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[redacted] SYSTEM PERFORMANCE/DESIGN

R EQUIREMENTS

GENERAL SPECIFICATION

April 1965

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[redacted] SYSTEM PERFORMANCE/DESIGN REQUIREMENTS

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1.0 SCOPE

This specification covers the requirements for the [redacted] Satellite Reconnaissance Search System which shall be capable of obtaining high resolution photography of specific geographical areas. The Satellite Vehicle flight system and ground support required for launching of the vehicle, its orbital control, and data retrieval will also be specified.

2.0 APPLICABLE DOCUMENTS

Requirements of the following documents shall be applicable as specified herein, in all other respects these documents shall be used as guides.

Specifications and Standards

Military

MIL-E-6051C	Interference Limits and Methods of Measurements, Electrical and Electronic Installation in Airborne Weapons Systems and Associated Equipment
MIL-I-26600	Interference Control Requirements, Aeronautical Equipment
MIL-Q-9858	Quality Control System Requirements
MIL-R-27542	Reliability Program Requirements for Aerospace Systems, Subsystems, and Equipment
MIL-S-6644A	Specifications, Equipment, Contractor-prepared, Instructions for the Preparal
MIL-STD-826	Electromagnetic Interference Test Requirements and Test Methods
MIL-STD-150	Photographic Lenses
MIL-STD-803	Human Engineering Criteria for Aircraft, Missiles and Space Systems, Ground Support Equipment
MIL-STD-810	Environmental Test Methods for Aerospace and Ground Equipment
MIL-STD-176	Weight and Balance Data Reporting Forms for Guided Missiles
MIL-STD-130A	Identification Marking of U.S. Military Property

Douglas Aircraft Company

DS-2345A	Production and Acceptance Specification DSV-2C (LV-2A) Thrust Augmented Thor Booster

Lockheed Missiles and Space Company

LMSC-1417524	Program [redacted] Satellite System Specification
LMSC-1414870	Detail Specification, S01B and SS01B Standard Agena Vehicle
LMSC-1416559A	Detail Specification, Model 39205 Vehicle, Program [redacted] Vehicle 1609 and up
LMSC-1417161	Detail Specification, Model 39205 Vehicle, Program [redacted] Vehicle 1625 and up
LMSC-T3-3-2B	Acceptance Test Specification, J-System Payload, Program [redacted]
LMSC-T3-4-001	Requirements Specification, J-System Payload, Program [redacted]
LMSC-T3-3-001	Design Control Specification, J-System Payload, Program [redacted]

LMSC-6117B 6117D	General Environmental Specification for Agena Satellite Systems
LMSC-447969B	Electromagnetic Interference Control Requirements and Electrical Interface for Agena Systems, Specification for.
LMSC-1414931	Specification, Shipping Preparations and Transportation Requirements for the Basic SO1B Vehicles.
LMSC-T3-4-505	Storage and Handling, J-System Payload

<u>Itek Corporation</u>

ITEK ATS-26	Acceptance Test Specification, Pan/Stereo Camera, J-System
ITEK DCS-19	Design Control Specification, Pan/Stereo Camera, J-System
ITEK ATS-28	Acceptance Test Specification, Film Supply Cassette, J-System
ITEK ATS-24	Acceptance Test Specification, Film Take-up Assembly No. 1, J-System
ITEK ATS-25	Acceptance Test Specification, Film Take-up Assembly No. 2, J-System
ITEK ATS-8A	Acceptance Test Specification, Stellar/Index Camera
ITEK DCS-13P	Design Control Specification, Stellar/Index Camera

<u>Eastman Kodak Company</u>

SP 62-65-0004A	Process Specification Film 4404

<u>General Electric Company</u>

GE SVS 3701A	System Design Specification, Satellite Recovery Vehicle, Mark VA.
GE SVS 3702	External Environments Specification, Satellite Recovery Vehicle.
GE SVS 3703	Internal Environments Specification, Satellite Recovery Vehicle.
GE SVS 3705	Acceptance Test Specification, Satellite Recovery Vehicle.
GE SVS 3706	Specification for Component Qualification, Requalification and Acceptance
GE SVS 3707	Specification, Recovery Subsystem - Satellite Recovery Vehicle.
GE SVS 3709	Specification, Orbit Ejection Subsystem - Satellite Recovery Vehicle.

<u>Bell Telephone Laboratories/Western Electric Company</u>

BTL G738648	Guidance Equations Specification, Ground Guidance Computer Programs
BTL G734173	Electrical Acceptance Test Specification, Series 600, Missile-Borne Guidance Equipment

<u>Reports</u>

Program Requirement Document - Program [redacted] National Range Division F 9026, Western Test Range Serial No. 00504, Headquarters WTR, Point Mugu California. (prepared as LMSC B 079855, 1 May 64)

Program [redacted] Orbital Requirements Document - 6595 ATW, VWZDE 4-17 (Prepared as LMSC B 079854, 1 May 64)

Program [redacted] Recovery Requirements Document

Program Support Plan, Program [redacted] F9012 Western Test Range

Program [redacted] Orbital Support Plan (Prepared as AS-65-0000-00214, 14 Jan 65)

Program [redacted] Recovery Support Plan

Thor/Agena Range Safety Report for the Western Test Range. LMSC A094529 A June 1963.

Test Operations Order No. 64-3, Program [redacted] 6594 ATW, May 1964

Recovery Group Operations Plan No. 1-64, Program [redacted] Hq 6594 Recovery Control Group, 6594 ATW AFSC.

[redacted] Letter dated 21 May 64, Subject: Procedures for Capsule Handling, Program [redacted]

Flight Termination System for the Agena Space Boosters Launched on the Western Test Range, LMSC A075675 B May 1964

Safety Procedures Manual, LMSC Operations at Vandenberg Air Force Base - LMSC 220580 Rev. 1, 14 March 64

Launch Stand Safety Plan, LMSC 224455 B

AFR 205-23 Special Security Procedures for Military Space Programs and Projects

Count Down Manual LMSC 226785, VAFB Complex 75

Count Down Manual LMSC 226791, PALC 1

Bulletins

ANA Bulletin 438A   AGE Control for Synthetic Rubber Parts.

3.0 REQUIREMENTS

3.1 Performance

The objective of the [redacted] System is to obtain terrain photographic data of specified geographical areas from a satellite vehicle for search-reconnaissance purposes. In accomplishing the mission, the following events shall be performed:

A. Boost and inject the satellite vehicle into a specified orbit.

B. Attitude Control of the Satellite Vehicle within specified tolerances.

C. Perform commanded functions for both programmed and real time commands.

D. Re-enter the recovery vehicles from orbit along a planned re-entry trajectory and impact at pre-selected ocean location.

E. Provide necessary telemetry, tracking, and commands - both vehicle and ground equipment.

F. Assembly, checkout and launch functions by Aerospace Ground Equipment.

G. Control of equipment temperatures throughout mission phases.

H. Stable re-entry with environment protection of payload data by recovery vehicles.

I. Recovery of data capsules during descent or after water impact.

The [redacted] System shall be capable of obtaining stereoscopic and monoscopic photos while operating in orbit at an altitude range of 90 to 240 nautical miles. The photographic scan angle shall be 70 degrees, yielding a swath width of 180 nautical miles at a Satellite Vehicle altitude of 125 nautical miles. For a single mission, the normal film capacity represents a total stereo ground coverage of 13.6 million square nautical miles. Supplementary provisions shall be provided to locate the vehicle position at exposure within one quarter minute of arc at the local horizon in relation to concentric earth coordinates, with a corresponding time determination within one millisecond. The system shall be designed such that image points of ground detail may be located by photogrammetric methods to an accuracy of at least one quarter nautical mile radius on a map of their true geodetic position.

The camera subsystem shall be capable of being programmed for the desired portion of the ground track on any given orbit. Nominal mission duration shall be eight days with early call down capability. The system shall be capable of performing missions of 11 days duration. Dual recovery vehicles each having a capacity for one half the total film load will be utilized in the nominal mission. In normal operation, recovery of the first capsule will precede continuation of the second half of the mission. However, continuation of the mission shall not be contingent upon re-entering the first recovery vehicle at the time its film capacity is reached. Additionally, a capability shall be provided for extending the time for mission accomplishment in orbit up to 20 days by means of a dormant mode for the Satellite Vehicle by commandable deactivation and reactivation sequences. The system will utilize a standard Thrust-Augmented-Thor Boost Vehicle (LV2A) and a modified standard Agena (S0-1B)* to launch from The Western Test Range (WTR) into high-inclination orbits. In addition to performing the function of a second stage boost vehicle, the Agena also serves as the Satellite Vehicle containing the photographic payload and re-entry vehicles. The re-entry vehicles containing the photographic record will be subject to air retrieval over water, or alternatively to water retrieval. Figure 1 illustrates the launch configuration for the [redacted] System.

The [redacted] System shall utilize flight qualified subsystems and/or components to the greatest extent feasible. On-orbit control will be performed utilizing the Satellite Control Facility. In addition to securing on-orbit payload data, the complete [redacted] System operation will be evaluated for each flight to advance satellite operation technology.

3.1.1 Characteristics

3.1.1.1 Operational Characteristics

The [redacted] System encompasses the total capability necessary to achieve search-reconnaissance photography by orbiting satellite, and includes all functional flight and ground based systems with support personnel necessary to attain this objective. The salient characteristics of the [redacted] System are as follows:

A. Communications and Control will be obtained through the use of the USAF Satellite Control Facility (SCF) tracking, telemetry and command net operating under the control of a centralized mission control center, the Satellite Test Center (STC), located at Sunnyvale, California. The Satellite Control Facility is under the operational cognizance of the 6594th Aerospace Test Wing. Tracking stations of the SCF are the [redacted] Tracking Station [redacted], the [redacted] Tracking Station [redacted] the [redacted] Tracking Station [redacted] the Shemya Auxiliary Station (SRS), the [redacted] Tracking Station [redacted], and the [redacted]

* Designations of S01B and SS01B are used interchangeably for the purposes of this Specification. Refer to Notes, Section 6.0.

Tracking Station [redacted] Other stations which are not a part of the SCF may support operations as necessary on an individual flight or flight series basis if required. The maximum number of orbits between station contacts is to be four.

The SCF shall be responsible for determining ephemeris data for the Satellite Vehicle immediately after orbit injection and updating the ephemeris by use of tracking data throughout the orbital mission. Telemetry data concerning vehicle state-of-health and verification of real time commands and programmed events shall also be obtained by SCF stations, and be made available for reduction, analysis and display at the STC.

Payload and Satellite Vehicle on-orbit functions will normally be preprogrammed on an orbital programmer tape for the desired nominal mission. Alternate stored programs for photographic operations shall also be provided for selection under non-nominal orbit conditions. Adjustment of the orbital programmer, the selection of stored programs and camera operating functions shall be provided through real time commands based on actual ephemeris conditions as determined from tracking data. Tracking and commanding capabilities shall be adequate to compensate for the following effects:

a. Accuracy tolerances on the orbital injection vector affecting orbital period, ecentricity, and location of perigee.

b. Orbit decay with time due to atmospheric drag effects on the Satellite Vehicle.

c. Apsidal rotation and nodal regression caused by the earth's oblateness as related to the orbit plane inclination.

To synchronize the stored commands with vehicle position, the orbital programmer shall be adjusted by real-time command so that the in-track error does not exceed 2 seconds of time. Accurate ephemeris data shall also be provided for real time selection of image-motion-compensation and camera cycle rate, camera operating program and camera operating mode.

At appropriate times following completion of the orbital missions, the re-entry vehicles will be separated and ejected from orbit using their own deboost propulsion capability to impact in the selected retrieval area. The primary impact area is a broad ocean area within the WTR, located between 10° and 26° N. Latitude, and 145° to 172° W. Longitude. The nominal impact latitude is 24° N. Retrieval will be accomplished by air recovery as the primary mode, or by water recovery in the event that air retrieval is not accomplished. The STC shall compute impact predictions for use in commanding deboost of the Satellite Re-entry Vehicles, and for deployment planning by the recovery forces.

Under normal operating conditions, vehicle and payload commands specified for the flight shall be implemented under the jurisdiction of the Flight Test Field Director (FTFD), 6594th Aerospace Test Wing. In the case of abnormal flight conditions or anomalies of the vehicle and/or payload, commands shall be subject to review and approval by the [redacted] Program Directorate, [redacted]

Commands transmitted by the SCF will be based upon ephemeris data obtained and reduced by equipment presently in use in the SCF. Available proven software and procedures will be utilized to the greatest extent possible without compromising system goals. Mission preparation time shall be compatible with the frequency of Program [redacted] launches.

B. Launch Operations

Program [redacted] Vehicles will be launched from Vandenberg Air Force Base. Launch operations will be under the cognizance of the 6595th Aerospace Test Wing. The following launch complexes will be utilized to assemble, check out and launch the LV2A/S01B boost vehicles.

Vandenberg Air Force Base

Launch Complex 75-1	Pads Number 1 and 2
Launch Complex 75-3	Pads Number 4 and 5
PALC I	Pad Number 1

Program [redacted] operations will be supported by the Western Test Range in areas of range safety, collection of down-range telemetry data, surface recovery ships and range interference control. Upon request, a ship and/or aircraft shall be made available for collection of down range telemetry data which is not within reception range of a land based station. In addition, the facilities of Vandenberg Air Force Base (VAFB), will be utilized as available and necessary to the implementation of the program.

Minimum modification to the existing launch facilities is desired to accommodate the LV2A/S01B booster vehicles. Negligible interference shall exist between schedules for preparation and launching from the above listed Pads. Launches shall be in accordance with the approved launch schedule. Program objectives require that at least one successful mission be completed every 30 days. Maximum launch rate will be two per calendar month, with an expected normal launch rate of three each two months. A turn-around time of 14 days from launch to launch for any given launch Pad is desired. Additionally, provisions shall be made to hold a

Program [redacted] launch vehicle in a state of readiness for extended periods of time, up to 20 days, so that launch can be accomplished within 24 hours when so directed. The system goal for successfully accomplishing the initial countdown and launch within the window shall be an 85 percent probability of success. The mission to be flown by this standby vehicle will remain fixed throughout the standby period. When this requirement is in effect, maintenance for this particular vehicle and ground equipment shall be scheduled and performed in such a manner as to not invalidate the ability to launch within 24 hours. At such time as this vehicle must be demated, an alternate vehicle and launch Pad may be required to phase into the standby status provided that the particular requirement is still in effect. Although the standby vehicle must be given highest priority at such time as its launch is directed, the capability to checkout and launch Program [redacted] vehicles from the remaining assigned Pads shall not be impaired during the standby period.

Final checkout, loading, and mating of the payload equipment to the satellite vehicle must be performed under conditions of the strictest security. Appropriate facilities and personnel must be provided to ensure that the nature of the equipment or program mission is not revealed to any unauthorized individual during the preparations for, and conduct of the launch operation.

C. Recovery Operations

The Satellite Vehicle recovery system shall provide a capability for recovery on any day following liftoff. The command to initiate recovery will be given from stations of the SCF.

The recovery sequence is divided into two phases: re-entry and recovery. The re-entry phase starts with the initiation of a programmed command. Following this command, the Satellite Re-entry Vehicle (SRV) beacon and telemetry are turned on to permit detection, tracking and data recording of the re-entry sequence. The Satellite Vehicle is pitched down 58 degrees from the local horizontal, the SRV separated from the Satellite Vehicle, and de-boost is achieved by means of spin jets, retro-rocket, and de-spin jets. Immediately after de-spin, the thrust cone is separated from the re-entry vehicle. The recovery phase consists of the deployment of the parachute system, ejection of the ablative shield, and activation of a flashing light following the exit of the SRV capsule from the ionization layer in the atmosphere.

The recovery force will consist of aerial recovery aircraft equipped with electronic detection and direction finding equipment. Five C-130 type aircraft shall normally be deployed in a North to South direction in accordance with the impact prediction provided by STC. The aircraft shall be equipped with special air retrieval gear to snare and secure the capsule/chute during its descent. The recovery force will also employ surface vessels with tracking/direction finding equipment and helicopters to retrieve a capsule that impacts the sea. Additional support will be rendered by Air Rescue Aircraft with para-rescue capability, weather reconnaissance aircraft, and land-based helicopters for sea surface recovery.

Two SRV's will normally be carried by each Satellite Vehicle. Recovery of the first SRV capsule will be accomplished in from one to five days after launch and the second SRV capsule will be recovered in from one to 20 days after the first recovery. It is required that the WTR ships be on station during both the first and second active periods of satellite vehicle operation. Each active period will normally be of four days duration.

Appropriate liaison and communications between all air and surface units of the recovery forces will be required. Communications between the units of the recovery forces and the STC shall be provided to enable monitoring of all pertinent phases of the recovery operations essentially in real time.

Recovery forces operations and specific deployment for each mission will be under the jurisdiction of the 6594 Recovery Control Group (RCG) Honolulu, Hawaii. Logistic support will be rendered by Pacific Air Forces Base Command (PACAFBASECOM). Overall responsibility for recovery operations rests with the 6594 Aerospace Test Wing, Sunnyvale, California.

Subsequent to recovery, capsule handling and disposition will be in accordance with the directives of the [redacted] Directorate. Until such time as the designated courier is able to assume physical custody, the Commander 6594 RCG will be responsible for the capsule's physical and security safeguarding per designation by the Commander 6594 ATW. Upon assuming physical custody, the courier shall be responsible for capsule handling, transportation and storage details.

3.1.1.1.1 Employment

A. Orbital Elements - For a particular flight the orbit parameters will be specified on the basis of payload search-area considerations and performance available from the launch vehicle subsystem. Parameters of primary importance are the orbit inclination, orbital period, location of perigee, and perigee altitude above the earth's surface. To cause the satellite vehicle to overfly the desired ground track with specified synchronization, the appropriate orbital period will generally be attained by selection of the proper orbit eccentricity.

With the LV2A/S01B booster vehicles, the system shall be capable of a range of missions with orbital parameters within the following limits:

1.	Range of orbit inclinations:	60 to 140 degrees
	(Most probable inclinations:	65 to 90 degrees)
2.	Range of perigee altitude:	90 to 220 n.m.
	(Most probable perigee altitude:	95 to 110 n.m.)
3.	Range of orbital period:	88 to 92 minutes
4.	Location of perigee:	20°N to 65°N. latitude.

Vehicle structural limitations may preclude flying all possible combinations of the above parameters. Hence, at inclinations below 75 degrees, the injection altitude shall be the lowest value compatible with LV2A/S01B structural capabilities. Mission duration shall be dependent upon the orbit inclination and perigee altitude selected.

Accuracy limits. Within the range of missions specified above, the following tolerances shall not be exceeded:

1.	Inclination:	Plus	0.25 degrees
		Minus	0.25 degrees
2.	Perigee Altitude:	Plus	5 nautical miles
		Minus	5 nautical miles
3.	Period:	Plus	0.20 minutes
		Minus	0.20 minutes
4.	Augment of Perigee:		
	(a) For eccentricities of 0.008 or less:	Plus	180 degrees
		Minus	180 degrees
	(b) For eccentricities greater than 0.008:	Plus	20 degrees
		Minus	20 degrees

B. Ascent Requirements

Program [redacted] launches using the LV2A/S01B vehicles will be conducted from the following launch sites of Vandenberg Air Force Base:

Complex	Pad No	Pad Azimuth from North	Geodetic Latitude	Longitude	Elevation above SL
75-1	1	259.50°	34.751629°	120.61830°	214 ft
	2	259.52°	34.755644°	120.62137°	184 ft
75-3	4	181.48°	34.757302°	120.62917°	125 ft
	5	218.50°	34.756153°	120.62522°	160 ft
PALC I	1	223.53°	34.643653°	120.59303°	428 ft

Western Test Range (WTR) facilities shall be utilized for tracking, telemetry, range safety and range frequency interference control.

The Launch Azimuth shall be compatible with orbit inclination requirements and range safety restrictions. For inclinations below approximately 85 degrees, a dogleg maneuver will be required because of range safety limitations on launch azimuth. For PALC launches, the launch azimuth is restricted to 170 degrees, and for VAFB complex launches the azimuth is restricted to 175 degrees. Hence, for trajectories which require orbit inclination angles of less than 80.1 and 84.3 degrees respectively, a yaw maneuver must be accomplished after the predicted downrange impact has passed the critical range safety boundary.

The Launch Window shall not be less than plus and minus one half hour about the optimum launch time. The optimum launch time shall be computed for each flight on the basis of required ground search-area lighting conditions and stellar field considerations based on the orientation of the orbit plane with respect to the Earth-Sun line for the dates of mission operations. Within the above constraints, the launch time and window may be varied to obtain the best thermal environment in-orbit for payload and satellite vehicle temperature-sensitive equipment.

Ascent Sequence of Events for a typical Program [redacted] mission is as follows. This sequence is representative of a 75 degree inclination orbit with injection at 90 nautical miles altitude.

Event	Time(Sec)	Distance(N. M.)
Launch	0	0
Solid Motor Burnout	40	2.5
Solid Motor Separation	60	8.5
Booster Main Engine Cutoff	145.7	83.7
Vernier Engine Cutoff	154.7	100.5
Booster Separation	163	115.8
Stage II Engine Ignition	177	141.2
Solid Motor Impact	284	18
Booster Impact	701	727
Stage II Engine Cutoff (orbit injection)	413	812.5

A representative ascent trajectory profile is shown in Attachment 1.

Launch Reaction Time is defined as the time span necessary to complete all prelaunch preparations and accomplish the launch, starting from the time a particular mission is defined by the [redacted] Directorate. For the LV2A/S01B vehicles the following items require hardware setting or other action based on mission peculiarities. and the launch pad to be utilized.

1. Satellite vehicle and payload fairing paint pattern application for thermal control as required by sun angle predictions.
2. Satellite vehicle recovery timer.
3. Satellite vehicle ascent timer, velocity meter and radio guidance antenna.
4. Satellite vehicle orbital programmer.
5. Stage I Booster autopilot programmer.
6. Ground Command Guidance Computer.
7. Batteries and Control gas loading
8. Range Safety flight data

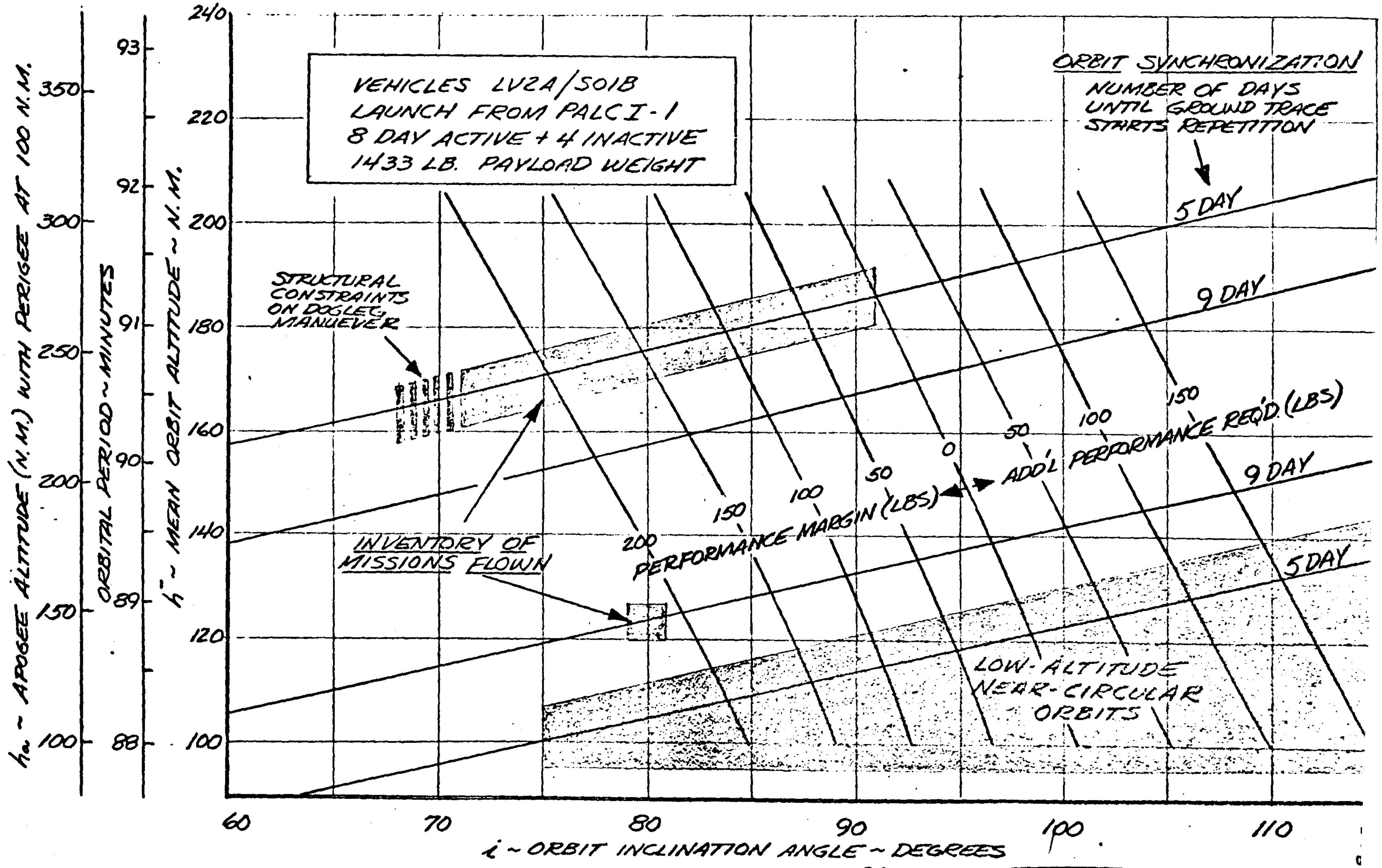
Preparatory work to support readiness of the above items involves trajectory computations and data exchanges between participating Contractors which normally require lead times from launch of 22 days for new missions and 17 days for previously flown missions. The launch reaction requirement for Program [redacted] is 8 days, or less, from mission assignment to launch readiness. Consequently, efforts shall be directed toward determining means of attaining this objective. Approval of the implementation of lead time reductions shall be at the discretion of the [redacted] Program Directorate. Similarily, practical concepts related to factory-to-pad hardware flow and extension of recycle time for vehicles erected on the launch pad are desired for the purpose of minimizing launch reaction time.

C. Mission Performance

Nominal Mission Duration shall be 8 days of active operation with a capability for operating in the dormant deactivate mode of not less than 1 day nor more than 20 days. For any particular mission, the number of active days duration shall be compatible with the orbit parameters required and the system orbital weight capability.

The capability of the system to place a given weight in orbit is a function of the required orbital elements, the ascent guidance, and the vehicle performance parameters. Figure 2 illustrates the nominal Orbital Weight Capability of the system using LV2A/S01B boost vehicles launched from PALC I, Pad 1. Performance capability is shown by lines representing constant values of excess weight, both positive and negative, that can be orbited in addition to the weight of a 1433 pound mission payload. Perigee altitude is 100 nautical miles, and the mission parameters are given in terms of orbit plane inclination and constant lines representing the number of days at which the sub-satellite point will repeat the ground trace initially covered. Repeat of the ground trace is termed orbit synchronization. A synchronous orbit is defined as an orbit having the descending node of the recovery pass of the specified "synchronous" day within one-half of one degree of longitude of the descending node of the first recovery pass after launch. Because of range safety restrictions concerning the drop time of expended solid rocket boost.

~~SECRET~~ [redacted]



~~SECRET~~ [redacted]

motors and launch azimuth restraints, the performance of vehicles launched from the 75 complexes is slightly less than for the PALC I launch shown in Figure 2. With a 75 complex launch, orbital weight capability is reduced by approximately 20 pounds for an orbit inclination of 90 degrees, and by 75 pounds at an inclination of 65 degrees.

Orbit Sustenance and Orbit Manuevering Control are not required provided that mission requirements for orbit plane inclination are achieved and that ground trace synchronization can be maintained for the specified mission duration. For inclinations between 70 degrees and 91 degrees, mission requirements can be satisfied by flying the longer period orbits of approximately 91 minutes (Westward closure) without a need for sustaining the satellite vehicles' orbital velocity. The eccentricity of these orbits tends to minimize the effects of atmospheric drag while maintaining an acceptable perigee altitude and location for payload operation. The operating regime of inclinations greater than 91 degrees with near-circular orbits of approximately 100 nautical miles offers payload advantages of improved scale, more constant compensation of image motion, and increased opportunities for payload operation on Northbound as well as Southbound passes. The desired synchronization can be attained by flying the shorter period orbits (Eastward closure). However, a satellite vehicle orbiting at these lower altitudes will be noticeably affected by the atmospheric environment and may require an orbit sustenance capability to make up the velocity decrement caused by drag. Requirements for drag make up provisions shall assure a 99.5 percent probability that the satellite vehicle altitude shall not decrease below 90 nautical miles during the active mission phases. Efforts shall be directed toward determining means of extending the system capability to attain near-circular orbits of approximately 100 nautical miles altitude for the highest orbit plane inclinations consistent with LV2A/S01B vehicle performance. Implementation of changes required to attain extended capability shall be approved by the Program [redacted] Directorate.

Deboost and Re-entry shall be achievable on either North to South or South to North passes over the Hawaiian recovery area by use of the satellite vehicle primary guidance and control subsystem. Normally, recovery will be effected on North to South passes only, with South to North passes utilized for emergency. A representative re-entry flight profile is shown in Attachment 2. A backup attitude control capability shall be provided in the satellite vehicle which will allow recovery on at least the North to South passes if the primary subsystem should fail.

The primary latitude for Hawaii recovery zone shall be 24 degrees North on North to South and 16 degrees North for South to North passes.

Re-entry impact dispersions are influenced by the following primary error sources:

1. Satellite vehicle attitude and attitude rates at re-entry vehicle separation.

2. Re-entry vehicle attitude and attitude rates after separation and during spin up and retro-rocket impulse.

3. Re-entry vehicle static and dynamic balance.
4. Retro-rocket impulse tolerance.
5. Uncertainty of orbit parameters at time of re-entry vehicle separation.
6. Event timing errors.
7. Uncertainties in actual ballistic parameter, atmospheric density and surface winds.

The following predicted impact dispersions due to the above error sources are representative of a typical Program [redacted] mission:

Orbit Inclination	85 degrees
Eccentricity	0.022
Period	91.06 minutes

Guidance & Control Subsystem	Pass Direction	Dispersion N. M.		
		Up-range	Down-range	Cross-range
Primary	N to S	73	102	± 8
Primary	S to N	132	186	± 11
Back up	N to S	104	176	± 13

Abort of the launch or the orbital phases of the mission shall not cause the nature of mission objectives to be revealed to unauthorized persons. In the event of an aborted launch, provisions shall be made to recover payload equipment under appropriate security conditions. Similarily, in the case of a catastrophic malfunction during booster ascent, a strict accounting shall be made of payload equipment salvage and/or disposition. In the event of orbit phase abort, the payload data shall be recovered. If recovery cannot be effected, the satellite vehicle shall provide a capability by command or other means to positively and permanently disable the command, telemetry and recovery subsystems.

Existing Range Safety Requirements shall be complied with.

The Reliability Objective for the Total System must consider the use of existing hardware provided as Government Furnished Equipment (GFE) to vehicle Contractors, and the adaptation of a ballistic missile vehicle and associated AGE for the [redacted] System.

Reliability shall be defined as the probability that the complete mission vehicle assembly will accomplish all primary mission performance requirements for the duration specified herein. The following values represent probability of success goals to be achieved in System operation:

1.	Scheduled count down to launch		.85
2.	Launch, Ascent, and Orbit Injection		.91
3.	Orbital Mission	(8 days active)	.86
		(9 days active)	.845
		(10 days active)	.83
		(11 days active)	.81
4.	Re-entry and Recovery (each SRV)		.92

On-Orbit Disposal of the satellite vehicle and payload equipment will occur by the natural re-entering of the vehicle into the earth's atmoshpere due to drag effects, with subsequent destruction of the vehicle by re-entry heating and loads. This action normally occurs approximately 25 days after launch for Program [redacted] satellites placed in a 91 minute orbit.

3.1.1.1.2 Deployment

A. Mission Responsibility- Figure 3 illustrates the management and mission responsibility relationships for [redacted] System. These relationships are stated in more detail as follows:

Program Management - Air Force [redacted] - Program [redacted] Directorate, [redacted] is responsible for overall planning and management of the [redacted] System.

General Systems Engineering and Technical Direction - GSE-TD will be provided by [redacted]. This work comprises the engineering effort dealing with the overall integration of the system, design compromises among subsystems, definition of interfaces, analysis of subsystems, and supervision of system testing, all to the extent required to assure that the system concept and objectives are being met in an economical and timely manner.

Launch Vehicle Contractor

The Launch Vehicle Contractor shall provide the Stage I Booster vehicle, and will also provide all services necessary to checkout and launch the vehicle. Douglas Aircraft Company is the Launch Vehicle Contractor for the LV2A, Thrust Augmented Thor used for Program [redacted].

Ascent Guidance Contractor

The Guidance Contractor shall provide the guidance equations and guidance-computer programming necessary to inject the Satellite Vehicle into specified orbits within allowable tolerances. Radio command guidance will be utilized to steer both the Stage I booster and the Satellite Vehicle during ascent. Western Electric Company/Bell Telephone Laboratory and [redacted] are the Guidance Contractors for Program [redacted]

Satellite Vehicle Contractor

The Satellite Vehicle serves as a Stage II Booster during ascent and operates in the orbit mode after orbit injection. The Satellite Vehicle Contractor shall provide this vehicle and all necessary services to checkout and launch the vehicle. Lockheed Missiles and Space Company is the Satellite Vehicle Contractor for the Agena S01B used for Program [redacted].

Payload Integration Contractor

The Payload Contractor shall provide the vehicle payload section, and shall integrate the installation of camera equipment, Satellite Recovery

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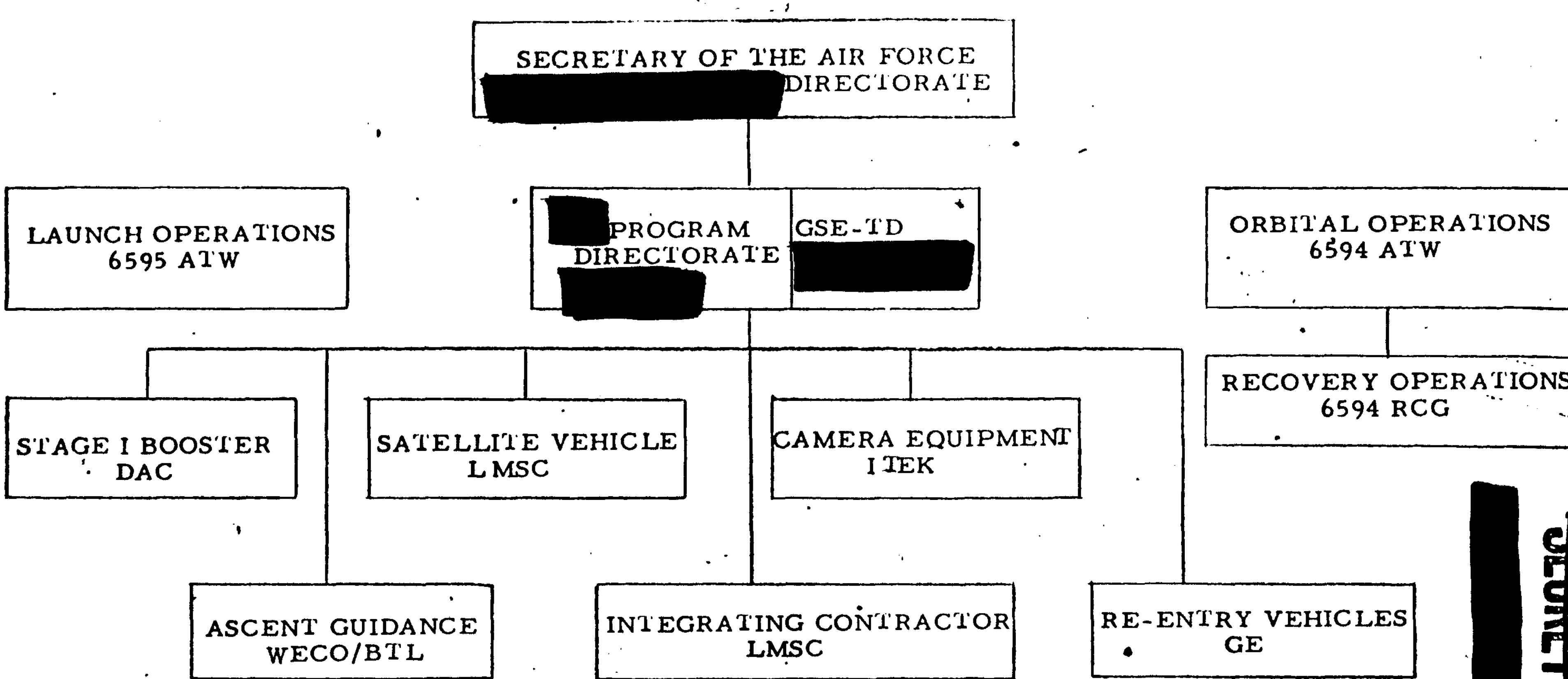

Figure 3 MAJOR ELEMENTS AND MISSION RESPONSIBILITIES - [redacted] SYSTEM

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Vehicles, and associated components necessary to operate the payload. Camera equipment and SRV's will be supplied to the Payload Integration Contractor as Government Furnished Equipment. Lockheed Missiles and Space Company provides the J-System payload integration for Program [redacted]

Camera Equipment Contractor

Panoramic cameras, film cassettes, and associated photographic equipment shall be provided for the [redacted] System by the Camera Equipment Contractor. This equipment will be supplied GFE to the Payload Integration Contractor. Itek Corporation provides the Corona J-System camera equipment for Program [redacted]

Re-entry Vehicle Contractor

The Re-entry Vehicle Contractor shall provide the Satellite Re-entry Vehicles (SRV) comprising the data capsule, ablative re-entry heat shield, parachute recovery system, retrieval aids and associated components required for SRV operation. This equipment will be supplied as GFE to the Payload Integrating Contractor. General Electric Company provides the Mark V-A re-entry vehicles for Program [redacted]

Launch Operations

The 6595 Aerospace Test Wing stationed at Vandenberg Air Force Base shall be responsible within the direction of the [redacted] Program Directorate for the integration and conduct of all prelaunch and launch operations for this program at the Vandenberg AFB launch sites.

Orbital Operations

Orbital operations shall be conducted by the U.S. Air Force Satellite Control Facility under the operational cognizance of the 6594 Aerospace Test Wing stationed at Sunnyvale, California. The 6594 ATW shall be responsible within the direction of the [redacted] Program Directorate for the integration and conduct of all orbital and recovery operations for this program.

Recovery Operations

Recovery forces operations and deployment will be under the jurisdiction of the 6594 Recovery Control Group as designated by the Commander, 6594 Aerospace Test Wing.

Integrating Contractor

The integrating Contractor shall be responsible to the [redacted] Program Directorate for the successful integration of all hardware into the Satellite Vehicle, and for the planning and conduct of prelaunch tests to verify a condition of flight readiness for this vehicle. Additionally, the Integrating Contractor shall be responsible for the preparation of preflight trajectories, orbital programmer tapes, and integrated documentation required to support the launch and orbital operations for this program. In the performance of these responsibilities, the Integrating Contractor shall coordinate with all other affected Associate Contractors as necessary. Lockheed Missiles and Space Company is the Integrating Contractor for Program [redacted]

3.1.1.2 Logistics

Each Contractor shall plan and logistically support his end-item hardware as necessary for the accomplishment of Program [redacted] mission objectives and schedules. Spares provisioning shall be defined in conjunction with the applicable procurement agencies to support Program [redacted] schedules. Spare equipment for Program [redacted] shall be subjected to appropriate levels of acceptance testing prior to being placed in logistical channels for all locations other than at the Contractor's primary assembly facility. In the latter case, spares shall pass applicable acceptance tests prior to being installed on a Program [redacted] vehicle. The Contractors also shall define controls for issuance of spares. Such controls shall provide for considerations of controlled life items and for re-acceptance testing of spare equipment where extended storage periods may be incurred.

3.1.1.3 Personnel and Training

The numbers of personnel, personnel pre-requisites, and required training needed to support the [redacted] System shall be identified by each major Contractor and participating Air Force Test Wing. Personnel requirements shall be specified for the following categories:

A. Operational Personnel

B. Maintenance Personnel

C. Organizational Maintenance Personnel

Existing facilities shall be utilized for training wherever possible. Unique requirements for training facilities solely related to the [redacted] System shall be identified together with supporting substantiation for the need.

3.1.2 System Definition

3.1.2.1 System Engineering Documentation

3.1.2.1.1 System Functional Flow Diagram

Figure 4 illustrates the functional flow sequence for the major elements of the [redacted] System. Functional interfaces are described in greater detail in Section 3.3.6 of this document. Figure 5 presents a tree of the operational requirements and planning documentation for the [redacted] System.

3.1.2.2 Functional Subsystem List

Functional Subsystems comprising the [redacted] System are identified as follows:

3.1.2.2.1 Satellite Vehicle Subsystem

The Satellite Vehicle Subsystem consists of an Agena, S01B, orbiting vehicle containing the payload equipment and re-entry vehicles. The satellite vehicle subsystem shall provide all necessary functions to fulfill the spaceborne mission requirements.

3.1.2.2.2 Launch Vehicle Subsystem

The Launch Vehicle Subsystem consists of a Thrust-Augmented-Thor, LV2A, for the first stage booster vehicle with the Agena, S01B, serving as the second stage booster. Ascent guidance is provided by Radio-Command Guidance utilizing radar tracking, and a ground based computer to compute the guidance commands. The launch vehicle subsystem shall provide all necessary functions to achieve injection of the satellite vehicle into the specified orbit within the allowable tolerances. The launch vehicle configuration is shown in Figure 1.

3.1.2.2.3 Mission Control and Communication Subsystem

The Mission Control and Communication Subsystem consists of the Satellite Test Center, Remote Tracking Stations of the Satellite Control Facility, and associated personnel, communications equipment, computers and computer programs necessary to track, command, and readout the telemetry of the Satellite Vehicle System.

3.1.2.3 Contract End-Item List

Figure 6 illustrates the Specification Tree for the major end-items comprising the [redacted] System. Detailed end-item lists shall be prepared and maintained by each applicable Contractor in accordance with his contract. Detailed Specs

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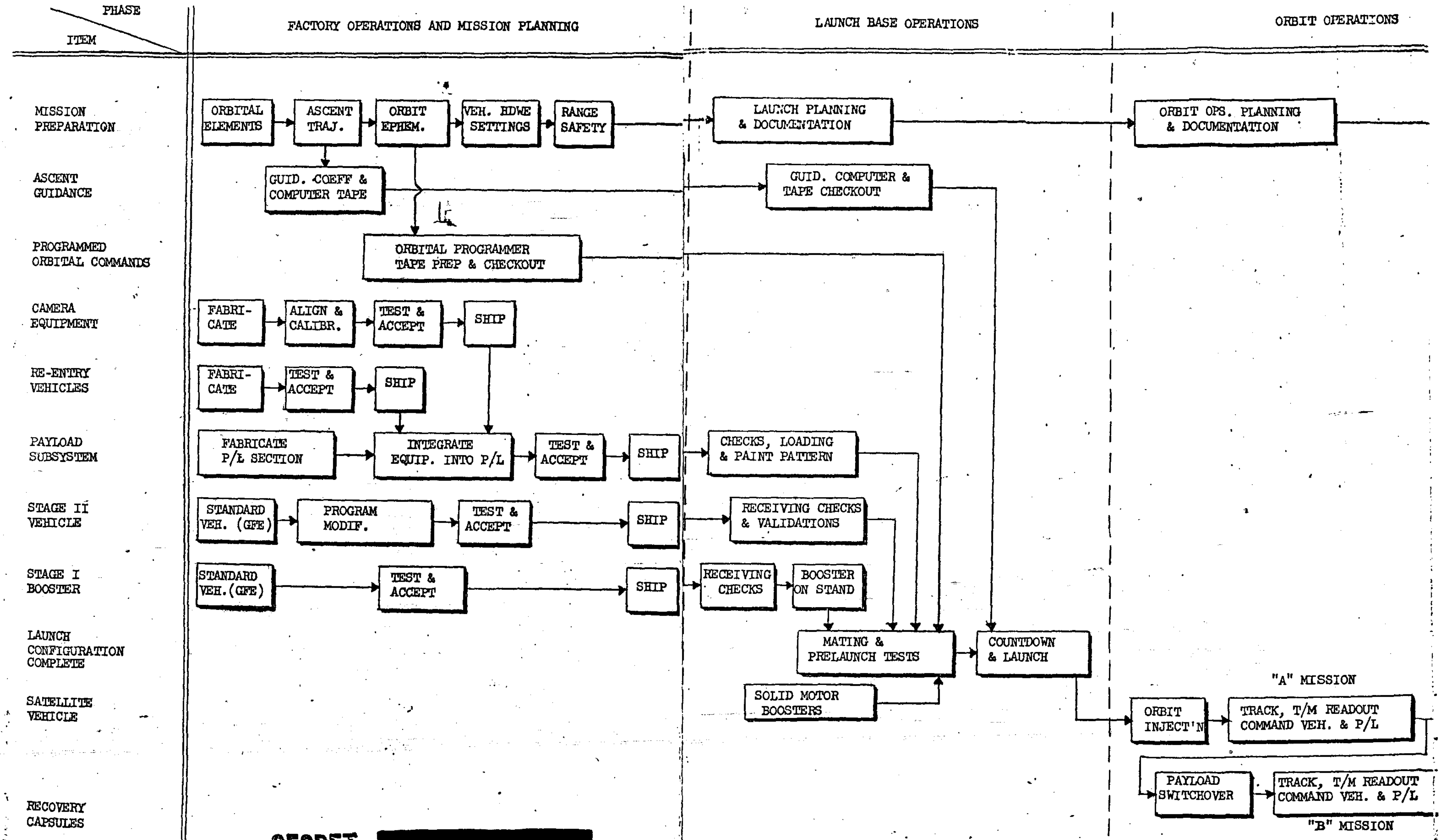
FUNCTIONAL FLOW DIAGRAM - MAJOR ELEMENTS [redacted] SYSTEM



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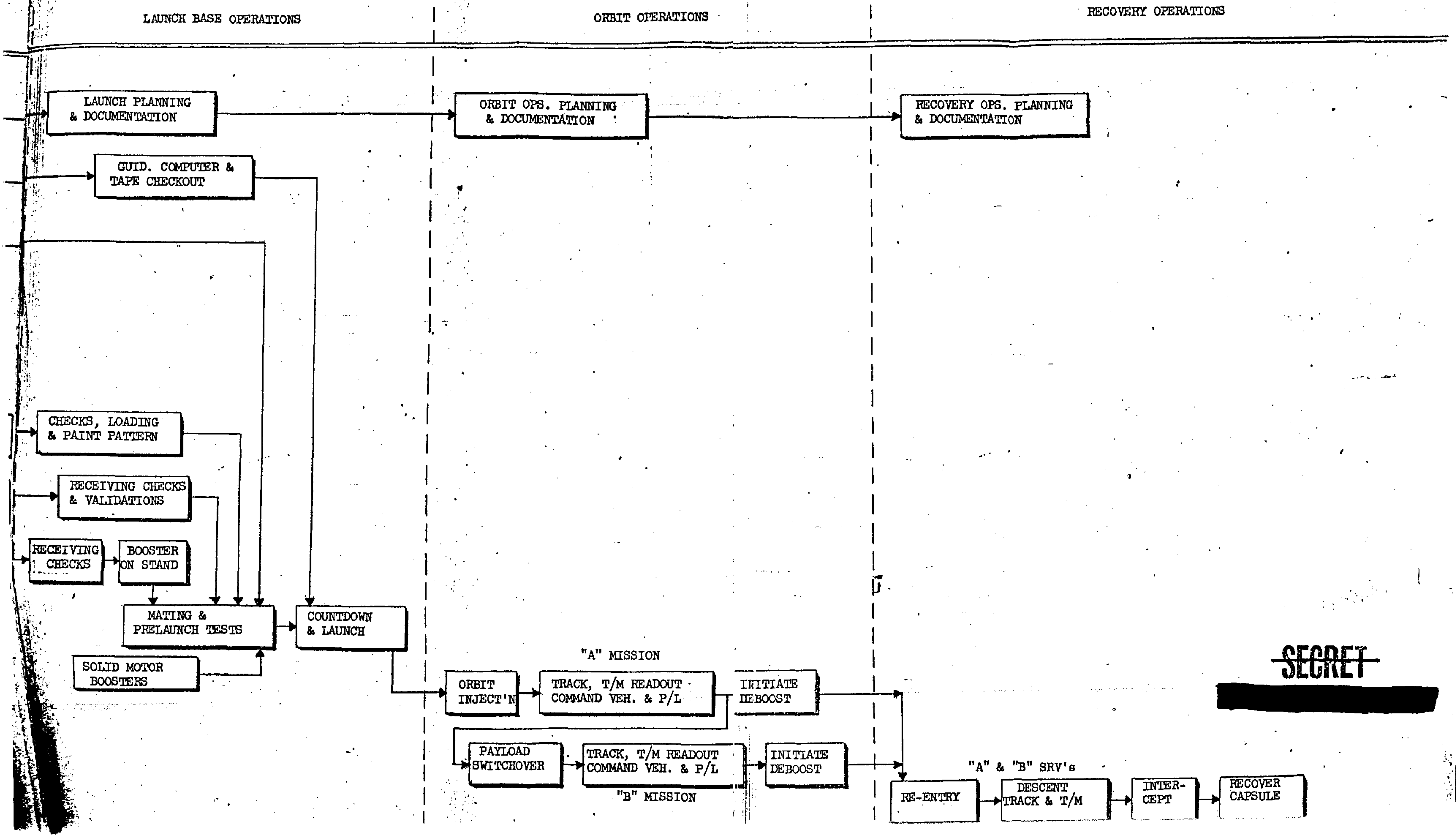
FUNCTIONAL FLOW DIAGRAM - MAJOR ELEMENTS [redacted] SYSTEM



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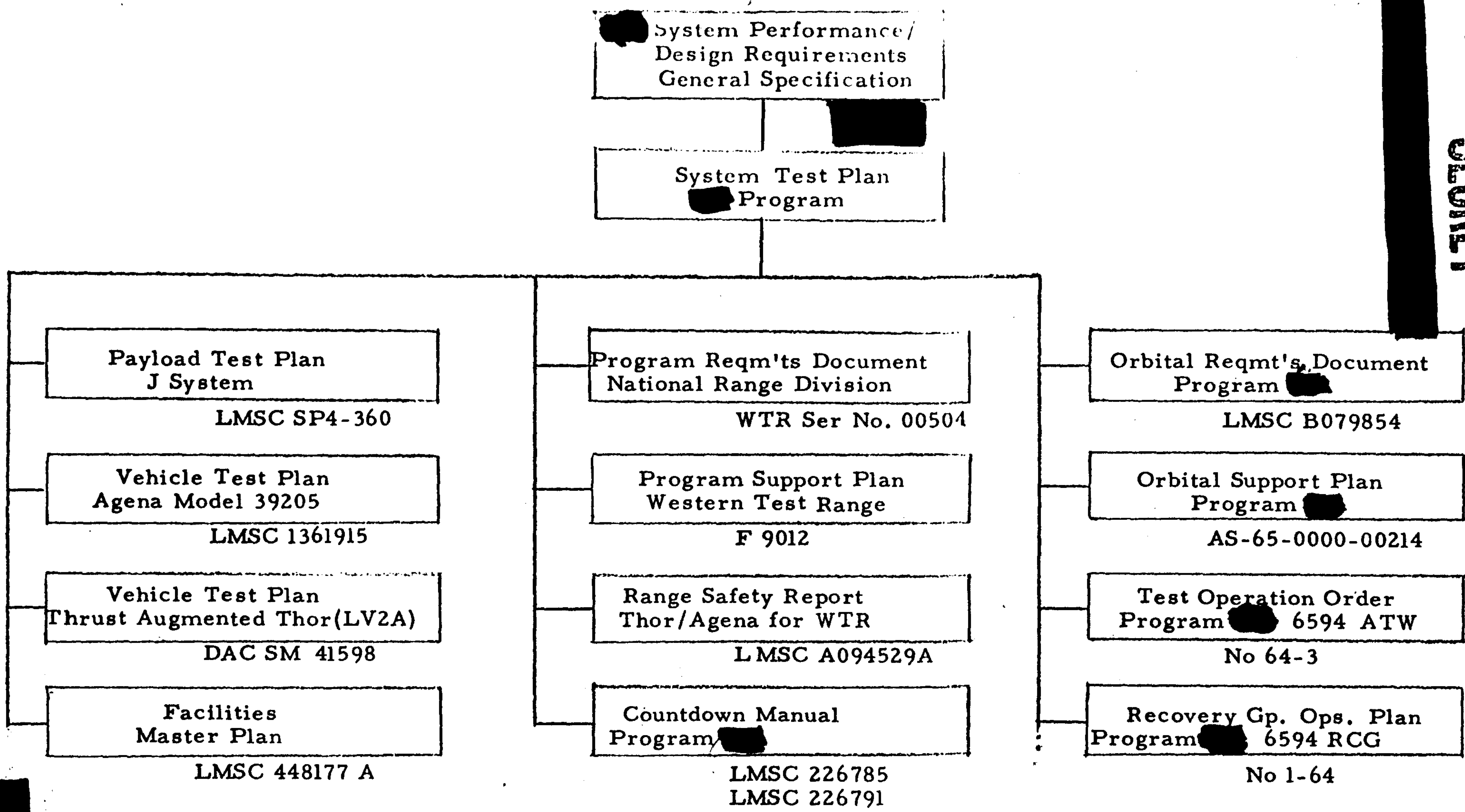

Figure 5 Operations Requirements and Plans - [redacted] System

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[redacted]

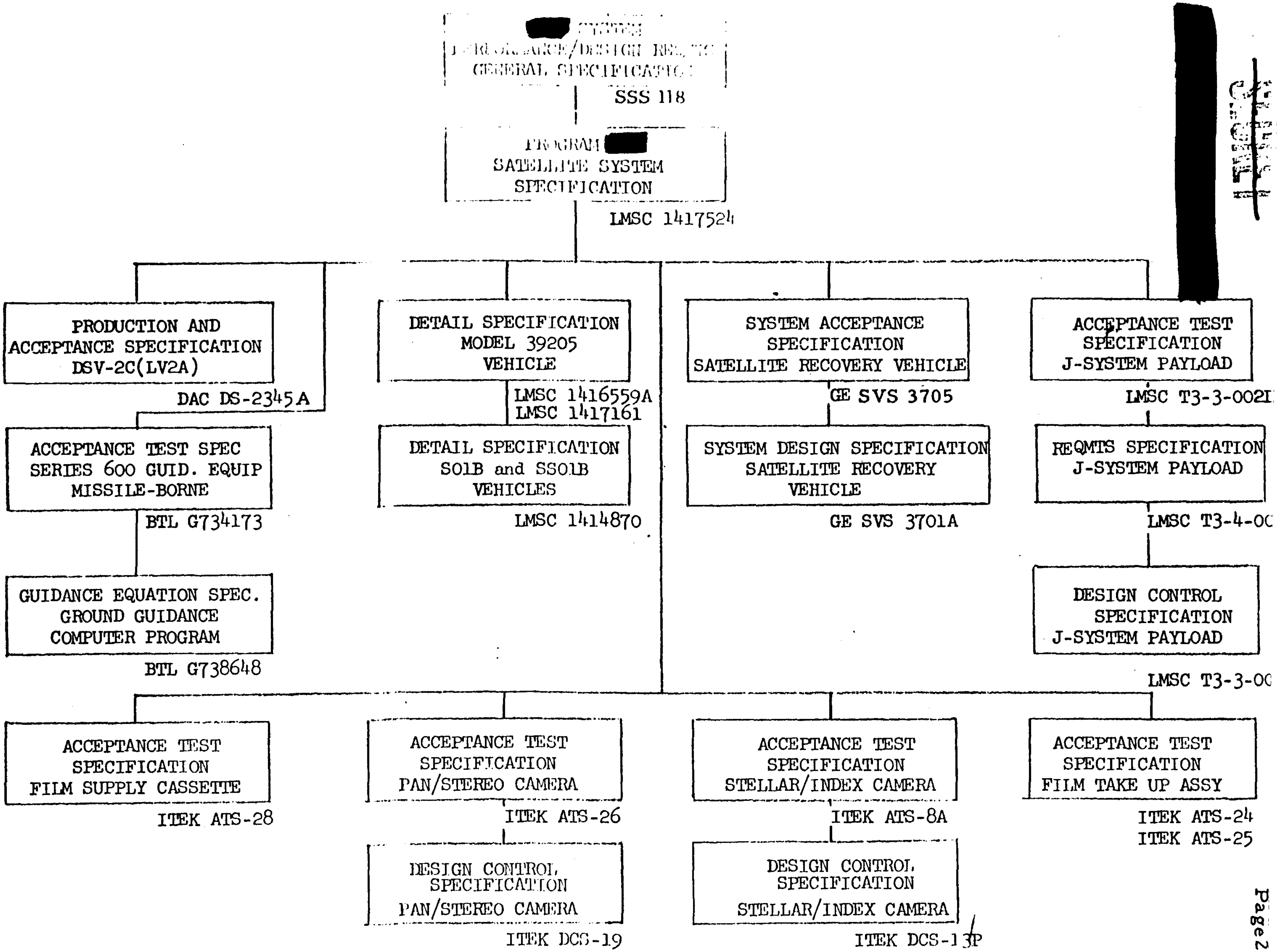


FIGURE 6 SPECIFICATION TREE MAJOR END ITEMS [redacted] SYSTEM

shall also be prepared, submitted to the applicable Procurement Agency for approval, and subsequently maintained by the Contractor for each end-item to be furnished.

3.1.3 Operability

3.1.3.1 Reliability Requirements

3.1.3.1.1 Equipment Life Requirements

As an objective, all flight equipment shall be designated and assembled of suitable materials so that all Program [redacted] calendar life and operating life requirements can be met under the environments encountered without an increased failure rate. These minimum life requirements are stated below. Where other objectives require that a less durable design be adopted, maintenance, repair, and scrap procedures shall be specified, and the life experience of each limited life item shall be recorded and controlled. The accumulated calendar life and operating life of all limited life items shall be reviewed as part of the final flight-readiness certification procedure to insure that sufficient useful life remains to meet the flight requirements without reliability degradation.

A. Limited Calendar Life

As an objective, materials, parts, and assemblies shall be used which are not subject to age or temperature deterioration within a calendar life of three years. Items which do not satisfy the minimum calendar life requirement shall be marked and controlled. Control procedures shall be based on pertinent factors of cure date, storage environments, and date of assembly into more complex configuration. Calendar life data shall be recorded, and reviewed for part suitability prior to flight.

B. Limited Operating Life.

As an objective, materials parts, and assemblies shall be used which are not subject to an increasing failure rate due to wear during an operating period of at least 2000 hours of nominal operation or equivalent cycles of operation after prime contractor acceptance. In no case shall the non-wear-out cyclical life be less than three times the maximum estimated or experienced Program [redacted] cyclical life. Where the above objectives are not met, the item shall be classified as a limited operating life item with specified maintenance, repair, and scrap procedures. Elapsed time indicators shall be utilized or an operating log shall be maintained to report accumulated operation time or operation cycles for all limited operating life equipment.

3.1.3.1.2 Numerical Reliability Goals

For the purposes of this program, reliability shall be divided into two categories: i.e., Reliability Design Objectives and Reliability Flight Goals. The [illegible]er category was discussed in Section 3.1.1.1.1 of this document.

Reliability design objectives shall be defined as those reliability [illegible]s assigned to the functional subsystems for the purposes of design apportion- [illegible]ment [illegible] respective vehicle subsystem and component levels.

The reliability figures given below represent the calculated probability that [illegible] in-line performance functions will successfully take place when an opportunity to perform is presented. For purposes of calculation, the Contractor's portion of the critical Aerospace Ground Equipment (AGE) shall be considered as an in-line function. Reliability degradation by items beyond the control of the contractor such as GFE hardware, weather, and recovery forces are not applicable.

3.1.3.1.2.1 Satellite Vehicle Subsystem

A. Program Modified Agena S01B (orbital mode, 8 days active operation) Design Objective .92

B. Payload Equipment (8 day operation) Design Objective .94

C. Re-entry Vehicles Design Objective .98

3.1.3.1.2.2 Launch Vehicle Subsystem

A. Stage I Booster Vehicle (LV-2A)

The reliability design objective of the basic booster vehicle is outside the scope of this specification. However, for the purpose of reliability apportionment, a figure of 0.96 is to be utilized.

B. Stage II Booster Vehicle (Agena S01B, ascent mode)

Design Objective .95

C. Radio Command Ascent Guidance

The reliability design objective of the radio command guidanc system is outside the scope of this specification. For the purpose of reliability apportionment, a figure of 0.985 is to be utilized for the combined ground based equipment and vehicle-borne equipment.

3.1.3.2 Maintenance Requirements

Maintenance and repair cycles for equipment and facilities required to operate and support the [redacted] System shall be scheduled and accomplished on a compatible basis with Program schedules and operations. Maintenance and repair requirements will normally be associated with the following areas:

A. Pad refurbishment after vehicle launch. This work shall be performed on a schedule that will support a pad turnaround time of 14 days from launch to launch.

B. Aerospace Ground Equipment (AGE) required to perform vehicles checkout, countdown and launch.

C. Satellite Control Facility equipment for tracking, communications, data handling and processing.

D. Vehicles and equipment of airborne and waterborne recovery forces.

In no instance shall the probability of successfully attaining Program [redacted] mission objectives be reduced by a failure to properly repair or maintain essential equipment.

3.1.3.3 [redacted] System Environments

3.1.3.3.1 Prelaunch Environment

The prelaunch operations phase shall include that period of time from when the equipment is removed from the assembly area until launch has been accomplished. Removable protective coverings may be utilized where necessary to prevent damage caused by the climate environments. All such coverings shall be easily removable prior to lift off. The following climatic environments are associated with the prelaunch operations:

A. Temperature - Surrounding air temperature from a minimum of 25°F to a maximum of 100°F.

B. Humidity - Relative humidity up to 100 percent with conditions such that condensation takes place in the form of water or frost.

C. Fungus - Exposure to high humidity is conducive to fungus growth. Materials which are fungus nutrients shall not be used.

D. Sand and Dust - Exposure to graded wind-blown sand and dust environment, equivalent to exposure in the sand and dust test outlined in Procedure 510 of MIL-STD-810 (USAF).

E. Sunshine - Exposure to radiant energy equivalent to exposure to the test outlined in Procedure 505 of MIL-STD-810 (USAF).

F. Rain - Exposure equivalent to four inches per hour for two hours rain in a rain chamber.

G. Salt Fog - Exposure to salt fog environment, equivalent to exposure in the salt fog test outlined in Procedure 509 of MIL-STD-810 (USAF).

H. Pressure - Sea level to 5,000 feet altitude.

I. Propellant Compatibility - Surfaces and areas adjacent to the vehicle propellant tanks may be exposed to propellant fumes or splashes. Materials which are employed in areas so subjected shall be capable of withstanding the effects of such exposure. Where current state-of-the-art prohibits meeting this requirement, protection coating or replacement shall be employed.

J. Explosive Atmosphere - Equipment intended to operate in areas where a possibility of an ambient explosive atmosphere exists shall operate in such an atmosphere without causing an explosion.

K. Ground Wind Velocity - Vehicle structural limitations shall not be exceeded due to the effects of surface wind steady state velocity or gusts. The ability of the launch configuration to withstand wind induced effects is dependent upon factors of tank pressurization, propellant loading, and structural support provided by launcher equipment. For the conditions specified, the launch vehicle configuration shall be capable of withstanding the following surface winds:

Propellant Tank Conditions

(P) indicates pressurized tank

(U) indicates unpressurized tank

Launcher Conditions

(A) unbolted toppling condition

(B) bolted at base only

(C) bolted at base and supported at Missile Station 151

Maximum Winds Conditions - Including Gusts*						
	Wind Velocity - Knots					
Launcher Condition	(A)		(B)		(C)	
Tank Pressure	(P)	(U)	(P)	(U)	(P)	(U)
Booster & S01B Empty w/o Solids		15		46		75
Booster & S01B Empty with Solids		31		46		75
Booster Empty S01B Fueled with Solids		35		41		75
Booster & S01B Fueled with Solids	61	35	64	35	75	75

3.1.3.3.2 Launch Ascent Environment

The ascent flight phase shall include that period of time from ign[illegible] of the Stage I Booster engine through injection of the Satellite Vehicle int[illegible]bit.

A. Vibration - A random vibration environment will be encou[illegible]ed during lift-off resulting from the acoustic noise generated by the launch vehicle's first stage engine and during the transonic portion of the flight due to aer[illegible]dynamic buffeting. Sinusoidal oscillations will be encountered as a result of lift-o booster engine shutdown, and start and shutdown of the second stage booster engine at

* For steady state surface winds, take 80% of the values listed.

various times throughout the flight, vehicle bending and compression structural vibration modes will be excited. The vibration levels given in Paragraphs 1 and 2 below are to be considered as motions of the primary structure of the satellite vehicle. For purposes of design it can be considered that the sinsoidal and random vibrations are encountered separately.

1. <u>Random Vibration</u> - The following random vibration spectrum is to be considered the motion of the satellite vehicle in each of the three primary axes.

Satellite Vehicle Area	Stage I Boost		Stage II Boost	
	Power Spectral Density g^2/CPS	Frequency CPS	Power Spectral Density g^2/CPS	Frequency CPS
Engine	.03	15-2000	.07	15-2000
Aft Rack	.05	15-2000	.01	15-2000
Propellant Tank	.05	15-2000	.01	15-2000
Fwd Rack & Payload Area	.03	15-2000	.005	15-2000

2. <u>Sinusoidal Vibration</u> - The satellite vehicle will be subjected to a number of transient vibration environments which are composed of decaying sinusoidal oscillation. These sinusoidal oscillations range in frequency from 2 to 200 cps with an envelope of the amplitudes as follows:

<u>Axial and Lateral</u>

1.0g (zero to peak) 5 to 50 cps

2.0g (zero to peak) 50 to 200 cps

<u>Torsional</u>

40 rad/sec^2 (zero to peak) 50 to 80 cps duration (1/2 sine = 200 milliseconds)

B. <u>Acoustic</u> - The most severe acoustic environment is encountered during lift-off and the transonic portion of the flight. Following is a tabulation of the estimated maximum octave band sound pressure levels to be encountered outside the satellite vehicle at the Stage II/Stage I vehicle interface.

Estimated Sound Pressure Levels over-all 151.6 db*

Frequency CPS		Octave band level (db)*
37.5	75	140
75	150	142
150	300	144
300	600	144
600	1200	144
1200	2400	143
2400	4800	140
4800	9600	137

*db reference 0.002 dynes/cm^2

C. Quasi-Static Acceleration - Quasi-static accelerations are produced by the thrust of the launch vehicle. The maximum expected acceleration levels are given below. For purposes of design it is assumed that the axial and lateral accelerations occur simultaneously:

Axial Direction	7.5 g's
Lateral Direction	2.5 g's

D. Temperature - During the powered flight ascent phase, the satellite vehicle and its payload will be subjected to aerodynamic heating. The vehicle skin and equipment temperatures to be encountered in flight depend upon the trajectory and equipment placement within the vehicle. The Contractor shall conduct analyses to determine temperatures of critical structure and equipment including the provisions and effectiveness of thermal protection where required. The trajectory used in these analyses shall represent the maximum heating case given by the root-sum-square effect of all applicable flight perturbations (3 sigma).

E. Shock - The shock or transient environment encountered as a result of engine ignition and shutdowns are taken into account by the vibration levels of Paragraph 3.1.3.3.2 A.1. The environment encountered as a result of pyrotechnic devices to sever vehicle skin and initiate ejection of doors/ fairings is ahigh-level, high-frequency, short duration load imparted to the structure. This shock is a potential problem for equipment located in the general area of the pyrotechnic device. The levels expected in the immediate vicinity of the pyrotechnics may be of the order of 3 to 4 thousand g's peak at high frequency for a very short duration. The design, installation and testing data for equipment to be located in these areas shall be reviewed by the [redacted] Program Directorate prior to granting approval to implement the installation.

F. Ambient Pressure - The satellite vehicle will be subjected to a decreasing ambient pressure during the ascent powered flight. The pressure will vary from that encountered at sea level to a pressure of 1×10^{-8} millimeters of Mercury encountered in space. The Contractors' design shall ensure that all equipment intended to operate with a zero pressure differential across various components, such as lens assemblies, shall be properly vented to accommodate the reduction in ambient pressure. Similarly, equipment intended to operate under a pressurized environment shall be adequately sealed.

G. Winds Aloft -The winds amplitude, rate of shear, duration of shear and atmospheric density are factors that may cause the ascent vehicle structural and control capability to be exceeded. Prior to launch, the Integrating Contractor shall compute the wind shears utilizing the winds aloft data from standard Rawinsonde observations, and provide the Launch Test Wing with percentages of structural and control capability. Standard observations will be accomplished at 0000 Greenwich time and subsequent 6 hour intervals. If sufficient safety margins cannot be attained because of the winds aloft, the launch shall be delayed.

H. Atmospheric Density Model: The ARDC Model Atmosphere 1959 is to be considered representative of atmospheric density for ascent condition design studies. The use of other atmosphere models may be permitted pending prior approval of the [redacted] Program Directorate.

3.1.3.3 Orbit Environment

During orbital flight, the satellite vehicle will be subjected to an environment consisting primarily of the following:

A.	Vacuum	1×10^{-8} mm of Mercury
B.	Solar Radiation	445 BTU/ft^2hr (nominal)
C.	Earth Shine	68.7 BTU/ft^2hr (nominal)
D.	Earth Albedo:	35% of the solar energy (nominal)
E.	Magnetic Field	560 milligauss at the poles to 280 mg at equator for an altitude of 125 n.m.
F.	Atmospheric Density	The Lockheed-Jacchia or the LDENSITY atmosphere models shall be used. The LDENSITY model is defined in LMSC 656536.

For the orbits flown in performing the [redacted] System mission, the effects of micrometeoroids and ionizing radiation can be considered negligible.

3.1.3.3.4 Descent and Recovery Environment

The descent and recovery phase shall include the period of time from parachute deployment for the recovery capsule until it has been retrieved in the air, or from the water. The following environments are encountered during this phase.

A. Upper Atmosphere - The ARDC Atmosphere Model 1959 shall be considered as representative of atmospheric conditions for the descent environment for analysis purposes. Winds aloft, cloud cover and visibility conditions shall be monitored in the recovery area during Program [redacted] operations. This meteorological data shall be utilized by the Recovery Control Group in planning the deployment of recovery forces, and by the SatelliteControl Test Wing for optimizing the re-entry and impact position.

B. Air Retrieval - At the time of parachute deployment, the capsule may be oscillating at ± 20 degrees and be rotating at a rate of 15 rpm.

The parachutes shall be capable of proper deployment under these conditions, and shall provide a nominal rate of descent of 25 ft/sec, under standard atmospheric conditions, at 10,000 ft. altitude above mean sea level. For weights of 90 to 180 lbs suspended under the main canopy, the chute/capsule shall be capable of sustaining retrieval loads by aircraft traveling at 135 knots. The maximum altitude for air recovery is 15,000 ft.

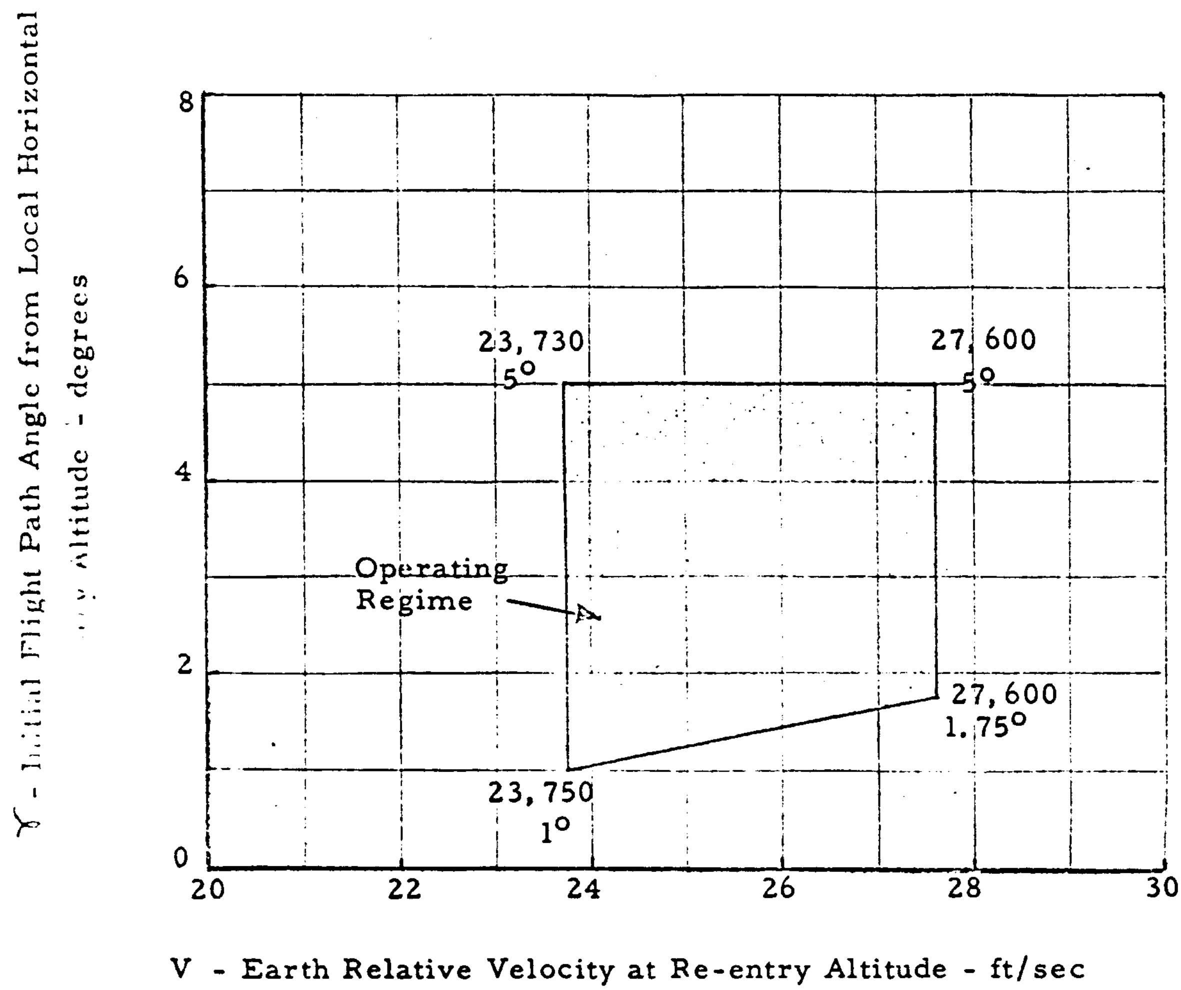
C. Surface Conditions - The recovery capsule shall be capable of sustaining water impact under conditions of a sea state of 3 with 18 knot surface winds. After water impact the capsule shall float in an upright position without capsizing in sea states of 3 or less. (Sea state as defined by U.S. Navy Hydrographic Office).

D. Aerodynamic Heating and Loads - During the re-entry phase, the SRV shall be subjected to heating and load conditions dependent upon the re-entry trajectory and SRV attitude. The design of the SRV shall be based upon analyses to determine temperatures of critical equipment and structure during re-entry, including the provisions and effectiveness of thermal protection where required. The trajectory used in these analyses shall represent the maximum heating case given by the root-sum-square effect of all applicable flight perturbations (3 sigma). Re-entry trajectories shall be confined to the performance envelope given on Figure 7.

3.1.3.4 Transportability

Under normal operating conditions, the Stage I Booster, the Satellite Vehicle, and the Payload Section of the satellite vehicle will be transported as separate items from their respective factory areas, and subsequently mated at the launch base. Transportation will normally be performed by road, using suitable trucks or special trailers as required. Payload equipment shall be capable of being transported by aircraft from the Contractor's plant to the Integrating Contractor's facility. During all periods of transportation, positive security safeguards shall be provided to physically protect System [redacted] flight equipment and preserve the required level of secrecy concerning its use in the mission.

Figure 7 - Operational Performance Envelope
Satellite Re-Entry Vehicles



Re-entry Altitude = 325,000 ft.

3.1.3.5 Safety

Normal transport, handling, and pad safety requirements will be recognized in the design and handling of Program [redacted] vehicles, equipment and components. Compliance shall be required with applicable regulations for pressurized lines and bottles, explosives and toxic propellants. Particular attention shall be given to selecting explosive ordnance which will meet the Western Test Range criteria for radio-frequency-electromagnetic environments.

Toxic or highly corrosive propellants shall be capable of being loaded and drained in the erected position on the pad, in a safe manner; even in the event of a launch abort at essentially "lift off" time. This is not intended to require that these functions be implemented through a "fly away" umbilical, but rather that situation control be maintained through such umbilical so that the vehicle can be safely approached for the attachment of fill and drain lines as required.

3.1.3.6 Vulnerability

Vulnerability is here defined as the susceptibility of the satellite vehicle and payload to hostile environments resulting from agressive attack. Such environments may be induced by means of particles or nuclear radiation. Practical consi[illegible]rations of weight and performance capability preclude the use of shielding [illegible]ening provisions on an extensive basis. However, where a design choice [illegible] in decreased vulnerability of critical mission components without imposing [illegible]cant performance penalty or degradation of function, the incorporation of [illegible]ch [illegible]tures is desired. Implementation of hardening provisions shall be contingent upon [illegible]rior review of the design and approval by the [redacted] Program Directorate.

3.2 System Design and Construction Standards

3.2.1 General Design and Construction Requirements

3.2.1.1 Specifications and Standards - The selection of specifications and standards pertinent to the design and construction of vehicles, subsystems, equipment and components of the [redacted] System shall be identified in the appropriate end item specification prepared by the Contractor. Specifications and standards selected, and the degree of imposition on the design and construction of System [redacted] end item hardware, shall be subject to review and approval by the [redacted] Program Directorate.

3.2.1.2 Materials, Parts and Processes - Materials, parts, and processes utilized in the fabrication of [redacted] System hardware shall be compatible with the applicable environments outlined in Section 3.1.3.3, and shall represent a current state-of-the-art approach to efficient design and construction. The requirements and use of advanced "state-of-the-art" research technology and/or "exotic" materials for [redacted] System hardware shall be reviewed by the [redacted] Program Directorate prior to fabrication of program hardware. The Contractor shall state requirements for materials, parts and processes in his specifications.

3.2.1.3 Moisture and Fungus Resistance - Moisture and fungus resistance shall be compatible with the environments outlined in Sections 3.1.3.3.1 and 3.1.3.3.4. The Contractor shall state requirements for moisture and fungus resistance in his specifications.

3.2.1.4 Corrosion of Metal Parts - Metal parts shall be resistive to the corrosive environments outlined in Section 3.1.3.3 and as required in each individual design application. Requirements for corrosion resistance shall be stated in the Contractor's specifications.

3.2.1.5 Interchangeability and Replaceability - The design of [redacted] System hardware shall specify tolerances necessary to achieve the interchangeability and replaceability required throughout the operational life of the system. The extent of establishing and maintaining interchangeability shall particularly apply to the mechanical interfaces between mating hardware of two or more Associate Contractors. Interchangeability and replaceability requirements for [redacted] System hardware shall be identified in applicable specifications for hardware and interfaces.

3.2.1.6 Workmanship - Workmanship shall conform with the standard practices prevalent in the Aerospace industry. Uniformity of shapes, dimensions, fit, and performance shall permit replaceability of items as dictated by their operational requirements. Any item showing evidence of poor workmanship shall not be accepted for use in the [redacted] System.

3.2.1.7 Electromagnetic Interference - All vehicle subsystems shall comply with MIL-E-6051C and MIL-I-26600* except as specifically authorized: components shall comply with MIL-I-26600. A demonstration of component compliance will be mandatory. An electrical interface control plan for the Satellite Vehicle shall be originated and employed in the design, fabrication and test of the Satellite Vehicle in accordance with the requirements of MIL-I-26600.

*For new or modified equipment procured subsequent to March 1965, MIL-STD-826 will supercede MIL-I-26600. The applicability of MIL-STD-826 shall be in conformance with the requirements of the procurement contract.

3.2.1.8 Identification and Marking - The identification marking requirements for vehicle equipment and components shall be in accordance with MIL-STD-130A. External marking requirements for vehicle external connections and thermal control paint patterns shall be identified on applicable drawings by the Contractor.

3.2.1.9 Storage - Specifications covering the storage and handling requirements for [redacted] System vehicles, equipment and components shall be provided by each Associate Contractor for his deliverable end-item hardware. The specifications shall identify requirements for environmental protection, maximum duration of storage for specified types of equipment, and requirements for equipment maintenance in a stored condition.

3.3 Performance Requirements for Functional Subsystems

3.3.1 Satellite Vehicle Functional Subsystem

The satellite vehicle functional subsystem consists of a satellite vehicle with attached payload and satellite re-entry vehicles (SRV), and the necessary ground facilities and aerospace ground equipment (AGE). The satellite vehicle, the payload, and the SRV are discussed individually under Sections 3.3.1, 3.3.2 and 3.3.3 respectively as follows:

3.3.1.1 Budgeted Performance and Design Requirements

3.3.1.1.1 Satellite Vehicle Weight Budget - The maximum weight for the S01B Agena for Program [redacted] missions shall be as follows for the conditions stated. These weights do not include the mission payload or SRV weights.

Item	Weight (lbs)	Total Weight (lbs)
Weight Empty		2437
Propellants	13565	
Helium	1	
Attitude Control Gas	74	
Aux Alt. Control Gas-L/B	11	
Gross Weight Without Payloads		16113
Less Adapter and Attach	- 293	
Less Retro Rockets	- 10	
Less Destruct System	- 6	
Separation Weight w/o Payloads		15804
Less Horizon Sensor Fairings	- 7	
Less Attitude Control Gas	- 1	
Ignition Weight w/o Payloads		15796
Less Propellants	- 13517	
Less Engine Start Change	- 2	
Less Attitude Control Gas	- 6	
Burnout Weight		2271
Less Residual Propellants	- 48	
Less Helium	- 1	
Weight on Orbit with Gas but w/o Payload		2222
Less Remaining Att. Control Gas	- 67	
Empty Weight on Orbit w/o Gas, w/o Payload		2155

3.3.1.1.2 Satellite Vehicle Reliability Budget Requirements

The reliability design objective for the satellite vehicle is 0.95 for the period of time during which the vehicle functions as a second stage booster in the ascent mode, and 0.92 for subsequent operation in the active orbital mode for a period of 8 days. These values are exclusive of the payload equipment and satellite recovery vehicle hardware. The Satellite Vehicle Contractor shall apportion the budgeted reliability goal to each of the following vehicle functional subsystems:

A. Space frame

B. Propulsion

C. Electrical

D. Guidance and Attitude Control

E. Tracking, Telemetry and Command

Equipment design and component selection shall consider budgeted reliability goals. Additionally, subsystem reliability shall consider improvements provided through redundancy. The Contractor shall prepare and maintain a detailed reliability model together with failure-rate data for the purpose of analyzing and reporting current reliability performance estimates for his end-item hardware.

3.3.1.2 Satellite Vehicle General Design Requirements

3.3.1.2.1 Satellite Vehicle Description

The satellite vehicle shall perform both ascent and orbital functions. The ascent functions shall be to:

A. Provide thrust required to attain injection of the satellite vehicle and payload into the specified orbit.

B. Maintain attitude control and respond to guidance steering commands so that injection into orbit is accomplished within allowable tolerances.

C. Provide a means for relaying radio guidance commands to the Stage I Booster from receivers mounted in the satellite vehicle during the first stage booster guided portion of flight.

D. Provide telemetry data concerning vehicle performance and equipment status during the ascent.

The on-orbit functions of the satellite vehicle shall be to:

A. Provide a stable earth-oriented platform for the payload.

B. Provide the required electrical power for vehicle and payload functions throughout the mission.

C. Provide a means for real-time commanding and stored-program commanding of vehicle and payload functions throughout the mission.

D. Provide environmental protection for all critical vehicle equipment during the orbital phase.

E. Provide a means for transmitting vehicle and payload information concerning status, operation, and environment back to the ground.

F. Perform necessary maneuvers and sequences to eject the two recoverable re-entry vehicles from the satellite vehicle for de-boost from orbit.

The Agena, S01B, vehicle is a liquid-fueled second stage booster, powered by a gimballed rocket engine. During powered flight, pitch and yaw control is provided by the rocket engine with roll control provided by cold-gas reaction jets. During coast and orbital flight, attitude control is effected by the three-axis pneumatic reaction nozzles. The vehicle is composed of four major sections; the forward equipment section, the propellant tanks section, the aft equipment section, and the Stage II/Stage I adapter section.

The forward equipment section contains mounting provisions for a payload section, and accommodates the major part of the guidance, electrical and communications equipment. The tank section is an integrally constructed dual chamber containing the fuel and oxidizer for the rocket engine. The aft equipment section provides mounting support for the rocket engine, gas reaction jets and hydraulic system. The booster adapter section attaches to the aft part of the tank section and is designed to support the entire satellite vehicle from the first stage booster during the ascent phase. The adapter section remains attached to the Stage I booster at the time the two vehicles are separated in flight. Figure 8 depicts the satellite vehicle configuration.

During ascent, the rocket engine develops a nominal thrust of 16,000 pounds. The engine is designed for a nominal thrust duration of 245 seconds burning time. Propellants consist of unsymmetrical dimethylhydrazine (UDMH) fuel,

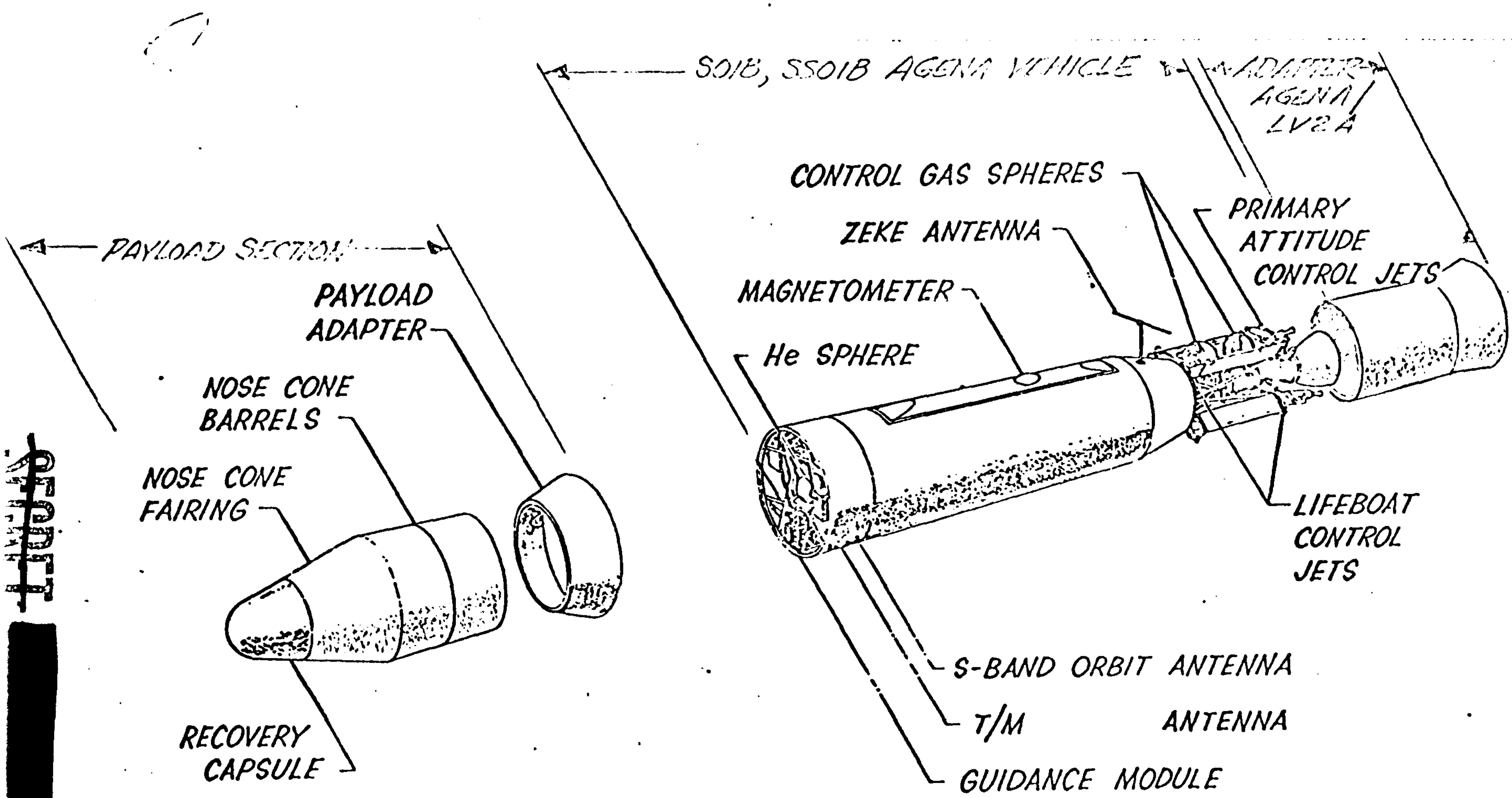


FIGURE 8 - SATELLITE VEHICLE, MAJOR COMPONENTS

and inhibited red fuming nitric acid (IRFNA) oxydizer. The propellant tanks are pressurized with helium to insure proper propellant pump operation and to maintain structural load carrying ability.

An electrical subsystem is provided to supply power to operate vehicle and payload electrical equipment. Batteries are used as the primary power source for the 24 volt direct current supply. Power conversion is accomplished by two switchable three phase inverters supplying single phase power from one leg, and DC to DC converters operating from the unregulated DC bus.

The guidance and control subsystem senses vehicle attitude by means of horizon sensors and an inertial-reference gyro package. Pneumatic reaction-control jets provide the necessary torques to maintain attitude control around the vehicle pitch, roll, and yaw axes. However, during powered-flight the pitch and yaw torques are supplied by engine gimballing activated by hydraulic servos. Vehicle velocity changes are sensed by a velocity meter consisting of an accelerometer and counter which perform the integration function to obtain velocity-gain information. During vehicle ascent and injection, a preset timer controls the sequence of vehicle events. Steering of the vehicle during ascent is accomplished by radio command from a ground based radar tracking and command station utilizing a computer to process tracking data and generate steering commands. A radio guidance command discrete is employed to enable the velocity meter to function.

The tracking, telemetry and command subsystem consists of vehicle-borne transmitters, receivers, decoders, and programmers. Real-time commands are transmitted to the vehicle by PRELORT, ZEKE, and Range Safety Command links. The Precision Long Range Tracking radar (PRELORT) operates in the S-Band frequencies and interrogates a vehicle-borne transponder for tracking. Commanding is accomplished for the Zorro and Analog Systems by modulation of the S-Band link. The Zorro system is digital, and the Analog system employs pulse position modulation using combinations of six audio tones. Analog commands are used for selection of programmed payload and vehicle functions, and the Zorro commands enable the primary re-entry sequence. The ZEKE command system employs a Very-High-Frequency (VHF) link where the carrier is amplitude modulated by audio tones. Zeke commands provide the selection and execution of the back-up "Lifeboat" recovery sequence as well as backup commands for the payload and orbital programmer adjustment. Command Destruct utilizes an Ultra-High-Frequency (UHF) link for range safety flight termination during vehicle ascent only until the time of Thor main engine cutoff.

Telemetry comprises two separate VHF Frequency Modulated (FM) links. Link 1 is used primarily to report vehicle and payload status and environmental data. Link 2 is used to telemeter backup information on payload status and diagnostic data. A tape recorder, utilized for the purpose of acquiring data while the vehicle is beyond the range of ground station contacts, is also played-back over the ground stations on Link 2. A capability exists for providing two additional telemetry links from the vehicle, if required, by installing additional transmitters.

The orbital programmer is used to store commands in the vehicle prior to launch. Four reels of punched 35 millimeter, 1.5 mil thick mylar tape provide vehicle and payload functions to be executed at specific times during the mission. Each reel of tape accommodates 13 brushes to make electrical contact with the grounded drum through the punched holes, thus providing a capability for 52 programmed commands. Tape speed is nominally 9 inches per orbit and tape length can be as long as 192 feet, providing programmed events for approximately 256 orbit revolutions. Tape speed and positioning are adjustable by analog command and backup Zeke commands to synchronize the programmer with the vehicle position and orbital period.

Two solid-state recovery timers are used to store commands for the recovery sequence through separation of the SRV from the satellite vehicle. The primary recovery timer is activated by stored command from the orbital programmer, and the backup recovery timer is activated by a Zeke command to control the Life boat backup recovery sequence.

3.5.1.2.2 Aerospace Ground Equipment (AGE)

The Satellite Vehicle Contractor shall provide the AGE required to check-out vehicle equipment, vehicle subsystems, and the complete satellite vehicle for proper operation within allowed tolerances and for flight readiness. Existing AGE will be used to the maximum extent practicable.

A vehicle systems test shall be conducted at the Contractor's plant prior to acceptance of the satellite vehicle by the procuring agency. The purpose of this test shall be to verify that all vehicle subsystems operate individually and concurrently within specification limits, and that the vehicle being offered for acceptance is flight ready. During the combined subsystems testing, functional test simulators may be used to represent hardware provided by another contractor across a mechanical or electrical interface. However, all simulators shall exhibit proper characteristics of loading and dynamic response as defined by the respective contractors and per interface specifications.

At the launch site, the satellite vehicle shall be inspected to ensure that no damage has been sustained as a result of shipment. Prior to installing the vehicle on the launch stand, functional tests shall be performed for those items of equipment requiring confidence testing at limited time intervals, to maximize the time that the vehicle may be held on stand prior to recycling. After erection on the launch stand and mating with the Stage I Booster, the vehicle shall be checked-out for joint compatibility with the balance of the launch configuration, and when directed it shall be counted-down and launched. Status monitoring shall be provided to support extended holds in a launch-ready status up to 20 days so that the vehicle can be launched within 24 hours when so directed.

Mechanical and electrical AGE shall be provided to implement the above stated concept. Ground handling dollies, slings, and fixtures shall be compatible with vehicle hardware to ensure that damage to vehicle is not incurred as a result of handling. The Satellite Vehicle Contractor shall substantiate the adequacy of his AGE provisions by the preparation and submittal of an AGE Plan. This plan shall provide for functional and reliability analyses of AGE, and for the collection and reporting of AGE failure data. Satellite vehicle AGE shall consider but not be limited to the following items:

A. Transportation and Handling
B. Servicing
C. Functional Test Simulators

D. Weight and Balance Equipment
E. Checkout Test Equipment
F. Loading Equipment for Expendables
G. Ground Electrical Power Equipment
H. Ground Environmental Control Equipment
I. Launch Control Equipment

<u>3.3.1.2.3 Facilities</u>

The Satellite Vehicle Contractor shall identify all facilities requirements necessary to support the factory-to-launch sequence for his end-item hardware. Facility requirements shall be in accordance with the vehicle test and AGE plans. All facility requirements that are uniquely related to the [redacted] Program shall be identified and substantiated. Facility requirements shall consider, but not be limited to the following:

A. Assembly Building
B. Test Facilities
C. Clean Room

<u>3.3.1.2.4 Satellite Vehicle Structural Envelope Requirements</u>

Satellite vehicle equipment shall be contained within the structural envelope of the S01B. Figure 9 presents the general arrangement of primary equipment and major dimensions. Installation of equipments external to the vehicle structural envelope, necessitating the use of additional aerodynamic fairings is to be avoided. The necessity for such installations shall be substantiated and accompanied by detailed analysis of the effect of aerodynamic heating and loads upon the vehicle structure and upon the equipment involved.

<u>3.3.1.2.5 Satellite Vehicle Effectiveness Requirements</u>

The satellite vehicle shall be compatible with functions, schedule, reliability, and utility requirements of the [redacted] System. Equipment and volume utilization shall be related directly to functions necessary in accomplishing the [redacted] mission. Vehicle reliability shall be enhanced by providing redundancy for subsystems and/or components critical to the mission, within the bounds of performance/weight constraints. Interchangeability and maintainability of components, together with logistical support of spares, shall be provided to support the launch and orbital objectives. Requirements for launch-holds and release from the launch pad shall not be more stringent for the satellite vehicle than for the payload.

~~SECRET~~ [redacted]

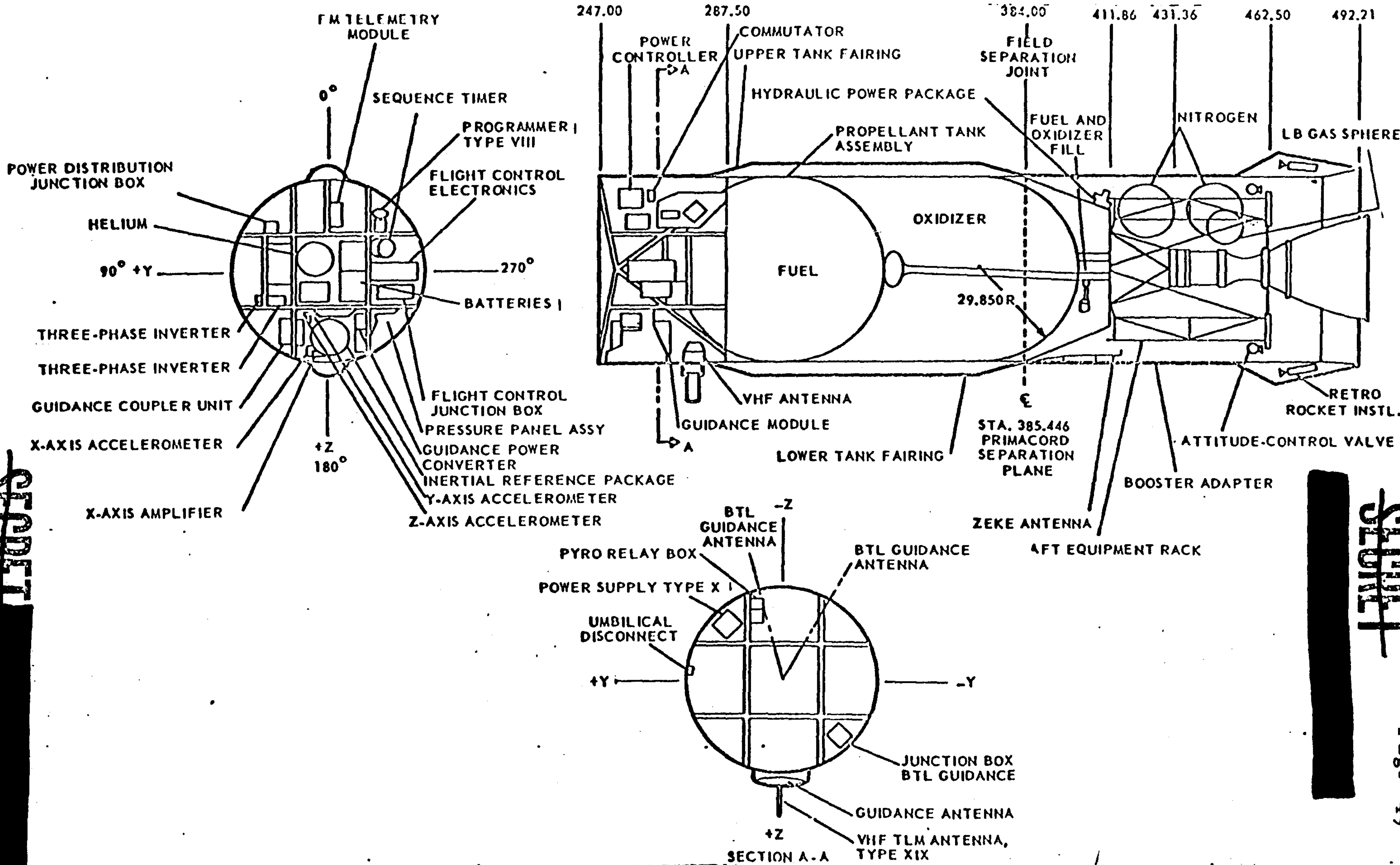


FIGURE 9 INBOARD PROFILE SATELLITE VEHICLE

~~SECRET~~ [redacted]

3.3.1.3 Specific Design Requirements

3.3.1.3.1 Communication and Control Requirements

3.3.1.3.1.1 Command Subsystem (Missile-Borne)

The satellite vehicle command subsystem shall provide real-time and stored commands for controlling all required events from powered flight through separation of the two re-entry vehicles. Critical functions shall be backed up in such a manner as to maximize assurance of successful commanding. A system of command interlocks shall be provided to minimize the effects of inadvertent or covert commands.

A. Real Time Commands

Each real time command shall be accompanied by a functional telemetry verification convenient in real time. Real time commands for Satellite Vehicle functions shall include but not be limited to those listed in Table 1. Real time commands for the payload shall include the specific commands listed in Table 2.

Table 1 Real Time Commands - Vehicle		
Function	No Commands	Command Mode
Orbital Programmer Adjust	4	Analog
Re-entry Orbit Select	2	Analog
Deactivate Vehicle Power	1	Analog
Inverter Select	1	Analog
Backup Recovery Mode Select	3	Zeke
Beacon Transmitter Enable/Disable	1	Zeke
Backup Functions	6	Zeke
Backup Recovery Execute	2	Zeke
Beacon & Telemetry On	1	Zeke
Reactivate Vehicle Power	2	Zeke
Re-entry Enable	2	Zorro

Table 2 Real Time Commands - Payload		
Function	No Commands	Command Mode
V/H Ramp Level Selection	1	Analog
V/H Ramp Amplitude Select	1	Analog
Program Selection On-Off	1	Analog
V/H Ramp Start Delay	1	Analog
Stero Mono Select	1	Analog
Intermix Selection	1	Analog
Intermix Mode Select	1	Analog

Radio Guidance Commands

The radio guidance command link shall use a continuous tracking X-Band rac which shall pulse-position modulate the command spacing between continuous pairs of address pulses.

Command Destruct

The UHF Command destruct subsystem shall be equipped to meet or exceed the minimum requirements of the Western Test Range as specified in AFMTC Regulation 80-7, "Airborne Flight Termination Systems", and as described herein in Section 3.3.1.3.8.

B. Stored Commands

Stored commands shall be provided to initiate vehicle and payload functions during ascent, orbital operations, and during ejection of recovery vehicles. Stored commands for satellite vehicle functions shall include but not be limited to those listed in Table 3. Stored commands for the payload shall include the specific commands listed in Table 4.

Table 3 Stored Commands - Vehicle

Function	No Commands	Command Mode
Ascent Phase		
Booster Separation Backup	1	Ascent Timer
Enable Radio Guid. Steering	2	" "
Engine Ignition Sequence	4	" "
Flt. Control & Attitude Functions	15	" "
Disable Radio Guid Steering	3	" "
Engine Shutdown Sequence	4	" "
Switch TM to Orbit Mode	2	" "
Switch Flt. Control to Orbit Mode	1	" "
Arm Backup Recov. Pyro Pwr.	1	" "
Enable Primary Recovery Select	1	" "
Vehicle 180° Yaw Around	2	" "
Stop Ascent Timer	1	" "
Orbit Phase		
Beacon & TM Link I on	1 Brush	Orbital Progr.
Enable Tape Reset Command	1 "	" "
Tape Index & Subcycle Identif.	1 "	" "
Beacon & TM Link I Off	2 "	" "
Disable Tape Reset Command	1 "	" "
Re-entry Execute-odd Orbits	2 "	" "
TM Link II On	1 "	" "
TM Link II Off	1 "	" "
Pre-launch Tape Index	1 "	" "
Re-entry Execute-even Orbit	2 "	" "
Deactivate		
Flt Control & Attitude Functions	8	Ascent Timer
Power Transfer to External	1	" "
Timer Shutdown Enable& Stop	2	" "
Reactivate		
Flt. Control & Attitude Functions	6	Ascent Timer
Lockout Timer Restart	1	" "
Timer Shutdown Enable & Stop	2	" "
Recovery Phase		
Vehicle Pitch-down Maneuver	4	Recovery Timer
Flt. Control & Attitude Functions	3	" "
Vehicle Pitch-up Maneuver	2	" "
Reset Recovery Timer	1	" "
Remove Rec. Timer Power	1	" "

Table 4 Stored Commands - Payload		
Function	**No Commands**	**Command Mode**
Ascent Phase		
Switch Power for Door Eject	1	Ascent Timer
Switch Power for Camera & Cassettes	1	" "
Orbit Phase		
Orbital Counter	1 Brush(s)	Orbital Progr.
Clock Interrogate	1 "	" "
V/h Programmer Start Pulse	1 "	" "
T/M Enable	1 "	" "
Camera Sequence Program 1	2 "	" "
" " " 2	2 "	" "
" " " 3	2 "	" "
" " " 4	2 "	" "
" " " 5	2 "	" "
" " " 6	2 "	" "
" " " 7	2 "	" "
" " " 8	2 "	" "
" " " 9	2 "	" "
" " " 10	2 "	" "
Redundant Off for All Prog	1 "	" "
Tape Recorder Read-in	1 "	" "
Tape Recorder Read-out	1 "	" "
Tape Recorder Stop	1 "	" "
Telemetry Off	1 "	" "
Deactivate		
Store Camera in Home Position	1	Ascent Timer
Recovery Phase		
Arm No. 1 Capsule Eject	1	Recovery Timer
Transfer Pwr to No. 1 Capsule	1	" "
Fire No. 1 Capsule Eject Squibs	1	" "
Switch Over to B Mission	1	" "
Arm No. 2 Capsule Eject	1	" "
Transfer Pwr to No. 2 Capsule	1	" "
Fire No. 2 Capsule Eject Squibs	1	" "

In addition to the orbital programmer commands listed, 10 brushes shall be available for use with experimental payloads when desired. Stored command equipment requirements shall be as follows:

1. <u>Ascent Timer</u>

The ascent timer shall provide capability to provide discrete time intervals with one to five functional events activated at each interval. The total running time of the ascent timer shall be at least 6000 seconds. Timer setting resolution shall be 1.0 second with a repeatability of 0.2 seconds.

2. <u>Orbital Programmer</u>

The orbital programmer shall provide a capability for 52 brushes, operating with 13 brushes each on 4 reels of punched tape. Tape length shall be compatible with the mission duration, and tapes shall be capable of being advanced or retarded to maintain synchronization of programmed events with the determined position of the Satellite Vehicle. Each of the four reels of tape shall be capable of controlling six equipment functions, on and off, in any preprogrammed arrangement. Accuracy of the orbital programmer shall be plus or minus 3.5 seconds, including the effect of tolerances on tape punching.

3. <u>Recovery Timer</u>

The recovery timer shall provide a capability to initiate at least 14 events. A reset function shall be provided to reset the timer counter to the initial count, and to reset the output relays. Normally, the reset pulse shall be generated by the recovery timer at the time of its last event. Timer accuracy shall be plus or minus 0.5 seconds or 0.1 percent between events, whichever is greater.

<u>3.3.1.3.1.2 Instrumentation and Telemetry</u>

Provisions shall be made to incorporate instrumentation in the Satellite Vehicle to provide timely and accurate data for the pre-launch, launch, orbital and the recovery separation phases of operation. Consideration shall be given to data requirements on a real-time and post-flight basis. The Satellite Vehicle shall provide the telemetry equipment for transmitting vehicle and payload real-time and stored data to the ground stations. FM/FM transmission in the 215 to 260 MC/sec band shall be employed.

A. <u>Sensors</u>

Sensors shall be provided as required for instrumentation of the satellite vehicle. Sensors shall have adequate dynamic range, frequency response characteristics and accuracy to meet the program requirements. Insofar as possible, sensors shall provide an output that can be directly correlated with the calibration points.

C. Reserved Subcarriers for Payload

The payload will require 8 continuous channels, two of which are commutated at a sample rate of 24 bits per second. In addition, 15 commutator points shall be provided at a sample rate of 24 bits per second for redundant payload data.

D. Telemetry Error and Signal

Overall vehicle telemetry subsystem error shall be defined to include, all error sources from the transducer's output terminals to the transmitted RF signal inclusive. Each error contributing elements maximum specified error shall be considered. All such error values shall be squared, the resulting squared values added together, and the square root of the resulting sum shall be taken to define overall error. Any error correction techniques shall not be considered in this definition of error. Maximum overall error for real time, commutated analog data shall be ±3%.

Maximum overall error for similar data which has gone through the cycle of vehicle tape recording and subsequent playback shall not exceed ±5%.

Under conditions of RF signal strength well above threshold, no [illegible] data channel shall contain hum, ripple or noise with a combined amplitude exceeding a 5% peak-to-peak value with respect to a full scale data range.

For telemetry signal reception and processing, receivers with an IF bandwidth of 300 KC/sec will be used. Subcarrier discriminators will utilize standard input tuners and standard output low pass filters. The satellite vehicles' telemetry subsystem design shall be compatible with these conditions.

E. Transmitters

Transmission shall be in the 215 to 260 MC/sec band and conform to IRIG requirements. Transmitter output power shall be a minimum of 2 watts (1.5 watts at the antenna terminal).

F. Subcarrier Oscillators

The subcarrier oscillators shall utilize standard IRIG bands. The maximum subcarrier frequency drift as a result of all causes shall not exceed ±2% of the bandwidth through which the subcarrier's frequency is deviated by full scale data. The subcarriers frequency deviation shall be proportional to the modulating voltage (positive frequency deviation for a positive modulating voltage) with a linearity within ±0.5%. Harmonic distortion shall not exceed ±1%.

A design objective shall be that no subcarrier oscillator shall be capable, under malfunction conditions, of generating an output frequency which interferes with other subcarrier oscillators.

G. Commutators

Commutators shall have proven reliability and compatibility with the SCF. A non-return-to-zero pulse train format shall be a design objective in order to conserve information capability. The total error contribution of any commutator for all combined causes shall not exceed ±1% of full scale. Calibration points shall be provided in each commutator's pulse train which are directly referenced to the applicable instrumentation points.

H. Commutated Data Grouping

Commutated data shall be grouped to facilitate the orbital commanding of the vehicle. All commutated data shall be capable of automatic de-commutation using standard equipment as now provided in the Satellite Control Facility.

I. Calibration Points

Each commutator shall contain calibration points for at least 0%, 50%, & 100%, of the subcarrier bandwidth. Calibration points shall be chosen so as to be compatible with SCF autocalibration techniques.

J. Calibration Books

A calibration book shall be provided covering all satellite vehicle instrumentation. In addition, a similar calibration book, shall be provided covering all payload instrumentation. The calibration books shall contain curves or tables for each instrumentation point relating the magnitude of the physical quantity measured, in engineering terms, to the related subcarrier oscillator's output. The subcarrier oscillator's output shall be expressed both as a percentage of full scale and as an absolute frequency on the same sheet previously mentioned.

The Agena S01B telemetry shall utilize two separate VHF FM links. Standard IRIG proportional bandwidth FM subcarriers shall be used for continuous channels and for commutated data. A representative assignment of data channels is shown in Table 5.

Table 5 Typical Assignment of Data Channels

Channel No.	Mode: Ascent	Mode: Orbit
1-2	Vehicle	Vehicle
3	Vehicle	Vehicle
4	Vehicle	Vehicle
5	Payload	Payload
6	Vehicle	Payload
7	Vehicle	Vehicle
8	Vehicle	Payload Commutated
9	Vehicle	Payload
10	Vehicle	Payload
11	Vehicle	Payload
12	Vehicle Commutated	Vehicle Commutated
13	Payload	Payload
14	Vehicle	Vehicle
15	Vehicle Commutated	Vehicle Commutated
16	Vehicle Commutated	Vehicle Commutated
17	Vehicle Commutated	Vehicle Commutated
18	Vehicle	Payload

Telemetry subsystem specification and the telemetry channel assignment lists for each flight shall require approval of the [redacted] Program Directorate prior to implementation.

3.3.1.3.1.3 Tracking

The primary means of tracking the satellite vehicle will be the S-Band radars operated by the Satellite Control Facility. The satellite vehicle shall contain an S-Band transponder and antenna compatible with the SCF tracking radars. The transponder and antenna system shall provide a system margin of 3 db. at 875 nautical miles slant range when operated with the ground equipment described in Section 3.3.7. The tracking transponder shall have a proven flight reliability.

3.3.1.3.1.4 Stored Data

A magnetic tape recorder/reproducer, having proven flight reliability, shall be included in the satellite vehicle for the purpose of storing vehicle and payload data during periods of time when the vehicle is not within range of an SCF ground station. The tape recorder shall have dual track data recording capability with a read-in-to-readout ratio of 26 to 1. The maximum readin time shall be 182 minutes from a 1 x 60 or a 0.4 x 60 commutator with equivalent readout time of approximately 7 minutes. The signal response shall be 300 cps, or DC to 60 pps commutated.

3.3.1.3.1.5 <u>Link Security Requirements</u>

Any stored command that could allow a controlled de-orbit maneuver shall be secured against unauthorized execution. If access to the communications were to be gained for a total of sixty passes, each having a six minute duration, the probability of controlled de-orbit of an SRV by an unauthorized agent shall be less than 10^{-5}.

It shall be a design objective to minimize the vulnerability of the system to unauthorized commands inserted with the intention of sabotaging the mission.

3.3.1.3.2 Equipment Environmental Requirements

The design, construction, and qualification of satellite vehicle equipment shall consider the environments described in Sections 3.1.3.3.1, 3.1.3.3.2, and 3.1.3.3.3. All Agena S01B equipment shall conform to the requirements of LMSC 6117B, "General Environmental Specification for the Agena Satellite Program," except as specifically authorized.

3.3.1.3.3 Guidance and Attitude Control Requirements

Attitude control shall be provided to stabilize the satellite vehicle during the last stage of powered flight, including any unpowered coast phases; during the on-orbit operation; and during the de-boost phase to initiate re-entry of the satellite recovery vehicles.

It is desired to use as few guidance and control components as possible. Hence, the same components should be used on-orbit as in the ascent and de-boost phases within the bounds of practicability and reliability. It shall be a system requirement to back-up critical de-boost sequences with a redundant attitude control and orientation subsystem.

Attitude control requirements for the above mentioned phases of flight shall be as follows:

A. Ascent Phase - At termination of Stage I Booster thrust by guidance command, the satellite vehicle ascent timer shall begin operation. Subsequently, at the time of Stage I vernier engine cutoff, the inertial reference gyros in the satellite vehicle shall be uncaged and the horizon sensor fairings ejected. A coast phase shall be initiated upon separation of Stage I from the satellite vehicle by radio guidance discrete. Immediately following physical separation, control of the satellite vehicle attitude and rates about all three axes shall be initiated, utilizing the vehicle reference attained at VECO. The horizon sensors shall reference the roll axis to the earth horizon. Engine ignition is initiated by signal from the ascent timer and engine shut-down is initiated by the velocity meter after the predetermined velocity-to-be-gained is achieved. The velocity meter is enabled by radio guidance discrete. During the burn period, pitch and yaw control are provided by hydraulic actuation of the gimballed engine while roll control is maintained by pneumatic reaction control jets. Radio guidance command is utilized through the major portion of the burn period to provide pitch and yaw steering commands to the satellite vehicle. At orbit injection, the required accuracies for [redacted] missions are as given in Table 6.

Table 6 Dispersions at Orbit Injection

Parameter	Requirement (3σ)	Objective (3σ)
Orbital Period	$\pm$ 0.20 min.	$\pm$ .025 min.
Altitude of Perigee	$\pm$ 5 n.m.	$\pm$ 1.5 n.m.
Argument of Perigee	$\pm 20^\circ$ ($e > .008$)	$\pm 5^\circ$ ($e > .008$)
	$\pm 180^\circ$ ($e \leq .008$)	$\pm 45^\circ$ ($e \leq .008$)
Inclination Angle	$\pm 0.25^\circ$	$\pm 0.25^\circ$

B. <u>Orbital Phase</u> - Subsequent to orbit injection, the satellite vehicle shall be yawed at a programmed rate, of approximately 60 degrees per second, through a total yaw angle of 180 degrees. Throughout the orbital mission, excluding de-boost sequences, the satellite vehicle shall remain oriented in the "tail first" position with the roll axis of the vehicle essentially aligned with the velocity vector in the orbit plane, and normal to the earth radius vector.

The following pointing accuracies and angular rates shall include all the errors from the true local vertical and true orbit plane to the reference camera axis. As such, these accuracies include the attitude control accuracies and the alignments from the control system to the payload optical or mechanical reference axes. While the payload equipment is operating, the satellite vehicle will be subjected to the following maximum torques:

Axis	Torque
Pitch	7 ft lb
Yaw	15 ft lb
Roll	2 ft lb

Poi ng accuracy requirements and maximum limit cycle rates are given in Tab 7.

Table 7	Pointing Accuracy and Rates	
Function*	Requirement (3σ)	Objective
Pitch Attitude	$\pm 1.5^{\circ}$	$\pm 0.75^{\circ}$
Yaw Attitude	$\pm 2.0^{\circ}$	$\pm 1.10^{\circ}$
Roll Attitude	$\pm 1.5^{\circ}$	$\pm 0.75^{\circ}$
Pitch Rate	.016 deg/sec	.008 deg/sec
Yaw Rate	.016 deg/sec	.008 deg/sec
Roll Rate	.022 deg/sec	.011 deg/sec

The above accuracies are required during payload operation. At times of no payload operation, reduced accuracies are acceptable. Additional to the above requirements for stabilization, the satellite vehicle shall be capable of yaw pointing accuracies within $\pm 0.25^{\circ}$ per step yaw affect with a stabilization time of 6 minutes per yaw step, in response to the payload yaw programmer when required.

During Deactivated periods of on-orbit operation, the satellite vehicle shall be programmed into a tumbling rate adequate for stabilization of temperatures. For the Agena S01B, the attitude control equipment shall be capable of providing a 3 degree per second ± 0.3 degree per second tumble rate about the pitch axis at the time of deactivation. The deactivate sequence is controlled from the ascent timer and is initiated by a real-time command. During deactivate periods, provisions shall be made to maintain the minimum environment, or to limit the exposure periods of temperature-sensitive attitude control equipment.

Reactivation of the satellite shall be initiated by a real-time command, with the subsequent sequence controlled by the ascent timer. Upon reactivation, the satellite vehicle shall restabilize to the active on-orbit attitude requirements within a period of 90 minutes (99% confidence level). Reactivation normally shall be initiated over the same ground station from which the deactivate command was transmitted.

C. De-Boost Phase

The de-boost sequence for the satellite vehicle is controlled by signals from the recovery timer. Upon the programmed command, the satellite vehicle shall pitch down a nominal 60 degrees from the local horizontal, while orbiting in the "tail-first" attitude, and hold this attitude with respect to the local horizontal until the recovery vehicle has been ejected. The time required to pitch down shall not exceed 60 seconds. After recovery vehicle ejection, the satellite vehicle shall return to normal on-orbit pitch attitude. Tolerances for attitude referenced to the local horizontal and orbit plane while in the pitch-down condition are given in Table 8.

Table 8 Pointing Accuracy, Pitched-Down Attitude

Function	Requirement (3σ)
Pitch Angle from Local Horizontal	$\pm 3.06^{\circ}$
Yaw Angle from Orbit Plane	$\pm 1.44^{\circ}$
Roll Angle from Radius Vector	$\pm 1.32^{\circ}$

In the event of a malfunction in the primary attitude control subsystem, the back-up stabilization subsystem (Life boat) shall be activated by real-time command. The Life boat subsystem shall be capable of redundantly performing all de-boost sequences necessary for properly ejecting one re-entry vehicle from the satellite vehicle. Upon initiation, the Life boat subsystem shall be capable of orienting the satellite vehicle from a tumbling mode of 20 degrees per second about any axis, attain a stabilized attitude within 90 minutes, and be capable of holding the de-boost orientation for a minimum of 30 seconds. Life boat attitude control is established by lining up the vehicle roll axis with the local magnetic vector. Life boat shall be capable of acceptable performance on North to South passes, and the ability to perform acceptably on South to North passes is desirable. Pointing accuracy required for Life boat is

$\pm 6.4^{\circ}$ maximum deviation, both in and out of the orbit plane referenced to the required 58 degree pitch down orientation. Vehicle re-orientation after Life boat is not required.

D. Satellite Vehicle Mass Characteristics

The following estimates of satellite vehicle mass characteristics are provided in Table 9 for preliminary attitude control design.

Table 9 Estimated Mass Properties, Satellite Vehicle

Condition	Wt (lbs)	Center of Gravity X-Station (in)	Moment of Inertia (Slug ft^2) I_y (Pitch)	I_z (Yaw)	I_x (Roll)
Separation from Stage I Booster	17329	329.5	14445	14429	277.3
Ignition Weight	17321	329.6	14467	14451	275.4
Burnout Weight	3796	277.1	8946	8930	274.5
Wt. on-Orbit	3747	275.5	8815	8800	274.1

3.3.1.3.4 Power Supply Requirements

The electrical power and distribution subsystem for the satellite vehicle shall comprise a direct current power source, a power distribution network, and power conversion and regulating equipment to satisfy the requirements of vehicle and payload operation. The power supply shall normally consist of batteries having sufficient capacity to supply electrical energy for all vehicle and payload requirements from lift-off, through orbital flight, and until separation of the second re-entry vehicle. Requirements and characteristics of the satellite vehicle electrical power subsystem shall be as follows:

A. Power Source - The power source, consisting of an adequate number of batteries, shall provide the electrical energy required to support vehicle and payload equipment for the assigned mission operations. Th[illegible] wer source shall maintain the unregulated voltage between limits of 22 an[illegible] 5 volts D.C. A capability to provide electrical energy for mission durations of 11 days shall be inherent in the design of the satellite vehicle, but employment shall be contingent upon mission requirements and the performance available during ascent flight. The number and type of batteries to be carried on any particular flight shall be based on a detailed electrical loads analysis for that mission. In satisfying the load requirement, the power source configuration shall consider the following factors:

a) Three sigma low capacity of the battery (a 99.87 percent probability of meeting or exceeding the rated power capability).

b) Battery temperature environment and the consequent effect on usable battery capability.

c) Activated stand time capabilities of the battery versus time from battery activation through mission completion.

d) Electrical power profile characteristics including surge characteristics of pyrotechnic devices and payload equipment.

e) Battery weight as a function of mission duration and allowable weight versus performance capability for the ascent flight.

f) A power margin of 5 percent over and above predicted load requirements shall be provided for each flight.

B. Payload Electrical Requirements

The satellite vehicle shall supply electrical power to the payload as stated below and in accordance with the Satellite Vehicle/Payload Interface Specification. Estimated maximum payload power requirement is 1100 watt hours per day with the following requirements:

a) Unregulated DC with an average load of 20 watts continuous, 350 watts for periods up to 20 minutes, and 450 watt peaks of less than 5 seconds.

b) Regulated DC (+) 28 $\pm$ 1/2 volt with average load of 10 watts continuous, and 60 watts for periods up to 20 minutes.

c) Regulated DC (-) 28 $\pm 1/2$ volt with an average load of 15 watts during telemetry subsystem operation.

d) 400 $\pm$ .004 cycles at 115 ± 2 volts AC with an average load of 3 watts continuous and 13 watts for periods up to 20 minutes.

e) Unregulated voltage for pyrotechnic actuation with peak currents of 60 amperes and peak duration of 100 milliseconds.

C. Electromagnetic Interference Control

The satellite vehicle shall be designed to and comply with MIL-E-6051C. All subsystems of the satellite vehicle shall comply with MIL-I-26600 and MIL-E-6051C except as specifically authorized: components shall comply with MIL-I-26600. An electrical interference control plan shall be originated and employed in the design, fabrication and test of the satellite vehicle.

D. Electrical Wire Harnesses

The electrical wire harnesses shall provide suitable electrical paths for the distribution of electrical power and signals to satellite vehicle and payload components and major elements. The harnesses shall be designed and installed on a subsystem concept. The wiring shall be such as to minimize noise and interference problems through use of twisted pairs, shields and coaxes. The harnesses shall be suitable for use in a vacuum environment. The maximum voltage drop in any individual circuit from battery to using component and return, attributable to the harness including connectors from battery to component, shall not exceed 0.5 volts DC. Voltage drops in primary leads of up to 1 volt are permissable where this can be shown to be consistent with voltage requirements at the component, and does not involve common wiring resistance of two or more components leading to an interference problem.

All pyrotechnic loads shall be connected to a separate, diode-isolated primary battery. These circuits shall be protected prior to launch by means of safe/arm plugs in the junction boxes for the pyrotechnic circuits. Additionally, fusistors shall be provided in the pyrotechnic circuit junction boxes to protect the vehicle power source and distribution network from short circuits that may occur after pyrotechnics firing. Wiring circuits to pyrotechnics and return shall be capable of handling four times the minimum all-fire current of the pyrotechnic device. Power and pyrotechnic harnesses may be grouped and routed together, but shall be separated from harnesses for instrumentation and test plugs.

E. External/Internal Power Transfer Switch

An external/internal power transfer switch shall be used beforeliftoff to transfer the AGE power supply to the satellite vehicle operational power supply. The transition shall be made so as to have no deleterious effect on the satellite vehicle's components or subsystems operation.

F. Grounding, Bonding and Shielding

Grounding, bonding and shielding of electrical subsystem components shall be accomplished in accordance with the applicable MIL Specifications.

G. Interface Connectors

The satellite vehicle shall provide electrical connectors at the forward interface for the payload at the aft interface for the Stage I booster. Separate interface connectors shall be provided for each of the following functional categories:

a) Pyrotechnics
b) Electrical Power
c) Commands
d) Telemetry Instrumentation

3.3.1.3.5 Structural Design Provisions

The design of the satellite vehicle shall provide mounting provisions for all subsystem equipment in a manner that affords compatible equipment environments, mass property compatibility, and structural integrity over the full range of mission conditions. Design provisions shall consider but not be limited to the following items:

A. Equipment accessibility
B. Aerodynamic heating protection

C. Ascent venting requirements

D. Pyrotechnics provisions and environments

E. Orbital environments and preservation of thermal surface radiation characteristics.

F. Attachment and separation provisions

G. Dynamic environments and physical clearances.

3.3.1.3.6 Propulsion Requirements

The satellite vehicle shall contain a propulsion system to provide thrust for second stage boost during ascent into orbit. The propulsion system shall consist of a rocket engine and components required to develop a nominal vacuum thrust of 16,000 lbs. The engine shall be designed for a nominal thrust duration of 245 seconds. The rocket engine thrust chamber shall be mounted on a gimbal ring and shall provide partial satellite vehicle attitude control during engine operation by means of yaw and pitch thrust chamber movement. A cylindrical shaped propellant tank shall be provided to contain a nominal propellant load of 13,560 lb. Propellants shall consist of Inhibited Red Fuming Nitric Acid (IRFNA) as an oxidizer, and Unsymmetrical Dimethylhydrazine (UDMH) as a fuel.

For the [redacted] Mission, a single propulsive interval will normally be required for the satellite vehicle, and restart of the rocket engine will not be required. Engine ignition time shall be within ± 0.7 seconds of the programmed value, and the engine shutdown impulse shall not exceed ± 600 lb-seconds. The Satellite Vehicle Contractor shall perform a performance error analysis for the purpose of defining the necessary propellant contingency required to accommodate guidance and performance tolerances contributing to performance degradation. Normally a propellant contingency shall be reserved in the satellite vehicle to cover the root-sum-squared effect of minus 3 sigma dispersions. If the Stage I Booster is utilized essentially to propellant depletion, the contingency carried in the satellite vehicle shall provide for flight from lift-off through orbit injection. However, if flight data should indicate that predicted dispersions are significantly conservative, it shall be permissible to readjust the propellant contingency to a more realistic value with prior approval of the [redacted] Program Directorate. Performance margin and propellant contingency shall be verified prior to each [redacted] vehicle launch. A [redacted] Program launch shall not be conducted under conditions that yield negative performance margin, or a propellant contingency corresponding to less than a 90 percent probability of achieving the desired orbit.

3.3.1.3.7 Guidance Requirements - Ascent

The satellite vehicle shall contain the vehicle-borne radio guidance group or command steering of both the Stage I Booster and the Satellite Vehicle during ascent flight. The vehicle-borne guidance group shall include a radar transponder to aid ground tracking, a command receiver, and circuitry for utilizing the RF commands to control the necessary functions of the satellite vehicle guidance and control subsystem. Command signals shall be provided from the satellite vehicle across the interface to the Stage I booster in accordance with provisions of the LV2A/S01B Electrical Interface Specification.

The function of the radio guidance shall be to increase guidance accuracy by providing real-time sequenced events and real-time steering corrections to the Thor and Agena vehicles during their boost phases. Steering corrections implemented by the radio guidance shall be in the nature of vernier corrections, and if none are received, the vehicle guidance and attitude control subsystem shall continue to function in the pre-programmed mode. Similarly, the sequenced events for separation of Stage I from the satellite vehicle, and start of the satellite vehicle ascent timer shall be actuated by programmed stored commands if they are not commanded by radio guidance. Thor MECO and Agena velocity meter enable are not backed-up by programmed command, and the engine shut-downs will occur upon propellant depletion.

Accuracy requirements for the ascent flight are given in Section 3.3.1.3.3.

3.3.1.3.8 Flight Termination Subsystem - Ascent

The satellite vehicle flight termination subsystem shall be capable of destructing the vehicle and payload in-flight upon command while attached to the Stage I Booster, or automatically in the event of an inadvertent premature separation from Stage I during ascent. The command destruct UHF receivers shall be carried in the Stage I Booster. The satellite vehicle shall provide the capability for carrying two redundant sets of destruct signals across the interface from Stage I. Additionally, the satellite vehicle shall provide the destruct pyrotechnic charge, the power to activate the charge upon receipt of a destruct signal, the activating destruct switch for inadvertent separation, and all necessary disarming circuitry to safe the satellite vehicle prior to launch and subsequent to Stage I boost.

The S01B flight termination subsystem shall be equipped to meet the minimum command destruct requirements of the Western Test Range as specified in AFMTC Regulation 80-7, "Airborne Flight Termination Systems." Command destruct signal provisions shall be in accordance with the LV2A/S01B Electrical Interface Specification.

3.3.2 Payload

3.3.2.1 Budgeted Performance and Design Requirement

3.3.2.1.1 Payload Weight Budget

Total weight of the J-System payload as defined herein shall not be more than 1445 lbs. at launch. Budgeted weight of payload components shall be apportioned to the following items:

A. Pan/Stereo Camera and Transport Rollers
B. Main Supply Cassette
C. Main Film Load
D. Stellar/Index Cameras (2)
E. S/I Film Load
F. Film Takeup Cassettes (2)
G. Digital Recording Clock Generator
H. Electrical Controls and Harnesses
I. Payload Section Structure
J. Satellite Re-entry Vehicles (2)
K. Instrumentation and Telemetry Harness

3.3.2.1.2 Payload Reliability Budget Requirements

The reliability design objective for the J-System payload is 0.94 for operation in the active orbital mode for a period of at least eight days. This is exclusive of the two SRV's operation subsequent to ejection from the payload section during the deboost sequences. However, the reliability goal shall include the probability of payload and SRV equipment surviving the ascent boost environment and subsequent inactive periods on-orbit until functional operation is required.

The Payload Integration Contractor shall apportion the budgeted reliability goal to the following major payload components:

A. Camera Equipment
B. Payload Section and Controls

Equipment design and component selection shall consider budgeted reliability goals. Additionally, the payload design shall consider reliability improvements provided through redundancy. The Payload Integrating Contractor shall record and maintain failure-rate data on all critical payload equipment for the purpose of reporting current reliability performance estimates for end-item hardware.

3.3.2.2 Payload General Design Requirements

3.3.2.2.1 General Description

The payload section comprises a cone-cylinder structure housing the camera equipment, satellite recovery vehicles, and the necessary payload control equipment and electrical cabling. The payload section shall mate to the forward bulkhead of the satellite vehicle as shown in Figure 1.

The primary camera equipment shall consist of two hi-acuity panoramic cameras mounted in a 30 degree converging stereoscopic configuration. Simultaneous operation of both cameras provides stereoscopic photography, or individual operation of either camera provides monoscopic photography. Each of the two cameras incorporates a 24 inch focal length, f3.5, Petzval lens system. During camera operation, the lens assembly rotates about its vacuum nodal point through 360 degrees to perform the pan function. The rotational motion of the lens is matched to a reciprocating scan head at the proper angular position to initiate an exposure sweep across the length of film which is held in position on the film track. As the exposure slit in the scan head traverses the main film frame, the two horizon-sighting cameras, located one on each end of the camera film track, record a vehicle attitude reference photograph on the film adjacent to the panoramic frame. The horizon cameras operate on alternate cycles of the panoramic camera. Time reference data and identifying information are recorded on the edge of the pan frame by a data block. Lens rotation, film transport, and an inertia counterbalance are powered from the camera electric drive-motor through a system of cams, belts and gears. Image motion compensation is provided by a cam that translates the lens assembly and scan head along its rotational axis at a velocity proportional to camera cycle rate. Camera cycle rate and the IMC programmer are directly proportional, selectable by command of the IMC programmer to match the actual V/h condition.

Film tension is provided on each side of the panoramic cameras by drive motors in the supply and takeup cassettes. Film from both pan cameras enters the rear recovery vehicle, where the takeup spools are locked to prevent rotation during the "A" mission, return from the rear SRV and are taken up on the cassette spools in the forward SRV. Upon command, the film entering the forward SRV is cut, the spools in the rear SRV unlocked, and takeup initiated for the "B" mission.

A separate Stellar/Index camera is installed adjacent to each SRV, one each providing terrain and stellar photography for "A" and "B" missions. The dual film of each Stellar/Index camera is taken up on two motor driven spools at the base of the takeup cassette. Stellar/Index photographs are nominally taken once for every seven frames of panoramic photography. A digital clock provides time data which is recorded on the pan frame data block for positional correlation of the Satellite Vehicle.

Programs for camera operation are provided as stored commands on the orbital programmer. Program selection, camera selection, stereo or mono mode selection, and V/h compensation are all provided through real-time commands.

At the conclusion of the "A" mission, the forward re-entry vehicle is deboosted on the desired recovery orbit. Initiation of the "B" mission shall not be contingent upon prior separation of the "A" SRV. The fairing between the forward and aft SRV's is retained on the Satellite Vehicle until jettisoned by the arm signal for deboosting the second SRV. The satellite re-entry vehicles are the modified Mark 5A recovery subsystems described in Section 3.3.3.

3.3.2.2.2 Aerospace Ground Equipment (AGE)

The camera equipment and satellite re-entry vehicles will be provided as Government Furnished Equipment (GFE) to the Payload Contractor for integration into the mission configuration. The flight payload shall be [illegible] tested at the Payload Contractor's facility prior to shipping to the launch base. It shall be a program objective to provide the payload section with all equipment in a flight-ready condition prior to shipping to the launch base. During transportation and payload handling, tension of the film drive shall be maintained.

At the launch site, the payload shall be inspected to ensure that no damage has been sustained as a result of shipment. Inspection shall require a minimum degree of disassembly to avoid invalidating the condition of the payload at the time of acceptance. Functional and compatibility tests shall be performed between the payload and the Satellite Vehicle at the time the payload is mated.

Subsequent to mating, status monitoring shall be provided to support either a countdown or extended hold in a launch-ready configuration. The payload shall be capable of being held in the latter condition for periods up to 20 days with the ability to be launched within 24 hours when so directed. At the launch pad, the payload shall be continuously conditioned at temperatures and relative humidity levels compatible with camera and film requirements. During extended standby, the payload environment shall be maintained between 40° F and 65° F with a relative humidity of 30 percent or less.

Mechanical and electrical AGE shall be provided at the Contractors' facilities and at the launch base to implement the above stated concept. Ground handling equipment, slings and fixtures shall be compatible with payload hardware to ensure that damage to the payload and its equipment is not incurred as a result of handling or shipping. The Payload Contractor shall prepare and submit a plan for the integrated testing of payload equipment including the AGE requirements necessary to support the plan. This plan shall provide for functional and reliability analyses of AGE, and for the collection and reporting of AGE failure data. Payload AGE shall consider, but not be limited to the following items:

A. Transportation and Handling
B. Setting and Checking Optical Alignments
C. Weight, Balance and Ballasting
D. Functional Test Simulators
E. Film Splicing and Loading Equipment
F. Checkout Test Equipment
G. Ground Electrical Power Supplies
H. Ground Environmental Control Equipment
I. Pre-launch Status Monitoring

3.3.2.2.2 Facilities

The Payload Contractor shall identify all facilities requirements necessary to support the factory to launch sequence for the payload. Facility requirements shall be in accordance with the payload test plan and AGE requirements. All facility requirements that are uniquely related to the [redacted] Program shall be identified and substantiated. Facility requirements shall consider, but not be limited to the following:

A. Assembly Building

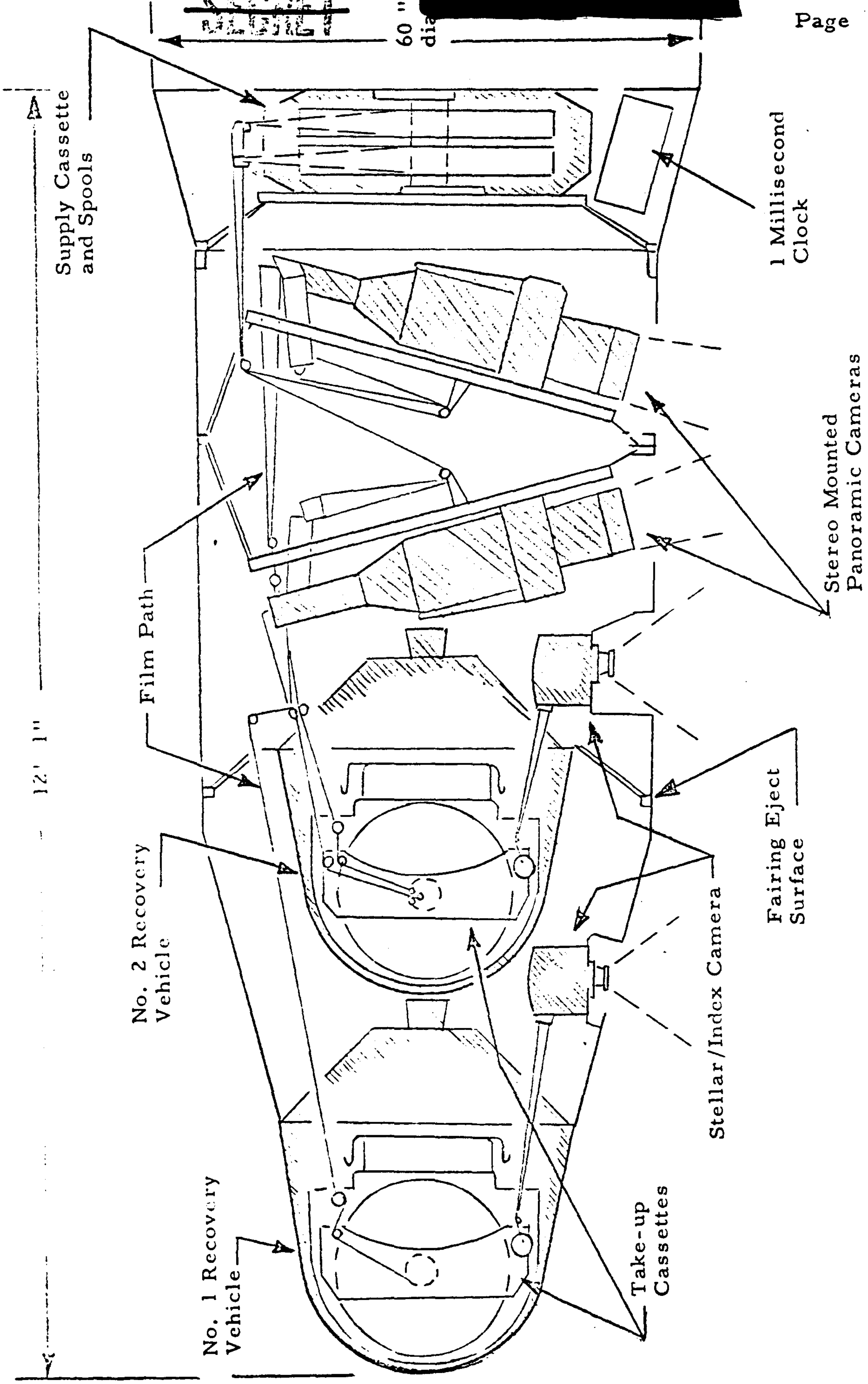
B. Test Facilities

C. Clean Room

3.3.2.3 Specific Design Requirements

3.3.2.3.1 Payload Structural Envelope Requirements

Payload equipment shall be contained within the structural envelope of the payload section of the Satellite Vehicle. Figure 10 presents the general arrangement of primary payload equipment and major dimensions. The maximum diameter of the payload section shall not exceed the outside diameter of the Satellite Vehicle at the mechanical interface station. Installation of payload equipment external to the payload structural envelope with consequent addition of aerodynamic fairings is to be avoided. The necessity for such installations shall be substantiated and accompanied by detailed analysis of the effect of aerodynamic heating and loads upon the payload structure and upon the equipment involved.


60 " dia
Supply Cassette and Spools
1 Millisecond Clock
Stereo Mounted Panoramic Cameras
Film Path
12' 1"
Fairing Eject Surface
Stellar/Index Camera
No. 2 Recovery Vehicle
No. 1 Recovery Vehicle
Take-up Cassettes

3.3.2.3.2 Communications and Control Requirements

All real-time and stored commands required for payload operations from lift-off through separation of the SRV's in the deboost phase shall be provided by the Satellite Vehicle command subsystem. The satellite vehicle shall also provide the telemetry link for payload instrumentation data prior to alunch, and from liftoff through separation of SRV's. The satellite re-entry vehicles shall each contain a tracking and telemetry subsystem for operation during the re-entry/recovery phases as described in Section 3.3.3.

3.3.2.3.2.1 Payload Commands

Payload operation shall be capable of being programmed for any portion of the ground track on any orbit during the active mission life, except during the deboost maneuver of the satellite vehicle. Preprogrammed stored-commands shall be capable of adjustments through real-time commands to match the payload periods of operation and photographic equipment functions to actual orbital and ground track conditions during the mission. Command functions shall provide adequate flexibility to accomodate the combined effects of ascent flight dispersions, and variations in orbital parameters due to atmospheric drag and earth oblateness. Payload real-time commands shall be accompanied by a functional telemetry verification convenient in real-time. Real-time and stored command requirements for the payload are stated in Section 3.3.1.3.1.1, and shall be defined in detail in the Satellite Vehicle/Payload Interface Specification.

3.3.2.3.2.2 Instrumentation and Telemetry

Provisions shall be made to incorporate instrumentation in the payload to provide timely and accurate data for pre-launch, launch, orbital, and recovery separation phases of operation. Consideration shall be given to data on a real-time and post-flight basis. Telemetered data shall provide information adequate for determining equipment mode status and state-of-health, and shall consititute a basis for real-time command decisions and malfunction diagnosis. Payload telemetry data shall consider, but not be limited to the following functions:

A. Camera Program Select Position
B. Camera Operating Mode
C. Camera Cycle Counter
D. Film Footage Transferred to Takeups
E. Clock Serial Readout
F. Camera Cycle Rate and V/h Settings
G. Regulated and Unregulated Supply Voltages
H. Equipment Operating Voltages

I. Light Leaks

J. Equipment Critical Temperatures

K. Payload Section Internal Pressure

L. Door and Fairing Separation Events

M. SRV Separation Sequence Events

The telemetry link for payload data shall be provided by the satellite vehicle by means of FM/FM transmitters located on the vehicle side of the interface. The satellite vehicle shall also provide a tape recorder to accomodate storage of payload data with subsequent playback over SCF ground Stations. Payload instrumentation signals shall be suitable conditioned for compatibility with telemeter equipment in the satellite vehicle. Requirements for payload telemetry are stated in Section 3. 3. 1. 3. 1. 2, and shall be defined in detail by the Satellite Vehicle/Payload Interface Specification.

3. 3. 2. 3. 3 Equipment Environmental Requirements

The J-System shall orbit the earth in the vacuum environment existing at altitudes from 90 to 240 nautical miles under conditions of solar radiation varying from direct sunlight to earth shadow. Photographic equipment will normally be operated at the lower orbit altitudes and under conditions of direct sunlight on the satellite vehicle.

The payload section of the satellite vehicle shall provide the necessary mountings for payload eqiuipment. Mounting provisions shall be capable of achieving the high degree of alignment accuracy required for camera optics and film transport, and for maintaining alignments under conditions of booster-induced environments during ascent and throughout the subsequent orbital phase. The payload section shall provide on-orbit environmental conditions of a special nature as required for the operation and survival of the photographic equipment. These shall be as follows:

A. The cameras and film tracks shall be provided with a light proof environment, free of all light leakage that could produce detectable film fogging.

B. A pressure environment shall be provided to supress corona discharge. The pressure makeup unit shall be capable of maintaining pressures of20 microns or higher in the payload cylindrical sections during camera operation.

C. Detachable doors shall be provided in the payload structure for viewing ports. The doors shall provide protection to optical equipment during ascent, shall be ejected prior to orbit injection of the satellite vehicle. The structure shall also provide boots or other similar devices to seal the camera equipment from external light.

D. Thermal control shall be provided for temperature-critical equipment. The external surface of the payload structure shall be utilized for passive temperature control. An optimum absorptivity/emmisivity shall be provided by surface coatings and mosaic patterns to maintain an average temperature of $70^{\circ} \pm 30^{\circ}$ Fahrenheit inside the cylindrical section of the payload.

Payload structure and equipment design, construction, and qualification shall consider the above stated requirements and the environments described in Sections 3.1.3.3.1, 3.1.3.3.2 and 3.1.3.3.3. Payload equipment shall conform to the requirements of LMSC 6117B, "General Environmental Specification for the Agena Satellite Program," except as specifically authorized. Equipment environmental requirements for the Satellite Re-entry Vehicles are covered in Section 3.3.3.

<u>3.3.2.3.4 Power Supply Requirements</u>

The satellite vehicle shall supply the electrical power required for operation of payload equipment. Electrical energy shall be made available for the duration of active mission time at a rate compatible with the payload electrical duty cycle. Power regulation and conversion shall also be provided by the satellite vehicle electrical subsystem to satisfy payload requirements. Payload electrical power requirements are stated in Section 3.3.1.3.4 and shall be specified in detail in the Satellite Vehicle/Payload Interface Specification.

The payload section shall provide the power distribution network for all payload equipment forward of the Satellite Vehicle/Payload interface. This shall include all junction-boxes, cables and connectors necessary for the control and monitoring of payload equipment. Cabling shall be provided for each of the following functional categories:

A. Pyrotechnics

B. Electrical Power

C. Commands

D. Telemetry Instrumentation

Requirements for the design of payload electrical wire harnesses, electromagnetic interference control, grounding, bonding, shielding, and interface connectors shall be compatible with and not less stringent than those specified for satellite vehicle design in Section 3.3.1.3.4.

3.3.2.3.5 Structural Design Requirements

Payload structural design shall afford a light-tight environment with provisions for achieving and maintaining a high degree of alignment accuracy for the panoramic cameras and film tracks. Design of the payload structure shall be compatible with the payload equipment environmental requirements specified in Section 3.3.2.3. Structural Design provisions shall be compatible with those specified in Section 3.3.1.3.5 for the satellite vehicle.

3.3.2.3.6 Design Requirements - Payload Equipment

A. Panoramic Camera

The two panoramic cameras shall be capable of operating simultaneously to generate stereoscopic photography or separately to generate monoscopic photography. Film width shall be 70 millimeters. The cameras shall have a sustained operational capability of 20 minutes operation per single orbit. Panoramic scan angle shall be 70 degrees, and image motion compensation shall be provided during scan exposure. Data required to identity mission reference time, camera serial number, stellar/index frame correlation, and end or start of pass shall be recorded during exposure of panoramic frames. Corona marking shall be limited to the first five consecutive frames from the end of pass mark at each instrument startup.

Specific requirements shall be as follows:

(a) Resolution

Each panoramic camera shall demonstrate a minimum dynamic resolution of 90 lines per millimeter utilizing a USAF standard low contrast (2 to 1) test target and 100 percent match of image motion compensation. Eastman Kodak 4404 film and Wratten 21 filters shall be used for this demonstration.

(b) Lens

The lens shall be a Petzval type with an aperture of f/3.5; a focal length of 24 inches ± 2 percent; and a field of view of 5°. The lens shall have a focus shift of less than $\pm$ .001 inches over a temperature range of $60^{\circ}F$ to $80^{\circ}F$ when the camera is installed in payload flight configuration. The lens shall be focused for orbital vacuum conditions.

(c) Image Motion Compensation

Each panoramic camera shall incorporate an IMC mechanism to provide a relative velocity between the lens cell and the film during scan exposure. The relative velocity shall be in a direction measured along the rotational axis of the lens cell, and shall be adjustable by command during orbital operations. The IMC mechanism shall be designed so that when properly matched to ground angular velocity, a 7.4 percent overlap of panoramic main frames shall result. The range of IMC adjustment shall accomodate orbital altitudes from 90 to 240 nautical miles with orbit eccentricities ranging from zero to 0.033.

(d) Cycle Period

Camera cycle period shall be adjustable by command in flight over the range from 2.15 seconds to 6.5 seconds. The two panoramic cameras shall operate at a cycle rate within 3 percent of each other. Cycle rates shall be repeatable within 1 percent at any given reference level.

(e) Scan Drive

The scanning drive mechanism shall drive the lens cell and scan arm in synchronism over the 70 degree panoramic scan angle. Velocity of scan shall be controlled so as to produce no visible banding in ground scenes. The drive subassembly shall be of modular construction for ease of installation and replacement. During the return sweep of the scan head between exposures, a light-tight blanking shutter shall be provided to eliminate light from striking film in the film track.

(f) Reaction and Momentum Balance

Residual reactions imparted to the satellite vehicle during camera operation shall be minimal. Installation of the cameras shall provide counter-rotation of moving components during simultaneous operation of both cameras for stereoscopic photography. During single camera operation, reaction torques shall be minimized by the incorporation of a momentum balance so the residual torques imparted to the satellite vehicle do not exceed the values specified in Section 3.3.1.3.3B.

(g) Homing

To eliminate light leakage through the optical train during extended periods when photographic operations are not required, provisions shall be made to orient the lens cell in a "homed" position such that lens elements are not exposed to the ambient light sources.

(h) Data Block and Other Reference Markings

A binary data head shall produce well defined spots comprising a 29 bit time word on each panoramic camera frame. Provisions shall also be incorporated to record constant frequency marks, natural fiducials, end-of-pass mark, S/I camera operation and the panoramic camera serial number on the pan frames. Data shall be recorded at the edge of the frame, and block design shall be such that the spots do not bloom into the main frame format nor off the edge of the film.

B. Horizon Cameras

Two 55 millimeter focal length, f/6.8 horizon cameras shall be incorporated as an integral part of each panoramic camera. The horizon cameras shall be capable of recording the earth horizons to the port and starboard sides of the satellite vehicle from orbital altitudes. Specific requirements shall be as follows:

(a) Paired horizon cameras on a particular panoramic camera shall operate simultaneously on alternate panoramic cycles. The horizon photographs shall be exposed adjacent to the panoramic format.

(b) Artificially illuminated fiducials shall be generated adjacent to each horizon format and shall be used to correlate optical and mechanical format centers.

(c) Horizon camera shutters shall be adjustable to exposure times of 1/100 second, and the shutters shall be designed to fail safe in the shut position to prevent fogging of the panoramic film in the event of a shutter malfunction.

(d) Horizon camera lenses shall be focused for vacuum conditions at a temperature of 70 degrees Fahrenheit. The lens shall have the following characteristics:

Focal length	=	55 millimeters ±2 percent
Field Angle	=	not less than 44 degrees
Aperture	=	f/6.8 with iris diaphram control from f/6.8 to f/22

C. Digital Recording Clock

The digital clock shall be capable of storing time unambiguously for a period of five days. Upon receipt of an interrogate command, the clock shall provide the signals required for auxiliary recording of the binary time word on the panoramic camera film. A 29 bit serial word shall be provided for telemetry verification, and two 29 bit parallel words (one to each panoramic camera) shall be provided for recording on film. The digital clock error shall not exceed 1.0 milliseconds in any 12 hour period after accounting for clock drift and offset. The least significant bit shall be 1 millisecond.

D. Stellar/Index Cameras

Stellar and Index photography shall be generated intermittantly and concurrent with panoramic photography. The S/I cameras shall be capable of simultaneously photographing the nominal area covered by six to ten panoramic exposures and a sufficiently large number of stars to permit photogrammetric analysis of the geocentric position of the satellite vehicle and its altitude in orbit. Capability of S/I photography when utilized with digital clock data shall enable the vehicle location to be determined within an accuracy of one quarter minute of arc at the local horizon and $\pm$ one millisecond of time. When located by photogrammetric methods, the location of image points of ground detail for index photography shall be determinable to within one quarter mile radius of the true geodetic position.

Two Stellar/Index cameras and film loads shall be provided, one for the "A" mission, and one for the "B" mission. Exposed film shall be transported to the takeup cassette in the respective satellite recovery vehicle. S/I camera circuits shall be fused to protect the panoramic camera power supply. Corona marking shall be limited to 10 percent of the programmed formats, and any such marking shall be at a density less than 0.4 above the base plus fog level. S/I photography shall be programmed to obtain 55 to 60 percent overlap on index photographs. One stellar photograph shall be generated for each index format, with a nominal capacity for 400 frames of photography.

Specific design requirements are as follows:

(a) The Index Unit shall accommodate 130 feet of 70 millimeter, 3.2 mil polyester base film. The lateral field of view shall be 70 degrees, and resolution shall be 90 lines per millimeter with a target contrast of 1000 to one. The index unit employs a 38 mm, f/4.5 lens, and has shutter speed adjustable from 1/125 to 1/500 second.

(b) The Stellar Unit shall accommodate 65 feet of 35 millimeter 3.2 mil polyester base film. The conical field of view shall be 16 degrees. The stellar unit employs an 85 mm, f/1.8 lens, and has a shutter speed of 2.0 seconds.

(c) Both the Stellar and Index units shall contain redundant film cutters in the film path. Upon command, the S/I camera shall be capable of transporting both the 70 mm and 35 mm to the takeup spools by slew. The slew rate requirement is a minimum of 2.5 feet of 35 mm film in 75 seconds in the slew mode.

(d) Reseau imagery shall appear in the stellar and index frames, and artificially illuminated fiducial marks shall be provided for the stellar frames.

(e) The time relationship between the opening of the index unit shutter and the stellar unit shutter shall be essentially constant, and shall be capable of being calibrated to within 10 milliseconds.

(f) The angular reliationship between the stellar and index optical axes shall be within 90 degrees $\pm$ 2 degrees, and capable of being measured within a tolerance of $\pm$ 5 seconds of arc. Calibration of the angular relationship shall be conducted and documented in the camera log book. The stellar unit shall be calibrated for radial distortion, and the index unit shall be calibrated for radial and tangential distortion.

E. Film Cassettes

Film cassettes shall comprise a supply for the panoramic cameras, and takeups located in each of the satellite re-entry vehicles.

The supply cassette shall consist of two film spools, mounting enclosure, and controls to assure film supply at proper tension. Each supply spool provides film for one of the panoramic cameras, and shall have a nominal film capacity of 15,600 feet of 3.2 mil thick 70 millimeter film. The supply spools shall be capable of operating together or independently as required for stereo or mono modes of photography.

The takeup cassettes shall be mounted within the satellite re-entry vehicles. The "B" mission cassette, installed in the aft SRV, shall be held to a fixed operational position until programmed to initiate movement to the takeup film. Additionally, this cassette shall provide rollers to permit panoramic film to enter and leave the film spool hubs during "A" mission operations while takeup is being performed by the cassette in the forward SRV.

Each takeup cassette shall consist of two panoramic camera film spools, one index unit film spool and one stellar unit film spool, one structural mount, three powered film takeup drives, a film measuring gage for panoramic film, and spool control features that assure film takeup. Each panoramic takeup spool shall accomodate 7,800 feet of 3.2 mil 70 mm film. The index spool shall accomodate 130 feet of 3.2 mil 70 mm film, and the stellar spool shall accomodate 65 feet of 3.2 mil 35 mm film. The design of the cassette shall conform to the basic configuration and space limitations of the satellite re-entry vehicle. The panoramic camera takeup spools shall be capable of operating together or independently. The takeup spools shall incorporate the following design features:

a) Each takeup spool shall be capable of being started and stopped at least 300 times in flight.

b) Each panoramic camera takeup spool shall incorporate an electro-mechanical anti-backup mechanism in the spool drive to prevent the spool from unwinding.

3.3.3 Satellite Re-entry Vehicle

3.3.3.1 Budgeted Performance and Design Requirement

3.3.3.1.1 SRV Weight Budget

Total weight of the Satellite Re-entry vehicle as defined here in shall not be more than 298 lbs, This weight is exclusive of the takeup cassette installed in the SRV. Budgeted weight of SRV components shall be apportioned to the following items:

A. SRV Thrust Cone Structure
B. Re-entry Heat Shield
C. De-boost Rocket Motor
D. Spin-Despin Subsystem
E. Recovery Capsule Structure
F. Parachutes
G. Telemetry Unit
H. Tracking Beacon
I. Ballast
J. Electrical-Electronic Programming & Harnesses
K. Retrieval Aids
L. Pyrotechnic Actuators

3.3.3.1.2 SRV Reliability Budget Requirements

The reliability design objective for the satellite re-entry vehicle is 0.98 for operation during the period of time from satellite vehicle separation through air retrieval, or through recovery from the water if air retrieval is not made. The reliability goal shall include the probability of SRV equipment surviving the ascent boost environment, and subsequent inactive periods on-orbit until functional operation is required.

Equipment design and component selection shall take cognizance of budgeted reliability goals. Additionally, the SRV Contractor shall consider reliability improvements provided through redundancy. The SRV Contractor shall record and maintain failure-rate data on all critical SRV components for the purpose of reporting current reliability performance estimates for the end-item hardware.

3.3.3.2 SRV General Design Requirements

3.3.3.2.1 General Description

The basic recovery subsystem consists of two Mark 5A Recovery Vehicles mounted in tandem on the payload structure as shown on Figure 10. The primary function of the SRV shall be the return of payload material from orbit. This shall be accomplished by separation of the SRV from the satellite vehicle, deboost from orbit, re-entry and subsequent parachute deployment and ablative heat shield separation. Recovery shall be effected by locating the descending capsule by means of recovery aids, and accomplishing aerial pickup by specially equipped aircraft. As a backup in the event air recovery is not successful, the capsule shall float and be acquired by a surface force.

The basic SRV shall be provided as government furnished equipment (GFE) to the payload integration contractor for installation of the takeup cassette and associated modifications. Attachment and separation provisions including command signals and instrumentation shall be provided by the Payload Contractor. Figure 11 illustrates the general arrangement and primary components of the basic SRV. SRV requirements and modifications shall be specified in detail in the Payload/SRV Interface Specification, and shall minimize the mechanical and electrical changes to the basic SRV.

3.3.3.2.2 Aerospace Ground Equipment (AGE)

The Payload Contractor shall provide AGE to handle the SRV in the modification and assembly cycle. The Payload Contractor shall also provide AGE for re-entry vehicle checkout after modification. Existing AGE will be used to the maximum extent practicable.

3.3.3.2.3 Facilities

No special facilities are expected to be required for the SRV. Any requirements of this type, uniquely related to Program [redacted] shall be identified and substantiated.

3.3.3.3 Specific Design Requirements

3.3.3.3.1 Communications and Control Requirements

All commands to initiate SRV operation shall be provided from the satellite vehicle. After separation from the satellite vehicle, the sequence of spin, firing the deboost rocket, despin, and ejection of the thrust cone, shall

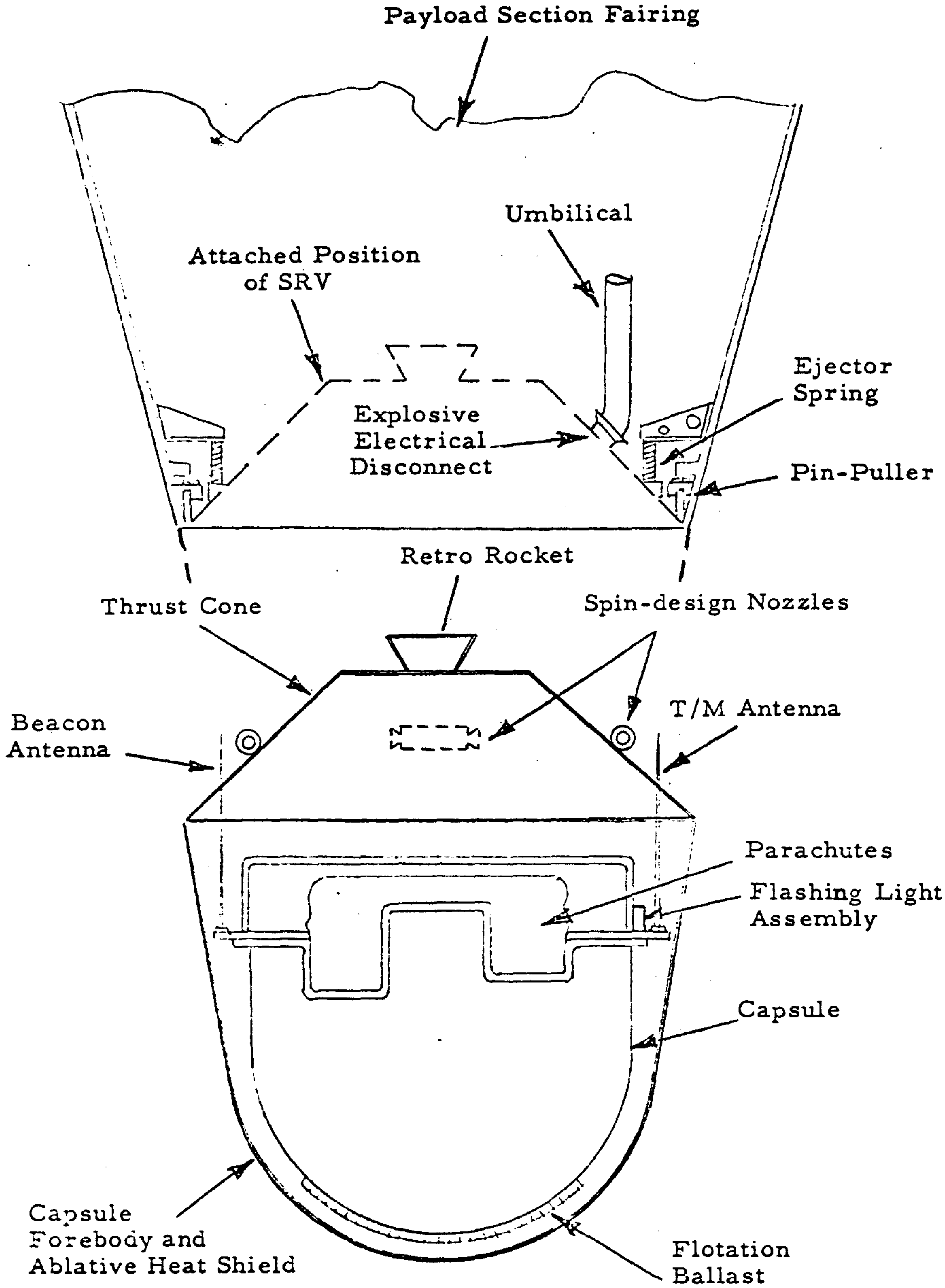


Figure 11 Satellite Re-Entry Vehicle, Inboard Profile

be controlled by a timer installed in the SRV. After the re-entry phase, deployment of the parachutes shall be initiated by circuitry contained with in the SRV. The SRV shall provide a 3 channel FM/FM telemetry transmitter for supplying key event and environmental data. A fixed-tuned pulse transmitter shall be provided as a beacon to assist in acquiring the capsule during recovery.

3.3.3.3.2 Equipment Environmental Requirements

The Payload Contractor shall provide sealing provisions for the capsule and other devices as necessary to protect the exposed film during the re-entry and retrieval operations. During re-entry, the film temperature shall not exceed 125 degrees Fahrenheit. The cassette assembly with film shall be capable of withstanding re-entry accelerations up to 35 g's longitudinally and 5 g's laterally.

3.3.3.3.3 Attitude Control Requirements

During the separation sequence, the satellite vehicle shall provide the initial orientation of 58 degrees from the local horizontal. Subsequently, the SRV attitude shall be maintained during retrorocket firing by spin stabilization. A minimum spin rate of 48 rpm shall be imparted to the SRV at a nominal time of 2.3 seconds after the separation command. After a time interval of 18.3 seconds to accomodate retro rocket firing, the SRV shall be despun to 10 rpm ± 2 rpm. During the balance of the re-entry phase, SRV attitude shall be maintained within acceptable structural limits by the aerodynamic damping characteristics inherent in the SRV design.

The modified SRV, with payload equipment installed, shall meet the following weight and balance requirements:

Condition	Weight Lbs.	Longitudinal CG	CG Offset from Roll Axis (in)
SRV at Separation with Thrust Cone	340 to 390		0.20
SRV at Re-entry Without Thrust Cone		Fwd. of STA 14 (Capsule Station)	0.08

Product of Inertia (lb. in^2)

Roll/Yaw	Roll/Pitch	Yaw/Pitch
300	300	

3.3.3.3.4 Power Supply Requirements

During on-orbit operation, power to operate the film takeup cassette shall be provided by the Satellite Vehicle. The Satellite Vehicle shall also provide power to accomplish activation of the SRV and separation from its mounting structure during the deboost sequence. Subsequent operation of SRV timers, telemetry, beacon, and pyrotechnic devices shall be powered by batteries contained in the SRV. Provisions shall be made to replace or trickle charge the GFE provided SRV batteries during countdown prior to launch. As an alternative, the Payload Contractor may substitute a remote activated battery. Implementation of a substitute battery installation shall have prior approval of the [redacted] Program Directorate. Electrical harnessing shall be provided by the Payload Contractor to connect payload equipment located in the SRV.

3.3.3.3.5 Structural Design Requirements

Components in the capsule and on the thrust cone shall be relocated and mounting trays shall be redesigned as necessary to provide space and access for payload components. Components redesign or repackaging is not expected.

The capsule modifications shall not alter its capability to float for at least 55 hours in the event that air-retrieval is not accomplished. The sink-port supplied with the capsule shall scuttle the capsule after 55 hours. Flotation shall not exceed 95 hours maximum. The capsule shall be capable of sustaining water impact under conditions of a sea state of 3 with 18 knot surface winds. After water impact, the capsule shall float in an upright position and shall not capsize in sea states of 3 or less, as defined by the U. S. Navy Hydrographic Office.

The parachute recovery subsystem shall be capable of effecting the necessary deceleration and stabilization of the recovery capsule during descent through the atmosphere. The suspended weight shall not exceed 180 lbs. excluding the parachutes. The desired rate of descent at 10,000 ft. MSL is 25 feet per second under standard atmospheric conditions. The main parachute canopy shall be designed for aerial recovery, with 90 to 180 lbs. suspended weight. Maximum aerial recovery altitude shall be 15,000 ft., and maximum aircraft speed shall be 135 knots indicated air speed.

3.3.3.3.6 Propulsion Requirements

Re-entry vehicle weight may vary between 340 lbs. and 390 lbs. depending upon the weight of payload material carried by the SRV. The retro-rocket shall provide a total impulse of 10,500 lb. seconds ±3 percent to impart an incremental deboost velocity to the SRV between 865 and 1000 ft/sec. The retro-rocket motor and all SRV pyrotechnic devices shall be compatible with the following requirements as related to Program [redacted]

A. Pyrotechnic Handling and Storage
Interstate Commerce Commission Regulations
Launch Site and Range Safety Regulations
Contractor Plant Safety Regulations

B. Service Life
Installation to activation time

3.3.3.3.7 Instrumentation

The satellite re-entry vehicle and payload components installed in the SRV shall incorporate adequate instrumentation to provide status and diagnostic data during orbital, separation, and re-entry phases of flight. During the orbital and separation phases, SRV-mounted payload components shall be monitored by means of the satellite vehicle telemetry link. During the deboost, re-entry and recovery phases, status data and key events shall be transmitted via SRV telemetry. Interface data requirements on-orbit and during separation shall be defined in the Payload/SRV Interface Specification.

3.3.3.3.8 Retrieval Aids

A flashing light, the VHF Beacon and the telemetry transmitter on the SRV will be used for tracking during the recovery phase. The flashing light shall have an output of 15 lumin-seconds per flash with a minimum flash rate of 60 per minute. Minimum operating time after water impact shall be 10 hours. The minimum life of the beacon shall also be 10 hours after water impact. Operating life of the SRV telemeter subsystem shall be a minimum of 40 minutes after separation of the SRV from the satellite vehicle.

Subsequent to recovery, simple and convenient means to turn-off retrieval aids and to disarm any unused pyrotechnics shall be provided. These provisions shall be designed to eliminate personnel hazards to the recovery crews.

3.3.4 Launch Vehicle Functional Subsystem

The launch vehicle functional subsystem consists of a first stage booster vehicle, the satellite vehicle functioning in an ascent mode as the second stage booster, and the radio command guidance equipment for tracking and steering the booster vehicles into the desired orbit. The Stage I booster and ascent guidance are discussed individually under Sections 3.3.4 and 3.3.5 respectively. Operation of the satellite vehicle in the ascent mode was discussed in Section 3.3.1.

3.3.4.1 Budgeted Performance and Design Requirements

3.3.4.1.1 Launch Vehicle Weight Budget - The nominal weights for the LV2A Thrust Augmented Thor booster shall be as follows:

Item	Weight (lbs)	Total Weight (lbs)
Weight Empty		6,873
Propellants	100,038	
Lube oil	127	
Pressuriz. gas	552	
Solid Motor Boosters (3)	27,828	
Stage I Weight at Liftoff		135,438
Less Expendables	- 66,162	
Less Solid Motor Cases (3)	- 5,652	
Wt.at Solid Motor Separation		63,624
Less Expendables	- 55,365	
Wt.at Main Engine Cutoff		8,259
Less Expendables	- 195	
Wt.at Vernier Engine Cutoff		8,064

3.3.4.1.2 Launch Vehicle Reliability Budget Requirements

The reliability design objective for the launch vehicle is outside the scope of this specification. However, for the purpose of reliability apportionment, a figure of 0.96 is to be utilized. The LV2A consists of the following functional subsystems and components:

A. Structure
B. Propulsion Group
C. Guidance and Control Group
D. Destruct and Range Tracking Equipment
E. Separation Provisions
F. Electrical Subsystem
G. Instruments and Telemetry

All changes to basic government furnished equipment shall consider the effect on demonstrated flight reliability as related to reliability goals for Program [redacted] Redesign of existing equipment shall consider reliability improvements through component selection and redundancy. The contractor shall collect and provide failure-rate data for the purpose of establishing reliability performance estimates for end-item flight hardware and aerospace ground equipment.

3.3.4.2 Launch Vehicle General Design Requirements

3.3.4.2.1 Launch Vehicle Description

The launch vehicle shall perform the following functions during the ascent phase of the [redacted] mission.

A. Provide thrust required to boost the satellite vehicle and payload from the launch pad to a sub-orbital velocity compatible with the mission profile and booster vehicle performance capabilities.

B. Perform preprogrammed maneuvers to orient the launch vehicle configuration to the desired flight azimuth, maintain heading within range safety boundaries, and execute yaw maneuvers when required to achieve the azimuth necessary for particular orbits.

C. Maintain attitude control and respond to guidance steering commands so that the suborbital burnout condition is achieved within specified tolerances. Guidance commands shall be transmitted to the Stage I Booster from a receiver located in the satellite vehicle.

D. Provide tracking signals during ascent for range safety impact calculations and be capable of receiving flight termination commands and destructing the booster when commanded. Destruct signals shall be forwarded to the satellite vehicle from the receivers located on the Stage I booster.

E. Separate from the satellite vehicle at the required suborbital flight condition by means of a retrovelocity maneuver without inducing rotational torques in the satellite vehicle.

F. Provide telemetry data concerning booster vehicle equipment status, environments, and occurrence of key events.

The LV2A Thrust Augmented Thor is a vertically launched, liquid-fueled space booster powered by a main gimballed rocket engine and three thrust augmentation solid propellant rocket motors. Pitch and yaw control is provided by gimballing the main engine in the pitch and yaw planes during powered flight. Two gimballed vernier rocket engines provide roll control, and augment the main engine in providing pitch and yaw attitude control prior to main engine cutoff (MECO). Liquid propellants consist of RJ-1 fuel and liquid oxygen.

The booster configuration is illustrated by Figure 1, and consists of five structural sections. From forward to aft, the sections are designated; transition section, fuel tank, center body section, oxidizer tank, and engine/accessories section. The solid propellant motors attach externally to the sides of the booster structure, and are jettisoned early in the flight after their burnout has occurred some 40 seconds after liftoff. At the time of booster separation from the satellite vehicle, a retro-velocity is imparted to the booster by solid rockets attached to the Stage I/Satellite Vehicle adapter. The adapter remains attached to the Stage I booster throughout separation, and carries with it the satellite vehicle's range safety destruct charge.

Booster attitude and stability are controlled by an autopilot flight controller which is activated at liftoff. Programmed maneuvers are implemented by a punched tape programmer/timer to actuate various portions of the control circuits. Subsequent to completing programmed orientation maneuvers, the guidance relay is locked-in and the booster responds to guidance command steering adjustments provided to the flight controller from the receiver located in the satellite vehicle. Booster main engine cutoff and satellite vehicle separation are commanded by radio guidance. All ascent guidance functions are backed-up by a nominal flight program of stored commands in the event of radio guidance failure, with the exception that MECO will occur through propellant depletion.

3.3.4.2.2 Aerospace Ground Equipment (AGE)

The Launch Vehicle Contractor shall provide the AGE required to check-out vehicle equipment, vehicle subsystems, and the complete booster for proper operation within allowed tolerances and for flight readiness. Existing AGE will be used to the maximum extend practicable.

A booster systems test shall be conducted at the contractor's plant prior to acceptance of the booster by the procuring agency. The purpose of this test shall be to verify that all booster subsystems operate individually and concurrently within specification limits, and that the booster being offered for acceptance is flight ready. During the combined subsystems testing, functional test simulators may be used to represent hardware provided by another contractor across a mechanical or electrical interface. However, all simulators shall exhibit proper characteristics of loading and dynamic response as defined by the respective contractors and per interface specifications.

At the launch site, the booster shall be inspected to ensure that no damage has been sustained as a result of shipment. Prior to erecting the booster on the launch stand, functional tests shall be performed for those items of equipment requiring confidence testing at limited time intervals, to maximize the time that the booster may be held on stand prior to recycling. After installation on the launch stand, the booster shall be checked-out for compatibility with launch AGE, and subsequently with the mated satellite vehicle and radio guidance subsystem. Solid rocket boost augmentation motors shall be attached after the satellite vehicle has been mated to the Stage I booster. Status monitoring shall be provided to support extended holds in a launch-ready status up to 20 days, so that the vehicle can be launched within 24 hours when so directed.

Mechanical and electrical AGE shall be provided to implement the above stated concept. Ground handling trailers, slings, and fixtures shall be compatible with booster and augmentation motor hardware to ensure that damage is not incurred as a result of handling. Launch vehicle AGE shall be identified and functionally described by the Launch Vehicle Contractor. AGE shall provide for but not be limited to the following functions:

A. Transportation and Handling
B. Servicing
C. Functional Test Simulations
D. Weight and Balance
E. Checkout Testing
F. Loading of Expendables
G. Ground Electrical Power
H. Ground Environmental Control
I. Launch Control

Maximum utilization shall be made of existing AGE equipment.

3.3.4.2.3 Facilities

The Launch Vehicle Contractor shall identify all facilities requirements necessary to support the factory to launch sequence for his end-item hardware. Facility requirements shall be in accordance with the booster test plan, and shall make use of existing facilities. All facility requirements that are uniquely related to the [redacted] Program shall be identified and substantiated. Facility requirements shall consider, but not be limited to the following:

A. Assembly Building
B. Test Facilities

3.3.4.3 Specific Design Requirements

3.3.4.3.1 Guidance and Control Requirements

From lift-off until initiation of radio command guidance, the booster shall be controlled by the autopilot flight controller. The flight controller shall maintain booster stability and shall direct the booster to the guidance initiation point as programmed for the flight. Radio guidance steering shall be enabled by the flight controller and shall be terminated for booster steering by the ground guidance equipment just prior to MECO.

With radio guidance, an accuracy of one percent of the computed radio guidance steering commands shall be achieved. If radio guidance is lost, the booster flight controller shall guide the booster to the burn-out condition within the operational tolerances of the equipment. These are: ± 4 degrees in flight path angle, 5 nautical miles in position, and 500 ft per second in velocity.

Programmed maneuvers and events for the Stage I booster are represented as follows for nominal flight conditions:

Event	Time from Lift-off (sec)
Lift-off	0
Start roll to Launch Azimuth	2
Start pitch and yaw rates	4
Pitch rate step	10
Stop roll and yaw rates	(at programmed time)
Solid Motor Burnout	40
Pitch rate step	55
Drop Solid Motor Cases (variable with inclin.)	60
Pitch rate step	80

Event (Cont'd.....)	Time from Lift-off (sec)
Start "Dogleg" yaw rate (if req'd)	80
Stop yaw rate, start roll rate	90
Stop roll rate	94
Enable Radio Guidance Commands	94
Enable Main Engine Cutoff (MECO)	138
End Radio Guidance Steering	143
Main Engine Cutoff (Radio Command)	148
Vernier Engine Cutoff, Stop Pitch Rate	157
Separate from Satellite Vehicle (Radio Comm)	161

Guidance commands shall be transmitted between the Stage I booster and satellite vehicle through a single interface connector carrying command signals only. Detail requirements shall be specified in the LV2A/S01B Electrical Interface Specification.

3.3.4.3.2 Propulsion Requirements

The propulsion subsystem of the LV2A consists of a main liquid-propellant engine, two liquid-propellant vernier engines, and three solid-propellant thrust augmentation rocket motors. Nominal performance characteristics for the propulsion subsystem shall be as shown in Table 10.

Table 10 Propulsion Performance Characteristics

Item	Nominal Value
Sea level thrust	332,400 lbs
Total impulse (flight)	32.0×10^6 lb sec.
Main Engine Burn time	146 sec
Vernier Engine Burn time	155 sec
Solid Motor Burn time	40.3 sec

Characteristics of the liquid-propellant rocket subsystem shall be as follows:

Fuel	RJ-1 conforming to Spec MIL-F-25558
Oxidizer	Liquid oxygen per Spec MIL-P-25508
Thrust (SL)	172,000 $\pm$ 2,000 lbs
Mixture ratio	2.15 $\pm$ 2 percent
Specific Impulse (SL min)	248 sec
Propellant Utilization(min)	99 percent
Fuel Tank Volume (total)	643.58 cubic ft
Liquid Oxygen Tank Volume	998.79 cubic ft

The booster shall contain valves, automatic sensors, relays, associated lines, and electrical circuitry to permit loading from the aerospace ground equipment. Gaseous nitrogen storage spheres shall be provided for pressurizing fuel and oxidizer tanks.

Characteristics of each of the thrust-augmentation solid motors shall be as follows:

Specific Impulse	223 seconds
Thrust (nominal during web-burn)	53,425 lbs
Total Impulse	1.637×10^6 lb seconds
Operational temperature range	10 to 110°F

Prior to attaching solid motors to the booster, each motor shall be fitted with a destruct charge.

3.3.4.3.3 Electrical Power Requirements

The booster electrical power subsystem shall provide a source of ac and dc power required by the various booster vehicle components and equipment. The booster shall not be required to supply electrical power across the interface to the satellite vehicle. Booster vehicle battery life shall be compatible with the [redacted] System pre-launch readiness hold requirements.

3.3.4.3.4 Flight Termination Subsystem

The range safety equipment installed in the booster vehicle shall conform to existing Western Test Range requirements. The installation shall consist of two command destruct receivers and two separate antennas. Each command destruct receiver shall be supplied with power independently of the other. The receiver outputs for destruct commands will be fed into the safety and arming mechanisms. A destruct command signal shall be provided to the Satellite Vehicle in accordance with the LV2A/S01B Electrical Interface Specificatior

3.3.4.3.5 Telemetry and Tracking

The booster vehicle shall provide a PDM/FM/FM telemetry subsystem. Telemetered data shall provide for post-flight analysis of booster performance, environments, and sequenced events. Diagnostic data, suitable for analysis of booster malfunctions, shall also be provided.

The booster shall not require a separate beacon for tracking purposes.

3.3.5 Ascent Radio Guidance Subsystem

3.3.5.1 Description

The Western Electric Company/BTL Guidance System shall be used to provide real-time sequenced events and real-time steering corrections to the LV2A and S01B during the powered flight phase. The guidance equations utilize velocity steering to guide both the first and second stages of powered flight. Controlled parameters for the first stage are apogee velocity, apogee radius and inclination. The Stage I booster engine is shut down by radio guidance command, as is separation of the Stage I booster from the Agena S01B, and enabling of the Agena velocity meter to control shutdown of the Agena engine.

Controlled parameters during Agena guidance are orbital period, orbital inclination and flight path angle at injection. The steering commands shall c attitude changes of the vehicle during powered flight to implement thrust vector corrections. These corrections shall result from the computer calculation of anticipated cutoff conditions repeatedly predicted from the radar tracking data and continuously compared with the desired velocity state of the vehicle at thrust cutoff. The radio guidance subsystem shall not force the vehicle to fly a nominal flight path, but shall command steering corrections to assure a specified velocity vector at cutoff. Guidance accuracy requirements are specified in Section 3. 3. 1. 3. 3

Normal preflight preparations require from 17 to 22 days to generate necessary performance data, guidance coefficients, guidance computer tape, and check-out the ground guidance equipment. It shall be a program requirement to reduce the time span for this work to 8 days or less to be compatible with [redacted] System reaction time. Modification of the equations and reprogramming for the guidance computer shall be accomplished to meet the System requirements. Implementation of this effort shall receive prior approval of the [redacted] Program Directorate.

3. 3. 5. 2 Facilities

The ground based components of the radio guidance subsystem shall include a radar tracking station, which tracks the vehicle and transmits RF commands; and a computer which processes the tracking data, computes trajectory corrections, and issues steering commands and timely discrete commands. The radar station operates at X-Band frequency, and pulse-position modulates the command spacing between continuous pairs of address pulses. The guidance computer is a Remington Rand "Athena" utilizing drum storage for the guidance equation program.

Currently available Radio Guidance facilities at the Western Test Range shall be utilized for the [redacted] System. Reprogramming of the guidance equations to attain quick reaction capability shall not dictate a need for new equipment or additional facilities.

3.3.6 Interface Requirements

3.3.6.1 System Interfaces

All functional interfaces occurring between the basic subsystems of the [redacted] System shall be identified as to characteristics, requirements, and implementation of action necessary to ensure compatibility with System Design. Each physical interface created by the junction of hardware supplied by two or more Associate Contractors, or by separate procurements for a single contractor, shall be documented by an adequate interface specification. The specification shall receive joint approval of the affected Contractors and the [redacted] Program Directorate. Interface Specifications shall provide for all pertinent requirements within the following categories:

A. Mechanical
B. Electrical
C. Environmental
D. Instrumentation
E. Assembly
F. Test
G. Schedule

Non-physical interfaces such as operational command links, and physical interfaces between a single Contractor's vehicle and its support equipment, shall be defined and implemented through an appropriate requirements document and/or test plan, as applicable.

Interfaces internal to the [redacted] System, and accomodation provisions shall be as specified in the following subsections.

3.3.6.1.1 Payload/Re-entry Vehicle

Interface specifications shall be provided for integration of the camera equipment into the payload section of the Satellite Vehicle. This specification shall include but not be limited to; mechanical structure and alignments, light and thermal environments, electrical power, commands, instrumentation, and testing.

An interface specification shall also be provided for integration of Satellite Recovery Vehicles into the photographic payload section. This specification shall include but not be limited to; mechanical structure and alignments, thermal environments, sealing provisions, electrical power, commands, instrumentation, and testing.

The Payload/Camera/SRV Interface Specification is contained in LMSC T3-3-004.

3.3.6.1.2 Satellite Vehicle/Payload Section

The interface specification between the Satellite Vehicle and its payload section shall include but not be limited to; mechanical structure, electrical power, commands, instrumentation, and testing. The Satellite Vehicle/Payload mechanical interface is defined by LMSC Drawing 1324216.

3.3.6.1.3 Satellite Vehicle/Satellite Control Facility

The interface between the Satellite Vehicle and The Satellite Control Facility is non-physical and comprises the general functions and requirements as specified in this document. Detailed requirements for tracking, telemetry and commanding shall be specified in the Orbital Requirements Document - Program [redacted] with implementation defined by the Orbital Support Plan and the Test Operations Order for Program [redacted]

3.3.6.1.4 Satellite Vehicle/Facilities - AGE

The interface between the Satellite Vehicle (excluding the Payload section) and AGE shall be accommodated by the Satellite Vehicle Test Plan, the Aerospace Ground Equipment System Design Analysis, Facilities Master Plan and the Program Requirements Document. LMSC Drawing 1399138 defines the mechanical and electrical interface between the Agena and AGE Launch Equipment.

3.3.6.1.5 Satellite Vehicle/Launch Vehicle

An Interface Specification shall be provided for the Satellite Vehicle and the Stage I booster vehicle. This specification shall include but not be limited to, mechanical structure, flight termination commands and guidance commands. The mechanical interface between LV2A and S01B vehicles is defined by LMSC Drawing 1345366B. The electrical interface is defined on LMSC Drawing 1398571.

3.3.6.1.6 Satellite Vehicle/Ascent Guidance

The interface between the Satellite Vehicle and ascent guidance shall be accommodated by Specifications for the airborne guidance equipment, the preflight performance data and trajectory reports, and the Program Requirements Document.

3.3.6.1.7 Launch Vehicle/Ascent Guidance

The Stage I booster vehicle shall not interface directly with the ground guidance subsystem, but receives guidance commands via the Satellite Vehicle/Launch Vehicle Interface.

3.3.6.1.8 Launch Vehicle/Facilities - AGE

The interface between the Stage I booster vehicle and its AGE shall be as described in the Booster Vehicle Test Plan, and the Program Requirements Document.

3.3.6.1.9 Payload/Facilities - AGE

The interface between the integrated Satellite Vehicle payload and its AGE shall be defined by the Payload Test Matrix Report.

3.3.6.1.10 Re-entry Vehicle/Recovery Forces

Interface requirements between the re-entry vehicles and the recovery forces shall be as defined in this specification, the Program [redacted] Test Operations Order, and the Recovery Group Operations Plan.

3.3.7 Requirements for Mission Tracking, Communications and Control Functional Subsystem (Ground Environment)

On-orbit control will be provided by the U.S. Air Force Satellite Control Facility operated by the 6594th ATW under the direction of the [redacted] Program Directorate. Control shall be performed so as to optimize photographic coverage of the pre-selected target areas. Operations for the [redacted] System shall be under the jurisdiction of the Flight Test Field Director (FTFD), 6594th ATW, with Test Advisory Support for mission provided in accordance with directives of the [redacted] Program Directorate. Hardware, software and procedures will be the same as those currently in use in so far as feasible.

The tracking stations identified in Section 3.1.1.1 will be utilized in conjunction with the Satellite Test Center (STC) to provide telemetry and tracking during orbit injection, tracking and telemetry readout at each station pass, and commanding as directed from the STC. Commands shall be issued for vehicle and payload operations as a result of ephemeris determination, mission optimization, and command selection at the STC. Support will also be supplied by SCF for operational tests, integrated systems test and associated program activities which will allow performance of the program objectives in an efficient and reliable manner.

3.3.7.1 Satellite Control Center

The Satellite Test Center shall perform the following functions:

A. Data Processing

The data processing equipment interfaces with the telemetry, tracking and command equipments to provide information flow between the STC and the tracking stations. Direct contact is maintained between the STC and SCF stations. The main functions are as follows:

1. Accept predicted acquisition and ephemeris data from the STC to provide positional data for tracking station antenna systems.

2. Accept tracking data from the radar tracking equipment and provide tracking data to the STC for use in ephemeris determination.

3. Accept command data from the STC and provide the radar equipment with information for transmission to the vehicle.

4. Accept telemetry data and provide it to the STC and/or other equipments for use in vehicle equipment status determination and control.

B. Ephemeris Determination

Ephemerides will be determined at the STC using tracking data gathered by the stations to provide information for acquisition and command selection for the Satellite Vehicle and payload. The orbital period, perigee altitude, argument of perigee, orbit eccentricity and probable errors in these parameters are vital to proper adjustment of programmed commands during the mission. The ephemeris will be utilized for computation of acquisition data, optimization of the mission, selection of real time commands for payload functions, impact prediction, post flight data correlation, flight evaluation and other such activities coincident with adequate program support. Ephemeris prediction capability must be very accurate and should have the capability of accounting for such factors as geopotential harmonics through the fourth order and a seventh parameter fit for average drag determination. Satellite spatial position errors on orbit for photographic command selection must be known to:

a. ± 4.0 N.M. in track
b. ± 1.0 N.M. cross track
c. ± 0.5 N.M. in altitude

C. Mission Optimization

Mission optimization programs will receive ephemeris, telemetry, payload capability, stored commands and alternative camera programs, and operational requirements including target search areas as inputs, and shall compute the optimum selection of commands necessary to maximize mission performance.

D Command Selection

With the stored commands available through the Satellite Vehicle orbital programmer, command selection shall be limited to the implementation of the optimum real-time commands available for adjusting payload and vehicle events. All commands selected for transmission to the vehicle will be verified at the STC prior to transmission by the tracking station to the Vehicle.

3.3.7.2 Ground Stations

The tracking stations to be utilized in conjunction with the STC are identified in Section 3.1.1.1. Functional interfaces with the STC are discussed in Section 3.3.7.1. Station characteristics for tracking, telemetry and command shall be as follows:

A. Tracking

The Precision Long Range Tracking S-Band radar (PRELORT) will be utilized for vehicle tracking. The radar operates in the 2.7 to 2.9 kmc frequency band and transmits an interrogation code to the Satellite Vehicle and tracks the response. The radar has a 14 foot diameter parabolic reflector antenna. Peak power output is 325 kw with pulse widths of 1.0 ± 0.2 microseconds at pulse repetition frequency rates of 410, 512, 584, and 630 as required to eliminate range ambiguity. A three pulse code is used for interrogation with the framing pulses providing the interrogation codes by six different spacings. Pulse position modulation of the third (center) pulse is used for transmission of command information. The TLM-18 and T&D VHF telemetry receiving antennas shall provide "angles-only" back-up tracking data to the PRELORT.

Tracking support during the launch and ascent phase shall be provided from [redacted] by tracking S-Band transponder signals with the PRELORT radar, seconded by TLM-18 tracking on the VHF telemetry signals. The WTR ship, if used, will sustain ascent support by acquiring vehicle signals with the SPQ-8 radars before [redacted] goes passive, and will track through injection until signal fade.

During the orbital phase, [redacted] and [redacted] will provide tracking support by acquiring and tracking on S-Band radar and transponder signals. Assurance of full coverage shall be provided by secondary, angle-tracking data acquired from the VHF telemetry antennas.

For the re-entry phases, tracking shall be provided by [redacted] and [redacted] utilizing the PRELORT radar to obtain tracking messages from the S-Band transponder in the satellite vehicle prior to SRV separation. [redacted] will receive S-Band tracking from the satellite vehicle as it continues in orbit, after SRV separation, and in addition, will receive angle-only data with the TLM-18 antenna from the re-entry capsule VHF telemetry signal. Capsule tracking is not used to determine capsule location if re-entry has followed predicted conditions, however, angle data is significantly valuable if an overshoot should occur.

B. Telemetry

The FM/FM ground station telemetry equipment operating in the 215-260 mc band will be utilized. Receiving antennas will include the Tri-Helix, the Disc-on-Rod, the TLM-18 and Telemetry and Data antenna. Anatenna use will be determined by availability. The following antennas shall be used to support the [redacted] Program mission at the station locations noted:

Station	Antenna
[redacted]	TLM-18 or Disc-on-Rod (Tri-Helix may be used for acquisition)
	Tri-Helix
	Flat-Plane Array or T&D
	Disc-on-Rod
	TLM-18
	Tri-Helix

The raw video output received by the ground station shall be routed to the subcarrier discriminators for immediate processing of status and control information. Transmission from the satellite vehicle provides non-return-to-zero (NRZ) modulation of the FM subcarrier data, with modulation in the positive sense only. Hence, the ground station discriminators will give a positive voltage for an upper-band edge signal. All channels used shall comply with standard IRIG designation, except Channel F, which has 98 kc center frequency with $\pm 15\%$ full scale deviation. Decommutation capability of each of the SCF ground stations used for Program [redacted] is as follows:

Station	Number of Decomms Available		
	30 pts/sec	60 pts/sec	20 pts/sec
[redacted]	(2)	---	---
	---	---	(2)
	(2)	(2)	---
	(2)	---	---
	(2)	(2)	---
	---	---	---

C. Command

Real-time commands are limited to the periods of time when the satellite vehicle is within communications view of the ground station.

The primary command system for Program [redacted] shall be the S-Band radar analog command mode, which comprises 6 tones and 15 commands. Two additional command systems are used to augment the primary command capability. These are the ZEKE (150 mc VHF) and ZORRO (S-Band digital). Depending upon the command, ZEKE commands will require transmission in a secure (encoded) or a clear mode. ZORRO commands will be transmitted in a secure mode. Antennas used by SCF ground stations to transmit commands to the [redacted] satellite vehicle shall be as follows:

	Command System		
Station	Analog	ZORRO	ZEKE
[redacted]	Prelort	Prelort	Helix
	Prelort	Prelort	Helix
	Prelort	Prelort	Crossed-Yagi
	Prelort	Prelort	Helix
	Prelort	Prelort	Crossed-Yagi

4.0 Quality Assurance

The [redacted] System shall incorporate provisions for quality assurance at all levels of design, fabrication and test as a means for attaining the program reliability objectives. The requirements of MIL-R-27542 and MIL-Q-9858, as modified herein and by the authority of individual contract, shall apply.

4.1 Design Review

Design reviews shall be performed for the purpose of evaluating the adequacy of design, design analysis, testing, and documentation. Design reviews shall be performed at the System, Subsystem, and Equipment levels in accordance with the following definitions, and shall apply to design modification as well as new design:

A. Concept Review - Prior to the initiation of detailed design effort, the requirements, preliminary design criteria, and design alternatives shall be reviewed. The purpose of this review shall be to ensure that the design approach is fully compatible with requirements; and that factors of technical risk, cost, schedule and utilization have been adequately assessed.

B. Detailed Review - Prior to engineering drawing release and/or prior to qualification testing, a detailed design review shall be conducted to assess the adequacy of design and testing.

C. Final Design Review - Prior to use in the [redacted] System, a final design review shall be conducted to verify compliance with requirements and compatibility with System characteristics. In the conduct of the final review, development and test history including all failure data shall be evaluated, and the readiness for application in the [redacted] System shall be substantiated.

Conduct of design reviews shall be a responsibility of each individual Contractor's management office for the [redacted] Program, utilizing the resources of affected design, analysis, test, manufacturing and inspection organizations as applicable within his company. The [redacted] Program Directorate shall be given adequate advance notice of all design reviews, and shall receive documented summaries of review proceedings. Requirements for formalized design reviews shall not lessen any Contractor's basic responsibilities for a continuing surveillance and reporting of design progress to the Contracting Agency. Requirements for final review of the System prior to each launch are specified in Section 4. 3. 2.

4. 2 Category I System Test Requirements

Category I test requirements shall encompass engineering, developmental, reliability and qualification testing. Test requirements shall specify the test planning, test conduct, data reduction and analysis, and test quality control.

4. 2. 1 Engineering Test and Evaluation

Engineering Tests are here defined as tests conducted for the purposes of determining feasibility, acquisition of state-of-art data, development phase demonstration, and exploratory failure mode simulation. For the [redacted] System, tests in this category are expected to be minimal, with the possible exception of last item. To achieve flight reliability goals through prevention of recurring in-flight malfunctions and anomalies, the mode of failure and positive corrective action shall be established for each occurrance by the responsible Associate Contractor. Engineering tests to simulate the failure mode shall be performed in cases where sufficient diagnostic telemetry data cannot be provided to substantiate the nature of the malfunction. Such tests shall be conducted on an expedited basis to provide implementation of corrective action commensurate with Program schedules. Under these circumstances, an informal test program shall be acceptable as regards prior publication of test plan and procedural documents. However, detailed logs of exploratory tests shall be maintained to account for all variations attempted. Requirements for data measurement accuracy and analytical correlation of the data with failure mode analyses, and post-test documentation shall not be less stringent than the requirements for a formalized test program. Engineering tests shall be conducted with the approval of [redacted] Program Directorate and in accordance with the Program Plan of the Procurement Contract.

4.2.2 Qualification (Pre-production) Tests

Qualification tests shall be conducted to verify design adequacy and to demonstrate a minimum level of equipment capability. The test conditions are intended to be representative of the extreme conditions to which the equipment may be subjected during its lifetime. Testing to these conditions shall provide assurance of locating faults, thus compensating to some extent for the statistical limitations of the small sample size. Qualification test conditions shall consider the environments described under Section 3.1.3.3. The Satellite Vehicle and Payload equipment shall be qualified to the requirements of 6117B, except as specifically authorized. Qualification test specifications, and all deviations from the above stated requirements, shall be reviewed and approved by the procuring agency and the [redacted] Program Directorate. Provisions for qualification testing shall be specified in the Contractor's Program Plan for the Procurement Contract. Detailed requirements for qualification testing of equipment shall be presented in the applicable Equipment Specification.

4.2.3 Reliability Testing

Reliability testing is here defined as the life testing of relatively large samples of particular items of equipment for the purpose of determining mean-time-to-failure, failure mode, statistical reliability and confidence factor. For [redacted] System equipment, the results of qualification, acceptance and flight testing shall provide data for establishing equipment reliability estimates. This shall be accomplished by the collection, reporting and analysis of equipment failure data down to the component level. Reliability testing, as such, shall be conducted on highly critical items of equipment for which a reliability base-line cannot be established by data from other types of testing. Reliability testing shall be conducted in accordance with the Reliability Program Plan of the Procurement Contract as approved by the Procuring Agency and the [redacted] Program Directorate.

4.3 Category II System Test Requirements

Category II test requirements shall encompass the acceptance testing of components, equipments, vehicles and functional subsystems for the [redacted] System. The culmination of Category II testing is the flight-readiness certification for the complete [redacted] System prior to each mission launch. Test requirements shall specify the test planning, test conduct, data reduction and analysis, and test quality control.

4.3.1 Acceptance Testing Functional Subsystems

Acceptance tests are intended to improve equipment reliability by disclosing workmanship defects in sufficient time to permit corrective action to be accomplished prior to the end use of the article. Acceptance test environments are intended to be comparable to nominal field environment in severity, but shall avoid fatiguing or wearing out of theequipment. Test requirements shall provide for the detection and elimination of early-life failures. Acceptance testing shall apply to all end-items delivered for use in the [redacted] System, including spares. The Functional Flow Diagram of Figure 4 illustrates the levels of acceptance testing for the major end-items and functional subsystems comprising the [redacted] System. Additionally, each Associate Contractor shall provide for acceptance testing below the levels shown, by incorporating acceptance requirements at equipment and component levels in his end-item specification shown in the tree of Figure 6. Requirements at the functional subsystem level are as follows:

4.3.1.1 Satellite Vehicle

The satellite vehicle comprising a basic S01B modified in accordance with [redacted] Program requirements shall be acceptance tested at the Contractor's facility in Sunnyvale, California. This test shall provide for a complete evaluation of individual vehicle subsystems and an integrated test for concurrent operation of vehicle subsystems in a simulation of all critical phases of flight. At the conclusion of testing, the vehicle with substantiating records shall be offered to the procuring agency as flight ready, with the exception of pyrotechnics, batteries, and similar items normally installed at the launch base. With the exception of a subsequent equipment malfunction prior to flight, there shall be no requirement for disassembling the vehicle or replacing wiring harnesses that would invalidate the prior condition at the completion of acceptance testing.

Testing shall be performed in accordance with a detailed Test Plan under the requirements of the Acceptance Test Specification shown on Figure 6. The test specification, plan and procedures shall be subject to review and approval of the [redacted] Program Directorate. Test results shall be presented in an acceptable data table format, and be substantiated by detailed log books containing the test history, operations, equipment removals, failure data, and documented equipment configuration at the time of acceptance. Supporting analyses shall be presented to define any test anomalies and corresponding corrective action.

4.3.1.2 Launch Vehicle

The Stage I booster vehicle consisting of a standardized LV2A, Thrust Augmented Thor, shall be acceptance tested at the Contractor's facility in Santa Monica, California. Acceptance requirements shall be essentially the same type as stated above for S01B. Testing shall be performed in accordance with a detailed Test Plan under the requirements of the Acceptance Test Specification shown on Figure 6. In this case, acceptance shall be the responsibility of the procuring agency.

4.3.1.3 Payload/Re-entry Vehicles

The camera equipment and the Satellite Re-entry Vehicles shall be tested and accepted at the respective contractor facilities as shown in the flow diagram of Figure 4. Acceptance shall be by the procuring agency to the requirements of the acceptance specifications as shown in Figure 6. Detailed log books, test logs, failure data, calibrations, and supporting analyses shall be provided in documented form accompanying the deliverable hardware.

After acceptance, the camera equipment and SRV's shall be provided as government furnished equipment (GFE) to the Payload Contractor for integration into the payload section of the Satellite Vehicle. Assembly of the integrated payload shall be in accordance with the requirements of the payload specification including all applicable interface requirements. The payload section of the Satellite Vehicle shall then be acceptance tested as a complete unit, and offered for delivery to the procuring agency. Payload requirements and acceptance specifications are shown on the tree of Figure 6. Specifications, test plans and procedures shall be subject to review and approval by the procuring agency and by the [redacted] Program Directorate.

The payload test philosophy shall be documented in a test matrix by the Payload Contractor. The test matrix shall minimize unnecessary redundant testing, disassembly and handling of the payload section components at the factory areas and at the launch base. The test matrix shall provide the basis for approval modifications to implement changes to optimize the Payload Test Plan referenced on Figure 5.

4.3.2 System Acceptance Test-Flight Readiness

The test plans shown on Figure 5 shall document the final testing requirements for individual subsystems and assemblies of hardware comprising the [redacted] System launch configuration. The interface requirements of Section 3.3.6 and applicable Interface Specifications shall be properly accommodated by the test plans. It is a program objective that all final testing prior to mating of the launch configuration be of confidence-test nature. Additionally, the tests shall minimize the necessity for disassembly and the disconnecting of electrical connectors in the flight wire harnesses.

All launch base tests shall be under the jurisdiction of the 6595th ATW. Each test shall be performed in accordance with a test procedure approved by the Launch-Base Test Wing. These requirements shall also apply to the pre-launch mated tests and countdown of the flight-ready vehicles.

Prior to launch, the [redacted] System shall be reviewed and certified as ready to accomplish the designated mission. The flight readiness review will normally be accomplished one day prior to launching. The review will be conducted for the [redacted] Program Directorate by the responsible organization of the 6595th Aerospace Test Wing. The readiness review shall include but not be limited to the following participants.

A. Launch Base Test Wing - 6595th ATW
B. [redacted] Program Directorate
C. [redacted]
D. Western Test Range
E. Satellite Control Wing - 6594th ATW
F. Ground Guidance Contractor - BTL/WECO
G. Stage I Booster Contractor - DAC
H. Satellite Vehicle Contractor - LMSC
I. Payload Integrating Contractor - LMSC
J. Supporting Organizations for the Launch Wing, Western Test Range, and Satellite Control Wing, such as Weather, Range Safety, etc.

The readiness review shall encompass all flight and ground hardware for the mission, operational personnel and facilities required, and the data and computer software utilized for mission preparation and operational control. The following categories shall be covered as a minimum.

Command Programming
Mission Profile and Sequence of Events
Launch Window
Launch/Hold Requirements
Weight and Balance Data
Performance Data and Margin
Instrumentation and Telemetry Schedules
Range Safety
Vehicle Hardware Settings
Alignment Verification
Electrical Load Summary and Capacity
Command Function Listings
Instrument Calibrations
Re-entry Range and Impact Dispersions
Thermal Analysis Summary
Aerodynamic and Structures Summary
Guidance and Control Summary
Interface Compatibility
Operational Support Requirements
Test History, Data and Audits
Reliability and Confidence Estimate for Performance
Countdown Manual
Certification of Readiness

4.3.3 Acceptance Test Documentation

Documentation provided by Contractors at the time end-item hardware is offered for acceptance shall include the items listed under Section 4.3.1. The Contractor end-item specifications of Figure 6 shall specify detailed documentation requirements. Documentation for operational organizations shall be in accordance with the requirements of operational plans as shown on Figure 5.

5.0 Preparation for Delivery

Specifications for all end-item hardware to be delivered into the [redacted] System shall specify detailed requirements for delivery preparation. These requirements shall provide for storage and shipment, with due consideration of environments to be encountered subsequent to leaving the contractor's facility. Special precautions shall be taken to protect payload equipment from damage and contamination.

6.0 Notes

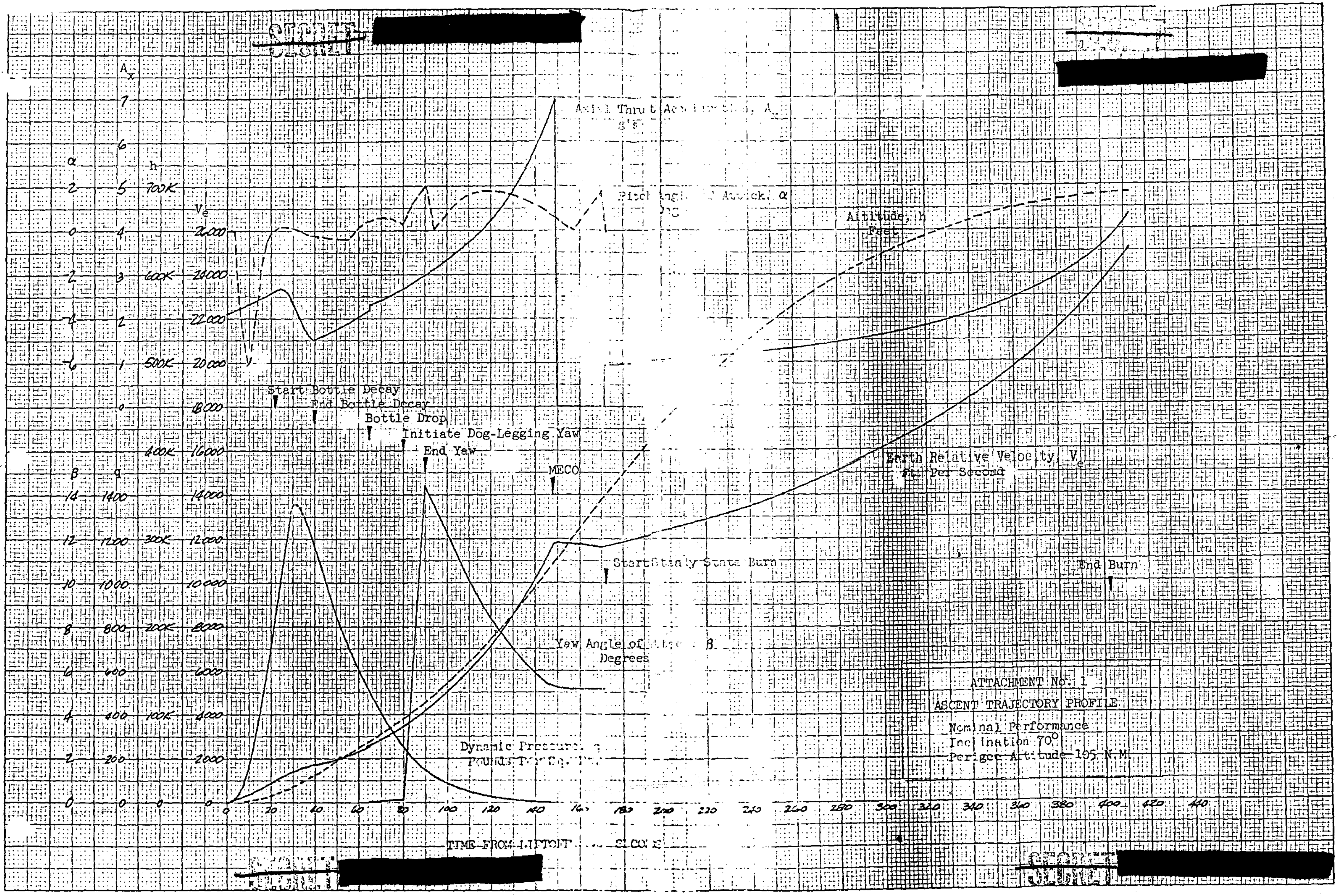
For certain missions, at orbit inclinations of 70° to 80°, a small excess performance capability is provided by the launch vehicle subsystem, over and above the requirements to place the mission payload into orbit. In these instances it may be desirable to utilize the full performance potential of the System and carry research payloads on the aft equipment rack of the satellite vehicle. Research payloads shall be functionally and electrically isolated from the satellite vehicle and its mission payload, so that a malfunction of the research payload can in no way compromise the primary mission. Approval to incorporate research payloads on the satellite vehicle shall be contingent upon presentation of a thorough failure-mode analysis to substantiate the element of risk involved to the [redacted] mission. Final approval to carry a research payload shall rest with the [redacted] Program Directorate. All secondary payloads to be carried shall be within the allowable weight margin for the mission, and shall conform to the interface requirements specification for Agena Model 39205/Research Payloads.

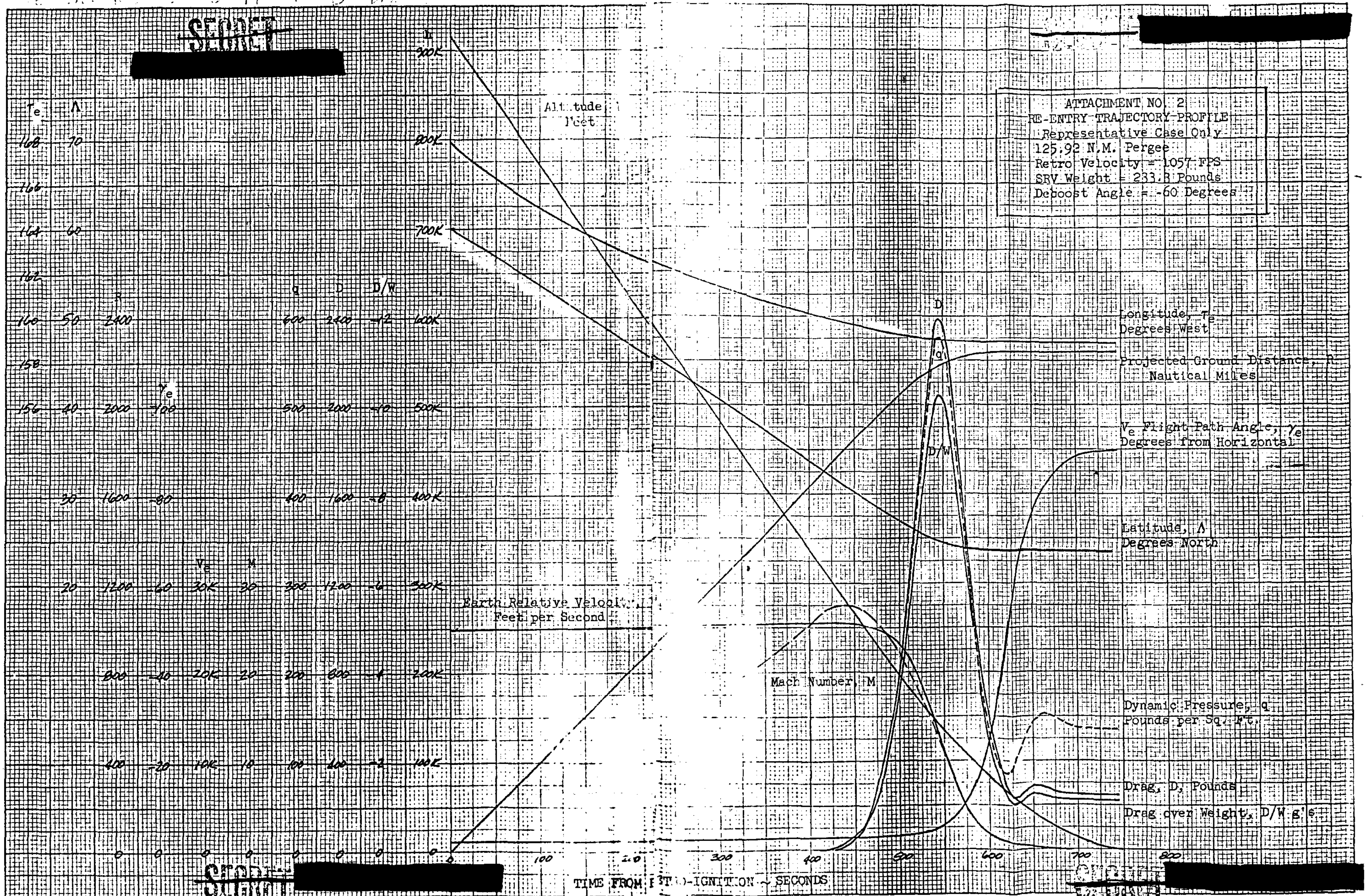
Agena S01B and SS01B vehicles are both covered by the single specification, LMSC 1414870, for standardized Agena vehicles. The SS01B designation applies to serial number vehicles delivered subsequent to First Article Configuration Inspection (FACI). The relatively minor differences in vehicl[illegible] [illegible]nent are defined by LMSC 1414870, "Detail Specification S01B and SS01B Vehicles".

6.1 List of Abbreviations

Abbreviations	Full Terminology
AC, ac	Alternating Current
AF	Air Force
AGC	Automatic Gain Control
AGE	Aerospace Ground Equipment
AP	Advanced Payload, Primary Payload
[redacted]	[redacted]
ATW	Aerospace Test Wing
BTL	Bell Telephone Laboratories
BTU	British Thermal Unit
C&C	Communications and Control
Cg	Center of Gravity
DAC	Douglas Aircraft Company
DB, db	Decibels
DC, dc	Direct Current
deg	Degrees
DPE	Data Processing Equipment
F	Fahrenheit
F/C	Flight Control
FM	Frequency Modulation
Ft, ft	Feet
G&C	Guidance and Control
GERZ	General Electric Return to Zero
GFE	Government Furnished Equipment
HG, hg	Mercury
HR, hr	Hour
H/S	Horizon Sensor
[redacted]	[redacted]
IF	Intermediate Frequency
IMC	Image Motion Compensation
IR	Infra Red
IRIG	Inter-Range Instrumentation Group
IRP	Inertial Reference Package
KIAS	Knots Indicated Air Speed
[redacted]	[redacted]
LMSC	Lockheed Missiles and Space Company
LV	Launch Vehicle

Abbreviations	Full Terminology Cont'd. . . . .
MIL	Military
MM, mm	Millimeter
MSL	Mean Sea Level
N	North
[redacted]	[redacted]
NM, nm	Nautical Miles
NRZ	Non Return to Zero
NTS	North to South
PAM	Pulse Amplitude Modulation
PALC	Point Arguello Launch Complex
PDM	Pulse Duration Modulation
PPM	Pulse Position Modulation
PU	Propellant Utilization
R/E	Re-entry
RF	Radio Frequency
RFP	Request for Proposal
RSC	Range Safety Command
RSS	Root Sum Square
RTC	Real Time Command
RV	Re-entry Vehicle
S	South
[redacted]	[redacted]
SCD	Subcarrier Discriminator
SCF	Satellite Control Facility
SCO	Subcarrier Oscillator
SLV	Standard Launch Vehicle
SPC	Stored Program Command
SRV	Satellite Re-entry Vehicle
STC	Satellite Test Center
STD, std	Standard
STN	South to North
SV	Satellite Vehicle
TAG	Tested and Guaranteed
TAT	Thrust Augmented Thor
TLM, T/M	Telemetry
TT&C	Telemetry, Tracking and Command
[redacted]	[redacted] Tracking Station
UHF	Ultra High Frequency
USAF	United States Air Force
VAFB	Vandenberg Air Force Base
VCO	Voltage Controlled Oscillator
VHF	Very High Frequency
V/M	Velocity Meter
[redacted]	[redacted]
WTR	Western Test Range


SECRET
Start Bottle Decay
End Bottle Decay
Bottle Drop
Initiate Dog-Legging Yaw
End Yaw
MECO
End Burn
Altitude, h
Feet
Earth Relative Velocity, V_e
Ft. Per Second
Yaw Angle of
Degrees
Dynamic Pressure, q
TIME FROM LIFTOFF
ATTACHMENT No. 1
ASCENT TRAJECTORY PROFILE
Nominal Performance
Inclination 70°
Perigee Altitude 105 N.M.


SECRET [redacted]
ATTACHMENT NO. 2
RE-ENTRY TRAJECTORY PROFILE
Representative Case Only
125.92 N.M. Pergee
Retro Velocity = 1057 FPS
SRV Weight = 233.8 Pounds
Deboost Angle = -60 Degrees
Altitude
Feet
Longitude, τ_e
Degrees West
Projected Ground Distance, R
Nautical Miles
V_e Flight Path Angle, γ_e
Degrees from Horizontal
Latitude, Λ
Degrees North
Earth Relative Velocity
Feet per Second
Mach Number, M
Dynamic Pressure, q
Pounds per Sq. Ft.
Drag, D, Pounds
Drag over Weight, D/W g's
TIME FROM RETRO-IGNITION ~ SECONDS
SECRET [redacted]

ATTACHMENT #3

ERROR BUDGETS

(This attachment will be incorporated into the Systems Specification at a later date.)

~~SECRET~~ [redacted]

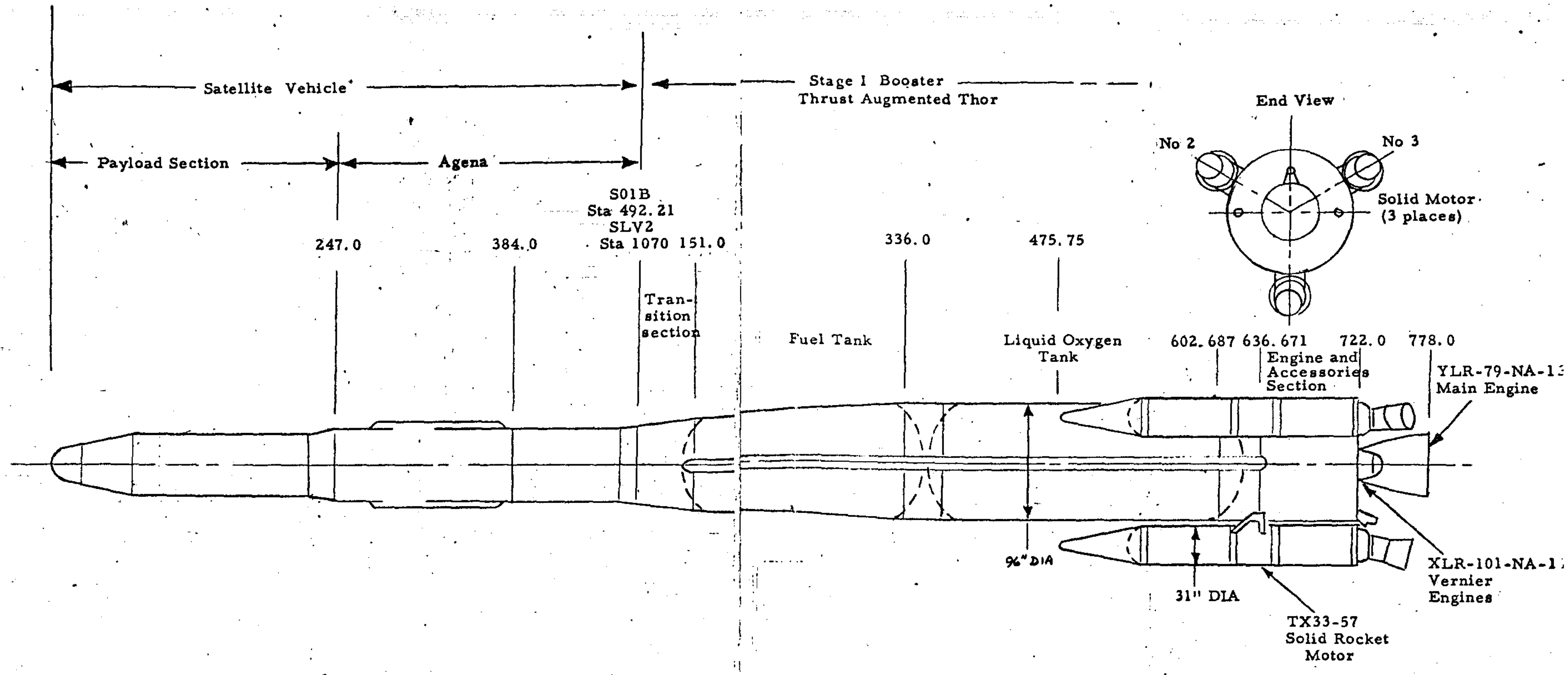


FIGURE 1 LAUNCH VEHICLE CONFIGURATION [redacted] SYSTEM

~~SECRET~~ [redacted]