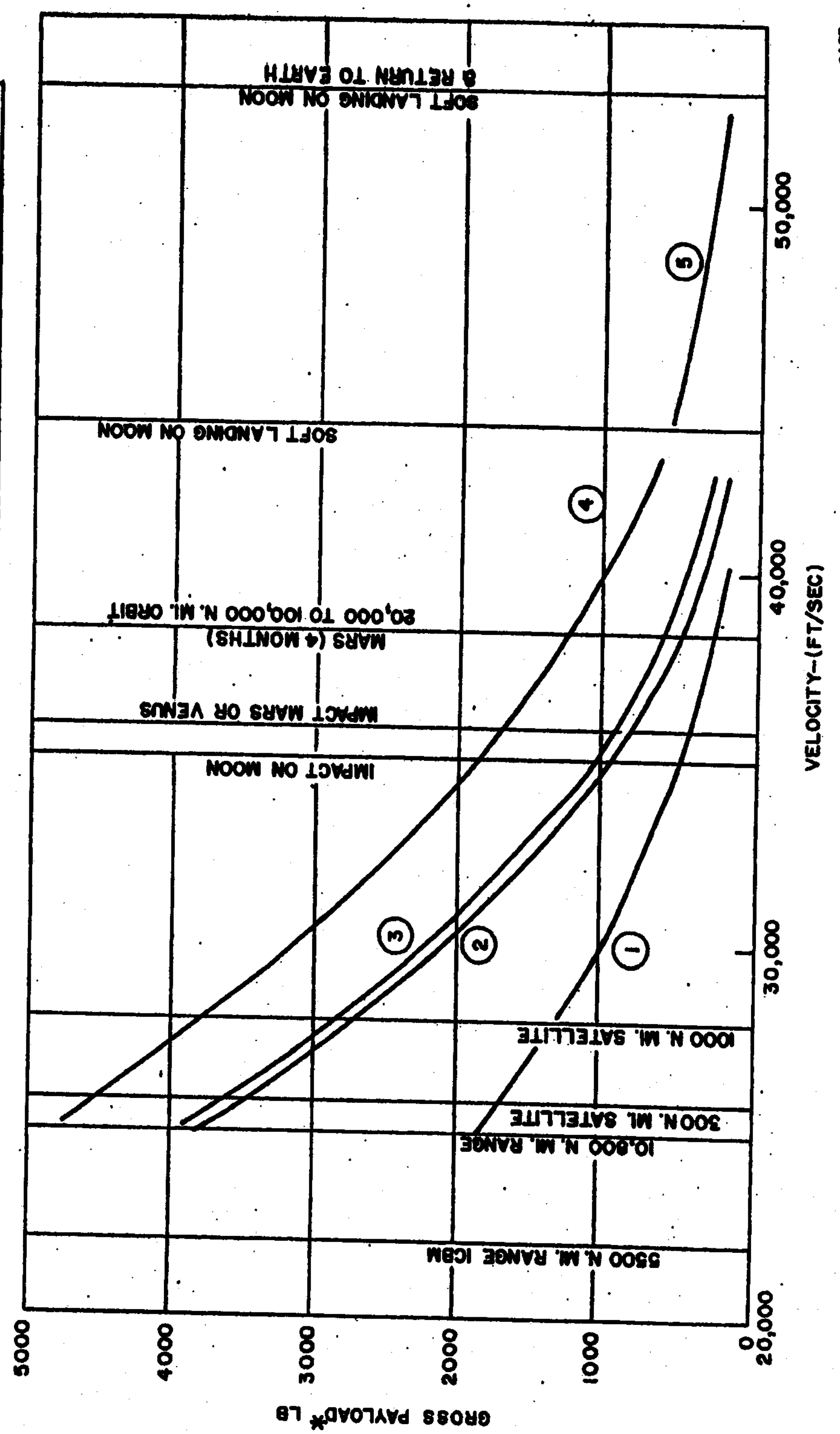


Figure 4. Atlas Capability with Added Stages.

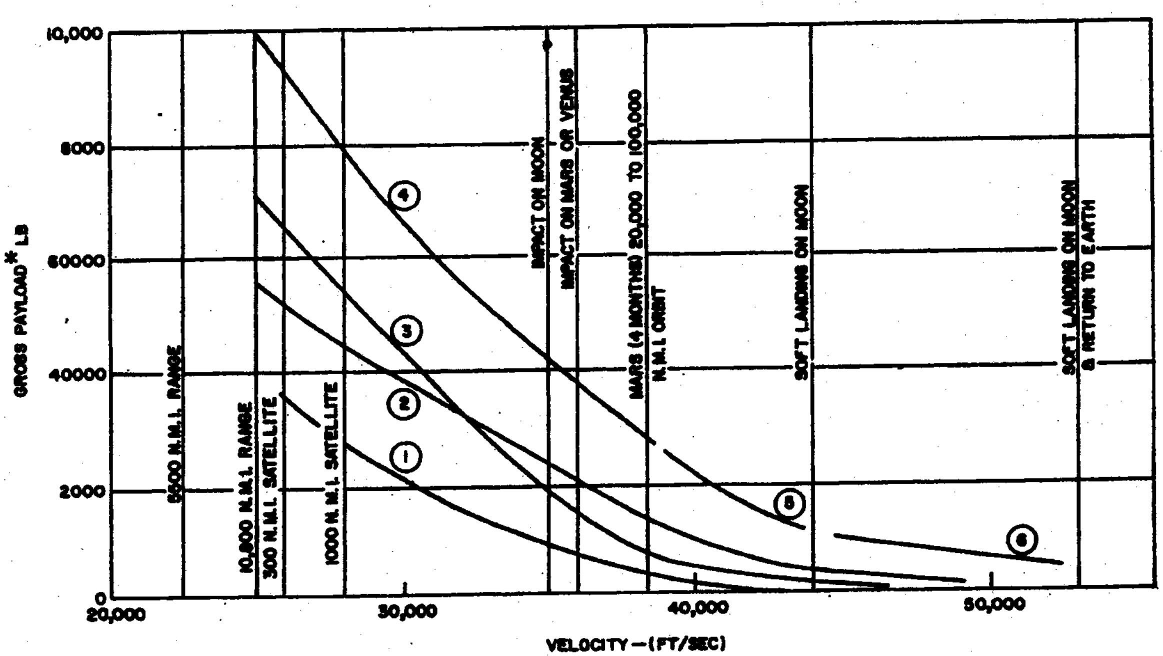
Figure 5 shows the Titan capability with added stages. The Titan, a two-stage missile with a very ruggedly designed booster stage, is particularly adaptable to third stage addition and to second stage modification for space missions. Two cases shown on Figure 5 are of particular interest. Case 2, where the Titan is used essentially as is, with a third fluorine stage of 15,000 lbs. gross weight and 10,000 lb. thrust (identical to the stage mentioned earlier for use on Thor), has many attractions. Case 4, in which the booster engine is uprated to 400,000 lbs., has a second stage adapted to fluorine and uses an enlarged version of the Thor fluorine stage (stretched to 20,000 lbs.) for Stage 3. Certain factors need to be discussed with respect to this combination. First the booster engine, which presently provides 300,000 lbs. thrust, obtained by two 150,000 lb. units, has already had tests run on a single unit at 194,000 lbs. thrust. A total testing time of 190 seconds at the high thrust level, with one run of 110 seconds' duration has been accumulated without difficulty. Hence, it appears that the development of this engine to the 400,000 lb. thrust level could be readily integrated into the main Titan propulsion development program. The change-over of the second stage to fluorine is somewhat more of a problem. Two cases have been considered:

1) Converting the present 80,000 lb. (altitude thrust) engine to fluorine, and

2) Developing a pressurized second stage operating at low chamber pressure.

The use of the pressurized stage, while slightly less efficient, avoids the problems of the turbo-pump system, which are expected to be considerable with fluorine. In addition, it is anticipated that it would greatly reduce the amount of testing required to get a suitable unit. The third stage, of 20,000 lbs. weight, is assumed here to be simply an enlarged version of the 15,000 lb. fluorine stage proposed for Thor and the unmodified Titan. The size increase would be achieved by lengthening the cylindrical portion of the pressurized tanks and increasing the pressure in the helium storage sphere. The total vehicle for Case 4 using the uprated booster, the 110,000 lb. fluorine second stage and the 20,000 lb. fluorine third stage, is referred to as Super-Titan for convenience. It will be noted from

	STAGE 1	STAGE 2	STAGE 3
1	TITAN BOOSTER AS IS	TITAN STAGE 2 STIFFEN INTERTANK STRUCTURE	15,000 LB STORABLE YLR-81 ENGINE
2	TITAN BOOSTER AS IS	TITAN STAGE 2 STIFFEN INTERTANK STRUCTURE	15,000 LB F_2-NH_3 10,000 LB THRUST
3	TITAN BOOSTER 400,000 LB ENGINE STIFFEN INTERTANK STR	110,000 LB F_2-NH_3 80,000 LB THRUST	15,000 LB STORABLE YLR-81 ENGINE
4	TITAN BOOSTER 400,000 LB ENGINE STIFFEN INTERTANK STR	110,000 LB F_2 NH_3 80,000 LB THRUST	20,000 LB F_2 NH_3 10,000 LB THRUST
5	SAME AS CASE 4 BUT WITH 3600 LB STORABLE 4TH STAGE		
	SAME AS CASE 5 BUT WITH 1200 LB STORABLE 5TH STAGE		



*GROSS PAYLOAD INCLUDES ALL FIXED EQUIPMENT, GUIDANCE, ETC, BUT DOES NOT INCLUDE THE BARE STRUCTURE OF THE FINAL STAGE

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Figure 5. Titan Capability with Added Stages.

Figure 5 that Super-Titan has an excellent capability; it can put approximately 10,000 lbs. into a low altitude orbit, impact 4,200 lbs. on the moon and nearly as much on Mars or Venus, and with the addition of suitable fourth and fifth stages it can land sizeable payloads on the moon and return them to earth. These payloads are sufficient for most of the military space missions discussed in Chapter I, and provide a major capability for exploration of the solar system.

C. Military Space Missions

Tables 2 through 4 and 7 through 11 compare the capabilities of the various basic booster upper stage combinations for a number of space missions. Where Figures 2 through 5 permit a comparison of different cases for the same basic booster, this set of mission figures permits comparison of booster/upper stage vehicle combinations for specific missions. The first case, Table 2, is for a low altitude satellite, which is suitable for visual and electromagnetic reconnaissance. Check marks by the numbers in the left-hand column indicate the cases which appear to be of primary interest for application.

The first of these is Case 4, which uses the Thor plus a Vanguard second stage, and would put 600 lbs. in a low-inclination orbit. A somewhat smaller payload, about 400 lbs., could be put in a polar orbit. This payload is sufficient to permit consideration for photo reconnaissance with recovery, tests of re-entry from satellite orbits, and animal experimentation on space flights with recovery. The Thor-Vanguard combination, being capable of delivering 1,300 lb., 5,500 nautical miles, is also suitable as detailed elsewhere in this test plan for re-entry test vehicles, and as a small ICBM.

The next combination of interest is the Thor with a 15,000 lb. pressurized fluorine stage. This combination is capable of putting 2,800 lbs. into orbit. The Atlas, with the presently planned 117-L 9,300 lb. stage (Case 7 of Figure 7), can put 3,200 lbs. into orbit. The great potential of using high-energy propellants in upper stages is indicated here. It should be recalled that an increase of a point of specific impulse in the final stage of a multi-stage vehicle reduces the launching gross weight for a given payload by the same percentage as a point of specific impulse in the booster stage. Hence, in developing space

Table 2. Low Altitude Satellites

	Stage I	Stage II	Stage III	Gross Payload*	Guidance Considerations
1	THOR "as is"	1 VANGUARD (Stage III)	---	50-100 lb (no guidance)	Autopilot only in THOR spun - Stage II 160 mi eccentricity
2	THOR "as is"	4 VANGUARDS (Stage III)	---	200-400 lb (no guidance)	Inertial guid. on THOR spun - Stage II 70 mi eccentricity
3	THOR "as is"	Sergeant	---	0	
4	THOR "as is"	VANGUARD (Stage II)	---	600 lb	
5	THOR 175,000 lb	15,000 lb storbl. YLR - 81 engine	---	1,000 lb	
6	THOR thrust 175,000 lb thicker fuel tanks	wt 15,000 lb F_2-NH_3	---	2,800 lb	
7	ATLAS boost stage "as is"	ATLAS sustainer "as is"	wt 9,300 lb storable propellants I_{sp} = 270 sec	3,200 lb	<u>Cases 3-10</u> Guidance in final stage <u>Guidance Weights</u>
8	ATLAS boost stage "as is"	ATLAS sustainer "as is"	wt 9,300 lb F_2 NH_3	4,760 lb	. Radio system~175 lb . All inertial sys.~460 lb
9	TITAN booster "as is" 300,000 lb thrust	TITAN Stage II "as is" (intertank struc. reinfor.)	wt 15,000 lb F_2 NH_3	5,900 lb	. Power supply~50 lb
10	TITAN booster airframe "as is" eng. uprated to 400,000 lb	wt. 110,000 lb F_2 NH_3	wt. 20,000 lb F_2 NH_3	10,000 lb	

*See Figure 2:

vehicles the optimum system will usually be realized when the upper stages have propulsion systems with maximum performance. While the lower stages must be large, the propulsion systems are developed primarily with respect to cost, reliability, non-toxicity (because the exhaust products discharge in the launch area), and ease of handling.

Should Case 6 of Table 2 prove feasible, it might very well become more attractive than Case 7 for the 117-L mission. The less than 5,000 lbs. of fluorine required for Case 6, even at the current exorbitant $5.00 per pound should be cheap when compared with the difference in cost between the basic Thor and Atlas boosters and numerous other items which enter into the reliability area.

Case 7, the Atlas plus the 9,300-pound nitric acid-RP stage, using the YLR-81 engine, is of interest primarily because the combination can be available somewhat earlier than the Thor-fluorine combination. Hence, Case 7 will permit early development of the full 117-L capability, including radio link transmission of the data to the ground.

Case 9, the Titan essentially as is, but with a 15,000 lb. fluorine third stage, puts 5,900 lbs. into low altitude orbit, or over twice the Thor-fluorine capability. Hence, the Titan with its fluorine third stage would appear to be the preferred combination for low altitude missions, where payloads of about 6,000 pounds are desired. Such payloads could obviously include manned satellite missions.

Case 10, the Super-Titan, with fluorine in both the second and the third stages, takes maximum advantage of the high energy available with this oxidizer and hence would be able to put 10,000 lbs. into a low altitude orbit. While there is no clear reason at present for such large payloads, it appears probable that developments in reconnaissance might make it desirable to carry heavy equipment to provide the desired resolution.

To summarize for the low altitude mission, there are five vehicle combinations of interest. Two, the Thor plus Vanguard second stage and the Atlas - 117-L combination are of interest primarily because of early availability. The other three, the Thor plus a fluorine second stage, and the two Titan-fluorine combinations, are of interest as giving three different payload combinations respectively of approximately 3,000, 6,000 and 10,000 lbs. on a later time scale. The present ICBM guidance is more than adequate with respect to performance for these missions. It might be preferable to take somewhat less accurate guidance and reduce the weight by eliminating some components of the present systems.

D. High Altitude Satellites

Table 3 shows the capabilities of the basic booster adaptations for high altitude satellites. Included here at approximately 25,000 miles altitude is the 24-hour satellite, or satellites which will stand over one point at the equator. By tilting the orbit, such satellites can be made to oscillate in a north-south pattern, retaining essentially their east-west position. Three cases are of interest here - the Thor-fluorine combination (Case 1); the Titan plus a fluorine third stage (Case 4), and the Super-Titan (Case 5). Case 1 for the Thor is perhaps of marginal interest, having a gross payload, including guidance, of a little under 500 lbs. However, the early availability possible for this combination makes it of interest for initial exploration of 24-hour satellite problems and capabilities. The main reliance later for this mission would tend to be on either the Titan combination (Case 4), or the Super-Titan (Case 5). The Super-Titan with a payload of 2,700 lbs. should make possible limited manned flight missions at this altitude. Particular consideration might be given to the use of the Super-Titan, manned for servicing or repair of 24-hour satellites. Thus, it would appear that the full early warning and communication satellite capabilities discussed in the chapter on "Missions" could be realized with the three-stage Titan and Super-Titan serving as vehicles. The present ICBM all in-inertial guidance system has more than sufficient accuracy to establish the basic orbit, but is heavier than would be desired. Also, new guidance systems, will be required to navigate into the 24-hour orbit and hold the satellite there. Very high impulse propulsion systems, possibly of the ionic type, with very low thrust will also be needed as part of the payload. The payload itself is, of course, a major development project.

E. Manned Exploration of Satellite and Re-entry Environment

Table 4 tabulates the weights of manned satellite re-entry vehicles which adaptation of the Atlas and Titan could launch for manned exploration of space environment.

Figure 6 illustrates the preliminary design concept of a manned re-entry test vehicle with a gross weight of approximately 3,000 lbs. The seat is swiveled mounted so that the man will experience acceleration in the same direction during the launch period as during the re-entry period. The vehicle is provided with ejection and de-boost rockets whose primary function would be to decelerate the body from the satellite orbit so that it would re-enter the atmosphere. The secondary functions of these rockets is emergency escape. Should difficulty be experienced with the booster during the ignition or early launch phase, the ejection rockets would be able to raise the re-entry vehicle to over 5,000 ft. where the recovery system parachute could go into action providing thereby for the occupant's escape. Re-entry from space flight would be initiated at a very flat angle, preferably less than 2° in order to hold decelerations down to the 15 g or less range tolerable to men. As indicated by Table 4, the re-entry vehicle could be launched by Atlas alone to velocities that would permit a number of minutes of free flight. If full advantage were taken of the earth's rotation from Cape Canaveral, a few satellite orbits might be made before re-entry. The capability of Titan without additional stages is essentially the same as Atlas, perhaps slightly better, based on present performance indications. Either Atlas or Titan with an additional stage under the manned vehicle are capable of launching much larger payloads into satellite orbits. The payloads listed for Titan in Cases 3 and 4 are essentially those that can be returned to ground, the weight of the heat-shield for re-entry and its attendant structure having been separately accounted for. Hence, the Titan of Case 3 would permit manned flight with at least two men, and the Super-Titan of Case 4 could provide equipment and supplies for explorations extending into durations of months.

In planning boost vehicles to be used in manned exploration of space, the initial choice is likely to be settled by conditions of availability and booster

Table 4. Manned Exploration of Satellite and Re-entry Environments.

	Stage I	Stage II	Stage III	Trajectory	Gross Payload*
1	ATLAS boost stage "as is"	ATLAS sustainer "as is"	None	$V_B \cong 24,000$ ft/sec	2,000 lb.
2	ATLAS boost stage "as is	ATLAS sustainer "as is"	wt. 9,300 lb storable ($I_{sp} = 270$) propellants	200 mi altitude satellite	3,500 lb.
3	TITAN booster "as is"	TITAN Stage II "as is," intertank stiffening	wt. 15,000 lb. $F_2 - NH_3$	300 mi altitude satellite	3,800 lb.
4	TITAN booster airframe "as is" 400,000 lb. engine	wt. 110,000 lb. $F_2 - NH_3$	wt. 20,000 lb. $F_2 - NH_3$	300 mi altitude satellite	6,500 lb.

*See Figure 3

NOTE: 1. Re-entry angle approximately 2°, ~5 minutes free flight

2. Re-entry angle approximately 2°, hrs to days of free flight

3. Time in orbit ~ days

4. Time in orbit ~ weeks

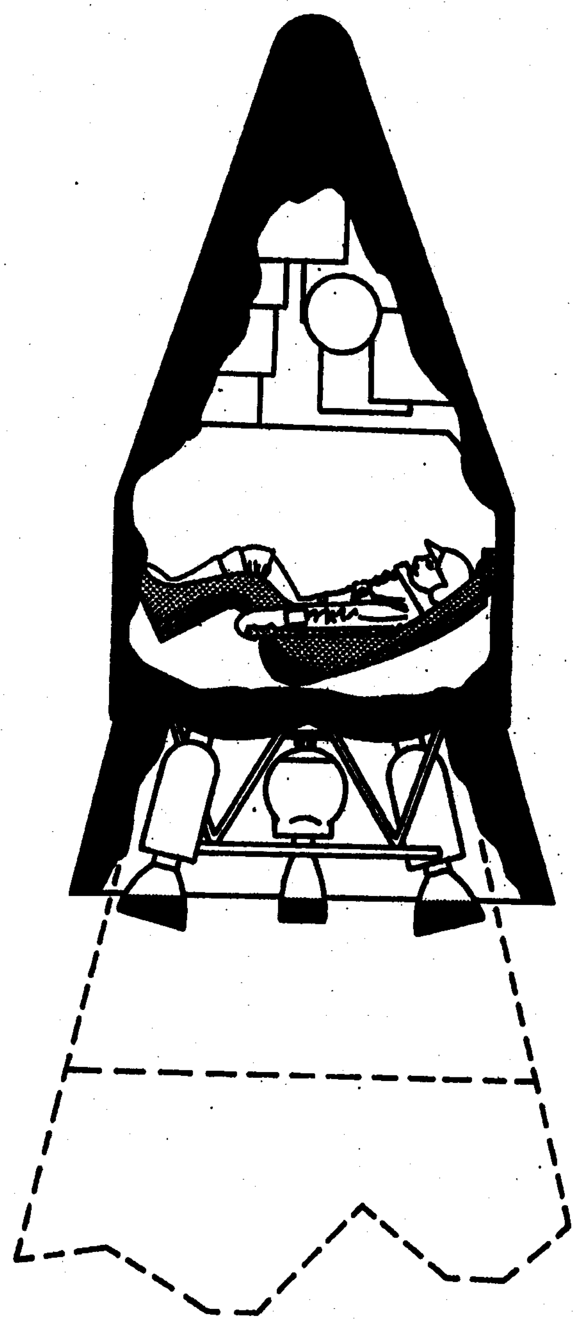

Figure 6. Manned Re-entry Test Vehicle Capsule.

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reliability reached at the time. Since Atlas goes into test first, it is likely to have an initial reliability lead over Titan. Hence, Atlas may very well be the choice for the initial tests without an additional stage. For those tests requiring additional stages, however, or needing additional payload, the Titan might be better. Accordingly, provisions will be made in planning on development programs for manned space explorations so that either Atlas or Titan may be selected for the initial investigation, with Titan preferred for the later tests of longer duration and larger payloads.

F. Impact on Moon

Table 5 compares the various space vehicles for impacting a payload on the moon. The interesting cases are Case 3 for the Thor with Vanguard second and third stages, Case 5 for the Thor plus the fluorine second stage, and the three-stage Titan and Super-Titan cases. Investigations made since Figure 2 was prepared have shown that if Case 3 is used with a spun third stage incorporating an integrating accelerometer to achieve thrust cut-off by blowing off the nozzle, the combination will put approximately 100 lbs. of payload, above these guidance weights, on the moon. The probability of hitting the moon with this guidance system is greater than 90%. While small, this payload permits such experiments as measurement of the moon's magnetic field (whose very existence is a subject of major scientific controversy), flash powder signals upon moon impact, or if safety considerations could be met, the explosion of a small atomic bomb on the moon's surface for spectroscopic data. Alternatively, a small TV camera and transmitter might be sent around the moon. Such a TV camera could also be used to view the earth from space and obtain basic data on atmospheric circulation, cloud patterns, etc.

The next case of interest is Case 5, the Thor-fluorine combination which could put 650 lbs. on the moon. This is sufficient to use guidance systems capable of selecting particular regions of the moon for exploration as well as permitting adequate marking of the impact for visual observation from earth. The Titan and Super-Titan three-stage combinations are able to impact very large payloads on the moon. Since it is questionable what purpose would be served by such impact, these payloads are perhaps of more interest for the

Table 5. Impact on Moon

	Stage I	Stage II	Stage III	Gross Payload*
1	THOR (guidance only "as is" in THOR)	4 VANGUARDS	1 VANGUARD	50-100 lb. (no guidance)
2	THOR	XM-34	VANGUARD Stage III	50 lb
√3	THOR 175,000 lb.	VANGUARD Stage II	VANGUARD Stage III	160 lb
4	THOR 175,000 lb.	15,000 lb.-storable YLR-81-engine	- - -	0
√5	THOR thrust 175,000 lb. thicker fuel tank	wt. 15,000 lb. F_2-NH_3	- - -	650 lb
√6	ATLAS boost stage "as is"	ATLAS sustainer "as is"	wt. 9,300 lb. I_{sp} = 270	700 lb
7	ATLAS boost stage "as is"	ATLAS sustainer "as is"	wt. 9,300 lb. F_2-NH_3	2,000 lb
√8	TITAN booster "as is"	TITAN Stage II "as is" intertank reinforcement	wt. 15,000 lb. F_2-NH_3	2,150 lb
√9	TITAN booster airframe "as is" thrust 400,000 lb.	wt. 110,000 lb. F_2-NH_3	wt. 20,000 lb. F_2-NH_3	4,200 lb

*See Figure 3.

NOTE: Case 1 thru 3: Between 10% and 50% chance of hitting moon

Cases 5 thru 9: Present radio guidance system (175 lb. + 50 lb. P. S.) would produce produce C. E. P. on moon ≅ 70 mi
Present all inertial guidance (460 lb. + 50 lb. P. S.) would produce CEP on moon ≅ 70 mi

missions to be discussed later. Table 5 indicates that the present GE guidance system would produce a cep on the moon of approximately 70 miles, and would do this for a weight of about 225 lbs. Hence, for all cases except the earliest Thor-Vanguard combinations only a small percentage of the payload would be needed for guidance. For purposes of development planning, it would appear that the use of Case 3, placing an early payload on the moon for basic scientific data and the accompanying international prestige, should be included. Subsequently Case 5, the Thor-fluorine combination, would serve for more extensive lunar impact exploration.

G. Circumlunar Flight and Soft Landing on Earth

Table 6 summarizes a number of cases for circumlunar flight with return to earth through aerodynamic re-entry and deceleration. The accuracy problem in returning to earth is rather severe, particularly if recovery is to be assured. Hence, only cases with payloads of at least several hundred pounds can be considered practical. While not included in Table 6, the Thor-Vanguard second and third stage combinations of Table 5 could take a 75 lb. payload around the moon and back towards earth. The probability of hitting the earth's atmosphere, however, would be so small that the circumlunar data might better be transferred by radio, rather than package recovery. Case 2 of Table 6 for the Thor-fluorine combination would put 400 lbs of payload back on the surface of the earth. These payloads, while they include final stage guidance, are above and beyond the weight of the structure and heat sink to achieve atmospheric re-entry. The present ICBM radio guidance is adequate to assure hitting the earth on return, although the use of corrections during the circumlunar phase would greatly facilitate hitting a particular region on earth. The three-stage Titan could deliver 1,350 lbs. back to an earth landing while the Super-Titan could bring back 2,700 lbs. The latter would appear to be sufficient to permit two men to make the five-day journey around the moon while carrying the necessary guidance and vernier system for a tangential re-entry to the atmosphere and recovery. Hence, the Super-Titan has the potential of gratifying the centuries old dream of man of actually flying around and looking at the back side of the moon.

Table 6. Circumlunar Flight and Soft Landing on Earth (Aerodynamic Deceleration)

	Stage I	Stage II	Stage III	Gross Payload* To Earth**
1	THOR "as is"	4 VANGUARDS	1 VANGUARD	50 lb. (marginal)
√2	THOR thrust 175,000 lb., thicker fuel tank	wt 15,000 lb. F_2-NH_3	(Three stage would be better)	400 lb.
3	ATLAS booster stage "as is"	ATLAS sustainer "as is"	wt. 9,300 lb. I_{sp} = 270 storable propellants	0 lb.
4	ATLAS booster stage "as is"	ATLAS sustainer "as is"	wt. 9,300 lb. F_2-NH_3	1,250 lb.
√5	TITAN booster "as is"	TITAN Stage II "as is" intertank stiffening	wt. 15,000 lb. F_2-NH_3	1,350 lb.
√6	TITAN booster airframe "as is" thrust 400,000 lb	wt 110,000 lb. F_2-NH_3	wt. 20,000 lb. F_2-NH_3	2,700 lb.

* See Figure 3
** Guidance Considerations
Case 1: THOR guidance in Stage I. Negligible probability of hitting earth on return.
Cases 2-6: Present radio guidance (170 lb + 50 lb P. S.) in Stage III (II for case 2) would produce ± 700 mile error in distance from back of moon. Error on return to earth would be about ± 800 miles on earth cross section. Error in impact point would be considerably greater.

H. Soft Landing on the Moon

Table 7 summarizes payloads for the more difficult case of making a soft landing on the moon. Since the moon is without any appreciable atmosphere, deceleration must be done by retro-rockets carried in the final stage. While the moon escape velocity is small (7,800 ft./sec.) compared to earth's (36,700 ft./sec.), it is very expensive in terms of the rocket stage that must be used to accomplish a smooth landing. It has been assumed that the landing deceleration stage would have to use storable propellants because of the two and one-half days required to reach the moon, during which time cryogenic propellants might boil off. Investigations currently under way indicate that it should be practicable with very small penalties to protect cryogenic propellants for these periods so that the use of fluorine in the final stages might increase these payloads somewhat. However, for maximum reliability, particularly for the first missions, it would appear preferable to use storable propellants. The fluorine-Thor vehicle with a third landing stage would deposit 180 lbs. smoothly on the moon, of which it is estimated perhaps 40 lbs. would have to be sacrificed for the deceleration guidance system. The landing deceleration stage is complicated by the problem that the engine must be able to be throttled to from 20 to 25 per cent of full thrust. This Thor-fluorine payload is sufficient for obtaining initial data on the nature of the moon's surface. Much more complete results would be achievable with the Super-Titan, which could place 1,400 lbs. on the moon's surface. This would seem to be sufficient to permit television-carrying tankettes to crawl around on the surface and to permit sampling of the moon's surface and radio relaying of the chemical analysis results. Major data should be obtainable on the moon using these payloads, and television cameras pointed at earth could provide significant information on weather patterns. Undoubtedly many other military uses will be originated for payloads that can be placed on the moon.

I. Soft Landing on Moon and Return to Earth

Table 8 indicates payloads for a number of vehicle combinations for landing gently on the moon and subsequently taking off again and returning to earth with

Table 7. Soft Landing on Moon

	Stage I	Stage II	Stage III (Includ. Boost Guid.)	Landing Deceleration Stage	Gross Payload* Include Guidance Deceleration**
√1	THOR thrust 175,000 lb, thicker fuel tank	wt. - 15,000 lb. F_2 - NH_3 (include boost guid.)	None - Stage III would be better	650 lb. Storable propell.	180 lb.
2	ATLAS boost stage "as is"	ATLAS sustainer "as is"	wt. - 9,300 lb. storable propell. I_{sp} = 270	700 lb. Storable propell.	200 lb.
3	ATLAS boost stage "as is"	ATLAS sustainer "as is"	wt. - 9,300 lb. F_2 - NH_3	2,000 lb. Storable propell.	600 lb.
4	TITAN boost stage "as is"	TITAN Stage II "as is" inter-tank reinforce.	wt. - 15,000 lb. F_2 - NH_3	2,150 lb. Storable propell.	750 lb.
√5	TITAN airframe "as is" engine up to 400,000 lb	wt. 110,000 lb. F_2 - NH_3	wt. 20,000 lb. F_2 - NH_3	4,200 lb. Storable propell.	1,400 lb.

*See Figure 3

**CEP on moon ± 70 mi with 170 lb radio system + 50 lb P.S. in last boost stage. Deceleration guidance system - weight about 40 lb for impact at less than 20 FPS carried in final stage.

[redacted]

Table 8. Soft Landing on Moon and Return to Earth (Aerodynamic Deceleration)

	Basic Vechile	Stage I	Stage II	Stage III	Moon Landing Deceleration Stage	Escape From Moon stage	Gross Payload* Including Terminal Guidance**
1	ATLAS	Boost stage "as is"	Sustainer "as is"	wt 9,300 lb storable fuels - I_{sp} = 270	700 lb Storable propellants	200 lb Storable propellants	0 lb
2	ATLAS	Boost stage "as is"	Sustainer "as is"	wt 9,300 lb F_2 - NH_3	2,000 lb F_2 - NH_3	820 lb F_2 - NH_3	250 lb
3	TITAN	Booster "as is"	Stage II "as is" but intertank stiffened	Wt 15,000 lb F_2 - NH_3	1,200 lb Storable propellants	460 lb Storable propellants	120 lb
4	TITAN	Booster air-frame "as is" 400,000 lb engine	wt 110,000 lb F_2 - NH_3	wt 20,000 lb F_2 - NH_3	4,200 lb F_2 NH_3	1,700 lb F_2 - NH_3	520 lb
5	Large military satellite vehicle	wt 1,500,000 lb thrust 2,000,000 lb Lox-RP	wt 500,000 lb thrust 400,000 lb F_2 - NH_3	wt 150,000 lb thrust 130,000 lb F_2 - NH_3	wt 39,000 lb thrust 20,000 lb	wt 16,000 lb thrust 8,000 lb	3,000 lb

*See Figure 3

**Guidance Considerations:
1. Present radio guidance system in Stage III would give 70 mi C. E. P. on moon.
2. Moon landing guidance system - 40 lb in deceleration stage.
3. Optical homing system in moon escape stage. No weight or accuracy estimates available.
4. These results appear to be optimistic, but probably are within ± 20%.

[redacted]

aerodynamic deceleration for the earth landing. Only one case using the present boosters appears to have sufficient payload to be of interest, and this is the case using the Super-Titan which would put 520 lbs. back on the surface of the earth above and beyond the weight of the re-entry vehicle, but including the guidance necessary in the moon escape stage. The impulse required to first decelerate onto the moon and subsequently to escape from the moon adds also the the basic requirements (See Table 1). The payloads, even using the Super-Titan, are quite small. It will be noted in Table 8 that for the 520 lbs. payload, fluorine fourth and fifth stages have been assumed. If storable propellants were used for these stages to avoid the problems of cryogenics, the payload returned to earth would be reduced to 300 lbs. A payload of 300 to 500 lbs. back on the earth is believed to bs sufficient to make possible the bringing back of moon samples, lunar photographs, etc., and hence to accomplish the initial scientific explorations requiring lunar round trips. The payloads of even Super-Titan, however, appear to be too small to consider this vehicle for any manned landing on the moon whatsoever. Manned lunar exploration appears to require a considerably larger vehicle than the Super-Titan.

In the first chapter on "Air Force Space Mission Requirements" a number of missions were indicated which would require a high altitude satellite, including the possibility of retaliation. It has also been pointed out that such missions may require manned vehicles for maximum effectivity, and the payloads required are likely to mount as the capabilities become better understood and technology advances. Hence, it would appear that basic planning should include a large military satellite vehicle for high altitude satellite applications . . . one capable perhaps of putting 40,000 lbs. into a high altitude orbit and returning to earth with perhaps 12,000 lbs. of it. Such a vehicle would permit the positioning of elaborate early warning stations, retaliatory launching stations carrying five to ten retaliatory missiles or elaborate space communication centers for communications transferring navigational fixes, etc. Since such a large vehicle would probably be required for these military applications, it is of interest to see what it might be able to accomplish in the way of lunar explorations.

Case 5 of Table 8 postulates a large military satellite vehicle of 1,500,000 lbs. weight and 2,000,000 lbs. thrust in the first stage using LOX and RP. The second and third stages employ fluorine and would weigh approximately 500,000 lbs. and 150,000 lbs. respectively. With suitable moon landing and escape stages, this vehicle would be capable of bringing 3,000 lbs. payload back to earth, including the terminal guidance system, above and beyond the structure and heat sink for the re-entry body. This 3,000 lbs. is believed to be sufficient to permit two men to make the two and one-half day trip to the moon, land on the surface and examine the surface for a period, perhaps walking around on it, if the surface proved to be suitable. They could then take off from the moon and return to earth with a tangential re-entry into the atmosphere to provide an easy deceleration. A description of this large military satellite vehicle will be given separately in a later section of this development plan.

J. Interplanetary Missions

The capabilities of a number of vehicle combinations for interplanetary missions are given in Table 9. Again, the most attractive cases are checked. Most promising are the Thor with the fluorine second stage which is indicated to be capable of sending 400 lb. payload probes near Mars or Venus, or of landing 240 lbs. on Mars, or 300 lbs. on Venus, including in every case the weight of the terminal stage guidance and control systems. These payloads appear to be sufficient to permit sending probes near Mars or Venus, which would transmit back photographs and facsimiles, and could make measurements of magnetic fields, reflected illuminations, etc. Assuming light-weight guidance systems, beacons could be placed on Mars or Venus with instruments for measuring atmospheric constituents, surface characteristics and perhaps television cameras for transmitting facsimile pictures back to earth. While the Thor-fluorine combination has marginal payloads, for these more advanced interplanetary missions, Super-Titan with the ability to put 1,500 lbs. on Mars, or 1,800 lbs. on Venus, could certainly carry the necessary midcourse system, power supply, etc., to place valuable payloads on these planets. Atmospheric deceleration is postulated again, as it permits a much more favorable ratio of payload to initial entering weight than any known retro-rocket systems. The payloads for landing on Venus

Table 9. Interplanetary Missions.

Mission		Gross Payloads* (Including Terminal Guidance)				
	Stage I	THOR 175,000 lb Engine Thicker Fuel Tank	ATLAS Boost Stage	ATLAS Boost Stage	TITAN Booster "As Is"	TITAN 400,000 lb Engine
	Stage II	15,000 lb F_2 - NH_3	ATLAS Sustainer	ATLAS Sustainer	TITAN Stage II "As Is"	110,000 lb F_2 - NH_3
	Stage III	None	9300 lb Storable I_{sp} = 270	9300 lb F_2 - NH_3	15,000 lb F_2 - NH_3	20,000 lb F_2 - NH_3
1. Probe near Mars or Venus		400 lb	450 lb	1100 lb	1300 lb	2400 lb
2. Beacon on Mars atmospheric deceleration		240 lb	270 lb	660 lb	800 lb	1450 lb
3. Beacon on Venus atmospheric deceleration		300 lb	340 lb	820 lb	950 lb	1800 lb
4. Mars landing and return atmospheric deceleration						110 lb Mars take-off stage, 1800 lb storable fuels
5. Solar probe inside orbit of Mercury						1000 lb Stage IV 4,200 lb F_2 - NH_3

*See Figure 3

Guidance Considerations: For Mars and Vensus an advanced guidance system (0.2 FPS - cutoff velocity error) would give probable miss of 10,000 miles; hence, a small homing correction would be sufficient to hit.

are somewhat greater than those for Mars because of the very thin Martian atmosphere. The thin Mars atmosphere is expected to require large parachutes with attendant complications and weight. The heavier atmosphere of Venus should permit re-entry efficiencies similar to those achievements on earth, barring unusual reactions with different atmospheric components. It is assumed that atmospheric spectroscopic data obtainable either from probes or previous space missions will provide sufficient data on the atmospheric constituents of the planets to permit rational design of aerodynamic deceleration systems. Perhaps they could be called atmospherodynamic deceleration systems, since the atmospheres most probably are not similar to air. The Super-Titan, it will be noted, also would permit putting a thousand pounds inside the orbit of Mercury and, hence, would permit impacting such payloads on this planet. Since Mercury's atmosphere is negligible, landing on Mercury would require retro-rockets, as on the moon, and, with the relatively large escape velocity of 14,100 ft./sec., the payload that could be placed on the surface would be negligibly small. The possibility of a Mars landing and return is listed for Super-Titan, but the payload and return to earth, including the final guidance, would be only 110 lbs. This small value precludes providing for the elaborate system and intelligence necessary to position the rocket for the return launch, to determine the optimum launch time, and to apply the guidance and control necessary for the return to earth. The large military satellite vehicle mentioned earlier could return about 600 lbs. to earth from Mars.

Manned landing on Mars and return would appear to be considerably more difficult, requiring payloads perhaps of 10,000 lbs. With manned control of the vehicles, however, the possibility exists that one of the small moons of Mars could be used as an intermediate satellite station. If this were done, it appears barely possible that a scheme might be worked out for a manned landing on Mars and return by using two large military satellite vehicles. One would carry the "ferry boat" stage, which would be used for the round trip from the small moon to the surface of Mars and back, while the other would put a 30,000 lb. vehicle on the Martian moon for the manned return trip. The practical details involved in carrying out such a journey are very many, and the difficulties very great. Further study is needed before definite conclusions can be reached. It is perhaps sufficient for this planning report to indicate that manned trips to and from Mars may be achievable by the use of two or three of the large military satellite vehicles programmed here. Mars with its two moons, its small escape velocity of 16,700 ft./sec., and its atmosphere is especially suited to this squadron staging type of exploration. Venus, on the contrary, with its complete absence of moons and an escape velocity of 34,100 ft./sec. (nearly as large as earth's), poses a much more formidable problem. While flight to Venus and return might be feasible using a fleet of the large military satellite vehicles, it appears difficult enough, and to require a sufficient advance in space technology so as not to warrant consideration at this time.

To summarize interplanetary missions, initial explorations of Mars and Venus can be made with Thor with the fluorine second stage. Much more extensive explorations, still unmanned, can be made with Super-Titan. An unmanned landing on Mars with a return to earth might be accomplished with the large military satellite vehicle, although with all things considered, the payload is marginal. By using one of the small moons of Mars as an intermediate station, it might prove feasible using two of the large military satellite vehicles to make a manned landing on Mars with return to earth.

III. VEHICLE DEVELOPMENTS FOR USE WITH PRESENT BOOSTERS

A. Primary Combinations Recommended with Present Boosters

Table 10 shows a primary combination recommended for use with present boosters to serve as the principal steps for the space missions of the next ten years. As has been mentioned earlier, some of these may be used with additional third, fourth, or fifth stages, depending upon the application. It will be noted that two cases are recommended for each basic booster, Thor, Atlas, and Titan, respectively.

In the case of Thor, the two different cases are recommended primarily because Case 1 with the Vanguard second stage can be available over a year before the much more effective Case 2 with the fluorine second stage.

Atlas is shown programmed with the 117-L stage because it gives the earliest possible complete 117-L capability, while Atlas with the 9,300 lb. Vanguard engine stage has been indicated on the assumption that Atlas will turn out to be the most suitable vehicle for manned re-entry tests. Should the Titan program progress expeditiously, this case might be eliminated in favor of the Titan with the 15,000 lb. fluorine third stage.

Two Titan cases are included, the basic Titan with a 15,000 lb. fluorine third stage for work horse missions requiring moderate payloads, and the Super-Titan with fluorine in the second and third stages for those operations requiring very large payloads and maximum performance over-all.

While six vehicle combinations are listed, the developments required to achieve them are very much smaller than might initially appear.

Table 10. Primary Combinations Recommended with Present Boosters.

Basic Vehicle	Stage II	Stage III
1. THOR	VANGUARD Stage II	VANGUARD Stage III
2. THOR	15,000 lb. F_2 stage	---
3. ATLAS	9300 lb. VANGUARD engine stage (manned re-entry only)	---
4. ATLAS	Present 117-L stage	---
5. TITAN Stage I	TITAN Stage II	15,000 lb. F_2 stage
6. TITAN Stage I	110,000 lb. F_2	20,000 lb. F_2 stage

B. Family of Additional Stages for Present Boosters

Table 11 shows the family of additional stages needed with the present boosters to achieve the vehicle combinations outlined in Table 10. One of the stages, as will be noted, is the existing Vanguard second stage with engine and tanks complete as already developed. The second uses the Vanguard second stage engine, provides a redesigned tank system to be suitable for manned satellite launchings from Atlas. The weight of this stage without its payload is only 7,300 lbs., since it uses an existing engine and a simple pressurized system, development effort required is quite small. The third and fourth additional stages are new developments. The first one, a 15,000 lb. gross weight fluorine stage, uses a 10,000 lb. thrust pressure fed engine. This stage weight is about optimum, for application as a second stage on Thor and as a third stage on Titan. Stretching of the cylindrical portions of the tanks to give a 20,000 lb, gross weight makes the unit optimum size for the Super-Titan configuration. The low thrust of 10,000 lbs. is acceptable, since the weight saved by reducing the engine weight more than makes up for the small gravity losses for an upper stage which comes into action with near satellite velocities. Fluorine is an obvious choice for the oxidizer because of its very high specific impulse which pays tremendous dividends for space missions. At the time this study was initiated it was believed that fluorine-ammonia would be preferable, but subsequent cooling investigations by both Rocketdyne and NACA lead to the conclusion that fluorine-hydrazine will probably be the most satisfactory propellant combination. An advantage of fluorine for this application is that it can operate at very low chamber pressures without instability. Preliminary investigations indicate that a chamber pressure of about 150 psi to be optimum for a pressure fed system. The sacrifice in payload for this low chamber pressure relative to an optimum turbo-pump fed system is quite small, being perhaps 5 to 10 per cent. This loss is well worthwhile, at least

Table 11. Additional Stages for Present Boosters.

	Existing VANGUARD Stage II Engine and Tanks	VANGUARD Stage II Engine New Tanks	New Engine and Tanks	New TITAN Stage II New Engine and Tanks
Stage Weight				
Nominal - lbs	3800	9300	15,000	110,000
Alternate - lbs	-	-	20,000	150,000
Propulsion				
Engine	VANGUARD Stage II	VANGUARD Stage II	New engine	New engine
Propellant	WFNA - UDMH	WFNA - UDMH	F_2-NH_3 or $F_2-N_2H_4$	F_2-NH_3 or $F_2-N_2H_4$
Pressure fed char.				
Thrust - lb.	7700	7700	10,000	80,000
I_{sp} - sec	270	270	370	370
Chamber press - psi	200	200	120	150
Turbopump fed				
Thrust - lb.	-	-	-	135,000
I_{sp} - sec	-	-	-	370
Chamber press - psi	-	-	-	250
Expansion ratio	20:1	20:1	30:1	25:1
Stage Used with Boosters	THOR	THOR, ATLAS, TITAN	THOR, TITAN, SUPER TITAN	TITAN

in the small final stage, for the added simplicity and reliability, together with the lower development costs and time that would result with a simple pressurized system. Since the engine operates only at altitude, the low chamber pressure does not compromise the specific impulse. While it requires a somewhat larger thrust chamber, this is largely balanced off by the saving in turbo-pump weight and gas generator propellant. Fluorine engines of up to 15,000 lbs. thrust have been run by Rocketdyne. The Rocketdyne engines were not regeneratively cooled and independent cooling systems were supplied. The other new stage postulated is of 110,000 lbs. nominal gross weight with an 80,000 lb. thrust engine. In the planning of this program it has been assumed that this would also be a pressure fed engine, and that the stage would in effect be a scaled up version of the 15,000 lb. system. The 15,000 lb. stage, incidentally, requires a thrust chamber of about the size of that currently in use at a higher chamber pressure for the Atlas and Titan second stage engines. The 110,000 lb. stage requires an injector and thrust chamber about the size of that presently used for the 150,000 lb. booster engines. The latter thrust chamber would have to be extended, of course, from the present 8:1 to the 25:1 expansion ratio desired for this stage. Alternatively, the 80,000 lb. thrust engine might be achieved by converting the present 80,000 lb. LOX-RP ICBM engines to fluorine. This would introduce the turbo-pump problem with all the attendant complications of gas generator control systems, starting sequence, etc. While the existence of a LOX-RP engine of about the right size for initial tests might facilitate a turbo-pump fluorine engine development, it has been assumed for the purposes of this development plan that the pressure fed engine would be preferred. It is proposed to develop the engine to operate pressurized at the 80,000 lb. thrust level initially. Later it would be converted to a turbo-pump fed engine of 135,000 lb. thrust to work with an extended second stage for Titan and to serve as a basic propulsion unit for the large military satellite vehicles mentioned earlier. The potential characteristics of the turbo-pump fed engines are also shown in Table 2 for the alternate stage weight of 150,000 lbs.

C. Program for Development of Additional Stages

Table 12 details the programs for the development of the four additional stages required for addition to the present boosters to achieve the space capability outlined previously.

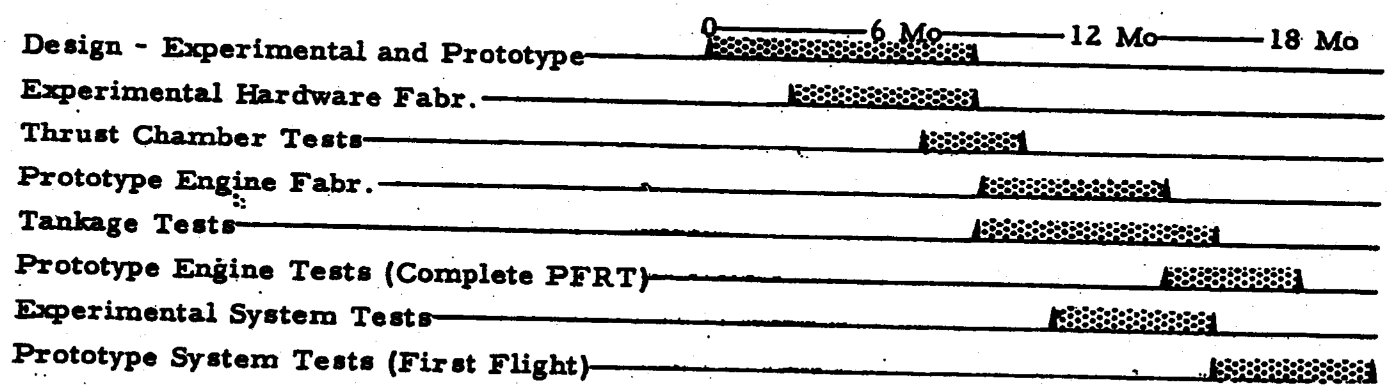
First, the Vanguard stage II takes no development, and is available upon reorder on 60-days' notice. The second stage shown in Table 12, using the Vanguard engine would require new tankage, and could be developed in approximately one year at a development cost estimated at about $3 million. These two items relate primarily to the interim program using earliest available hardware on Thor and Atlas, and need no further discussion.

The development program for the 15,000 lb. fluorine stage has been outlined in considerable detail. The first eight months are used primarily to design experimental hardware and conduct tests using existing hardware adapted for the application. In particular here, the present sustainer thrust chambers of the ICBM are of about the right size for consideration for this application and, while probably unduly heavy, could provide battleship hardware for experimental work. Tests of these units could be followed with experimental hardware more directly related to the prototype and subsequently with prototype engine tests.

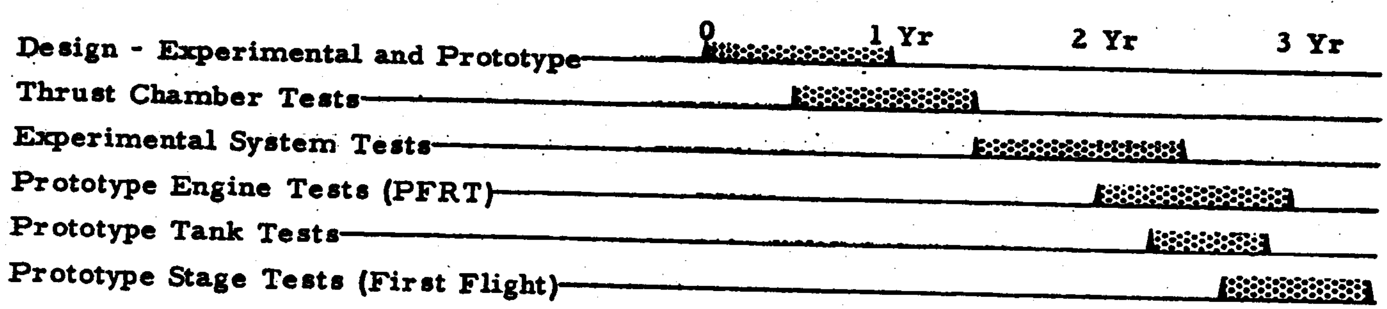
Table 12, which has been laid out on the basis of the extensive experience from the ICBM development, indicates that the 15,000 lb. fluorine stage could be available for first flight 20 months after initial go-ahead. The fluorine 110,000 lb. stage, takes three and one-third years for development to first flight. This greater time is primarily the result of the requirement for new facilities in a remote location to permit extended test durations with fluorine.

Table 12. Development of Additional Stages for Present Boosters.

1. Vanguard Stage II - Available on 60 Day Notice - No Development Required
2. Storable 9300 Lb Stage Vanguard Eng. Available - Stage Development Cost ~ $3 Million
   Stage Development Time ~ 1 Year
3. Fluorine 15,000 Lb Stage - Estimated Cost $8.0 Million
   6.0 Million Engine
   2.0 Million Tankage and System Integration



4. Fluorine 110,000 Lb Stage - Estimated Cost $50 Million
   40 Million for Engine
   10 Million for Tank and Integration



The three years indicated from work initiation to PFRT is identical to the time required for development and PFRT of the second stage engines for the ICBM's, including the construction of facilities, development of the turbo-pump system and the use of LOX-RP propellants in place of the lower energy combinations previously used by both engine manufacturers concerned. With large-scale hardware available which would permit initial testing, the background in large engine development accumulated during the last three years, and the absence of the turbo-pump, it is believed the time schedule laid out for this stage can be readily met.

D. Manned Satellite and Re-entry Test Vehicle Development Program

Earlier in this study the capabilities of the ballistic missile boosters for launching of manned re-entry vehicles were outlined. Figure 6, showing a preliminary design for a manned test vehicle for exploring satellite environments and re-entry conditions, was discussed. Figure 7 shows a development schedule for such a vehicle to include 10 flights with apes and 12 flights of gradually increasing duration with men. The estimated cost for the development of these vehicles and their fabrication and tests, exclusive of the costs of the launching system is $45 million. It is expected that the ICBM program will provide the heat shield material data, that the recovery system can be an adaptation of the Cook system developed for nose cone recovery, and that the contractor would be one involved in the present nose cone programs to use acquired background to the maximum practical extent. The program outlined here should be preceded by a program using Thor-Vanguard and Thor-fluorine stage systems to run zero gravity tests on small animals. The three years allotted for development prior to first flight compares with about two years and eight months for the present nose cone programs, and about a year and a half for improved nose cones presently under development for the IRBM. Hence, the three-year period is believed to be conservative and to adequately account

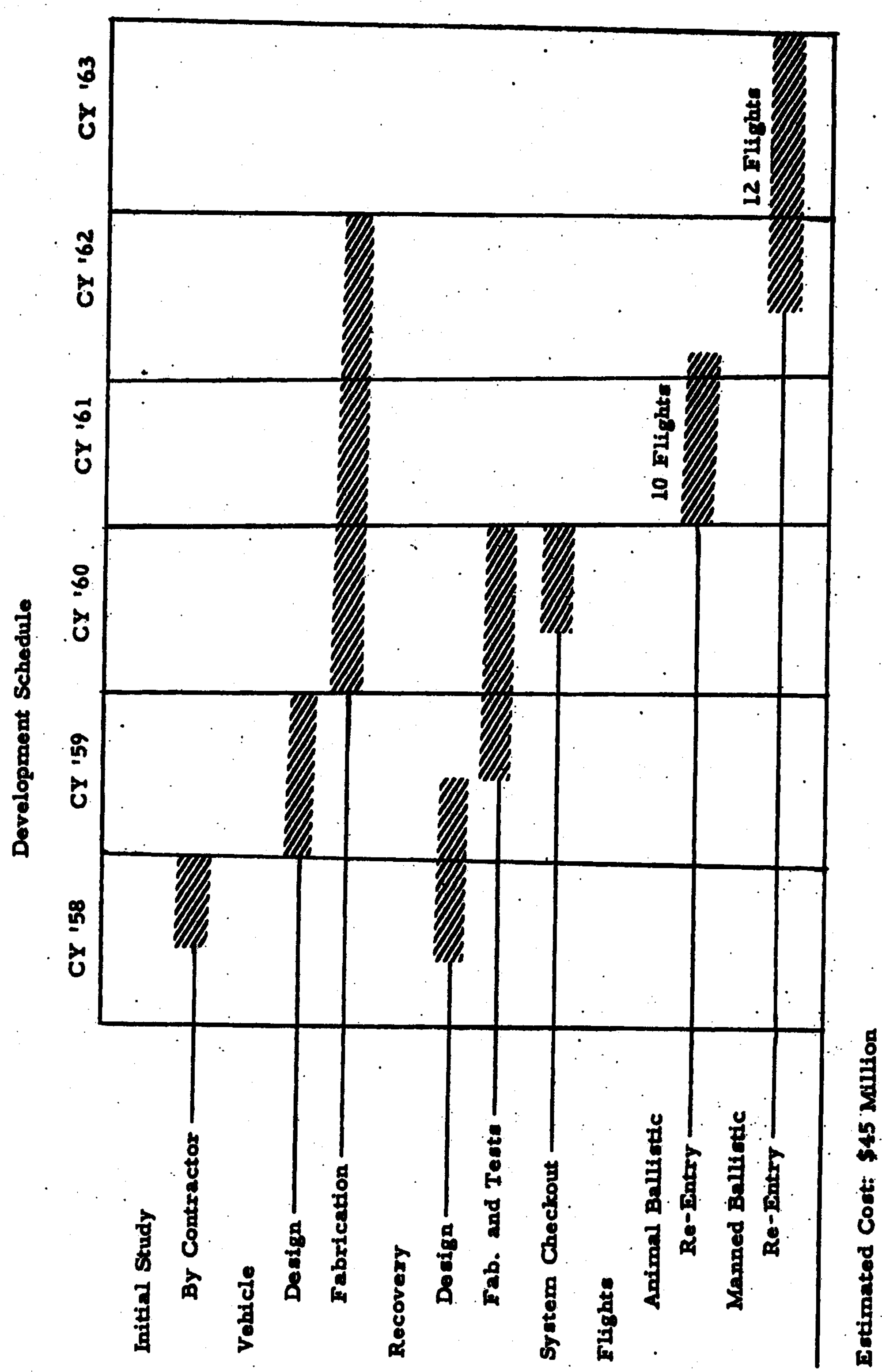


Figure 7. Manned Ballistic Re-entry Development Schedule.

for the complications introduced by the manned requirement. The development schedule indicates ten flights in approximately a 14-month period for animal re-entry prior to manned re-entry. Obviously the decision to attempt or not to attempt manned re-entry will depend upon the evaluation of many factors at the time. In particular, the reliability of the boosters, results of tests of the escape measures, and the extent of the success of the animal flights will all have to be considered in determining whether or not to attempt manned re-entry. It appears that if the program is carried out properly, the initial risk can be much less than that accepted by the pilot in the first supersonic flight in the X-1 airplane. At least the vehicles will have been checked for a number of times for the exact sequence and operation it will have to perform with a man on board. Once manned flight is successfully achieved, it will of course be desirable to make tests with more than one subject, and to gradually increase the duration in the zero gravity field until the human factors information on operations in this environment has been acquired. The 12 flights scheduled here are certainly a minimum to obtain this data. However, it is believed that it will be sufficient to provide the information required for the design of a militarily useful manned satellite, if requirements at that time dictate such a development.

E. Assignment of Vehicle Combinations to Missions

A proposed initial assignment of vehicle combinations for the principal and immediate military missions is shown in Table 13. In addition to the cases listed, the Thor with the 15,000 lb. fluorine stage might be suitable for the 117-L mission, and the Super-Titan will very probably perform an effective early warning mission. With respect to Item 4, "Technical Development and Experimental Support,", the Thor-Vanguard combination is recommended here for use as an ICBM re-entry test vehicle to explore new high-performance nose cone designs, and for exploring satellite re-entry and recovery problems. In addition, as mentioned earlier, when the Vanguard third stage is added to it

[redacted]

Table 13. Applicable Vehicle Combinations.

Mission	Base Vehicle	Additional Stages	Remarks
1. Reconnaissance - (117L)	THOR	VANGUARD	Early experimental flights
	ATLAS	YLR-81	
2. Communications	THOR	15,000 lb. F_2	
	TITAN Stage I	TITAN Stage II 15,000 lb. F_2	
	TITAN Stage I	110,00 lb. F_2 20,000 lb. F_2	
3. Manned space flight	THOR	VANGUARD	Animal experiments; data gathering
	TITAN Stage I	TITAN Stage II 15,000 lb. F_2	
	TITAN Stage I	110,000 lb. F_2 20,000 lb. F_2	
4. Technical development and experimental support			All vehicles applicable as dictated by payload requirements; e.g., ATLAS for initial manned re-entry tests

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with a suitable guidance system, it can perform an initial impact on the moon. The Thor plus the 15,000 lb. fluorine stage can serve for considerable initial lunar and planetary explorations, although the Super-Titan is the vehicle primarily indicated for these missions.

A summary sheet and the payloads for various space missions for the recommended Thor-Titan combinations is given in Figure 22.

IV. LARGE MILITARY SATELLITE VEHICLE

Mention has been made in a number of places earlier in this development plan of a possible large military satellite vehicle for high altitude satellite missions requiring large payloads. These missions might include early warning, anti-ICBM control, communications, navigation, and possibly even the provision for launching retaliatory missiles. The retaliatory missiles would be an extension in concept of current air-to-ground types, being, in effect, Satellite-to-Surface ballistic missiles. The vehicle postulated here (see Figure 8) which would have a gross weight of approximately 15,000,000 lbs. and would be capable of putting 20,000 lbs. into a high altitude orbit, of which 12,000 lbs. could be returned to earth. The first stage is conceived as having a 2,000,000 lb. boost system using LOX-RP, probably composed of two 1,000,000 lb. thrust chamber units. An alternate possibility would be four half-million lb. systems obtained by the use of large thrust chambers now under development by WADC, together with large turbo-pumps under development by AFBMD for use on the Atlas. Based on results of a previous Rocketdyne large engine study made for WADC, and general considerations of reliability, it would appear that the two 1,000,000 lb. thrust units would be preferable technically, while the four one-half million lb. systems would probably be somewhat cheaper to develop.

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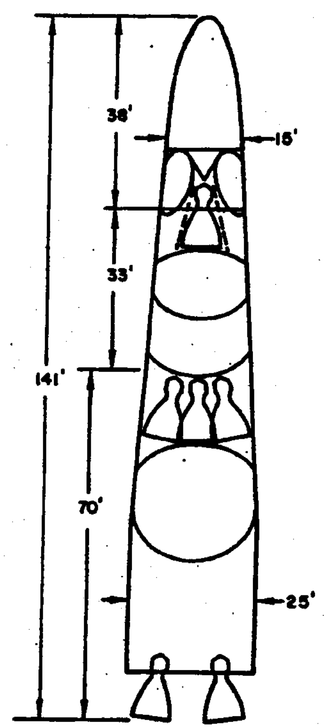

SATELLITE STAGE (Manned)

Payload in 20,000 mi. orbit	= 20,000 lbs.
Recoverable Payload	= 12,000 lbs.
Gross Weight	= 40,000 lbs.

3RD STAGE

Weight (Including Later Stage)	= 150,000 lbs.
Thrust -	= 135,000 lbs.
Engine-F_2-NH_3 or F_2-N_2H_4 with Turbopump (Pressurized 80,000 lb. Engine Plus Turbopump)	
Thrust Chamber Available from Pressurized Engine	
Estimated Engine Development Time	- 4 Years
Estimated Engine Development Cost	- $50 Million

2ND STAGE

Weight (Including Later Stages)	= 500,000 lbs.
Thrust -	= 405,000 lbs.
Prop. System - Three 135,000 lb. 3rd Stage Engines (F_2-NH_3 or F_2-N_2H_4)	
Estimated System Development Time 5 Years Total	- 1 year after 3rd Stage Engine Available
Estimated System Development Cost	$20 Million (3rd Stage)

1ST STAGE

Weight (Including Later Stages)	=1,500,000 lbs.
Thrust -	=2,000,000 lbs.
Engine - Two 1,000,000 lb. Lox-RP (Alternate-four 500,000 lb. Lox-RP)	
Estimated Engine Development Time	- 4 Years
Estimated Engine Development Cost	-$135 Million

Figure 8. Large Military Satellite Vehicle.

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Before discussing the second stage of this vehicle it would be desirable to examine the third stage. The engine for this section is conceived to be a 135,000 lb. thrust unit, obtained by adding a turbo-pump to the 80,000 lb. fluorine engine used on Stage 2 of the Super-Titan. The increase in chamber pressure resulting from a turbo-pump would raise the thrust to 135,000 lb., while the lower pressure required in the tanks should achieve a significant weight saving. The third stage is conceived as having a basic weight of 110,000 lb., which would be topped by a 40,000 lb. payload, to make a total weight of 150,000 lb.

The second stage has a net weight of 350,000 lb., leading to a gross weight, when topped by the third stage, of 500,000 lb. The propulsion system for this stage is conceived as a cluster of three of the 135,000 lb. fluorine engines mentioned earlier for use on the third stage. The successful development of three engine clusters for the Atlas indicates that a three-chamber system would be feasible, and the high cost of developing large fluorine engines and the difficulties involved in their testing make it highly advantageous to use this approach.

As indicated in Figure 8, it is estimated that the development of the first stage engine would take four years, and cost about $135 million. The third stage engine development has likewise been estimated to take four years, and to cost $50 million above the cost for the development of the pressurized 80,000 lb. engine from which it would be derived. Assembly of three of these three-stage engines into a system for the second stage is estimated to take an additional year and another $20 million for system integration. The total propulsion development costs, accordingly, for this vehicle are estimated at $205 million, and the propulsion development time approximately five years. The cost for design and development of the tankage systems, together with the propulsion systems, and for the initial prove-in flight tests could be expected to be an additional 200 to 300 million dollars, bringing the total cost for this development through the initial prove-in flight tests to approximately 500 million dollars. Development in an orderly fashion would result in a vehicle being available for initial use about 1965. Particularly urgent at the present time is the initiation of the development of thrust chambers and turbo-pumps in the million pound thrust capacity to provide the propulsion background and components for launching this, or equivalent large future military vehicles. Rocket engines have very long lead times and, if we are to surpass other nations in propulsion developments, it is essential that active hardware programs for high-thrust engines be initiated immediately. The exact definition of a future large military satellite vehicle can be an item of study for the next two years, by which time the payload requirements and other desired characteristics should be much better defined. Vehicle development should be initiated about 1960, so that the vehicle would be available for both military and scientific missions during the 1965 to 1970 period.

V. MISCELLANEOUS VEHICLE REQUIREMENTS

In addition to the basic vehicles outlined up to this point, which serve primarily to launch bodies into space, it is obvious from the previous discussion that developments will have to be planned for propulsion systems to stabilize and position satellites for vehicles to land on the moon and planets, and for launching vehicles from these locations. Further needs include development of satellite recovery vehicles, of instrumentation for recovery vehicles and for use in outer space, and of atmospheric decelerating systems suitable for use in other planetary atmospheres. Many special requirements arise for these cases, such as long duration power plants for auxiliary power, heating and cooling systems, and of the auxiliary systems necessary to make manned flight of long duration feasible. Planning for these developments is not detailed here, not because this is not an area of major importance, but because it is believed initial emphasis should be given to getting launching vehicles developed within the required time once the basic launching vehicle program is firmly established.

VI. GUIDANCE SYSTEMS

Reviews of the guidance requirements for space missions have shown the need for three general classes of guidance: launch and injection into orbit, midcourse, and terminal. Analyses of the accuracy requirements of representative space missions show that the guidance systems developed as a result of the ballistic missile program are adequate for the launch phase of space missions and with slight modifications meet the injection guidance requirements. Advanced developments in radio guidance systems being carried out by STL for AFBMD are presently being used for midcourse guidance for space missions. Terminal guidance systems are under development by Lockheed Aircraft Company for AFBMD.

A brief discussion of the characteristics of guidance systems under development for AFBMD and of the application of these guidance systems to the Air Force space missions and to other space missions follows.

The Air Force has two radio guidance systems and two inertial guidance systems in production. Two additional inertial guidance systems, improved in both accuracy and weight, are under development. Finally, a study is continuing of a wide baseline radio guidance system which can be used both in weapon and space missions. Several attitude control systems are being developed by Air Force contractors for space vehicles.

The present Thor missile utilizes the first inertial guidance system (Figure 9) used in an operational IRBM. It is being built by AC Spark Plug. Gyro drifts and accelerometer errors couple with computer errors to give

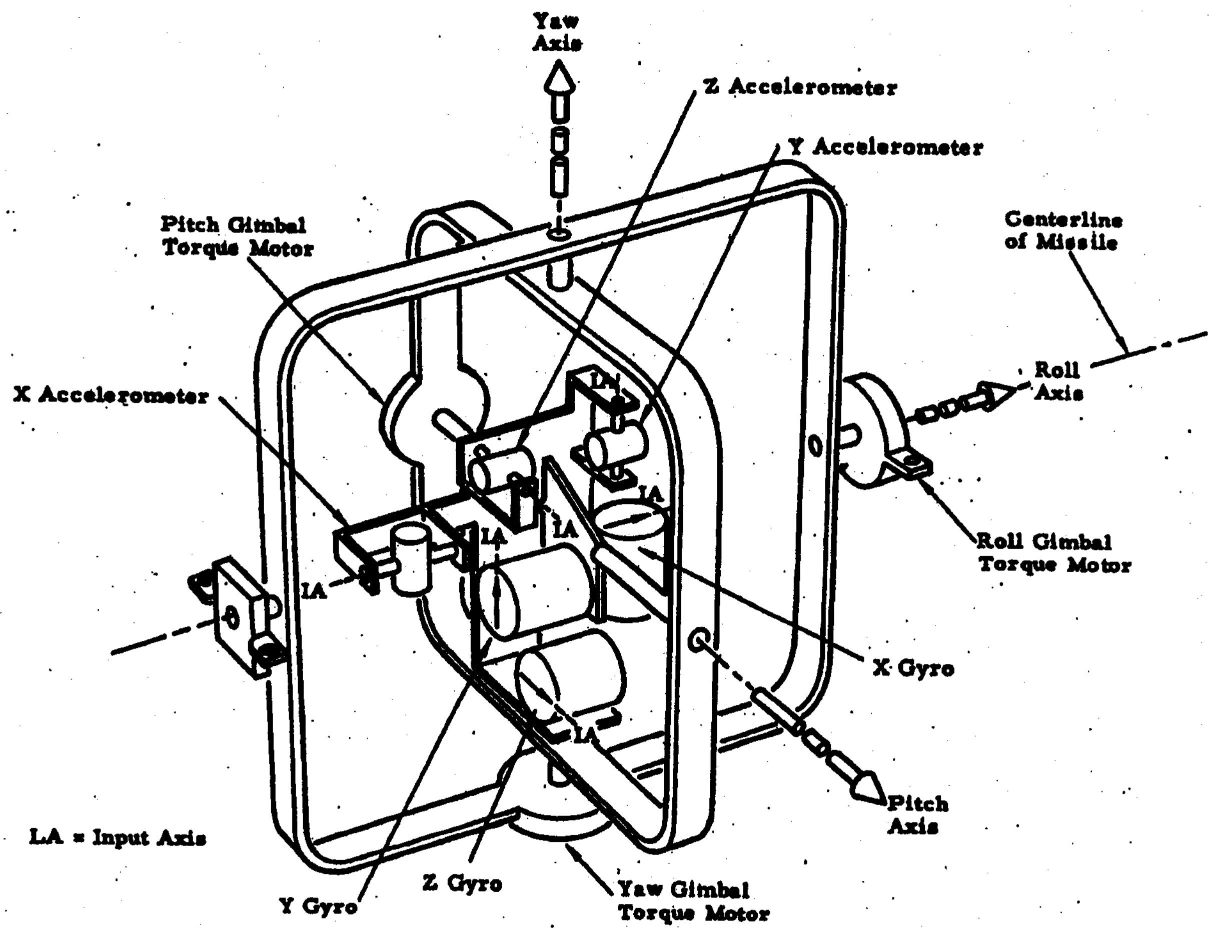


Figure 9. Thor Inertial Guidance Platform,

errors at burnout of approximately eight feet per second in velocity, of 0.02 degrees in flight path angle, and of 0.02 degrees in azimuthal angle*. At 1500 nautical miles the CEP of this system is 1.8 nautical miles. The system weighs about 807 lb. and occupies about 39 cubic feet.

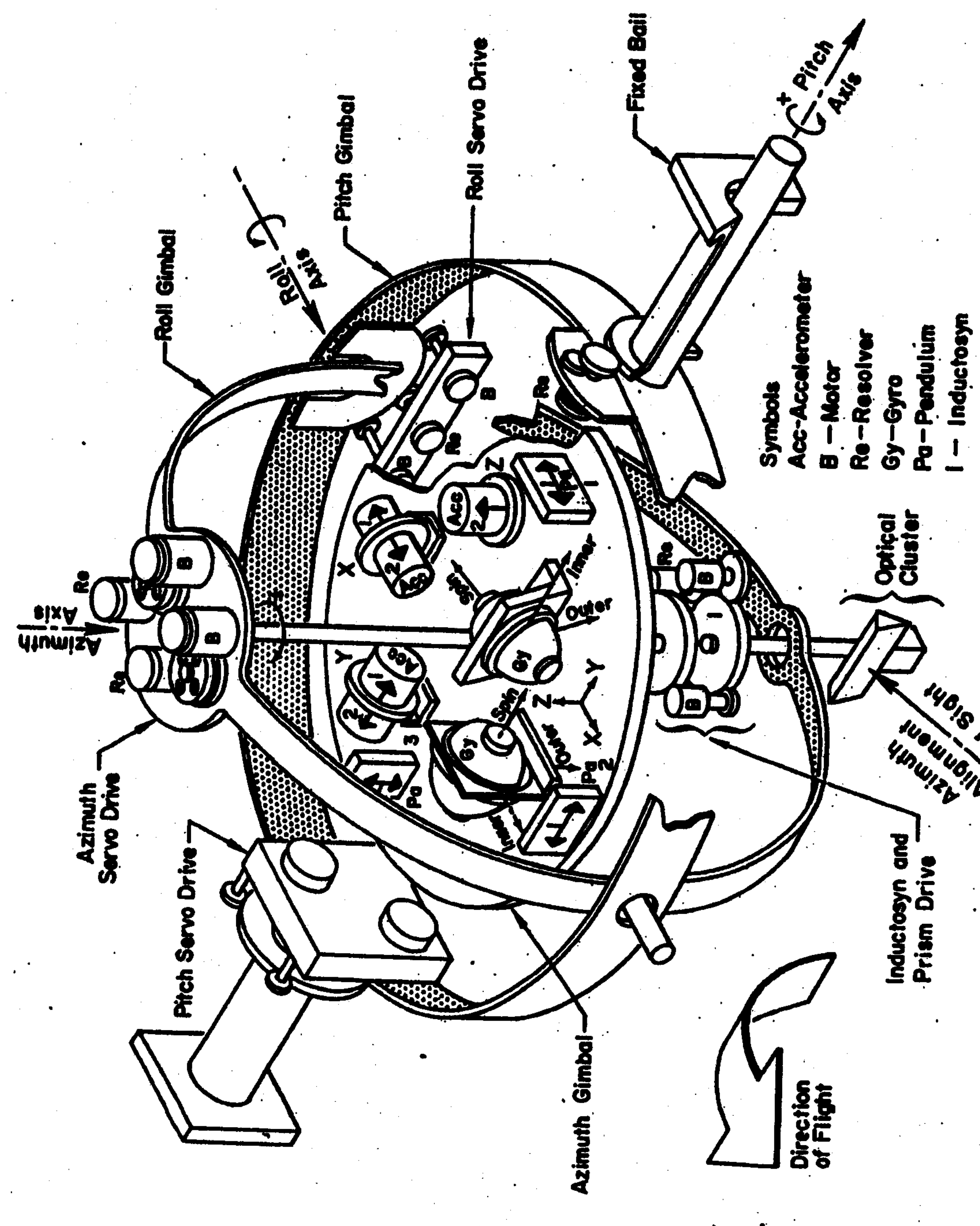
In one configuration, the Atlas missile will utilize an inertial guidance system (Figure 10) produced by The Arma-Bosch Corporation. Burnout accuracies in velocity of approximately 2 feet per second, in flight path angle of 0.005 degree and azimuthal angle of 0.005 degree are provided by this system. At 5,500 nautical miles, the CEP of this system is 1.8 nautical miles. This system weighs about 480 lbs. and occupies about 16 cubic feet.

AC Spark Plug is designing and building a second inertial guidance system to be used in one configuration of the Titan. Predicted errors at burnout with this system are: in velocity, 1 foot per second, in flight path angle 0.0025 degree and azimuthal angle, 0.0025 degree. This system will weigh about 165 lbs.

For the Minuteman, Autonetics Corporation is developing an inertial guidance system (Figure 11) with a specified CEP at 5,500 nautical miles of 1.1 nautical mile. In this system, at burnout the expected velocity accuracy is 1.5 feet per second, flight path angle is 0.004 degree and azimuthal angle is 0.004 degree. Weight is 232 lbs., volume 25 cubic feet.

The Air Force has three radio guidance systems presently under development. It should be noted that because radio guidance systems can use the same equipment for powered flight guidance, for free flight tracking and for issuing commands to the payload for midcourse or terminal guidance or for other purposes, they have a general advantage over inertial guidance systems.

* It must be noted, of course, that the errors at burnout given for each of these systems are nominal values and do not represent what would occur on an actual flight where the specific trajectory would have a marked effect on these values.



* Beginning with Lot B, the No. 3 pendulum motion will be in the X-Z plane.

Figure 10. Atlas Inertial Guidance Platform.

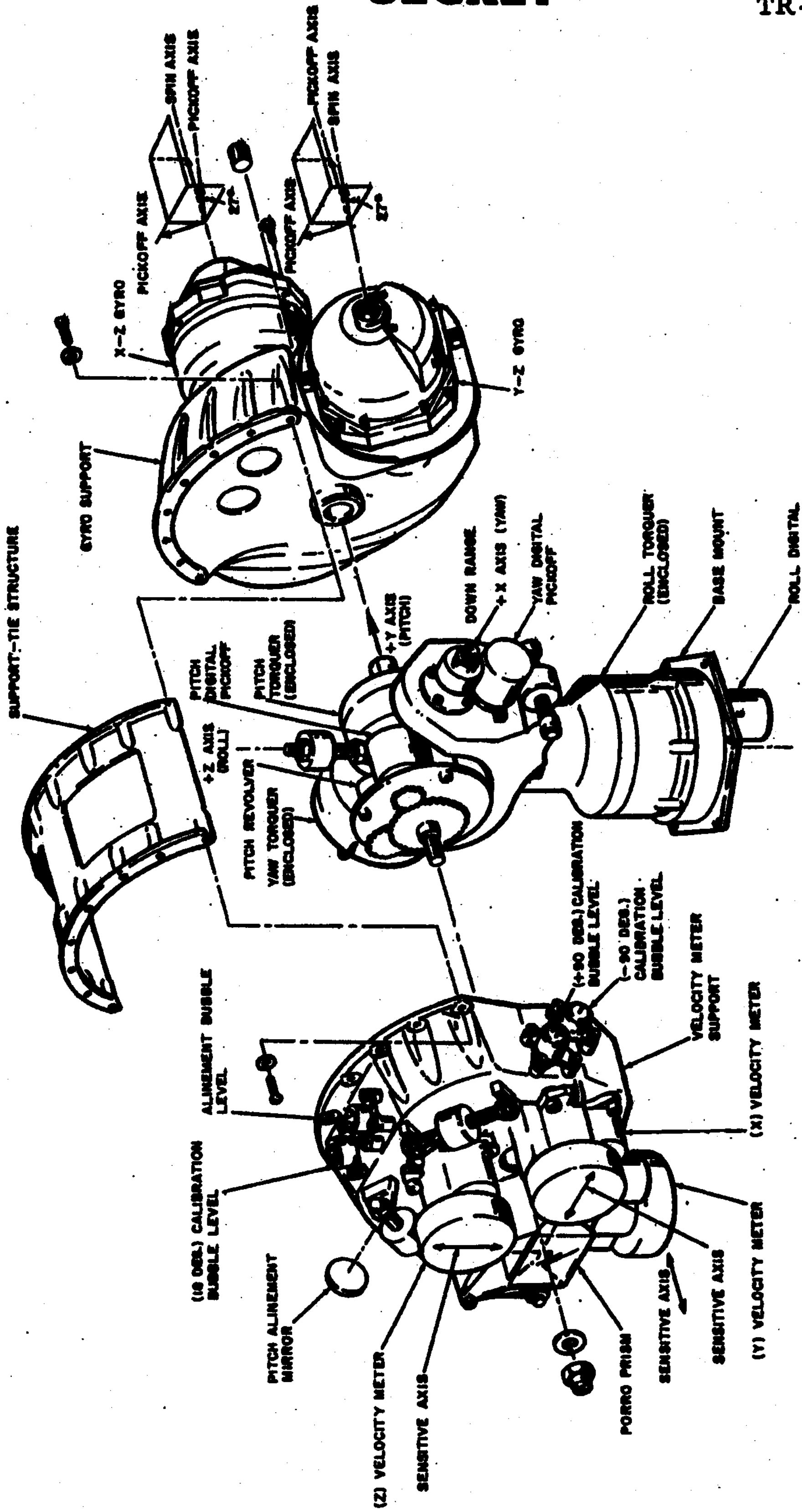


Figure 11. Minuteman Inertial Guidance Platform.

The BTL system (Figure 12), to be used with the other configuration of the Titan missile, is a pulse radar guidance system: the airborne unit weighs 150 lbs., and occupies a little more than four cubic feet. The system provides burnout accuracies of 0.5 feet per second in velocity, 0.005 degree in flight path angle, and 0.005 degree in azimuth. The guidance contribution to the CEP on the Titan at 5,500 nautical miles is 1.4 nautical miles. The GE radio guidance system designed for the Atlas is an interferometric system obtaining range and position measurements by means of a pulse tracking radar and velocity and angular rate measurements by means of a doppler radar. This system weighs 152 lbs. and occupies 4.8 cubic feet. It achieves a 0.5 foot per second error in velocity, 0.005 degree in both flight path angle and azimuth. CEP for the Atlas at 5,500 nautical miles range with this guidance system is 0.8 nautical miles.

In addition to the foregoing systems, a wide baseline radio guidance system is presently under study by STL and is being used on the Able space missions. This system utilizes three doppler radars with microwave links between the outlying stations and central station. The airborne unit weighs 8.5 pounds and occupies 0.2 cubic foot. This system, with a baseline of 15,000 feet will provide burnout accuracies of approximately 0.1 foot per second in velocity (exclusive of ionospheric effects) and 0.003 degree in flight path angle and azimuth. This system is adapted to extremely long-range guidance having apresent range in excess of 50,000,000 miles, and is thus suitable for midcourse guidance on lunar and interplanetary probes.

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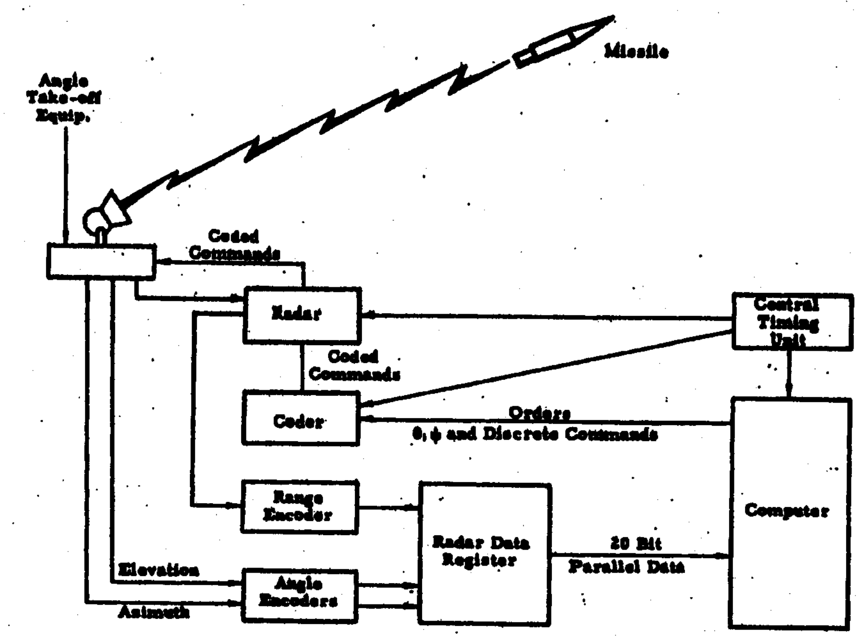

Figure 12. Simplified Block Diagram of the BTL Radio Guidance System.

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A terminal guidance system is being developed by the Lockheed Aircraft Company for the Discoverer program. Its accuracy from launch to terminal burnout is about 5 ft/sec in velocity and approximately 0.1 degree in angle. It weighs about 122 lbs.

A more accurate final guidance system is being developed by the Communication Satellite Program to meet the station keeping requirements of these programs.

In Table 14 the burnout accuracies possible with these nine guidance systems are listed; and Table 15 gives the approximate guidance accuracy requirement for several types of space missions.* These guidance systems are being applied to four basic missions: reconnaissance and early warning, communications, navigation and satellite rendezvous. The reconnaissance and early warning satellite system being developed by AFBMD in the Discoverer, Sentry, and Midas programs makes use of the Lockheed optical inertial guidance and attitude control system for the injection phase of the launch maneuver. The Thor autopilot is used for first stage guidance in the Discoverer program, and the GE radio system is used for guidance of the Atlas first stage in the Sentry/Midas programs. The recoverable data gathering package of the Discoverer program is guided by a combination of radio tracking and command and the optical inertial attitude control system.

AFBMD's communication satellite program is divided into five parts. Of these the most difficult guidance problem is encountered in the 24-hour communication satellite program. This requires development of a guidance and control system capable of placing the payload in a circular equatorial orbit at an altitude of 19,650 nautical miles. The guidance system proposed here is a combination of the Sentry/Midas tracking and command system, an optical

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* Compare Appendixes

Table 14. Guidance System Accuracy.

Guidance System	Type	Approximate Burnout Accuracy		
		Velocity (ft/sec)	Azimuth (deg)	Flight Path Angle (deg)
ACSP (THOR)	Inertial	8	0.02	0.02
Arma (ATLAS)	Inertial	2	0.005	0.005
Autonetics (MM)	Inertial	1.5	0.0025	0.0025
ACSP (TITAN)	Inertial	1	0.004	0.004
BTL (TITAN)	Radio	0.5	0,005	0.005
GE (ATLAS) 2,000 ft base line X-Band	Radio	0.5	0.005	0.005
AGS 15,000 ft base line UHF	Radio	0.1*	0.003*	0.003*

* Exclusive of ionospheric effects.

Table 15. Approximate Accuracy Requirements at Burnout.*

Mission	Velocity (ft/sec)	Azimuth (deg)	Flight Path Angle (deg)
Earth satellite in approximate circular orbit at 1000 nautical miles altitude ($\pm$100 nautical miles). Launch guidance.	250	1	0.6
Recoverable earth satellite (accuracies at reinjection)	30	10	10
Lunar impact (or satellite)	10	0.4	0.2
Interplanetary probe	10	0.2	0.2
Planetary impact (or satellite) with midcourse guidance	10	0.2	0.2
Without midcourse guidance	1	0.02	0.02
Lunar soft landing with midcourse and terminal guidance. Launch guidance.	1	0.1	0.06
Terminal guidance	10	0.5	0.5

* For some missions these requirements have been chosen to represent selected days for permissible (range safe) azimuths. There are, of course, many alternative days but the requirements are close to the optimum, i.e., the best possible days for launch.

attitude control similar to that used in the Sentry vehicle, and an inertial system based on either the Autonetics or AC Spark Plug inertial platform. The polar communication satellite program requires the establishment of a circular orbit at an altitude of 5,000 nautical miles. It will use the General Electric guidance system for first stage control and a simple inertial guidance system for the injection maneuver.

The navigation satellite program is a joint venture with the payload supplied by the Applied Physics Laboratory of Johns Hopkins University and the vehicle supplied by AFBMD/STL. The vehicle is a Thor/Able Star configuration using the AGS system for launch guidance and a simple inertial guidance system for injection.

A program for the development of a satellite interception system is being prepared. Either radio or inertial guidance using the present systems appear to provide adequate accuracy for successful completion of the satellite rendezvous maneuver. Terminal guidance is accomplished by existing homing radar and infared systems.

In addition to the DOD space flight missions, STL is studying other space flight missions ranging from earth satellites with scientific payloads to a lunar soft landing. For each of these missions certain common guidance requirements can be distinguished, while in some cases special guidance requirements which require special implementation can be identified. The earth satellite studies include the highly elliptical earth satellite orbits where the guidance need be only a simple autopilot on both the first and second stages or an earth satellite with a circular orbit which could make use of the BTL guidance system to meet the increased accuracy requirement.

For lunar impact or a lunar satellite, first stage guidance may be again a simple autopilot but for second stage guidance either the BTL, GE, or AGS system is required. In addition, midcourse correction of the velocity vector is necessary.

Interplanatary probes, if aimed at a particular point, require sophisticated guidance, while a so called "solar probe" can be adequately guided by a simple autopilot.

A recoverable earth satellite as exemplified in the Discoverer program, requires accurate tracking in orbit and a command and terminal guidance system such as is used in the Lockheed recovery vehicle.

Interplanetary flights which either impact or orbit about neighboring planets require a higher order of accuracy in guidance both during the launch and mid-course phase. This can be achieved with existing radio guidance systems. An example is the proposed Venus probe using the Atlas Able-4 vehicle configuration with a combination of the GE guidance of the first stage and AGS guidance of the second stage and for the fourth stage midcourse guidance.

Finally the most difficult task of all is encountered in the lunar soft landing mission. The accuracies required are such that only a wide-base radio guidance system will provide adequate launch and midcourse guidance. A study has shown that the AGS system, with suitable modifications, will serve the functions. However, the study also shows that a new terminal guidance system is required.

VII. OTHER ELECTRONIC REQUIREMENTS

Before proceeding to the mission sections it might be well to review a few considerations involved in other electronics required for astronautics.

Figure 13 indicates schematically some possibilities for earth to Mars navigation with midcourse guidance being supplied by signals from four earth satellites or, alternatively, from celestial navigation, while solar energy might power the electronic systems. Infrared, optical, or radar systems would be used for homing, or other guidance at the Mars terminus.

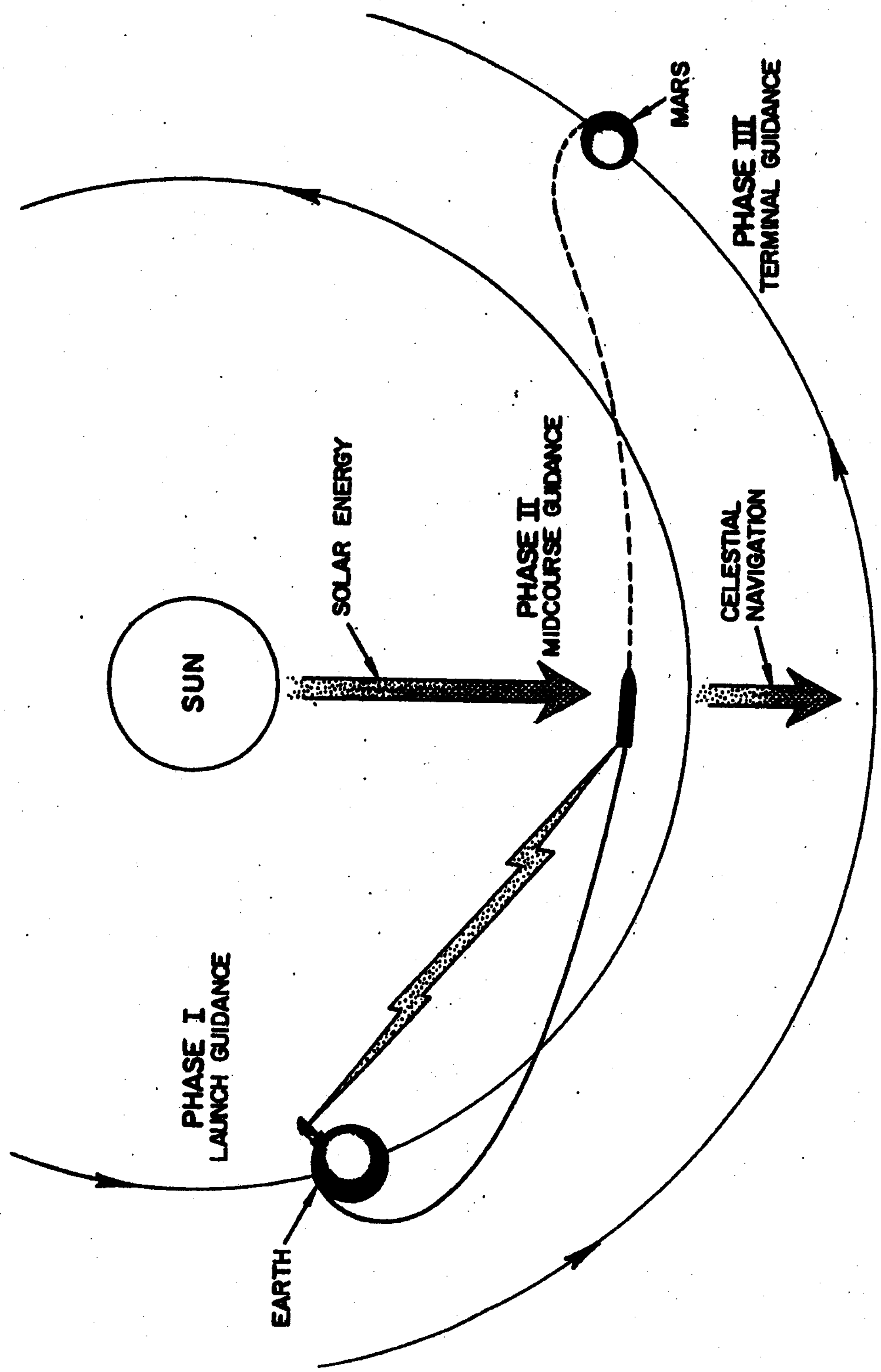

Figure 13. Earth to Mars Navigation.

Figure 14 indicates possibilities for moon exploration by television. Preliminary studies indicate that as little as one watt radiated power, if suitably employed, would be sufficient to transmit television pictures from the moon to earth. Hence, very light payloads could suffice to bring pictures of moon details back to earth.

Figure 15 indicates possibilities with respect to Mars exploration where facsimile pictures could be transmitted to earth from a television camera, using as little as 10 watts of transmitter power.

Problem areas anticipated in interplanetary exploration are listed in table 16.

VIII. PLANS FOR VEHICLE USE

A. Introduction

The preceding sections have outlined the principal Air Force space missions, together with plans for the development of vehicles and electronic systems for carrying out these missions. This section will present an over-all schedule for the various developments, a calendar for the accomplishment of the major missions, and a suggested allocation of vehicles for astronautics through 1965.

B. Over-all Development Schedule

Figure 16 shows an over-all schedule for the development and application of the vehicle combinations incorporated in this development plan. The chart will be of use principally in indicating the successive phasing of the increasing space capability, beginning in 1958 with the Vanguard second stage on Thor, and extending to the availability in 1965 of the large military satellite. Achievement of the later missions on or about the dates indicated would be feasible only if an active program such as outlined here is initiated immediately and carried on vigorously over the next ten years. While such planning must be considered to be extremely tentative, it is believed desirable to lay out such a calendar from a number of standpoints. In particular, it makes it possible to show the very great potential of the Air Force ballistic missiles for military space vehicles and for

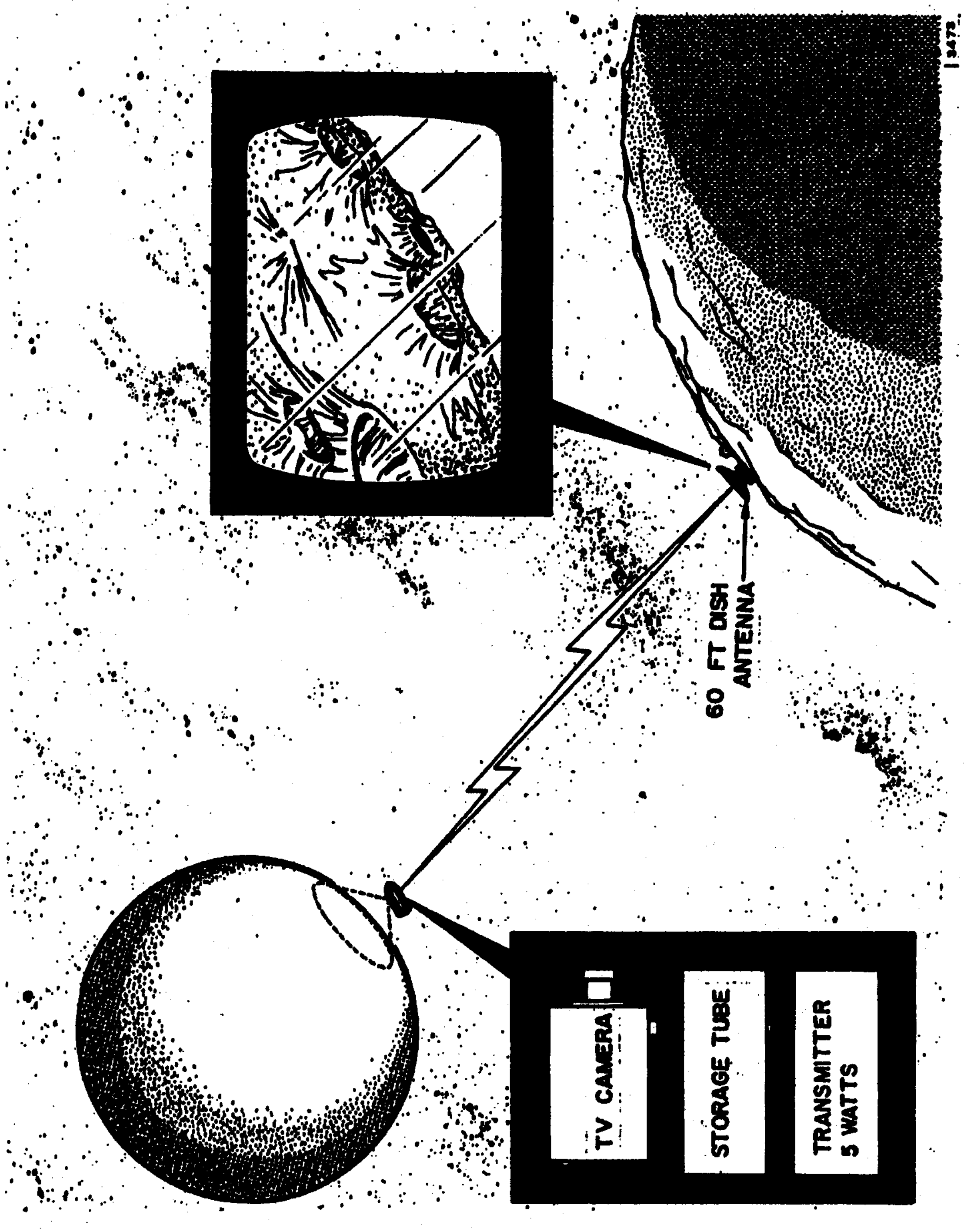

Figure 14. Televized Moon Exploration.

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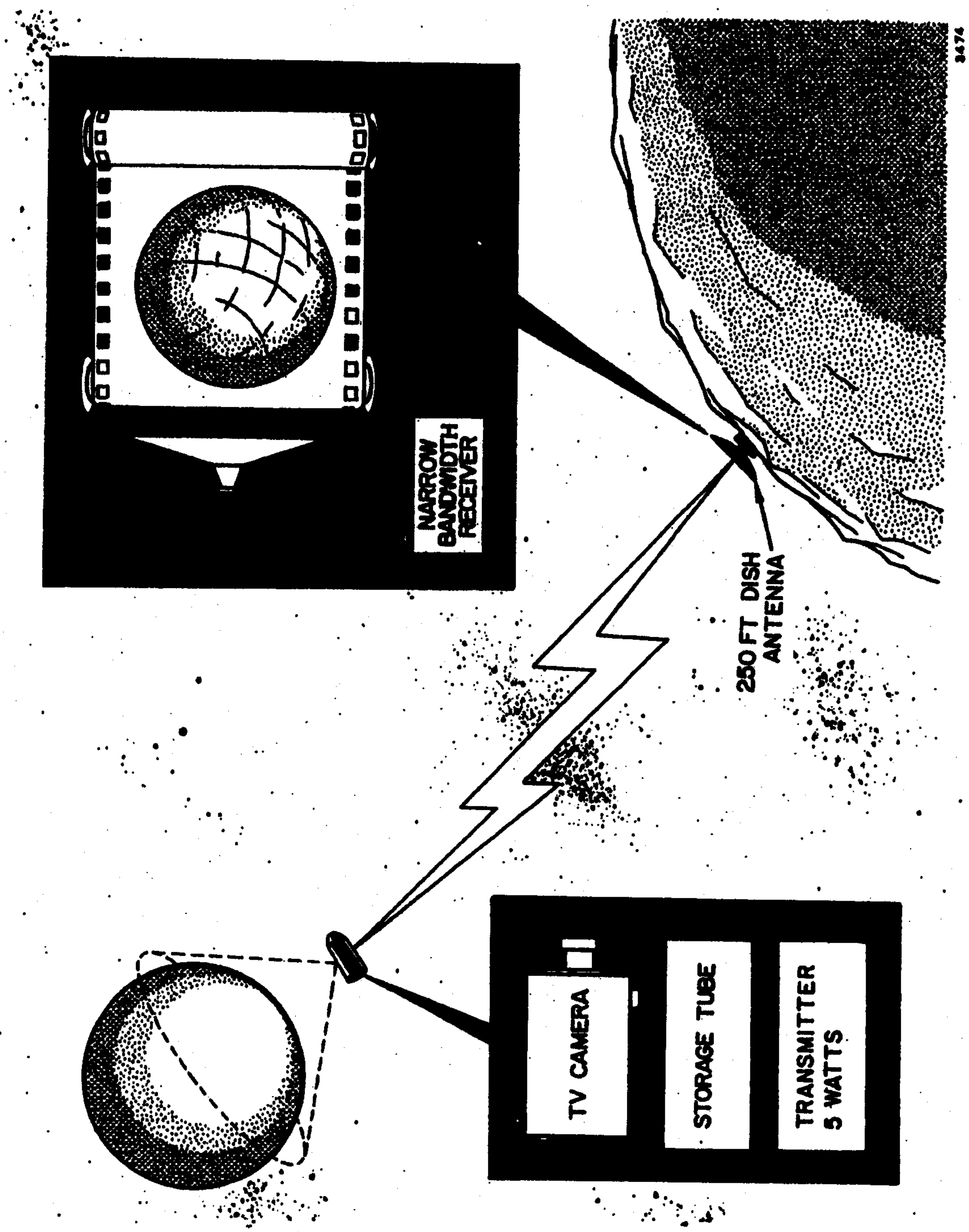

Figure 15. Facsimile Mars Exploration.

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Table 16. Problem Areas.

1. Astrophysical and Planetary Constants
   - A. Astronomical Unit
   - B. Interplanetary Dust and Meteorite Density
   - C. Planetary Atmospheres
2. Comprehensive Study of Trajectories and Miss Coefficients
3. Soft Landing Procedures and Corresponding Guidance Requirements
4. Light Weight, Low-Power Computers
5. Propulsive Energy Requirements for Mid-Course or Terminal Guidance
6. Orbital Stability of Transponder Satellites Used in Long-Base Guidance Systems
7. Millimeter Wave Components for Long Distance Transmission
8. High Efficiency Electrical Energy Sources
9. Terrestrial Component Test Procedures

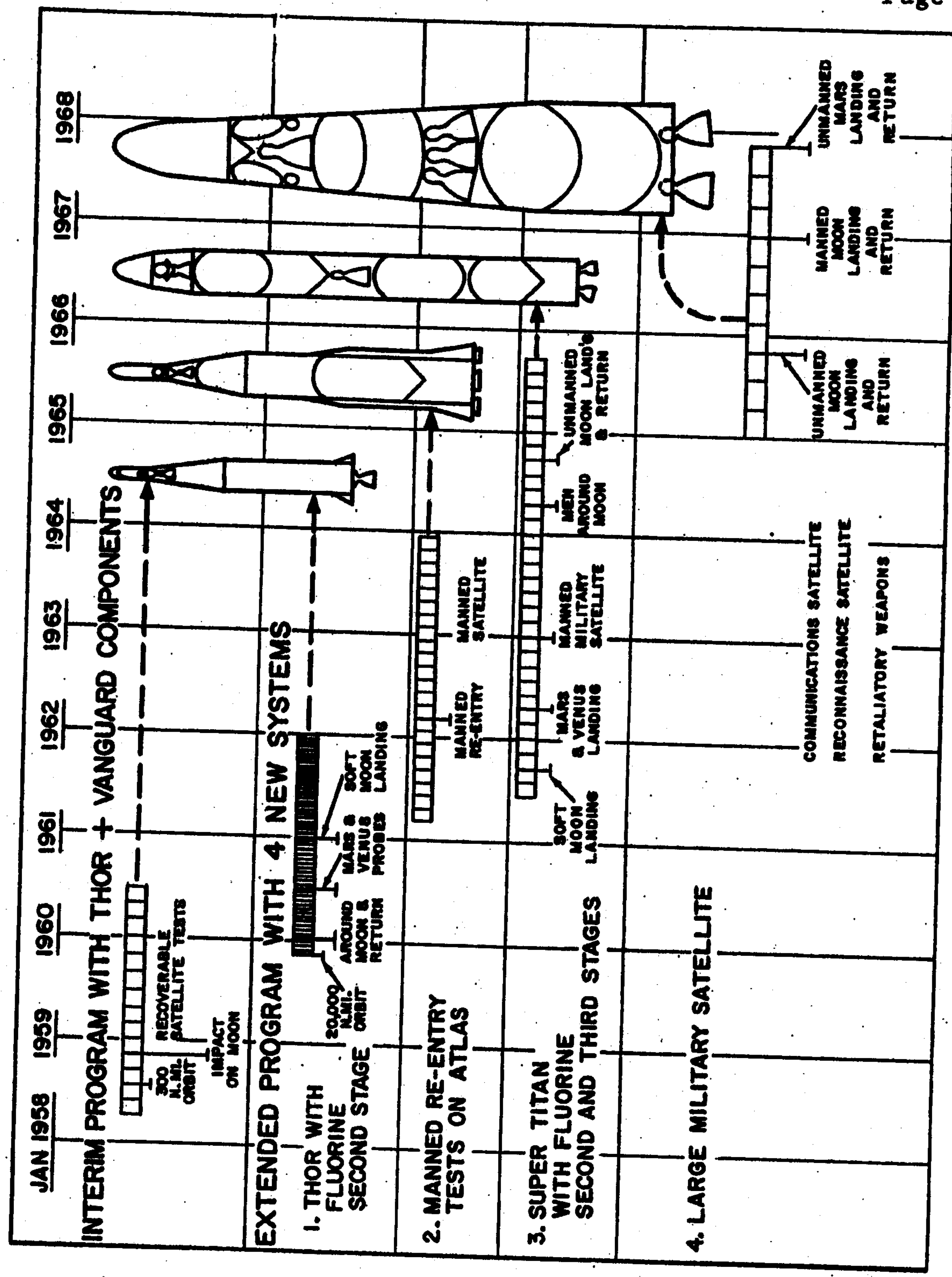

Figure 16. Air Force Military Satellite and Space Technology Calendar.

space exploration. It should also serve to crystallize thinking on astronautics development and lead to carrying out a sound program to keep the United States ahead in this vital area.

C. Supplemental Research Program

The specific nature of the developments required in the way of astronautic vehicles over the next decade have been laid out in this plan, together with a somewhat less detailed description of the guidance, instrumentation and payload developments for astronautics. It is obvious that to accomplish these developments a large supporting research program of basic technology in the areas of rocket propulsion, rocket vehicle structures, satellite and re-entry vehicles, nuclear and solar power supplies, electronics in all its aspects, and astrogeophysics will be essential.

D. Conclusions

The preceding sections covering the technical aspects of a development plan for astronautics lead to some significant general conclusions. These may be summarized as follows:

1. The present Atlas, Titan, and Thor provide the basic booster capacity for space missions of primary interest for the next five years. The principal investment for first generation space vehicles has been made already.

2. Development of a few added stages of small size as "building blocks" can in proper combination with the boosters provide vehicles for all space missions for the next five to seven years.

3. The guidance systems for present and second generation ICBM's are basically adequate for the launch phase of space missions. Guidance systems for midcourse, for satellite recovery, and for lunar and planetary landings can be developed using presently available guidance concepts.

4. The basic characteristics of "Payloads" for many of the astronautics missions (including "Payloads" for animal and manned experiments) should be defined immediately and development initiated.

5. Research and technical development required on critical problems and on basic space phenomena can be defined and initiated now for the second generation of space vehicles and space missions.

6. The limited investigations carried out to date show the military value of astronautics program to be very great. In addition the possible return from the scientific aspects of such a program would produce a further contribution of inestimable value to civilization.

The Appendix consists of seven reports. which are reprinted unchanged in content. Only the pagination has been altered.

The unclassifiedportion of the appendix is issued as a separate document for the convenience of all users.

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APPENDIX

Volume I

A. Vehicles

A.1 Performance Estimates for Some Air Force Ballistic Missile Stage Combinations

Volume II

B. Space Trajectories

B.1 On the Motion of a Satellite in the Gravitational Field of the Oblate Earth

B.2 Lunar Trajectories

B.3 Accuracy Requirements for Interplanetary Ballistic Trajectories

B.4 Three-Dimensional Interplanetary Ballistic Trajectories

C. Re-entry Trajectories

C.1 Analytic Description of the Re-entry Trajectory

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APPENDIX A.1

PERFORMANCE ESTIMATES FOR SOME AIR FORCE BALLISTIC MISSILE STAGE COMBINATIONS

Published as GM-TR-0165-00426, 7 July 1958

ABSTRACT

The payload capabilities of vehicles using Air Force Ballistic Missile stages are studied with reference to their suitability for space experiments. Sixteen combinations of 18 separate vehicles are shown in terms of their payload capabilities for various space flight missions ranging from a 10,000-naut-mi ballistic missile range to a round trip from earth to Mars.

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CONTENTS

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CONTENTS (Continued)

ILLUSTRATIONS

ILLUSTRATIONS (Continued)

ILLUSTRATIONS (Continued)

TABLES

PERFORMANCE ESTIMATES FOR SOME AIR FORCE BALLISTIC MISSILE STAGE COMBINATIONS

1.0 INTRODUCTION

This report presents the results of a study of the payload capabilities for space experiments of vehicles using Air Force Ballistic Missiles as stages. The rocket stages presently being developed as part of IRBM and ICBM programs are suitable for large-scale space experiments. Other rocket vehicle programs are producing smaller stages which can be used in combination with IRBM and ICBM equipment. In addition, advanced high-performance stages are being considered specially for space experimentation. The combinations of such stages studied in this report do not represent all feasible possibilities but they do include a number of arrangements with some very attractive capabilities for launching space experiment payloads.

Section 1.0 of this report defines the terminology of the report and summarizes the possible combinations. Section 2.0 presents the physical and performance characteristics of each stage studied. A standard scale of 1 ft = 100 ft is used for outline drawings of each stage in order to show relative sizes. Section 3.0 gives a description of the various stage combinations together with sketches to a scale of 1 ft = 360 ft. Section 4.0 shows the procedures used in obtaining performance estimates and the requirements for some typical experiments. Section 5.0 shows curves of payload for each vehicle considered as a function of equivalent velocity and mission requirements.

The terminology used in this report is shown in Figure 1-1. Only gross payloads are described in this report in order to separate comparisons of vehicle performance from various possible guidance, heat shield, power supply, and structural weights. The definitions shown for stages, vehicles, and combinations have been adopted for convenience in referring to equipment throughout the text of the report.

The basic measure of vehicle performance used here is a curve of equivalent velocity versus gross payload. Equivalent velocity is the velocity that a stage combination would impart to a payload if all rocket impulse were

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DEFINITIONS

Stage: A separable element, with propulsion.

Vehicle: The entire launch configuration. (Also first stage vehicle)

Nth Stage Vehicle: The assembly of stages with the Nth stage for propulsion.

Terminal Vehicle: The final portion of a vehicle containing the net payload, with or without propulsion.

Combination: The assembly of stages excluding the terminal vehicle.

Gross Payload: The burnout weight of the terminal vehicle.

Code: The numbers in parentheses behind the stage designation refer to the thrust and weight (in thousands of lb) of the stage respectively. For example, Fluorine (12-12) stage denotes a stage thrust of 12,000 lb and a weight of 12,000 lb.

4-Stage Space Vehicle

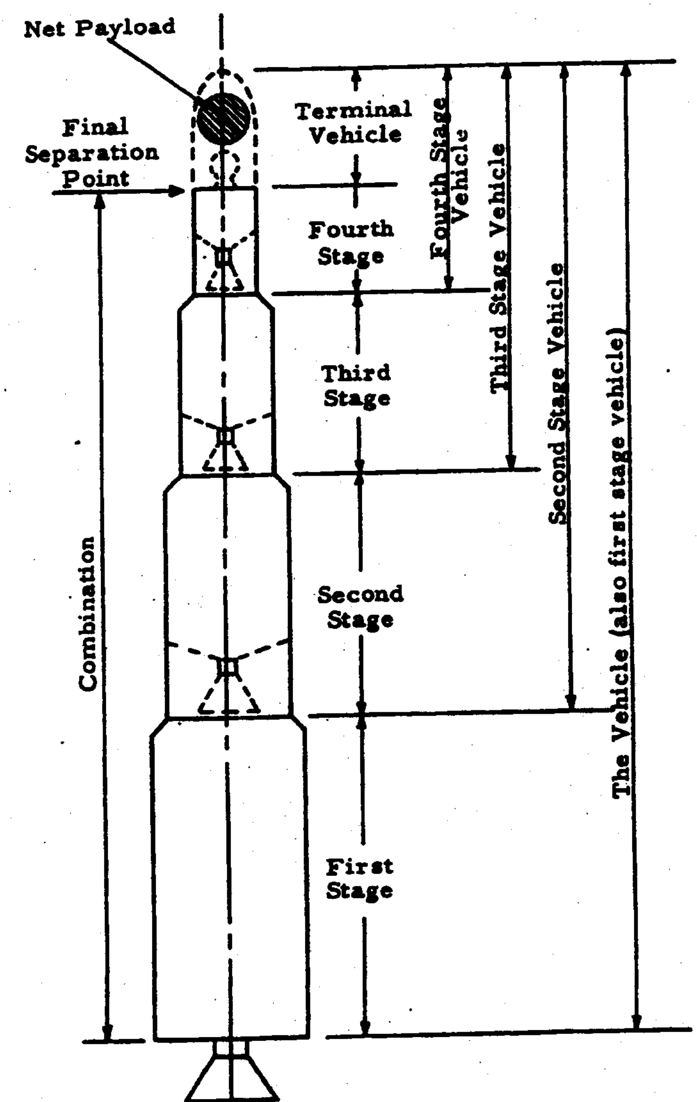

Figure 1-1. Space Vehicle Terminology Employed.

used for acceleration in one direction without gravity loss or drag. Each stage would impart a velocity increment

$$\Delta V = Ig \log \frac{W_o}{W_b}$$

where

I = specific impulse (lb-sec/lb)

g = 32.2 ft/sec^2

W_o = initial weight of vehicle

W_b = burnout weight of vehicle.

The total velocity would be the sum of the individual stage velocity increments.

In actual space flight, various velocity increments might be used to accelerate, maneuver, or decelerate, and consequently the total velocity required need not necessarily equal the equivalent velocity. Designation of performance by equivalent velocity is merely a convenient means for separating vehicle capabilities from specific flight trajectories. It is convenient to include in equivalent velocity the gravity and drag losses associated with achieving some typical satellite orbit. A 200-naut-mi altitude circular orbit has been used for this purpose. Figure 1-2 shows a summary of equivalent velocity requirements for some typical space flights.

Table 1-1 shows all the stages and combinations considered in this report. Data shown for each stage includes weight, thrust, propellants, and development status (current, product improvement, modification, and new). The stages which are combined to form the vehicles studied are arranged on the horizontal lines. The combinations in which each stage is used can be found in the vertical columns.

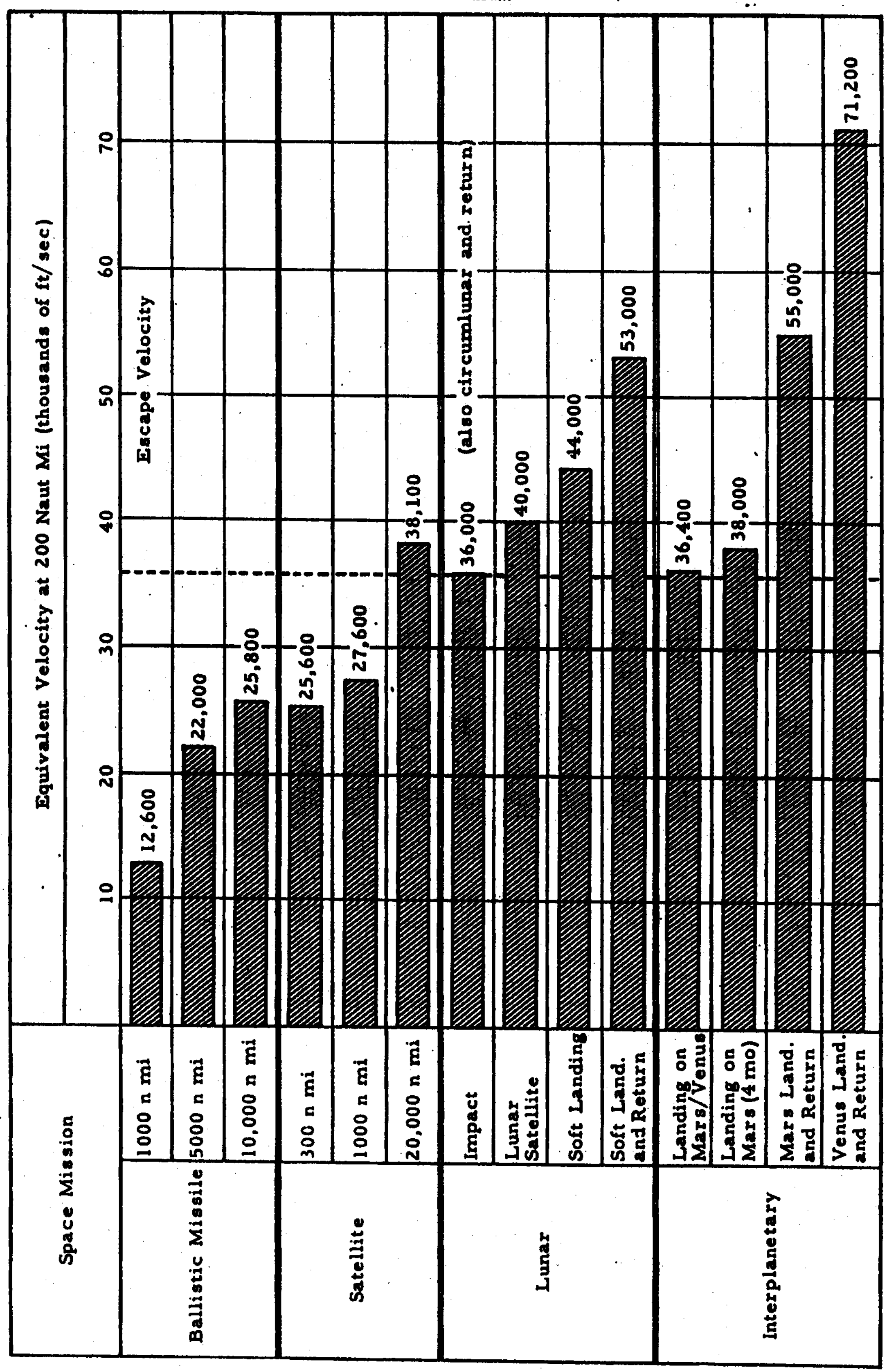

Figure 1-2. Equivalent Velocity Requirements.

Combinations Studied.

Thor (175-107)	Fluorine (12-12)	Fluorine (12-30)	Vanguard II	Vanguard II (Mod)	(4) Vanguard III's	XM-34	WS 117L (Mod)	Sergeant	Vanguard III
2-9	2-10	2-11	2-12	2-13	2-14	2-15	2-16	2-17	2-18
106,600	12,000	30,000	4,232	6,300	1,800	8,635	6,000	8,185	435
175,000	12,000	14,000	7,700	7,700	9,400	62,000	15,000	50,000	2,350
Lox-RP	Fluorine Hydrazine	Fluorine Hydrazine	WFNA-UDMH	WFNA-UDMH	Solid	Solid	RFNA-RP	Solid	Solid
Product Improvement	New	New	Current	Modified	Modified	Product Improvement	Current	Current	Current
■			■						
■	■								
■									■
■					■				
■								■	
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A summary of the payload capabilities of the combinations considered in this report is shown in Table 1-2. Payloads for various satellite, lunar, and interplanetary experiments have been collated in this figure from the general performance curves in Section 5.0 of this report.

Comb. No.	Space Mission	Earth Satellites			Lunar				Interplanetary			
		300 n mi	1000 n mi	20,000 n mi	Impact Also Circumlunar and Return	Lunar Satellite	Soft Landing also Lunar Satellite	Soft Landing and Return	Impact on Mars or Venus	Impact on Mars (4 months)	Mars Landing and Return	Venus Landing and Return
1	Thor Comb.	340										
2		3,200	2,600	450	670	670			630	450		
3		60										
4		180	50									
5												
6		680	520	43	105				72	43		
7		290	220		40				25			
8		500	340		45				30			
9	Atlas Comb.	3,000	2,200	430	680	290	70		560	430		
10		3,300	2,500	570	800	410	130		700	570		
11		7,850	6,100	1,900	2,600	1,470	720		2,250	1,900		
12	Titan Comb.	5,800	4,600	1,400	1,950	1,070	420		1,700	1,400		
13		11,200	9,500	4,250	5,300	3,500	2,000	210	4,850	4,300	70	
14	Boosted Comb.	50,000	41,000	17,500	21,000	15,800	9,400	3,300	19,000	17,500	2,400	
15		48,000	36,000	11,000	15,000	8,200	3,500		13,000	11,000		
16		20,000	15,000	4,600	6,000	3,700	2,300	500	5,400	4,600	300	

Table 1-2. Gross Payload Capabilities of Combinations Studied for Selected Space Missions.

2.0 STAGES FOR SPACE VEHICLES

2.1 Description of Stages

The various space vehicle systems are illustrated and discussed in this report. In this section detailed information is presented on the stages which may be combined in order to complete the various space vehicle assemblies. Weight and performance data are given for each of the stages listed in Table 1-2. Note that the over-all length does not include interstage structure.

2.1.1 Booster (2000-1275) Stage

High-velocity missions (such as manned flights to the Moon) required a new class of vehicle combinations comprised of existing ballistic missiles plus new first-stage boosters. The Titan booster is shown in Figure 2-1. The propulsion system is composed of two engines of 1 million lb of thrust each, and utilizes a Lox-RP propellant combination. The specific impulse (vacuum) is expected to be 290 sec with an expansion ratio of 8. A turbopump feed will be employed.

The maximum diameter of the booster is 22 ft, with a length of 64 ft, exclusive of the interstage structure. The over-all vehicle (booster plus advanced Titan plus terminal vehicle) will be about 200 ft in length.

The full-up weight of this stage is 1.28 million lb.

2.1.2 Booster (1000-640) Stage

This stage is similar in purpose and design to the Titan booster described above (see Figure 2-2). The booster has one engine of 1.0 million lb of thrust, and uses the same Lox-RP propellant combination. The engine is identical to that employed in the Titan booster.

The weight of this stage is 640,000 lb. The maximum diameter is 19 ft, and the over-all length is 60 ft.

2.1.3 Atlas Stages

The standard Atlas (360-265) vehicle is shown in Figure 2-3. The vehicle is intended for use as is, without any structural modifications. It is estimated that a 13,000-lb terminal vehicle could be carried by the Atlas without altering the present structure.

The Atlas is commonly described as a one and one-half-stage vehicle; at staging, the two booster engines (in the outboard pods) are jettisoned along with structural portions of the pods and certain other equipment. The central sustainer engine then drives the missile during the next burning period. The Atlas has but one set of propellant tanks, which supply both the booster and sustainer engines. The tanks depend upon pressurization for adequate strength to meet structural loads. A more detailed description of the Atlas can be found in reference 1.

The Atlas (415-262) booster (Figure 2-6) is similar to the standard stage with the exception that the booster engine thrust is increased from 150,000 lb to 175,000 lb. The total thrust at take-off is thus uprated to 415,000 lb (including sustainer).

2.1.4 Titan I (301-170) Stage

The standard Titan first stage (Figure 2-4) employs two engines giving a combined thrust of about 301,000 lb. A Lox-RP propellant combination is employed, with a turbopump propellant feed system. The total weight of the first stage is about 170,000 lb. The Titan is described in more detail in reference 2.

2.1.5 Titan II (81-45) Stage

The standard Titan second stage (Figure 2-7) has one engine which produces 81,000 lb of thrust at start of burning. The engine has an expansion ratio of 25 and a chamber pressure of 660 psi, giving a specific impulse (vacuum) of 310 sec. A Lox-RP propellant combination is used, with a turbopump propellant feed system. The initial weight of the stage is about 44,500 lb.

Vernier motors are used on the second stage to adjust the cutoff velocity of the stage to the desired value.

2.1.6 Titan I (400-170) Stage

The improved Titan first stage (Figure 2-5) is identical to the standard Titan first stage except for thrust rating, which is increased from a total of 301,000 lb to 400,000 lb. This thrust uprating is achieved by increasing the propellant flow by means of higher turbopump speeds. The engines already have demonstrated this capability. No increase in propellant storage capacity is envisioned for this stage.

2.1.7 Fluorine (135-100) Stage

The fluorine-hydrazine second stage for the Titan vehicle is shown in Figure 2-8. The stage as shown has a total weight of 100,000 lb.

Two versions of the propulsion system are contemplated for this stage: one employing a pressurized tank system for propellant feed and producing 81,000 lb of thrust; the second or follow-on version using a turbopump feed and producing 135,000 lb of thrust. The pressurized feed system would be used first because of the relatively short development time. Helium would be used as the working gas and would be stored in the fluorine tank to reduce the storage tank weight.

The turbopump configuration for a high-altitude stage such as the present one is simplified considerably over a sea-level design, by the feasibility of using low combustion chamber pressure. The pressure rise, for example, is about one-tenth that of the usual propellant feed pump, thus reducing the turbine horsepower to a very manageable level. The use of hydrazine as a monopropellant provides a readily available turbine-drive gas supply. Replacing the pressurized gas propellant feed system with a turbopump installation should save about 1000 lb of structural weight, provided that low-pressure tanks are installed with the turbopump. The thrust uprating would be achieved through increased chamber pressure.

The general features of the fluorine-hydrazine second stage are similar to those of the Fluorine (12-12) stage described later. The motor expansion ratio is limited to about 18 in order to keep the nozzle exit diameter within the 10 ft outside diameter limit of the Titan. A specific impulse of 364 sec is expected for the motor when operated at high altitude. Cylindrical tanks with elliptical bulkheads have been used to reduce the over-all length of the vehicle to a minimum.

2.1.8 Thor (175-107) Stage

The Thor first stage is shown in Figure 2-9. The stage is similar to the standard Thor vehicle except for two modifications: first, the engine has been uprated to 175,000 lb at launch; and secondly, an additional 2000 lb of propellants have been provided. The Thor is described in reference 3.

The stage uses a Lox-RP propellant combination with a turbopump propellant feed system. The engine has an expansion ratio of 8, a chamber pressure of 560 psi, and produces a specific impulse (vacuum) of 290 sec.

No structural modifications to the current Thor are required. The vehicle has a maximum diameter of 8 ft, and an over-all length (excluding the guidance compartment) of 53 ft.

2.1.9 Fluorine (12-12) Stage

A preliminary design has been made on a high performance final power stage of 12,000 lb of weight which will fulfill several important space flight missions when used with existing or follow-on ICBM and IRBM boosters. Payload capability extends up to 7000 lb (see Figure 2-10).

The stage utilizes a fluorine-hydrazine propellant combination, which gives a specific impulse of about 364 sec at high altitudes. The chamber pressure of 150 psia provides near optimum stage performance for the engine of expansion ratio 18:1. Regenerative cooling of the engine using hydrazine as coolant appears feasible, with some film cooling required to ensure adequate throat cooling.

The pressurized propellant tanks are separate spheres of approximately equal volume, connected by a short cylindrical member. A nested tank configuration offers some advantage in reduced weight, but presents the difficult problem of intertank insulation needed to prevent freezing of the hydrazine by the low temperatures of the adjacent liquid fluorine.

A helium pressurization system has been used because of the short development time as compared to a turbopump feed. The helium tank is stored within the fluorine in order to reduce the size and weight of tank. The helium is heated to about 400°F by means of a motor heat exchanger before entering the propellant tanks.

Because of the small angular movements of the engine (± 4 deg), a universal flexure has been used for thrust vector control, rather than the conventional gimbals, which require moving or sliding parts. The flexures are designed to offer negligible resistance to small angular movements. Small servoactuators of conventional design are used for control.

A structural weight ratio of about 0.060 can be expected for this type of stage.

2.1.10 Fluorine (12-30) Stage

This stage (Figure 2-11) is a modification of the Fluorine (12-12) stage described above in that the propellant tank capacity has been increased to bring the over-all stage weight up to 30,000 lb. The engine and propellant feed system are the same as in the (12-12) stage. The Fluorine (12-30) stage is intended for use primarily with the boosted Titan.

2.1.11 Vanguard Second Stage

A weight statement and other performance data on the second stage of the Vanguard are given in Figure 2-12. This stage, with a gross weight of 4232 lb, uses the storable propellants white fuming nitric acid and unsymmetrical dimethyl-hydrazine stored in cylindrical tanks. A specific impulse of about 278 sec is obtained from the propellant combination.

The engine expansion ratio of 20:1 is near optimum for the chamber pressure of 200 psi. Helium pressurization is provided from a storage sphere located between the fuel and oxidizer tank. A small solid-propellant gas generator is used to raise the temperature and add additional charge before entering the propellant tanks.

The engine is gimbaled for thrust vector control. In the Vanguard installation, attitude control for the third stage is accomplished by spin stabilization.

2.1.12 Vanguard II (Modified) Stage

The pertinent design data for the Vanguard second-stage engine adapted to the Atlas booster is shown in Figure 2-13. The configuration shown is designed for a stage weight of 6300 lb with the manned re-entry test vehicle as the payload.

In this design the stage is shaped to fit between the Atlas nose and the re-entry nose cone with a minimum of length. The small Vanguard engine is located in the center, surrounded by toroidal tanks for the WFNA-UDMH propellants. The specific impulse, chamber pressure, expansion ratio, etc., are the same as the second-stage Vanguard engine. In order to reduce the volume of the helium pressurization system, a storage pressure of 5000 psi is used in each of two spheres located within the propellant tanks.

2.1.13 Clustered Vanguard III Stage

Four Vanguard III engines can be coupled together to form a final power stage for use in combination with the improved Thor booster. The Vanguard III units are unmodified. The total width of the package is slightly less then 40 in. and the length is about 45 in. The assembly band is carried to the top of the stage to provide support for the payload. The weight of additional structure required is estimated at 20 lb (see Figure 2-14).

2.1.14 XM-34 and Sergeant Stages

The XM-34 and Sergeant stages have nearly the same external dimensions, but differ in propellant characteristics, and hence, in thrust output.

The Sergeant stage (Figure 2-17) uses a T-17E2 propellant which gives a thrust of 50,000 lb and a specific impulse of 191 sec. The average chamber pressure is 535 psi.

The XM-34 stage (Figure 2-15) has an uprated propellant TRX-H-606A, which yields a thrust of 62,000 lb and a specific impulse of 226 sec. The average chamber pressure increases to 616 psi, and the gross weight of the stage to 8635 lb. The motor size also has been increased slightly.

2.1.15 WS 117L (Modified) Final Stage

The WS 117L Modified final stage (Figure 2-16) is similar in design to the Vanguard II modified stage since it is intended for use with the Atlas booster. The stage plus terminal vehicle is limited in weight to 9,300 lb.

The propellant tanks employed for this stage are identical to those used on the Vanguard stage. However, a new engine of 15,000-lb thrust, the YLR-91, has been used in place of the Vanguard II engine. The YLR-91 engine is longer, so that the stage length is increased to 17.3 ft. It is noted that the extended skirt used on this stage serves as interstage structure, since it couples directly onto the main structural frame atop the Atlas booster.

2.1.16 Vanguard Third Stage

Information on the Vanguard third stage engine is presented in Figure 2-18. A solid-propellant rocket with a gross weight of about 435 lb provides 2350 lb of thrust at altitude. A specific impulse of about 240 sec is obtained from the solid propellant polysulphide-ammonium-perchlorate. The chamber pressure operates in the range from 140 psi to a maximum of 250 psi with a nozzle expansion ratio of 18:1. Two small (1-1/2 in. OD x 5 in. long) propellant rocket motors are used to impart the necessary torque for spin-up.

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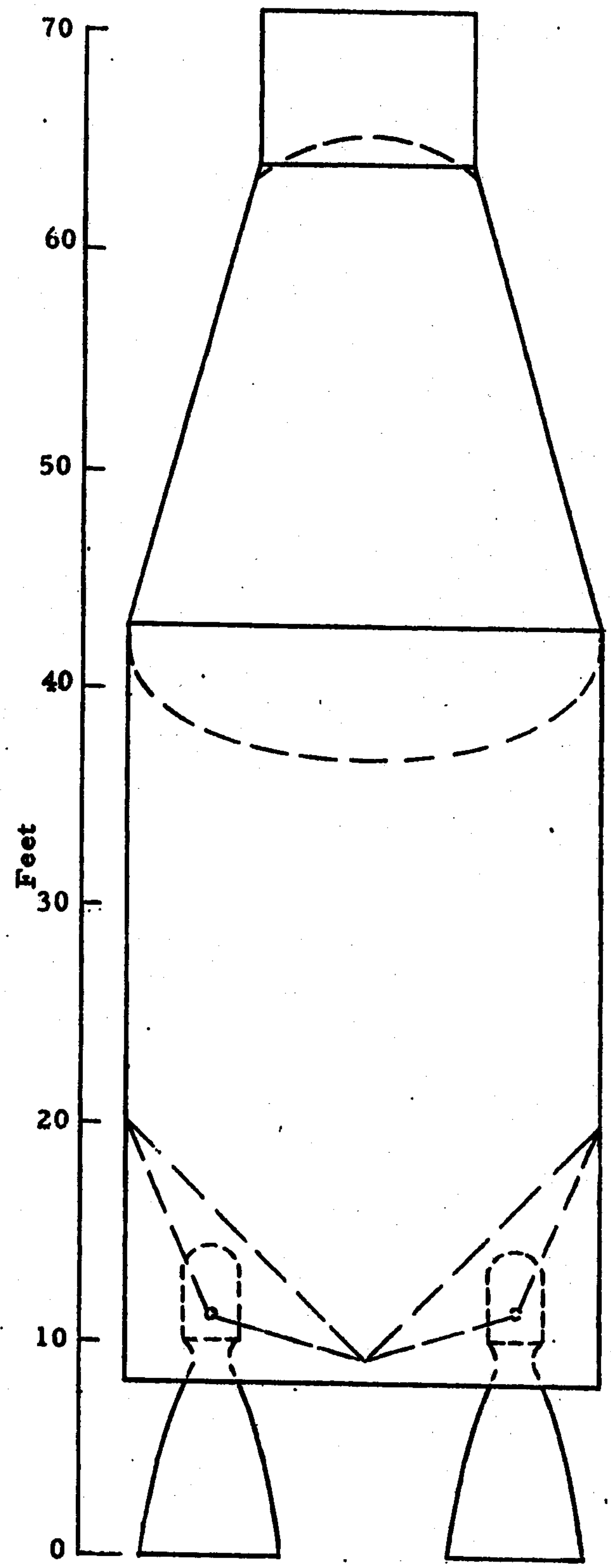
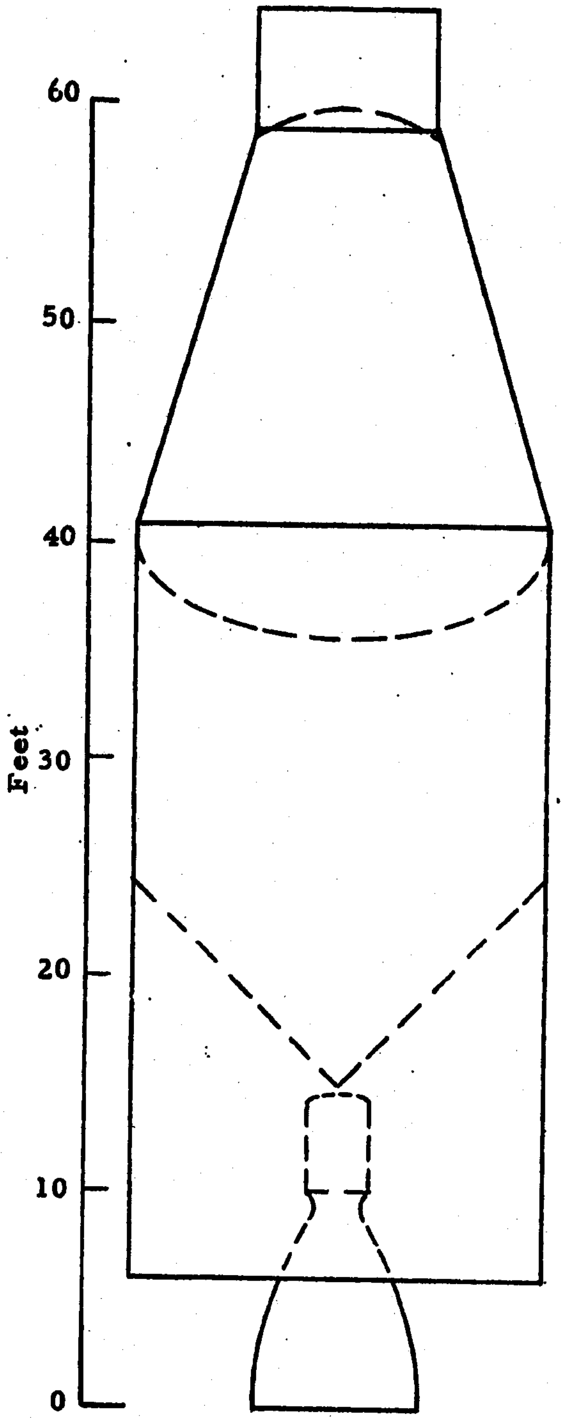


STAGE CHARACTERISTICS

Stage Weight	1.275 million lb
Burnout Weight	38,300 lb
Thrust	2.0 million lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	22 ft
Length	64 ft
Specific Impulse	290 sec
Total Impulse	3.37×10^8 lb-sec
Expansion Ratio	8
Chamber Pressure	750 psi
Exit Diam	7.5 ft

Figure 2-1. Booster (2000-1275) Stage.



STAGE CHARACTERISTICS

Stage Weight	640, 000 lb
Burnout Weight	19,100 lb
Thrust	1.0 million lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	19 ft
Length	60 ft
Specific Impulse	290 sec
Total Impulse	1.8×10^8 lb-sec
Expansion Ratio	8
Chamber Pressure	750 psi
Exit Diam	7.5 ft

Figure 2-2. Booster (1000-640) Stage.

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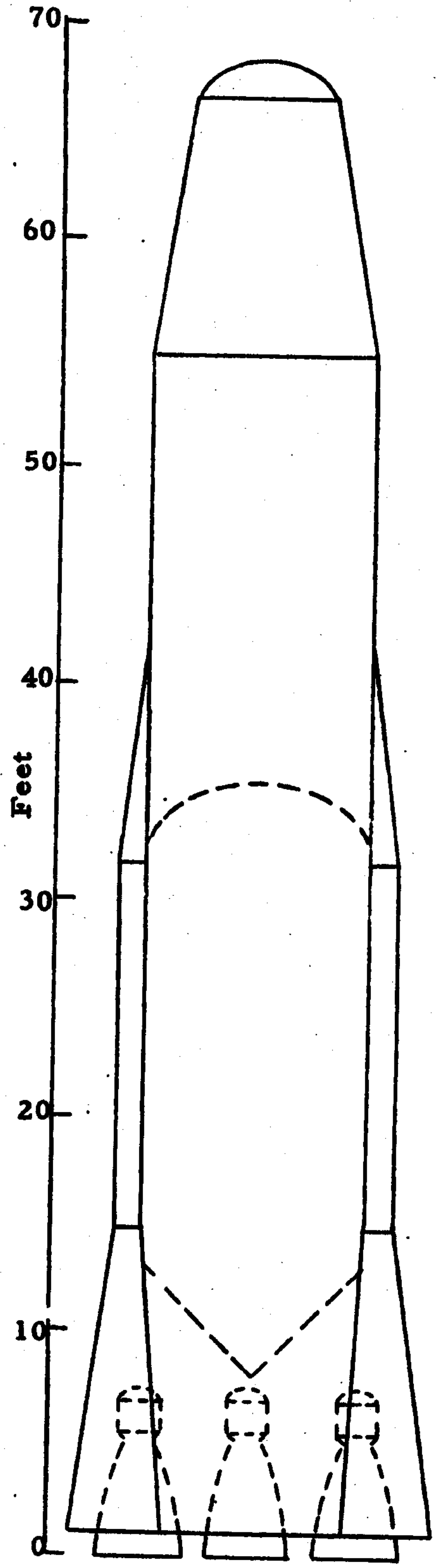

STAGE CHARACTERISTICS

Stage Weight	265,000 lb
Burnout Weight	8,300 lb
Thrust	360,000 lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	10 ft
Length	68 ft
Specific Impulse	290 sec
Total Impulse	7.2×10^7 lb-sec
Expansion Ratio	8 (25)
Chamber Pressure	530 (650 psi)
Exit Diam	4 (4) ft

(Sustainer engine)

Figure 2-3. Atlas (360-265) Stage.

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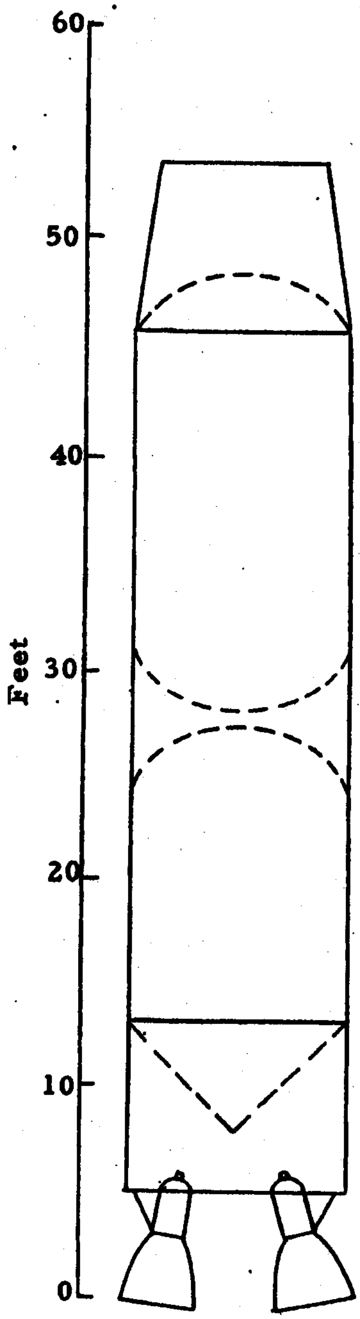
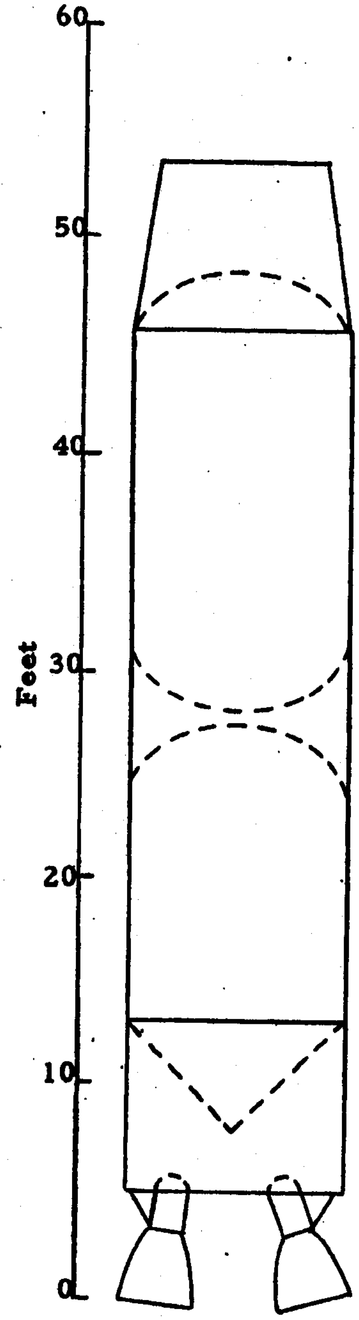


STAGE CHARACTERISTICS

Stage Weight	170, 000 lb
Burnout Weight	10, 000 lb
Thrust	301, 000 lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	10 ft
Length	48 ft
Specific Impulse	290 sec
Total Impulse	4.6×10^{7} lb-sec
Expansion Ratio	8
Chamber Pressure	560 psi
Exit Diam	3. 7 ft

Figure 2-4. Titan (301-170) First Stage.



STAGE CHARACTERISTICS

Stage Weight	170, 000 lb
Burnout Weight	10, 000 lb
Thrust	400, 000 lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	10 ft
Length	48 ft
Specific Impulse	290 sec
Total Impulse	4.6×10^7 lb-sec
Expansion Ratio	8
Chamber Pressure	750 psi
Exit Diam	3. 7 ft

Figure 2-5. Titan (400-170) First Stage.

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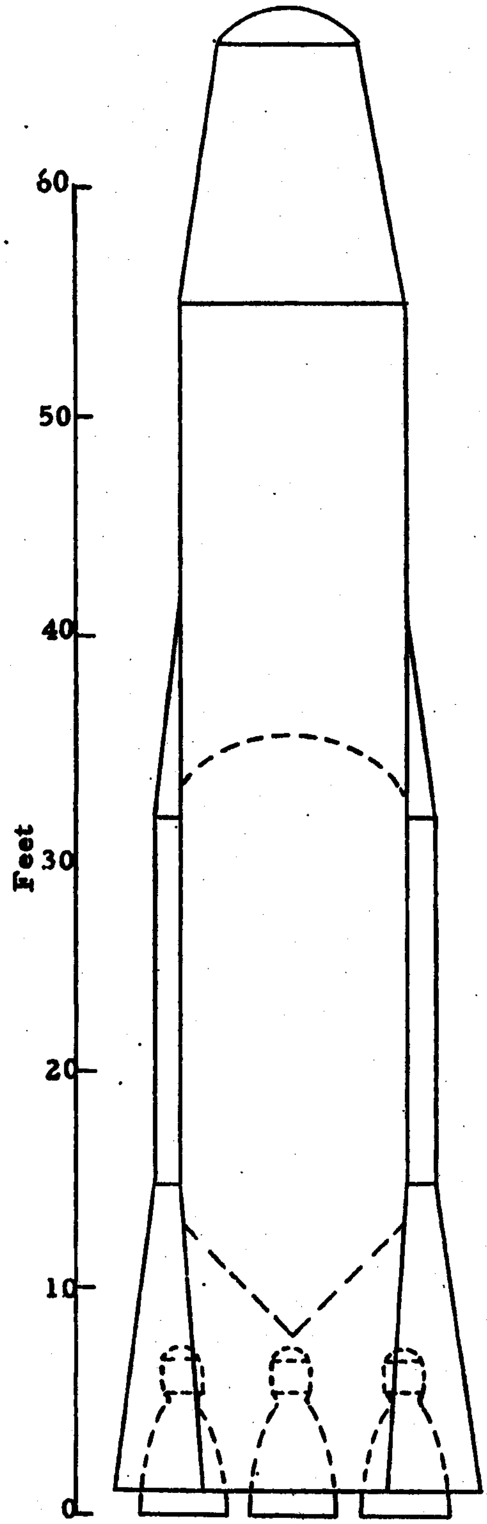

STAGE CHARACTERISTICS

Stage Weight	262,000 lb
Burnout Weight	8,300 lb
Thrust	415,000 lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	10 ft
Length	68 ft
Specific Impulse	290 sec
Total Impulse	7.2×10^7 lb-sec
Expansion Ratio	8 (25)
Chamber Pressure	530 (650) psi
Exit Diam	4 (4) ft
	(Sustainer engine)

Figure 2-6. Atlas (415-262) Stage.

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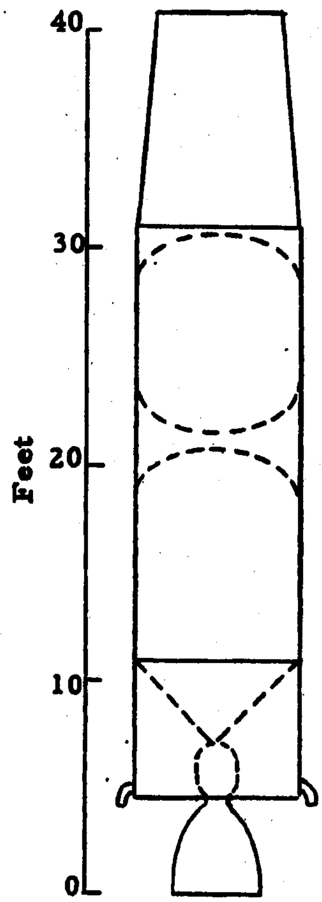

STAGE CHARACTERISTICS

Stage Weight	44, 500 lb
Burnout Weight	5, 800 lb
Thrust	80, 800 lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	8 ft
Length	31 ft
Specific Impulse	310 sec
Total Impulse	1.2×10^{7} lb-sec
Expansion Ratio	25
Chamber Pressure	660 psi
Exit Diam	3. 9 ft

Figure 2-7. Titan (81-45) Second Stage.

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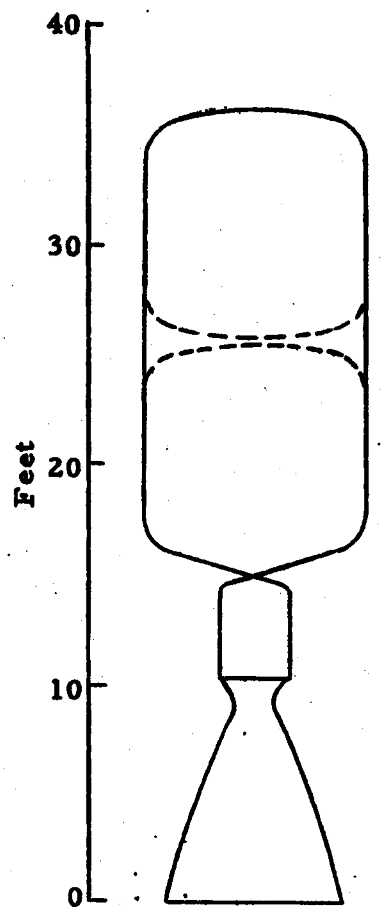

STAGE CHARACTERISTICS

Stage Weight	100,000 lb
Burnout Weight	6,600 lb
Thrust	135,000 lb
Propellants	F_2-N_2H_4
Propellant Feed	Turbopump
Maximum Diam	10.0 ft
Length	36.0 ft
Specific Impulse	364 sec
Total Impulse	3.4×10^7 lb-sec
Expansion Ratio	18
Chamber Pressure	150 psi
Exit Diam	7.5 ft

Figure 2-8. Fluorine (135-100) Stage.

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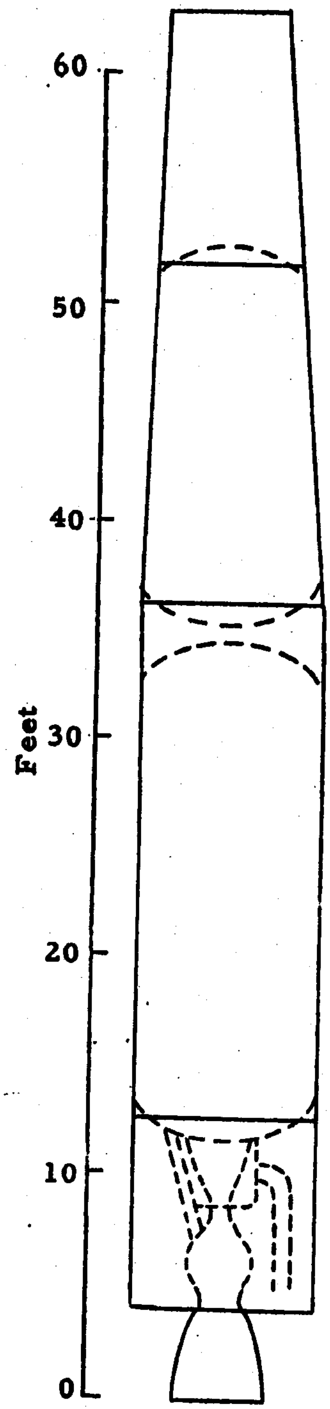

STAGE CHARACTERISTICS

Stage Weight	106,600 lb
Burnout Weight	10,100 lb
Thrust	175,000 lb
Propellants	Lox-RP
Propellant Feed	Turbopump
Maximum Diam	8 ft
Length	53 ft
Specific Impulse	290 sec
Total Impulse	2.9×10^7 lb-sec
Expansion Ratio	8
Chamber Pressure	560 psi
Exit Diam	3.7 ft

Figure 2-9.. Thor (175-107) Stage.

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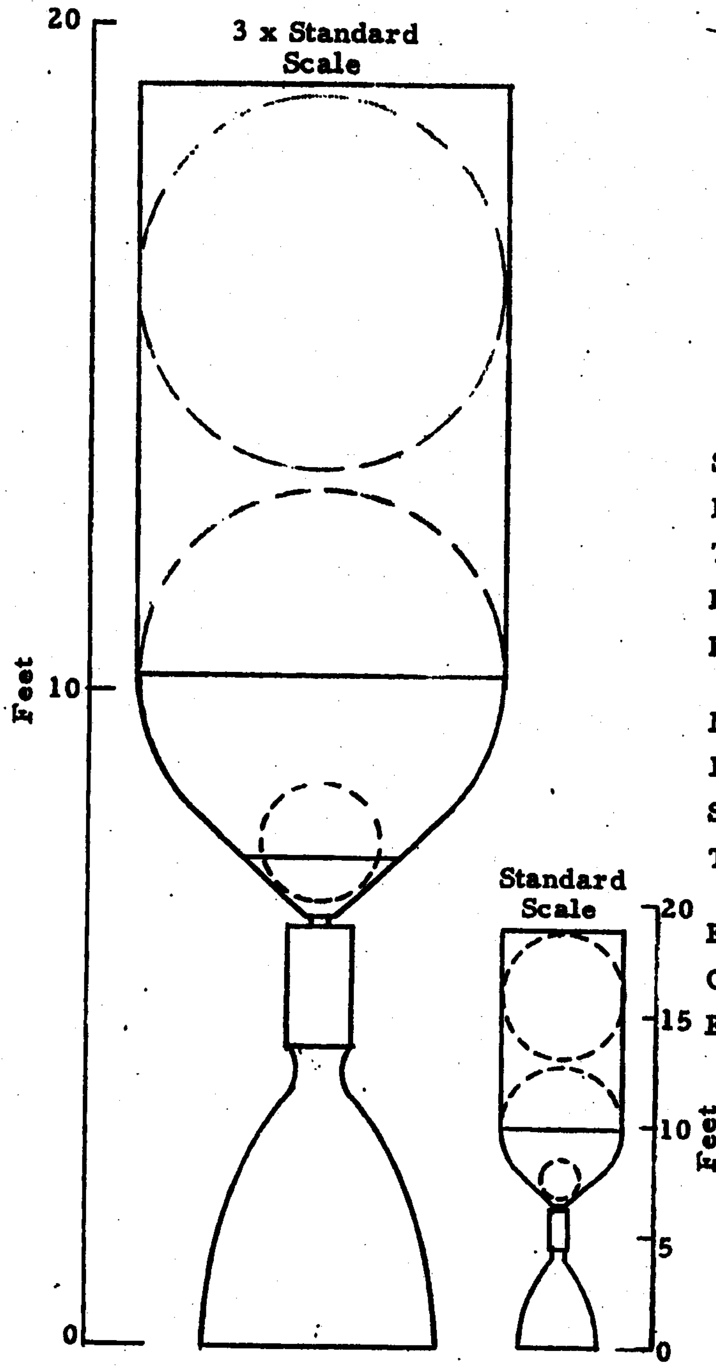

STAGE CHARACTERISTICS

Stage Weight	12,000 lb
Burnout Weight	720 lb
Thrust	12,000 lb
Propellants	F_2 - N_2H_4
Propellant Feed	Pressurized Helium
Maximum Diam	5.4 ft
Length	19 ft
Specific Impulse	364 sec
Total Impulse	4.1×10^6 lb-sec
Expansion Ratio	18:1
Chamber Pressure	150 psia
Exit Diam	2.8 ft

Figure 2-10. Fluorine (12-12) Stage.

[redacted]

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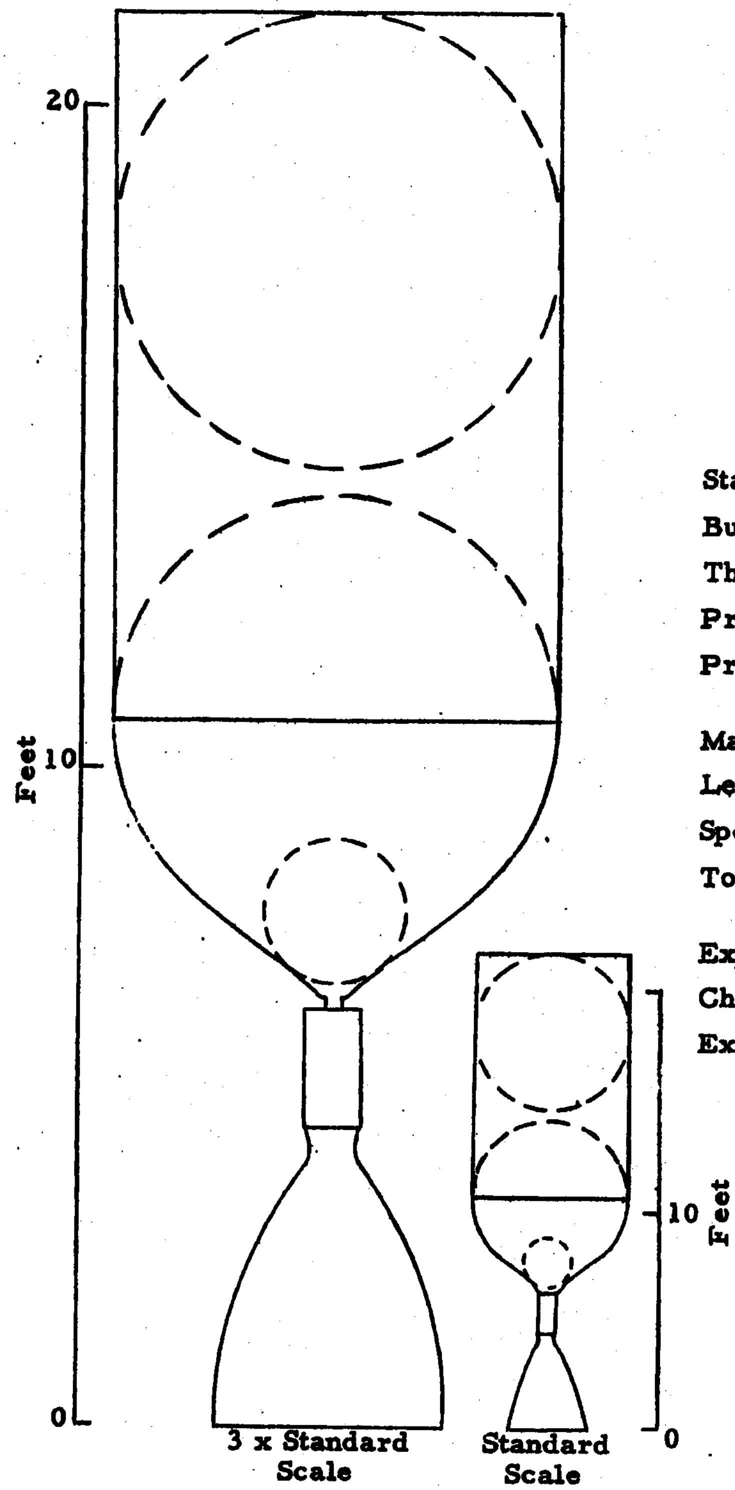

STAGE CHARACTERISTICS

Stage Weight	30,000 lb
Burnout Weight	1,800 lb
Thrust	12,000 lb
Propellants	$F_2 - N_2H_4$
Propellant Feed	Pressurized Helium
Maximum Diam	7.2 ft
Length	22 ft
Specific Impulse	364 sec
Total Impulse	10.3×10^6 lb-sec
Expansion Ratio	20
Chamber Pressure	150 psi
Exit Diam	2.8 ft

Figure 2-11. Fluorine (12-30) Stage.

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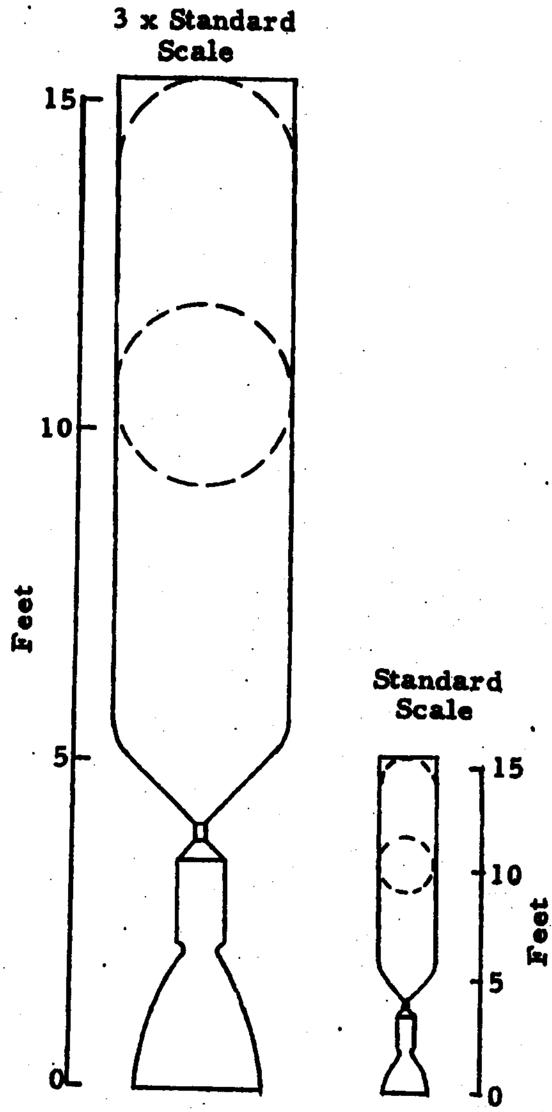

STAGE CHARACTERISTICS

Stage Weight	4, 232 lb
Burnout Weight	1, 037 lb
Thrust	7, 700 lb
Propellants	UDMH-WFNA
Propellant Feed	Pressurized Helium
Maximum Diam	2.7 ft
Length	15.0 ft
Specific Impulse	278 sec
Total Impulse	8.9×10^5 lb-sec
Expansion Ratio	20
Chamber Pressure	200 psi
Exit Diam	1.9 ft

Figure 2-12. Vanguard II Second Stage.

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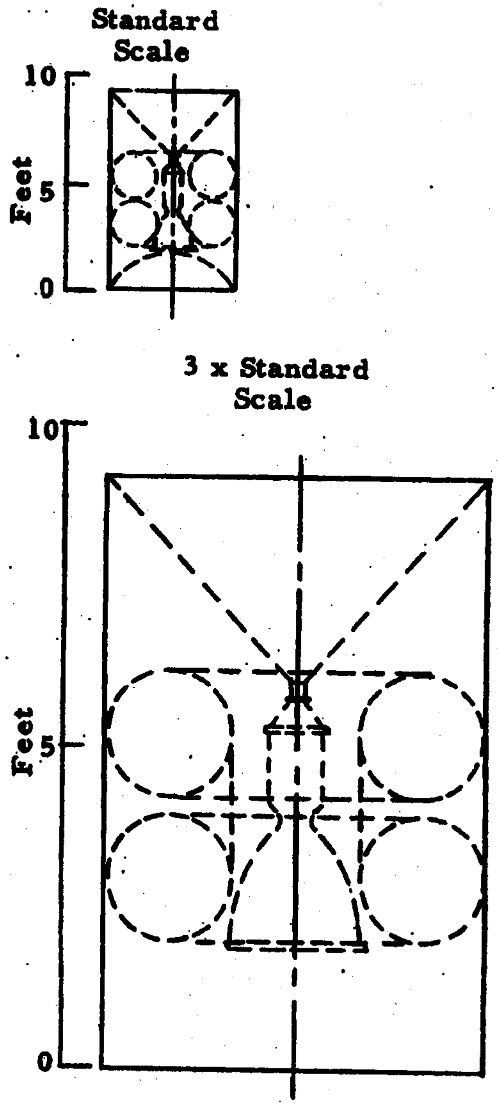
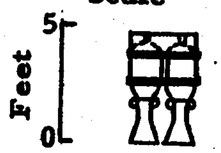


STAGE CHARACTERISTICS

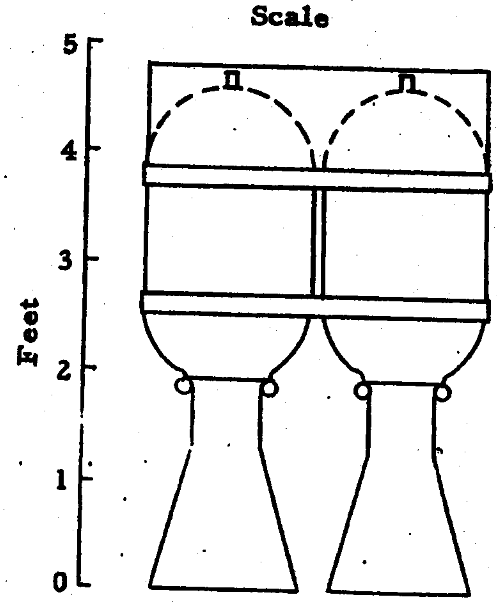
Stage Weight	6,300 lb
Burnout Weight	600 lb
Thrust	7,700 lb
Propellants	UDMH-WFNA
Propellant Feed	Pressurized Helium
Maximum Diam	6.3 ft
Length	9 ft
Specific Impulse	278 sec
Total Impulse	1.6×10^6 lb-sec
Expansion Ratio	20
Chamber Pressure	200 psi
Exit Diam	2.0 ft

Figure 2-13. Vanguard II (Modified) Stage.

Standard
Scale



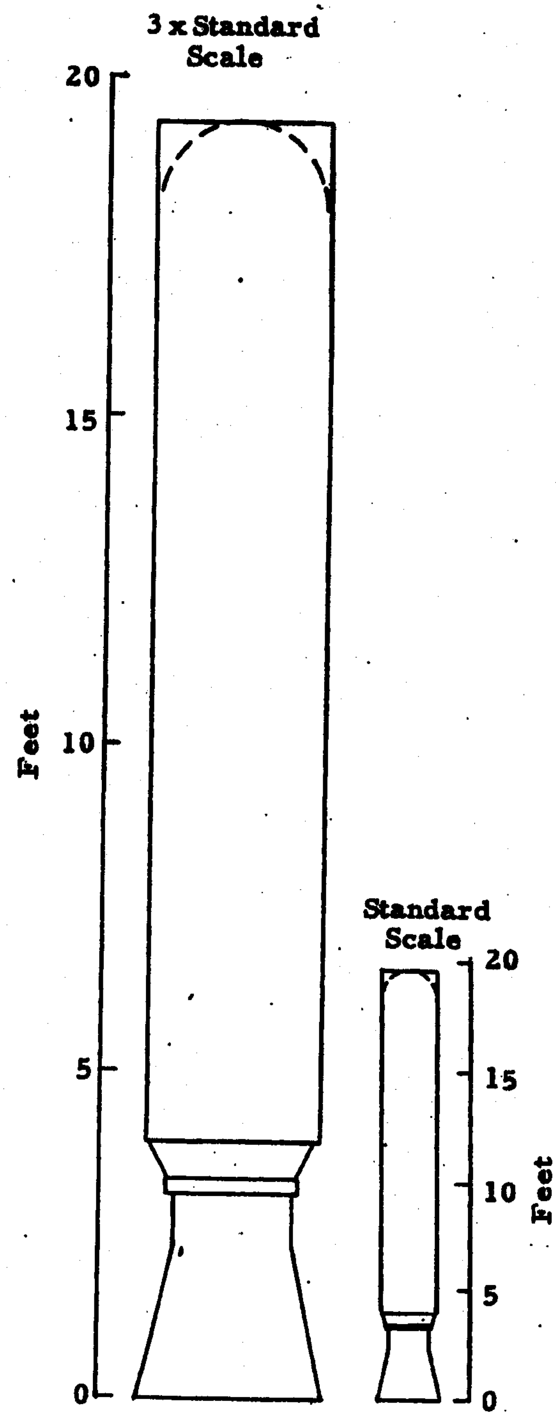
5 x Standard
Scale



STAGE CHARACTERISTICS

Stage Weight	1,800 lb
Burnout Weight	240 lb
Thrust	9,400 lb
Propellants	Solid
Maximum Diam	3.1 ft
Length	4.7 ft
Specific Impulse	240 sec
Total Impulse	3.7×10^5 lb-sec
Expansion Ratio	18
Chamber Pressure	140 psi
Exit Diam	1.3 ft

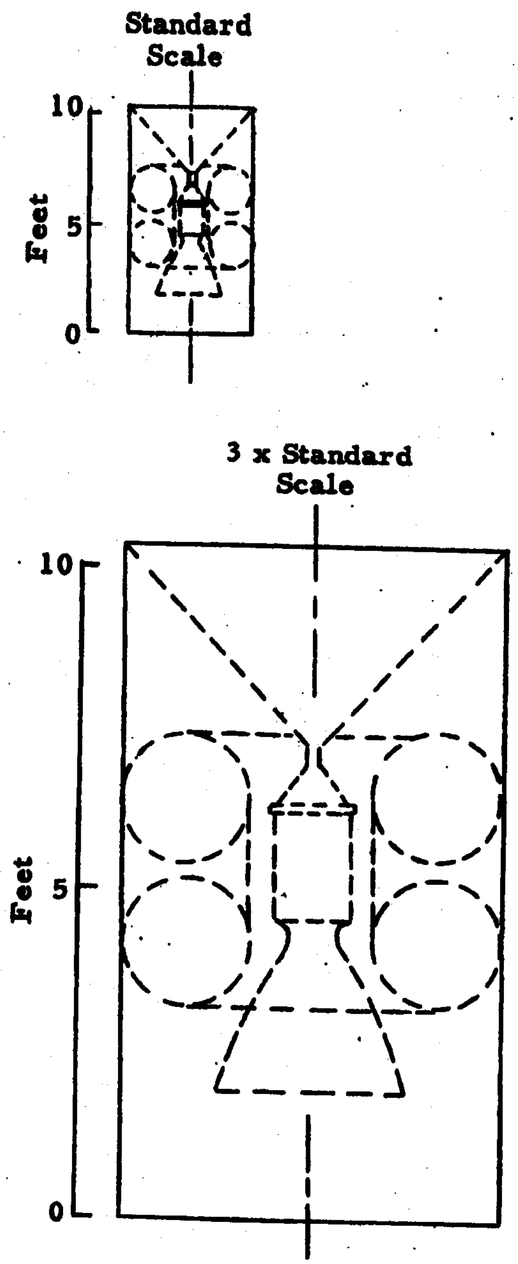
Figure 2-14. Four Vanguard III Stage.



STAGE CHARACTERISTICS

Stage Weight	8,635 lb
Burnout Weight	1,385 lb
Thrust	62,000 lb
Propellants	Solid
Maximum Diam	2.6 ft
Length	19.3 ft
Specific Impulse	226 sec
Total Impulse	1.6×10^6 lb-sec
Expansion Ratio	6.1
Chamber Pressure	616 psi
Exit Diam	1.9 ft

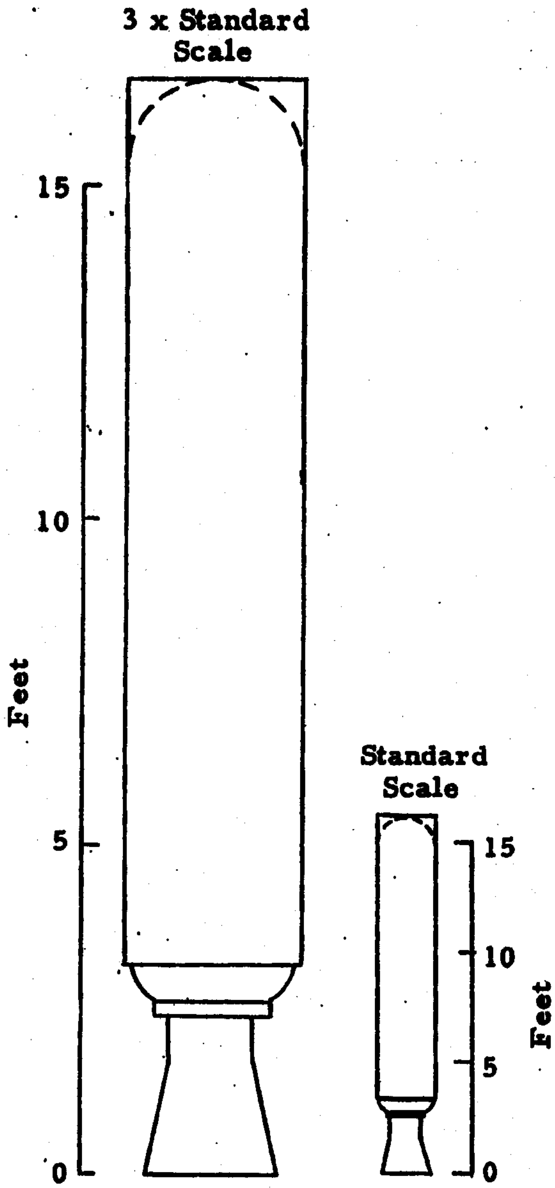
Figure 2-15. XM-34 Stage.



STAGE CHARACTERISTICS

Stage Weight	6,000 lb
Burnout Weight	700 lb
Thrust	15,000 lb
Propellants	RFNA-RP
Maximum Diam	6.3 ft
Length	17.3 ft
Specific Impulse	270 sec
Total Impulse	1.5×10^6 lb-sec
Expansion Ratio	20
Chamber Pressure	200 psi
Exit Diam	2.8 ft

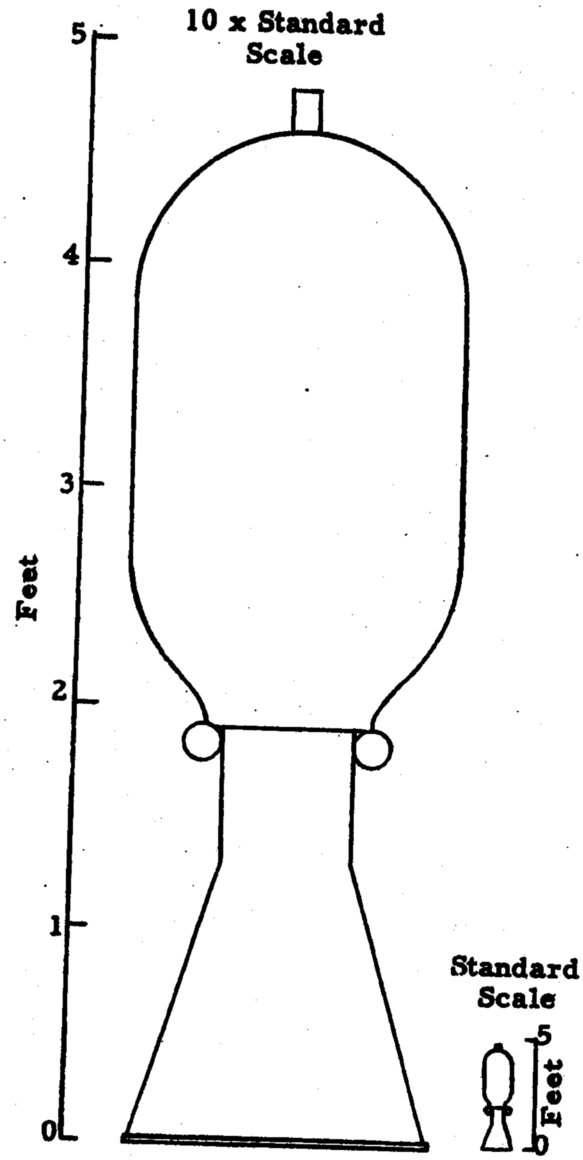
Figure 2-16. WS 117L (Modified) Final Stage.



STAGE CHARACTERISTICS

Stage Weight	8,185 lb
Burnout Weight	1,145 lb
Thrust	50,000 lb
Propellants	Solid
Maximum Diam	2.6 ft
Length	19.3 ft
Specific Impulse	191 sec
Total Impulse	1.3×10^6 lb-sec
Expansion Ratio	6.1
Chamber Pressure	535 psi
Exit Diam	1.9 ft

Figure 2-17. Sergeant Stage.



STAGE CHARACTERISTICS

Stage Weight	435 lb
Burnout Weight	55 lb
Thrust	2, 350 lb
Propellants	Solid
Maximum Diam	1. 5 ft
Length	4. 7 ft
Specific Impulse	240 sec
Total Impulse	9.2 x 10^4 lb-sec
Expansion Ratio	18
Chamber Pressure	140 psi
Exit Diam	1. 3 ft

Figure 2-18. Vanguard III Third Stage.

3.0 STAGE COMBINATIONS

This section contains descriptions and sketches of combinations of the stages described in Section 2.0. Detailed analyses of interstage structure, staging procedures, or load transmission between stages have not been conducted comprehensively. Neither has detailed consideration been given for all combinations to location of guidance, autopilot, power supply, and other similar equipment.

The sketches of combinations show configurations with typical interstage structure and a payload shape representative of a typical body for re-entry into the earth's atmosphere. Vehicle weights shown with the sketches consist of the sum of the stage weights plus the payload weight which the vehicle could put into a 300-naut-mi altitude circular satellite orbit.

3.1 Description of Space Vehicles

Most of the space vehicles of the next few years will be built around the present IRBM and ICBM boosters because the performance of these boosters is adequate for the requirements of the space flight program, and their production capacity has already been established. Furthermore, their reliability will have been developed to an acceptable level by the weapon system program.

The characteristics of the three weapons systems in their Initial Operational Capability (IOC) are given in this section. The description of additional stages which may be combined in various ways with the boosters to provide space vehicles are presented in Section 2.0. Sketches of the various vehicles are included with the descriptive material, along with a brief summary of payload capabilities.

3.1.1 Thor Space Vehicles

The Thor vehicle is produced and tested in sufficient quantities to ensure reliability, low cost, and availability for the space flight program. It is scheduled for availability on a noninterference basis after September, 1958. The nominal thrust of 150,000 lb can be increased to 175,000 lb by July, 1958, by adjustment of the propellant flow rates.

Many combinations involving the basic Thor stage have been studied. The most important applications include the Thor plus Vanguard II for low-altitude satellite missions, the Thor plus Vanguard II plus Vanguard III for a lunar shot, and Thor plus a fluorine final stage for general purposes. In addition, the Thor has been studied in combination with several solid stages.

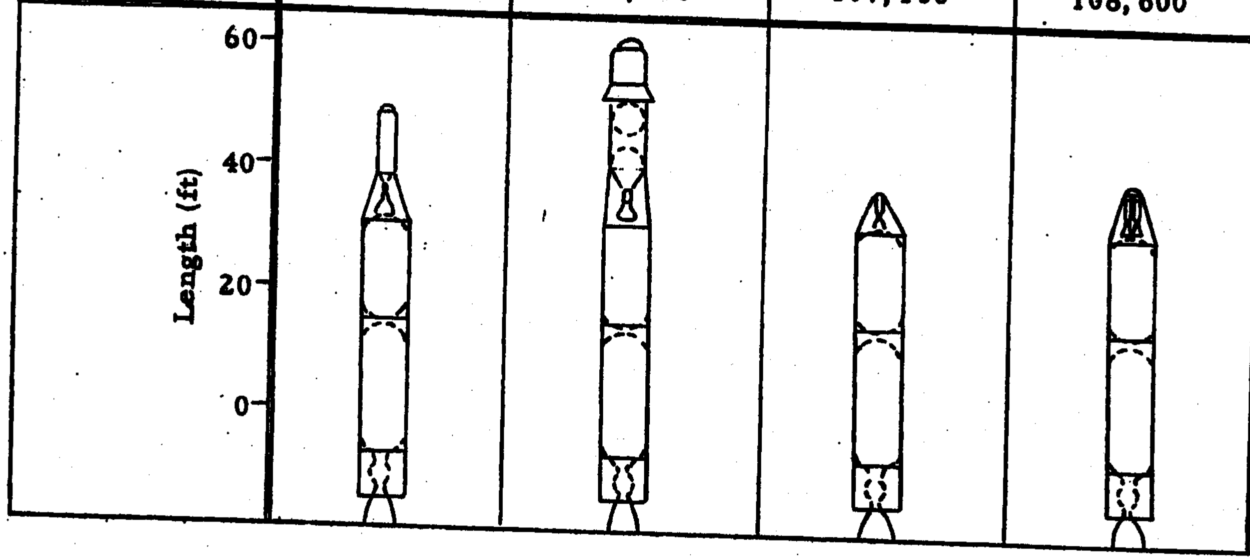
The Thor plus Vanguard Stage 2 can be assembled in a few months with available components, for an early operational capability as a low-altitude satellite with 300 to 400 lb net payload. The standard IOC Thor may be used, with nose cone and guidance platform removed. The Vanguard Stage 2 is attached to the nose cone ring at Station 50. The 32-in. diameter forward compartment of the Vanguard is provided with GE Mod III guidance system and autopilot, and extended forward to enclose the payload compartment with suitable nose fairing. Specific trajectories are given in Section 3.0.

Thor plus Vanguard Stage 2 plus Vanguard Stage 3 is capable of sending a 100-lb payload to the moon in the near future. It can be assembled in the same manner as the two-stage Thor-Vanguard above, with the addition of a spin stabilized Stage 3 mounted on the forward skirt of Stage 2. Stage 3 is designed to be spun-up by means of small powder charges to about 3 rps before separation from Stage 2 which reduces the effect of thrust and mass misalignment of the final stage firing period to less than 1 deg. The 200-lb guidance system mounted in Stage 2 detracts only slightly from the payload of the final stage. After separation, Stage 3 will maintain its attitude in space, but it needs to be attitude controlled by stage before spin-up.

Thor plus Fluorine (12-12) stage provides a low-altitude satellite payload from seven to nine times as great as the Thor-Vanguard. The second stage, which is scheduled for flight readiness in 18 months, consists of two spherical tanks and the engine is tandem. A modified adaptor cone, which transmits thrust loads to Stage 2 from the Thor, extends forward from Station 150 to engage the equator of the rear tank. The two tanks are joined by a short cylinder, from which the nose shell projects forward to house the payload compartment. Guidance equipment weighing 200 lb is carried, in addition to the payload of 3240 lb for a 200-stat-mi orbit.

The performance of the Thor in combination with various solid propellant stages has also been computed. A list of the combinations is given in Table 1-2. Sketches of the Thor vehicles are shown in Figure 3-1.

Combination No.		1	2	3	4
Stages Employed	1st	Thor (175-107)	Thor (175-107)	Thor (175-107)	Thor (175-107)
	2nd	Vanguard II	Fluorine (12-12)	Vanguard III	4 Vanguard III's
Gross Weight (lb)		111,200	121,700	107,100	108,600



Combination No.		5	6	7	8
Stages Employed	1st	Thor (175-107)	Thor (175-107)	Thor (175-107)	Thor (175-107)
	2nd	Sergeant	Vanguard II	4 Vanguard III's	XM-34
	3rd	---	Vanguard III	Vanguard III	Vanguard III
Gross Weight (lb)		114,700	112,000	109,100	116,200

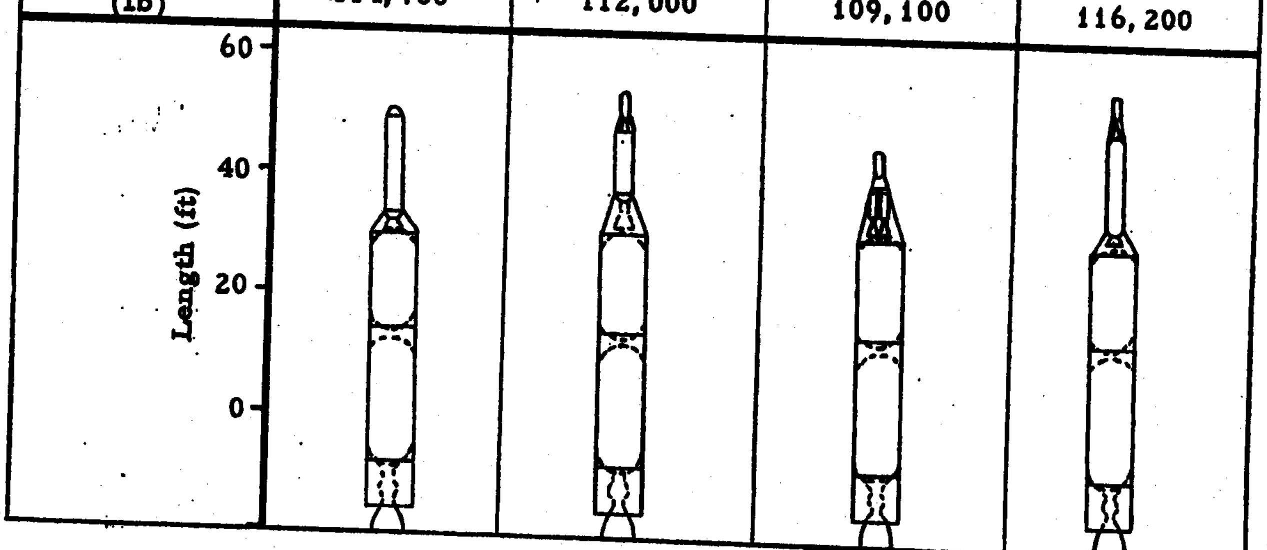


Figure 3-1. Thor Vehicles.

Combination No.		9	10	11
Stages Employed	1st	Atlas (360-265)	Atlas (360-265)	Atlas (415-262)
	2nd	Vanguard II (Mod)	WS 117L (Mod)	Fluorine (12-12)
Gross Weight (lb)		274,000	274,600	281,700

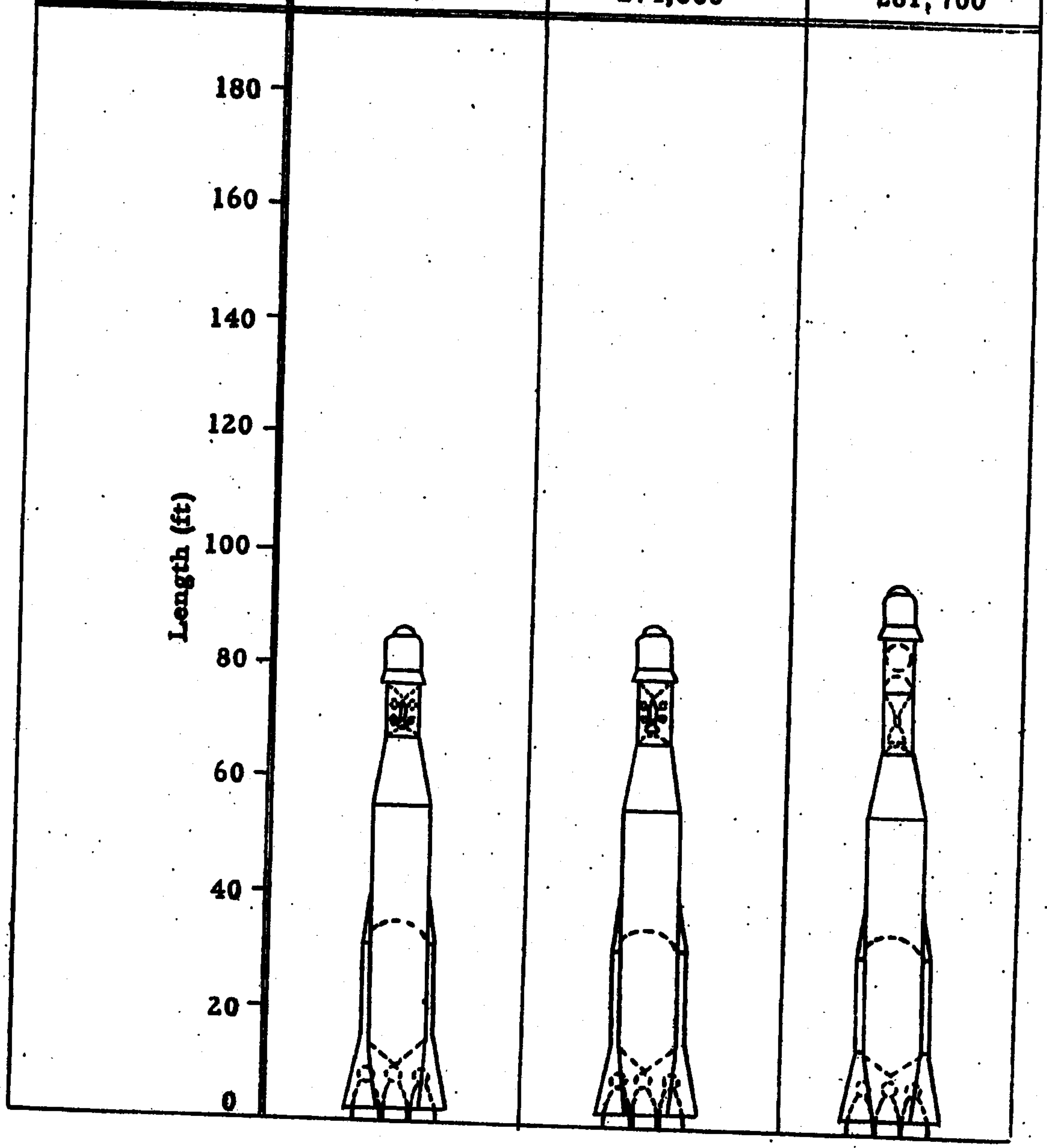


Figure 3-2. Atlas Vehicles.

3.1.2 Atlas Vehicles

Three combinations of Atlas vehicles are described in this section. The Atlas stages are scheduled for availability by mid-1959, and by that time will have been tested in sufficient numbers to improve the reliability.

The first space flight Atlas will have for the final stage the Vanguard Stage 2 engine, driving a 9300-lb stage. Because of the large diameter of the nose cone for this purpose, the final stage is shaped to fit between the Atlas nose and the re-entry nose cone with a minimum of length. The small Vanguard engine is placed at the center, surrounded by toroidal tanks for the WFNA-UDMH propellants. This compact arrangement minimizes the structural problems of mounting the long nose on the thin-walled Atlas tanks. The payload for a 300-naut-mi orbit is 3200 lb in addition to 500 lb of guidance and electronic gear.

The second final stage considered for use with the Atlas booster is very nearly the same as the Vanguard II modification. The only difference is the use of the slightly larger YLR-91 engine in place of the Vanguard II engine. The propellant tanks are arranged in toroidal fashion, as before. This latter stage is designated the WS 117L (modified) stage.

Also considered is the Atlas (415-262) plus Fluorine (12-12) stage. The standard Atlas booster thrust has been increased from 150,000 lb per engine to 175,000 lb for this application.

Sketches of the Atlas vehicles are shown in Figure 3-2.

3.1.3 Titan Vehicles

As shown in Figure 3-3, two variations of the Titan are illustrated, the standard Titan and an advanced Titan version. Availability of Titans on a noninterference basis is scheduled for late 1959 with thrust uprated to 400,000 lb.

The standard Titan plus Fluorine final stage is capable of carrying 5900 lb (plus guidance) in a 300-naut-mi orbit. The Fluorine final stage as illustrated is the same stage as used on Thor. The alternate shorter arrangement as on the Atlas could be used if desired. (See Section 2.0 for description of the final stages.)

Combination No.		12	13
Stages Employed	1st	Titan I (301-170)	Titan I (400-170)
	2nd	Titan II (81-45)	Fluorine (135-100)
	3rd	Fluorine (12-12)	Fluorine (12-12)
Gross Weight (lb)		232,500	293,500

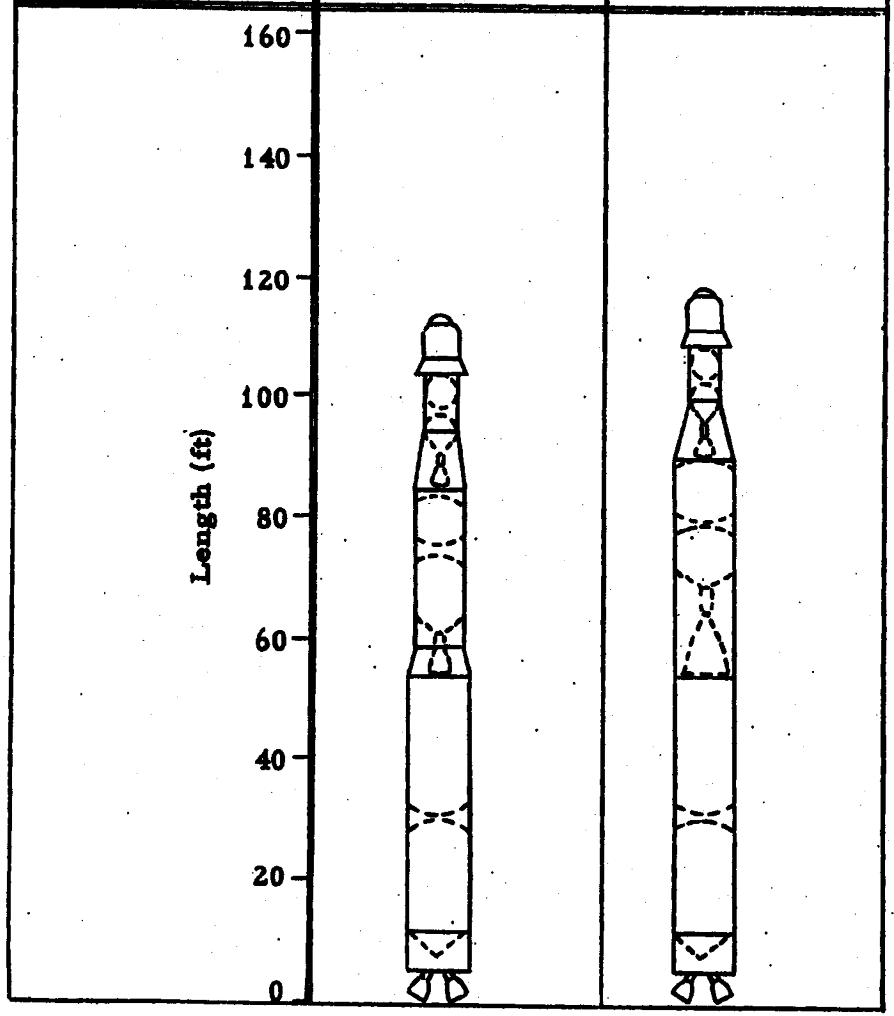


Figure 3-3. Titan Vehicles.

The advanced Titan consists of the standard Titan booster with a fluorine-hydrazine second stage weighing 50,000 lb and a thrust of 135,000 lb and a fluorine-hydrazine third stage weighing 12,000 lb, the same unit as developed for the Thor. The advanced Titan could be scheduled for operation after a three-year development period for the second stage propulsion system. The payload of this combination is about 9800 lb (plus 200-lb guidance) in a low-altitude orbit.

3.1.4 Boosted Thor and Titan Vehicles

The next step in space flight involves the addition of boost stages to the existing or improved Thor and Titan missiles. The primary use for the boosted configurations lies in the high velocity missions, such as high-altitude satellite applications, lunar shots, or interplanetary missions. Payload capabilities for these applications are sufficient for manned trips to the moon. For example, in the case of the 20,000-naut-mi satellite mission, the payload capability of the IOC Titan with the Fluorine final stage is increased from 1000 lb to about 6500 lb through addition of a 2-million-lb thrust booster stage, as shown in Figure 3-4.

One combination was studied using a boosted Thor stage. It consists of the Booster (1000-640), the Thor (175-107) stage, and the Fluorine (12-12) stage. One million lb of thrust is available at sea level to lift a gross weight of 759,000 lb exclusive of payload.

Two combinations using boosted Titan stages were investigated. One consists of the Booster (2000-1275), with standard Titan first and second stages, i.e., Titan I (301-170) and Titan II (81-45). The other is the largest and best performing combination studied, consisting of the Booster (2000-1279), the Titan I (400-170), the Fluorine (135-100) and the Fluorine (12-30) stages. These combinations weigh 1,480,000 lb and 1,575,000 lb, respectively, exclusive of payload. Two million lb of sea-level thrust is available at launch.

Combination No.		14	15	16
Stages Employed	1st	Booster (2000-1275)	Booster (2000-1275)	Booster (1000-640)
	2nd	Titan I (400-170)	Titan I (301-170)	Thor (175-107)
	3rd	Fluorine (135-100)	Titan II (81-45)	Fluorine (12-12)
	4th	Fluorine (12-30)	--	--
Gross Weight (lb)		1,625,000	1,536,000	778,600

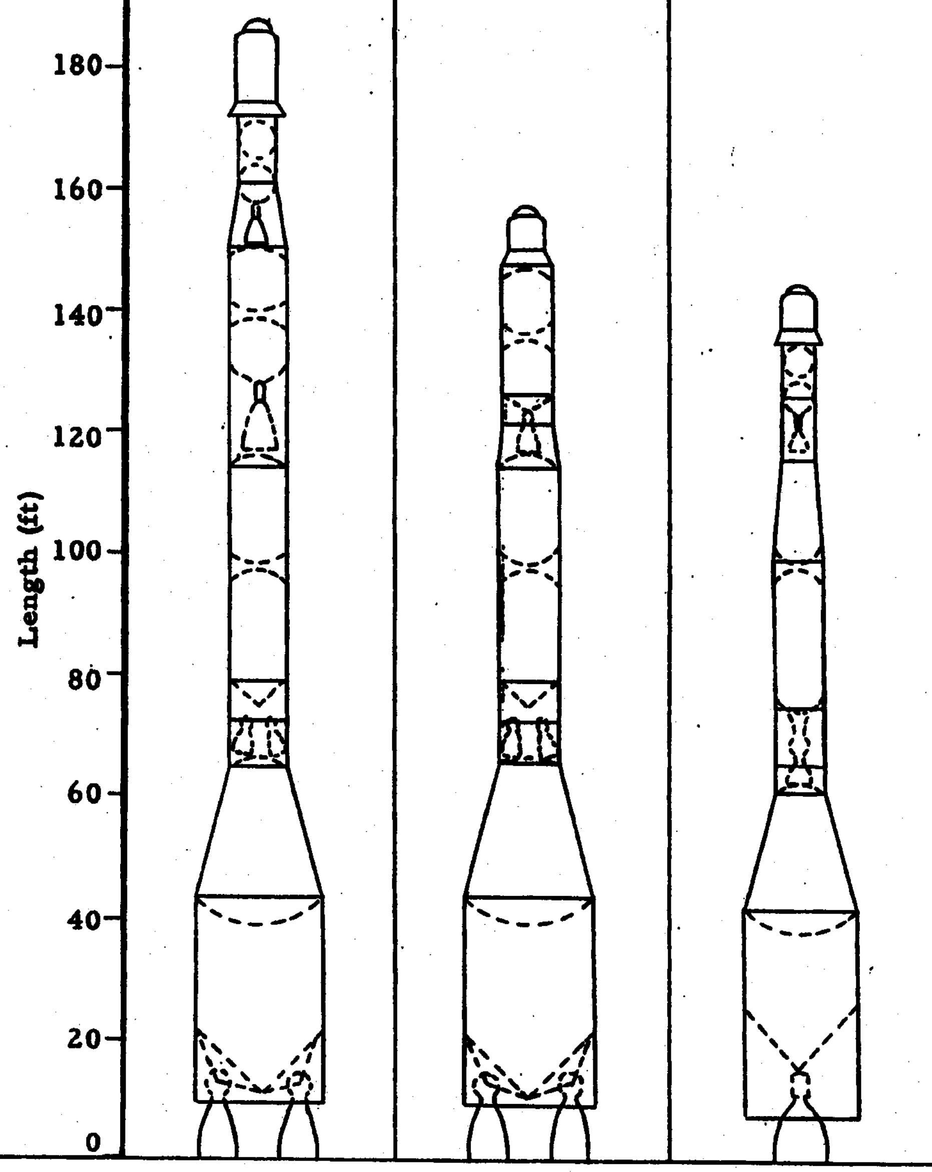


Figure 3-4. Boosted Vehicles.

4.0 PERFORMANCE CALCULATION METHODS

4.1 Methods and Assumptions

In practically all cases, machine computations were available to determine payload capabilities for the various vehicle combinations studied. In the few remaining cases, preliminary design methods were used.

Trajectory results in reference 5 were of great help in carrying out the computations. The report contains accurate and detailed calculations of the ICBM and IRBM missile trajectories for a wide range of payloads and burnout angles. Many of the performance results of the vehicles reported herein were obtained simply by adding final power stages to the standard boosters.

An appreciable amount of work has been done to optimize some of the new stages used in various vehicle combinations. Efforts were made to size the stages for general utility, and to select thrust-to-weight ratios that would yield near maximum payloads. A fairly detailed preliminary design was carried out on the Fluorine final power stage, for example, as shown in reference 4.

4.1.1 Thor, Atlas, and Titan Performance Computations

Reference 5 presents burnout conditions (velocity, altitude and angle) for the Thor, Atlas, and Titan missiles for a wide range of payloads and kick angles. The trajectories are carried out for the rotating earth case, and take full account of drag and gravity losses.

The combinations using the above stages as boosters utilized the data in reference 5 at the appropriate payload weight. A burnout angle of 80 deg was assumed in all cases.

Final stage(s) performance was computed by ignoring drag and gravity losses, which are negligible for near horizontal flight at high altitudes. The velocity increment was computed from the expression for ideal velocity.

4.1.2 Boosted Vehicle Performance

Trajectories for several combinations involving the large booster stages were computed. Since no previous computations had been performed on these boosters, complete machine solutions were carried out in order to properly account for drag and gravity losses in the sensible atmosphere. Again final stage(s) performance was computed from the ideal velocity expression since the final stage(s) were fired at high altitudes in near horizontal flight.

4.2 Mission Trajectories

4.2.1 Satellite Missions

The optimum manner in which to place a payload in orbit is to burn all stages consecutively, and then let the payload drift up to apogee altitude. Since the orbit will be slightly elliptic, another short burning period is required at apogee to alter the orbit from elliptical to circular.

An alternate scheme (and simpler from an operational standpoint) is to burn all but the final stage consecutively. The final power stage then coasts along with the payload to apogee altitude. Final stage burning is initiated so that burnout occurs precisely at the desired apogee altitude, and at circular velocity. Actually, if gravity losses are ignored, either of the above schemes will give near maximum velocities.

As mentioned previously, gravity effects were ignored during final stage burning for the cases reported herein. The equivalent velocities required to achieve orbital paths at various altitudes are listed in a following section.

4.2.2 Lunar Missions

For hard impact on the moon, the vehicle must be accelerated to escape velocity during initial burning as it leaves the earth (theoretical escape velocity from the surface of the earth is 36,800 ft/sec). The vehicle will then follow a near parabolic escape path until the null point is reached (that point between the earth and moon where the respective gravitational forces are equal). The moon's gravitational pull then causes the vehicle to impact on the moon with about 8000 ft/sec, assuming a null velocity of zero. No retrorocket or navigational corrections are assumed necessary. The vehicle must be steered properly during initial burning periods to ensure reasonable cep's on the moon.

For a soft landing on the moon, the vehicle must be decelerated by the application of retrorocket thrust. The velocity increment imparted by the retrorocket must be equal to or greater than 7900 ft/sec (which occurs when the null velocity is zero). Specific impulse values range from 250 to 300 sec for solid propellant and from 250 to 400 sec for liquid propellants. Structural factors from 0.06 to 0.10 may be attained.

It is noted in Figure 4-1, which summarizes the lunar landing case, that the weight of retrorocket is equal to or greater than the payload for the soft landing case. For specific impulse of 300 sec and $\Delta V = 9000$ ft/sec, $P = 3$; hence, the retrorocket weight will be 2 times that of the payload. For this reason, the soft landing mission is a difficult one for a vehicle of limited payload.

In establishing a lunar satellite, the lunar vehicle is aimed at a point ahead of the moon's position at time of arrival, as shown in Figure 4-1, and a retrorocket is fired to retard the velocity to values required for circular or elliptical orbits.

The requirements for the retrorocket are less severe for this case than for the soft landing. Velocity increments of a few thousand feet per second normally will be required. Figure 4-1 shows the ΔV requirements to produce a moon satellite, with various approach velocities, V_a.

For example, to establish a satellite orbit at $h = 2R_m = 2000$ miles above the surface of moon, with an approach velocity of 6000 ft/sec, a retrorocket weight of about 0.47 of the final payload W_1 is required. An additional requirement is attitude control with respect to the moon to ensure that the retrorocket thrust is applied in the proper direction.

In the case of the circumlunar flight with return to earth, the vehicle is aimed to effect a "near miss" ahead of the lunar position so that the trajectory will be sharply turned by moon's gravitational field, causing it to return to the vicinity of earth, as shown in Figure 4-1 . This mission offers great attraction for exploratory flights, including picture gathering, magnetic field measurements, and other scientific data of conditions in moon's vicinity.

The velocity requirements for the circumlunar mission are approximately the same as for other missions. Because of the need to clear the moon at a correct altitude to receive a prescribed turning effect, the timing error is quite important. Hence, it is desirable to have a slight excess velocity at the null point, in order to minimize the time errors. During the transit, it may be desirable to check the velocity slightly by retrorocket. For this reason, attitude control would also be desirable, to cause the vehicle

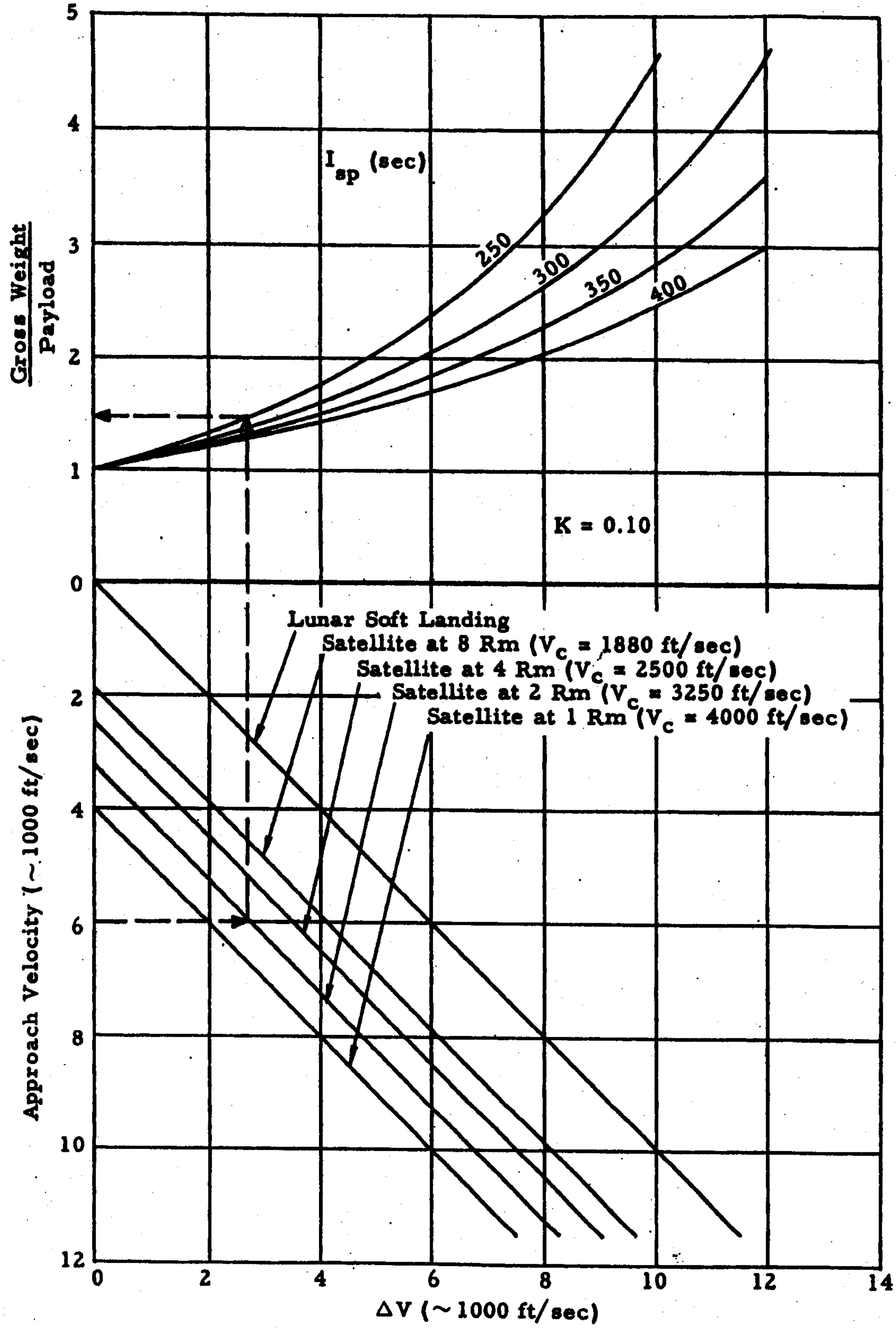


Figure 4-1. Determination for Payload for Lunar Landing or Lunar Satellite.

to have a proper aspect to the moon at all times. After the turn, the accuracy problem of returning to earth is rather severe, and some adjustment of velocity may be desirable to improve the recovery. Terminal guidance on earth would be very beneficial to increase the chances of recovery. The problem of re-entry into the earth's atmosphere with aerodynamic deceleration offers an additional requirement, to provide sufficient heat shielding to withstand the high temperatures.

4.2.3 Interplanetary Missions

The interplanetary missions considered herein are those involving a reconnoiter of Venus or Mars, with return to the earth. These two planets lie close to the earth, and thus are of immediate interest to the space flight experimental program.

The mission may be stated as follows. A chemically propelled missile is required to leave earth, travel to Mars or Venus and return to earth. It is assumed that guidance equipment of infinite accuracy is available, so that no mid-flight corrective accelerations will be required. Aerodynamic braking may be used in the atmospheres of both earth and the target planet.

It can be shown that the minimum energy transfer involves a semiellipse, the ellipse being tangent to the earth's orbital path at one apogee, and tangent to the orbital path of the target planet at the other apogee. This is commonly known as a Hohmann transfer.

Some remarks should be made concerning the attainment of transfer speeds. As mentioned above, the vehicle would leave the earth along an escape hyperbola and enter the transfer ellipse. Escape speed from the earth is about 36,800 ft/sec. A drag-free vehicle accelerated to this speed at the surface of the earth would be able to recede to infinity, but would arrive there with zero speed. This follows from the definition of the escape velocity, which involves a statement of the conservation of energy:

$$\frac{1}{2} m v_{\infty}^2 = \frac{1}{2} m v_o^2 - m g_o r_o$$

where the subscript o refers to conditions on the surface of the planet and ∞ refers to conditions at an infinite distance away from the planet. By setting v_{∞} equal to zero, the escape velocity is found to be $v_o^2/\sqrt{2 g_o r_o}$.

Actually the vehicle must be accelerated beyond the escape speed in order to have the residual velocity required to fly the interplanetary transfer. This additional velocity increment can either be added during the initial burning period, or at some time later, after escape from the earth has been achieved. A significant advantage is gained by adding the incremental velocity during the initial burning period, as can be seen from the energy equation above. For example, to enter upon a voyage to Venus would require a characteristic escape velocity of 36,800 ft/sec, plus an incremental velocity of 8,200 ft/sec to enter upon a Hohmann transfer to Venus (a total of 45,000 ft/sec). Now, if the total velocity is added during the initial burning period the energy equation gives

$$\frac{1}{2} m (8{,}200)^2 = \frac{1}{2} m (v_o)^2 - m g_o r_o$$

$$= \frac{1}{2} m (v_o)^2 - \frac{1}{2} m (36{,}800)^2$$

$$v_o = 37{,}700 \text{ ft/sec}$$

Hence, small velocity increments added during initial burning periods produce large residual velocities at infinity. This technique has been used in computing the required equivalent velocities for flights to Venus and Mars.

4.3 Equivalent Velocity Requirements

Several possible space flight missions were singled out for special attention. Typical of the missions thus considered are:

- Ballistic Missile - 10,000-naut-mi range
- Satellite in Circular Orbit
  - 300-naut-mi altitude
  - 1,000-naut-mi altitude
  - 20,000-naut-mi altitude
- Hard Lunar Impact
- Soft Lunar Landing
- Soft Lunar Landing and Return to Earth
- Landing on Mars or Venus
- Landing on Mars with Reduced Transit Time.

The flight trajectories assumed for each of these missions will be described in the following section.

During the climb up through the atmosphere, appreciable amounts of the rocket thrust of the combination are used in overcoming aerodynamic drag and the component of the gravitational attraction which acts parallel to the thrust vector of the rocket engine in the vehicle. However, it will be found that by the time the vehicle has reached an altitude of 200 miles it is well above the sensible atmosphere and moving almost horizontally. Consequently, the need for additional allowances for aerodynamic drag and gravitational loss is eliminated; the fundamental rocket equation

$$\Delta V = Ig_o \ln\left(\frac{W_o}{W_b}\right)$$

can be used in computing any additional velocity increases which may be needed during the early part of the mission. In order to take advantage of this analytical simplification, the numerical value used for the speed requirement in this study was selected to be that required by a vehicle starting at a point 200 naut mi above the surface of the earth.

The performance of the various combinations in approaching the starting point (at 200-naut-mi altitude) was computed on a digital computer. In all cases (except ballistic missile missions) an eastward launch from Cape Canaveral was assumed. Beyond the "starting point" the fundamental rocket equation was used.

4.3.1 Ballistic Missile (10,000-Naut-Mi Range)

For this case the assumption of an eastward launch from Cape Canaveral, as used in the other cases, was deemed unduly restricted, and hence unwarranted; a nonrotating earth was assumed. Under these conditions, the initial speed required (at 200-naut-mi altitude) for a 10,000-naut-mi range is 25,800 ft/sec, assuming a minimum energy trajectory is used.

The curve of initial speed available versus range is quite flat. Essentially the same initial speed will yield a 10,800-naut-mi range, i.e., halfway around the world.

4.3.2 Satellite Orbits

The orbital speed of a terrestrial satellite moving in a circular orbit at an altitude of 200 naut mi is 25,214 ft/sec (measured in a nonrotating coordinate system). In transferring to a circular satellite orbit at any other altitude, a Hohmann ellipse should be followed if the total equivalent speed required is to be minimized. Numerical values are given in Table 4-1.

4.3.3 Lunar Impact

With perfect guidance and starting from the assumed altitude of 200 naut mi, a total speed of only about 35,000 ft/sec is required for ballistic impact on the lunar surface. However, in order to ease the guidance requirements, an additional allowance of roughly 1000 ft/sec was included, making a total requirement of 36,000 ft/sec. This value was used in computing payload capability.

4.3.4 Soft Landing on Moon

The velocity of escape from the moon is about 7800 ft/sec. If a vehicle approaches the moon with an initial speed of 2000 ft/sec, the speed of impact will be increased to about 8000 ft/sec. In order to obtain a soft landing this approach speed must be checked by rocket deceleration. The total equivalent velocity required for a soft landing is thus the equivalent speed required for impact plus the 8000 ft/sec required for deceleration, giving a total of 44,000 ft/sec.

4.3.5 Soft Lunar Landing and Return to Earth

To return to earth starting from the lunar surface would require an initial acceleration equal to the lunar escape velocity, i.e., 8000 ft/sec. If, as in the soft lunar landing case, an additional allowance of 1000 ft/sec is allowed for possible mid-course guidance and gravitational losses when accelerating away from the lunar surface, the total budget will be 44,000 plus 9000 = 53,000 ft/sec.

If unchecked, the returning terminal vehicle would approach and strike earth's surface with prohibitive speed. However, with proper vehicle design and flight trajectory, the atmosphere can be used as a brake. The aerodynamic

Table 4-1. Total Equivalent Speed Required for Satellite Missions.

Orbital Altitude (naut mi)	Speed Increment to Enter Hohmann Ellipse (from 200-naut-mi orbit) (ft/sec)	Speed Increment to Leave Hohmann Ellipse (at orbital altitude) (ft/sec)	Total Speed to Transfer from 200-Naut-Mi Orbit (ft/sec)	Total Mission Requirement (ft/sec)
300	176	174	350	25,600
1,000	1,240	1,180	2,420	27,600
20,000	8,060	4,850	12,910	38,100
∞	10,430	0	10,430	35,640

forces which act on the terminal vehicle during its rapid flight through the atmosphere can be used to decelerate the vehicle and make possible a non-destructive landing on the earth's surface.

4.3.6 Landing on Mars or Venus with Reduced Transit Time (Aerodynamic Deceleration)

If properly directed and timed, an initial vehicle speed of about 38,000 ft/sec at the reference 200-naut-mi altitude will result in an interorbital transfer ellipse (no longer a Hohmann ellipse) to Mars or Venus having an elapsed travel time of less than 4 months. The methods used to compute the properties of the ellipse were developed in reference 6.

5.0 VEHICLE PERFORMANCE

The following pages contain curves of gross payload versus equivalent velocity for the stage combinations studied in this report. Drag and gravity losses incurred in reaching a 200-naut-mi altitude circular orbit have been included in the performance curves. The orbital velocity for this case is 25,300 ft/sec so that all equivalent velocities shown on the curves fully reflect these losses. The equivalent velocity requirements for some typical space flights are indicated on the performance sheets for reference purposes.

Stage No.	1	2
Stage Designation	Thor (175-107)	Vanguard II
Stage Weight (lb)	106,600	4232
Propellants	Lox-RP	WFNA UDMH
Thrust (lb)	175,000	7700
I_{sp} (lb-sec/lb)	290	278

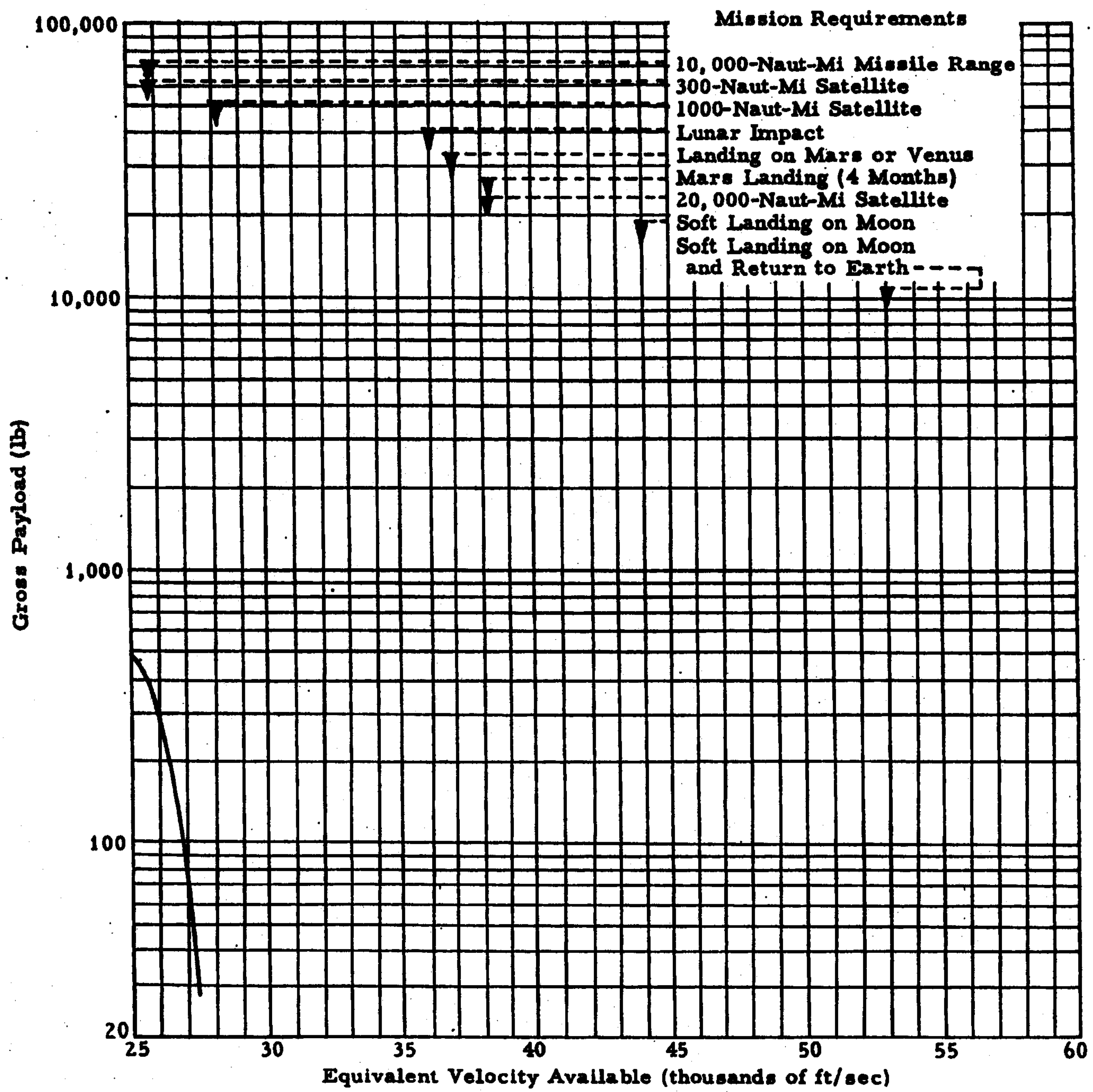


Figure 5-1. Gross Payload Versus Equivalent Velocity, Combination Number 1.

Stage No.	1	2
Stage Designation	Thor (175-107)	Fluorine (12-12)
Stage Weight (lb)	106,600	12,000
Propellants	Lox-RP	Fluorine-Hydrazine
Thrust (lb)	175,000	12,000
I_{sp} (lb-sec/lb)	290	364

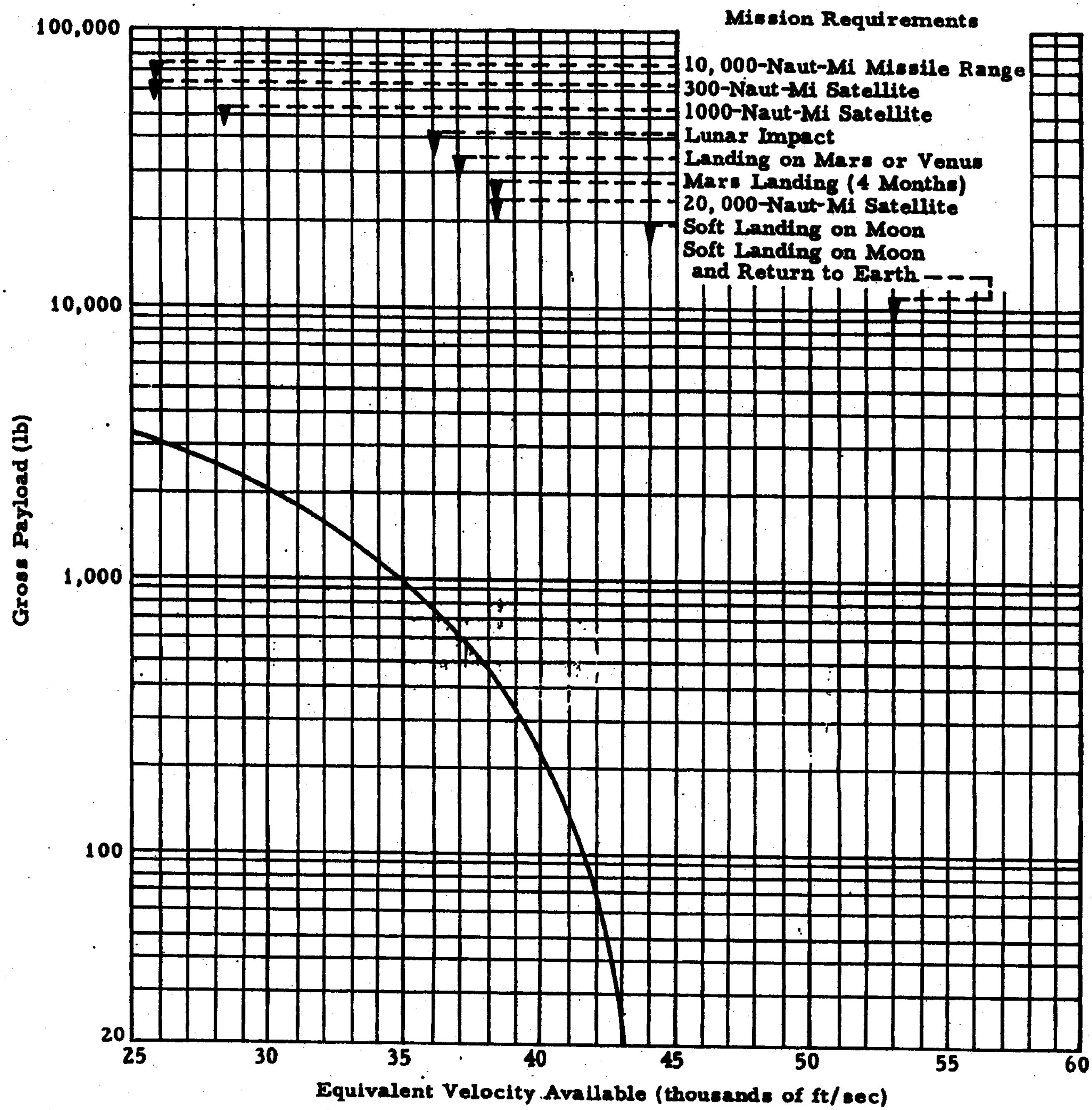


Figure 5-2. Gross Payload Versus Equivalent Velocity, Combination Number 2.

Stage No.	1	2
Stage Designation	Thor (175-107)	Vanguard III
Stage Weight (lb)	106,600	435
Propellants	Lox-RP	Solid
Thrust (lb)	175,000	2350
I_{sp} (lb-sec/lb)	290	240

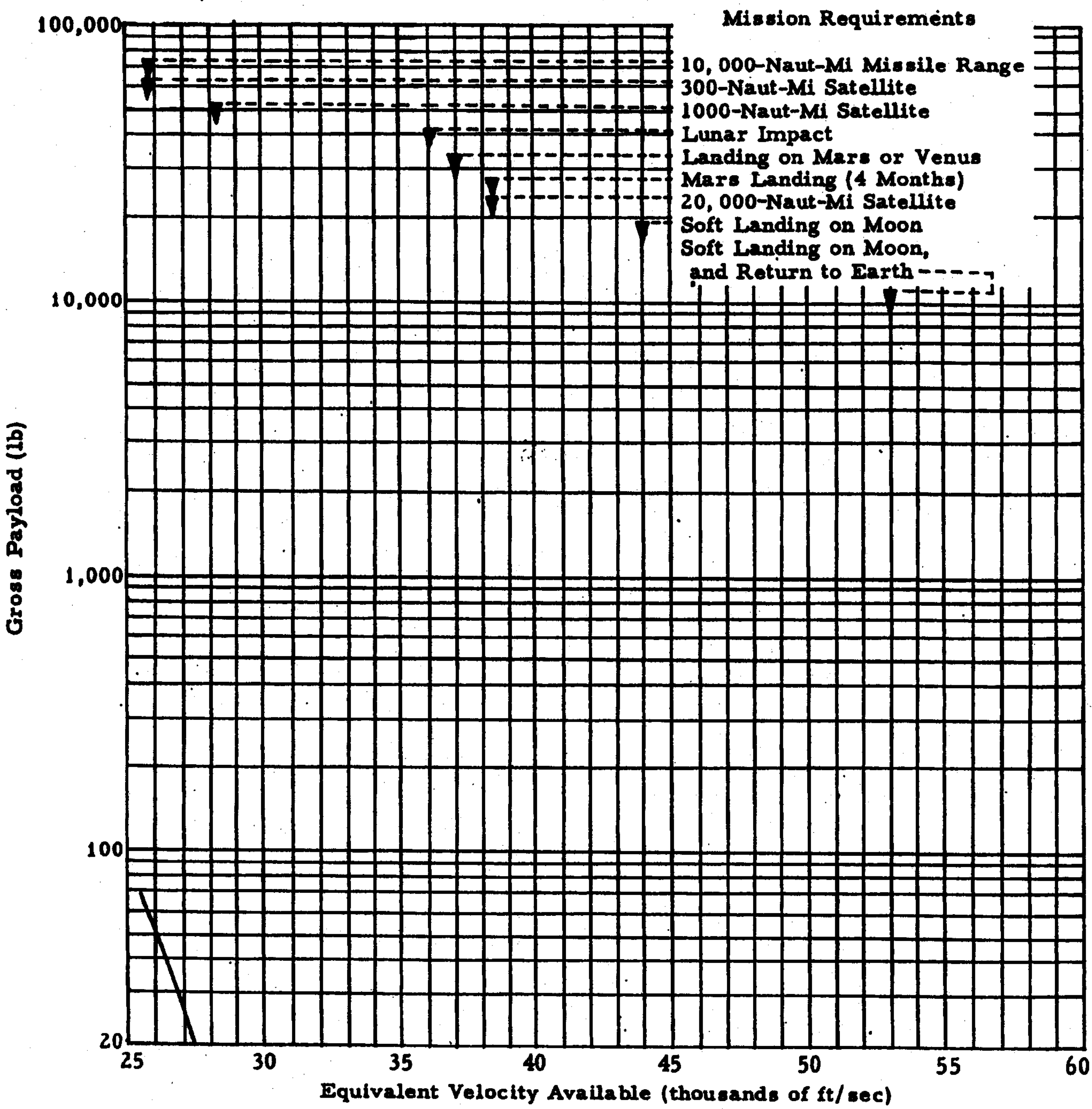


Figure 5-3. Gross Payload Versus Equivalent Velocity, Combination Number 3.

Stage No.	1	2
Stage Designation	Thor (175-107)	4 Vanguard III's
Stage Weight (lb)	106,600	1800
Propellants	Lox-RP	Solid
Thrust (lb)	175,000	9400
I_{sp} (lb-sec/lb)	290	240

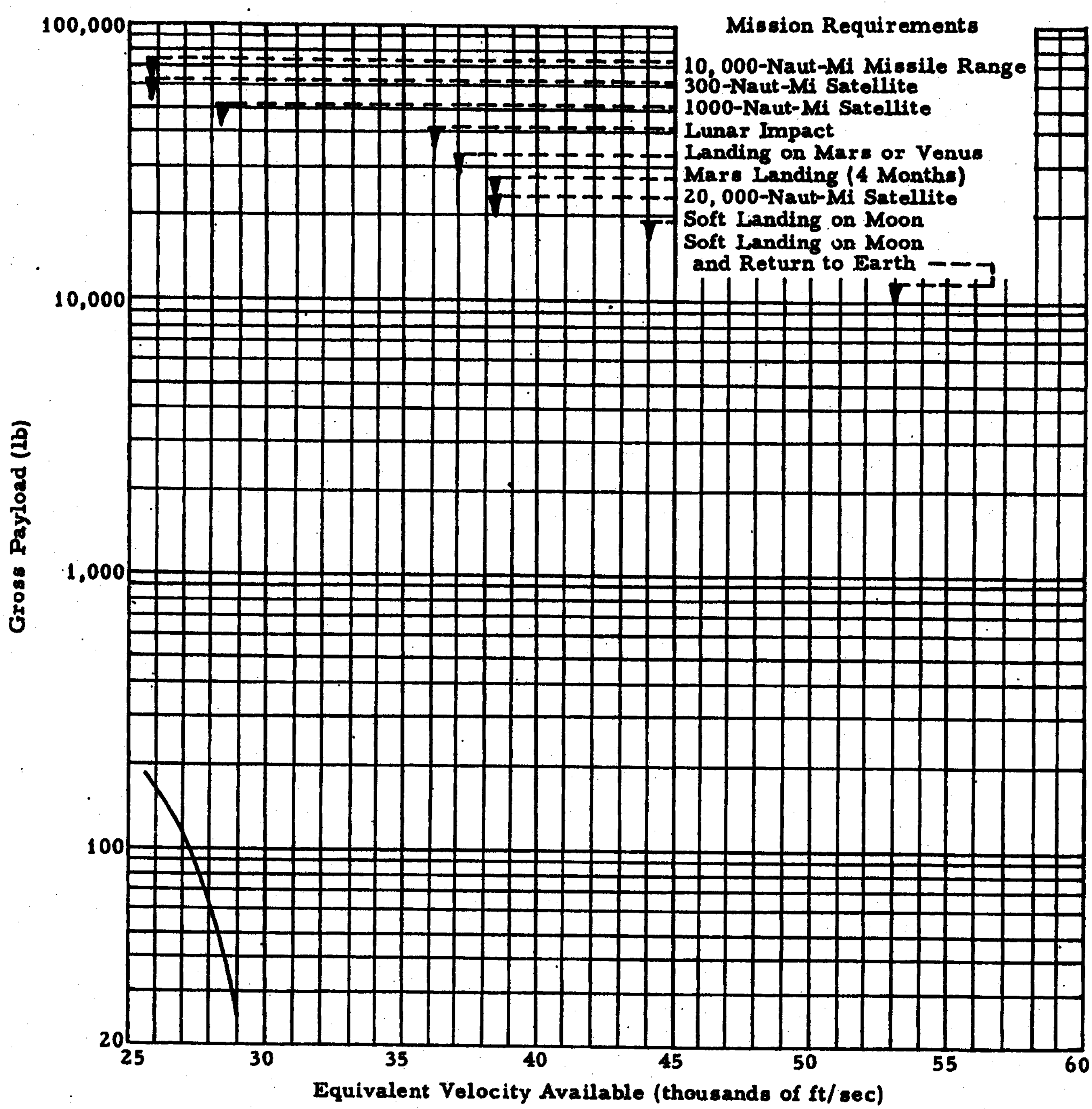


Figure 5-4. Gross Payload Versus Equivalent Velocity, Combination Number 4.

Stage No.	1	2
Stage Designation	Thor (175-107)	Sergeant
Stage Weight (lb)	106, 600	8185
Propellants	Lox-RP	Solid
Thrust (lb)	175, 000	50, 000
I_{sp} (lb-sec/lb)	290	191

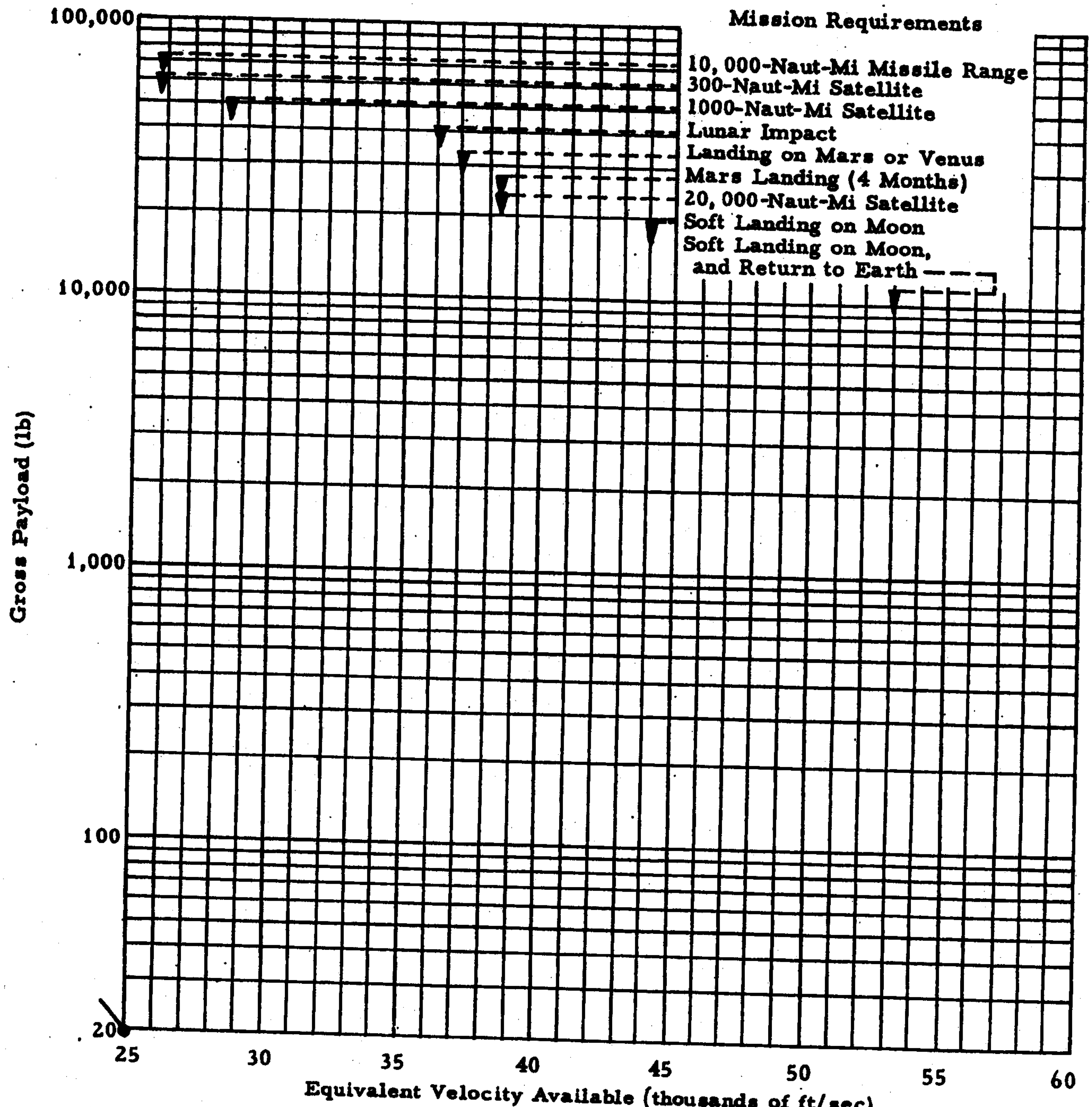


Figure 5-5. Gross Payload Versus Equivalent Velocity, Combination Number 5.

Stage No.	1	2	3
Stage Designation	Thor (175-107)	Vanguard II	Vanguard III
Stage Weight (lb)	106, 600	4232	435
Propellants	Lox-RP	WFNA UDMH	Solid
Thrust (lb)	175, 000	7700	2350
I_{sp} (lb-sec/lb)	290	278	240

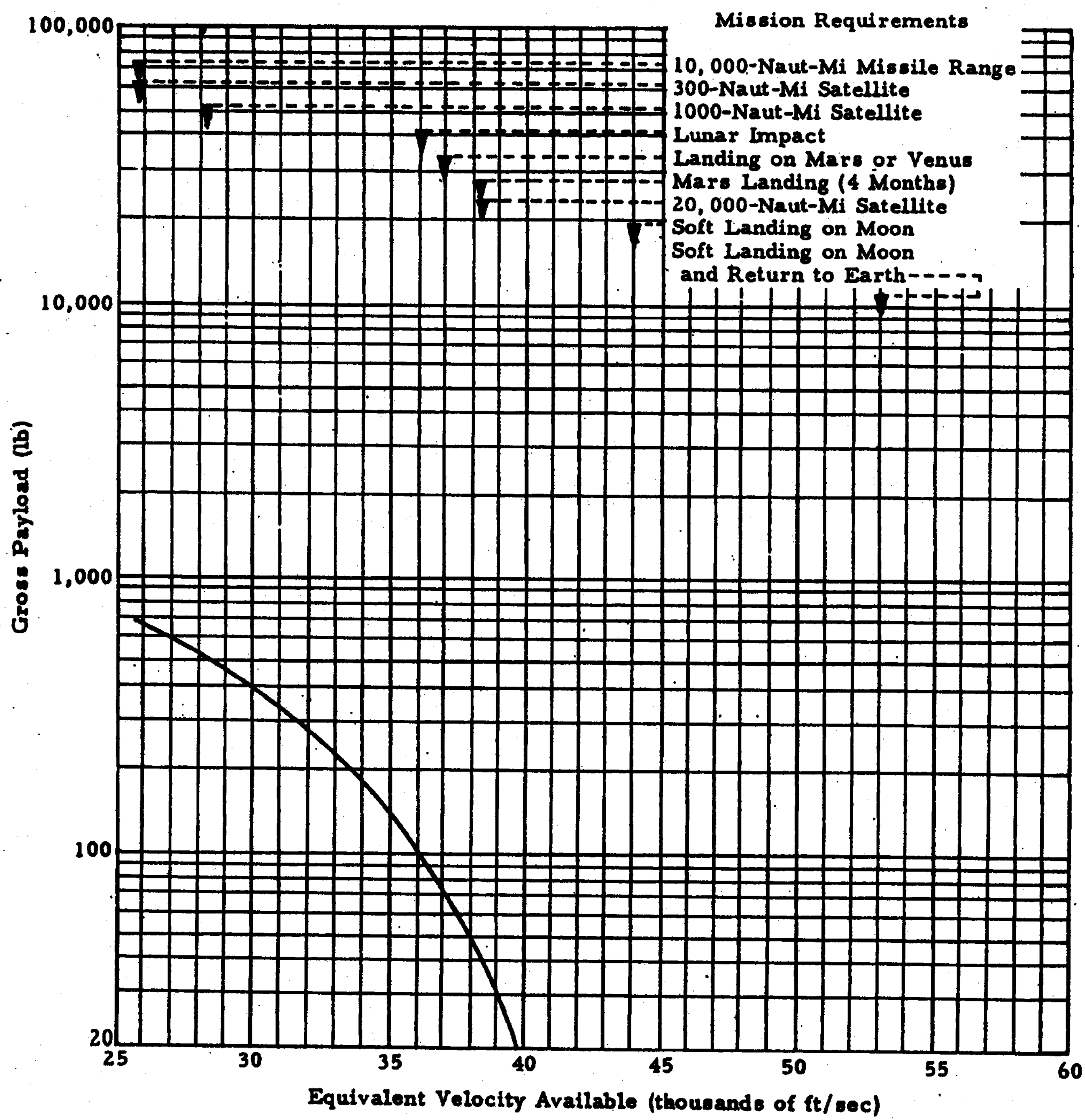


Figure 5-6. Gross Payload Versus Equivalent Velocity, Combination Number 6.

Stage No.	1	2	3
Stage Designation	Thor (175-107)	4 Vanguard III's	Vanguard III
Stage Weight (lb)	106,600	1800	435
Propellants	Lox-RP	Solid	Solid
Thrust (lb)	175,000	9400	2350
I_{sp} (lb-sec/lb)	290	240	240

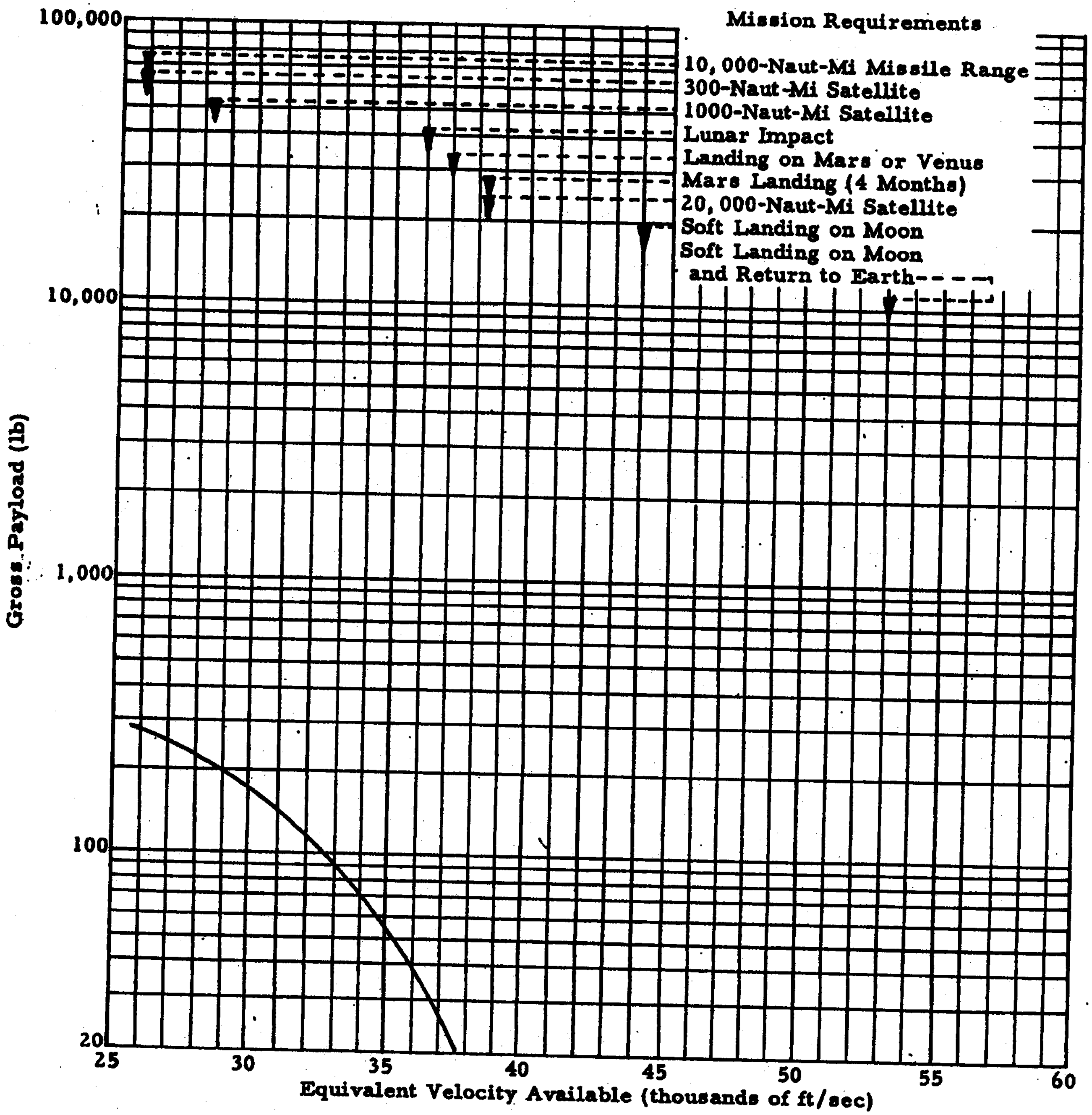


Figure 5-7. Gross Payload Versus Equivalent Velocity, Combination Number 7.

[redacted]

Stage No.	1	2	3
Stage Designation	Thor (175-107)	XM-34	Vanguard III
Stage Weight (lb)	106,600	8635	435
Propellants	Lox-RP	Solid	Solid
Thrust (lb)	175,000	62,000	2350
I_{sp} (lb-sec/lb)	290	226	240

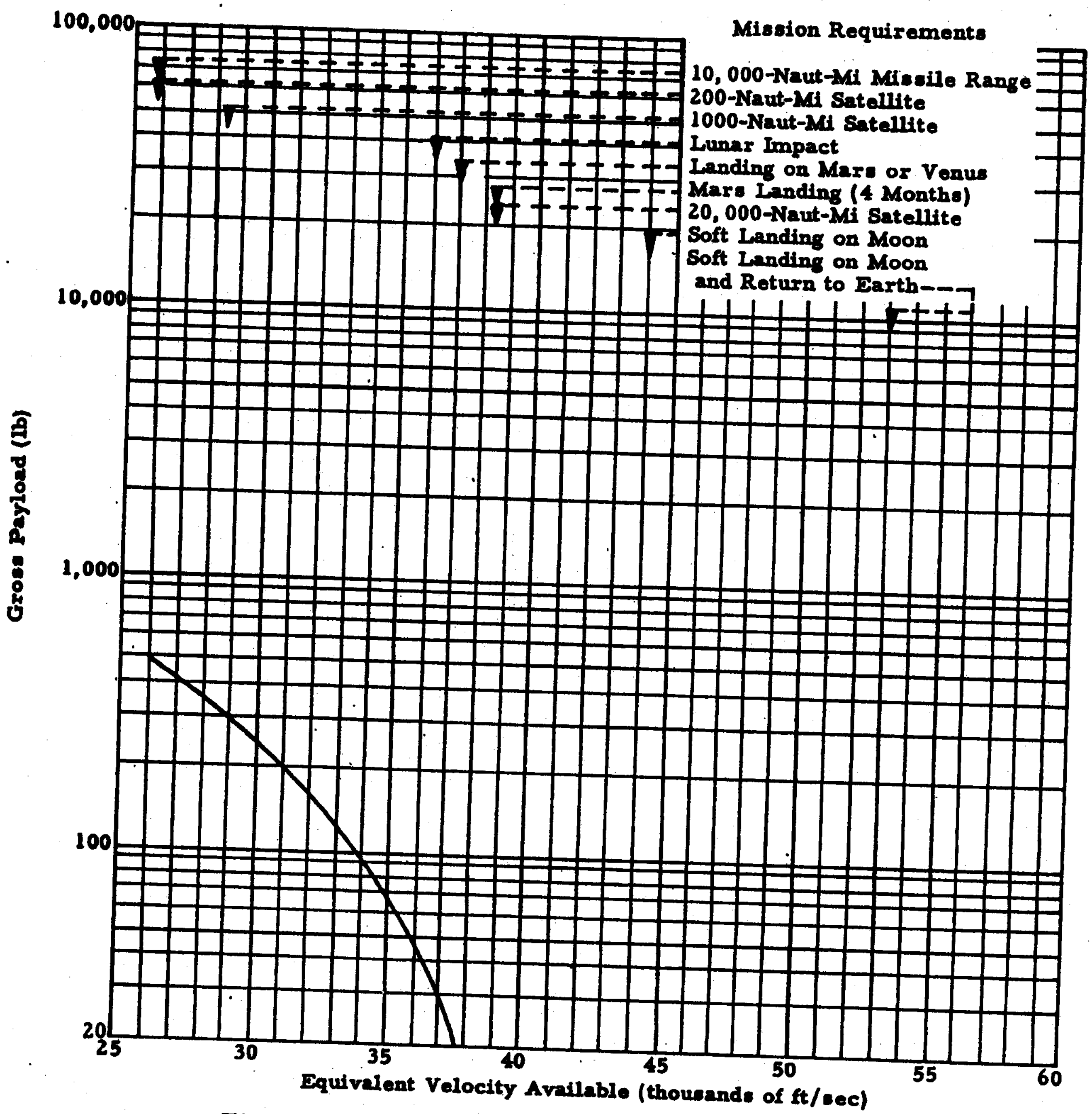


Figure 5-8. Gross Payload Versus Equivalent Velocity, Combination Number 8.

Stage No.	1	2
Stage Designation	Atlas (360-265)	WS 117L (Mod)
Stage Weight (lb)	265,000	6000
Propellants	Lox-RP	RFNA-RP
Thrust (lb)	360,000	15,000
I_{sp} (lb-sec/lb)	290	270

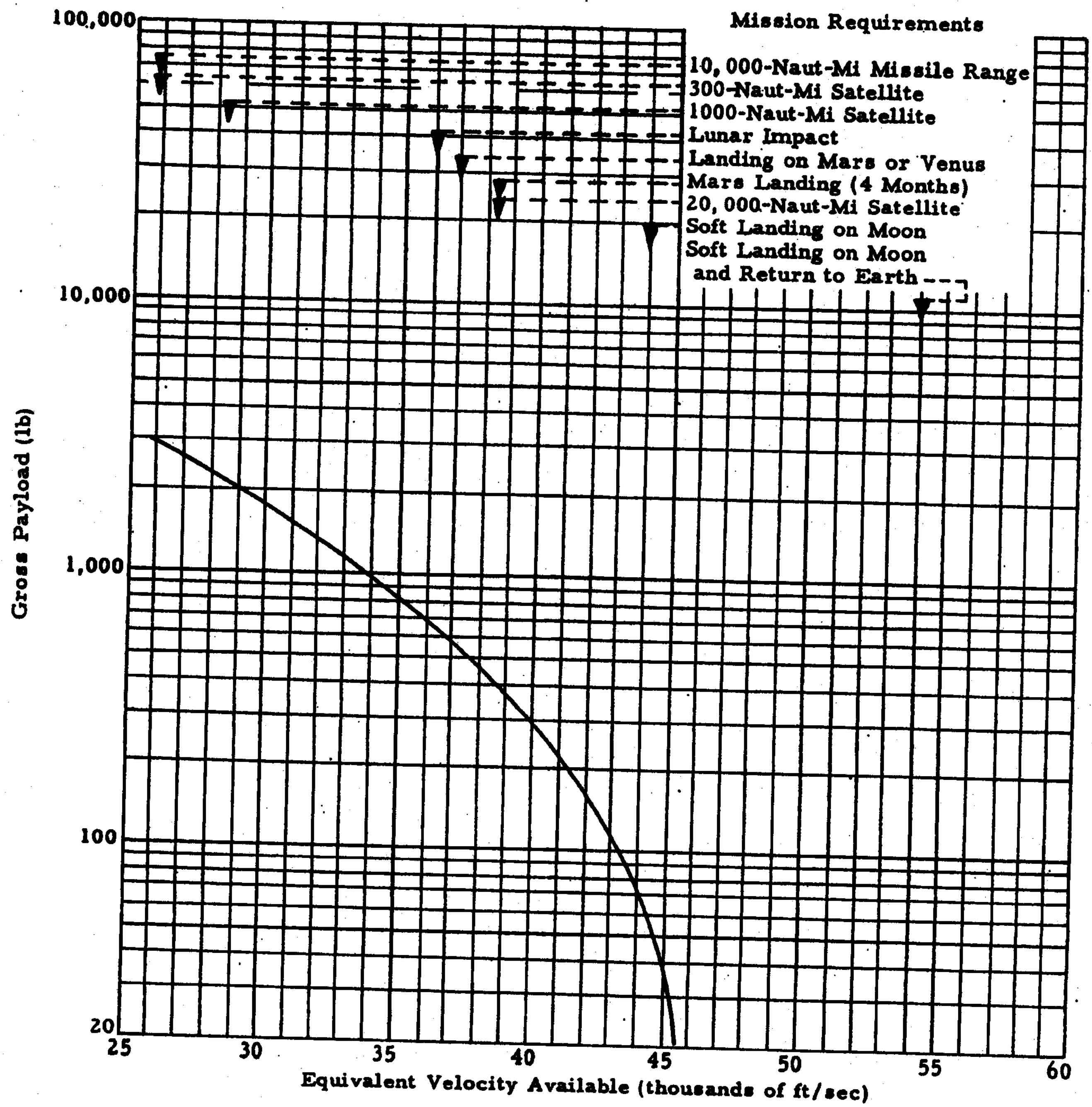


Figure 5-9. Gross Payload Versus Equivalent Velocity, Combination Number 9.

Stage No.		
Stage Designation	Atlas (360-265)	Vanguard II (Mod)
Stage Weight (lb)	265,000	6300
Propellants	Lox-RP	WFNA UDMH
Thrust (lb)	360,000	7700
I_{sp} (lb-sec/lb)	290	278

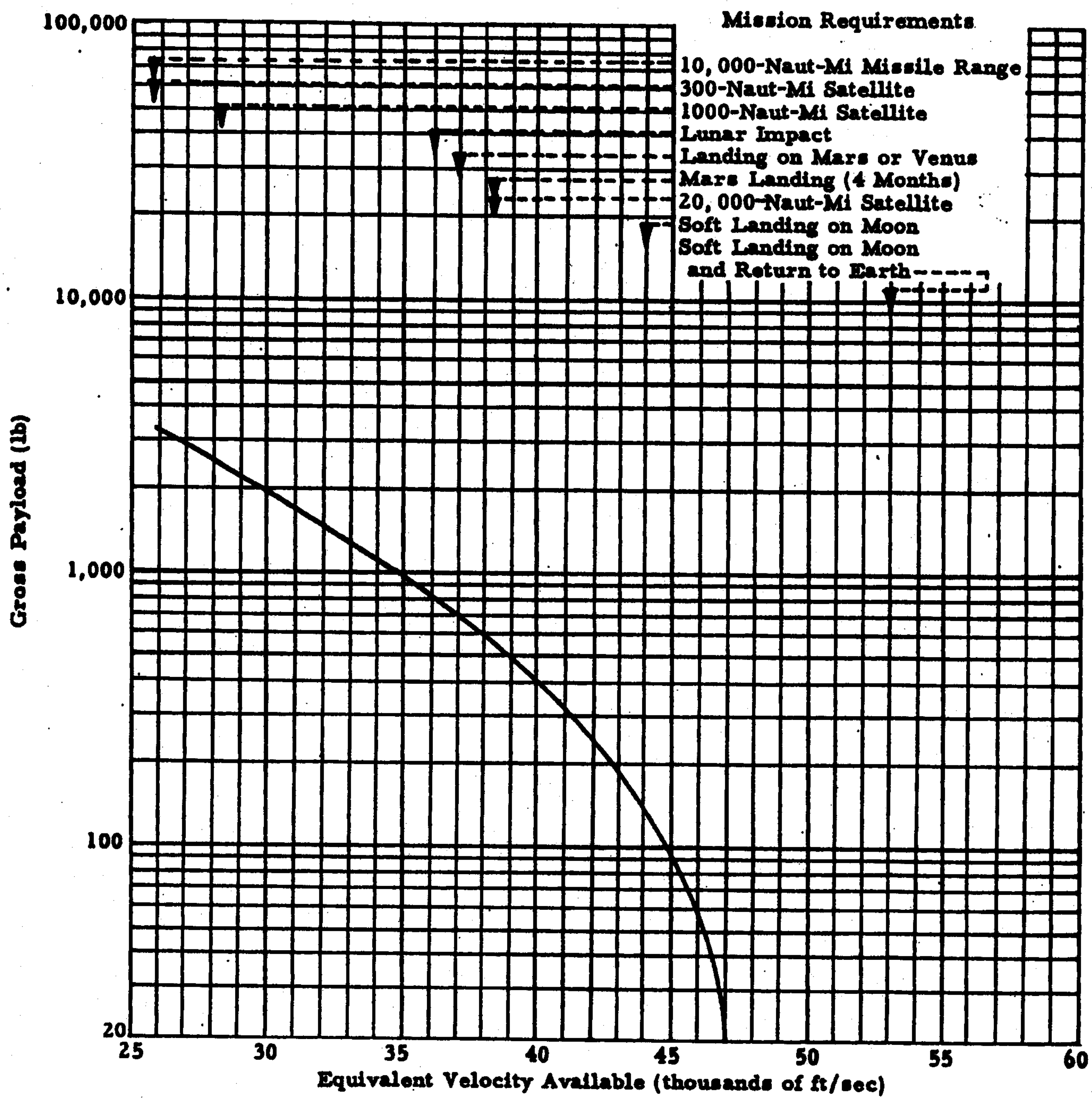


Figure 5-10. Gross Payload Versus Equivalent Velocity, Combination Number 10.

Stage No.	1	2
Stage Designation	Atlas (415-262)	Fluorine (12-12)
Stage Weight	262,000	12,000
Propellant	Lox-RP	Fluorine-Hydrazine
Thrust (lb)	415,000	12,000
I_{sp} (lb-sec/lb)	290	364

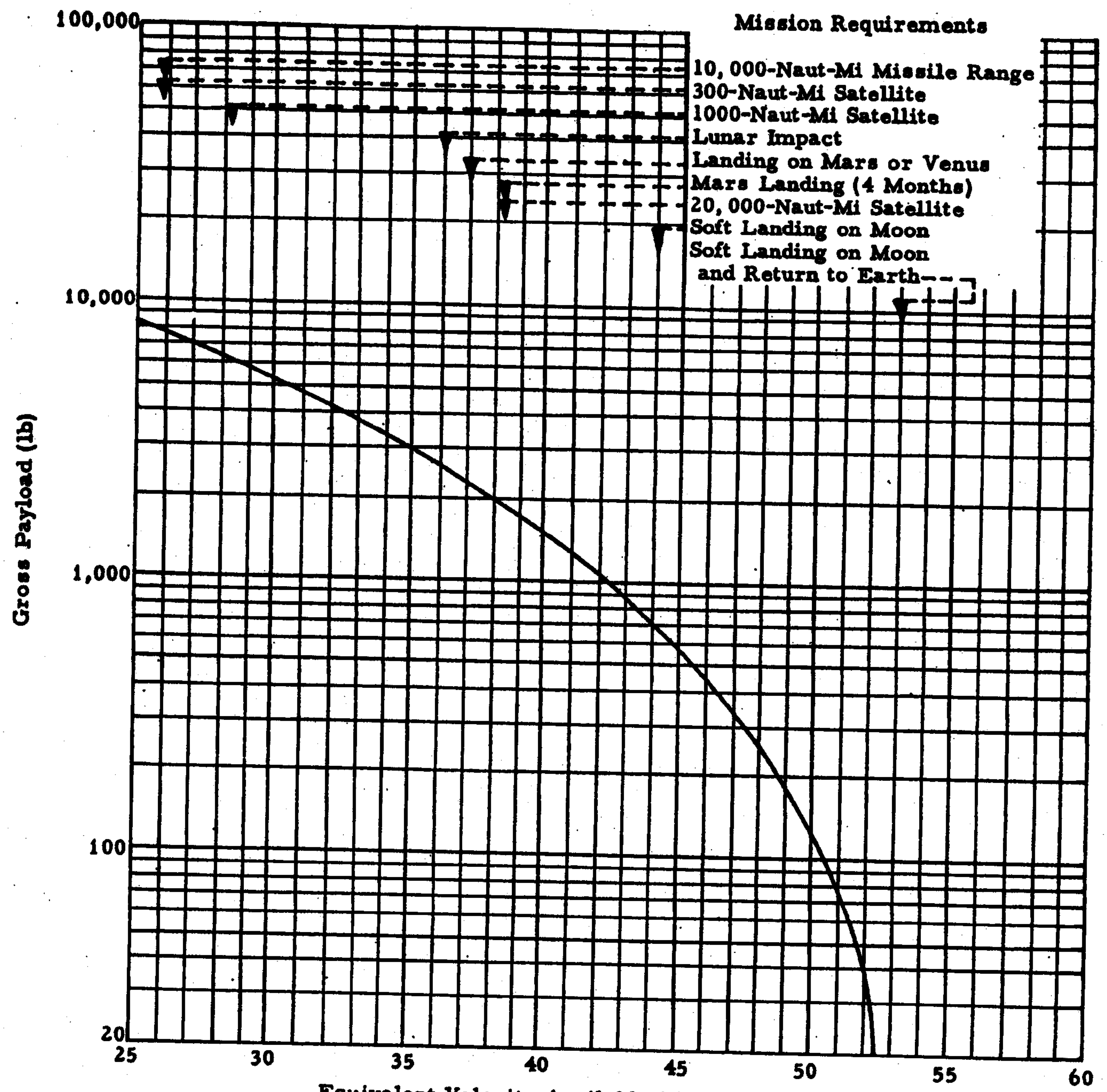


Figure 5-11. Gross Payload Versus Equivalent Velocity, Combination 11.

Stage No.	1	2	3
Stage Designation	Titan I (301-170)	Titan II (81-45)	Fluorine (12-12)
Stage Weight (lb)	170,000	44,500	12,000
Propellants	Lox-RP	Lox-RP	Fluorine Hydrazine
Thrust (lb)	301,000	80,800	12,000
I_{sp} (lb-sec/lb)	290	310	364

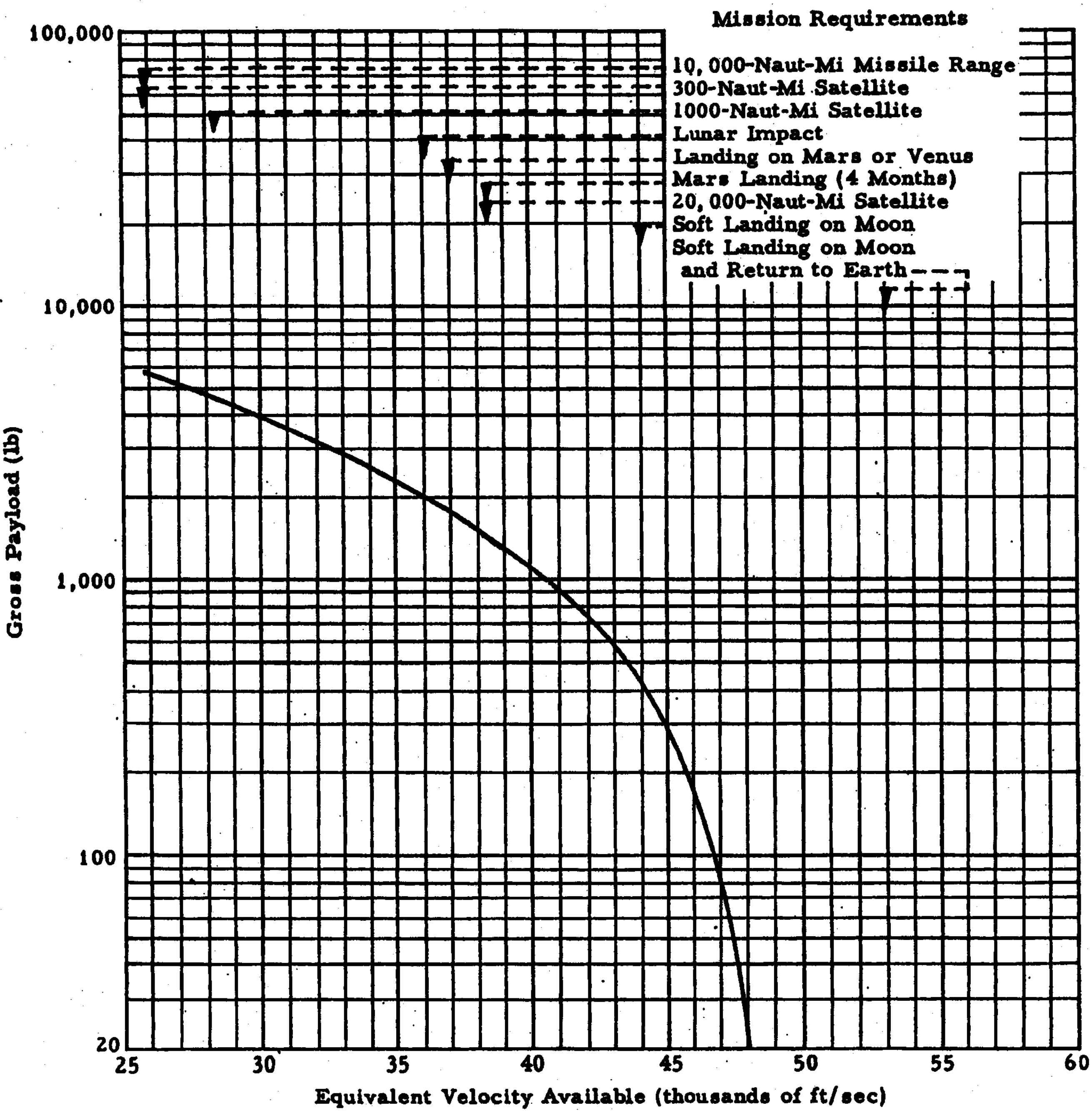


Figure 5-12. Gross Payload Versus Equivalent Velocity, Combination Number 12.

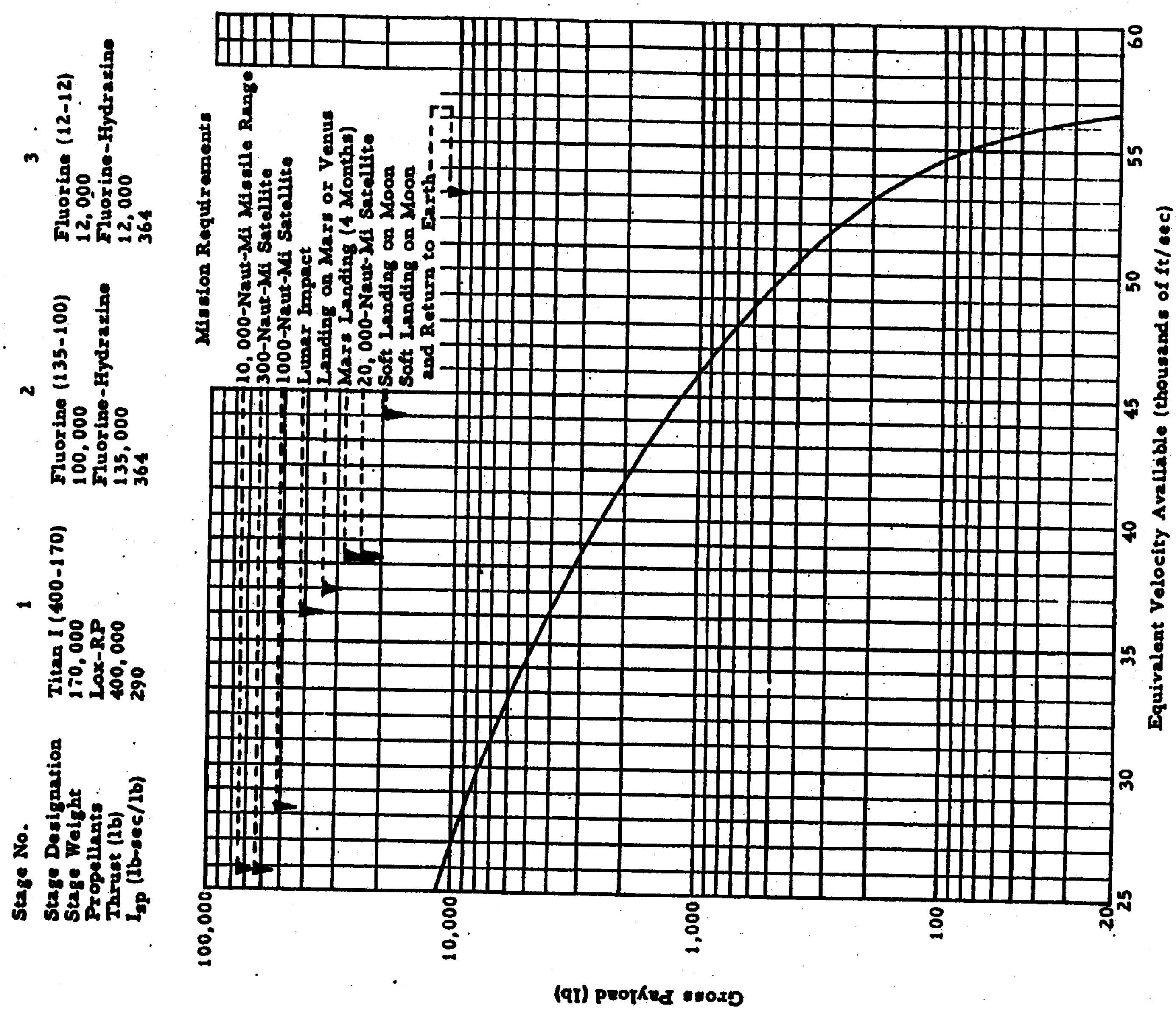


Figure 5-13. Gross Payload Versus Equivalent Velocity, Combination Number 13.

Stage No.	1	2	3	4
Stage Designation	Booster (2000-1275)	Titan I (400-170)	Fluorine (135-100)	Fluorine (12-30)
Stage Weight	1,275,000	170,000	100,000	30,000
Propellants	Lox-RP	Lox-RP	Fluorine-Hydrazine	Fluorine-Hydrazine
Thrust (lb)	2,000,000	400,000	135,000	12,000
I_{sp} (lb-sec/lb)	290	290	364	364

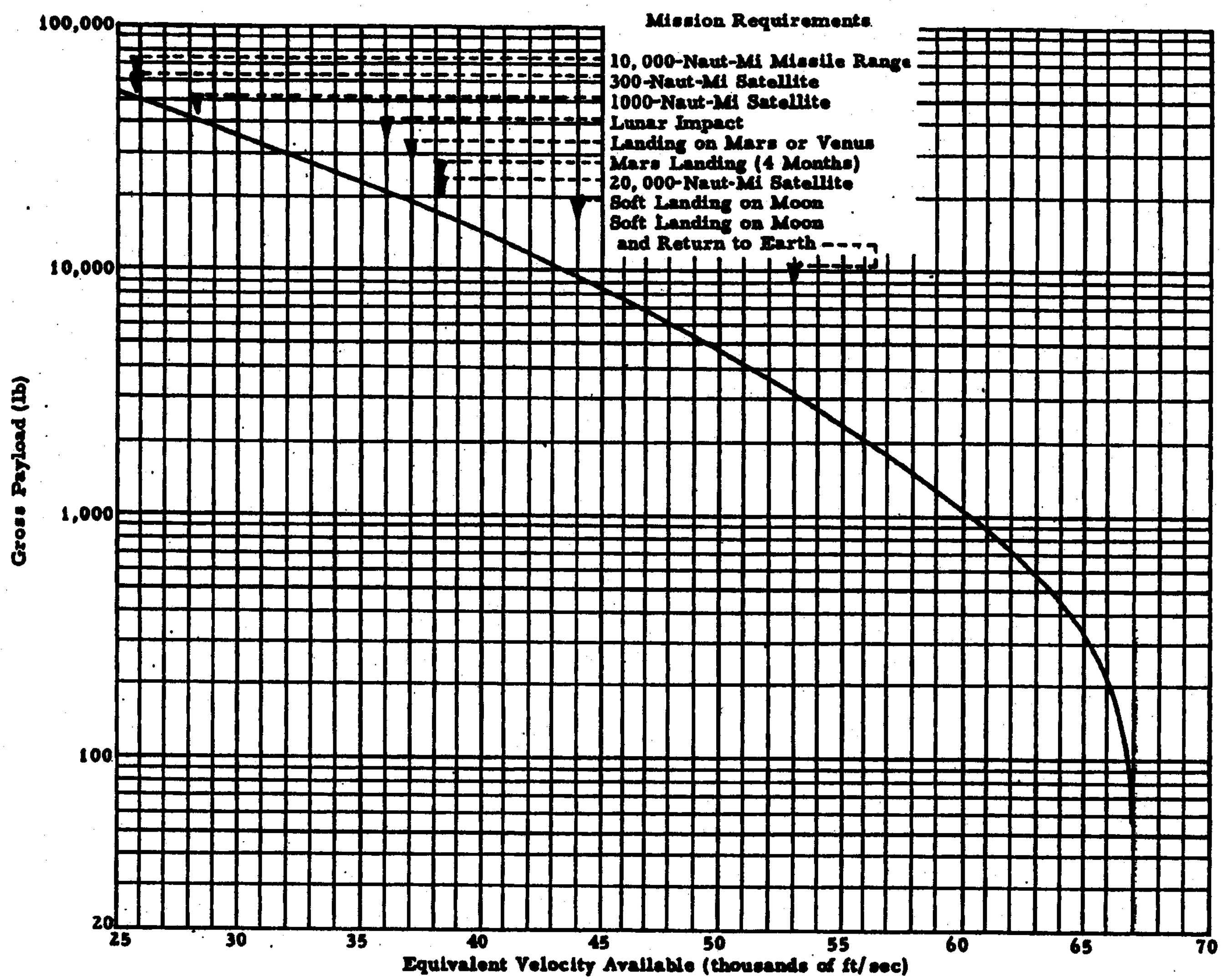


Figure 5-14. Gross Payload Versus Equivalent Velocity, Combination Number 14.

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Stage No.	1	2	3
Stage Designation	Booster (2000-1275)	Titan I (301-170)	Titan II (81-45)
Stage Weight (lb)	1,275,000	170,000	44,500
Propellants	Lox-RP	Lox-RP	Lox-RP
Thrust (lb)	2,000,000	301,000	80,800
I_{sp} (lb-sec/lb)	290	290	310

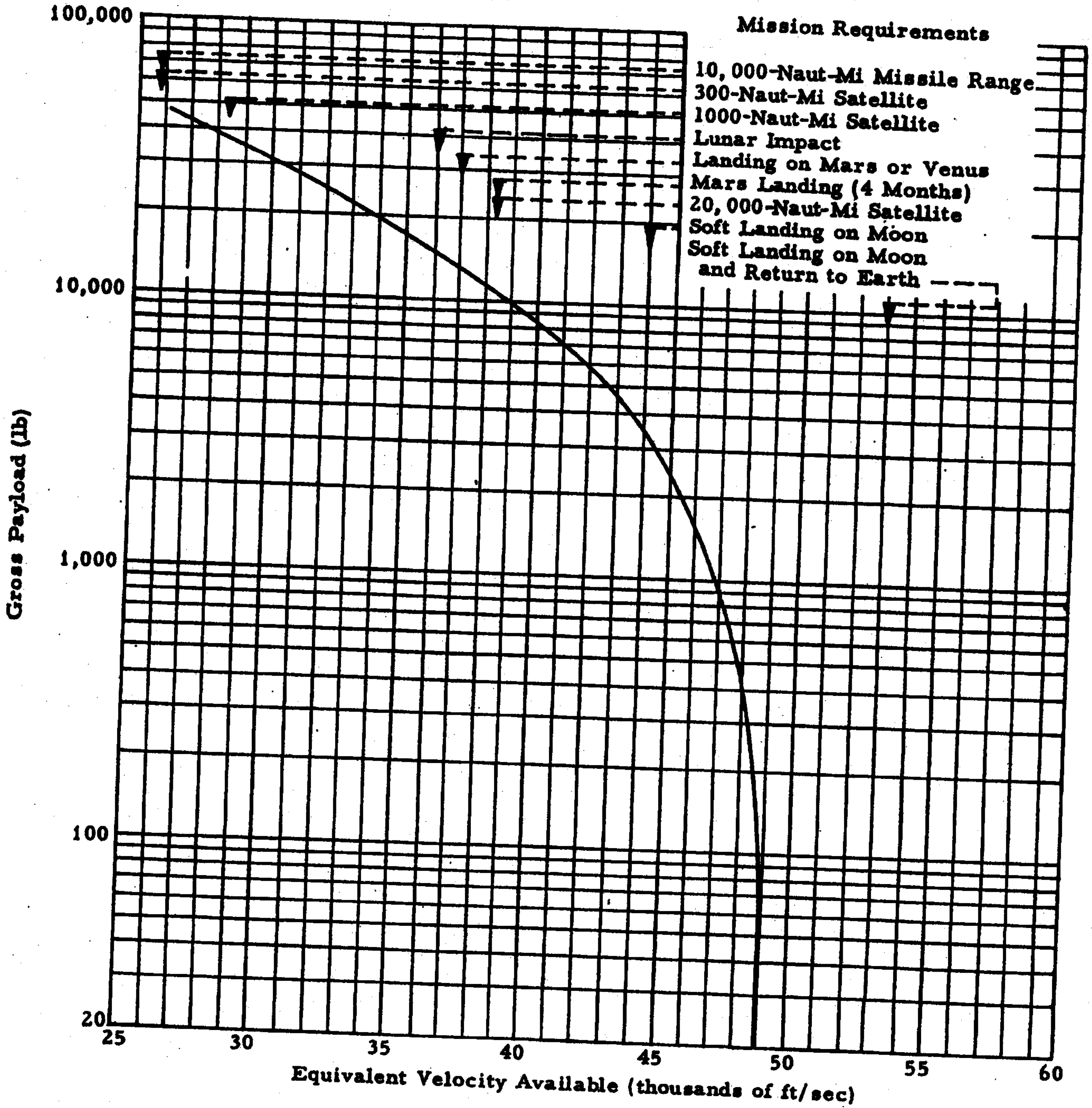

Figure 5-15. Gross Payload Versus Equivalent Velocity, Combination Number 15.

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Stage No.	1	2	3
Stage Designation	Booster (1000-640)	Thor (175-107)	Fluorine (12-12)
Stage Weight (lb)	640,000	106,600	12,000
Propellants	Lox-RP	Lox-RP	Fluorine-Hydrazine
Thrust (lb)	1,000,000	175,000	12,000
I_{sp} (lb-sec/lb)	290	290	364

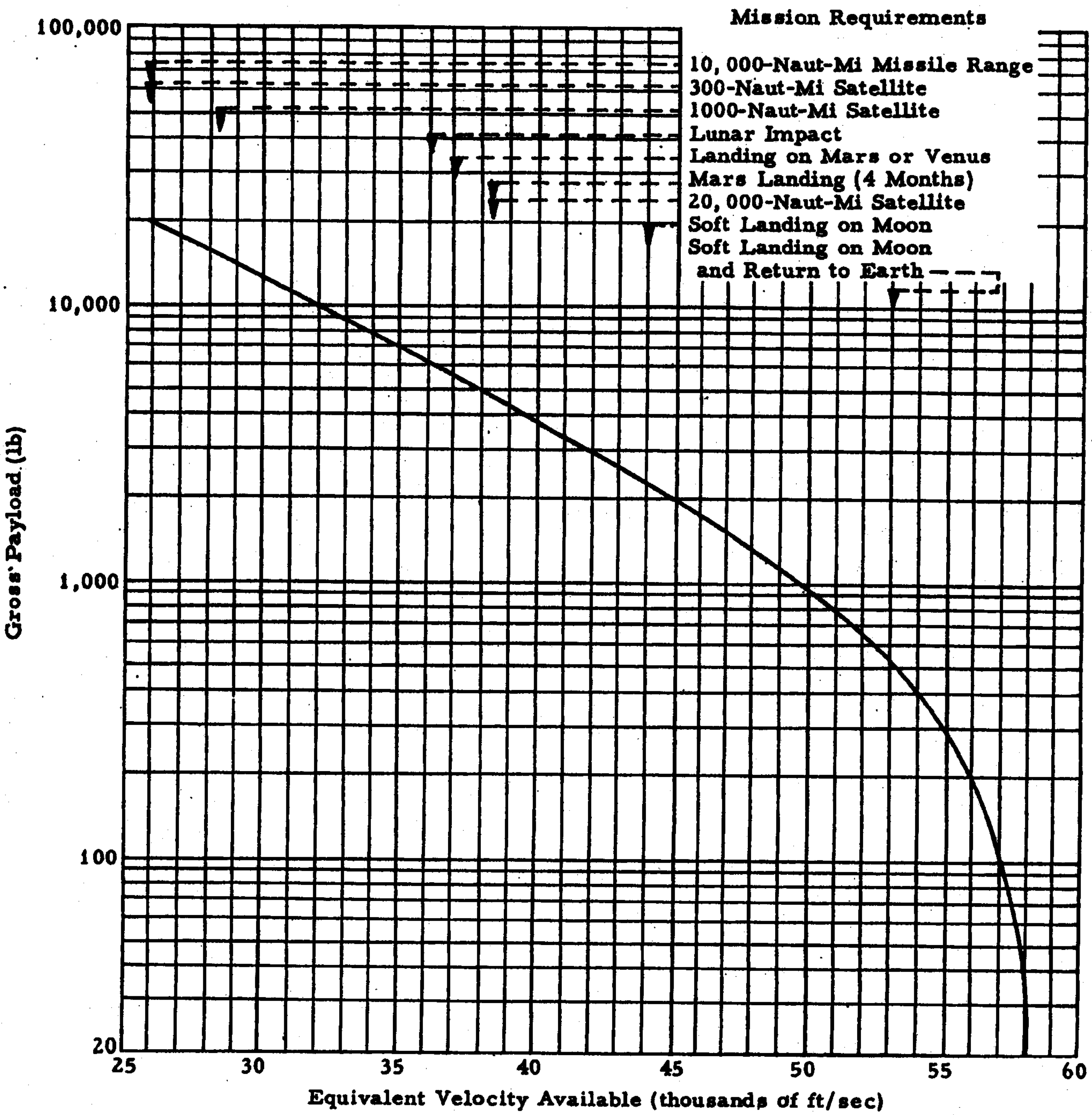

Figure 5-16. Gross Payload Versus Equivalent Velocity, Combination Number 16.

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