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NASA Contract NAS 5-302

PROJECT APOLLO

A Feasibility Study of an Advanced Manned Spacecraft and System

FINAL REPORT

VOLUME IV. ON-BOARD PROPULSION

Book 2 — Appendix P-A

[redacted]

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Prepared for:

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

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III. REQUIREMENTS

A. GENERAL

Basic references for the final requirements used in the design of the Apollo onboard propulsion system are General-Electric Work Statement 730-A-12 and a letter, dated 7 March 1961, to T.P. Browne of Aerojet-General from A.D. Cohen of General Electric.

The onboard propulsion system is designed for a 14-day mission. A propellant reserve of 10% of the total liquid propellant is carried. The propulsion system envelope is as described in the letter to T.P. Browne from A.D. Cohen.

Two vehicles are considered: first, a direct re-entry vehicle and second, a glide re-entry vehicle. The glider propulsion requirements are considered to be the same as for the ballistic vehicle, except for launch-abort escape requirements.

Two versions of the direct re-entry vehicle are considered: (1) a 1963 version with a gross weight of 15,715 lb plus abort and separation motors and (2) a 1966 version with gross weight of 14,715 lb plus abort and separation motors. Actually, the latter should be considered a growth version of the 1963 vehicle and not a different unit. The weight reductions will be made in the non-propulsive as well as the propulsive components.

B. LAUNCH-ABORT ESCAPE PROPULSION

The requirements for the launch-abort escape subsystem for the direct re-entry vehicle are given by the following set of parameters:

III, B, Launch-Abort Escape Propulsion (cont.)

Aborted vehicle weight (main propulsion module is jettisoned)		7,000 lb (1963) 6,500 lb (1966)
Initial abort thrust-to-weight ratio (in direction of thrust)		20:1
Thrust-rise rate (maximum)		300 g/sec
Burning time (Nominal)	6 Units	1.0 sec
	and	
	2 Units	2.0 sec
	or	
	8 Units	2.0 sec
Units jettisoned at booster first-stage burnout		4
Units jettisoned at booster second-stage burnout		2
Units jettisoned at escape velocity		2
Mounting angle of motors		25 degrees to vehicle centerline
Net thrust vector during first second of firing		15 degrees off vehicle centerline through the center of gravity

The thrust level for all abort units should be approximately the same, although a thrust deviation on the order of 10% between the 1.0-sec and 2.0-sec burning-time units can be tolerated. Optimum performance should be achieved for sea-level conditions.

In the design of the escape-propulsion subsystem, the primary consideration is reliability. Weight is not as critical, because the units are dropped when they are no longer necessary.

III, B, Launch-Abort Escape Propulsion (cont.)

Launch-abort escape subsystem requirements for the glide re-entry vehicle are presented in Appendix A, along with the discussion of the subsystem design.

C. SEPARATION ROCKETS FOR THE DIRECT RE-ENTRY VEHICLE

The separation rockets separate the re-entry vehicle from the spacecraft shell prior to re-entry. Two different requirements exist: first, separation at high dynamic pressure and second, separation outside of the atmosphere.

Requirements for separation at high dynamic pressures are:

Total vehicle weight	6,000 lb
Re-entry vehicle weight	4,000 lb
Total drag-force on spacecraft shell	33,000 lb
Separation distance	20 ft in 1 sec

For separation outside the atmosphere, prior to re-entry, sufficient force to achieve positive separation is required.

D. ATTITUDE CONTROL

Total impulse	60,000 lb-sec
Number of starts	3,000
Maximum single total impulse	200 lb-sec
Number of units	12
Thrust per unit	3 lb

III, Requirements (cont.)

E. COURSE CORRECTION--OUTBOUND

Velocity increment	250 fps
Minimum acceleration	0.1 g
Maximum acceleration	1.5 g
Number of starts (maximum)	5

F. LUNAR-ORBIT ENTRY

Velocity increment	3,500 fps
Minimum acceleration	0.25 g
Maximum acceleration	1.5 g
Number of starts (maximum)	5

G. LUNAR-ORBIT EXIT

Velocity increment	3,500 fps
Minimum acceleration	0.3 g
Maximum acceleration	2.0 g
Number of starts (maximum)	2

H. COURSE CORRECTION--INBOUND

Velocity increment	250 fps
Minimum acceleration	0.1 g
Maximum acceleration	3.0 g
Number of starts (maximum)	5

III, Requirements (cont.)

I. RE-ENTRY SPIN CONTROL JETS

These units generate a moment around the re-entry vehicle longitudinal axis.

Total impulse	7,000 lb-sec
Number of starts	Many
Number of units	4
Thrust per unit	18 lb

IV. SUBSYSTEM DESIGN

The various liquid propellant subsystems integrated in the main propulsion module are shown in Figure IV-1. The location and mounting of the solid propellant motors used for launch abort and separation are shown in Figure IV-2.

A. LAUNCH ABORT ESCAPE AND SEPARATION SUBSYSTEMS

1. Considerations

Solid propellant motors were selected for this subsystem. The main criteria for this selection were the requirements for short response time, very high thrust level, and short duration--while maintaining high reliability. These requirements, and the fact that very high specific impulse was not essential, led to the conclusion that solid rockets are best suited to this application.

In the design of the launch-abort escape and separation subsystems, only the requirements for 1963 were considered. The 1966 version of the Apollo vehicle may require an escape system with a different thrust level. Since weight is not of prime importance--at least in the early portion of the trajectory--during first and second stage boost, requirements for 1966 can best be met with the application of more or fewer units of 1963 design.

2. Propellant

The high reliability standards set for this system mandate a propellant of proved performance and aging stability. Propellant formulation ANP-2913 CD meets these requirements. Its excellent mechanical and casting and curing properties, coupled with a measured specific impulse at sea level of 247 lbf-sec/lbm, make it a logical choice for use in an escape motor.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

This propellant--formulated with 68% ammonium perchlorate, 16% aluminum, 0.30% ballistic additive, and 15.70% polyurethane binder--has been widely tested and is completely developed. The burning rate for ANP-2913 CD, at the present time, is 0.54 in./sec, but the propellant can be modified to provide burning rates over the range of 0.3 to 0.7 in./sec at a 1,000 psi chamber pressure. This range of burning rates adequately covers the requirements for the escape and separation rockets. ANP-2913 CD is now being qualified over a temperature range of -65°F to +165°F.

To produce a lighter casing and to reduce motor development cost, however, a temperature cycling requirement of 70°F ± 40°F is suggested as more consonant with the operational requirements for the escape and separation motors (1KS-23,800, 2KS-23,800, and 1.9KS-18,100 motors).

Accelerated aging studies indicate the aging stability of formulation ANP-2913 CD to be very good. Propellants of this type have been successfully aged for six months at 180°F. Correlation of data for aging a similar propellant, ANP-2639-AF, at 180°F and at ambient temperature indicates a storage life of three to five years for ANP-2913 CD at ambient temperatures.

The autoignition temperature is 510°F.

Based on more than 100 static test firings, the motor-to-motor variation in total impulse among motors cast from the same propellant batch is ± 0.6%. The thrust deviation under the same conditions is ± 1.2%. Maximum variation in total impulse and thrust among motors cast from different batches is ± 2% and ± 4%, respectively. This variation can be cut in half by more rigid manufacturing control.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

3. Igniter Design

For the ignition of the propellant grain, the use of Alclojet igniters is proposed. The Alclojet igniter offers several advantages when compared with the usual basket-type igniter.

In the Alclojet igniter, the Alclo pellet is enclosed in a chamber that is built to withstand a combustion pressure of 11,000 psi for 0.25 millisec. The combustion products are directed through nozzles at high velocity toward the propellant surface. This confinement of the charge results in smooth, rapid burning. Heat from the igniter is delivered at a maximum rate at the beginning of the motor ignition transient and regresses before steady-state motor pressure is reached. Therefore, ignition pressure peaks are reduced to values of 300 to 475 psi. Ignition delay is short, 6 to 10 millisec, as compared with 16 to 150 millisec for basket igniters. The igniter is self-pressurizing, which is an important feature in attaining reliable ignition at high altitude.

Because the charge weighs about two-thirds as much as the charge in a basket-type igniter, the Alclojet igniter is smaller and lighter than the basket-type igniter. The unit is designed to maintain its mechanical integrity in short-duration motors (approx. 2 sec). No debris is expelled through or against the nozzle.

The Alclojet igniter is being used at present in the following systems: Eagle (booster and terminal), improved Tartar, Skybolt (first and second stage), improved Genie, XM59, Asp, Army Drone Booster, and Titan gas generator. In its development and application to highly diverse systems, enough data have been collected to qualitatively establish reliability. The quantitative determination of reliability will depend upon developmental and qualification testing of the specific design chosen for use in Apollo motors.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

The igniter charge is initiated by an electrically-fired squib. Low-tension and low-current squibs or high-tension exploding bridgewire squibs can be used.

Typical values for low-tension squibs are a maximum no-fire current of 0.25 amps and a 100% fire current of 0.5 amps. For a short function time, however, higher currents are used.

Reliable exploding bridgewire squibs have been developed with a 1,000-v maximum no-fire and 1,500-v minimum all-fire voltages. Typical ignition function times varies between 0.1 millisec at 3,000 v and 1 millisec at 2,200 v. The exploding bridgewire concept eliminates accidental ignition from static discharge, high-energy radio-frequency fields, and low voltage potentials. Firing times can be made faster and more consistent. The power requirements for one power-supply unit for eight igniters are approximately 28 v dc, 1.5 amp surge, and 750 milliamps steady within 30 sec (standby).

Since adequate electrical power will be available, the exploding bridgewire squib is recommended for use in the Apollo solid rocket motors to ensure against ignition by stray currents and to provide short and consistent-firing times. The alternative to this would be a low-tension squib and a safety-and-arming device which puts a mechanical barrier between the squib and igniter charge. Because a two-step action is necessary for ignition with the alternative system, it is less reliable for the particular conditions of the Apollo mission, where a malfunctioning squib or safety-and-arming device cannot be replaced.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

4. <u>1KS-23, 800 Escape and Separation Motor</u> (Figure IV-3)

Requirements for the launch-abort escape propulsion subsystem call for two motors with 2-sec burning times and six motors with 1-sec burning times. The necessary thrust is 23, 800 lb per motor.

Requirements for the two motors used to separate the re-entry vehicle from the rest of the spacecraft while it is in the atmosphere are similar to those for the 1 sec duration escape motor. Therefore, the 1KS-23, 800 motor is used for both escape and separation purposes.

The 1KS-23, 800 motor contains 102 lb of propellant. The motor weighs 126.5 lb, and the resulting mass fraction is 0.807. A dendrite configuration for the burning surface of the case-bonded propellant grain has been selected; it combines relatively high volumetric loading (68%) with a high-burning area for this short-duration motor.

A small amount of ANP-2319 CD propellant is case in the aft head, and the joint between chamber propellant and aft-head propellant is restricted from burning at the interface. A thrust-versus-time curve for the motor is shown in Figure IV-4.

For the motor case, high-strength nickel steel (25% Ni) is used. This material has high strength and a high elastic modulus of 30 x 10^6 psi. The yield strength is 260, 000 psi and the ultimate strength is 285, 000 psi. Minimum strengths of 225, 000 psi at weld points have been adequately demonstrated. In this specific application, homogenous steel cases are competitive with filament-wound cases because the weight advantage of the fiber-glass case is not realized if throat areas are required.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

A chamber wall thickness of 0.030 in. has been determined for this motor. This thickness is based on a designed yield pressure of 1.42 times the maximum expected operation pressure (MEOP) of 1,200 psi. This value, which is recommended by Aerojet-General, is higher than the normally-used safety factor of 1.25 MEOP for solid propellant missile motors used on manned systems such as Skybolt and Genie. The safety factor used on unmanned vehicles, such as Minuteman is 1.1 MEOP.

A shear-pin joint of full case diameter is provided at the aft closure in this motor design to allow access for core removal from the propellant grain after casting. This type of joint has been selected because it is two to three times lighter than a bolter joint, and it has demonstrated excellent reliability on Aerojet-General-produced Eagle, Sparrow, and Tartar motors. The heavy structure at the shear-pin joint can efficiently be used for motor-attachment fittings.

The average chamber pressure for this motor is 1,130 psi. Selection of this pressure is based on the tradeoff between nozzle weight and case weight.

Internal motor case insulation has been omitted from this motor because heat transfer calculations show that the temperature rise in the bare metal case is negligible for a 1-sec duration.

The nozzle consists of a 0.4 in.-thick, silica-reinforced, phenolic throat insert backed up by a 0.030 in.-thick-steel shell. The shell also forms the nozzle attachment flange. The silica-phenolic exit cone is wrapped on and varies in thickness from 0.4 in. downstream of the throat to 0.125 in. at

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

the exit plane. Selection of these thicknesses is based on structural requirements only. Since erosion and charring of the exit cone is negligible for shor firing durations, no extra thickness is included to compensate for these effects. Net ejection forces on the exit cone are forward.

The phenolic throat insert will erode, resulting in a throat area increase up to a maximum of 3% during the 1-sec firing duration of this motor. This area change is based on an erosion rate of 0.05 in./sec and is responsible for a loss of only 0.2% of available motor impulse.

A nozzle expansion ratio of 9:1 has been selected for optimum motor performance at sea level, since the most severe operating conditions are at sea level. The thrust coefficient at sea level for this nozzle is 1.59.

Two Alclojet igniters are used in this motor to provide a redundant system for ignition reliability. Each igniter charge weighs 26 grams, and the total weight of each igniter is 0.390 lb. Both igniters will receive an ignition signal and will be joined by a flame tube to assure that they are both ignited. The resultant chamber pressure will be controlled to stay below the nominal case design pressure.

The dual igniter system is still under investigation. This continued study may indicate that a single igniter with dual squibs would make an equally reliable unit with less weight and complexity.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

5. 2KS-23, 800 Escape Motor (Alternative Separation Motor)

The 2KS-23, 800 escape motors shown in Figure IV-5 are similar in design to the 1-sec duration motor. Two of the 2KS-23, 800 motors are provided to supply thrust along the axis of the spacecraft after burnout of the 1-sec duration motors.

In case a maximum altitude of 5, 000 feet from the launching pad is desired, the six 1KS-23, 800 escape motors are replaced by six 2KS-23, 800 motors. The escape system is then composed of eight 2KS-23, 800 motors. If eight 2KS-23, 800 motors are used in the abort system, two 2KS-23, 800 motors should be used for separation at high dynamic pressure, to minimize the number of unique types of motors required. This arrangement is illustrated in Figure IV-2. The weight penalty resulting from this "over-powering" is not great, since the motors are jettisoned early in the launch trajectory.

As is shown in Table IV-1, the propellant for this motor weighs 210 lb, and the total motor weighs 242.9 lb. The motor mass fraction is 0.864; total impulse is 51, 500 lbf-sec. Volumetric loading of the dendrite grain is 71%. Data showning thrust versus time for sea level and vacuum conditions are presented in Figure IV-6.

The chamber wall of this motor is 0.041 in. thick, and the average chamber pressure is 1, 130 psi.

Internal motor case insulation is a 0.030-in. thick layer of silica-filled nitrile rubber. Heat transfer calculations show that this is sufficient to limit the temperature rise in the metal case due to heat influx during combustion to a negligible amount.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

The basic nozzle is the same as was used with the 1KS-23, 800 motor. A 0.6% loss of available motor impulse is expected because throat area will increase a maximum of 7% due to erosion.

The charge in each of the two Alclojet igniters used in this motor weighs 45 grams. Each igniter weighs a total of 0.582 lb.

6. 2.5KS-560 Separation Motor

For the separation of the re-entry vehicle from the spacecraft, two 2.5KS-560 motors are used as shown in Figure IV-7. The motor performance curve is plotted in Figure IV-8, and weights and performance are tabulated in Table IV-1.

Since these motors are carried along during the entire mission, that they be light in weight is important. A glass-fiber-resin composite case is wrapped around the propellant grain, with premodled insulation and nozzle in place. The need for mechanical joints is thus eliminated, and the lower weight of inert parts is achieved. This manufacturing technique is described in Ref. 5.

As previously mentioned, the propellant is qualified over a temperature range of -65°F to +165°F. Since the motors are mounted inside the skin of the spacecraft, the temperature of the motors during the mission is expected to be well within this limit.

These motors are also exposed to space environment during the fullmission time. The propellant is less sensitive to radiation than is the crew. Serious degrading effects on propellant and case caused by the hard

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

vacuum are not to be expected. The location of the motors inside the vehicle skin minimizes the dange of meteorite damage, eliminates excessive solar heat radiation, and shields the motors completely against ultraviolet radiation.

The calculation of chamber wall thickness was based upon a combined ultimate stress of 135,000 psi for the cylindrical portion of the motor case. The longitudinal ultimate stress of 186,000 psi was used to determine the thicknesses of the aft head at the aft head-to-cylindrical section knuckle. (Ultimate stress levels are quoted because glass fibers do not yield before failing.) The safety factor used is 1.42 times MEOP. The nominal chamber pressure is 500 psi.

The cylindrical wall of the separation-motor case is 0.036 in. thick, and the aft head is 0.015 in. thick at the knuckle. The grain is covered by a 0.030 in. thick premolded insulation of nitrile rubber. Temperature rise in the fiber-glass case is negligible when this amount of insulation is used.

The nozzle entrance and throat section consists of a 0.25 in. thick molded, silica-reinforced, phenolic piece which is bonded to the grain insulation before the motor case is wrapped around the grain. The exit cone section is wrapped to adjoin the throat after the motor case is completed. The thickness of the exit cone section ranges from 0.2 in. downstream of the throat to 0.125 in. at the exit plane.

This nozzle is designed for optimum performance in a vacuum and has an expansion ratio of 24:1. The expansion ratio is limited here only by the space available in the installation. The thrust coefficient is 1.725.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

The charge in the single Alclojet igniter weighs 4.76 grams; the igniter as a whole weighs 0.171 lb. The small separation motors are not fitted with a second igniter because a single motor can impart a sufficient velocity increment to the space capsule to safely separate it from the re-entry vehicle. Dual igniters would thus contribute only second-order redundancy.

7. <u>Mounting of the Escape and Separation Motors</u>

The six 1KS-23, 800 and two 2KS-23, 800 motors (alternative: 8 2KS-23, 800 motors) are mounted aft of the cylindrical section of the spacecraft, as shown in Figure IV-2. The motors are arranged so as to produce a resultant thrust vector which passes through the vehicle center of gravity at an angle of 15° to the vehicle longitudinal axis.

The two motors for separation at high dynamic pressure are attached to the nose cone of the spacecraft. They are also mounted with the net thrust vector at an angle to the vehicle axis so that the spacecraft shell will follow a trajectory after separation different from that of the re-entry vehicle, thus avoiding the possibility of collision.

One of the many possible jettisoning devices for these motors is also shown in Figure IV-2. The motors are attached by a pair of rollers at the forward end of the motor case, and a tubular piece is linked on the shear-pin flange at the aft end of the case. The rollers slide in rails built into the fairing fixed on the spacecraft. The tubular piece slides in a cylindrical guide and is held against a preloaded steel spring by a single explosive bolt. Two explosive charges are contained in the bolt to improve the reliability of jettisoning.

IV, A, Launch Abort Escape and Separation Subsystems (cont.)

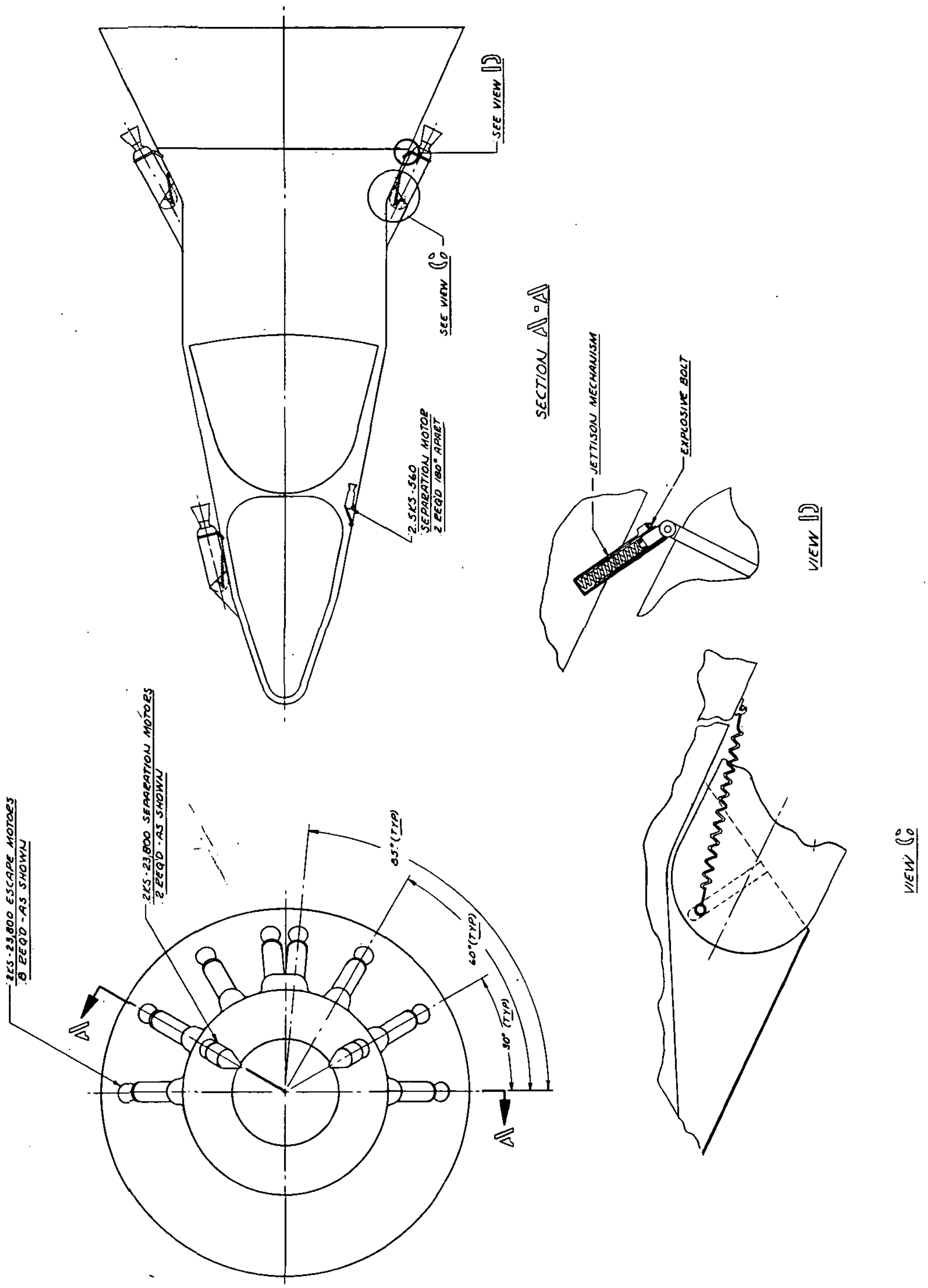
The rails and the cylindrical guide are inclined at an angle of 30° to the motor centerline. The aft mounting point carries all aerodynamic and thrust forces parallel to the motor axis. The front mounting point carries forces perpendicular to the motor axis. Aerodynamic and inertial side loads are carried by both mounting points.

Two longitudinal springs are located at both sides of each motor. The motor is jettisoned by firing the explosive bolt and allowing the energy of the spring to be imparted to the motor. The inclination of the forward and aft guide rails forces the motor sideways so that it will not collide with the aft conical skirt of the space vehicle.

The jettisoning devices, capable of imparting a side acceleration of 2 g to the escape motor, weigh 3.4 lb and 3.7 lb for the 1- and 2-sec motors, respectively. These figures are based on an assumed dynamic pressure of 120 lb/sq ft acting on the motor.

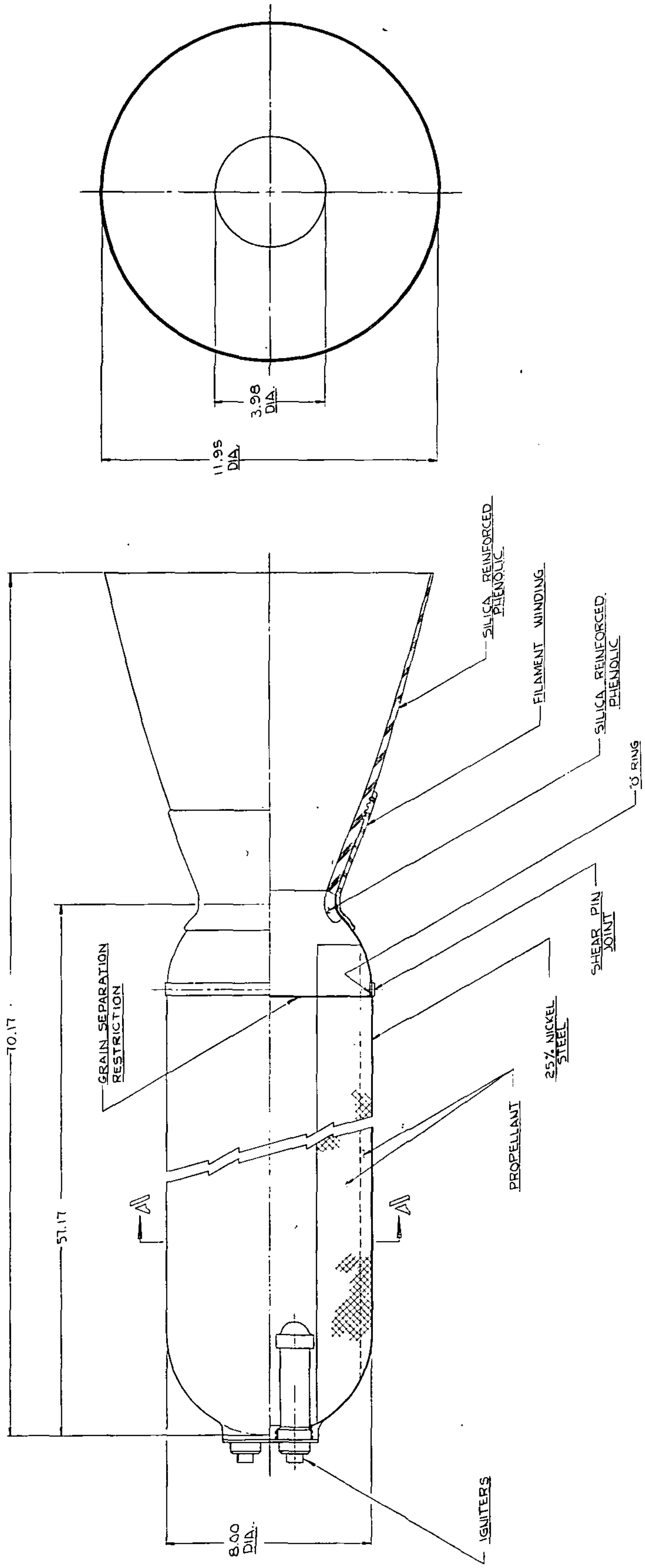
Each of the two 2.5KS-560 motors is provided with a wrapped-on skirt, which is rigidly attached to a mounting structure inside the spacecraft.

The motor jettisoning mechanisms are conceptual designs only and thus are not included in the cost and development schedule. If these jettisoning devices were to be an Aerojet-General responsibility, design investigation would be necessary before arriving at the final configuration.



Apollo Escape and Separation System

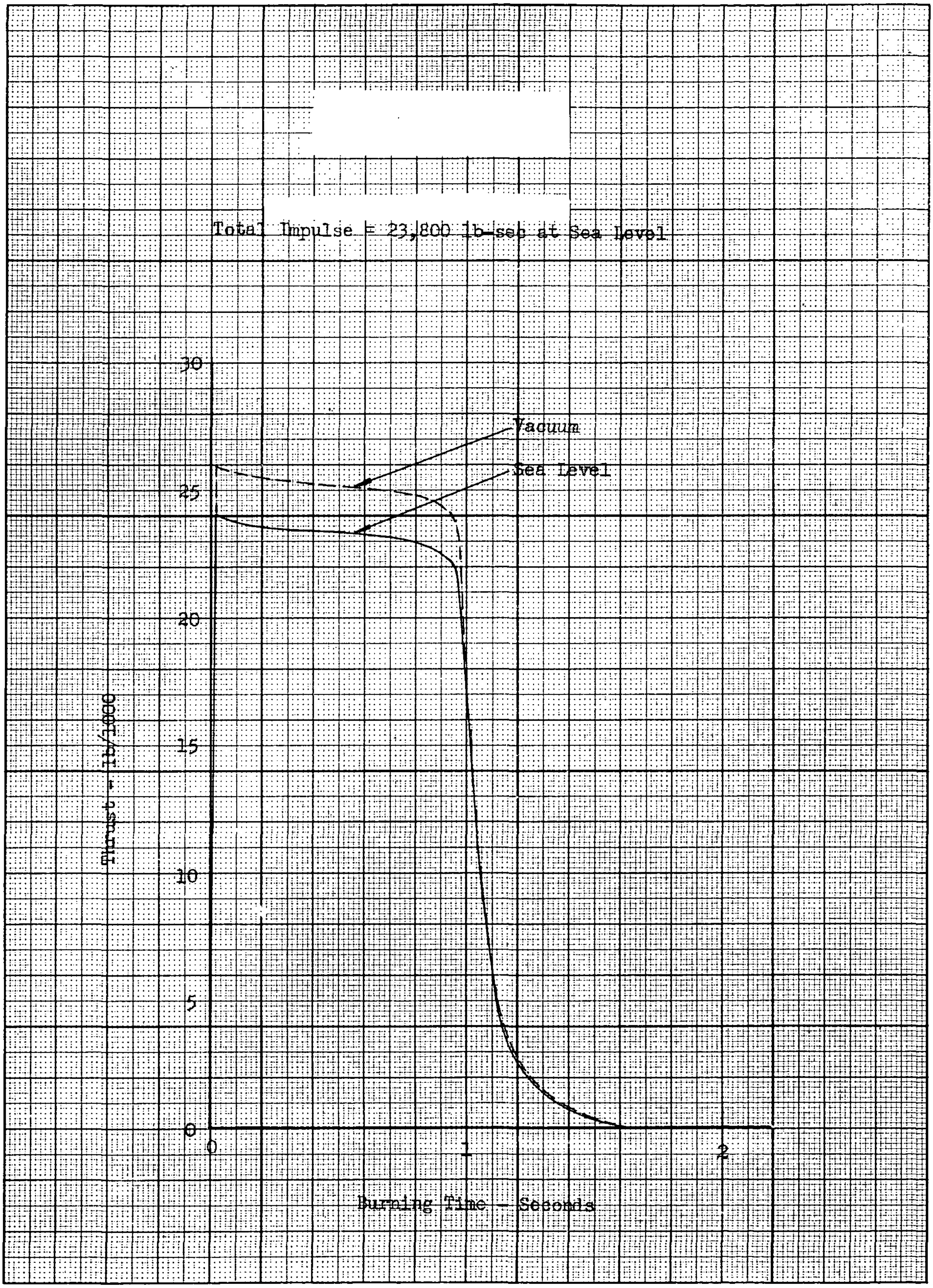
Figure IV-2



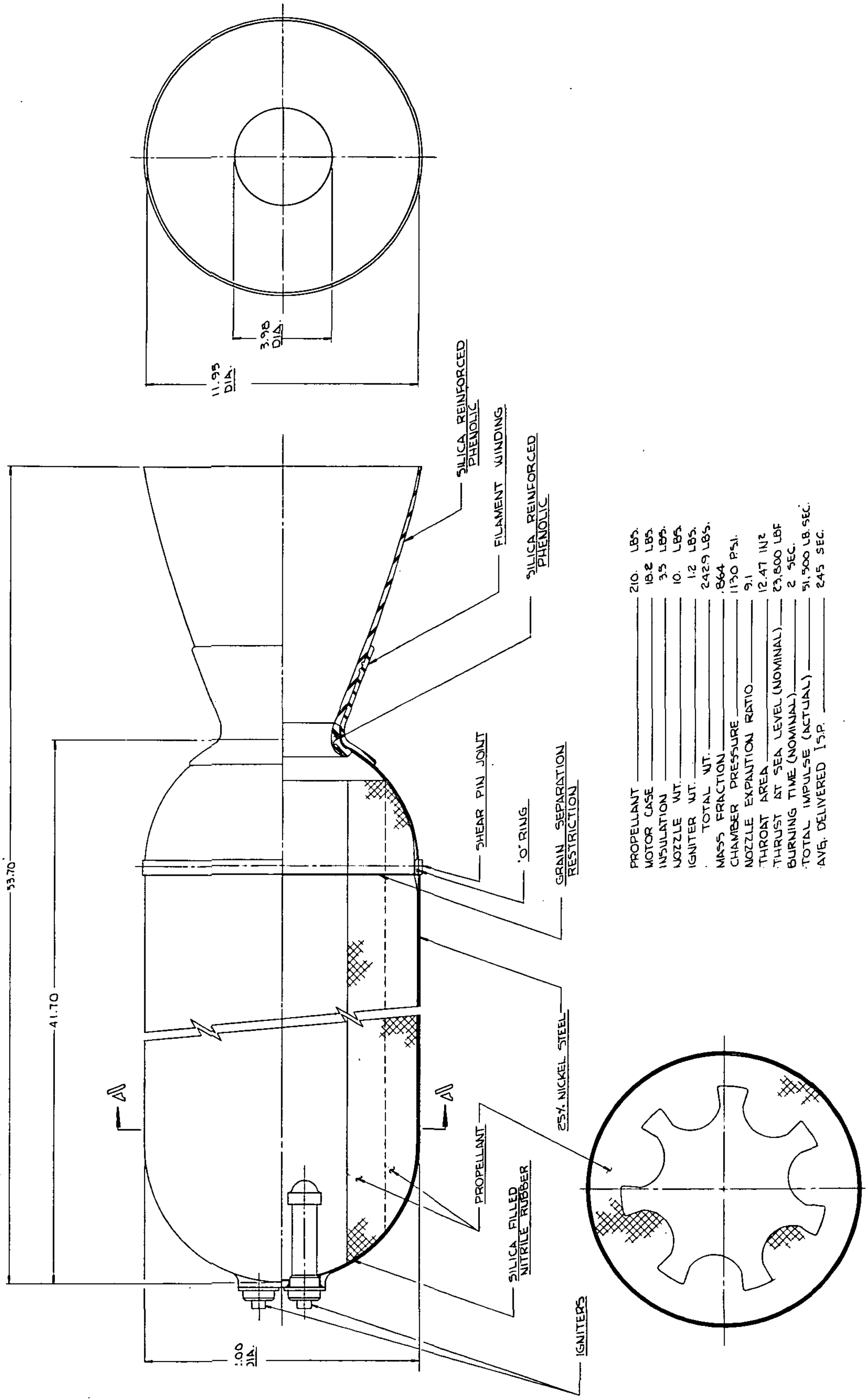
PROPELLANT WT.	102 LB.
MOTOR CASE WT.	13.6 LB.
NOZZLE WT.	10 LB.
IGNITER	.8 LB.
TOTAL WT.	126.4 LB.
MASS FRACTION	.807
CHAMBER PRESSURE	1130 P.S.I.
NOZZLE EXPANSION RATIO	9:1
THROAT AREA	12.47 IN²
THRUST AT SEA LEVEL (NOMINAL)	23,800 LB.
BURNING TIME (NOMINAL)	1.0 SEC.
TOTAL IMPULSE (ACTUAL)	25,000 LB-SEC.
AVERAGE DELIVERED Isp	245 SEC.

1KS-23,800 Escape Motor

Figure IV-3

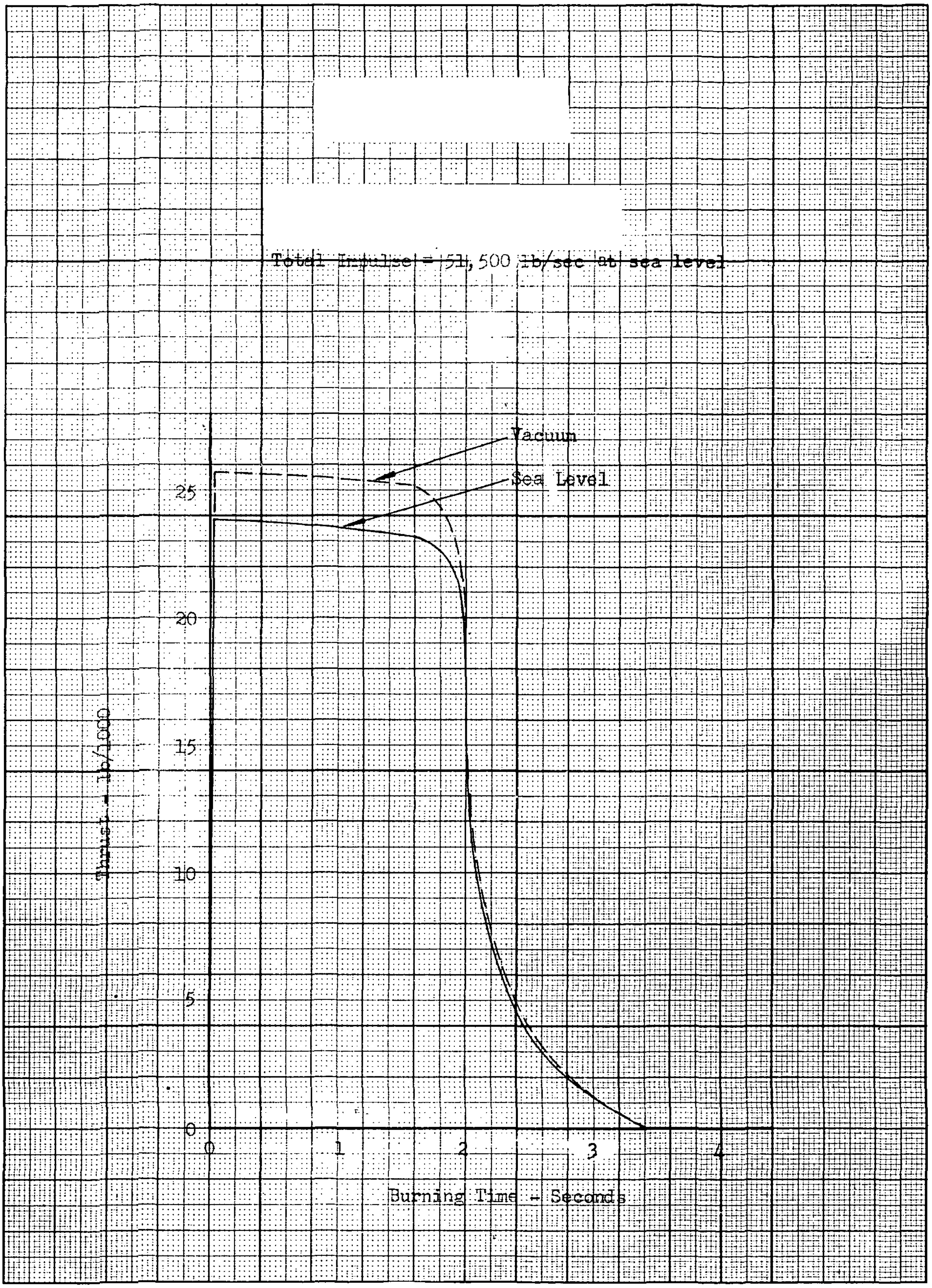


Apollo Escape Motor 1KS-23,800; Thrust vs Burning Time
Figure IV-4

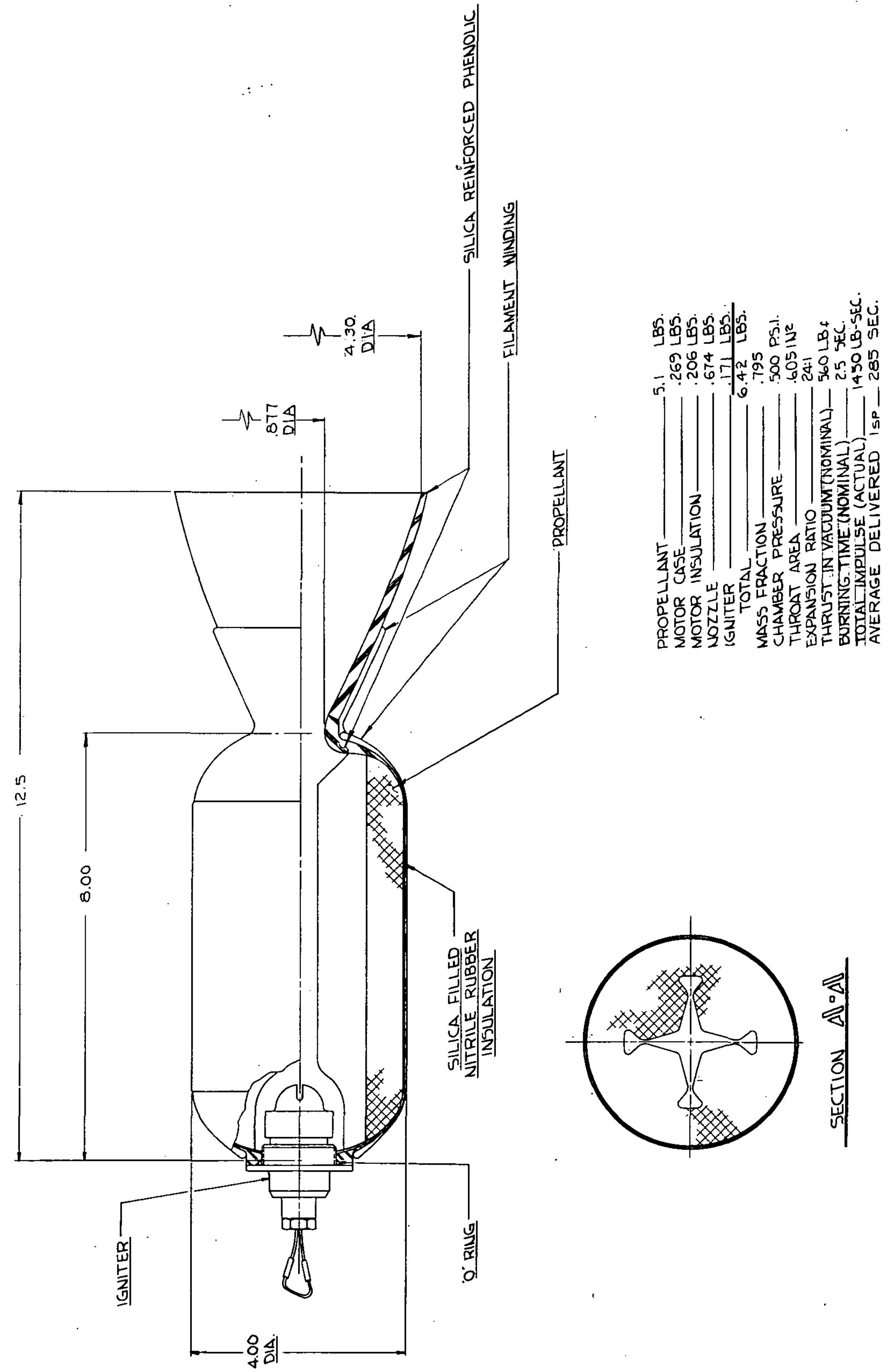


2KS-23,800 Escape Motor

Figure IV-5

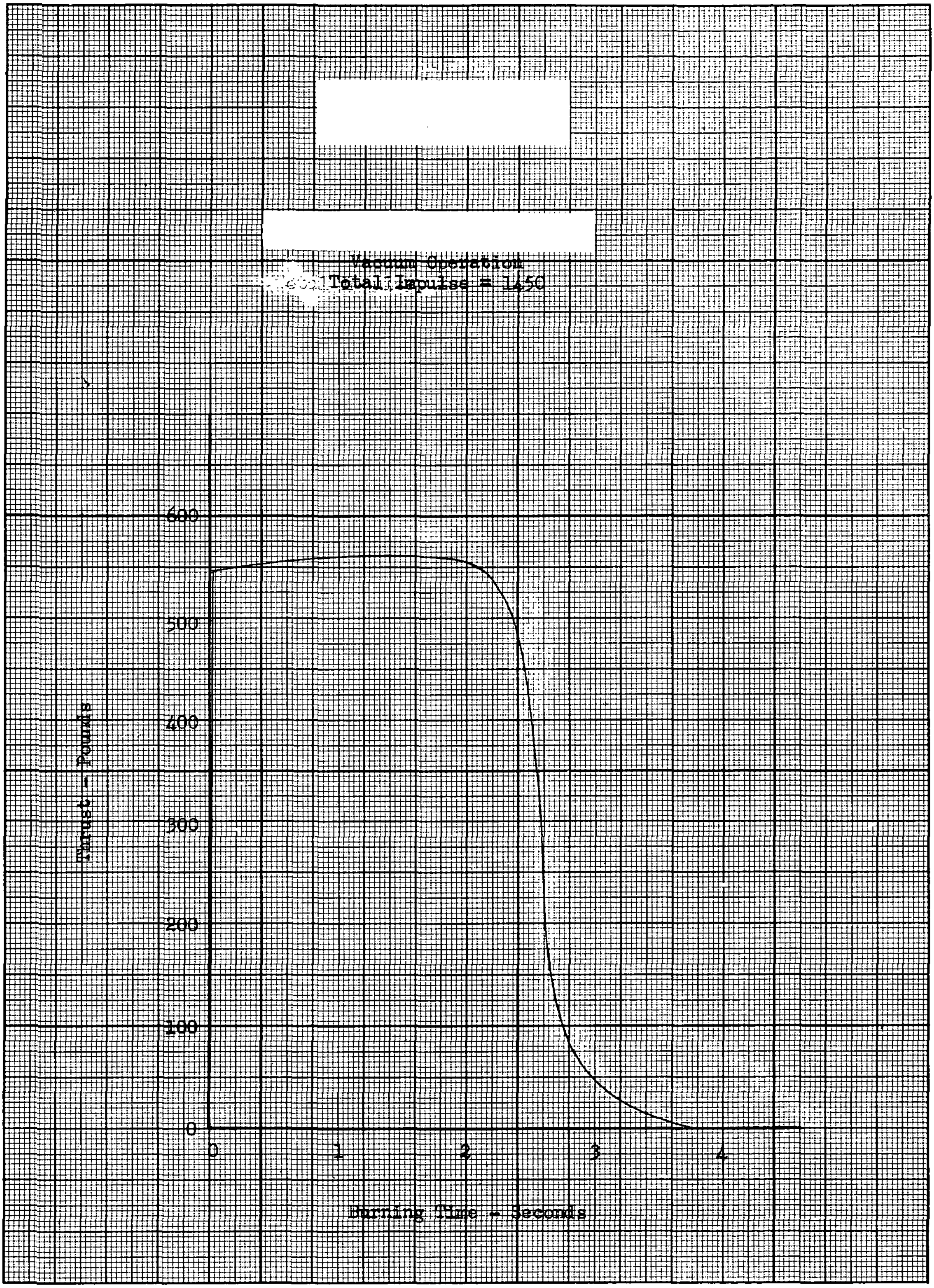


Apollo Escape Motor 2KS-23,800; Thrust vs Burning Time
Figure IV-6



2.5KS-560 Separation Motor

Figure IV-7



Apollo Separation Motor 2.5 KS-560; Thrust vs Burning Time

Figure IV-8

APPENDIX A

GLIDE VEHICLE LAUNCH-ABORT ESCAPE SYSTEM

TABLE LIST

FIGURE LIST

I. REQUIREMENTS FOR LAUNCH-ABORT ESCAPE PROPULSION FOR THE GLIDE VEHICLE

The requirements for this subsystem are as follows:

Aborted weight exclusive of abort escape subsystem (lb)	6,000 lb (1963) 5,500 lb (1966)
Initial thrust to weight ratio (weight includes abort propulsion)	15
Burning time (sec)	1.9
Thrust vector angle in relation to vehicle longitudinal axis (degrees)	20
Number of units	6
Performance perfected at sea-level conditions	

II. DESIGN DESCRIPTION OF LAUNCH-ABORT ESCAPE PROPULSION

A 1.9KS-18,100 escape motor, shown in Figure 1, has been designed for use with the re-entry glider. The nozzle is inclined at an angle of 28 degrees to the motor case, so that the thrust can be directed through the glider center of gravity.

As is shown in Table 1, the propellant weighs 156 lb, and the total weight of the motor is 182.47 lb. The resulting motor mass fraction is 0.855. The volumetric loading of the dendrite grain is 70.5%. A thrust-time curve for the motor is shown in Figure 2. A small part of the propellant is cast into the aft head, and the propellant interface is restricted from burning.

The wall of the 25% nickel-steel chamber is 0.041 in. thick. The average chamber pressure is 1,100 psi.

II, Design Description of Launch-Abort Escape Propulsion (cont.)

Internal motor case insulation is a 0.030-in.-thick layer of silica-filled nitrile rubber.

The nozzle design is similar to that used for the 2KS-23,800 motors, except that the throat diameter and nozzle exit diameter are 3.68 in. and 11.05 in., respectively.

A 0.2% loss in available motor impulse is expected because of throat erosion.

Ignition is accomplished with two Alclojet igniters similar to those used on the 2KS-23,800 motors.

Six of these motors are mounted on the back of the glider vehicle as shown in Figure 3 . The motors are mounted in two packages of three motors each, by shoes sliding in short rails.

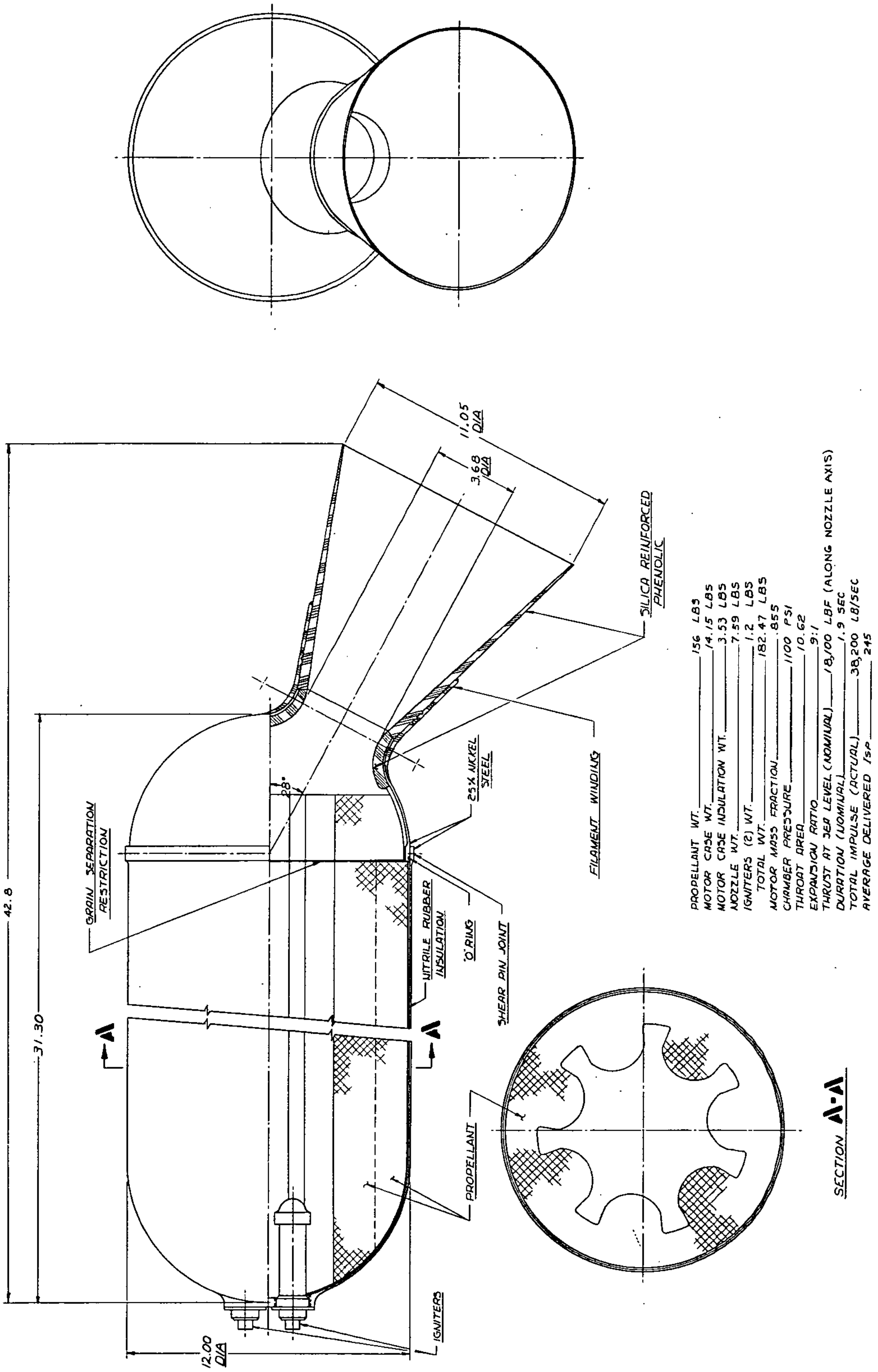
All six motors have the same angle between nozzle and chamber axis. The thrust vector of each motor is directed through the pitch axis of the vehicle. A top view of the vehicle shows that the thrust vector of the two inboard motors is parallel to the vehicle axis, and the nozzles of the outboard motors are inclined sideways so that the thrust vectors are crossing at the centerline forward of the center of gravity. The six motors are jettisoned by applying a side force by use of jettison mechanisms and explosive bolts, as is done with the escape-separation motors on the direct re-entry vehicle.

Development of the 1.9KS-18,100 motor will follow the same plan as for the 2KS-23,800 motor. Motor development cost data are provided separately.

TABLE 1

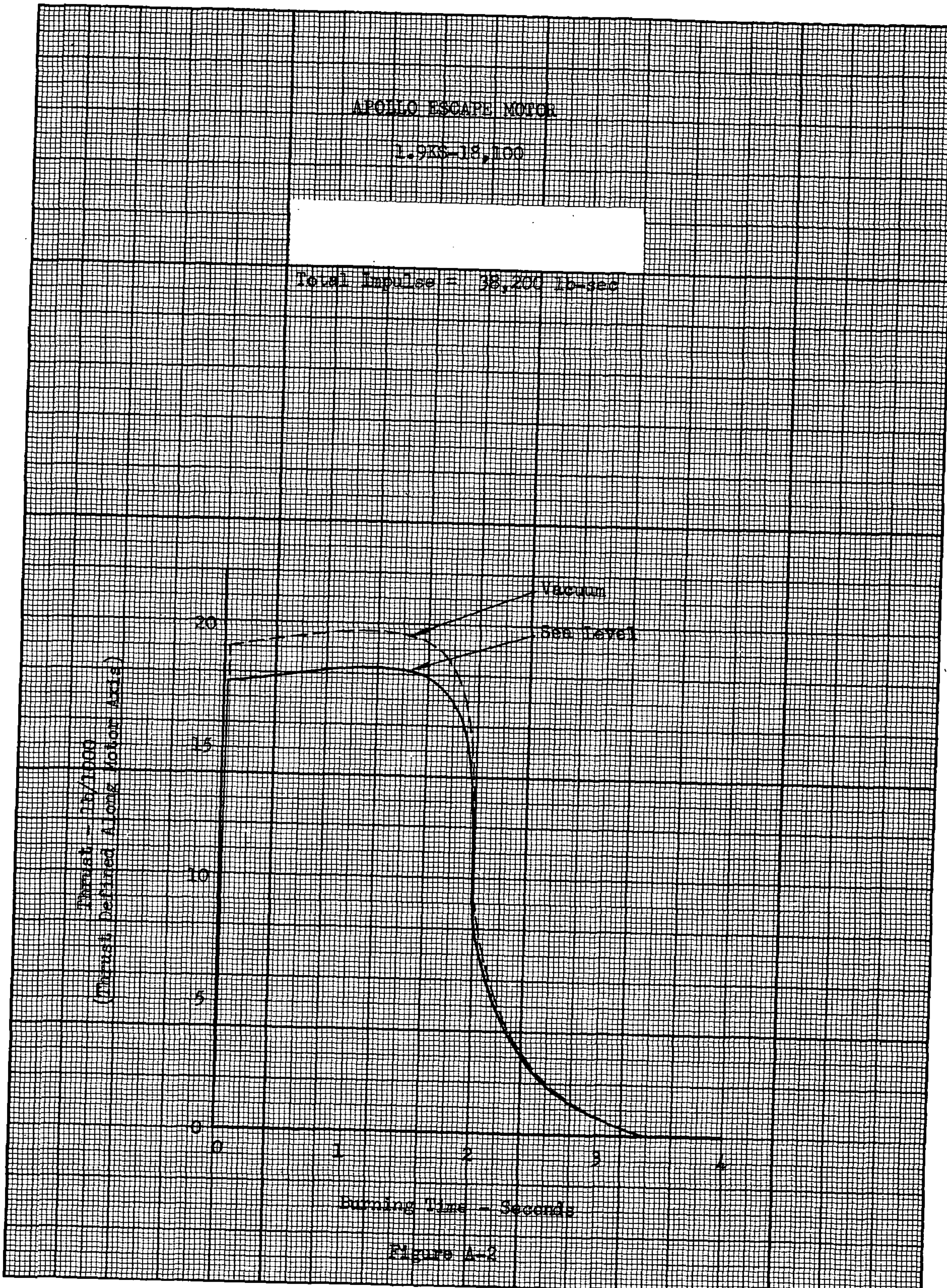
1.9KS-18,100 ESCAPE MOTOR FOR GLIDER VEHICLE

Weight	
Propellant (lb)	156
Case (lb)	14.15
Case insulation (lb)	3.53
Nozzles (lb)	7.59
Igniters (lb)	1.2
Motor weight (lb)	182.47
Mass fraction	0.855
Chamber diameter (in.)	12
Chamber length (in.)	31.3
Overall length (in.)	42.8
Nozzle throat area (sq in.)	10.62
Throat diameter (in.)	3.68
Exit diameter (in.)	11.05
Chamber pressure (psi)	1,100
Motor thrust at sea level (lb)	18,100
Total impulse (lb-sec)	38,200
Grain volumetric loading (%)	70.5
I_s, average, delivered (sec)	245

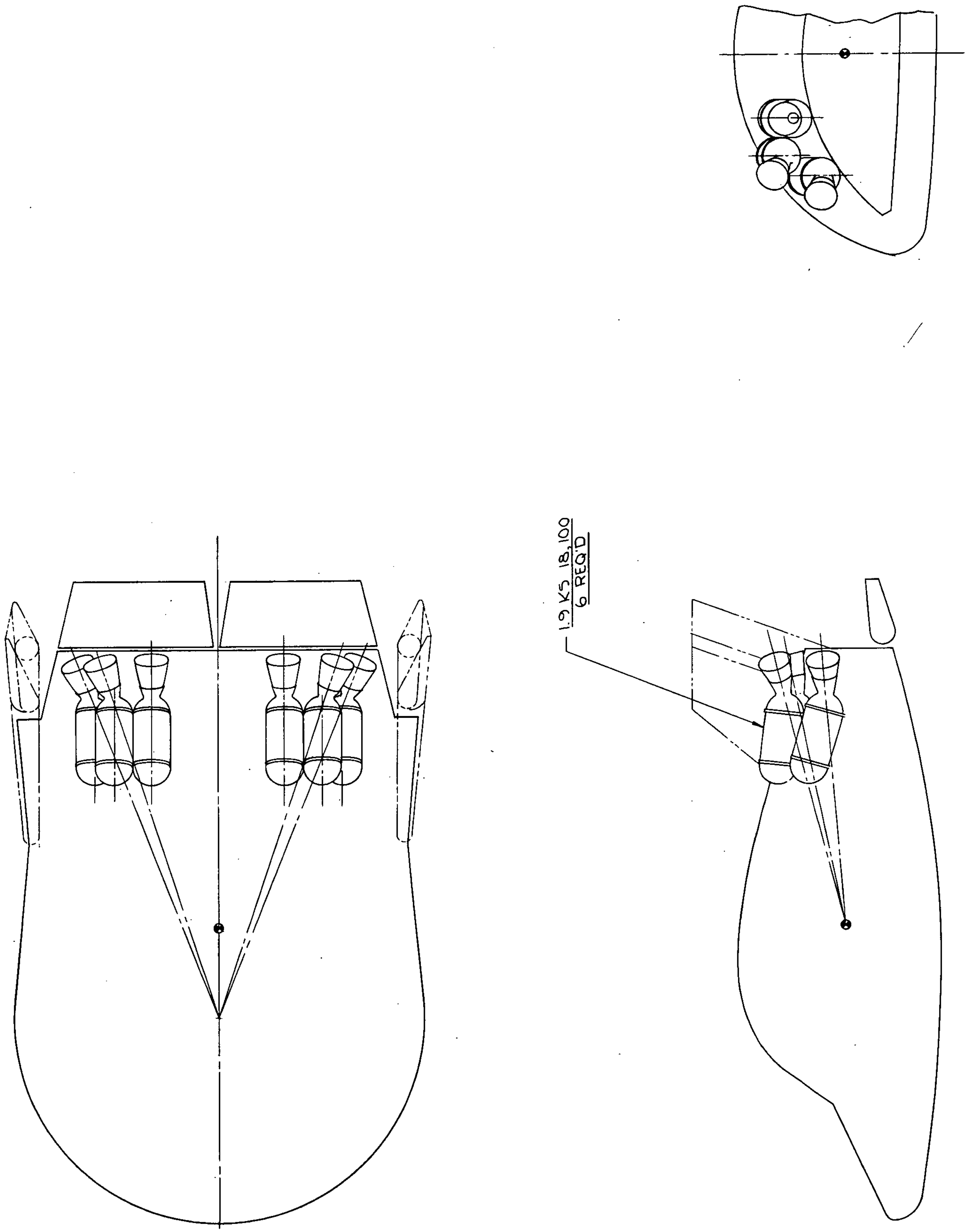


1.9KS-18,100 Glider Configuration Escape Motor

Figure 1



Thrust vs Burning Time--Sea Level Operation
Figure 2



Apollo Glider Configuration Escape Motor Location

Figure 3