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IMPROVING THE UPRATED SATURN I

by

C. E. Tharratt
C. Eng., F.R.Ae.S., F.S.E., F.B.I.S.

Program Manager - Advanced Vehicle Systems
Chrysler Corporation Space Division,
New Orleans

formerly Chief Rocket Development Engineer,
Westland Aircraft Ltd.
Cowes, Isle of Wight

INTRODUCTION

The October 1966 issue of 'Spaceflight' reported the award by NASA of contracts for the continuation of studies directed toward the improvement of the Uprated Saturn I (formerly Saturn IB) launch vehicle.

This paper discusses five improved versions of the Uprated Saturn I that were studied by the Chrysler Corporation Space Division, supported by the Douglas Aircraft Corporation. The configurations evaluated all employed solid rocket motor (SRM) strap-ons in either Zero Stage or boost assist applications. The objective of the study was to investigate several methods by which the performance capability of the Uprated Saturn I can be increased, and to determine the design changes and the impact upon facilities, ground support equipment, schedule and costs. The study was also supported by the United Technology Center, and the Thiokol Chemical Corporation, with solid rocket motor data, and by Rocketdyne, with liquid engine data. The study was directed for NASA through the Marshall Space Flight Center. A parallel study of the launch facility requirements was separately conducted by the Martin Company, for the Kennedy Space Center.

Although this paper is largely confined to discussing the payload capability of the aforementioned five configurations, it is important to recognize that the singular achievement of maximum payload does not in itself lead to the most attractive vehicle. Investment cost and total program cost-efficiency (dollars per lb of payload in orbit), each have a decisive role to play in vehicle selection and can have a sobering effect on enthusiasm, unless the mission requirement is of such importance that cost takes a subordinate role. With this qualification, this paper will demonstrate how dramatically the payload and mission capability of the powerful, reliable, Uprated Saturn I can be increased by the addition of simple state-of-the-art methods of vehicle improvement.

The current Uprated Saturn I is shown in figures 1 and 2; it is an improved version of the Saturn I. Altogether, ten Saturn I launch vehicles have been built and successfully launched from Cape Kennedy. Three of the Uprated Saturn I series have also been successfully launched and a fourth and fifth are currently being prepared. The Uprated Saturn I has already demonstrated its power to orbit bulky payloads, however, as knowledge and skill in executing space activities increase, so the need for improving the payload capability of the Uprated Saturn I becomes more pressing. Therefore, trade studies were conducted on a number of important variables to provide sufficient performance, engineering design, costs and schedule data on each configuration with its variations, to permit a selection of the most attractive vehicles for further in-depth study. It should be noted that an Uprated Saturn I with a payload capability of 38,000 lbs. into a low earth orbit, was used throughout this study as a reference vehicle. Current Uprated Saturn I launch vehicles have a capability of over 40,000 lbs. Thus, the payload capability of the launch vehicle configurations studied and discussed herein are all approximately 2000 lbs low.

TRADE STUDIES

The vehicles studied are shown in figure 3. The MLV-11 has a zero stage consisting of four 120-inch UA 1200 solid rocket motors. The eight H-1 engines on the S-IB boost stage are ignited at altitude. MLV-12 also has four 120-inch SRM, but in this case the S-IB stage of the core vehicle has only four H-1 engines, and all engines (solid and liquid) are ignited on the pad. The fins on both the MLV-11 and MLV-12 configurations were deleted. The MLV-13 configuration is boosted by two 120-inch SRM. The MLV-14 and MLV-15 configurations are boost assisted respectively by four and by eight Minuteman SRM's. All configurations employ 205 K H-1 engines on the S-IB stage and a standard J-2 engine on the S-IVB stage. In each case, programmed mixture ratio shift was utilized for the S-IVB stage, as required. During the course of the study, a variation of the MLV-11 and MLV-12 configurations showed such promise that is was also studied. This configuration was designated MLV-11A and consisted of igniting the four SRM and four H-1 engines on the pad and the remaining four H-1 engines just prior to SRM separation.

As an aid in the selection of vehicles, flight performance computations were made to determine payloads, maximum dynamic pressure (max q), SRM impact points, vehicle controllability, and vehicle lift-off motion. These analyses were conducted parametrically with prime variables of S-IB propellant tank extension, number of 120 inch SRM segments (5 or 7), ignition sequence, and SRM jettison time.

The results of the payload versus S-IB tank length studies are presented in figures 4 and 5 along with the corresponding maximum dynamic pressures. These results indicate that for the MLV-11 and MLV-12 configurations, S-IB tank extensions yields either no payload gain or a payload loss. For the MLV-13 and

MLV-11A configurations, tank extensions of 20 feet or greater result in significant payload increases with corresponding reductions in maximum dynamic pressure. For the MLV-14 and MLV-15 configurations, S-IB tank extension also results in a payload increase, but the gain to be expected in progressing from a 10-foot to a 20-foot extension is not sufficient to justify the additional vehicle and facility costs involved. For these two configurations the effect of igniting half the number of MM SRM on the pad and the remainder at 15 seconds or 70 seconds was also investigated. In each case there was a limit to the permissible tank extension, for tower collision was possible with thrust to weight ratios of less than 1.2 g.

Program costs, as a function of S-IB tank length, were also prepared for each of the study configurations. These costs were non-dimensionalized, figure 6, by referencing them to MLV-14 with a zero tank extension as this proved to be the least expensive study vehicle. To assist in this evaluation, the cost efficiency (dollars per pound of payload in low earth orbit) for each configuration was prepared. These values were also non-dimensionalized by referencing all configurations to the MLV-11.7A configuration with a 20-foot tank extension, as it was estimated that this vehicle had the best cost effectiveness. This cost analysis clearly indicated the desirability of tank extensions of 10 feet to 20 feet for MLV-11A, -13, -14, and -15, while no tank extensions were indicated for the MLV-11 and MLV-12 configurations. At the conclusion of the trade studies, the five configurations shown in figures 7 thru 11 were selected for further in-depth study, and the results that follow all refer to these launch vehicles.

PERFORMANCE AND MISSION APPLICATIONS

The Uprated Saturn I is playing a major role in the Apollo program and the Improved Uprated Saturn I configurations extend this capability to support the national space program objectives of earth orbital operations, extended lunar operations and manned planetary exploration. Some of the more important performance and mission capabilities of the Improved Uprated Saturn I configurations are presented in the following sub-sections.

a) General Earth Orbital Performance

The payload capability of each configuration for circular orbits ranging between 80 to 300 nautical mile altitudes, assuming a 72 degree launch azimuth out of Cape Kennedy, are presented in figure 12. It will be noted that a payload capability of over 100,000 lbs is possible with MLV-11.7A in a 105 nautical mile circular orbit.

b) Earth Orbital Operations

The payload capability for general space station applications in low inclination and polar orbits of interest is given in figure 13. Only the payloads for MLV-11.7A and the Uprated Saturn I are shown as the capabilities of the remaining configurations fall between these boundaries in a similar fashion to that indicated on figure 12. The types of space station considered are the Manned Orbiting Telescope (MOT), Manned Orbiting Laboratory (MOL), National Research Observatory (NRO), and a Manned Orbiting Research Laboratory (MORL). The shaded area of the diagram indicates a payload/altitude range for these applications. Note that the payload can be increased to the maximum capability of the MLV-11.7A launch vehicle configuration. Thus the payload can be increased to over 100,000 lbs in low inclination orbits or 80,000 in a polar orbit.

One of the most talked about "space stations" for the Uprated Saturn I is the S-IVB orbital workshop which is planned as part of the Apollo Applications Program. This, in its essentials, consists of an empty S-IVB with a means of ingress and egress for the astronauts. In this application the S-IVB forms part of the payload, and figure 14 shows that the MLV-11.7A could loft over 130,000 lbs into a low inclination orbit, or over 100,000 lbs into a polar orbit. Again, the range between the limits for the Uprated Saturn I and MLV-11.7A is filled by the capabilities of the other configurations.

c) General High Energy Performance

High energy missions mean those missions having an injection velocity greater than that required for escape (i.e. $> 36,000$ fps.). The payload capabilities of the study vehicles were determined for this velocity range both with and without a Centaur type upper stage. In all cases, a direct injection at a 105 n.m. perigee altitude from a due east launch was used. The high energy performance with S-IVB injection (two starts) and with a Centaur upper stage is summarized in figure 15. The energy parameter used throughout these analyses is defined as $C_3 = (V_{burnout})^2 - 2\frac{\mu}{r}$ where r is the radius corresponding to an altitude of 105 n.m. The curves (A thru F) on the left of figure 15 represent the performance attainable with S-IVB injection and those on the right (A' thru F') illustrate the dramatic increase in payload capability when a Centaur type high energy upper stage is added. Figure 16 summarizes the payload capability with Centaur injection for a number of important high energy missions. From this diagram, it will be noted that MLV-11.7A can inject 40,000 lbs to escape or 28,000 lbs to the vicinity of Mars for a Voyager mission. Similar capability using S-IVB injection only would result in 28,000 lbs to escape or 9,000 lbs for a Voyager mission.

d) Synchronous Orbit Capability

One mission that has injection requirements within the high energy range discussed in the preceding sub-section, is the synchronous (24-hour) circular orbit mission. This orbit is of particular interest because of its unique characteristics that can be used for a wide variety of applications.

The synchronous orbit capability for each of the vehicles considered, as shown in figure 17, was based on injection by the S-IVB stage using 3 burns of the J-2 engines. No data is presented for the MLV-14 configuration since the payloads achievable are either impractically small or non-existent. The mission profile used was the 105 n.m. circular parking orbit accomplished at the completion of the S-IVB first burn with sufficient flight propellant reserves provided during the suborbital flight to ensure the required injection accuracy in the parking orbit. The first coast period in this parking orbit was assumed to be 4.5 hours. The second burn provides the velocity increment required to place the payload and the S-IVB stage, with its remaining propellant, into the desired transfer ellipse. The second coast period is spent in this transfer ellipse and takes approximately 5 hours. The velocity for circularization and any required plane change is provided concurrently by the third and final S-IVB burn. These synchronous orbit studies using S-IVB injection investigated only one mission profile, whereas the advantages and disadvantages of three mission profiles were examined during the analysis of synchronous orbit capability utilizing a Centaur type injection stage. For this case, each profile involved a due east launch and injection into a parking orbit at an altitude of 105 n.m. Following a parking orbit coast time (assumed to be 4.5 hours), a velocity increment was applied at the 105 n.m. altitude sufficient to place the vehicle in a transfer orbit with an apogee at the desired synchronous orbit altitude. A 5-hour transfer time was used. At the synchronous orbit altitude, the velocity required for circularization and any desired plane change was provided. The three profiles are presented schematically in figure 18, and differ only in their staging sequence and the characteristics of their parking orbits.

Profile A assumed injection into a 105 n.m. circular parking orbit (CPO) at S-IVB burnout. At the end of the parking orbit phase, the Centaur was ignited to provide the velocity increment required to place the vehicle in the desired transfer orbit (TO). This increment was applied when the vehicle was in a nodal crossing position in the parking orbit. When synchronous altitude was reached, the Centaur was ignited the second time to provide concurrently the velocity increment required for circularization and any required plane change.

Profile B assumed that the S-IVB injected the fully loaded Centaur and payload into an elliptical parking orbit (EPO) at the perigee altitude of 105 n.m. At the completion of the parking orbit phase, the Centaur was ignited to achieve the desired transfer orbit. This ignition occurred at the 105 n.m. perigee of the parking orbit. At synchronous altitude, the Centaur was re-ignited to provide the velocity required for circularization. Unlike Profile A, for Profile B the plane change maneuver is not combined with the circularization maneuver since the transfer orbit apogee does not coincide with a nodal crossing. To provide the velocity required for the plane change necessary for synchronous equatorial orbit missions, a third burn of the Centaur is necessary.

Profile C follows the profile described as Profile A with one notable exception. Rather than penalize the performance of the vehicle by off-loading S-IVB propellant, an S-IVB restart capability was assumed. The transfer velocity increment was, therefore, provided by a combination of S-IVB second burn and Centaur first burn. Additionally, since the circular parking orbit was maintained, allowing transfer to be effected at a nodal crossing point, the advantage of concurrently providing the circularization and plane change velocity can be retained.

The synchronous orbit performance of each of the vehicles with a Centaur type upper stage is shown in figure 19, and the most advantageous profile is indicated.

e) Extended Lunar Exploration

The escape capability ($C_3 = 0$) of 28,000 lbs and 40,000 lbs for S-IVB and Centaur injection respectively referred to in subsection C, general high energy performance, gives MLV-11.7A an excellent potential as a transport vehicle for lunar exploration support in the role of a soft lander, lunar orbiter, or for a circum-lunar/return mission. For instance, assuming MLV-11.7A with S-IVB injection, the payload in the vicinity of the moon is approximately 28,000 lbs. This could be used to orbit approximately 17,000 lbs of useful payload or soft land 5,000 - 7,000 lbs of useful payload on the lunar surface. Thus an entirely new field of activity is available for this configuration when a mission of this type becomes a reality.

f) Manned Planetary Exploration

The family of improved Uprated Saturn I launch vehicles have the capability to play a major role in the precursor, development and scientific support missions associated with determining the nature of the environment of the far planets that will eventually be visited by astronauts from the earth. Typical missions are the Voyager type missions to Mars, Venus and Jupiter. Assuming a conservative energy level of $C_3 = 25\ km^2/sec^2$ for a Mars Voyager mission, including allowances for a 30 day launch window at any opportunity, accounting for parking orbit losses and a $\pm$ 40° asymtotic declination of the outgoing geocentric asymtote, the Uprated Saturn I with a third stage such as Centaur, could place 8,000 lbs of useful payload in the vicinity of Mars. Alternatively the payload with MLV-11.7A is 11,000 lbs and that for MLV-11.7A with a Centaur type third stage is 28,000 lbs. Figure 20 shows how these payloads change for a Venus Voyager ($C_3 = 11\ km^2/sec^2$). For Jupiter Voyager ($C_3 = 90\ km^2/sec^2$) the important parameter is flight time. In this case it would take the Uprated Saturn I with a third stage 2.5 years flight time to transport approximately 3,000 lbs of useful payload to the vicinity of Jupiter or 15,000 lbs with MLV-11.7A/third stage. Alternatively, this latter vehicle could execute a Jupiter flyby flight time of just over one year with a payload of 2,000 lbs.

The payload capabilities for the missions previously discussed are all based on the use of the high energy chemical propellants LH_2 and LO_2. Space technology is at the threshold of the introduction of new propulsion systems such as the solar electric propulsion system (SEPS) and nuclear propulsion. One of the extremely attractive advantages of SEPS lies in the fact that it utilizes solar energy collected on solar panel arrays and transformed into useful electrical energy. This method of transforming solar energy into electrical energy is a necessary requirement for systems operation on the spacecraft at the planet of interest, or during transit from earth to the planet, due to the long flight times involved. It is, therefore, logical to utilize these solar energy collecting arrays during transit to provide the source of energy for a small but continuous thrust. It has previously been shown that the payload to escape ($C_3 = 0$), for MLV-11.7A with S-IVB injection, is 28,000 lbs. Under the most stringent conditions of injection from a parking orbit and making an allowance for a $\pm$ 40° declination of the outgoing geocentric asymtote this payload reduces slightly to 25,000 lbs. Under these injection conditions and utilizing a SEPS upper stage, a typical Mars mission could soft land 9,500 lbs, including 1,000 lbs for pure science, on the Martian surface and still retain a 350 lb orbiter for transmitting data back to earth. Alternatively, a 5,300 lb Mars Mapper could be orbited to spiral into the Martian atmosphere. For a Jupiter flyby a 7,000 lb payload, including 2,250 lbs of pure science, could be transported to Jupiter with a two year flight time. Figure 21 shows the payload capability with a nuclear upper stage. The nuclear propulsion system chosen as the basis for this investigation had a thrust of 10,000 lbs and used NERVA technology. The payloads attainable for Venus, Mars, Mercury and Jupiter flyby's are indicated for both the Uprated Saturn I and MLV-11.7A. Finally, the performance for very high missions using a chemical third stage and a third stage with a chemical kick stage, is shown in figure 22. This diagram shows that, using a kick stage, a Jupiter flyby mission with a 4,000 lb package of instruments, is feasible with a flight time of one year, or a 1,000 lb package in a Saturn flyby in just over 18 months. Alternatively, very useful instrument payloads for solar probes to within 0.10 AU of the sun, or at high inclinations to the ecliptic, are possible.

g) Mission Summary

From the foregoing discussions it is clear that, in addition to playing a major role in the Apollo program, the improved Uprated Saturn I configurations are ideally suited for a wider range of missions associated with earth orbital operations, they have a transport/logistics capability for extended lunar exploration, and the capability for a major supporting role in the preliminary phases of manned planetary exploration.

SYSTEMS OPERATIONS

From a systems operations viewpoint, the simplest uprated configuration is MLV-14. In this instance the payload capability is increased to over 50,000 lbs by the simple addition of the four Minutemen SRM in a boost assist application. Following boost thrust tailoff, the spent cases are jettisoned on the release of a simple mechanical attachment. The control of the vehicle throughout this operation is exercised through gimballing the four outboard H-1 engines of the core vehicle. The operation and control of the remaining four configurations is somewhat more demanding for the added weight of the 120-inch SRM requires additional power for control. This is made available by injecting a liquid propellant (N_2O_4) through ports strategically situated around the exhaust nozzle of each SRM. The resultant chemical reaction causes a deflection of the exhaust jet to provide a thrust vector of adequate magnitude to control the vehicle. This system is known as Thrust Vector Control (TVC) and is used either solely, or in conjunction with the gimballed nozzle H-1 engine control system. After liftoff, a preprogrammed roll and pitch command originating in the Instrument Unit (IU) flight control computer (FCC), see figure 23, controls the individual SRM thrust vector control systems. The input signals to the TVC system, initiate the release of pressurized nitrogen tetroxide to the appropriate valve quadrants to provide thrust vectoring. The resultant vehicle motions are sensed in the IU, which initiates signals to null the control action. During SRM operation, surplus TVC fluid is uniformly released and dumped overboard. The start dump programmer is located on the individual SRM's and is commanded by a signal originating from either the ground equipment at liftoff, the stage command receivers during flight, or the IU.

Figure 24 indicates a typical flight sequence for MLV-11.7A. At liftoff, the four SRM and the four outboard H-1 engines are ignited. Ignition of the four inboard H-1 engines follows at approximately 115 secs, just prior to SRM thrust tail-off. Separation and jettison of the spent cases, following SRM burnout is effected, first, by initiating the ignitors of solid propellant translation motors located at the forward and aft ends of the SRM's, and second, by activation of separation charges on the forward and aft attachment structures. The translation motors on the SRM's combined with aerodynamic forces create a SRM/vehicle motion which ensures adequate vehicle clearance of the spent SRM cases during separation. In essence, this separation system closely follows the method so successfully proven on the Titan IIIC.

PROGRAM OPTION

One important result to emerge from the study of these five launch vehicle configurations was the recognition that a variable payload capability was possible by the staged addition of 120-inch SRM to a core vehicle and that this is an attractive method of meeting future mission requirements. This program option, see figure 25, can be provided in two ways. In the first instance, the core vehicle of MLV-11.5 (zero S-IB tank extension) can loft 37,000 lbs into a low earth orbit. Adding four five-segment 120-inch SRM, increases this capability to 78,000 lbs, or by factor of two. In the second instance, the core vehicle of MLV-11.7A (20 foot S-IB tank extension) can loft 35,000 lbs into a low earth orbit; the reduction in payload capability is due to the increased tank length and structural weight. Adding two seven-segment 120-inch SRM increases this payload capability by more than a factor of two to 82,000 lbs and adding a further two SRM increases the payload capability by a factor of three to 106,000 lbs, thus providing an extremely attractive payload and mission flexibility. The ellimination of the SRM for missions with reduced payload requirements, also leads to important reductions in the vehicle unit cost, thus substantially reducing the overall program cost. As stated in the introduction, these payloads are approximately 2000 lbs low due to the capability of the reference vehicle used for study purposes.

CONCLUSIONS

This study has suggested that the payload capability of the Uprated Saturn I can be increased nearly three fold, using proven methods and state-of-the-art techniques. Because of this attractive feature, the investment cost, or development cost, is low. Furthermore, the step by step addition of a solid rocket motor strap-ons for boost assist, provides a payload and mission flexibility hitherto unprecedented in large boosters. It has also been shown how the addition of a third stage such as Centaur, dramatically increases the ability of these launch vehicles to support planetary exploration.

Following the first lunar landings, the whole universe remains to be explored - Mars, Venus, Jupiter, and the other planets, the sun - the entire solar system. Space stations and orbiting research laboratories must be launched, supplied and resupplied, and kept functioning. This paper has attempted to show how the Improved Uprated Saturn I launch vehicle family can play an important role in achieving these goals.

ACKNOWLEDGEMENTS

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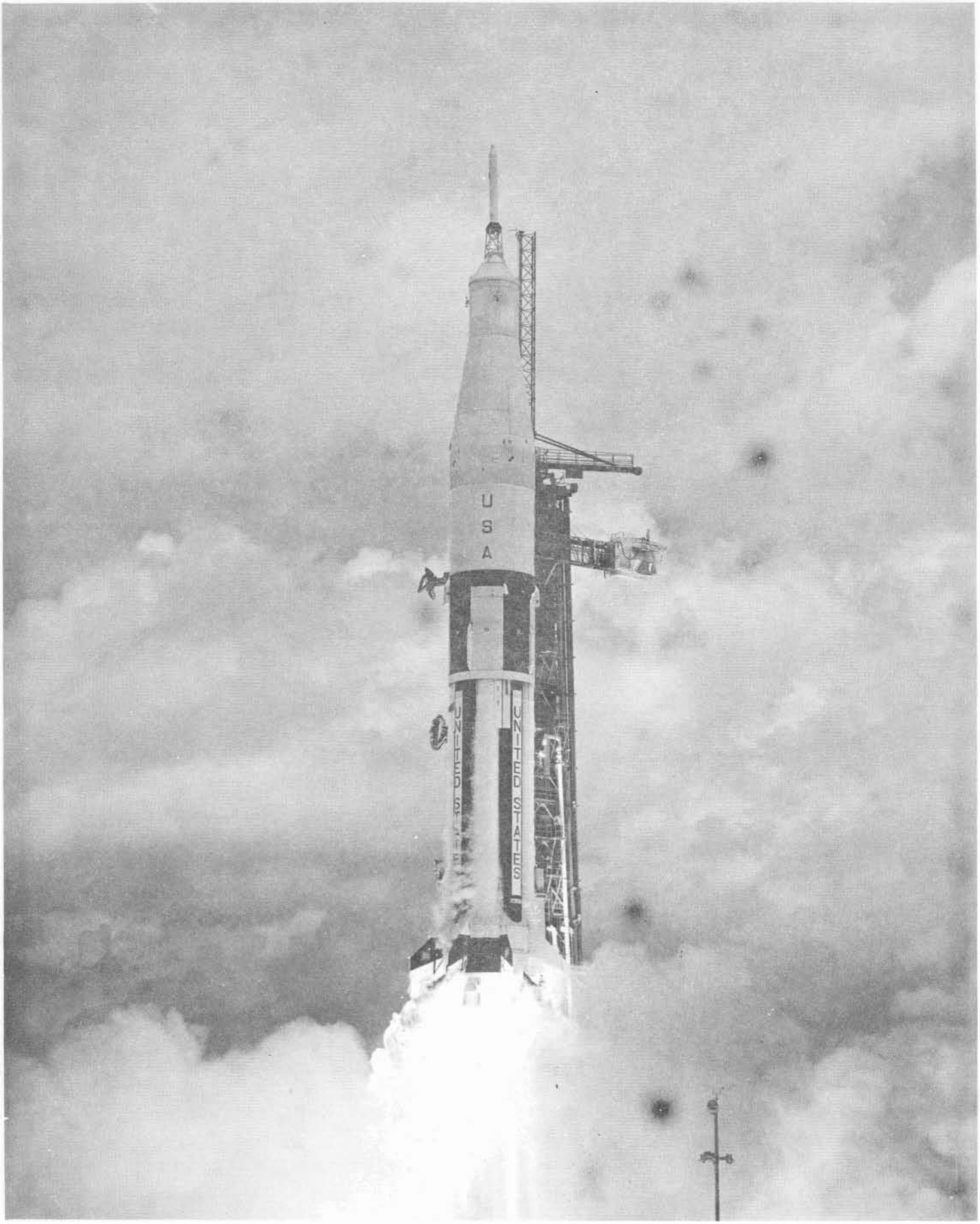


Fig 1

BASELINE

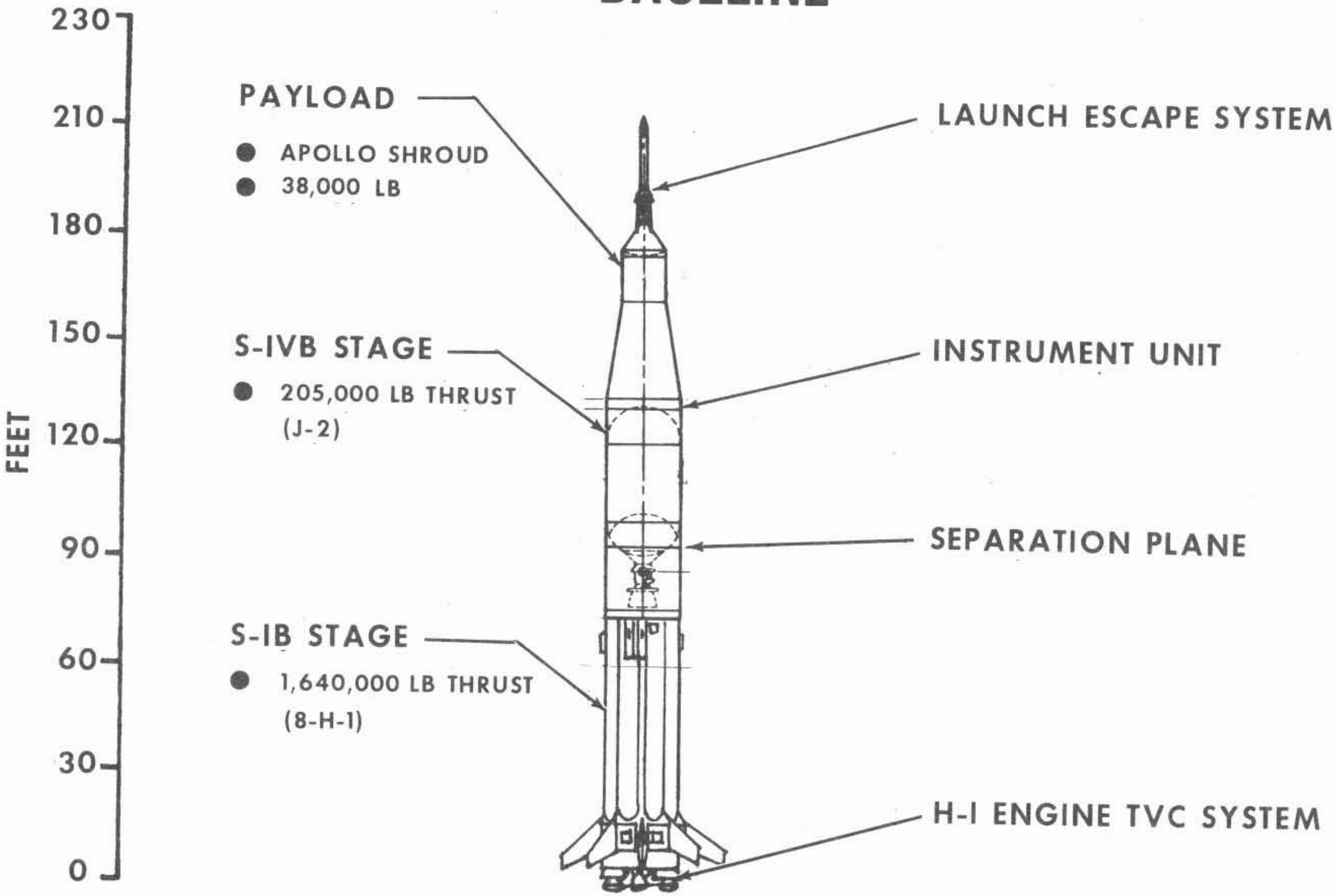


FIG 2

TRADE STUDY CONFIGURATIONS

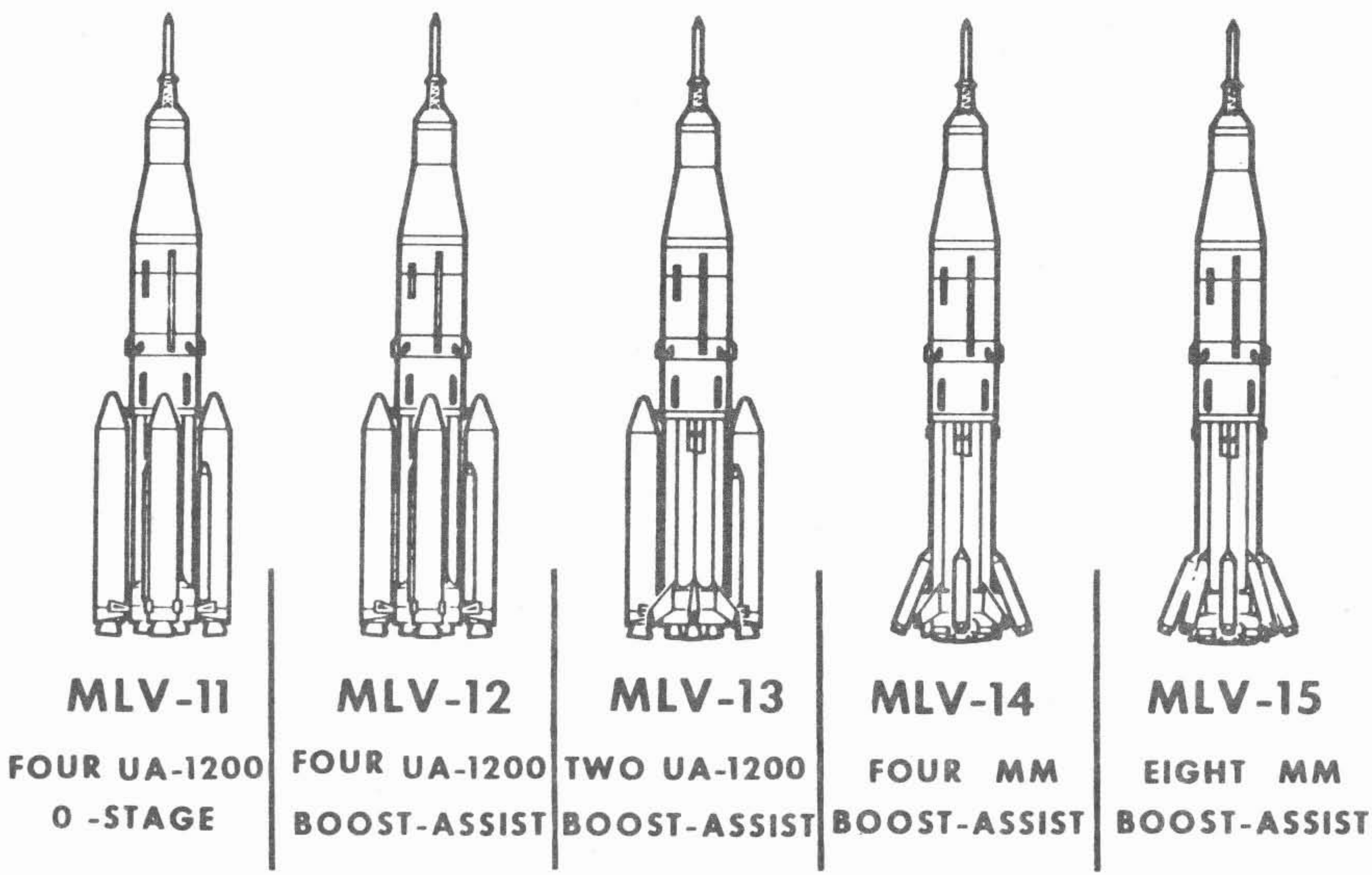


FIG 3

TRADE STUDY PERFORMANCE
OF CONFIGURATIONS USING UTC SRM

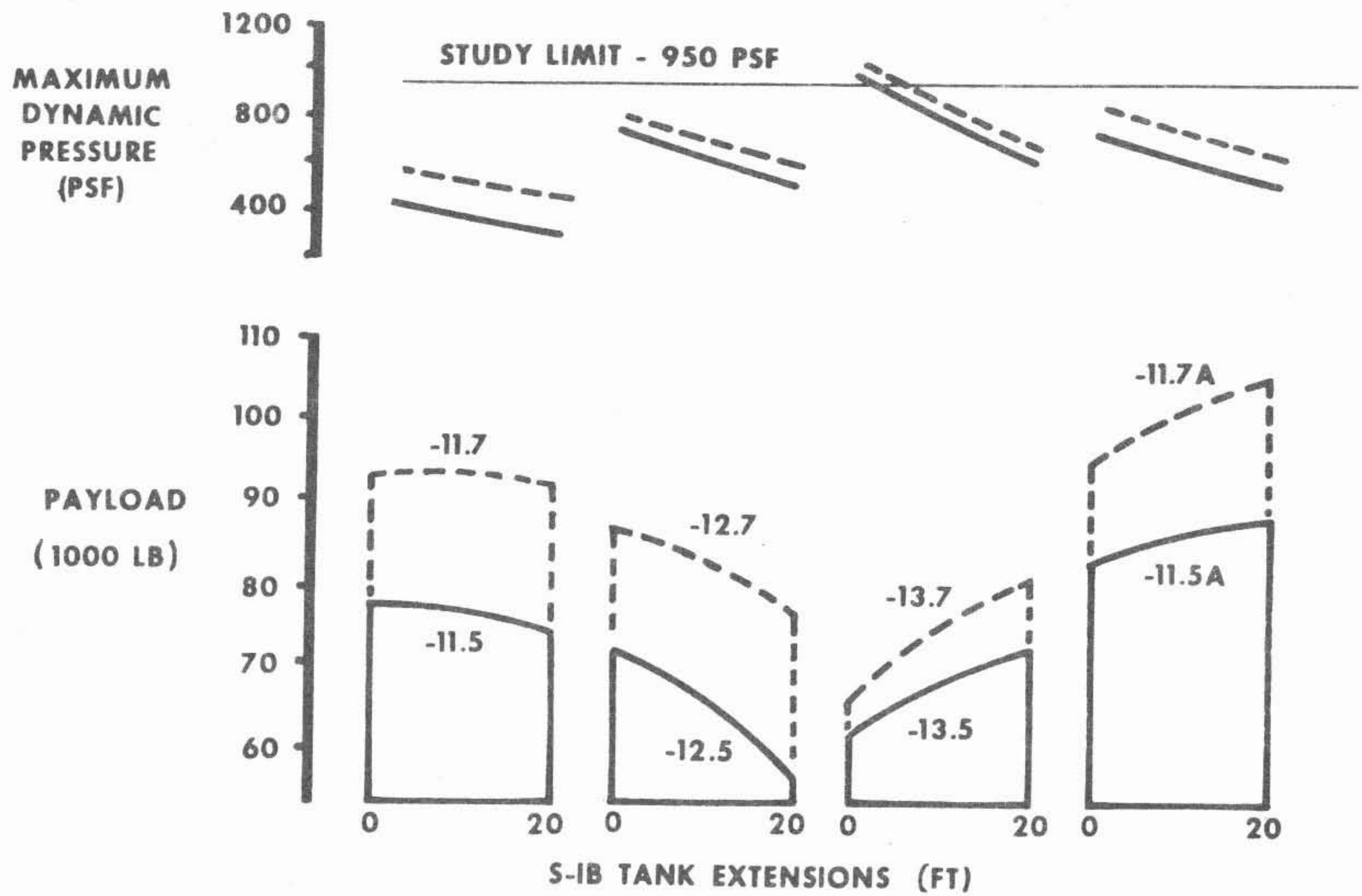


FIG 4

TRADE STUDY PERFORMANCE
OF CONFIGURATIONS USING MM SRM

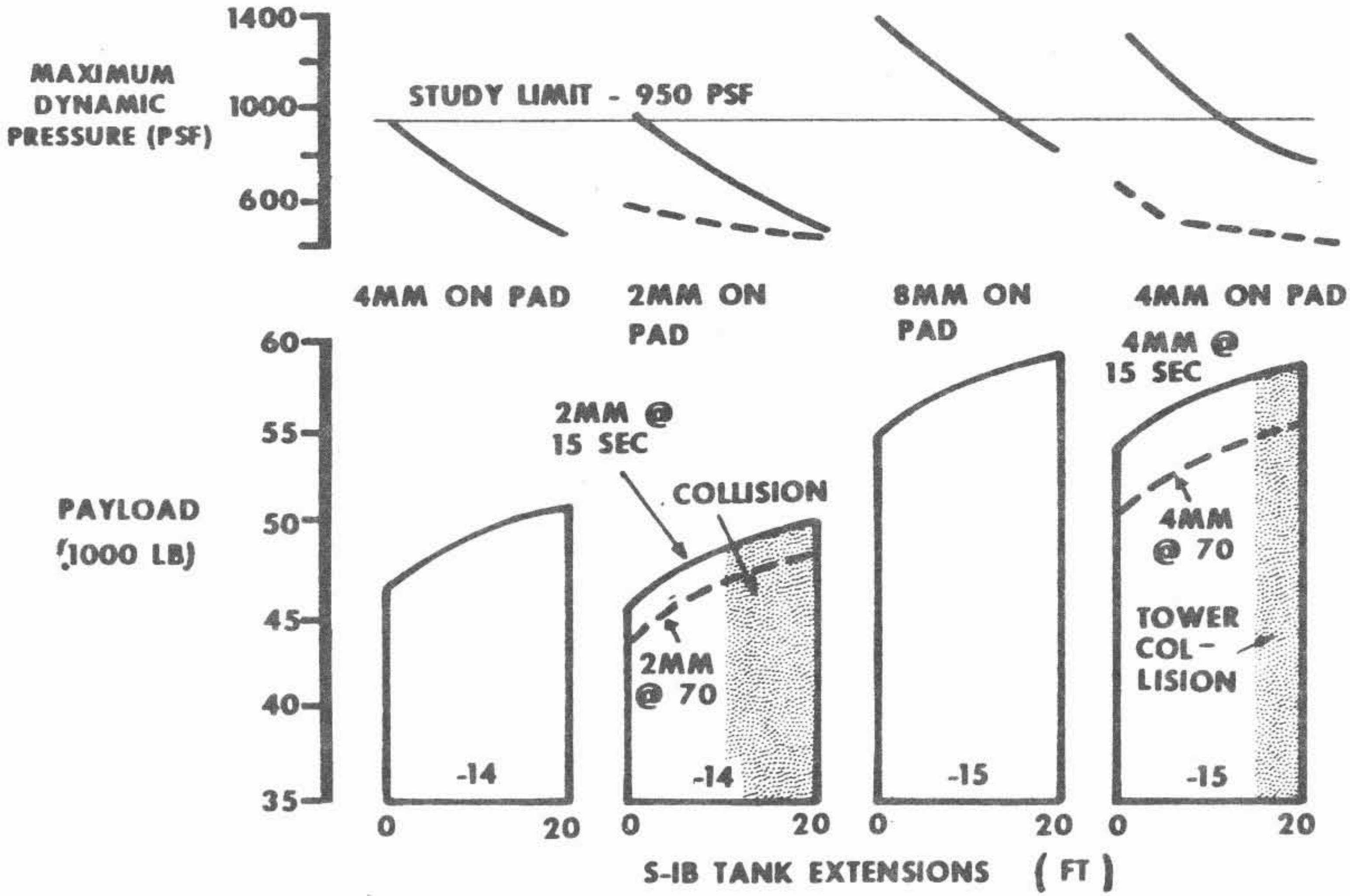


FIG 5

TRADE STUDY COST IMPACT

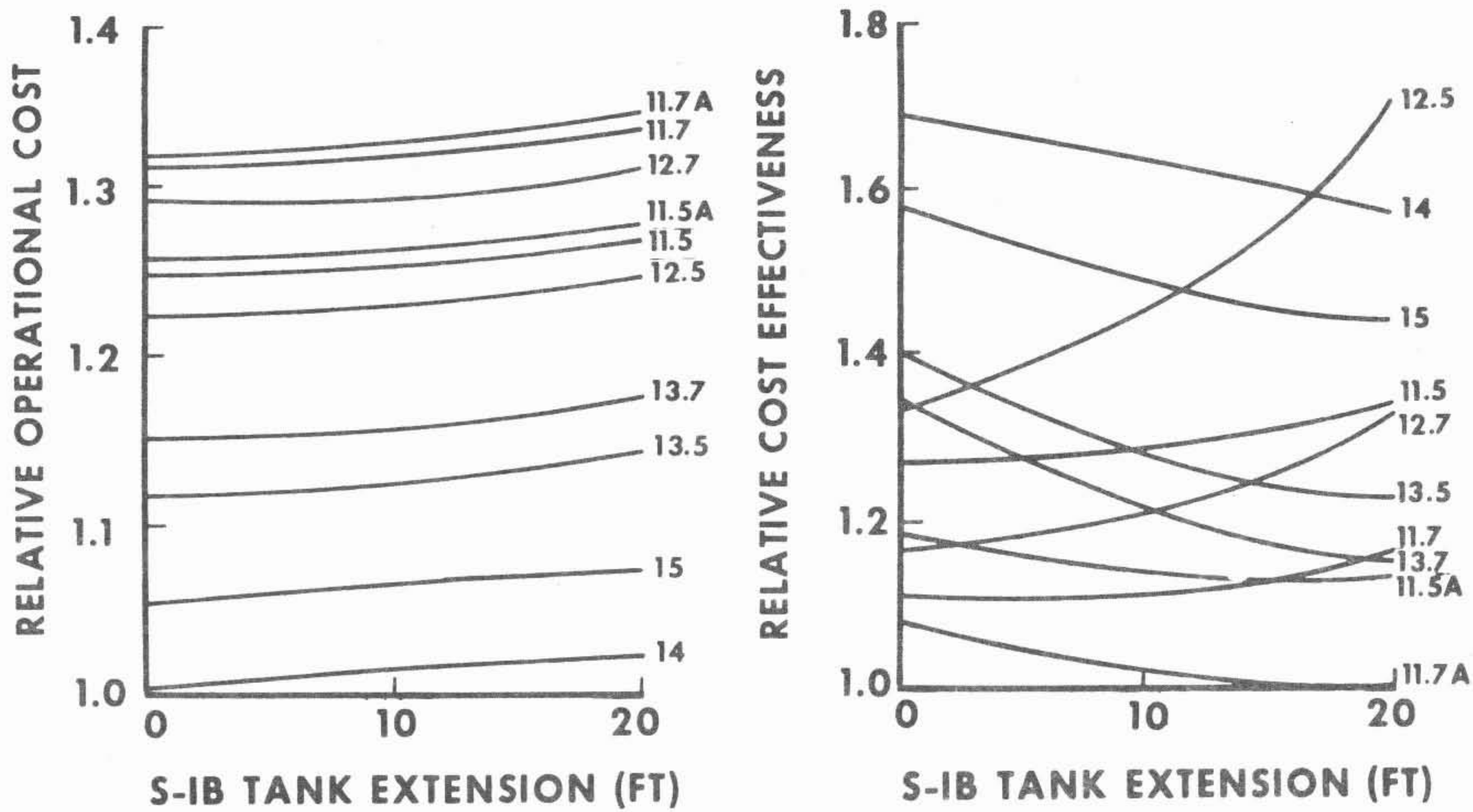


FIG 6

UPRATED SATURN I/4 UA-1205 BOOST ASSIST
(MLV-SAT-IB-11.5)

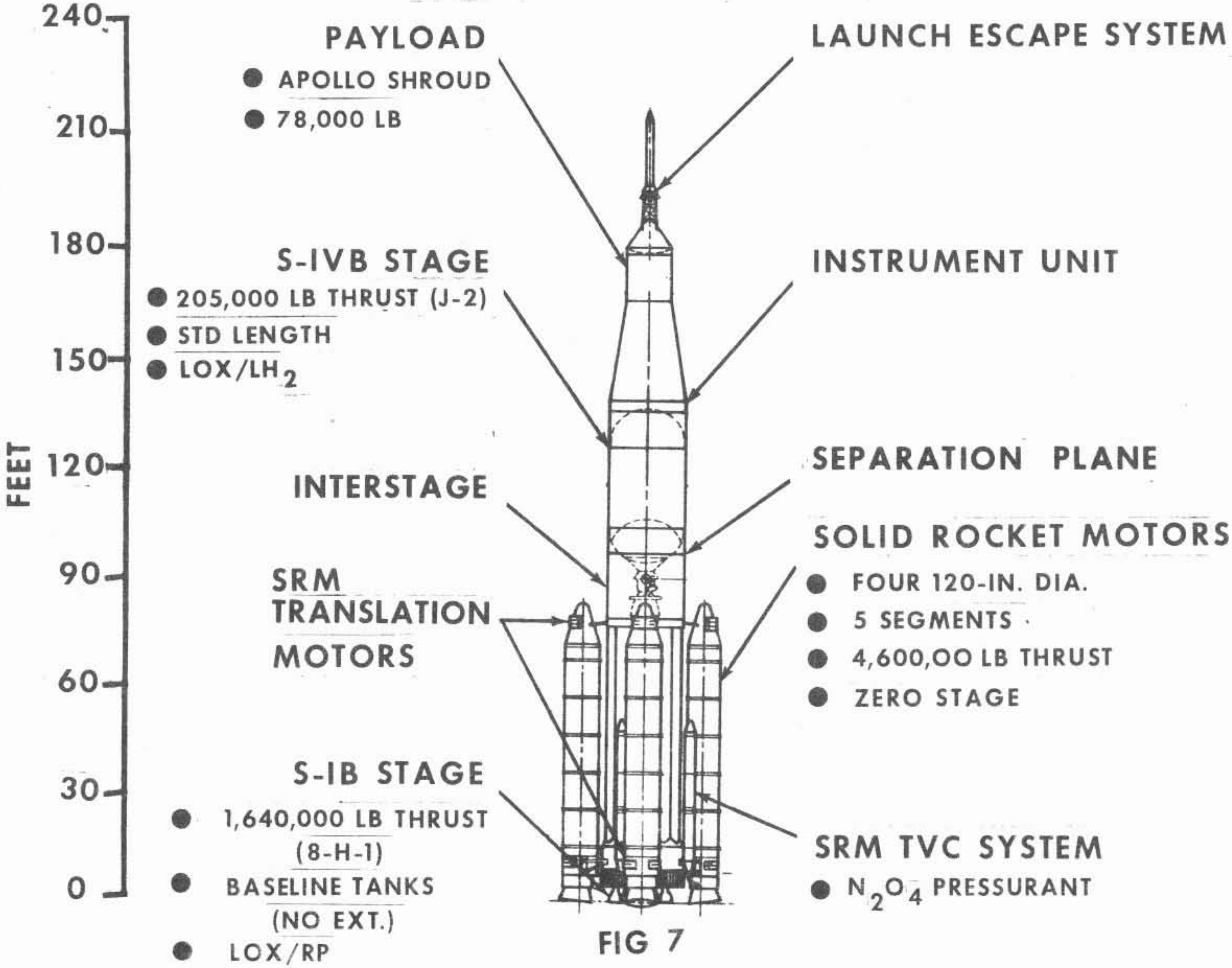


FIG 7

UPRATED SATURN I/4 UA-1205 BOOST ASSIST
(MLV-SAT-IB-11.5A)

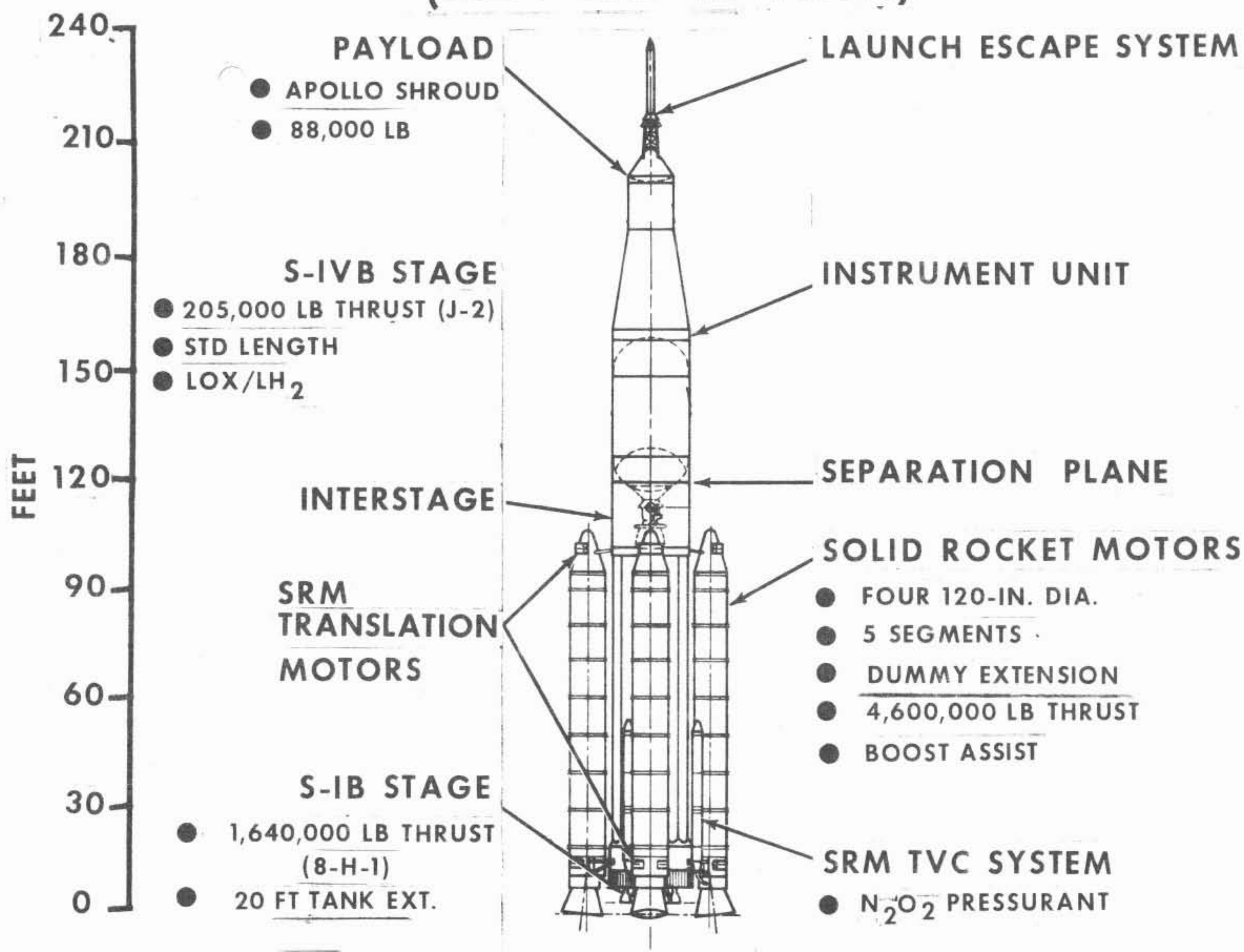


FIG 8

UPRATED SATURN I/4 UA-1207 BOOST ASSIST
(MLV-SAT-IB-11.7A)

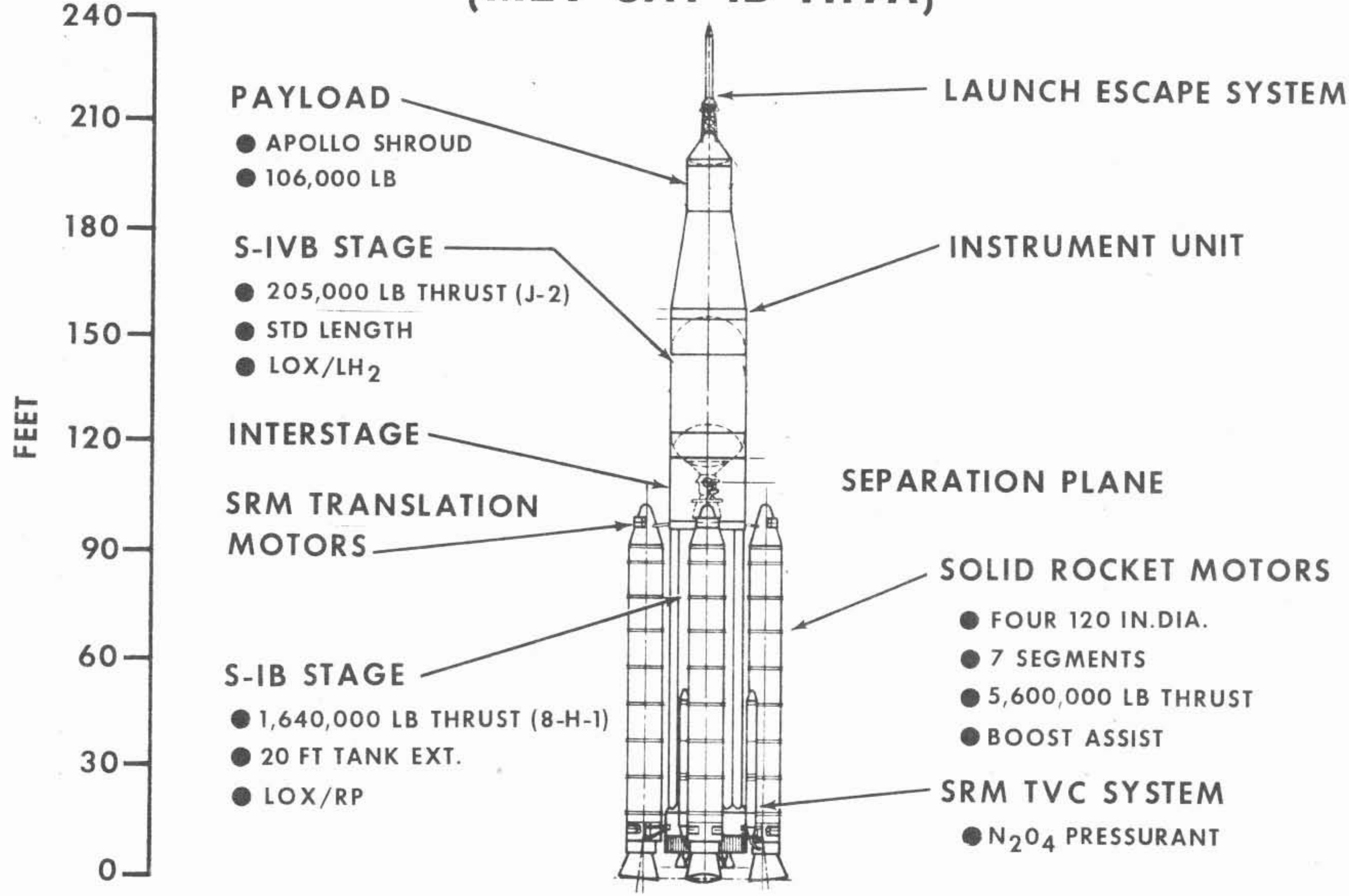


FIG 9

UPRATED SATURN I/2 UA-1207 BOOST ASSIST (MLV-SAT-IB-13.7)

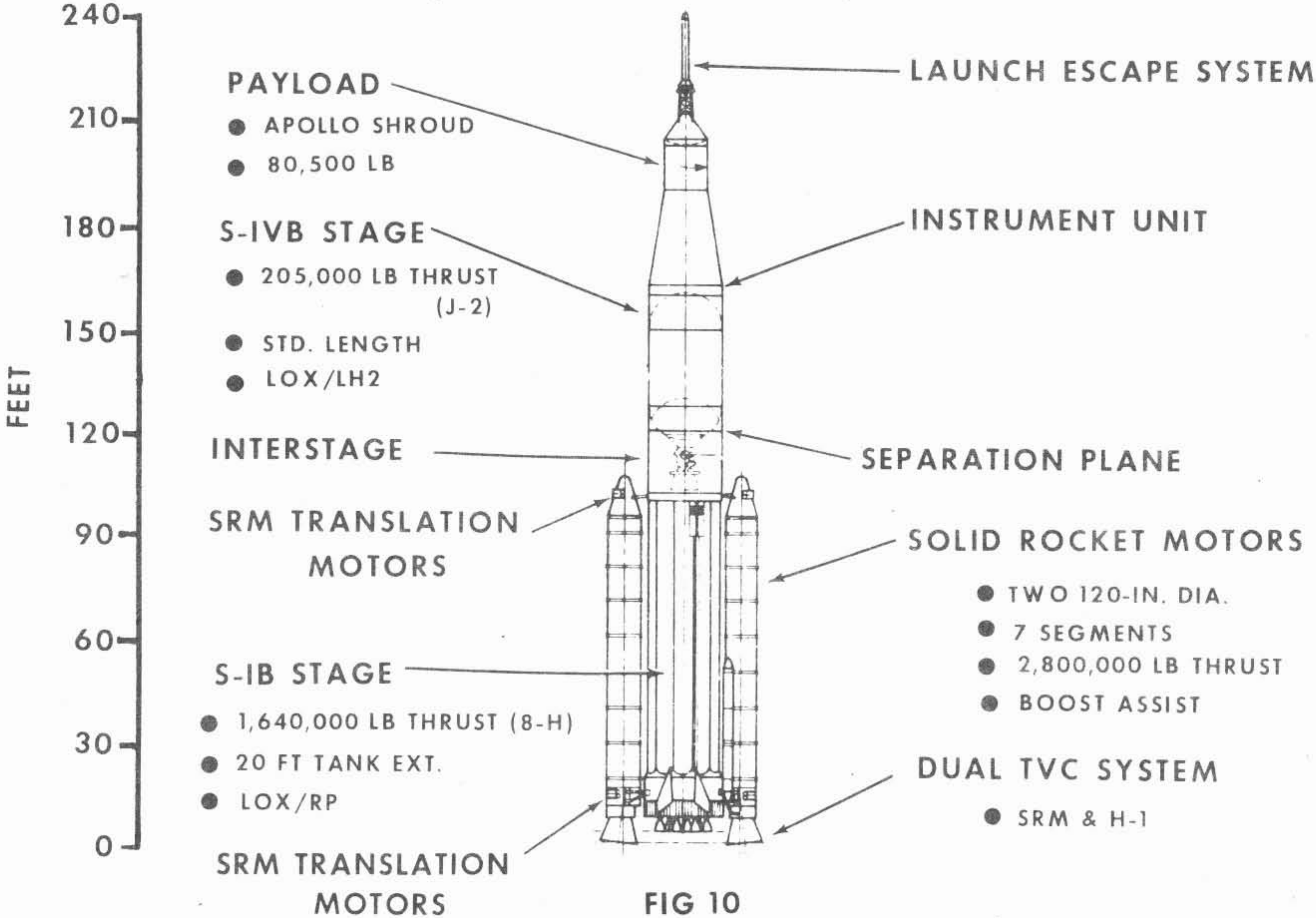


FIG 10

UPRATED SATURN I/MINUTEMAN BOOST ASSIST (MLV-SAT-IB-14)

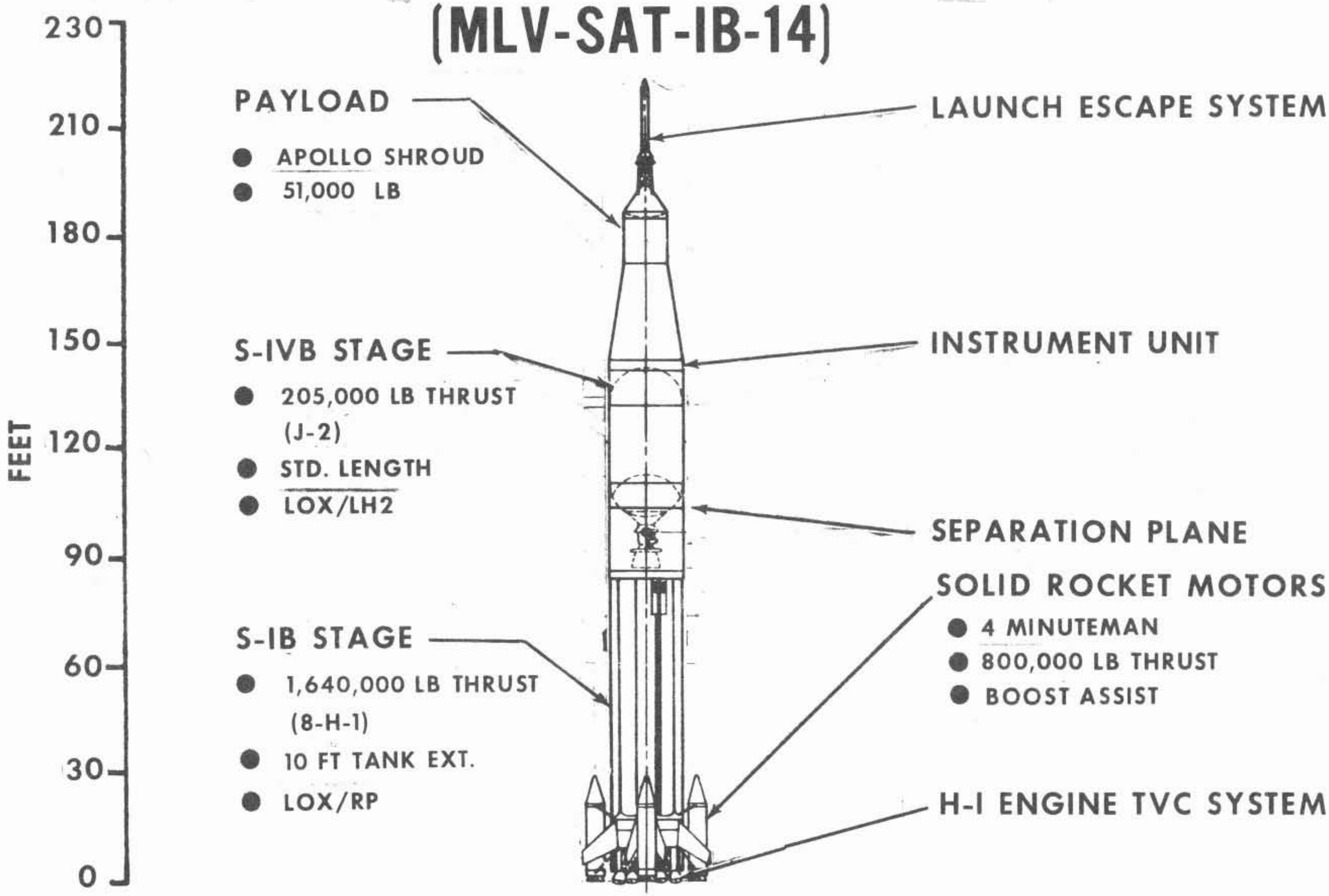


FIG 11

CIRCULAR ORBIT PERFORMANCE WITH ALTITUDE VARIATION

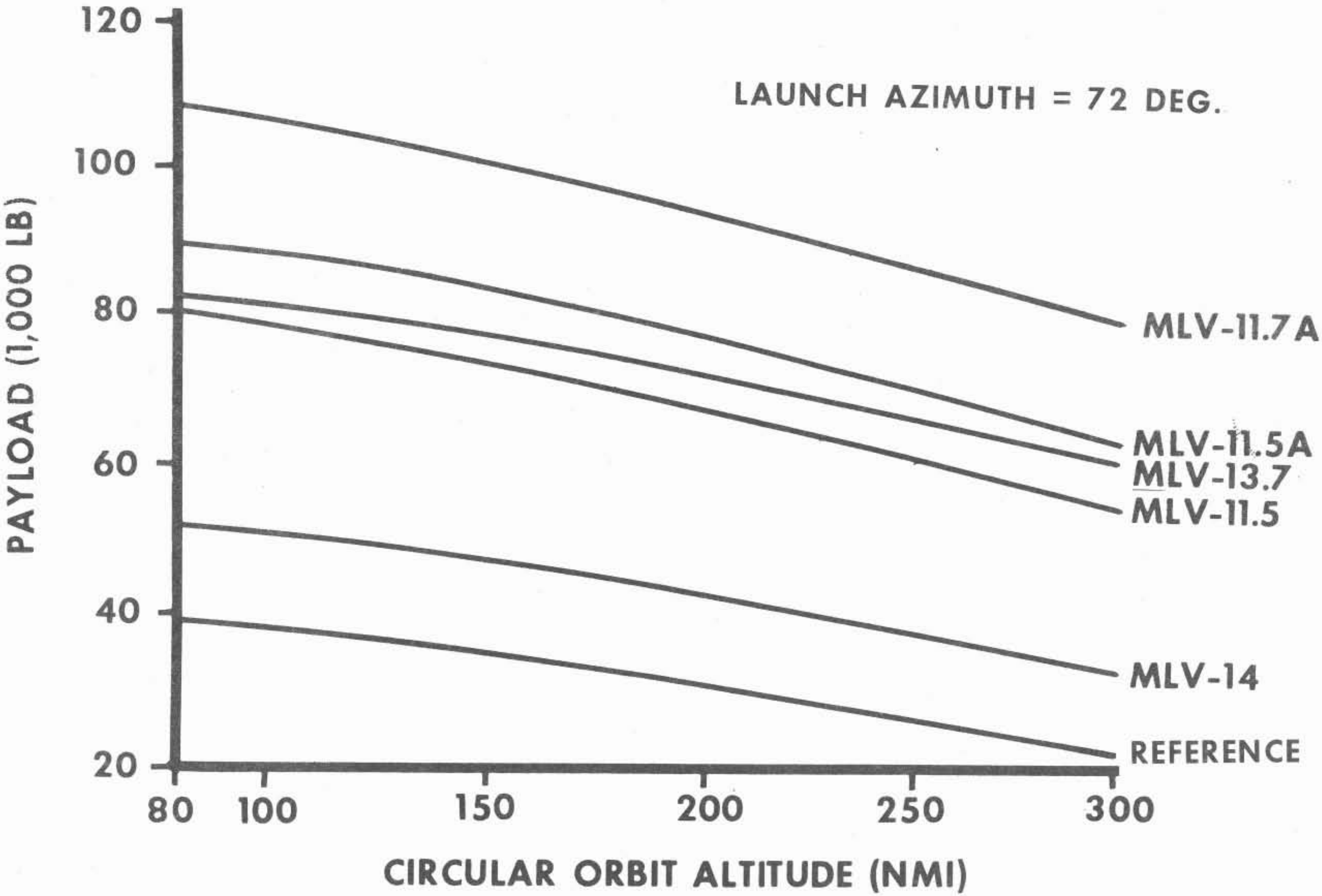


FIG 12

GENERAL SPACE STATION APPLICATIONS

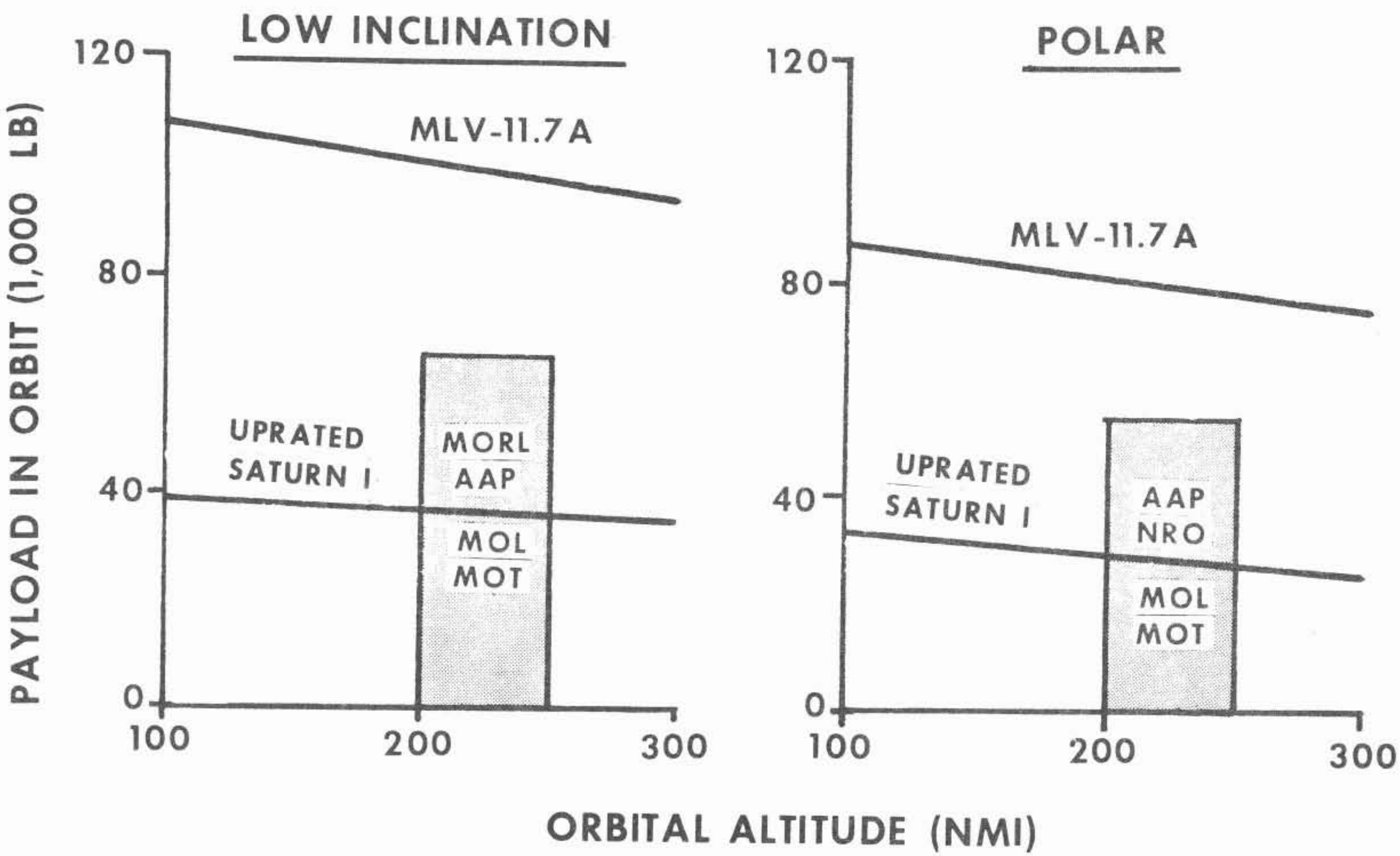


FIG 13

S-IVB ORBITAL WORKSHOP

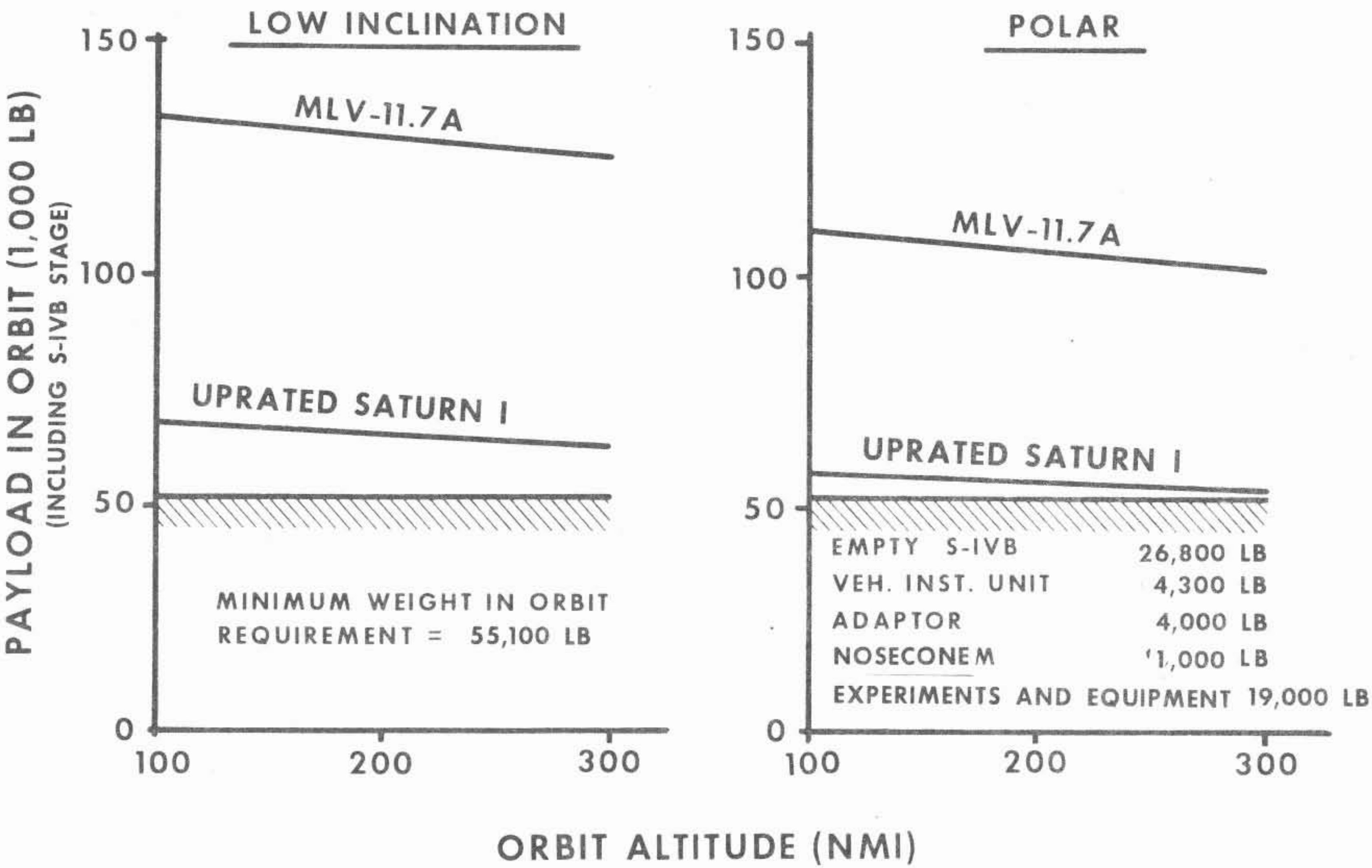


FIG 14

PAYLOAD VARIATION WITH ENERGY PARAMETER

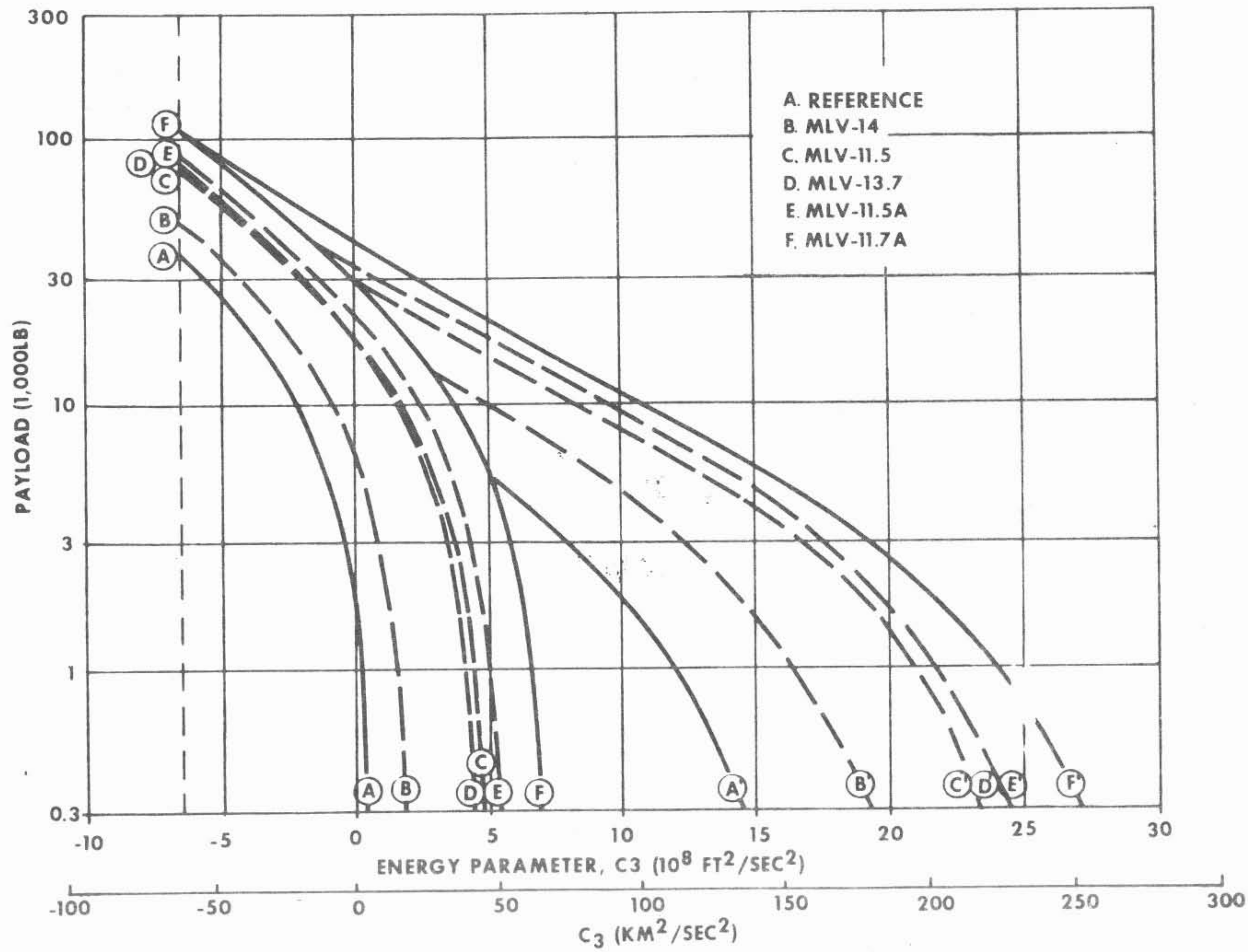


FIG 15

HIGH-ENERGY PERFORMANCE
SUMMARY WITH CENTAUR THIRD STAGE

PAYLOAD (LB) / CONFIG-URATION	ESCAPE 0 KM^2/SEC^2	VOYAGER 25 KM^2/SEC^2	JUPITER FLYBY 90 KM^2/SEC^2	SOLAR ESCAPE 160 KM^2/SEC^2
MLV-11.5	30,200	21,000	8,500	2,500
MLV-11.5A	33,900	23,000	9,500	3,000
MLV-11.7A	40,200	28,000	11,000	4,500
MLV-13.7	30,700	21,000	8,500	2,500
MLV-14	20,300	14,000	5,000	500

FIG 16

SYNCHRONOUS ORBIT PERFORMANCE WITHOUT CENTAUR (3 STARTS ON S-IVB)

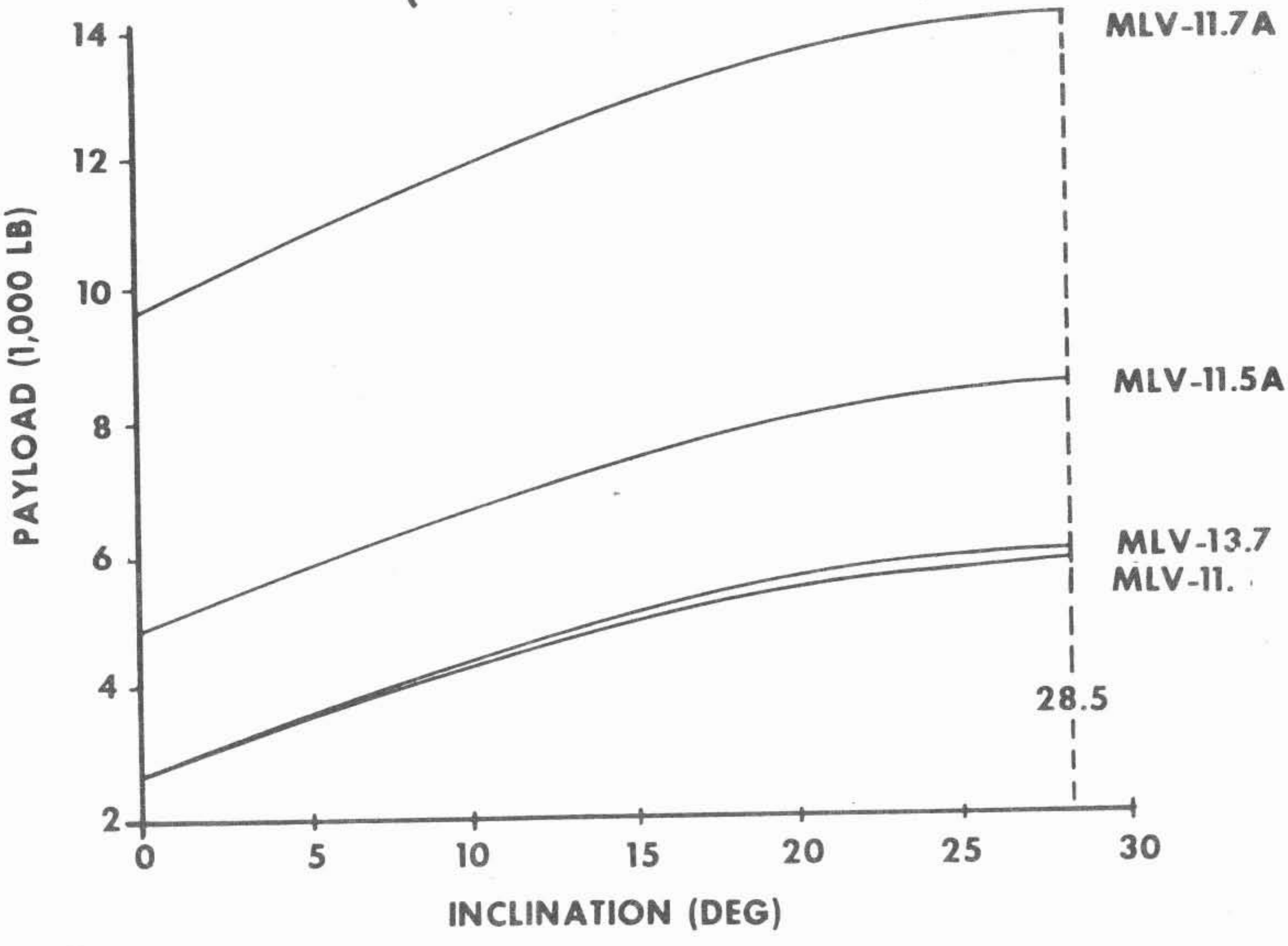
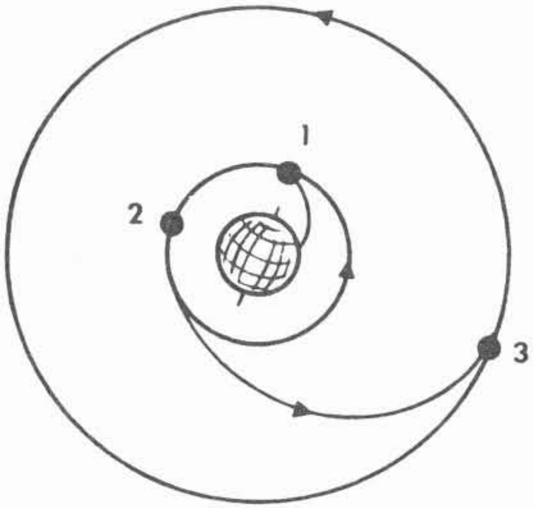


FIG 17

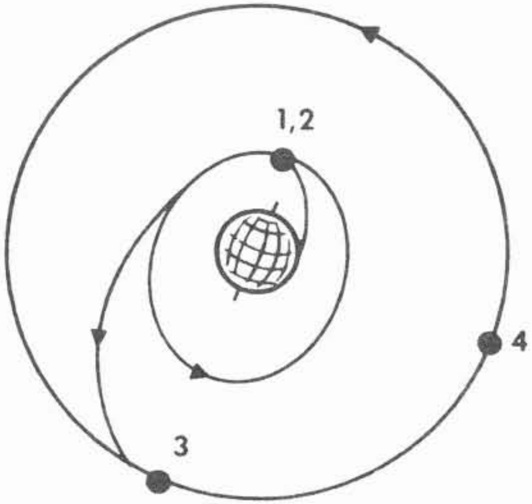
SYNCHRONOUS ORBIT MISSION PROFILES (MLV WITH CENTAUR)

PROFILE A



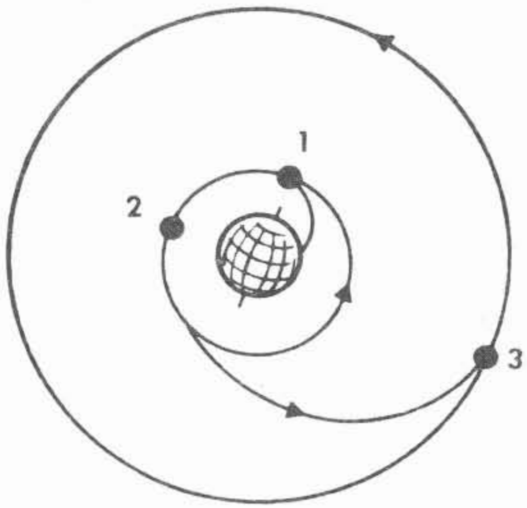
1. INJECT INTO C.P.O. (S-IVB BURNOUT)
2. INJECT INTO T.O. AT NODAL CROSSING (CENTAUR FIRST BURN)
3. CIRCULARIZE AND CHANGE PLANE (CENTAUR SECOND BURN)

PROFILE B



1. INJECT INTO E.P.O. (S-IVB BURNOUT)
2. INJECT INTO T.O. AT P.O. PERIGEE (CENTAUR FIRST BURN)
3. CIRCULARIZATION (CENTAUR SECOND BURN)
4. CHANGE PLANE AT NODE (CENTAUR THIRD BURN)

PROFILE C



1. INJECT INTO C.P.O. (S-IVB FIRST BURN)
2. INJECT INTO T.O. AT NODAL CROSSING (S-IVB SECOND BURN, CENTAUR FIRST BURN)
3. CIRCULARIZE AND CHANGE PLANE (CENTAUR SECOND BURN)

FIG 18

SYNCHRONOUS ORBIT PERFORMANCE WITH CENTAUR

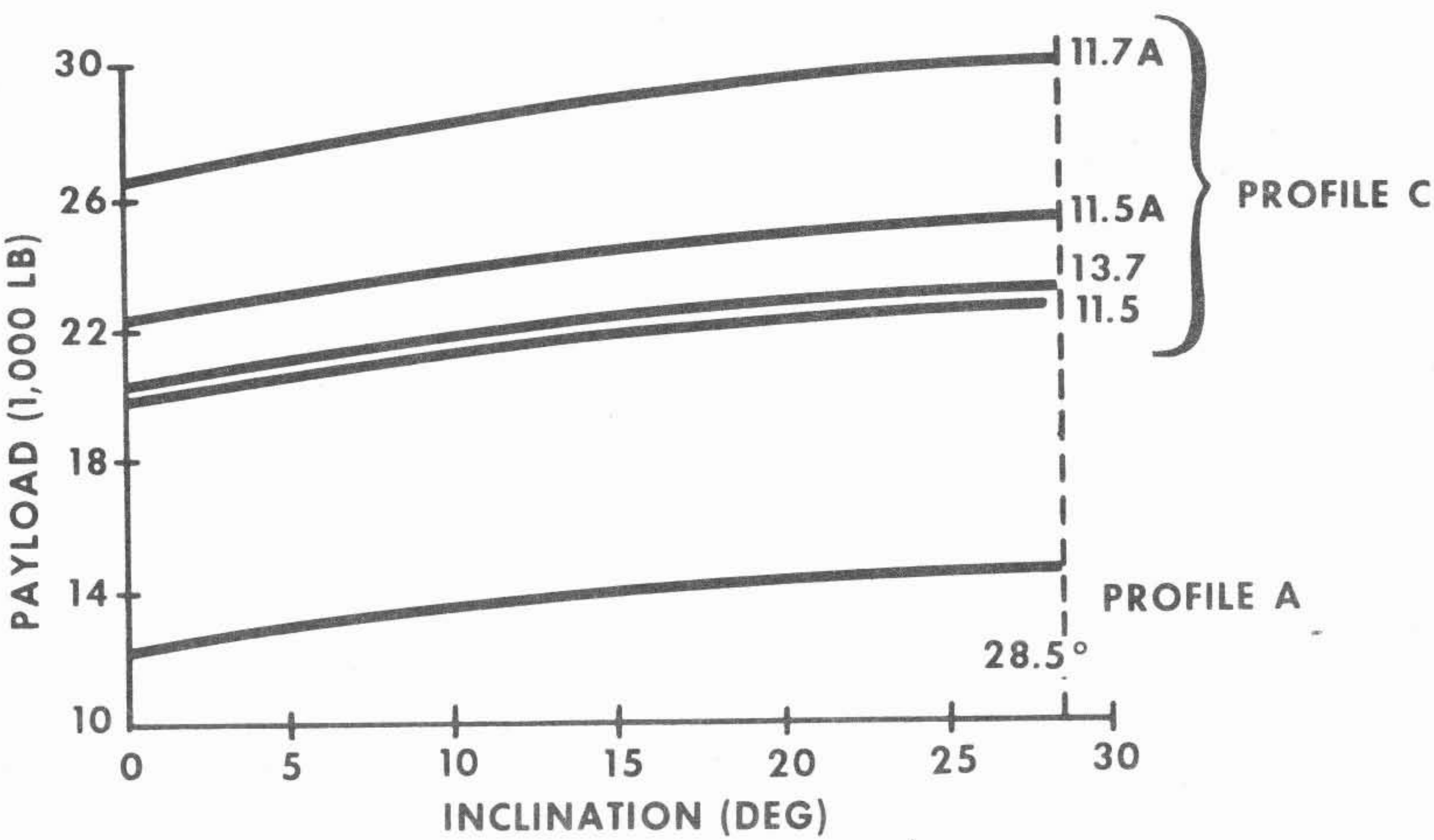


FIG 19

VOYAGER

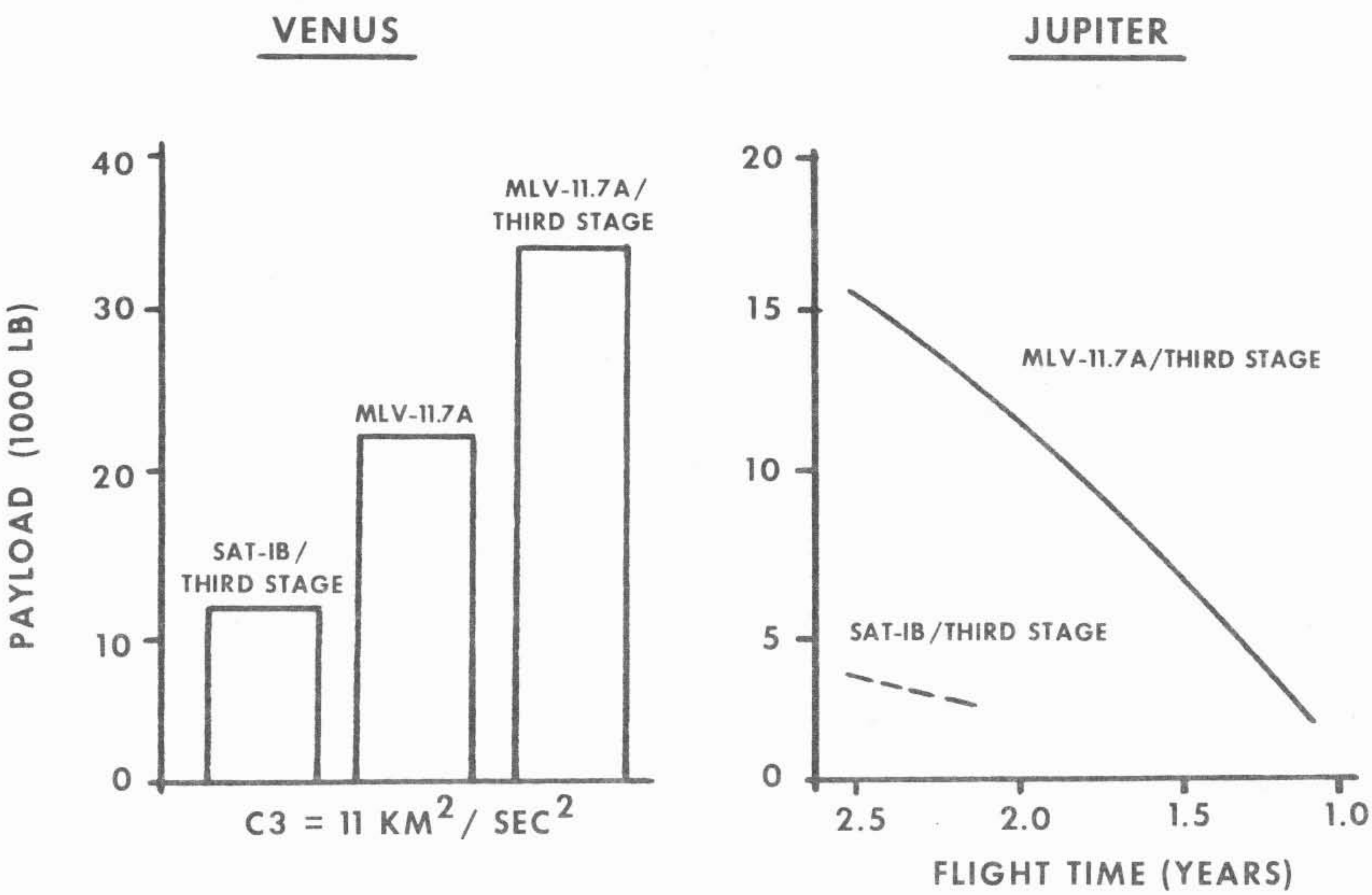


FIG 20

UPRATED SATURN I FAMILY PERFORMANCE WITH NUCLEAR UPPER STAGE

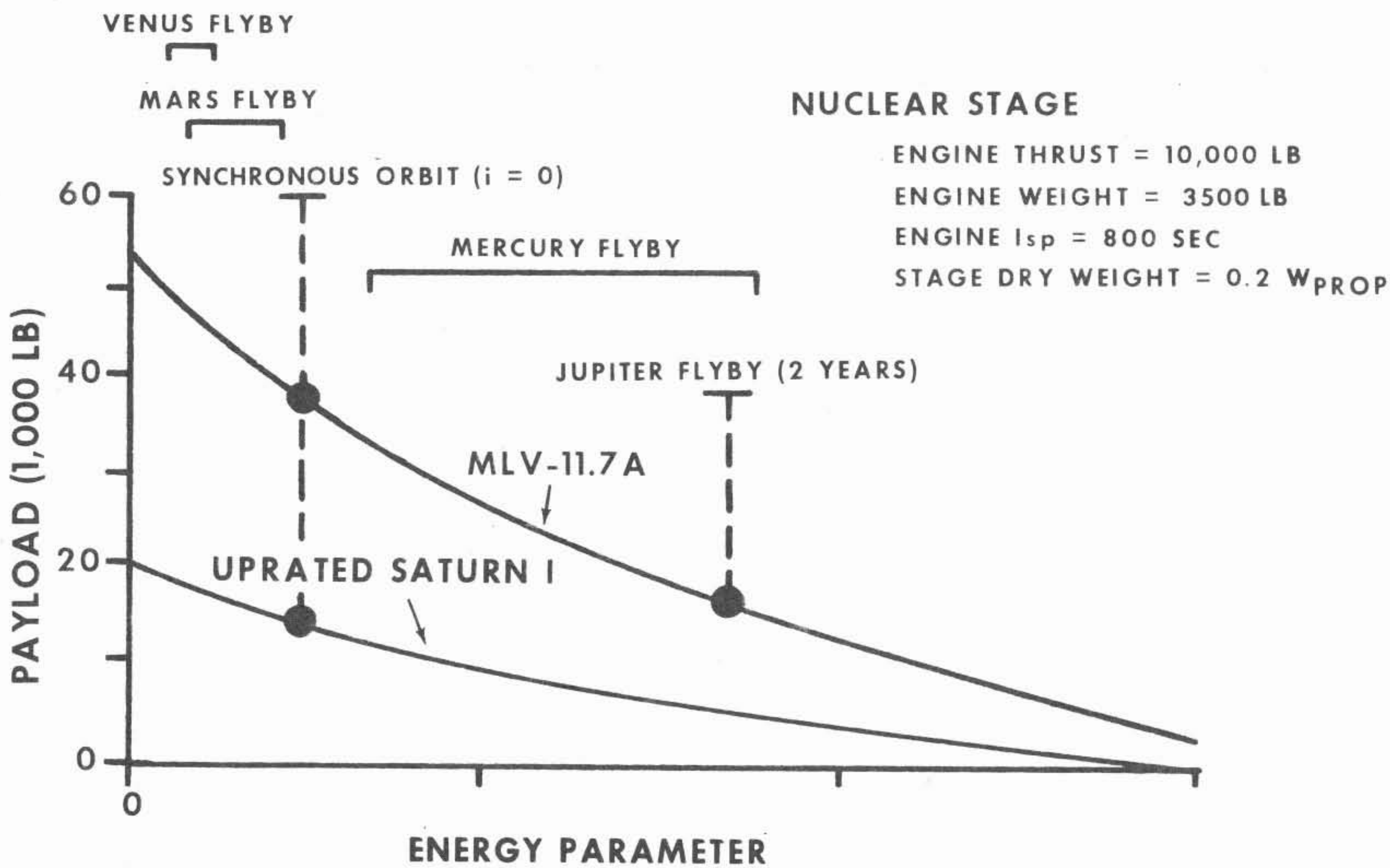


FIG 21

HIGH ENERGY PERFORMANCE
MLV-11.7A WITH UPPER STAGES

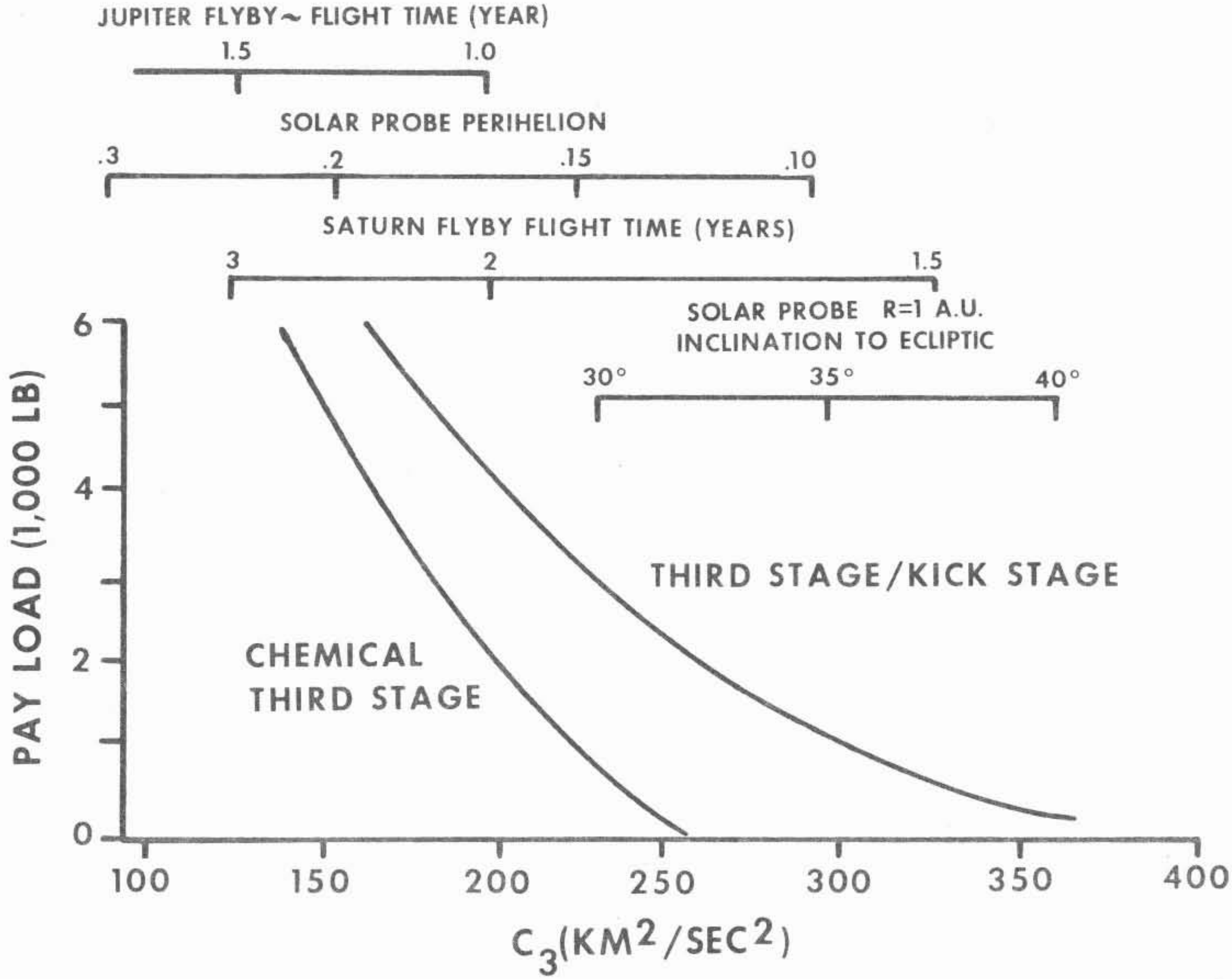


FIG 22

GUIDANCE AND CONTROL

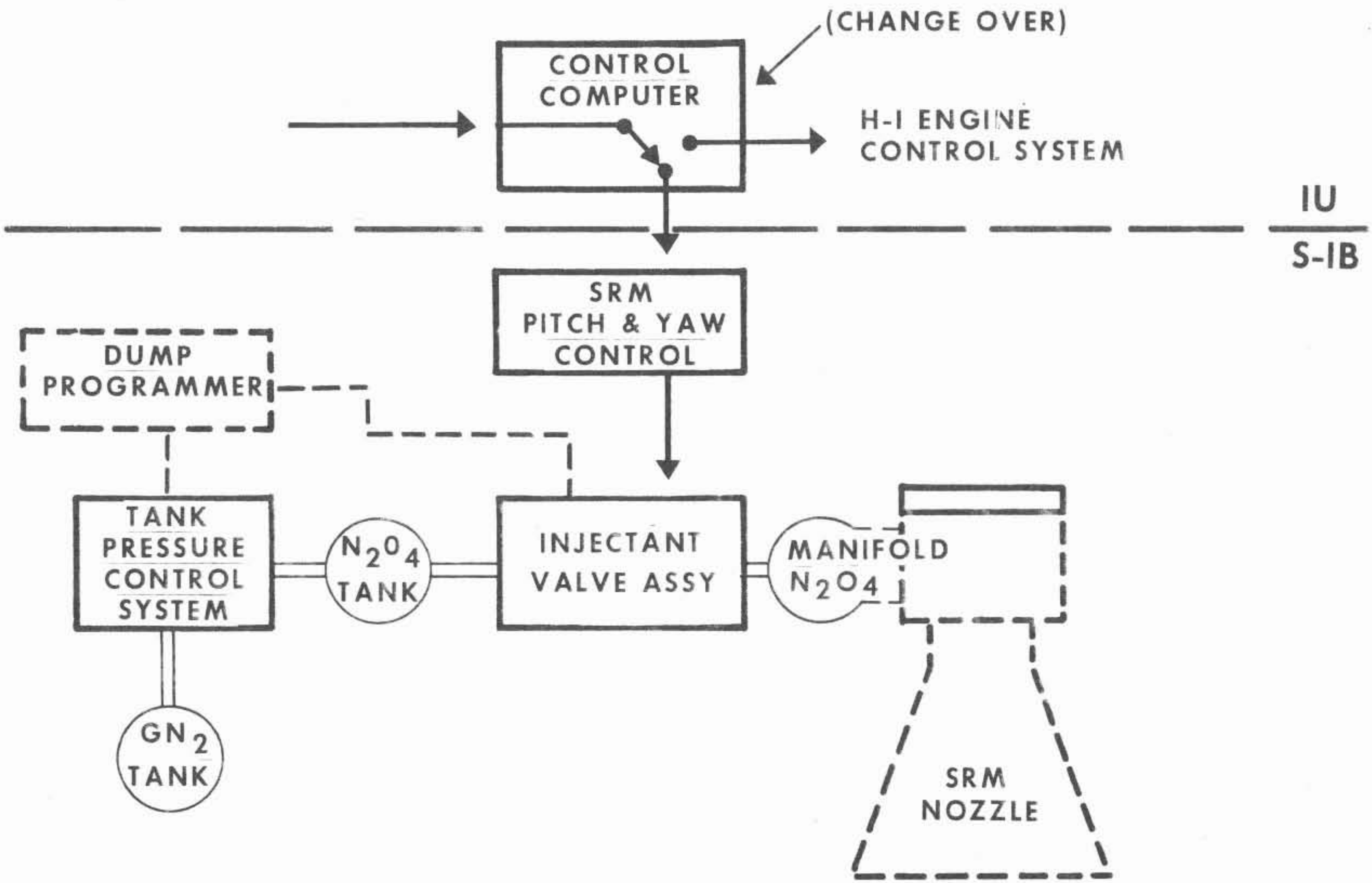
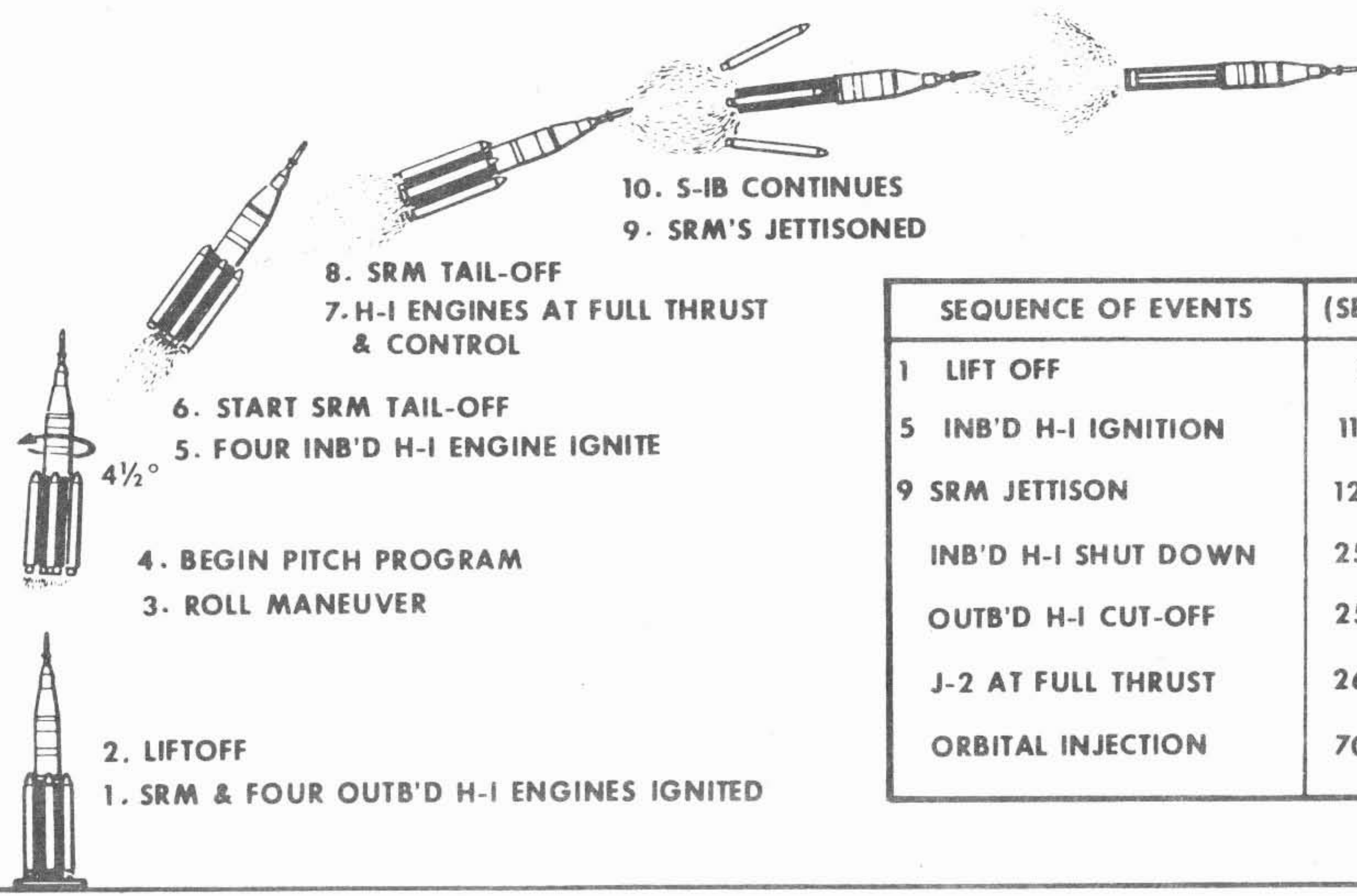


FIG 23

MLV-11.7A OPERATION



SEQUENCE OF EVENTS	(SEC)
1 LIFT OFF	0
5 INB'D H-I IGNITION	115
9 SRM JETTISON	122
INB'D H-I SHUT DOWN	251
OUTB'D H-I CUT-OFF	257
J-2 AT FULL THRUST	263
ORBITAL INJECTION	705

FIG 24

PROGRAM OPTIONS

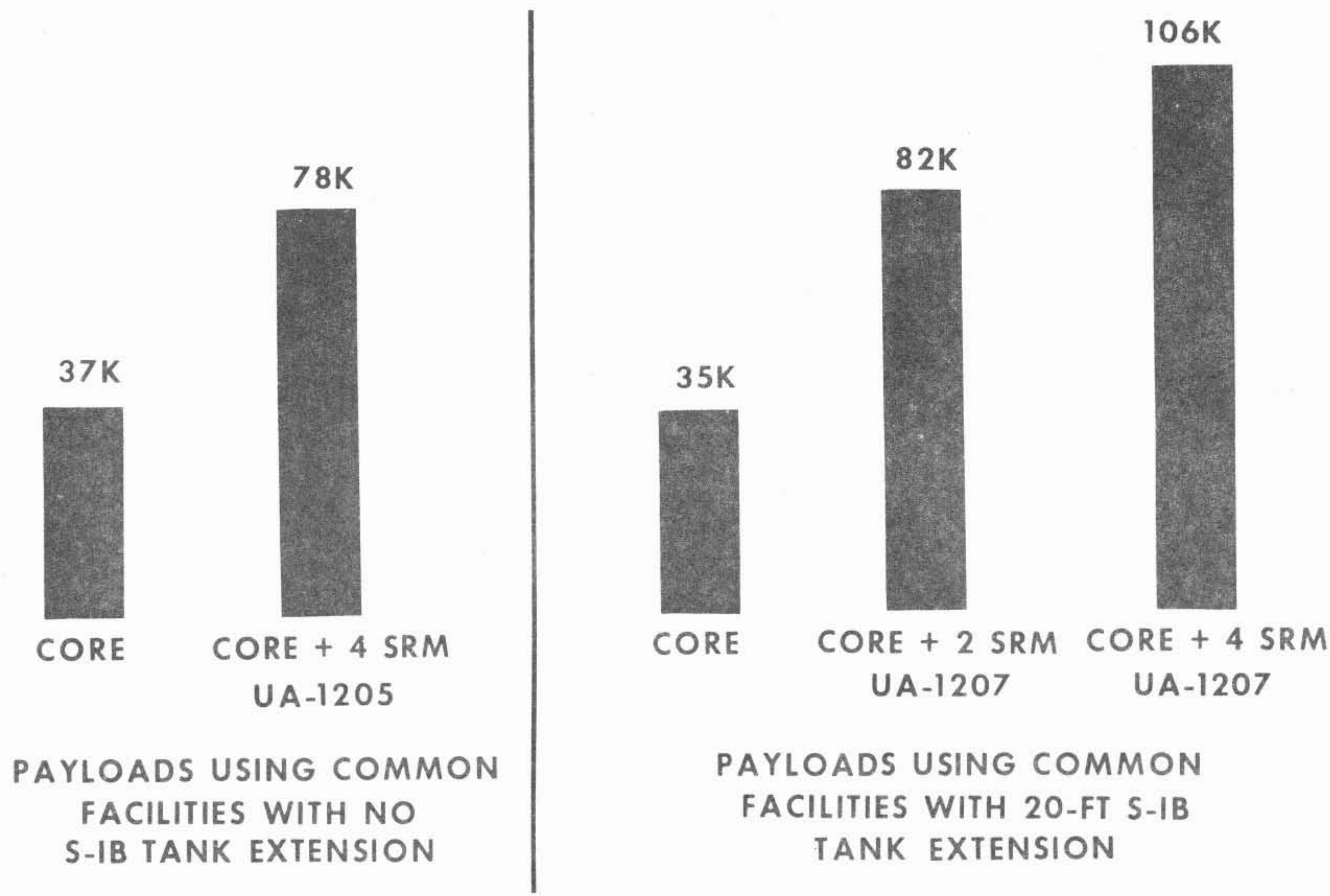


FIG 25